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AEROSPACE GROUP
SPACE SYSTEMS DIVISION
EL SEGUNDO, CALIFORNIA

31 October 1963

SUBJECT: Advanced Syncom Summary Technical Report
for Period May through October 1963
Contract NAS 5-2797

TO: Mr. Robert J. Darcey
SYNCOM Project Manager
Goddard Space Flight Center, Code 621
Greenbelt, Maryland

Attached are copies of the Advanced Syncom Summary Technical Report for the period May through October 1963. This seven-volume report provides engineering information summarizing all elements of study, research and development, fabrication and testing, and specification of subsystems and components conducted during the report period. Engineering schematic, circuit, mechanical, and product drawings prepared during this period were transmitted separately and are not repeated in this report.

During the 6-month period reported here, the engineering model structure (T-1) was subjected to qualification level dynamic environmental loads; new communication transponder, telemetry encoder, command decoder, central timer, phased array control, and jet control circuit designs were completed; communication system test equipment, control item test equipment, and synchronous controller designs were established; alternate structural design approaches were evaluated and a versatile structure selected and designed; and feasibility demonstration tests were conducted on the bipropellant reaction control system and propellant exhaustion test runs were completed at simulated mission conditions on the engineering model reaction control unit.

Advanced engineering models of the communication transponders, phased array antenna, antenna control electronics, and central timer were fabricated and tested. In addition, a welded module model of the phased array control electronics was fabricated.

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Sources were evaluated for possible future procurement of high-reliability components, and detailed component specifications and requirements for inspection, test, serialization, burn-in, and power aging were prepared.

Dynamic load tests were conducted on a solar panel designed for the revised structure. Solar cell manufacturers were surveyed and sample quantities of n on p cells were procured. Flat plate nickel-cadmium cell batteries were obtained in sample quantities for cycle tests under simulated stationary orbit mission requirements.

HUGHES AIRCRAFT COMPANY

Paul E. Norsell

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Advanced SYNCOM

SUMMARY REPORT

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Volume 1

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NASA Contract 5-2797

APPROVED: *Paul E. Nonell*

Manager, Systems Development
Advanced Syncom

SSD 3118R

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HUGHES AIRCRAFT COMPANY
SPACE SYSTEMS DIVISION

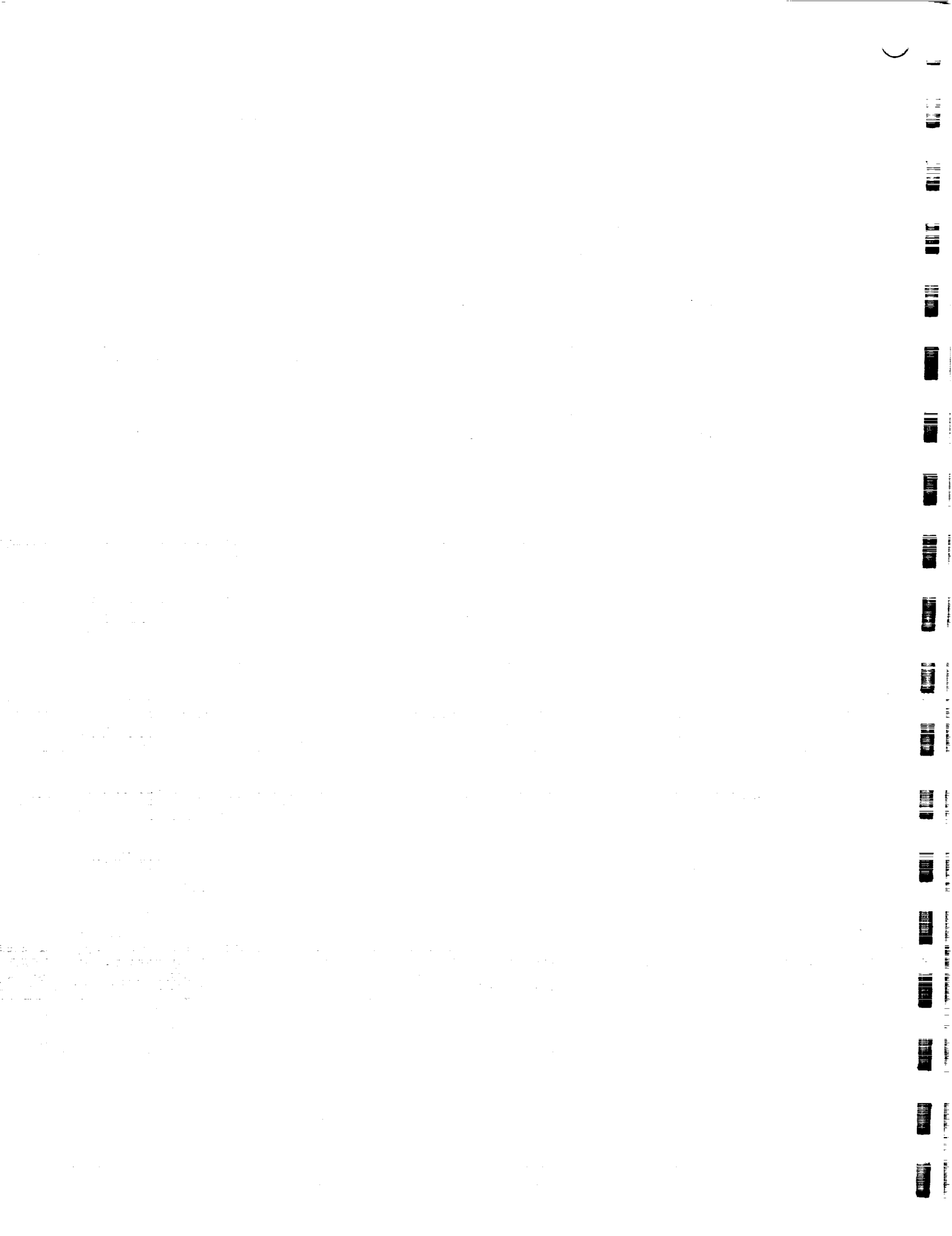


CONTENTS

	Page
1. INTRODUCTION	1-1
2. SPACECRAFT SYSTEM SUMMARY	2-1
3. COMMUNICATION SYSTEM DESIGN	
Requirements	3-1
Modes of Operation	3-1
System Calculations	3-2
Frequency Allocations and Interrelations	3-4
Special Studies	3-5
RF Interference Analysis	3-11
4. LAUNCH AND ORBIT CONSIDERATIONS	
General Sequence of Events	4-1
Fixed versus Variable Apogee Ignition Timer	4-2
Translation due to Initial Orientation Impulse	4-5
Orbital Perturbations due to Sun and Moon	4-6
Justification for Spin Speed Control	4-10
Axial Jet Induced Nutation	4-14
Emergency Procedures (Ascent Sequence)	4-18
Earth Sensor for Advanced Syncom	4-21
5. SPACECRAFT SYSTEM DESIGN	
Spacecraft Block Diagram	5-1
Communication Transponders	5-1
General Physical Arrangement of Transponder	5-2
Minor Control Item Description	5-2
Regulators	5-16
Communication Transmitters	5-17
384H Traveling-Wave Tube	5-17
Traveling-Wave Tube Power Supply	5-33
Ferrite RF Power Switch	5-41

	Page
Communication Antenna System	5-42
Phased Array Antenna System	5-42
Phased Array Control Electronics (PACE)	5-54
Receiver Antenna Development	5-77
Multiplexers	5-80
Telemetry and Command System	5-81
Telemetry and Command Antenna System	5-81
Telemetry Transmitter	5-82
Command Receiver	5-87
Telemetry Encoder	5-87
Command Decoder	5-110
Command Allocation	5-128
Central Timer	5-130
Progress Items	5-131
Functional Description	5-132
Block Diagram Description	5-132
Core Frequency Scalers	5-135
Product Design	5-135
Circuit Descriptions	5-135
Electrical Power	5-145
Power Supply System Design	5-145
Battery Power System	5-153
Solar Cell Assembler	5-154
Electrical Power System Analog	5-154
Particulate Radiation Damage	5-157
Structure	5-167
General Arrangement and Structural Design	5-167
Structural Dynamic Test and Analysis	5-196
Mass Property Analysis	5-196
Thermal Control	5-198
Reaction Control System	5-210
System Description	5-210
Development Status	5-216
Apogee Engine	5-225
System Description	5-225
Development Status	5-225

	Page
6. RELIABILITY AND QUALITY ASSURANCE	
Failure Mode Analysis	6-1
Critical Component Test Plans	6-6
Quality Assurance	6-57
7. MATERIALS, PROCESSES, AND COMPONENTS	
Spacecraft Bill of Material	7-1
System Test Equipment	7-1
Preferred Parts, Materials, and Processes	7-17
Radiation Environment Studies	7-21
Material and Process Studies	7-28
8. GROUND SUPPORT EQUIPMENT	
Introduction	8-1
Synchronous Controller Auxiliary	8-10
Transponder RF Test Set	8-12
Major and Minor Control Item Test Positions	8-21
Digital Subsystem Major and Minor Control Item Test Equipment	8-26
9. SPACECRAFT HANDLING EQUIPMENT	
Weight, CG, and Moment of Inertia Fixture	9-1
Systems Test and Spin Fixture	9-2
Mobile Assembly Stand	9-5
Balancing Machine	9-5
10. NEW TECHNOLOGY	
New Items	10-1
Previously Reported Items	10-1
Subcontracts	10-2
11. PROJECT REFERENCE REPORTS	11-1



1. INTRODUCTION

The use of communication satellites provides the practical solution to the need for greatly expanded global communications capability. A major effort of the United States Government and of industry has been in process since the late 1950s to develop a satellite relay system at the earliest possible time.

The National Aeronautics and Space Administration, having management responsibility for developing the space technology leading to a communication satellite system, has investigated nonsynchronous passive satellites and nonsynchronous active repeater satellites. The Goddard Space Flight Center Project Syncom is assigned the synchronous-orbit active repeater satellite investigations.

Under NASA Goddard Space Flight Center Contract NAS5-1560, Hughes Aircraft Company developed and constructed three Syncom spacecraft for launch from the Atlantic Missile Range by Delta launch vehicles for conduct of inclined synchronous-orbit communications experiments during 1963. The Syncom spacecraft have demonstrated a simple spin-stabilized, active repeater satellite design capable of being placed in a synchronous orbit. Similarly, it has been demonstrated that a simple pulse jet control system can provide the stationkeeping necessary to maintain a synchronous orbit.

The Advanced Syncom spacecraft, currently under study for feasibility and advanced technological development, will demonstrate the stationary, or equatorial, synchronous orbit with a vehicle providing a relatively large, adaptable payload capability, achievement of long life in

orbit, an electronically steerable antenna beam, continuous wide-band communications, and new multiple-access communications. Scientific instruments will be carried to measure the radiation environment and to assess radiation damage occurring during the orbiting process and throughout satellite life in the synchronous, equatorial orbit.

The Advanced Syncom study program has included research and development of engineering models of a multielement phased array transmitting antenna and associated control circuits; a dual-mode communications transponder operating at 6-gc receiving frequency and 4-gc transmitting frequency and providing alternate modes of operation as a multi-channel SSB-PM multiple-access transponder or as a wide-band FM frequency translation transponder; a traveling-wave tube final power amplifier for the transponders; a spacecraft structure; and a bipropellant rocket jet control system.

In May 1963, the NASA expanded the advanced technology program to include design effort on all elements of the spacecraft and the communication system test equipment. Currently under way is the fabrication and testing of advanced engineering models of the communication transponders, transmitting and receiving antennas, and traveling-wave tube power amplifiers. Breadboard circuits of the telemetry encoders and command decoders are similarly in process. The system test equipment being developed will permit quantitative measurements of communication system performance to be obtained. The above activities will be completed by the end of October 1963.

Hughes Aircraft Company

This Summary Report covers the technical progress achieved during the contract period and details the system configuration and specifications resulting from system studies. The report is divided into seven volumes:

Volume 1: Advanced Syncom Summary Report

Volume 2: Major and Minor Control Item Test Plans and In-process Specifications

Volume 3: Interface Reports

Volume 4: System and Subsystem Performance Requirements

Volume 5: System Test Plans

Volume 6: Engineering Data on Transponder Control Items

Volume 7: T-1 Structural Vibration Test Report

2. SPACECRAFT SYSTEM SUMMARY

The Advanced Syncom satellites, as in the Delta-launched Syncom, will utilize spin stabilization for attitude stabilization. The spacecraft physical parameters are increased over those of Syncom to accommodate a large increase in communications capacity. In addition, the self-contained apogee injection stage removes approximately 29 degrees of inclination from the orbit while circularizing the elliptical transfer orbit at the synchronous radius. The parameters of an Advanced Syncom spacecraft are summarized in Table 2-1. Figures 2-1 and 2-2 illustrate the general arrangement and the structural engineering model of the Advanced Syncom.

The communication capacity of each of the satellite transponders is 600 two-way telephone conversations. The design for each satellite contains four such transponders, providing a total system capacity through the satellite of 2400 two-way voice channels. Alternately, the system can accommodate television or other wide-bandwidth signals through each of the transponders.

Ground station characteristics for which the full communications capacity is achieved would include an 85-foot-diameter antenna, a 10 kw saturated power transmitter, and an 80° K receiver.

Smaller stations can be used with a proportionate reduction in capacity. With 40-foot-diameter antennas and the same transmitters and receivers, the voice channel capacity is reduced to 120 two-way channels and the television signal noise level, and therefore picture quality, falls below CCIR standards.

The system has two alternate modes of operation possible in each assigned frequency band. The first mode accommodates a wide-band FM transmission to the spacecraft, which translates the signal-carrier frequency and repeats the sig-

TABLE 2-1. PARAMETERS OF ADVANCED SYNCOM

Physical configuration	58-inch-diameter cylinder
Weight	1518 pounds at launch 760 pounds in 24-hour, equatorial orbit
Apogee injection motor	Solid propellant, JPL
Control systems	Liquid bipropellant Fuel: monomethylhydrazine Oxidizer: nitrogen-tetroxide Two independent systems: capacity per system adequate for correcting initial errors and providing 3 years stationkeeping
Communications	Self-contained spin rate control Four independent dual-mode transponders Redundant 4.0-watt traveling-wave tube power amplifiers in each transponder 8-db collinear array receiving antenna 18-db phased array transmitting antenna 6019, 6108, 6212, 6301 gc ground-to-spacecraft 3992, 4051, 4120, 4179 gc spacecraft-to-ground
Telemetry	Four 1.25-watt transmitters at 136.470 or 136.980 mc Four encoders; GSFC PFM standard
Command	Four receivers at 148.260 mc Four decoders; GSFC FSK standard
Electrical power	147-watt, n-p solar cell array 650-watt-hour rechargeable nickel-cadmium energy storage system

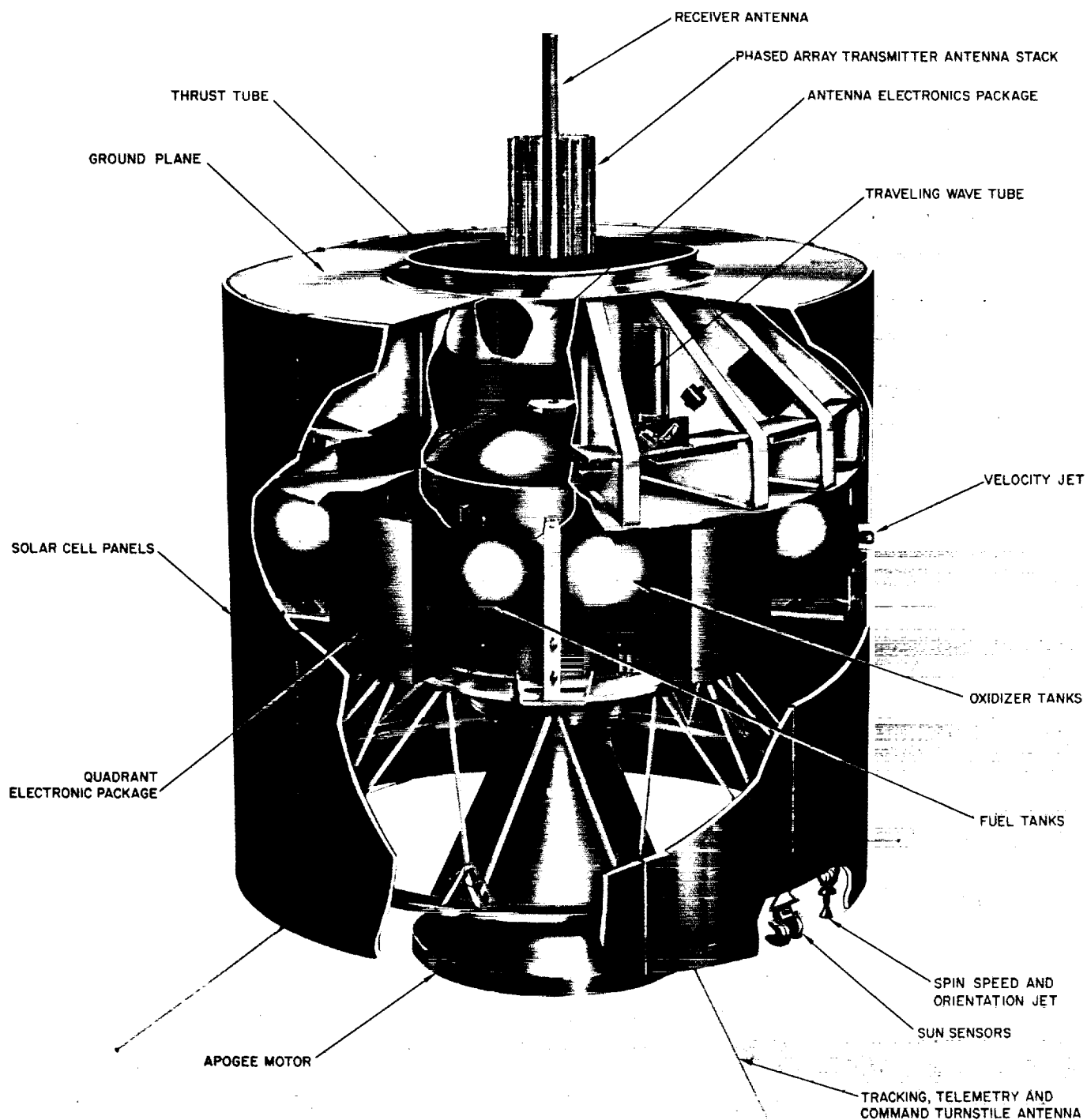


FIGURE 2-1. GENERAL INTERNAL ARRANGEMENT OF ADVANCED SYNCOM

nal with no conversion in modulation. This mode is used for television or other wide-band data originating from a single station. The spacecraft transponder mode for such signals is termed the frequency translation mode.

The second mode of operation involves the transmission, simultaneously from a large number of ground stations, of frequency division

multiplexed, single-sideband, suppressed carrier voice channels. The signals are converted into phase modulation of a single carrier in the spacecraft and are retransmitted back to all stations in this form. This mode permits simultaneous two-way interconnection of all combinations of the ground stations. The spacecraft transponder mode for these signals is termed the multiple access mode.

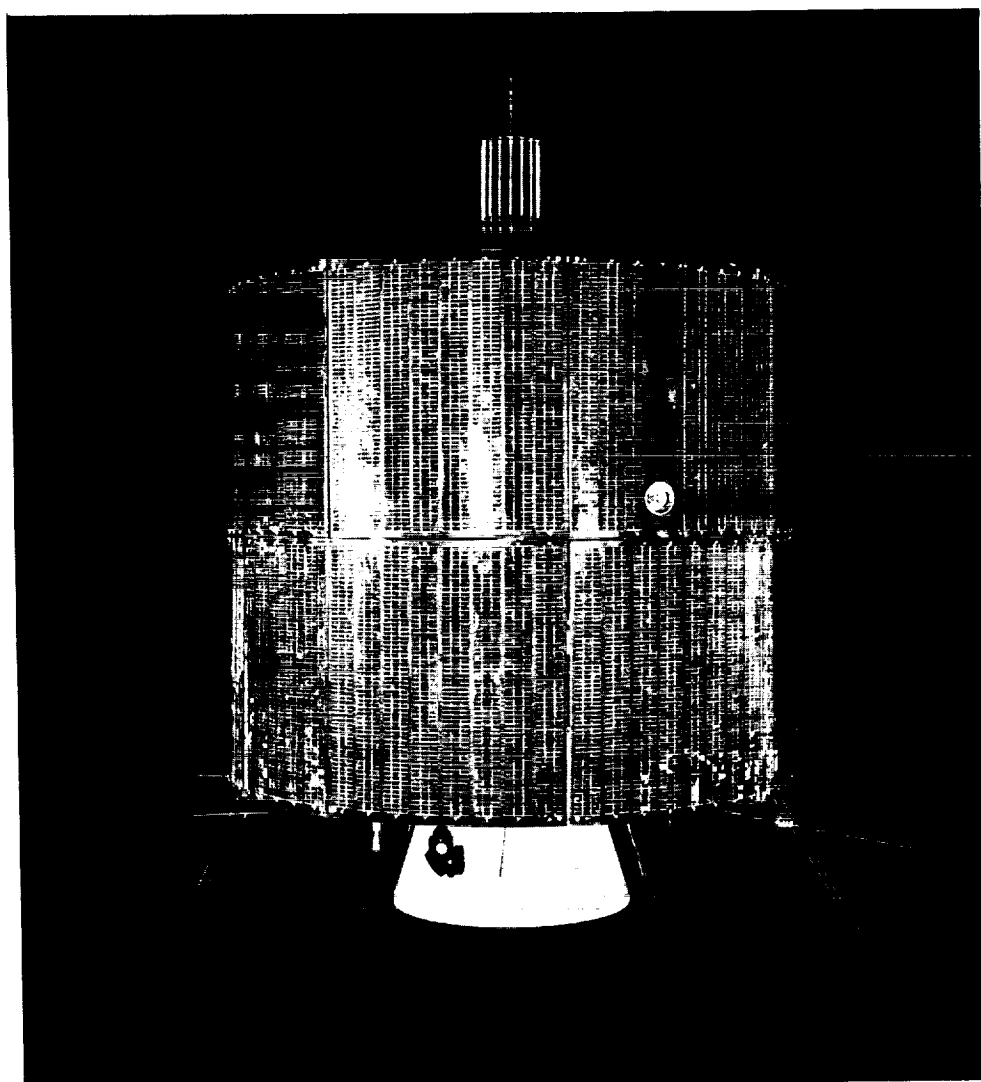
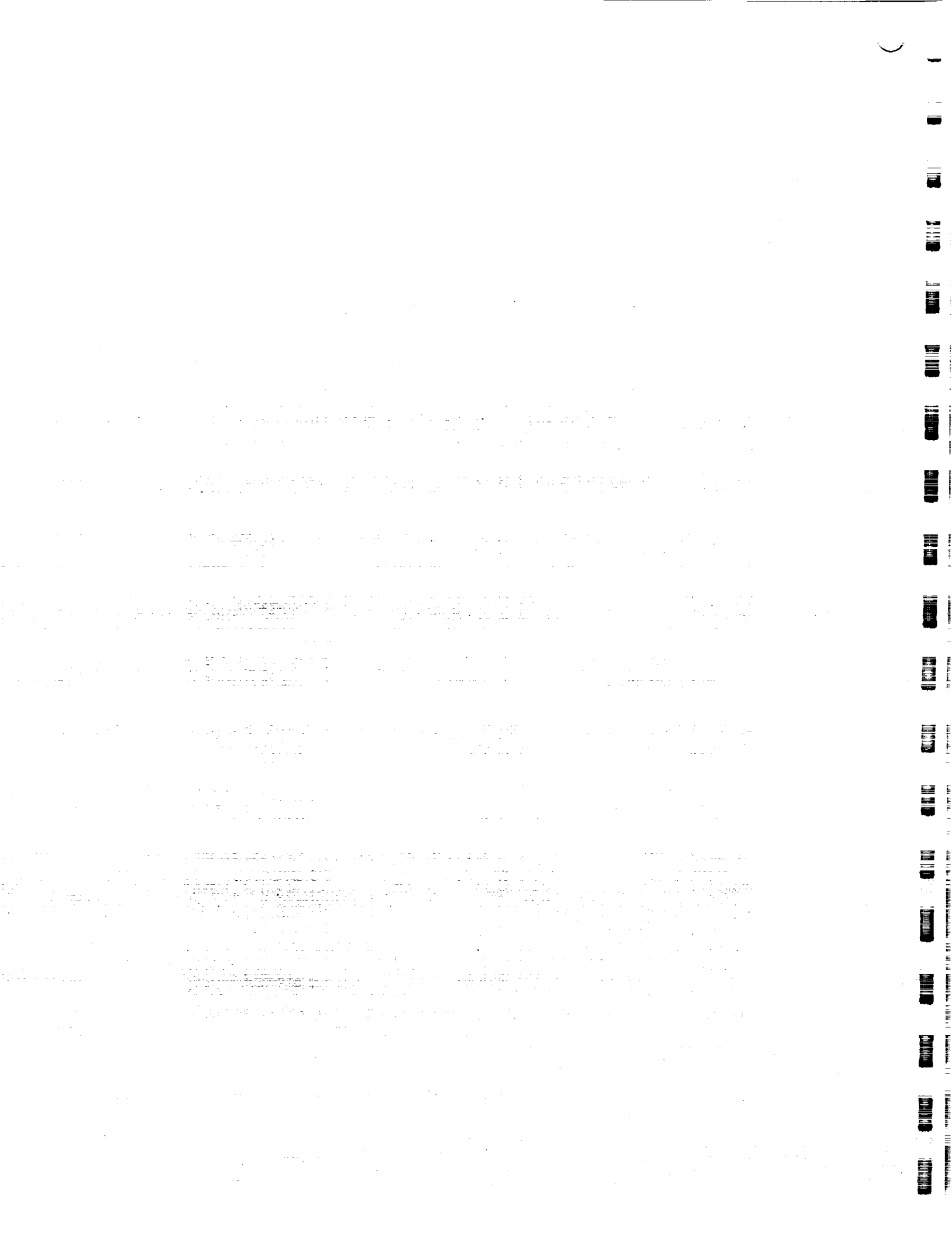


FIGURE 2-2. STRUCTURAL ENGINEERING MODEL OF ADVANCED SYNCOM



3. COMMUNICATION SYSTEM DESIGN

REQUIREMENTS

The communications requirements which have determined the system design are as follows:

Capacity: The system should accommodate 600 two-way voice channels or one monochrome or color television station in each of the four assigned frequency bands. The voice channels can originate from as many as 100 ground terminals simultaneously and can accommodate multiplexed teletype signals.

Quality: The quality of the communications links should exceed the appropriate standards established by the International Radio Consultative Committee (CCIR) of the International Telecommunication Union (ITU).

The assumed ground station characteristics for which the full communications capacity is achieved are listed in Table 3-1.

Smaller stations can be used with a proportionate reduction in capacity. With 40-foot-diameter antennas and the same transmitters and

receivers as in Table 3-1, the voice channel capacity is reduced to 120 two-way channels and the television quality falls below CCIR standards.

The standards used to establish the design of this system are given in Table 3-2. The use of the CCIR standards in the design of this system follows the procedures recommended by Bray.*

MODES OF OPERATION

The system has two alternate modes of operation possible in each assigned frequency band. The first mode accommodates a wide-band frequency modulated transmission to the spacecraft, which translates the signal carrier frequency and repeats the signal with no conversion in modulation. This mode is used for television or other wide-band data originating from a single station. The spacecraft transponder mode which accommodates such signals is termed the frequency translation mode.

*W. J. Bray, "The Standardization of International Microwave Radio Relay Systems," Proceedings of the IEE, Part B, March 1961.

TABLE 3-1. GROUND STATION CHARACTERISTICS

Transmitter (for each frequency assignment)	
Saturated power	10 kw
Frequency band	6 kmc
Bandwidth	25 mc
Diplexer loss	-1 db
Frequency stability	1 part in 10^{10} , short term 1 part in 10^7 , long term
Antenna	
Diameter	85 feet
Efficiency (transmitting and receiving)	54 percent
Receiver	
Noise temperature (all sources, including antenna)	80°K

TABLE 3-2. COMMUNICATION SYSTEM STANDARDS

Television signal-to-noise ratio	
Peak-to-peak signal to weighted noise	51 db
Television video bandwidth	
Monochrome	4 mc
Color	4.5 mc
Voice channels	
Test tone/noise ratio	50 db
Total channel bandwidth	4 kc
Voice portion of bandwidth	3.1 kc
Multiplexed teletype signals	
Maximum error in frequency	2 cps

The second mode of operation involves the transmission simultaneously from a large number of ground stations of frequency division multiplexed, single-sideband, suppressed carrier voice channels. These signals are converted into phase modulation of a single carrier in the spacecraft, and are retransmitted back to all stations in this form. This mode permits simultaneous two-way interconnection of all combinations of the ground stations. The spacecraft transponder mode which accommodates these signals is termed the multiple access mode.

SYSTEM CALCULATIONS

FM Frequency Translation (TV). Tables 3-3 and 3-4 list the signal and noise levels at several key points in the system when used for the relaying of monochrome television signals. The overall peak-to-peak signal to weighted noise ratio is 54.2 db, which exceeds the CCIR standard by 3.2 db. The noise weighting factor and the appropriate standard were interpolated from curves given in the previously cited Bray report.

Single Sideband-Phase Modulation Multiple Access (Voice). Tables 3-5 and 3-6 list the signal and noise levels at key points in the system

TABLE 3-3. GROUND TO SPACECRAFT, FREQUENCY MODULATION (TV)

Transmitter power	33.0 dbw
Diplexer loss	-1.0 db
Ground antenna gain	62.1 db
Space attenuation	-200.8 db
Receiving antenna gain	8.0 db
Off beam center allowance	-1.5 db
Multiplexer and miscellaneous loss	-3.0 db
Received carrier power	-103.2 dbw
Receiver noise power density	-194.4 dbw/cps
Receiver noise bandwidth	74.0 db
Receiver noise power	-120.4 dbw
Carrier/noise ratio	17.2 db

TABLE 3-4. SPACECRAFT TO GROUND FREQUENCY MODULATION (TV)

Spacecraft transmitter power	6 dbw
Diplexer and phase shifter losses	-3 db
Spacecraft antenna gain	18 db
Space attenuation	-197.1 db
Off beam center allowance	2 db
Ground antenna gain	58.4 db
Received carrier power	-119.7 db
Receiver noise power density (80°K)	-209.62 dbw/cps
Receiver bandwidth	74.0 db
Receiver noise power	-135.6 dbw
Carrier/noise ratio	15.9 db
Carrier/noise ratio — up link	17.2 db
Carrier/total noise ratio	13.5 db
Top modulation frequency	4 mc
Modulation index, M	2.5
Improvement factor, $3M^2 \frac{25}{4(2)}$	17.7 db
Average signal-to-noise ratio	31.2 db
Noise weighting factor	14 db
Peak-to-peak signal/weighted noise	54.2 db

TABLE 3-5. GROUND TO SPACECRAFT, SINGLE SIDEBAND MODULATION (VOICE)

Transmitter peak power capability	40 dbw
Transmitter average power	31.7 dbw
Channel test tone power	18.9 dbw
Diplexer loss	-1.0 db
Ground antenna gain	62.1 db
Space attenuation	-200.8 db
Receiving antenna gain	8.0 db
Off beam center allowance	-1.5 db
Multiplexer and miscellaneous loss	-3.0 db
Received test tone power	-117.3 dbw
Receiver noise power density	-194.4 dbw/cps
Channel bandwidth	34.9 db
Psophometric noise weighting factor	-2.5 db
Receiver channel noise (weighted)	-162.0 dbw
Test tone/fluctuation noise ratio	44.7 db
Test tone/modulation noise ratio	47.7 db
Test tone/noise ratio	42.9 db

TABLE 3-6. SPACECRAFT TO GROUND,
PHASE MODULATION (VOICE)

Spacecraft transmitter power	6 dbw
Diplexer and phase shifter losses	-3 db
Spacecraft antenna gain	18 db
Space attenuation	-197.1 db
Off beam center allowance	-2 db
Ground antenna gain	58.4 db
Received carrier power	-119.7 dbw
Receiver noise power density (80°K)	-209.6 dbw/cps
Receiver noise bandwidth (25 mc)	74.0 db
Receiver noise power — total	-135.6 dbw
Carrier/total noise ratio	15.9 db
Weighted channel noise power	-177.2 dbw
Carrier/channel noise ratio	57.5 db
Channel test tone modulation index	0.35
Test tone/noise ratio	48.5 db
Transponder test tone/intermodulation noise ratio	40.0 db
Overall link test tone/noise ratio	38.9 db
Compandor improvement factor	15 db
Overall link test tone/effective noise ratio	53.9 db

for conditions of full channel loading, when 600 two-way circuits are in use in each band. The ground station calculations are for the worst case, with all 600 conversations flowing through one station. In computing the noise contributed by various elements of the systems, intermodulation noise as well as thermal noise was considered.

Two voice channel signal processing techniques are employed. Echo suppressors are used to prevent the annoying effect on the speaker of delayed echoes produced by long transmission delays and mismatched ground distribution networks. The newer type of suppressor, which allows interruptions to be heard, are used to allow smoother conversational exchanges. Com-

pandors, consisting of a voice dynamic range compressor at the transmitting end and a corresponding expander at the receiving end, are used to substantially improve the capacity and quality of the voice channels. Although tests have indicated improvements of the order of 22 db in signal-to-noise ratio through the use of compandors, the conservative figure of 15 db is used in the calculations to give adequate allowance for the increased channel loading which occurs.

The calculations are based on the channel loading factors and procedures given by Bray and on the detailed analysis of this system given by Braverman and Williams.* The resulting test tone/noise ratio of 53.9 db exceeds CCIR standards by 3.9 db.

Frequency and Gain Adjustments. Because of the stringent frequency error tolerances in the single sideband mode, special consideration must be given in the system design to meeting the frequency stability requirements. The major sources of frequency error are associated with the spacecraft — its residual motion relative to the earth produces a small but significant doppler shift, and the transponder receiver oscillator instability must also be taken into consideration. The design philosophy employed has been to accommodate both sources of spacecraft-induced frequency errors by compensating adjustments made continuously and automatically by the ground stations through the use of transponded pilot tones from each station. The same pilot tones used for frequency control are used by the ground stations to maintain gain control as well, to achieve the desired test tone modulation index in the spacecraft.

*D. Braverman and D. D. Williams, "SSB-PM Multiple Access Communication System Analysis," Hughes Aircraft Company TM 721, October 1962.

FREQUENCY ALLOCATIONS AND INTER-RELATIONS

Transponder Frequency Relationships. The frequency relationships between the various parts of the transponder are shown in Figure 3-1.

Multiple Access Baseband Frequency Arrangement. The frequency structure of the multiple

access system baseband is shown in Figure 3-2.* Each of the four mastergroups consists of 300 voice channels (five supergroups) so that a total of 1200 channels is represented in the figure. The voice channel frequencies extend from 316 kc to 5.564 mc. Pilot frequencies may be added

*Taken from CCITT IInd Plenary Assembly, New Delhi, Redbook, Vol. III, 8-16, December 1960, p. 128.

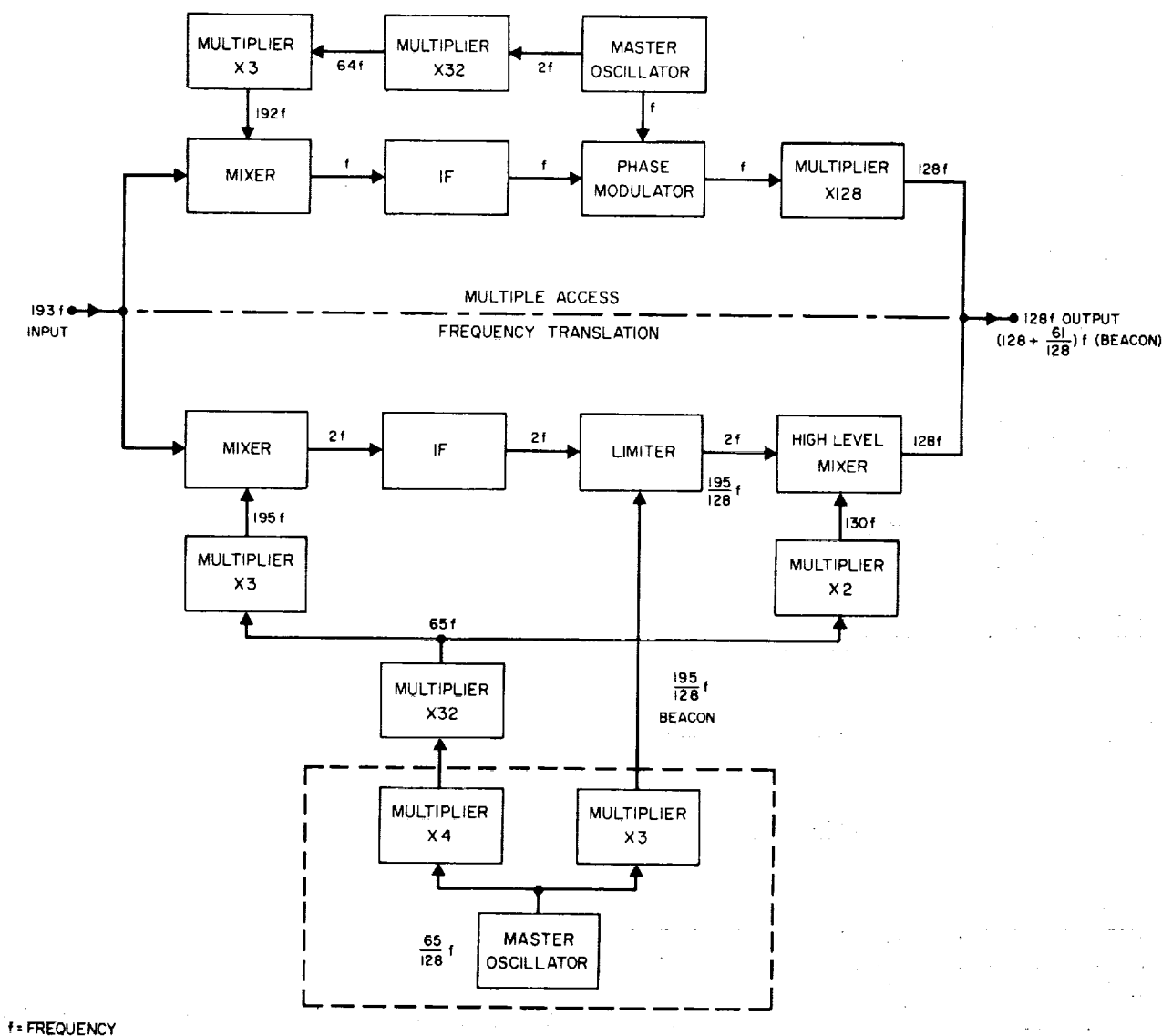


FIGURE 3-1. FREQUENCY DIAGRAM

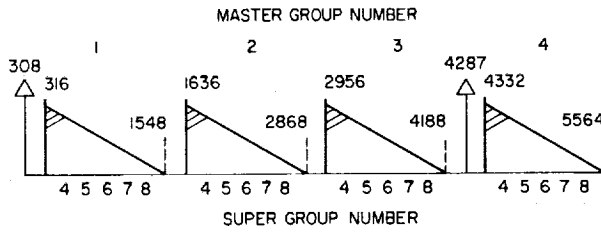


FIGURE 3-2. MULTIPLE ACCESS BASEBAND FREQUENCY ARRANGEMENT

to the lower end of the baseband, and one is shown at 308 kc. The CCITT also allows a pilot frequency at 4.287 mc, as indicated in Figure 3-2.

SPECIAL STUDIES

Carrier Level in Multiple Access Mode as Function of Modulation Index

Ratio of Carrier Power to Total Power as Function of RMS Modulation Index. An analysis was made of the ratio of carrier power to total power as a function of rms modulation index for a carrier phase-modulated by random noise. In the multiple access mode of the spacecraft-to-ground link the carrier is phase-modulated by a signal with the characteristics of random noise. To determine the feasibility of tracking the carrier using a phase-locked tracking receiver, the amount of power in the carrier as a function of the rms modulation index should be determined.

Assume a carrier is phase-modulated by random noise of bandwidth B . The rms modulation index is therefore the rms value of the phase deviation of the carrier:

$$\text{rms modulation index} = \bar{\Theta}$$

or

$$\text{peak modulation index } M = \sqrt{2} \bar{\Theta}$$

The mean square value of the modulation index is proportional to signal power, which in this

case is proportional to the bandwidth B of the modulating spectrum (noise).

$$\bar{\Theta}^2 = KB$$

It may be assumed that B is the sum of N contiguous narrow bands of noise, each of width B/N . Therefore each of the N bands contributes $\bar{\Theta}^2/N$ to the mean square value of the modulation index, or the rms modulation index due to a narrow band is $\sqrt{\bar{\Theta}^2/N}$. Now the power is represented in each small noise band by a sine wave of the same equivalent power of peak amplitude M_n . The peak modulation index is thus

$$M_n = \sqrt{2} \sqrt{\frac{\bar{\Theta}^2}{N}} = \bar{\Theta} \sqrt{\frac{2}{N}}$$

The ratio of carrier power to total power in a signal modulated by N sinusoidal subcarriers is

$$\frac{P_c}{P_{\text{tot}}} = \prod_{n=1}^N [J_0(M_n)]^2$$

If all N signals have the same modulation index M ,

$$\frac{P_c}{P_{\text{tot}}} = [J_0(M)]^{2N}$$

For small values of M , $J_0(M)$ may be approximated by

$$J_0(M) \approx 1 - \frac{M^2}{4}$$

Therefore

$$\begin{aligned} \frac{P_c}{P_{\text{tot}}} &\approx \left(1 - \frac{M^2}{4}\right)^{2N} = \left(1 - \frac{2\bar{\Theta}^2}{4N}\right)^{2N} \\ &= \left(1 - \frac{\bar{\Theta}^2}{2N}\right)^{2N} \end{aligned}$$

To simplify this, consider the expression

$$\begin{aligned} \ln \left(1 - \frac{\bar{\Theta}^2}{2N}\right)^{2N} &= 2N \ln \left(1 - \frac{\bar{\Theta}^2}{2N}\right) \approx 2N \left(-\frac{\bar{\Theta}^2}{2N}\right) \\ &= -\bar{\Theta}^2 \quad (\text{for } \frac{\bar{\Theta}^2}{2N} \text{ small}) \end{aligned}$$

Therefore

$$\left(1 - \frac{\bar{\Theta}^2}{2N}\right)^{2N} \approx e^{-\Theta^2}$$

or

$$\frac{P_c}{P_{tot}} \approx e^{-\Theta^2}$$

or

$$10 \log \frac{P_c}{P_{tot}} = -4.34 \bar{\Theta}^2 \text{ decibels}$$

This relationship is plotted in Figure 3-3.

Carrier Reduction in Multiple Access Mode.

A demonstration was made to show that, under full modulation conditions, the carrier can be used instead of a beacon for frequency control. The unmodulated carrier level was measured and then modulated by a continuous 5-mc spectrum of noise simulating full modulation; the drop in carrier level was noted.

The configuration used for the demonstration (Figure 3-4) consists basically of:

- 1) HP-608A signal generator to determine the power level of the modulating noise simulating full modulation
- 2) Noise source to modulate the carrier
- 3) HP-431B power meter to monitor the modulating signal
- 4) Spacecraft electronics under test
- 5) Panoramic SPA4a spectrum analyzer to monitor the carrier level
- 6) HP-618 signal generator as a calibrated source with which to compare the carrier level.

The demonstration procedure is as follows. A single test signal from the HP-608A signal generator, adjusted for a peak modulation index of 1.6, determines the noise power level required

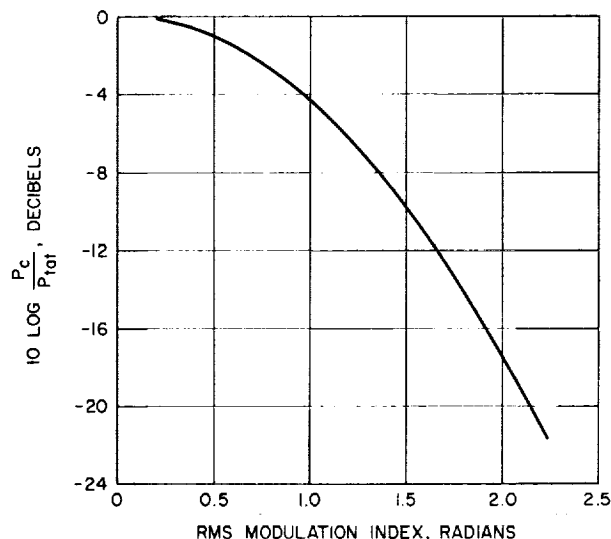


FIGURE 3-3. RATIO OF CARRIER POWER TO TOTAL POWER AS FUNCTION OF RMS MODULATION INDEX FOR CARRIER PHASE - MODULATED BY RANDOM NOISE

for full modulation. The reading on the power monitor is noted and the test signal removed.

The unmodulated carrier level is observed on the spectrum analyzer and a signal from the HP-618 generator at the same level and frequency is fed into the spectrum analyzer; the reading on the calibrated dial of the generator, which is equal to the strength of the unmodulated carrier, is noted. The modulating noise level is then increased until the power monitor indicates the same level as the single test signal. The height of the carrier is noted and the HP-618 generator at the same frequency is again fed into the spectrum analyzer and the dial adjusted until the signal level from the signal generator is the same as the fully modulated carrier. The dial reading equals the modulated carrier level. Subtracting the dial reading (in decibels) obtained from the unmodulated carrier from that obtained for the modulated carrier gives the drop in carrier level caused by full modulation.

The relative magnitude of sideband to carrier for the test signal is calculated based on the

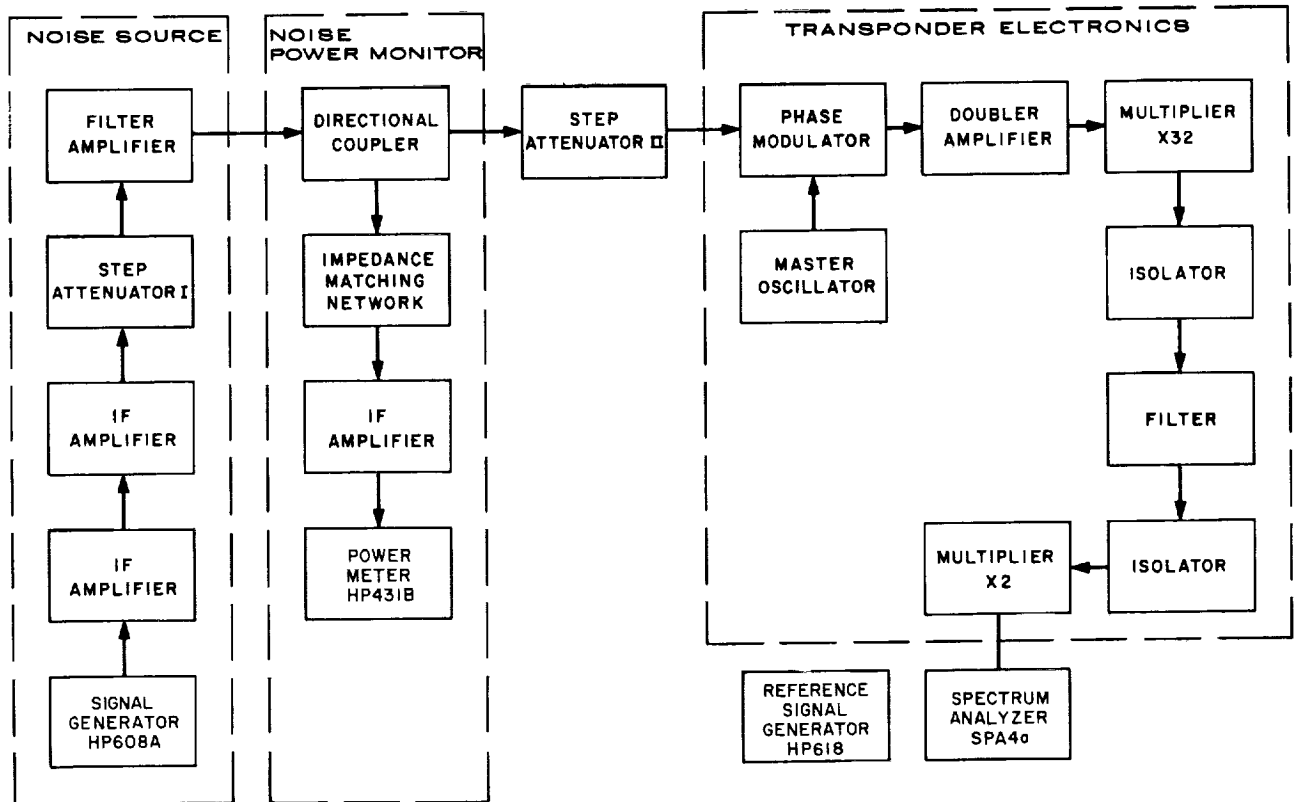


FIGURE 3-4. DEMONSTRATION SETUP BLOCK DIAGRAM

determination that 40 test tones at a peak modulation index of 0.002 radian per test tone at the output of the phase modulator will simulate full modulation. At the output of the multiplier chain, $0.002 \times 2 \times 32 \times 2$ peak radians per test tone = 0.256 peak radians per test tone.

The equivalent test signal required to determine the necessary modulating noise level is one in which the ratio of first sideband to carrier at the output is thus 1.25. The power monitor reading for the test signal was 1.24 milliwatts. The unmodulated carrier level was -14 dbm.

By removing attenuation at step attenuator 1 in the noise source, noise modulation was increased until the power monitor indicated 1.4 milliwatts (the closest level to 1.24 milliwatts attainable due to the discrete steps of the attenu-

ator). The modulated carrier level was -22 dbm. Therefore, the drop in carrier level was 8 db.

Comparison of Measured Carrier Level with Theoretical Carrier Level. When the test signal was 1.24 milliwatts, the modulation index was 1.6 radians (peak) or 1.15 radians rms. However, when the noise modulation source was substituted, it was necessary to increase the modulating power to 1.4 milliwatts. The resulting rms modulation index was therefore

$$\bar{\Theta} = 1.15 \sqrt{\frac{1.4}{1.24}} = 1.22$$

or

$$\bar{\Theta}^2 = 1.49$$

In the analysis, it was shown that

$$10 \log \frac{P_c}{P_{tot}} = -4.34 \bar{\Theta}^2 \text{ decibels} \\ = -6.5 \text{ decibels}$$

This theoretical value agrees within 1.5 db of the measured value of -8 db.

Intermodulation and Thermal Noise in Ground-to-Spacecraft Link. The noise at the output of the Advanced Syncom ground receiving system will be of several types, with thermal noise and intermodulation noise predominating. The CCIR makes recommendations on the limits of the total amount of system noise but does not attempt to divide the allotted amount among individual components of the system. As the system design progresses, however, it becomes essential to assign reasonable noise objectives to individual system components, such as transmitters and receivers, based on both the state of the art in component design and optimum theoretical design considerations. The latter is considered in this discussion.

The Tenth Plenary Session of the CCIR approved a number of Study Group IV documents relating to satellite communications. Document No. 2173, *Active Communication-Satellite Systems for Frequency Division Multiplex Telephony—Allowable Noise Power in the Basic Hypothetical Reference Circuit*, recommends:

- "1) that the psophometrically weighted noise power at a point of zero relative level in any telephone circuit in the basic hypothetical reference circuit as defined in Recommendation . . . (Document 337) should not exceed the provisional values of 10,000 pW mean in any hour, and 80,000 pW, one-minute mean, for more than 0.2 percent of any month;

- 2) that the higher noise levels that may be tolerated for very small percentages of the time should be the subject of further study;
- 3) that the following notes should be regarded as part of the Recommendation:
- Note 1: Noise in the multiplex equipments is excluded from the above."

Only the 10,000-picowatt figure stated in the above recommendation will be considered at this time.

At the "point of zero relative level" mentioned in the CCIR recommendation, the level of a test tone used as a reference signal has been established as 0 dbm or 10^{-3} watt. Therefore, a test-tone-to-noise ratio (TT/N_p) is established, in effect, at the point of zero relative level. The value of this ratio is

$$\frac{TT}{N_p} = \frac{10^{-3}}{10,000 \times 10^{-12}} = 10^5 \text{ or } 50 \text{ db} \quad (3-1)$$

By establishing a suitable test-tone level at the output of the ground receiving system in the Advanced Syncom system, it may be made a point of zero relative level, and therefore a point at which the test-tone-to-noise ratio should be 50 db.

The Advanced Syncom communications system consists of two basic parts, the ground-to-spacecraft link and the spacecraft-to-ground link. It has been shown** that the noise contributed by each of these links should be approximately equal. In terms of the CCIR recommendation, this means that about half of the total system noise, or 5000 picowatts, should be assigned to each link. Furthermore, since noise-to-test-tone ratios in the overall system may be added directly

* Additional notes are omitted.

** D. Braverman and D. D. Williams, *op. cit.*

3. Communication System Design

$$\left(\frac{N}{TT}\right)_o = \left(\frac{N}{TT}\right)_{GS} + \left(\frac{N}{TT}\right)_{SG} = -50 \text{ db} \quad (3-2)$$

where

$$\left(\frac{N}{TT}\right)_o = \text{noise-to-test-tone ratio at output of ground receiver}$$

$$\left(\frac{N}{TT}\right)_{GS} = \text{noise-to-test-tone ratio in ground-to-spacecraft link}$$

$$\left(\frac{N}{TT}\right)_{SG} = \text{noise-to-test-tone ratio in spacecraft-to-ground link}$$

Since

$$\left(\frac{N}{TT}\right)_{GS} = \left(\frac{N}{TT}\right)_{SG} \quad (3-3)$$

it is evident that

$$\left(\frac{N}{TT}\right)_{GS} = -3 \text{ db} - 50 \text{ db} = -53 \text{ db} \quad (3-4)$$

or

$$\left(\frac{TT}{N}\right)_{GS} = +53 \text{ db} \quad (3-5)$$

The noise in Equation 3-5, however, is psophometrically weighted noise power, and it is generally more convenient to use white gaussian noise in analysis. Psophometrically weighted noise has a level that is 2.5 db below that of white gaussian noise because of the filter characteristics of telephone circuits. Therefore, if N in Equation 3-5 is interpreted as white noise,

$$\left(\frac{TT}{N}\right)_{GS} = 53 - 2.5 = 50.5 \text{ db} \quad (3-6)$$

As will be shown later, this ratio, $(TT/N)_{GS}$, severely limits the permissible intermodulation noise from the ground transmitter when the system is statistically loaded with 600 channels. The transmitter power tube design problem may be eased considerably if the use of compandors

is considered in the system. Compandors permit a significant increase in the noise power level while maintaining the output signal-to-noise ratio at its original level, based on subjective listener tests. A factor of 15 db is often used as the compandor improvement factor, and this appears to be a conservative value. Therefore, $(TT/N)_{GS}$ may be decreased by 15 db to give

$$\left(\frac{TT}{N}\right)_{GS} = 35.5 \text{ db} \quad (3-7)$$

The two main noise contributions in the ground-to-spacecraft link are thermal noise at the spacecraft receiver and intermodulation noise at the ground transmitter. The CCIR excludes noise in the multiplex equipments from the 10,000-picowatt noise limit. For the ground-to-spacecraft link

$$\left(\frac{N}{TT}\right)_{GS} = \left(\frac{N}{TT}\right)_T + \left(\frac{N}{TT}\right)_X \quad (3-8)$$

where

$$\left(\frac{N}{TT}\right)_T = \text{thermal noise-to-test-tone ratio at spacecraft receiver}$$

$$\left(\frac{N}{TT}\right)_X = \text{ground transmitter intermodulation noise-to-test-tone ratio}$$

It has been shown* that the following relations result from an optimum system design, assuming a statistically loaded, 600-channel system:

$$\left(\frac{N}{TT}\right)_T = \frac{2}{3} \left(\frac{N}{TT}\right)_{GS} \quad (3-9)$$

$$\left(\frac{N}{TT}\right)_X = \frac{1}{2} \left(\frac{N}{TT}\right)_T \quad (3-10)$$

*Braverman and Williams, *op cit*, Equations 5-22 and 5-23.

Therefore,

$$\left(\frac{N}{TT}\right)_x = \frac{1}{3} \left(\frac{N}{TT}\right)_{GS} \quad (3-11)$$

From Equation 3-7,

$$\left(\frac{N}{TT}\right)_{GS} = -35.5 \text{ db} \quad (3-12)$$

Therefore,

$$\left(\frac{N}{TT}\right)_x = -35.5 - 4.8 = -40.3 \text{ db}$$

or

$$\left(\frac{TT}{N}\right)_x = +40.3 \text{ db} \quad (3-13)$$

and

$$\left(\frac{N}{TT}\right)_T = -35.5 - 1.8 = -37.3 \text{ db}$$

or

$$\left(\frac{TT}{N}\right)_T = +37.3 \text{ db} \quad (3-14)$$

Equations 3-13 and 3-14 may be summarized in words as follows:

- 1) *Ground transmitter intermodulation noise.* The single-channel test-tone-to-intermodulation-noise (unweighted) ratio at the output of the ground transmitter should not be less than +40.3 db if the compandor improvement factor is 15 db.
- 2) *Spacecraft receiver thermal noise.* The test-tone-to-thermal-noise (unweighted) ratio at the spacecraft receiver in a noise bandwidth of 3.1 kc should not be less than 37.3 db if the compandor improvement factor is 15 db.

These results assume both an optimum choice of test-tone power as derived by Braverman and

Williams and a statistically loaded, 600-channel system.

The CCIR states that the test tone psophometrically weighted noise ratio in a 600-channel system can be converted to the noise power ratio (NPR) by subtracting about 19 db from the test tone to psophometrically weighted noise ratio.* The NPR is defined as the ratio of noise in a freely transmitted band to that in the measurement slot in which the noise at the input to the transmitter is suppressed by at least 80 db. It is often convenient to make NPR measurements using a slot noise test equipment. Therefore,

$$NPR = \left(\frac{TT}{N}\right)_{PW} - 19 \text{ db} \quad (3-15)$$

where $(TT/N)_{PW}$ = test tone to psophometrically weighted noise ratio.

It is evident that the test tone to psophometrically weighted noise ratio is 2.5 db greater than the test tone to unweighted noise ratio. Therefore, 2.5 db should be added to all values of $(TT/N)_T$ and $(TT/N)_x$ or other ratios including *unweighted* noise when using Equation 3-15.

Table 3-7 summarizes test tone to unweighted noise ratios and noise power ratios in a statistically loaded 600-channel system (ground-to-spacecraft link) for several compandor improvement factor values.** If fewer channels are used, the intermodulation noise per channel will be less.

The ratios $(TT/N)_x$, $(TT/N)_T$, and NPR are plotted as a function of compandor improvement

*W. J. Bray, *op cit*, p. 196.

**In Table 2-1 of the August 1963 Advanced Syncom Monthly Progress Report the psophometric weighting factor of 2.5 db should be added to all values in the NPR column. In Figure 2-1 of that report, the NPR curve should be raised by 2.5 db.

TABLE 3-7. TEST-TONE-TO-UNWEIGHTED-NOISE AND NOISE POWER RATIOS FOR VARIOUS COMPANDOR IMPROVEMENT FACTORS

Compandor Improvement Factor, db	$\left(\frac{T}{N}\right)_{gs}$ db	$\left(\frac{T}{N}\right)_x$ db	$\left(\frac{T}{N}\right)_r$ db	NPR at Transmitter Output caused by Intermodulation Noise, db
0	50.5	55.3	52.3	38.8
6	44.5	49.3	46.3	32.8
8.4	41.7	46.5	43.5	30.0
10	40.5	45.3	42.3	28.8
13	37.5	42.3	39.3	25.8
15	35.5	40.3	37.3	23.8
17	33.5	38.3	35.3	21.8
20	30.5	35.3	32.3	18.8
23	27.5	32.5	29.3	15.8

factor in Figure 3-5 for a 600-channel system. A frequently used value of 30 db for NPR requires a compandor improvement factor of about 8.4 db in a statistically loaded, 600-channel system.

RF INTERFERENCE ANALYSIS

An analysis of possible areas and effects of RF interference in the Advanced Syncom spacecraft has been made. Five segments of the included frequency spectrum were uncovered to which more than one frequency source contributes. The effects of these five frequency segments on single quadrant and multiple quadrant operation are delineated in Table 3-8.

This analysis does not include the effects of intermediate harmonics and subharmonics generated by frequency multipliers nor does it take into account undesired spectral components derived from nonlinear circuit elements. The study of interference from these sources will be accomplished during subsequent analyses.

In summary, although five areas of frequency overlap were discovered, it is not expected that any of these singly or in combination will significantly degrade operation of the spacecraft.

RF interference between transponders is not expected to be a problem for the following reasons:

- 1) There is a large physical separation of transponders, within the spacecraft (approximately 3.5 feet between adjacent transponders and 5 feet between diametrically opposed transponders).
- 2) All RF cabling between multiplexers, traveling-wave tubes, and transponders will be kept as short as possible and physically separate.
- 3) The transponder package will be enclosed in an RF shield.

RF interference within a transponder mode will be minimized by RF shielding of circuit elements, by keeping inter-element connections as short as possible, and by judicious placement of circuit elements themselves.

RF interference between the multiple access mode and frequency translation mode of one transponder cannot exist since both modes of one transponder can never be operated simultaneously.

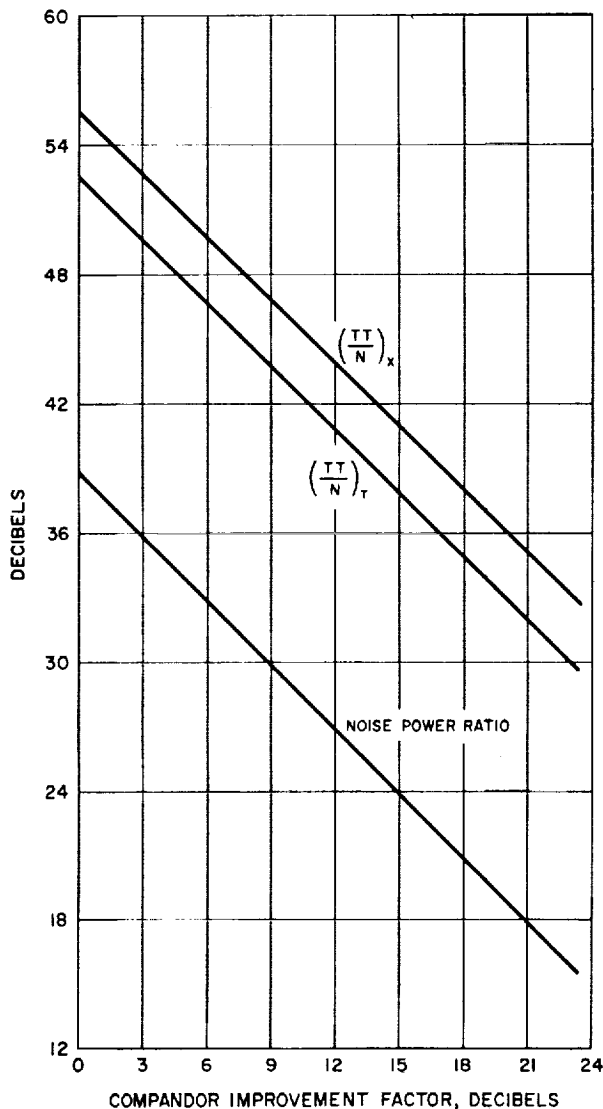


FIGURE 3-5. TEST TONE TO UNWEIGHTED NOISE AND NOISE POWER RATIOS IN GROUND-TO-SPACECRAFT LINK FOR 600-CHANNEL LOADING

Interference between the telemetry subcarrier oscillator output, ψ_2 angle counter, and command audio tones could present the following problems:

- 1) After the command decoder has received its word sync, command address, and command number, a continuous zero tone is

received until the execute tone is received. If the zero tone frequency were to blot out or bias the telemetry subcarrier oscillator (SCO) output, incorrect telemetry information (in particular, command verification) would be transmitted.

- 2) Both the command address and command number consists of 6-bit binary-coded groups such that the probability of the telemetry SCO generating a specific coded group of ones and zeros is small, but does exist.
- 3) The ψ_2 angle counter may interfere with the telemetry SCO output and the command decoder audio inputs.

These interference areas are not expected to cause a problem for the following reasons:

- 1) The command receiver/decoder, telemetry encoder, telemetry transmitter, and ψ_2 angle counter are all on physically separate circuit boards.
- 2) The command receiver and decoder are on the same circuit board. This arrangement eliminates external routing and minimizes the length of the audio signal connection between the receiver and decoder.
- 3) Shielded wire will be used for the telemetry subcarrier oscillator output.

Table 3-9 is a compilation of frequencies existing in the spacecraft with discussions of overlapping spectra. In the five subdivisions of Table 3-9, the five frequency segments in which interference might occur are displayed along with notations of possible effects.

TABLE 3-8. FREQUENCY S

Frequency Spectrum	Mutual Area	Single Quadrant Communication Effects		
		Multiple Access	Frequency Translation	Multiple Access Multiple Access
3.62 kc 10.239 kc	1			
31.6491 mc 32.6479 mc	2			1a) Master oscillator f quadrant II, is in IF passband of quadrant I 1b) Master oscillator f quadrant III, is in IF passband of quadrants and II. 1c) Master oscillator f quadrant IV, is in IF passband of quadrants II, and III.
62.3764 mc 68.490 mc	3	1) Master oscillator amplifier output f is in passband of doubler amplifier.	2) X32 input is in passband of limiter amplifier	1a) Each master oscill output frequency is in passband of all double amplifier outputs. 1b) Passbands of all d amplifiers overlap with other.
1996.0448 mc 2092.1536 mc	4	1) Local oscillator X32 output f is in passband of high-level X32 output.		
4054.4640 mc 4184.3072 mc	5			



PECTRUM OVERLAP SUMMARY

Multiple-Quadrant Communication Effects			Noncommunication Mutual Effects	
	Frequency Translation — Frequency Translation	Multiple Access — Frequency Translation	Telemetry Transmitter — Multiple Access / Frequency Translation	Mutual Effects
				1) "0" and "1" tones of command signal are in operating range of telemetry subcarrier oscillator. 2) ψ_2 angle counter register can operate at frequencies in operating range of telemetry subcarrier oscillator. 3) ψ_2 angle counter register can operate at frequencies identical to command tone frequencies for "0", "1", and "execute".
ator r- oubler- i each	2a) Each X32 input f is in passband of all limiter amplifiers. 2b) Passbands of all limiter amplifiers overlap with each other.	3a) Each multiple access master oscillator f lies in passband of all frequency translation limiter amplifiers. 3b) Each frequency translation X32 input f is in passband of all multiple access doubler amplifiers 3c) Passbands of all multiple access doubler amplifiers overlap with passbands of all frequency translation limiter amplifiers.	3) Both telemetry transmitter local oscillator frequencies are in passbands of multiple access doubler amplifier and frequency translation limiter amplifiers.	
		1a) Frequency translation X32 output f of quadrant I is in passband of multiple access high-level X32 output, quadrant II. 1b) Frequency translation X32 output f of quadrant II is in passband of multiple access high-level X32 output of quadrant II. 1c) Frequency translation X32 output f of quadrant III is in passband of multiple access high-level X32 output of quadrant IV.		
	1a) X2 output f of quadrant I is in passband of the frequency translation mode output of quadrant II. 1b) X2 output f of quadrant II is in passband of frequency translation mode output of quadrant III. 1c) X2 output f of quadrant III is in passband of frequency translation mode output of quadrant IV.	1a) X2 output f of quadrant I is in passband of multiple access mode output of quadrant II. 1b) X2 output f of quadrant II is in passband of multiple access mode output of quadrant III. 1c) X2 output f of quadrant III is in passband of multiple access mode output of quadrant IV.		



TABLE 3-9. FIVE ARE.

MUTUAL AREA 1

Quadrant	Telemetry SCO Output	Command			ψ_2 Angle Counter
		"O" Tone	"1" Tone	Execute Tone	
I through IV	4.5 kc to 15.6 kc	7.4 kc	8.6 kc	3.62 kc	3.413 kc to 10.239 kc

Interference Effects

- 1) The "O" and "1" tones of the command signal are in the operating range of the telemetry SCO.
- 2) The ψ_2 angle counter register can operate at frequencies which are in the operating range of the telemetry SCO.
- 3) The ψ_2 angle counter register can operate at frequencies identical to the command tone frequencies for "O," "1" and "execute."

MUTUAL AREA 2

Quadrant	Multiple Access Master Oscillator Output, mc	Multiple Access, IF Passband, mc
I	31.1882	31.191 — 36.691
II	31.6491	31.648 — 37.148
III	32.1870	32.196 — 37.696
IV	32.6479	32.66 — 38.16

Multiple Quadrant Effects

- 1a) The multiple access master oscillator frequency of quadrant II is in the multiple access mode IF passband of quadrant I.
- b) The multiple access master oscillator frequency of quadrant III is in the multiple access mode IF passband of quadrants I and II.
- c) The multiple access master oscillator frequency of quadrant IV is in the multiple access mode IF passband of quadrants I, II, and III.

MUTUAL AREA 3

Quadrant	Multiple Access Doubler Amplifier Output, mc	Multiple Access Master Oscillator Amplifier Output, mc	Frequency Translation Limiter Amplifier Output, mc	Frequency Translation X32 Input, mc	Telemetry Transmitter Local Oscillators, mc
I	62.3764 $\pm$ 12.5	62.3764	62.371 $\pm$ 12.5	63.3510	68.235
II	63.2982 $\pm$ 12.5	63.2982	63.2962 $\pm$ 12.5	64.2872	68.490
III	64.3740 $\pm$ 12.5	64.3740	64.3608 $\pm$ 12.5	65.3798	68.235
IV	65.2958 $\pm$ 12.5	65.2958	65.2899 $\pm$ 12.5	66.3160	68.490

Single Quadrant Effects

- 1) A multiple access master oscillator amplifier output is in the passband of the multiple access doubler amplifier output.
- 2) The frequency-translation X32 input frequency is in the passband of the frequency-translation limiter amplifier output.
- 3) The telemetry transmitter local oscillator frequencies are in the passbands of both the multiple access doubler amplifier outputs and the frequency translation limiter amplifier outputs.

Multiple Quadrant Effects

- 1a) Each multiple access master oscillator output frequency is in the passband of all multiple access doubler amplifier outputs.
- b) The passbands of all multiple access doubler amplifier outputs overlap with each other to some extent.
- 2a) Each frequency translation X32 input frequency is in the passband of all frequency translation limiter amplifier outputs.
- b) The passbands of all frequency translation limiter amplifier outputs overlap with each other to some extent.
- 3a) Each multiple access master oscillator output is in the passband of the frequency translation limiter amplifier outputs.
- b) Each frequency translation X32 input frequency is in the passband of all multiple access doubler amplifier outputs.
- c) The passbands of all multiple access doubler amplifier outputs overlap with the passbands of all frequency translation limiter amplifier outputs, to a limited extent.

AS OF SPECTRAL OVERLAP

MUTUAL AREA 4

Quadrant	Multiple Access High Level X32 Output, mc	Multiple Access Local Oscillator X32 Output, mc	Frequency Translation X32 Output, mc
I	1996.0448 $\pm$ 12.5	1996.0448	2027.2320
II	2025.5424 $\pm$ 12.5	2025.5424	2057.1904
III	2059.9680 $\pm$ 12.5	2059.9680	2092.1536
IV	2089.4656 $\pm$ 12.5	2089.4656	2122.1133

Single Quadrant Effect

- 1) The multiple access local oscillator X32 output is in the passband of the multiple access high level X32 output.

Multiple Quadrant Effects

- 1a) The frequency translation X32 output of quadrant I is in the passband of the multiple access high level X32 output of quadrant II.
 b) The frequency translation X32 output of quadrant II is in the passband of the multiple access high level X32 output of quadrant III.
 c) The frequency translation X32 output of quadrant III is in the passband of the multiple access high level X32 output of quadrant IV.

MUTUAL AREA 5

Quadrant	Multiple Access Mode Output, mc	Frequency Translation Mode Output, mc	Frequency Translation X2 Output, mc
I	3992.0896 $\pm$ 12.5	3992.0950 $\pm$ 12.5	4054.4640
II	4051.0848 $\pm$ 12.5	4051.0846 $\pm$ 12.5	4114.3808
III	4119.9360 $\pm$ 12.5	4119.9464 $\pm$ 12.5	4184.3072
IV	4178.9312 $\pm$ 12.5	4178.9367 $\pm$ 12.5	4244.2266

Multiple Quadrant Effects

- 1a) The frequency translation X2 output of quadrant I is in the output signal bandwidth of both modes of quadrant II.
 b) The frequency translation X2 output of quadrant II is in the output signal bandwidth of both modes of quadrant III.
 c) The frequency translation X2 output of quadrant III is in the output signal bandwidth of both modes of quadrant IV.

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3. Communication System Design

Table 3-10 is a complete list of frequencies above 1.5 kc existing in the spacecraft. This frequency table is divided into the following equipment group frequency sources for each spacecraft quadrant: multiple access mode, frequency translation mode, telemetry transmitter, telem-

etry encoder, command receiver, ψ_2 angle counter, and central timer.

In addition to these tables, an RF spectrum diagram (Figure 3-6) and explanatory functional block diagrams (Figures 3-7 through 3-12) are presented.

TABLE 3-10. FREQUENCY TABLE

Equipment Group	Signal Sample Points*	Frequency, mc	Power Level, mw	Power Level, dbm	Bandwidth, mc	Functional Block Diagram Figure
QUADRANT I — INPUT FREQUENCY 6019.325 MC						
Multiple access mode	1) f_{in}	6019.325	2.9×10^{-5}	-93.4	5.3	3-6, 3-7
	2) f_{ir} — input to phase modulator	31.191 to 36.691	10^{-5}	-50	5.3	3-6, 3-7b
	3) Doubler amplifier output	62.3764	100	20	11.0	3-6, 3-7b
	4) High level X32 output	1996.0448	11	10.42	16.0	3-6, 3-7b
	5) f_{out} (antenna)	3992.0896	1.6×10^1	32	25	3-6, 3-7
	6) Master oscillator output	31.1882	7	8.5		3-6, 3-7b
	7) Master oscillator amplifier output	62.3764	240	23.85		3-6, 3-7b
	8) Low level X32 output	1996.0448	35	15.5		3-6, 3-7b
	9) X3 output	5988.1344	3.9	5.94		3-6, 3-7b
Frequency translation mode	1) f_{in}	6019.325	2.9×10^{-5}	-93.4	25	3-6, 3-7
	2) Limiter amplifier output					
	Signal	62.371	30	14.85	25	3-6, 3-7a
	Beacon**	47.5133	7.5	8.75		3-6, 3-7a
	3) X2 output	4054.464	9	9.56		3-6, 3-7a
	4) f_{out} (antenna)	3992.095	1.6×10^1	32	25	3-6, 3-7
	Beacon output (antenna)**	4006.953	400	26		3-6, 3-7a
	5) Master oscillator frequency	15.83776	$\sim \frac{10^1}{2}$	27		3-6, 3-7a
Telemetry transmitter	6) X32 input	63.3510	350	25.45		3-6, 3-7a
	7) X32 output	2027.232	70	18.5		3-6, 3-7a
	8) X3 output	6081.696	3.9	5.9		3-6, 3-7a
	1) Power amplifier	68.235	2.5×10^1	34	~ 0.100	3-9
Telemetry encoder	2) Antenna output	136.470	10^1	30	~ 0.100	3-9
	Subcarrier oscillator	0.0045 to 0.0156 and 0.00362	0.045	-13.5		3-8
Command receiver	1) Local oscillator	118.31	3	4.9		3-10
	2) Command — IF	29.95	3	4.9	0.075	3-10
	3) Input signal	148.26	10^{-10}	-100	0.075	3-10
	4) Tones					
	Zero	0.0074	~ 10	10		3-10
	Ones	0.0086	~ 10	10		3-10
ψ_2 angle counter	Execute	0.00362	~ 10	10		3-10
	Output of angle encoder register	0.003413 to 0.010239	~ 0.05	-13		3-11
Central timer	Tuning fork oscillator	0.001553	$\frac{1}{4}$	-6		3-12

*For the multiple access and frequency modes, the sample point designations are referenced from Figure 3-7.
 **Beacon output power levels were computed in the absence of a communications signal.

TABLE 3-10. (Continued)

Equipment Group	Signal Sample Points	Frequency, mc	Power Level, mw	Power Level, dbm	Bandwidth, mc	Functional Block Diagram Figure
Multiple access mode	QUADRANT II — INPUT FREQUENCY 6108.275 MC					
	1) f_{in}	6108.275	2.9×10^{-5}	-93.4	5.3	3-6, 3-7
	2) f_{in} — input to phase modulator	31.648 to 37.48	10^{-5}	-50	5.3	3-6, 3-7b
	3) Doubler amplifier output	63.2982	100	20	11.0	3-6, 3-7b
	4) High level X32 output	2025.5424	11	10.4	16.0	3-6, 3-7b
	5) f_{out} (antenna)	4051.0848	1.6×10^3	32	25	3-6, 3-7
	6) Master oscillator output	31.6491	7	8.5		3-6, 3-7b
	7) Master oscillator amplifier output	63.2982	240	23.9		3-6, 3-7b
	8) Low level X32 output	2025.5424	35	15.5		3-6, 3-7b
	9) X3 output	6076.6272	3.9	5.9		3-6, 3-7b
Frequency translation mode	1) f_{in}	6108.275	2.9×10^{-5}	-93.4	25	3-6, 3-7
	2) Limiter amplifier output					
	Signal	62.2962	30	14.9	25	3-6, 3-7a
	Beacon**	48.2154	7.5	8.75		3-6, 3-7a
	3) X2 output	4114.3808	9	9.6		3-6, 3-7a
	4) f_{out} (antenna)	4051.0846	1.6×10^3	32	25	3-6, 3-7
	Beacon output (antenna)**	4066.1654	400	26		3-6, 3-7a
	5) Master oscillator frequency	16.07181	$\sim \frac{10^3}{2}$	27		3-6, 3-7a
Telemetry transmitter	6) X32 input	64.2872	350	25.5		3-6, 3-7a
	7) X32 output	2057.1904	70	18.5		3-6, 3-7a
	8) X3 output	6171.5712	3.9	5.9		3-6, 3-7a
Telemetry encoder	1) Power amplifier	68.235	2.5×10^3	34	~ 0.100	3-9
	2) Antenna output	136.470	10^3	30	~ 0.100	3-9
Telemetry encoder	Subcarrier oscillator	0.0045 to 0.0156 and 0.00362	0.045	-13.5		3-8
Command receiver	1) Local oscillator	118.31	3	4.9		3-10
	2) Command — IF	29.95	3	4.9	0.075	3-10
	3) Input signal	148.26	10^{-10}	-100	0.075	3-10
	4) Tones					
	Zero	0.0074	~ 10	10		3-10
	Ones	0.0086	~ 10	10		3-10
ψ_2 angle counter	Execute	0.00362	~ 10	10		3-10
ψ_2 angle counter	Output of angle encoder register	0.003413 to 0.010239	~ 0.05	-13		3-11
Central timer	Tuning fork oscillator	0.001553	$\frac{1}{4}$	-6		3-12

*For the multiple access and frequency modes, the sample point designations are referenced from Figure 3-7.

**Beacon output power levels were computed in the absence of a communications signal.

3. Communication System Design

TABLE 3-10. (Continued)

Equipment Group	Signal Sample Points	Frequency, mc	Power Level, mw	Power Level, dbm	Bandwidth, mc	Functional Block Diagram Figure
QUADRANT III — INPUT FREQUENCY 6212.05 MC						
Multiple access mode	1) f_{in}	6212.05	2.9×10^{-5}	-93.4	5.3	3-6, 3-7
	2) f_{if} — input to phase modulator	32.196 to 37.696	10^{-5}	-50	5.3	3-6, 3-7b
	3) Doubler amplifier output	64.3740	100	20	11.0	3-6, 3-7b
	4) High level X32 output	2059.9680	11	10.4	16.0	3-6, 3-7b
	5) f_{out} (antenna)	4119.936	1.6×10^3	32	25	3-6, 3-7
	6) Master oscillator output	32.1870	7	8.5		3-6, 3-7b
	7) Master oscillator amplifier output	64.3740	240	23.9		3-6, 3-7b
	8) Low level X32 output	2059.9610	35	15.5		3-6, 3-7b
	9) X3 output	6179.9040	3.9	5.9		3-6, 3-7b
Frequency translation mode	1) f_{in}	6212.05	2.9×10^{-5}	-93.4	25	3-6, 3-7
	2) Limiter amplifier output					
	Signal	64.3608	30	14.9	25	3-6, 3-7a
	Beacon*	49.0349	7.5	8.75		3-6, 3-7a
	3) X2 output	4184.3072	9	9.6		3-6, 3-7a
	4) f_{out} (antenna)	4119.9464	1.6×10^3	32	25	3-6, 3-7
	Beacon output (antenna)*	4135.2723	400	26		3-6, 3-7a
	5) Master oscillator frequency	16.34496	$\sim \frac{10^3}{2}$	27		3-6, 3-7a
	6) X32 input	65.3798	350	25.5		3-6, 3-7a
Telemetry transmitter	7) X32 output	2092.1536	70	18.5		3-6, 3-7a
	8) X3 output	6276.4608	3.9	5.9		3-6, 3-7a
Telemetry encoder	1) Power amplifier	68.490	2.5×10^3	34	~ 0.100	3-9
	2) Antenna output	136.980	10^3	30	~ 0.100	3-9
Telemetry encoder	Subcarrier oscillator	0.0045 to 0.0156 and 0.00362	0.045	-13.5		3-8
Command receiver	1) Local oscillator	118.31	3	4.9		3-10
	2) Command — IF	29.95	3	4.9	0.075	3-10
	3) Input signal	148.26	10^{-10}	-100	0.075	3-10
	4) Tones					
	Zero	0.0074	~ 10	10		3-10
	Ones	0.0086	~ 10	10		3-10
	Execute	0.00362	~ 10	10		3-10
ψ_2 angle counter	Output of angle encoder register	0.003413 to 0.010239	~ 0.05	-13		3-11
Central timer	Tuning fork oscillator	0.001553	$\frac{1}{4}$	-6		3-12

*Beacon output power levels were computed in the absence of a communications signal.

TABLE 3-10. (Continued)

Equipment Group	Signal Sample Points	Frequency, mc	Power Level, mw	Power Level, dbm	Bandwidth, mc	Functional Block Diagram Figure
Multiple access mode	QUADRANT IV — INPUT FREQUENCY 6301.05 MC					
	1) f_{in}	6301.05	2.9×10^{-5}	-93.4	5.3	3-6, 3-7
	2) f_{in} — input to phase modulator	32.66 to 38.16	10^{-5}	-50	5.3	3-6, 3-7b
	3) Doubler amplifier output	65.2958	100	20	11.0	3-6, 3-7b
	4) High level X32 output	2089.4656	11	10.4	16.0	3-6, 3-7b
	5) f_{out} (antenna)	4178.9312	1.6×10^3	32	25	3-6, 3-7
	6) Master oscillator output	32.6479	7	8.5		3-6, 3-7b
	7) Master oscillator amplifier output	65.2958	240	23.9		3-6, 3-7b
	8) Low level X32 output	2089.4656	35	15.5		3-6, 3-7b
	9) X3 output	6268.3968	3.9	5.9		3-6, 3-7b
Frequency translation mode	1) f_{in}	6301.05	2.9×10^{-5}	-93.4	25	3-6, 3-7
	2) Limiter amplifier output					
	Signal	65.2899	30	14.9	25	3-6, 3-7a
	Beacon*	49.7370	7.5	14.9		3-6, 3-7a
	3) X2 output	4244.2266	9	9.6		3-6, 3-7a
	4) f_{out} (antenna)	4178.9367	1.6×10^3	32	25	3-6, 3-7
	Beacon output (antenna)*	4194.4896	400	26		3-6, 3-7a
	5) Master oscillator frequency	16.57901	$\sim \frac{10^3}{2}$	27		3-6, 3-7a
Telemetry transmitter	1) Power amplifier	68.490	2.5×10^3	34	~ 0.100	3-9
	2) Antenna output	136.980	10^3	30	~ 0.100	3-9
Telemetry encoder	Subcarrier oscillator	0.0045 to 0.0156 and 0.00362	0.045	-13.45		3-8
Command receiver	1) Local oscillator	118.31	3	4.9		3-10
	2) Command — IF	29.95	3	4.9	0.075	3-10
	3) Input signal	148.26	10^{-10}	-100	0.075	3-10
	4) Tones					
	Zero	0.0074	~ 10	10		3-10
	Ones	0.0086	~ 10	10		3-10
ψ_2 angle counter	Execute	0.00362	~ 10	10		3-10
	Output of angle encoder register	0.003413 to 0.010239	~ 0.05	-13		3-11
Central timer	Tuning fork oscillator	0.001553	$\frac{1}{4}$	-6		3-12

*Beacon output power levels were computed in the absence of a communications signal.

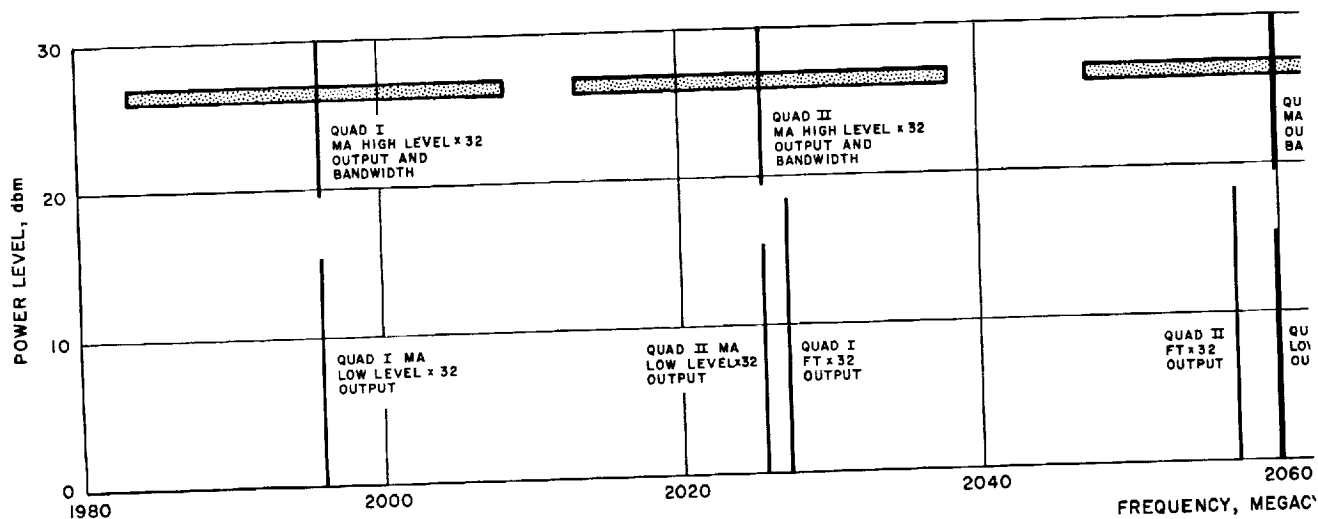
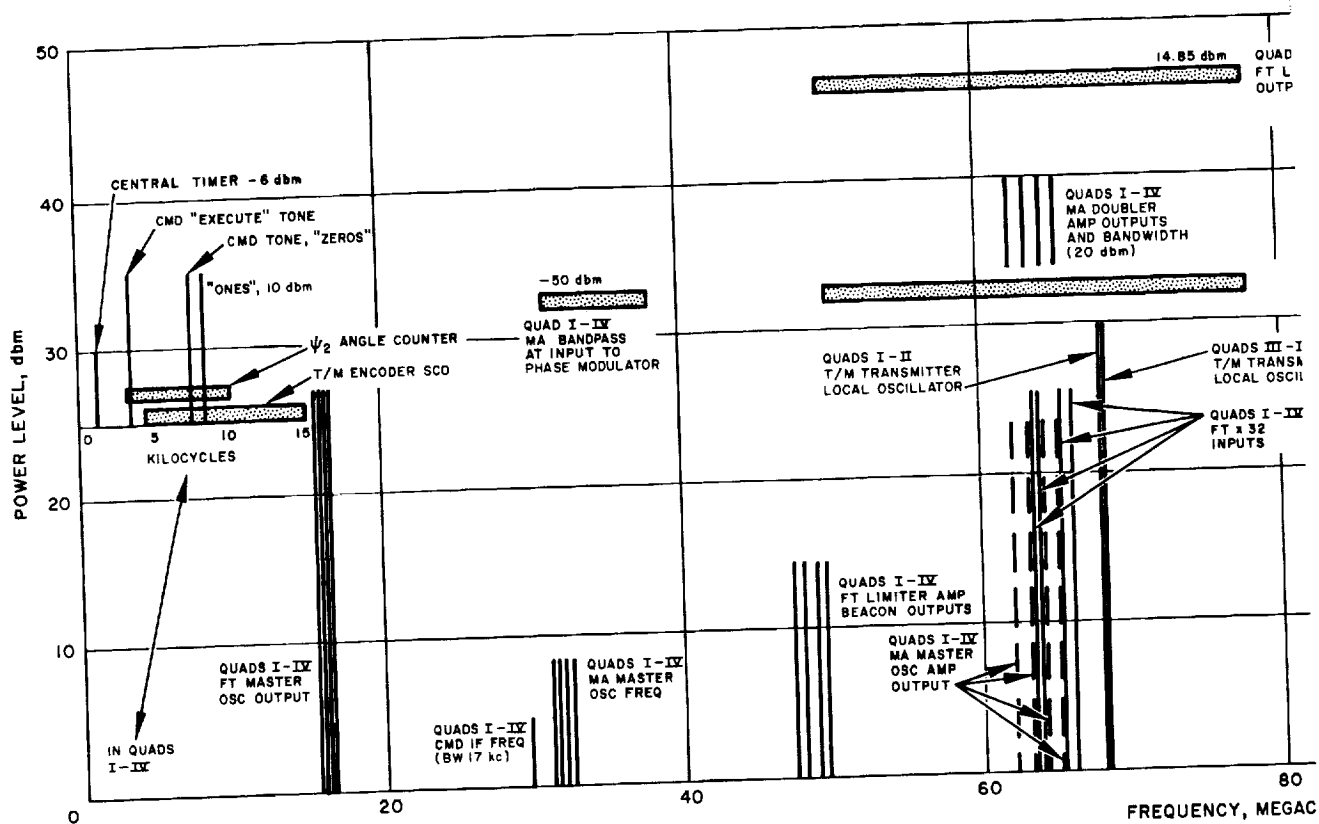
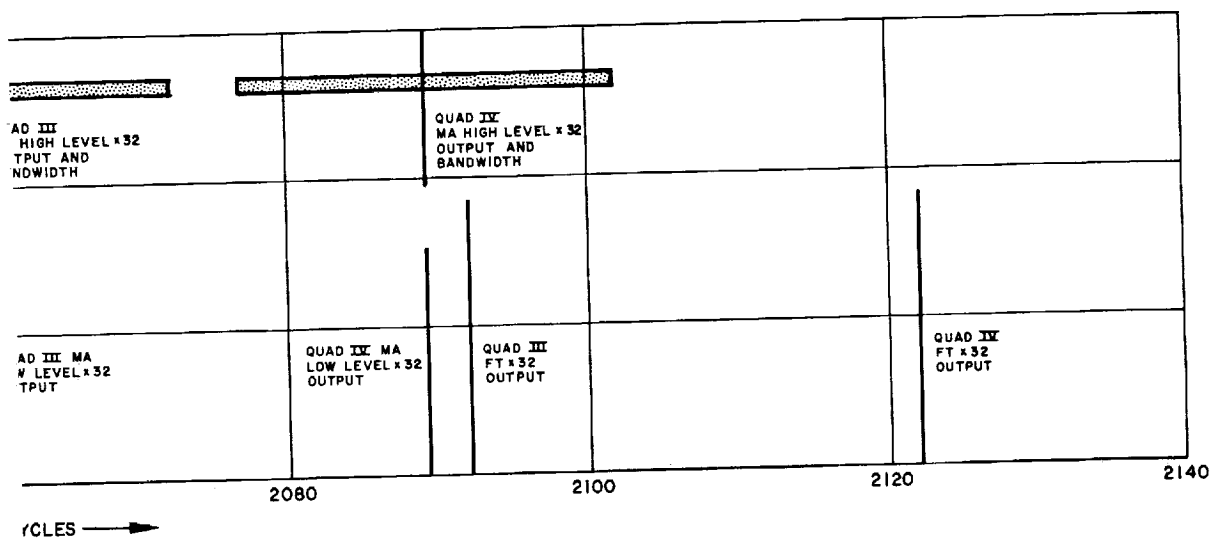
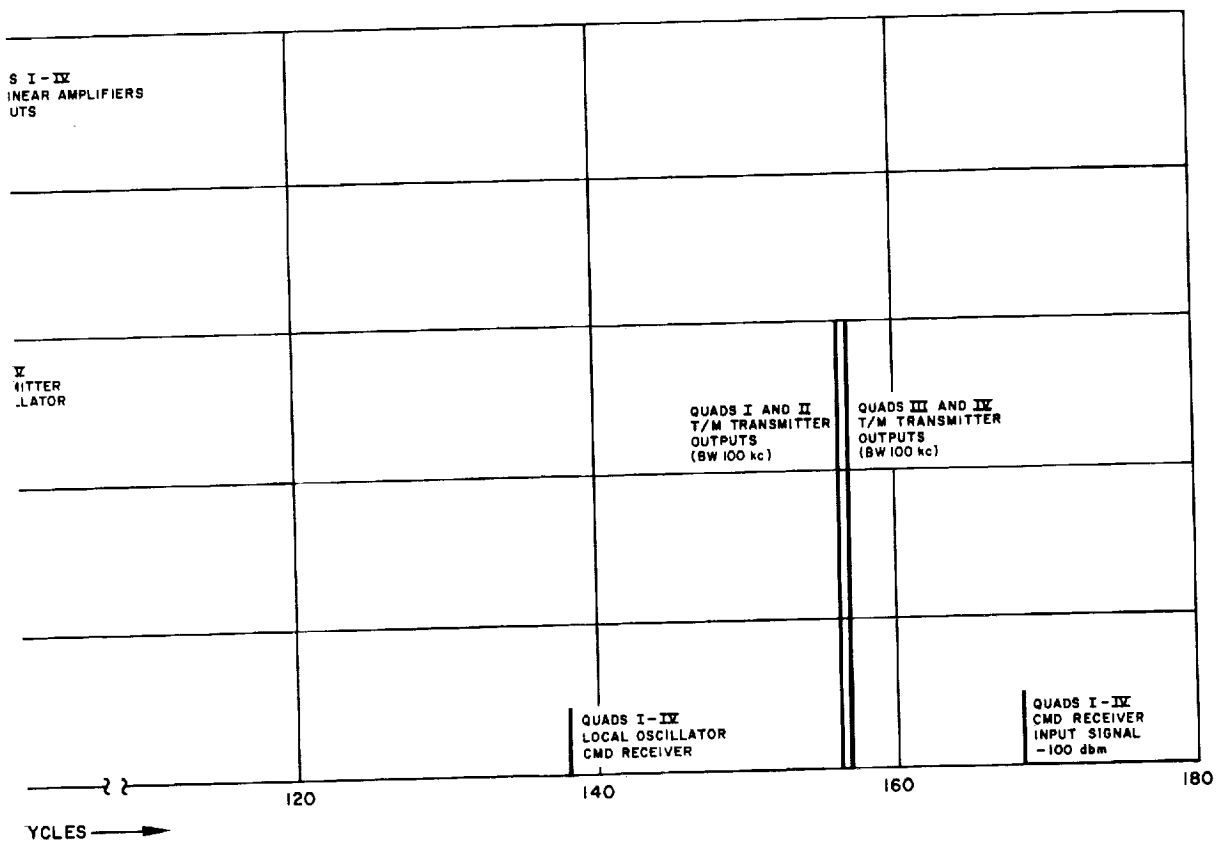


FIGURE 3-6. RF SPECTRUM DIAGRAM





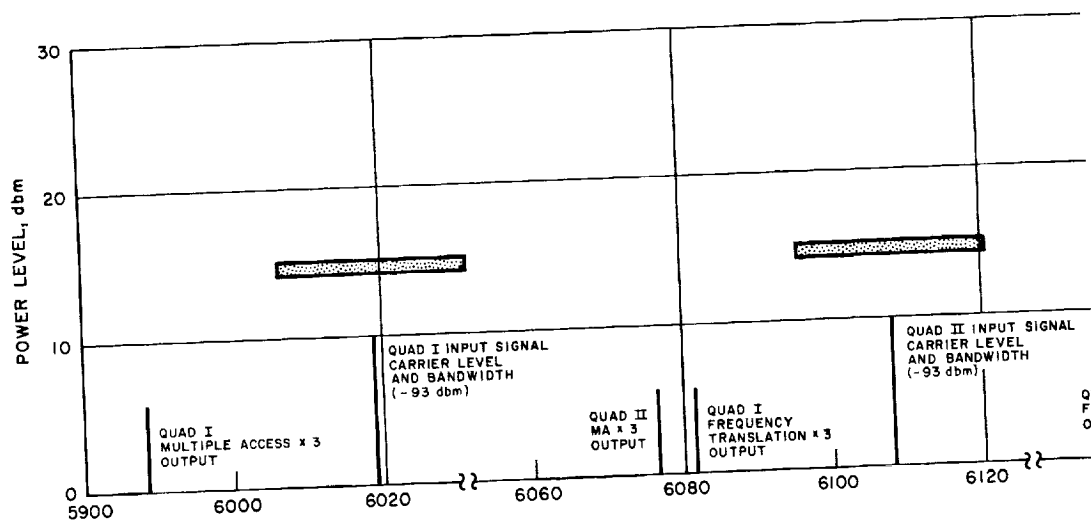
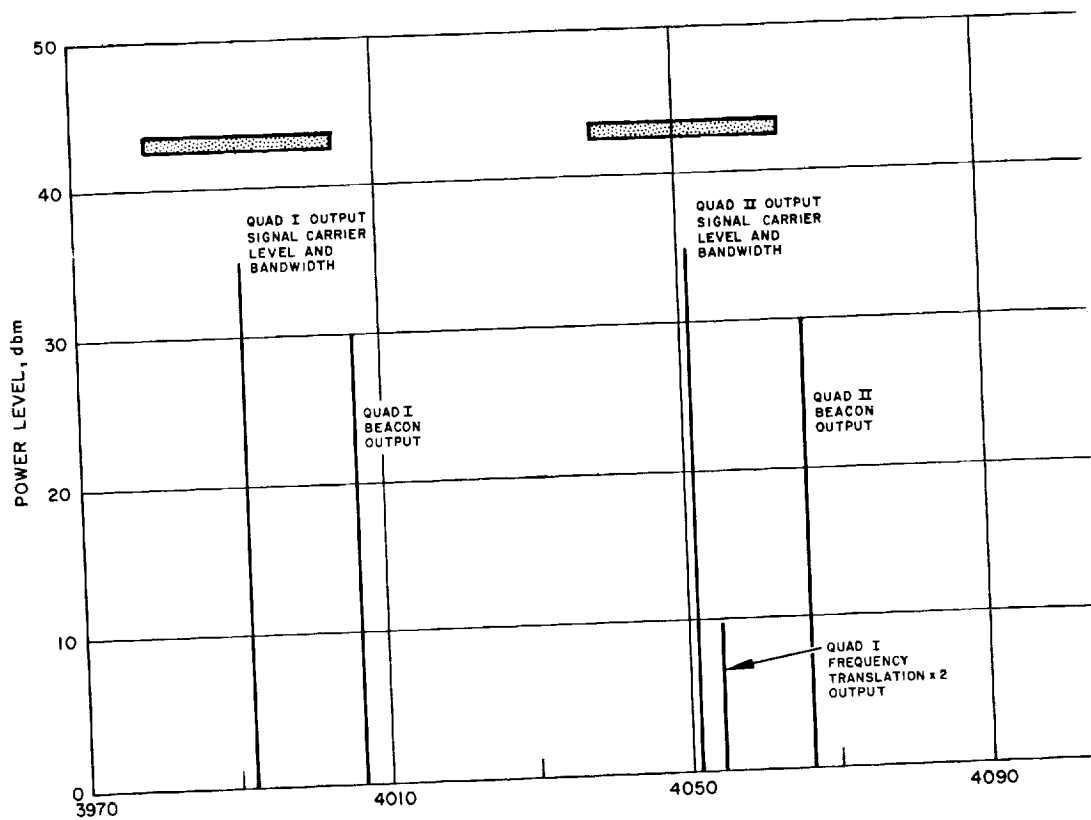
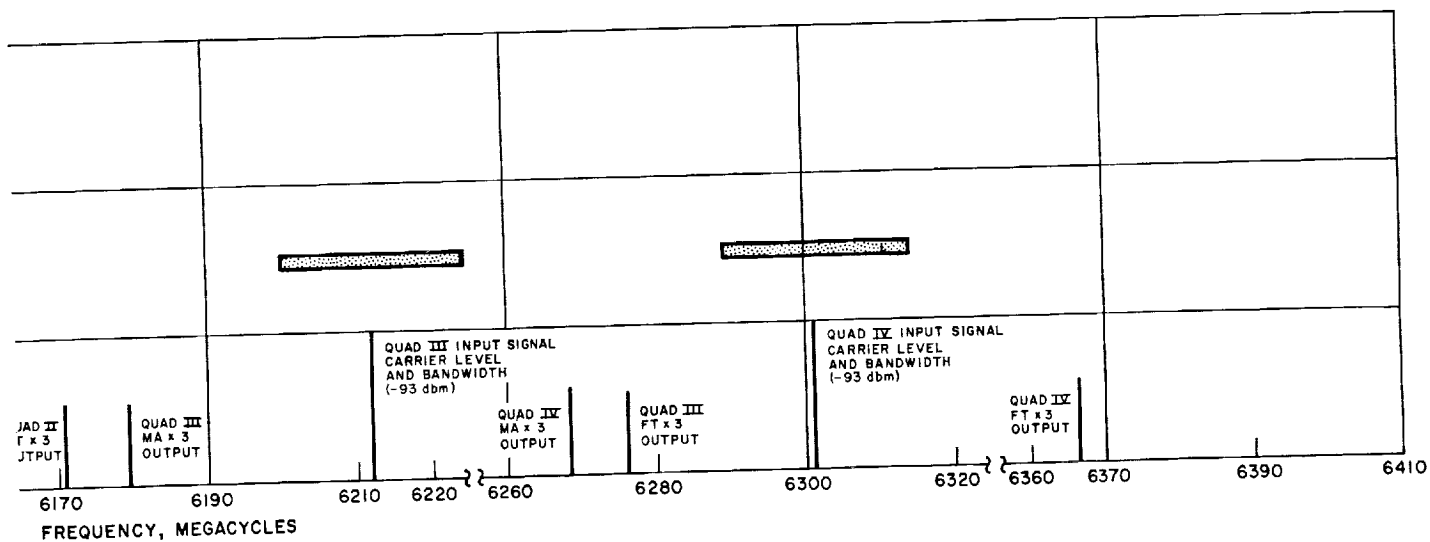
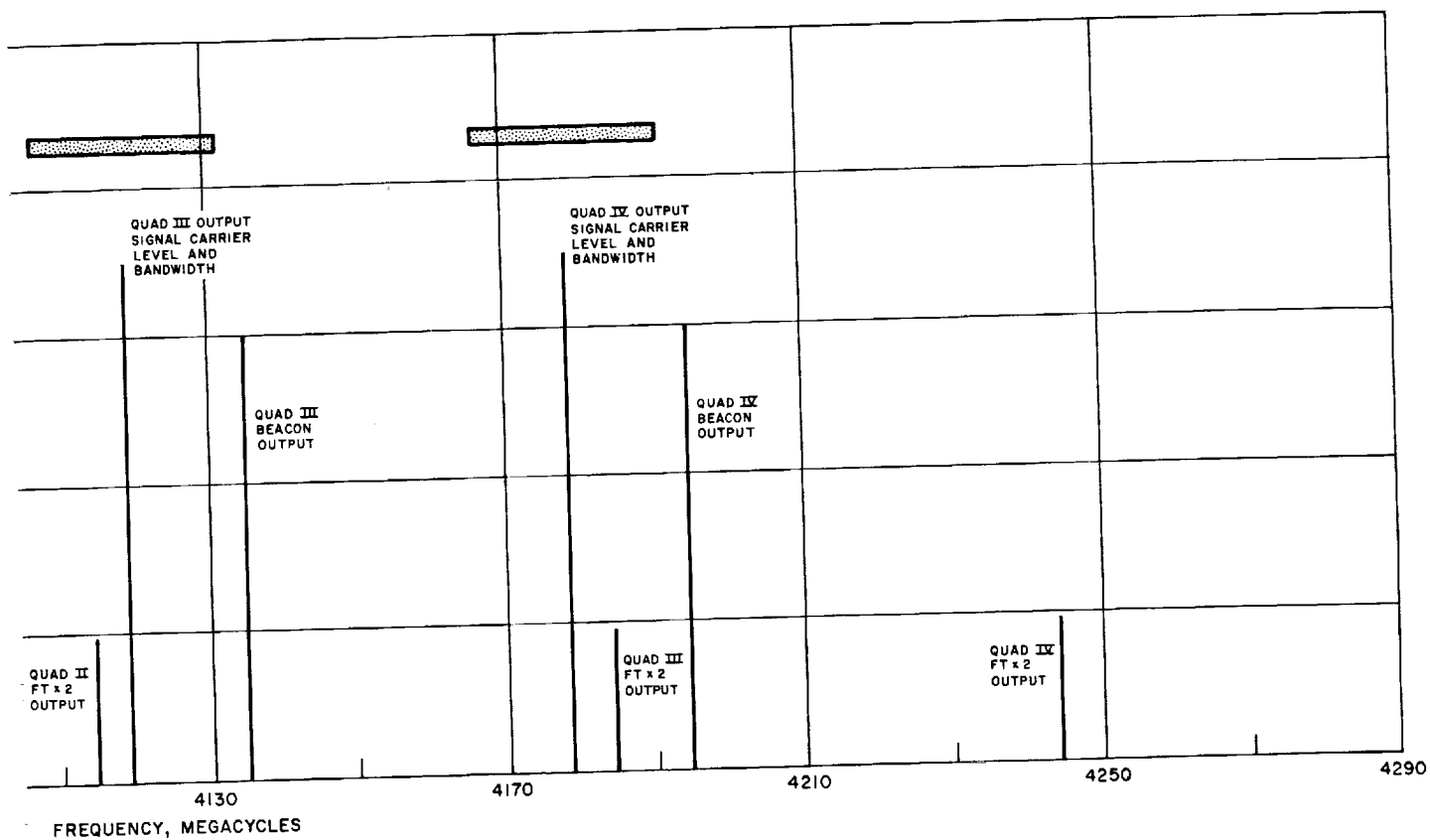
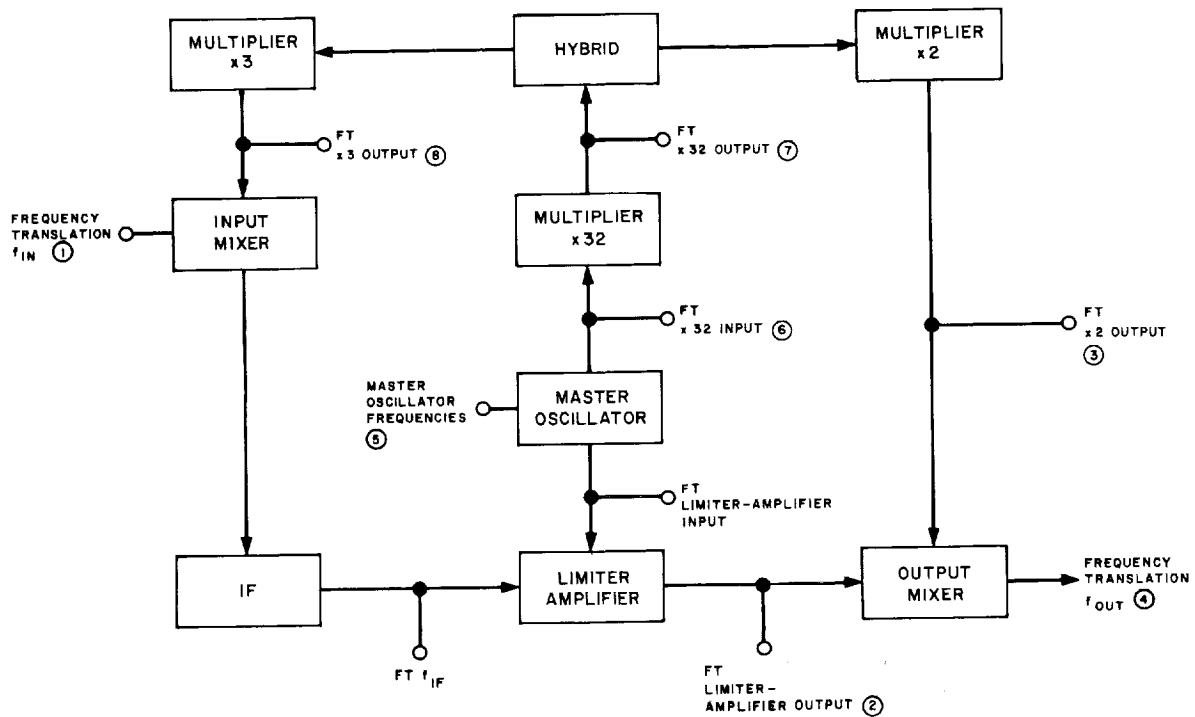


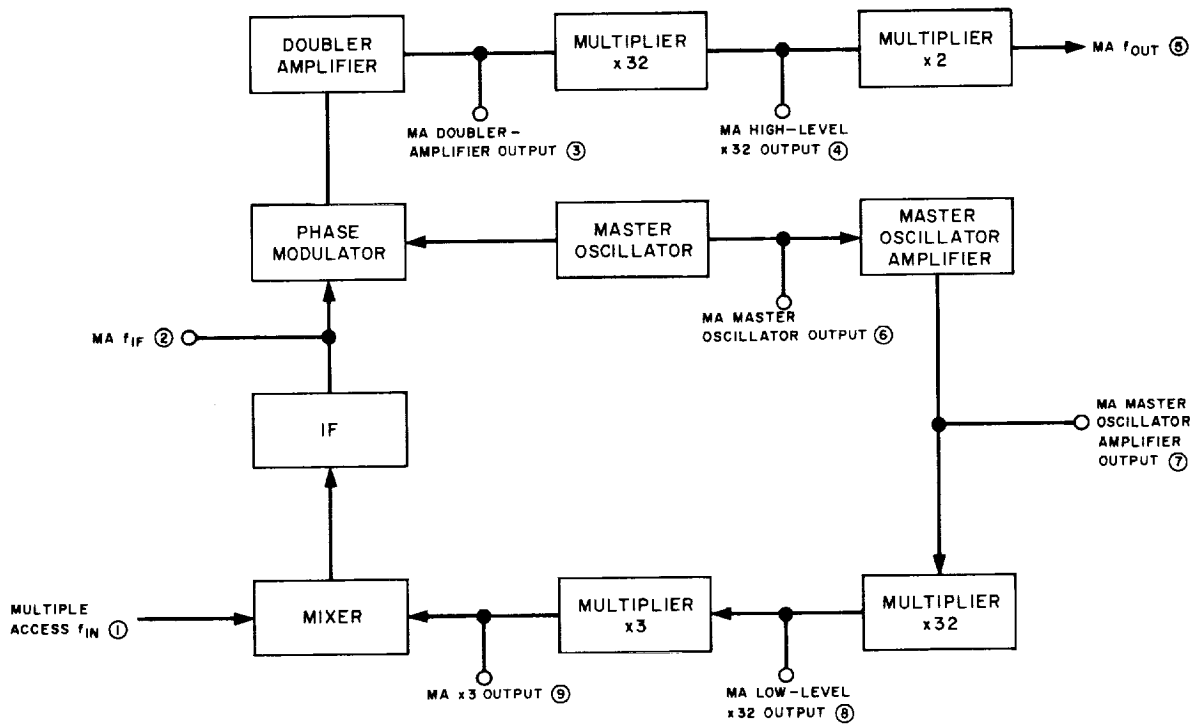
FIGURE 3-6 (CONTINUED). RF SPECTRUM DIAGRAM



3. Communication System Design



a) Frequency Translation Mode



b) Multiple Access Mode

FIGURE 3-7. SIMPLIFIED TRANSPONDER BLOCK DIAGRAM

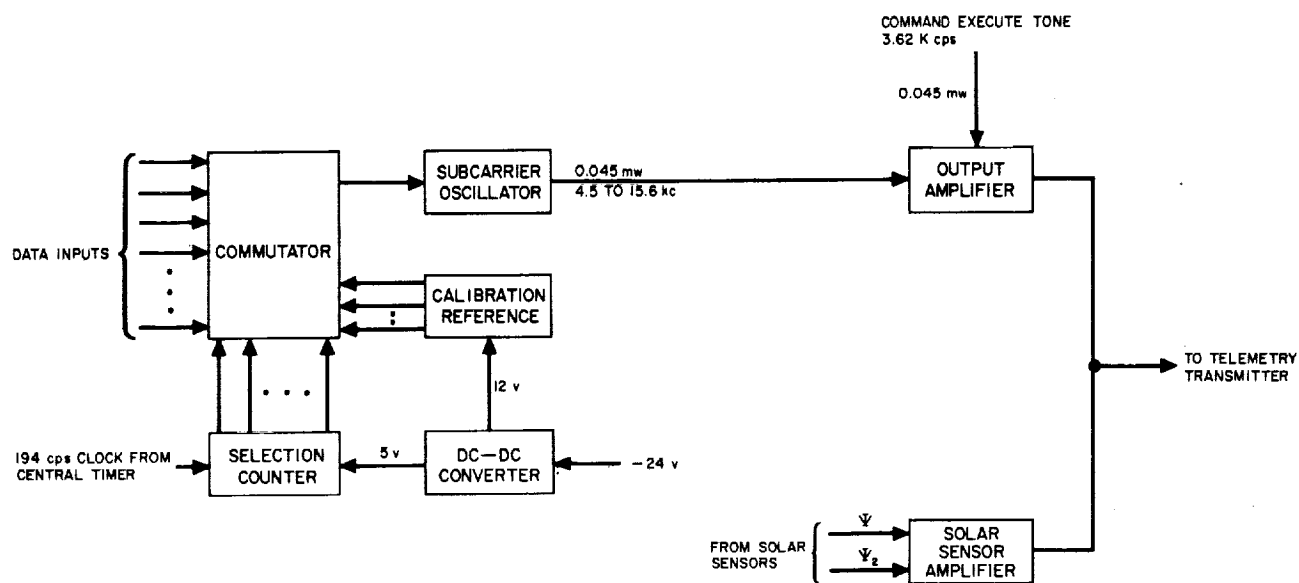


FIGURE 3-8. TELEMETRY ENCODER BLOCK DIAGRAM

3. Communication System Design

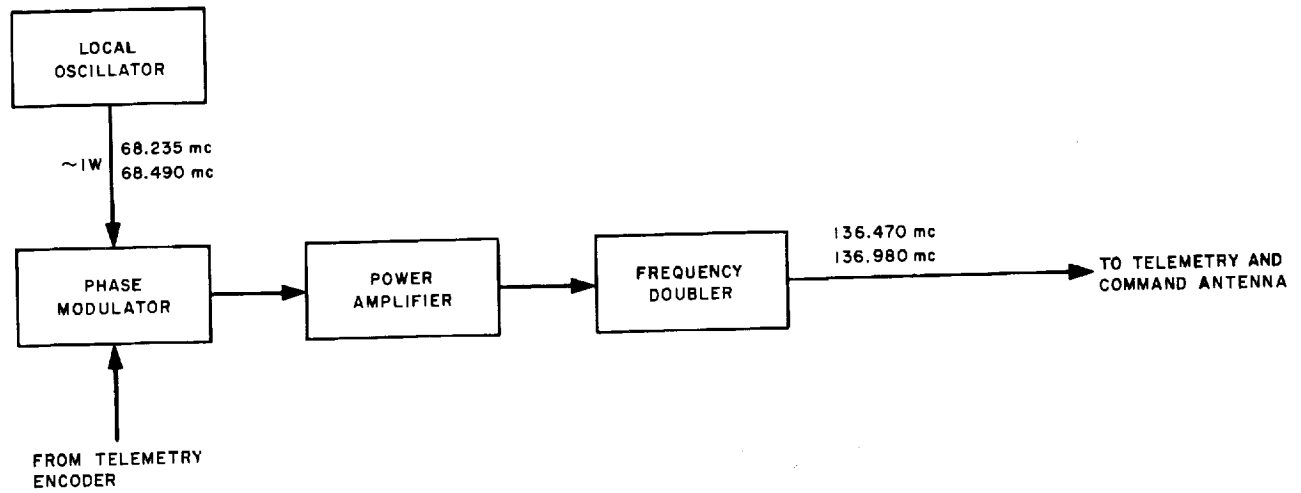


FIGURE 3-9. TELEMETRY TRANSMITTER BLOCK DIAGRAM

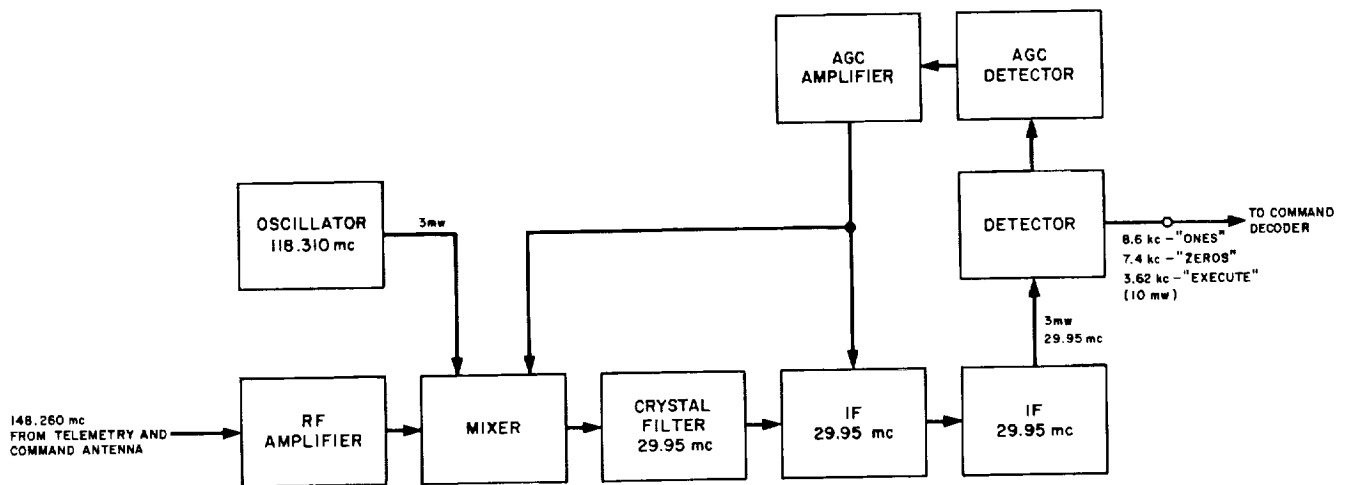


FIGURE 3-10. COMMAND RECEIVER BLOCK DIAGRAM

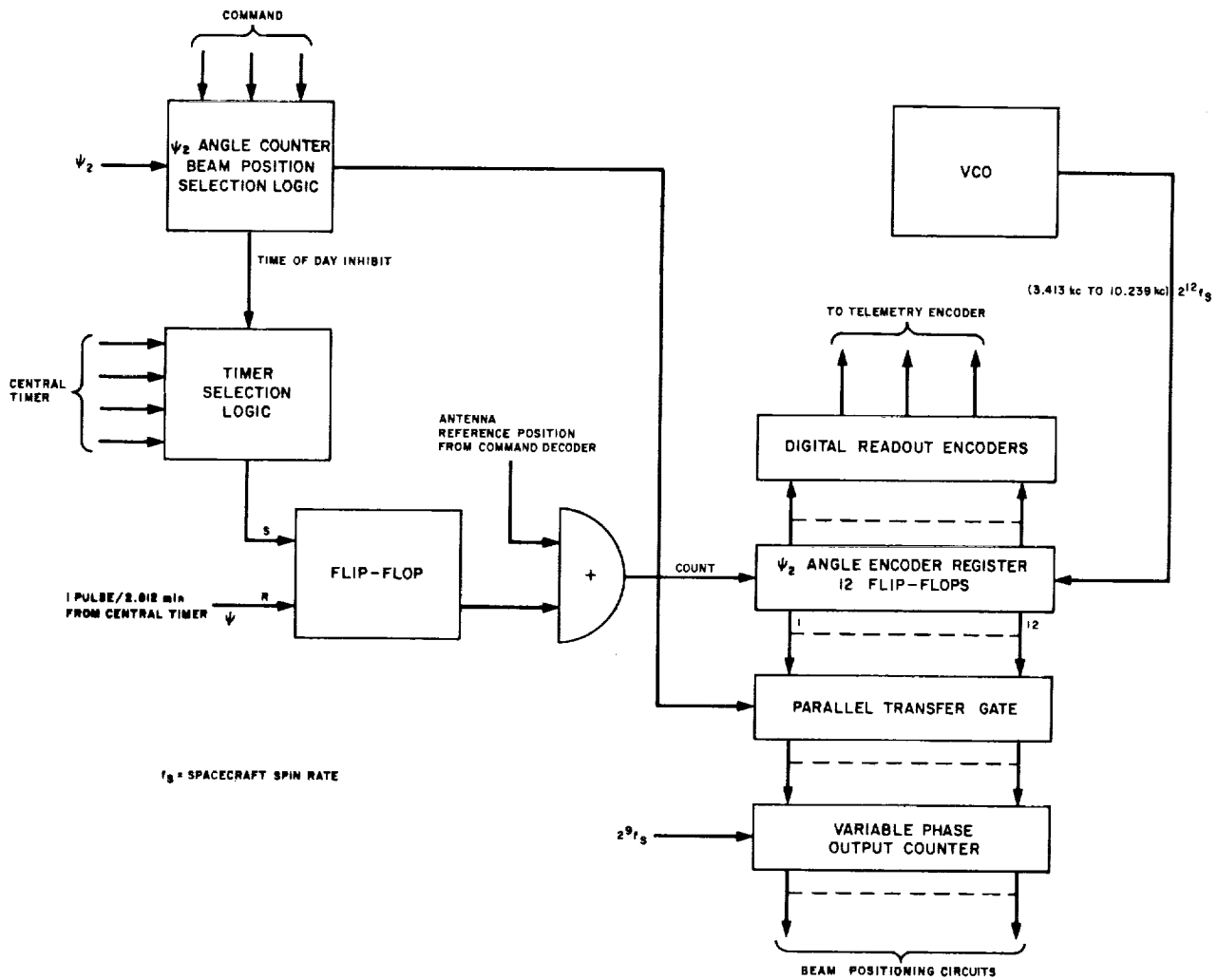


FIGURE 3-11. ψ_2 ANGLE COUNTER BLOCK DIAGRAM

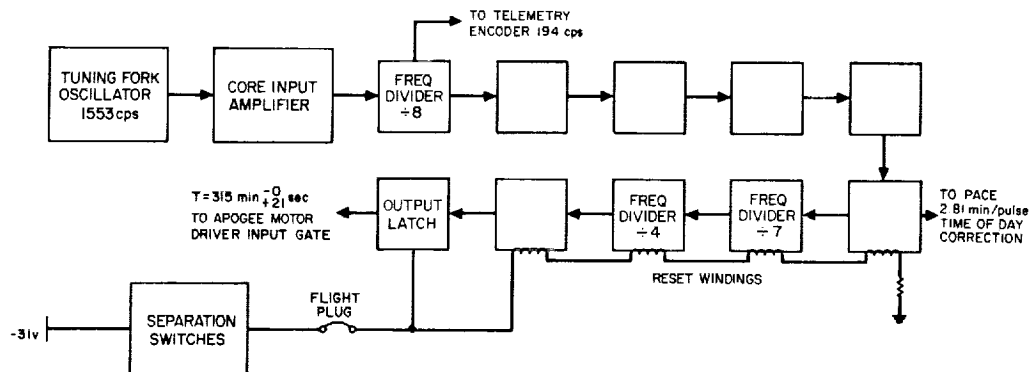


FIGURE 3-12. CENTRAL TIMER BLOCK DIAGRAM

4. LAUNCH AND ORBIT CONSIDERATIONS

The following section summarizes the system-oriented studies associated with the launch and in-orbit procedures and events of the Advanced Syncom mission. These studies were undertaken to facilitate the specification of procedural, performance, and spacecraft parameter requirements as early as possible.

GENERAL SEQUENCE OF EVENTS*

The Syncom spacecraft will be launched from the Atlantic Missile Range (AMR) (28.5 degrees latitude) at 93 degrees azimuth ($\sim$ east), using an Atlas-Agena D with the Atlas and first Agena burn to place the spacecraft in an 85 to 100 n.mi. nominally circular orbit. A second Agena burn at the first ascending node (second equatorial crossing) will place the spin-stabilized spacecraft into an elliptical transfer orbit with an apogee radius equal to the synchronous orbit radius of 22,752.3 n. mi. An apogee boost is provided by a JPL solid propellant motor fixed in the spacecraft. The JPL motor is sized to the maximum payload injection capability of the Agena for this mission and the payload will be ballasted to this requirement. After cutoff of the second Agena burn, the Agena guidance package will program a yaw angle of about 54 degrees to the right of the flight path in the horizontal plane to pre-align the third-stage apogee motor thrust axis (Syncom spin axis) prior to spin-up and separation of the spacecraft from the expended Agena stage. The spacecraft will then be spun at 100

rpm on the spin table mounted on the Agena, and separated.

As it traverses the transfer orbit, the spacecraft will maintain the spin axis attitude established by the Agena guidance system prior to separation. This is the correct attitude for apogee motor thrust to simultaneously accelerate the spacecraft to circular synchronous velocity and remove the 29.1-degree inclination when the spacecraft reaches apogee at the stationary orbital radius. The final longitude attained at apogee motor burnout will be about 93° W (over the Pacific, west of Ecuador). If the apogee motor is ignited at the second apogee crossing, the final longitude will be about 112° E (West Borneo). Other longitudes are also possible, depending on the number of transfer orbit periods allowed before apogee motor ignition or the node at which the Agena begins a second burn to leave the parking orbit. Figure 4-1 illustrates the geometry of the launch for the case in which the transfer ellipse is entered at the second node of the parking orbit, and apogee boost is at the first apogee of the transfer orbit. Figure 4-2 is a ground trace of a similar ascent sequence studied by Lockheed and launched from AMR at 93 degrees azimuth, giving an initial inclination of 29.1 degrees and a longitude at apogee boost of 93° W.

The satellite spin axis attitude and velocity are controlled by a pulsed-jet control system based on the same principles as the control system of the present Delta-launched Syncom satellite. Means for controlling the spacecraft spin speed throughout the lifetime of the vehicle will be incorporated in the form of a spring-restrained centrifugally actuated axial jet.

*A detailed description of the planned launch and in-orbit procedures and parameters is given in Section 5 of "Syncom II Summary Report," March 1963.

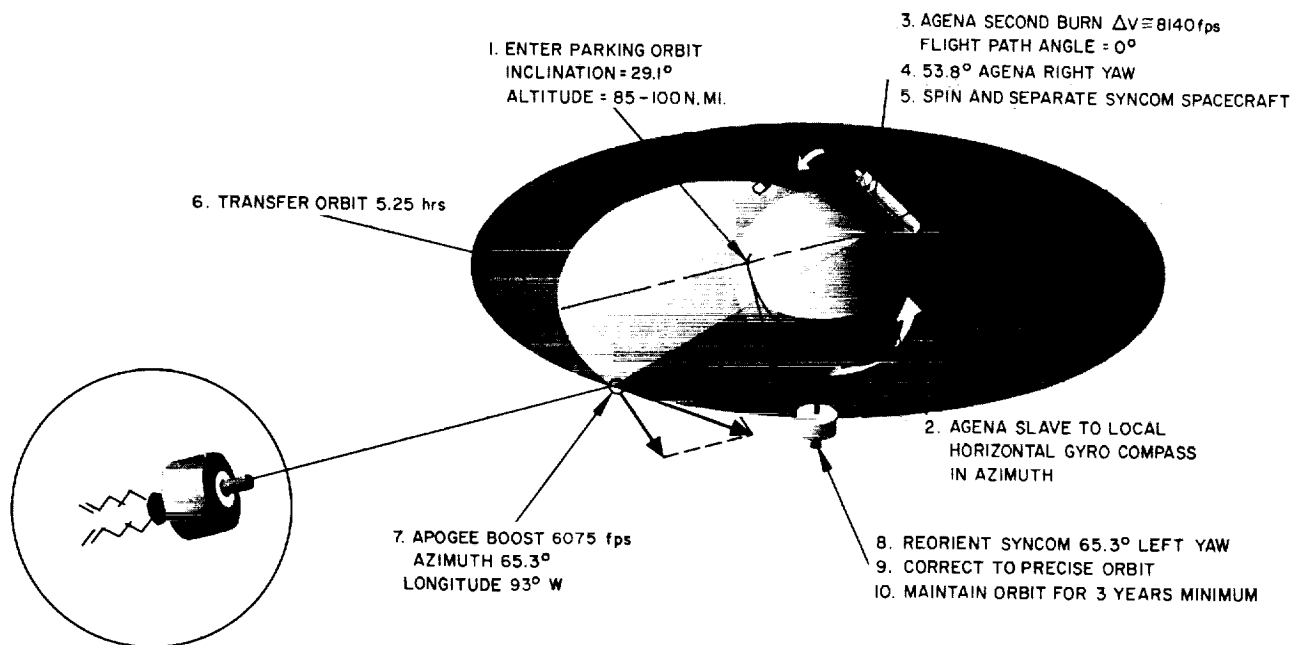


FIGURE 4-1. ADVANCED SYNCOM LAUNCH SEQUENCE

After apogee motor burning, the spacecraft will be precessed through approximately 65 degrees in yaw until the spin axis is parallel to the earth's spin axis. This attitude permits the transmitting antenna beam to point at the earth at all times, gives efficient solar power supply illumination, and provides control of period and elimination of orbital eccentricity by the velocity control jet system. The orientation control jet system, operated continuously rather than in a pulsed mode, permits elimination of inclination in this attitude.

FIXED VERSUS VARIABLE APOGEE IGNITION TIMER

This discussion will assist in ascertaining the errors in the synchronous orbit resulting from planned changes of parking orbit altitude without a corresponding correction in the time from Agena D separation to apogee motor firing. The

noncorrectable timing device employed for apogee motor firing is referred to as a fixed timer.

The use of such a fixed timer in conjunction with any planned parking orbit altitude change will necessarily result in apogee motor firing at a time other than apogee; i.e., a timer error exists. The purpose here, therefore, is to decide if the magnitude of the resulting errors is significant to the extent that a variable timer is required.

To obtain the magnitude of the resulting errors, a Syncom 2 Keplerian apogee firing computer program (FUSIT)* was used with Advanced Syncom apogee firing geometry (Figure 4-3) and run for a timer error of $T = \pm 0, 1, 2, \dots, 10$ minutes. Other parameters used for

*"Syncom I 7090 Computer System Description - FUSIT," Hughes Aircraft Company, 5 September 1962.

4. Launch and Orbit Considerations

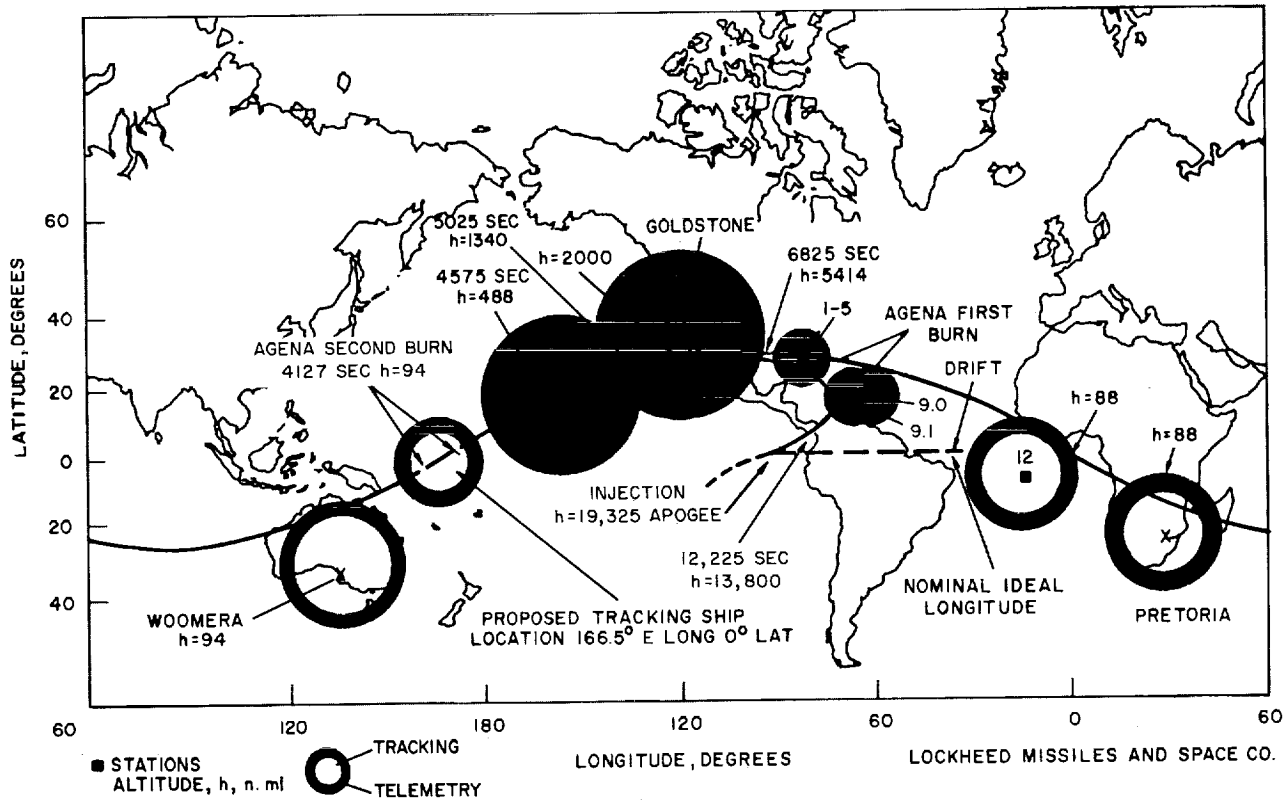


FIGURE 4-2. GROUND TRACE ASCENT SEQUENCE

the calculations are shown in Table 4-1. Residual errors are present at $T = 0$ due to inaccuracies in nominal parameters.

The results of immediate interest are shown in Table 4-2. An examination of the plot of $|V_a|$ (velocity correction necessary to remove drift)

TABLE 4-1. ORBITAL PARAMETERS

T at apogee = 0	
Transfer ellipse	
Apogee radius	22,752.3 n.mi.
Apogee longitude	93° W (descending node)
Apogee velocity, V_a	5250 fps (0.863 n.mi./sec)
Perigee radius	~ 3529.7 n.mi. (88 n.mi. altitude)
Perigee velocity	~ 33,760 fps
Semimajor axis	13.141 n.mi.
Nominal period, τ	37,800 seconds
Inclination	29.1 degrees
Longitude of ascending node	~ 166.5° E
Period	~ 37,800 seconds (~10.51 hours)
Apogee boost, ΔV_a (descending node)	
ΔV_a	6075 fps
Burning time	~ 45 seconds
V_a , synchronous velocity	~ 10,087.5 fps
Elevation	~ 0 degrees
Azimuth	65.3 degrees from north (clockwise)
Timer setting $\frac{\tau}{2} \pm N$ minute, $N = 0, 1, 2, \dots, 10$	

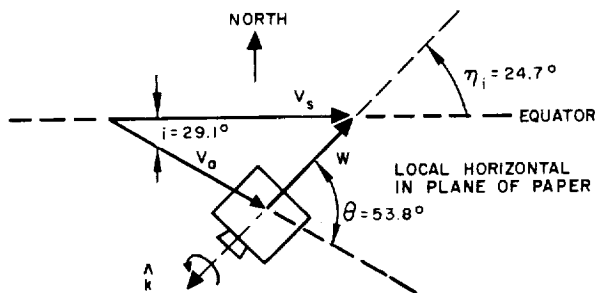


FIGURE 4-3. APOGEE MOTOR BOOST GEOMETRY

TABLE 4-2. EFFECTS OF TIMER SETTING ON PERIGEE CHANGE

ΔT	Change in one-half period (timer error)
$T = 315$ minutes	One-half nominal period
Δr_p	Change in parking orbit altitude
$a = 13,141$ n.mi.	Nominal semimajor axis
Δr_s	Change in synchronous radius
Δa	Change in semimajor axis
$\frac{\Delta T}{T} = \frac{3}{2} \frac{\Delta a}{a}$	
$\Delta a = \Delta r_p + \Delta r_s$	
$\Delta r_s \approx 0$	
$\therefore \Delta r_p = \Delta a$	
$\Delta T/T = (3/2) (\Delta r_p/a)$	
$\Delta r_p = (\Delta T/T) \left(\frac{2}{3} \right) (a)$	
$\Delta r_p = \left(\frac{2}{3} \right) \left(\frac{1}{315} \right) (13,141) \Delta T$	
$\Delta r_p = 27.8$ n.mi./min	

versus T (time from apogee) (Figure 4-4) shows that $|V_d| \approx kT^2$ where $k \approx 0.02$ fps/min².

The 3σ ascent guidance errors, including an inherent 2-minute timer error, require a drift velocity correction, $|V_d|$, of 120 fps.* The data in Table 4-2 shows that, for a ± 10 -minute timer error, an additional $|V_d|$ of 1.79 fps is required. This is an increase of approximately 1.49 percent. Therefore, the additional timer error causes an insignificant increase in the required drift velocity correction.

An examination of the plot of the velocity correction (Figure 4-5) necessary to remove inclination, V_i , shows the $\Delta V_i/\Delta T \approx 11.1$ fps/min.

*"Syncom II Summary Report," Hughes SSD 3128R, NASA Contract 5-2797, 31 March 1963.

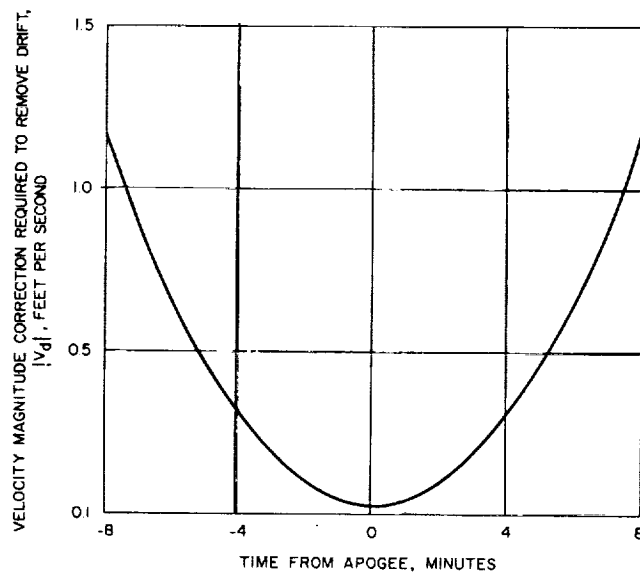


FIGURE 4-4. VELOCITY MAGNITUDE REQUIRED TO REMOVE DRIFT, $|V_d|$, VERSUS TIME FROM APOGEE, T

A ± 10 -minute timer error requires an additional $|V_i|$ of 111 fps. This correction is of significant value.

The significance of the additional velocity correction required must be assessed with respect to the change in planned parking orbit altitude necessary to cause such a timer error. The correlation of planned parking orbit altitude change and resulting timer error is shown in Table 4-2.

Table 4-3 shows a 10-minute timer error will result from a planned change in parking orbit altitude of 278.0 n.mi. This is clearly an unrealistic change of plan. A more realistic change in altitude would be at most 30 n.mi. For ease of calculation, assume the change to be 27.8 n.mi. This corresponds to a 1-minute timer error. The resulting $|V_d|$ is 0.046 fps and $|V_i|$ is 11.7 fps. $|V_d|$ is clearly insignificant. $|V_i|$ is a 10.4 percent increase in the $|V_i|$ of the 3σ ascent guidance errors, which is not significant.

This study indicates that errors resulting from changes in planned parking orbit altitude with

4. Launch and Orbit Considerations

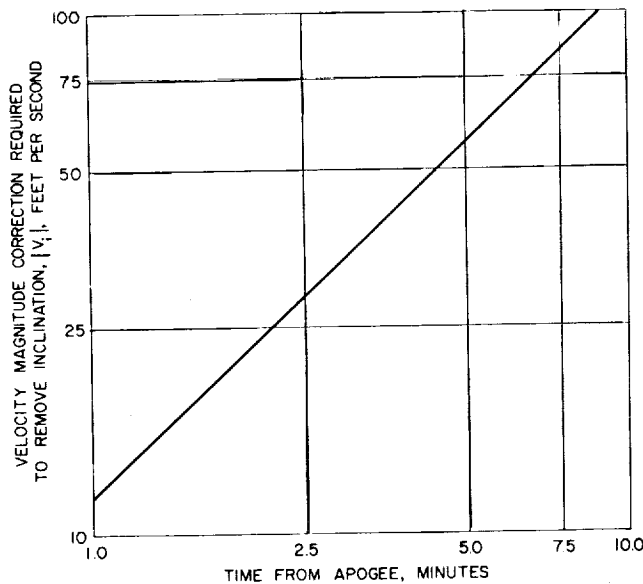


FIGURE 4-5. VELOCITY MAGNITUDE REQUIRED TO REMOVE INCLINATION, $|V_i|$, VERSUS TIME ERROR

a fixed timer are not significant to the extent that a variable timer is required. If it should be desirable to reduce the slight change of $|V_i|$ resulting from the use of a fixed timer, a change can be made in the Agena D yaw programmer, a readily variable unit.

TRANSLATION DUE TO INITIAL ORIENTATION IMPULSE

During an orientation maneuver, the axial jet produces simultaneously both a torque and a translatory impulse. Since any translation of the spacecraft represents a potential need for stationkeeping control action, it is desirable to estimate the magnitude of the translation impulse.

The greatest number of axial pulses fired in one sequence is expected during initial orientation (~1250 pulses, calculated by means of the orientation dynamics IBM computer model*).

*"Advanced Syncom Monthly Progress Report," Hughes Aircraft Company, May 1963.

TABLE 4-3. TIMER ERROR EFFECTS

T from Apogee, minutes	$ \Delta r_p $, n.mi.	$ V_i $, fps	$ V_d $, fps	Eccentricity
10	278.0	111	1.79	0.023
9	250.2	100	1.46	0.021
8	222.4	89.0	1.16	0.019
7	194.6	77.9	0.842	0.016
6	166.8	66.8	0.66	0.014
5	139.0	55.7	0.465	0.011
4	111.2	44.7	0.306	0.0095
3	83.4	33.6	0.186	0.0071
2	55.6	22.5	0.093	0.0047
1	27.8	11.7	0.046	0.0023
0	0	4.0	0.0279	0.00001
-1	27.8	11.7	0.046	0.0023
-2	55.6	22.5	0.093	0.0047
-3	83.4	33.6	0.186	0.0071
-4	111.2	44.7	0.306	0.0095
-5	139.0	55.7	0.465	0.011
-6	166.8	66.8	0.66	0.014
-7	194.6	77.9	0.842	0.016
-8	222.4	89.0	1.16	0.019
-9	250.2	100	1.46	0.021
-10	278.0	111	1.79	0.023

Thus, the calculation below assumes this number of impulses.

Starting with the basic assumptions that the spin axis traces a great circle instead of a rhumb line, and the large number of discrete jet pulses closely approaches a continuum, the North-component of the translation impulse (Figure 4-6) is

$$\int_{\theta_0}^{\pi/2} \Delta V_0 \sin \theta d\theta$$

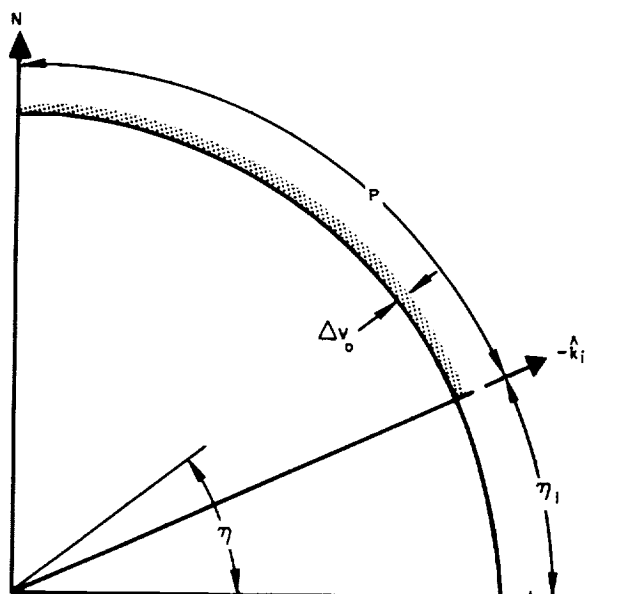


FIGURE 4-6. TRANSLATION IMPULSE DUE TO INITIAL ORIENTATION IMPULSE

where θ_o (the inclination of the spin axis prior to orientation) = 24.4* degrees and $\Delta V_o = 18/1.145 = 15.72$ fps/rad is $15.72 \times 0.911 = 14.3$ fps. ($\Delta V = 18$ fps is the equivalent ΔV required of the control system for the orientation maneuver, and 90 degrees - 24.4 degrees = 65.6 degrees = 1.145 radians is the precession angle.) It should be noted that this impulse is always in a northerly direction.

14.3 fps is ~12 percent of the estimated ΔV (112 fps, 3σ) required to correct for ascent guidance inclination errors, and hence will be taken into account in estimating total ΔV requirements for the mission. Similarly,

$$\begin{aligned} \Delta V_{\text{east}} &= \int_{\theta_o}^{\pi/2} \Delta V_o \cos \theta \, d\theta = \Delta V_o (1 - \sin 24.4 \text{ degrees}) \\ &= (15.72) (0.587) \\ &= 9.23 \text{ fps} \end{aligned}$$

* (24.7 degrees - 0.3 degree) = initial geometric attitude minus inefficiency due to rhumb line precessional motion.

ORBITAL PERTURBATIONS DUE TO SUN AND MOON

This discussion describes a proposed method of computing perturbations of a satellite orbit, especially those due to the sun and moon. It is intended to implement this method for the IBM 7094 computer in the near future.

Nature of Method. The usual methods of computing perturbations fall into two classes, general and special perturbations. Use of general perturbations has the advantage that analytic expressions are obtained, leading to the ability to evaluate results for any value of the time without computing for intervening times. More important, an insight is obtained into the general nature of the motion. However, a rather overwhelming amount of labor is involved in a development in general perturbations. Use of special perturbations or numerical integration, on the other hand, requires no heavy analysis but also gives almost no physical insight into the results. Furthermore, evaluation of position and velocity at arbitrary times is computationally inefficient.

The method planned is an intermediate approach which largely preserves the flexibility and insight offered by general perturbations, while avoiding the time-consuming (and, for all but the experienced celestial mechanic, error-promoting), series developments by taking advantage of the speed of modern computers. The method rests on knowledge of the functional form, that of general trigonometric series in the time plus constant and secular terms, and of the solutions obtained using general perturbations. The coefficients are determined by numerically fitting trigonometric series, whose frequencies are determined analytically, to expressions giving the rates of change of the orbital elements.

The resulting series may be integrated analytically. The method may be carried out for any reference orbit (e.g., the orbit given by Brouwer's theory accounting for oblateness), and may be iterated to obtain perturbations of any order, with the restriction, however, that the number of frequencies included is restricted by the capacity and speed of the computer in simultaneous solution of the linear equations for determination of the coefficients. The method may be termed "semi-analytic" since the result is a closed-form expression with coefficients determined by a numerical method which in itself does not recognize the analytic form of the perturbing potential or the corresponding accelerations.

Analysis. Let E_i be a set of quantities defining an osculating orbit. The E_i are constants for Keplerian motion; the differential equations which they satisfy for perturbed motion may be written in the form of Gauss's variation equations (e.g., Moulton, *Celestial Mechanics*, pp 404-405):

$$\frac{dE_i}{dt} = \sum_j M_{ij} a_j \quad (4-1)$$

where the a_j represent components of the perturbing acceleration and the M_{ij} are functions of the E_i and the time. There may already be a solution in which part of the perturbing acceleration is accounted for; say E_{oi} taking into account a_{oj} ; in this case let

$$E_i = E_{oi} + E_{ii}, a_j = a_{oj} + a_{ij} \quad (4-2)$$

Then

$$\frac{dE_{ii}}{dt} = \sum_j M_{ij} a_{ij} \quad (4-3)$$

Equation 4-1 will provide the exact solution only if the instantaneous values of the E_i are used in evaluation of both the matrix M_{ij} and the accel-

erations a_{ij} . If constant values of the E_i are used in the right-hand side of Equation 4-1, the resulting solution includes only first-order perturbations. If Equations 4-2 and 4-3 are used, with the values of E_{oi} substituted in the right-hand member of Equation 4-3, the results will contain some but not all of the second-order terms proportional to products of a_{oj} and a_{ij} . Physically, the cumulative effect of the a_{oj} , will be taken into account in computing the future effect of the a_{ij} , but not *vice versa*. Let it be assumed for the time being that this order of accuracy is all that is required.

In developments in general perturbations, it is usual to develop Equation 4-1 or Equation 4-3 in series. When the perturbations are due either to bodies other than the primary in the system or to harmonics in the gravitational field of the primary, the result is of the form

$$\frac{dE_{ii}}{dt} = C_{ii} + \sum_{k=1}^{\infty} \left(C_{i(2k)} \cos W_k t + C_{i(2k+1)} \sin W_k t \right) \quad (4-4)$$

The coefficients C_{il} are functions of the elements only and not of the time. The W_i are linear combinations of the form

$$W_k = \sum_i M_{ki} n_i + \sum_{j,m} L_{kjm} N_{jm}$$

where the M s and L s are integers and, in the case of first-order perturbations of an orbit by other bodies in unperturbed Keplerian orbits, the only n_i required is n and the only N_{jm} are N_m where n is the mean motion of the body considered and the N_m are the mean motions of the perturbing bodies. In the case where Equations 4-2 and 4-3 are used, the n_i include other frequencies; for example, if the a_{oi} include oblateness, the n_i are linear combinations of the

mean-mean motion and the mean motions of the apogee and node. If a perturbing body is itself perturbed, the corresponding N_m include similar frequencies.

Rather than develop the expansion (Equation 4-4) analytically, the proposed method is to solve for a limited set of the $C_{1\ell}$ numerically. This may be done by solution of the set of linear equations resulting from numerical evaluation of Equation 4-1 or 4-3 at a properly chosen set of values of the time. If more times are used than the number of values of ℓ , the $C_{1\ell}$ may be determined by least squares.

Determination of the $C_{1\ell}$ requires the solution of six sets of $(2n_r + 1)$ equations (or the inversion of a square matrix with $(2n_r + 1)^2$ elements when n_r is the number of frequencies. It is evident that some intelligence and judgment (and possibly numerical experimentation) will be required in the choice of the W_k as well as of the times at which the rates of change of the elements are evaluated; aliasing of frequencies must be considered.

Having the coefficients, it is easy to see that

$$E_{11} = E_{11}^{(0)} + C_{11}t + \sum_{k=1}^{\infty} \left(\frac{C_{1(2k)}}{W_k} \sin W_k t - \frac{C_{1(2k+1)}}{W_k} \cos W_k t \right) \quad (4-5)$$

One further remark is needed. In putting down the functional form, Equation 4-4, it was assumed that the set of elements used did not introduce a factor proportional to the time. It is usual in celestial mechanics to avoid this by the use of an element such as σ , such that the mean anomaly M is given by

$$M = \sigma + \int_0^t n dt = \sigma + \int_0^t \frac{k}{a^{3/2}} dt$$

Considering only first-order perturbations with $a = a_{00} + a_1$, this may be reduced to

$$M = \sigma + \frac{k}{a_{00}^{3/2}} t - \frac{3}{2} k a_{00}^{-5/2} \int_0^t a_1 dt$$

and the coefficients in the series for the required integral readily derived from that for a_1 . However, some caution regarding initial conditions is necessary.

For generality, suppose that Equations 4-2 and 4-3 are used. Let $[\sigma_1]$ and $[a_1]$ be any indefinite integrals of

$$\frac{d\sigma_1}{dt} \quad \text{and} \quad \frac{da_1}{dt}$$

Then if $\sigma_0(0)$ and $a_0(0)$ are the osculating elements at $t=0$,

$$\sigma_1(t) = [\sigma_1(t)] - [\sigma_1(0)]$$

$$a_1(t) = [a_1(t)] - [a_1(0)]$$

and

$$M = \sigma_0(t) + \sigma_1(t) + \int_0^t \frac{k dt}{[a_0(t) + a_1(t)]^{3/2}}$$

or to the first order

$$M = \sigma_0(t) + \sigma_1(t) + \int_0^t \frac{k dt}{[a_0(t)]^{3/2}} - \frac{3}{2} \frac{k}{a_{00}^{5/2}} \int_0^t ([a_1(t)] - [a_1(0)]) dt$$

let

$$M_0 = \sigma_0(t) + \int_0^t \frac{k dt}{[a_0(t)]^{3/2}}$$

and let $[J]$ be any indefinite integral of $[a_1(t)]$.

4. Launch and Orbit Considerations

Then

$$M = M_1 + \sigma_1(t) + \frac{3}{2} \frac{k}{a_{00}^{5/2}} [a_1(o)] t - \frac{3}{2} \frac{k}{a_{00}^{5/2}} \left([J(t)] - [J(o)] \right) \quad (4.6)$$

High Order Perturbations. The solution given by Equation 4-5 (plus the E_{oi} , if Equation 4-3 is used) could be substituted back into Equation 4-1 and an iteration performed to obtain a new set of equations of the form Equation 4-4. Alternatively, the initial solution could be used in place of E_{oi} in Equations 4-2 and 4-3; this would, however, be somewhat more complex as the a_{11} would now involve the differences of perturbing accelerations.

The initial numerical application of the method will be based on first-order results only, utilizing Brouwer's formulation to obtain the E_{oi} .

A Property of the Method for a Circular Reference Orbit. Let a be the position angle relative to a fixed reference in unperturbed motion in a circular reference orbit. Let $\delta\lambda$ be the deviation in position angle, δr the deviation in radius, and z the displacement normal to the reference orbit. Let a_o be the unperturbed semimajor axis, n_o the mean motion. If the motion is linearized about the reference orbit (e.g., Hughes Aircraft Company TM 649, *Dynamic Analysis and Design of the Synchronous Communication Satellite*, pp 5 ff), it is possible to show that for unperturbed motion

$$\left. \begin{aligned} \delta r &= \delta a + r_X \cos a + r_Y \sin a \\ X &= a_o \delta\lambda = X_o - \frac{3}{2} n_o \int \delta a \, dt \\ &\quad - 2r_X \sin a + 2r_Y \cos a \\ z &= Z_X \cos a + Z_Y \sin a \end{aligned} \right\} \quad (4.7)$$

where

$$(\delta a, r_X, r_Y, Z_X, Z_Y, X_o)$$

are constants. It is also possible to show that in perturbed motion, these elements satisfy the equations

$$\left. \begin{aligned} -n_o \frac{dZ_X}{dt} &= a_z \sin a \\ n_o \frac{dZ_Y}{dt} &= a_z \cos a \\ \frac{d\delta a}{dt} &= \frac{2}{n_o} a_\lambda \\ \frac{dX_o}{dt} &= -\frac{2}{n_o} a_r \\ \frac{dr_X}{dt} &= -\frac{2}{n_o} a_\lambda \cos a - \frac{1}{n_o} a_r \sin a \\ \frac{dr_Y}{dt} &= -\frac{2}{n_o} a_\lambda \sin a + \frac{1}{n_o} a_r \cos a \end{aligned} \right\} \quad (4.8)$$

where (a_r, a_λ, a_z) are radial, tangential, and normal perturbing accelerations respectively.

If the set in Equation 4-8 is developed in series, it is evident that the arguments of the trigonometric functions have the form

$$Ma + \sum_{j,m} L_{kjm} N_{jm}$$

and that the same is true of the indefinite integrals for the elements. The conclusion is that once a set of coefficients C_{ij} is obtained a reference orbit for which

$$a = a_o + n_o t$$

the result can be extended to any choice of a_o . As an example, lunisolar perturbations of a synchronous equatorial satellite may be found from a single set of C_{ij} independent of the longitude.

JUSTIFICATION FOR SPIN SPEED CONTROL

The following study was undertaken to examine the effects of jet damping, skew torque (due to asymmetric transverse moments of inertia), and jet thrust misalignments on the spin speed history of the spacecraft throughout its nominal operating time. Results summarized below show that a spin speed change of 50 to 90 percent is not unlikely (primarily due to jet misalignment thrust components).

Summary. The effects of the jet misalignment in spin speed negate the simplifying assumptions made below for jet damping and skew-torque calculations, but the general conclusions on the necessity for spin speed control are still valid from jet misalignment effects alone.

Jet damping (approximately $\sim \omega_z$) $\Delta\omega_z$ — 7.1 rpm

Skew torque — $\left(\text{approximately } \sim \frac{1}{\omega_z^2} \right)$ ± 10.6 rpm

Jet misalignments (assume 0.5° , $0.5^\circ, 0.25''$)

$$\left. \begin{array}{l} \Delta\omega_{ZA} = 63.2 \\ \Delta\omega_{ZR} = 9.5 \\ \Delta\omega_{ZC} = 10.5 \end{array} \right\} \sum_{\text{rss}} = \pm 64.8 \text{ rpm}$$

It should be noted that spin speedup minimizes the skew-torque effect while increasing the jet damping. Certainly speedup would cause the spacecraft to exceed its structural design limitations. Slowdown could bring spacecraft spin speed to less than 10 rpm.

General Equations Including Jet Damping. By the principle of angular momentum

$$\bar{M}_Q = \dot{\bar{H}}_Q \quad (4-9)$$

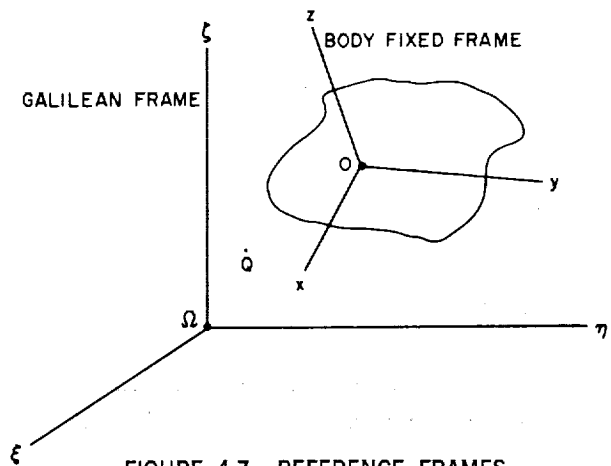


FIGURE 4-7. REFERENCE FRAMES

where $\bar{M}_Q$ is the applied vector moment and $\dot{H}_Q$ is the time derivative of the total angular momentum. The subscript Q indicates these quantities are measured with respect to a point Q fixed in the Galilean frame (A). Referring the angular momentum to frame (B) (see Figure 4-7).

$$\dot{\bar{H}}_Q = \dot{\bar{H}}_B + [\omega_B] \times \bar{H}_B \quad (4-10)$$

If (B) is principal and dropping the subscripts (B) the well known forms appear*:

$$\bar{H} = A\omega_x \bar{i} + B\omega_y \bar{j} + C\omega_z \bar{k} \quad (4-11)$$

$$\begin{aligned} \bar{\omega} &= \omega_x \bar{i} + \omega_y \bar{j} + \omega_z \bar{k} \\ \dot{\bar{H}} &= (\dot{A}\omega_x + A\dot{\omega}_x) \bar{i} + (\dot{B}\omega_y + B\dot{\omega}_y) \bar{j} \\ &\quad + (\dot{C}\omega_z + C\dot{\omega}_z) \bar{k} \end{aligned} \quad (4-12)$$

where

ω_x = pitch rate, ω_y = yaw rate, and

ω_z = roll rate,

and

$$\begin{aligned} \bar{\omega}_x \bar{H} &= (C - B) \omega_y \omega_z \bar{i} + (A - C) \omega_z \omega_x \bar{j} \\ &\quad + (B - A) \omega_x \omega_y \bar{k} \end{aligned}$$

*A, B, and C are pitch, yaw, and roll moments of inertia about body axes x, y, z.

4. Launch and Orbit Considerations

NOTE: By implication, B is assumed to remain principal for the interval under consideration. Combining, the Euler equations are obtained for a system of particles of varying inertia.

$$\left. \begin{aligned} M_x &= A\dot{\omega}_x - (B - C)\omega_y\omega_z + \dot{A}\omega_x \\ M_y &= B\dot{\omega}_y - (C - A)\omega_z\omega_x + \dot{B}\omega_y \\ M_z &= C\dot{\omega}_z - (A - B)\omega_x\omega_y + \dot{C}\omega_z \end{aligned} \right\} \quad (4-13)$$

NOTE: No particles are permitted to enter or leave the system.

Equation 4-15 is next modified to account for expulsion of mass. Let x_i, y_i, z_i be the coordinates of the i^{th} point on body B where mass particles will leave the system.* Practically speaking, they would be the coordinates of the i^{th} jet. Nothing is said of the orientation of this jet. The loss of angular momentum from the system of particles due to the expulsion of Δm_i is

$$\Delta H_i = \bar{R}_i \times \Delta m_i \bar{V}_i \quad (4-14)$$

where

$$\begin{aligned} \bar{V}_i &= \bar{\omega} \times \bar{R}_i \\ \Delta \bar{H}_i &= \bar{R}_i \times \Delta m_i (\bar{\omega} \times \bar{R}_i) \end{aligned} \quad (4-15)$$

$$\dot{\bar{H}}_i = \frac{d\bar{H}_i}{dt} = \dot{\bar{H}}_i = \bar{R}_i \times \frac{dm_i}{dt} (\bar{\omega} \times \bar{R}_i) \quad (4-16)$$

and since

$$\bar{R}_i = x_i \bar{i} + y_i \bar{j} + z_i \bar{k} \quad (4-17)$$

$$(\dot{\bar{H}}_i)_x = \left(\frac{dm_i}{dt} \right) [(y_i^2 + z_i^2) \omega_x - x_i y_i \omega_y - x_i z_i \omega_z]$$

$$(\dot{\bar{H}}_i)_y = \left(\frac{dm_i}{dt} \right) [(z_i^2 + x_i^2) \omega_y - y_i z_i \omega_z - y_i x_i \omega_x]$$

$$(\dot{\bar{H}}_i)_z = \left(\frac{dm_i}{dt} \right) [(x_i^2 + y_i^2) \omega_z - z_i x_i \omega_x - z_i y_i \omega_y] \quad (4-18)$$

Note the symmetry of these equations which indicates that only the exit position of the leaving mass particles, without regard to exit direction, determines the loss in angular momentum for the system of remaining particles (m_i is positive for mass expulsion). Combining Equations 4-18 and 4-13 gives for N jets

$$M_x = A\dot{\omega}_x - (B - C)\omega_y\omega_z + \left\{ \dot{A}\omega_x + \sum_1^N \dot{m}_i [(y_i^2 + z_i^2) \omega_x - x_i y_i \omega_y - x_i z_i \omega_z] \right\} \quad (4-19a)$$

$$M_y = B\dot{\omega}_y - (C - A)\omega_z\omega_x + \left\{ \dot{B}\omega_y + \sum_1^N \dot{m}_i [(z_i^2 + x_i^2) \omega_y - y_i z_i \omega_z - y_i x_i \omega_x] \right\} \quad (4-19b)$$

$$M_z = C\dot{\omega}_z - (A - B)\omega_x\omega_y + \left\{ \dot{C}\omega_z + \sum_1^N \dot{m}_i [(x_i^2 + y_i^2) \omega_z - z_i x_i \omega_x - z_i y_i \omega_y] \right\} \quad (4-19c)$$

In Figure 4-8, showing the body Euler angle and test geometry, the terms inside the braces are the "jet damping" terms.

$$\begin{cases} x_1 = 0, y_1 = 2.16, z_1 = 2.0 \\ x_2 = 2.16, y_2 = 0, z_2 = 0 \end{cases}$$

$$\left\{ \begin{aligned} \omega_x &= \text{pitch rate} \\ \omega_y &= \text{yaw rate} \\ \omega_z &= \text{roll rate} \\ \theta &= \text{nututation angle} \\ \phi &= \text{precession angle} \\ \psi &= \text{spin angle} \\ 0 &\dots \text{center of gravity} \\ \textcircled{1} &\text{inch} \dots \text{axial jet} \\ \textcircled{2} &\text{inches} \dots \text{radial jet} \end{aligned} \right.$$

*Rosser, "Mathematical Theory of Rocket Flight," p. 11 (Principle V).

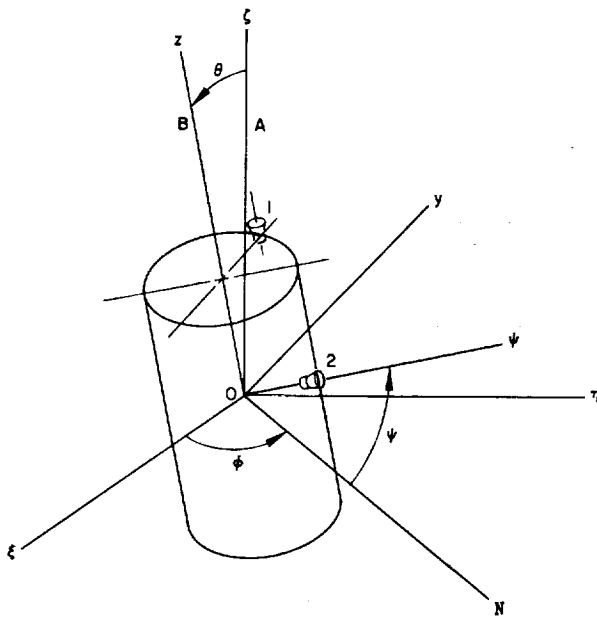


FIGURE 4-8. BODY EULER ANGLE AND TEST GEOMETRY

Inertial Properties. An estimation of inertial properties is given in Table 4-4.

The vernier velocity corrections are:

Continuous — axial jet	1013 fps	
Pulsed — axial jet	22	
Subtotal	1035 fps	87 percent
Pulsed — radial jet	155	13 percent
Total ΔV	1190 fps	100 percent

Nominal spacecraft mass in final orbit configuration is

$$\log \frac{m_1}{m_2} = \log \frac{W_1}{W_2} = \frac{\Delta V}{g I_{sp}}$$

TABLE 4-4. INERTIAL PROPERTIES

	W, pounds	I _{zz} , slug-ft ²	I _{xx} , slug-ft ²	R/P
Final orbit configuration*	618.3	55.8	50.3	1.11
Apogee motor burnout	754.4	69.7	57.2	1.22
Δ	136.1	13.9	6.9	0.11

*Complete bipropellant depletion — limit condition.

Assume

$$I_{sp} = 250 \text{ seconds}$$

$$\log \frac{W_1}{W_2} = \frac{1190}{32.2 (250)} = 0.148$$

$$W_2 = 754.4 e^{-0.148} = 754.4 (0.863) = 651 \text{ pounds}$$

$$\Delta W_{\text{NOMINAL}} = 754.4 - 651 = 103.4 \text{ pounds}$$

$$\frac{\Delta W_{\text{NOM}}}{\Delta W} = \frac{103.4}{136.1} = 0.76$$

$$I_{zz \text{ NOM}} = 69.7 - 0.76 (13.9) = 59.7$$

$$I_{xx \text{ NOM}} = 57.2 - 0.76 (6.9) = 52.0$$

$$R/P_{\text{NOM}} = 1.22 - 0.76 (0.11) = 1.14$$

$$\frac{I_{zz \text{ NOM}}}{I_{xx \text{ NOM}}} = 1.14$$

Average spacecraft properties during operation of jets are:

$$W_{\text{AVE}} = 754.4 + 651/2 = 702.5 \text{ pounds}$$

$$C = I_{zz \text{ AVE}} = 69.7 + 59.1/2 = 64.4 \text{ slug-ft}^2$$

$$A = I_{xx \text{ AVE}} = 57.2 + 52.0/2 = 54.6 \text{ slug-ft}^2$$

$$R/P = 1.22 + 1.14/2 = 1.18$$

Rewriting Equation 4-19c:

$$\dot{C}\omega_z = - \left\{ \dot{C}\omega_z + \sum_i^N \dot{m}_i \right.$$

$$\left. \left[(x_i^2 + y_i^2) \omega_z - z_i x_i \omega_i - z_i y_i \omega_y \right] \right\}$$

$$+ (A - B) \omega_x \omega_y + M_z$$

Assume that the variation of the quantities above will be small so that a difference equation will suffice.

4. Launch and Orbit Considerations

$$\begin{aligned}
 C\Delta\omega_z = & - \left\{ \Delta C\omega_z + \sum_I^N \Delta m_I \right. \\
 & \left. \left[(x_I^2 + y_I^2) \omega_z - z_I x_I \omega_x - z_I y_I \omega_y \right] \right\} \\
 & + (A - B) \omega_x \omega_y \Delta t + M_z \Delta t \\
 \Delta\omega_z = & - \left\{ \frac{\Delta C}{C} \omega_z + \sum_I^N \frac{\Delta m_I}{C} \right. \\
 & \left. \left[(x_I^2 + y_I^2) \omega_z - z_I x_I \omega_x - z_I y_I \omega_y \right] \right\} \\
 & + \left(\frac{A - B}{C} \right) \omega_x \omega_y \Delta t + \frac{M_z}{C} \Delta t
 \end{aligned}$$

Consider the jet damping term in the braces

$$\frac{\Delta C}{C} \omega_z = \frac{59.1 - 69.7}{64.4} (10.47) = -1.725 \text{ rad/sec}$$

$$\Delta m_1 = .87 (103.4) = 90 \text{ pounds} = 2.80 \text{ slugs}$$

$$\Delta m_2 = .13 (103.4) = 13.4 \text{ pounds} = 0.416 \text{ slugs}$$

For a body of symmetry subjected to a transverse moment with negligible jet damping,

$$M_x = A\dot{\omega}_x - (A - C) \omega_y \omega_z$$

$$M_y = A\dot{\omega}_y + (A - C) \omega_x \omega_z$$

$$0 = C\dot{\omega}_z$$

$$\omega_x = q \quad \omega_y = r \quad \omega_z = p$$

since

$$\dot{p} = 0 \quad p = p_0$$

$$M_x = A \dot{q} - (A - C) p_0 r$$

$$i M_y = i A \dot{r} + (A - C) p_0 q$$

$$M_T \equiv M_x + i M_y = A \dot{\omega} - i \lambda p_0 \omega$$

where

$$\lambda \equiv \frac{C - A}{A}$$

$$\omega \equiv \omega_x + i \omega_y = q + i r$$

For M_T constant

$$\omega = \omega(0) e^{i \lambda p_0 t} + \frac{i M_T}{\lambda p_0 A} (1 - e^{i \lambda p_0 t})$$

Neglecting the transcendental terms,

$$\omega_x + i \omega_y = \frac{i M_x}{\lambda p_0 A}$$

$$= \omega_y = \frac{M_x}{(C - A) p_0} = \frac{-2.16 (5.0)}{(9.8) 10.47}$$

$$= -0.1055$$

$$-z_1 y_1 \omega_y = - (2.0 (2.16) (-0.1055)) = +0.456$$

$$y_1^2 \omega_z = (2.16)^2 10.47 = +48.8$$

$$-z, y, \omega_y + y^2, \omega_z = +49.256$$

$$\frac{\Delta m_1}{C} \left[(x_1^2 + y_1^2) \omega_z - z_1 x_1 \omega_x - z_1 y_1 \omega_y \right]$$

$$= \frac{2.80}{64.4} [49.26] = 2.15 \text{ rad/sec}$$

$$\frac{\Delta m_2}{C} \left[(x_2^2 + y_2^2) \omega_z - z_2 x_2 \omega_x - z_2 y_2 \omega_y \right]$$

$$= \frac{0.416}{64.4} [48.8] = 0.316 \text{ rad/sec}$$

$$\Delta\omega_{z \text{ JET DAMPING}} = - \left[-1.725 + 2.15 + .316 \right]$$

$$= -0.74 \text{ rad/sec} = -7.1 \text{ rpm.}$$

Since variation of ω_z due to jet damping is only 7 percent, the previous assumptions concerning small $\dot{\omega}_z$ are permissible.

Consider the $(A - B/C) \omega_x \omega_y \Delta t$ term. The effect of skew torque can be evaluated at this time by assuming a variation in A and B and orienting the axial jet in a most disadvantageous position.

$$A = 54.6 (0.975) = 53.2 \text{ sl-ft}^2 \quad (\text{i.e.} - 2\frac{1}{2} \text{ percent})$$

$$B = 54.6 (1.025) = 56.0 \text{ sl-ft}^2 \quad (\text{i.e.} + 2\frac{1}{2} \text{ percent})$$

$$C = 64.4 \text{ sl-ft}^2$$

$$\Delta\omega_z = \left(\frac{A-B}{C} \right) \frac{(M_T^2) \sin \phi \cos \phi}{(C-A)(C-B)p_0^2} (\Delta t)$$

$$|\sin \phi \cos \phi|_{\max} = 0.5 \quad \text{worst case}$$

$$I_{\text{AXIAL}} = m\Delta V_A = \frac{702.5}{32.2} (1035) = 22,500 \text{ lb-sec}$$

$$\Delta t = \frac{22,500}{5} = 4500 \text{ seconds}$$

$$|\Delta\omega_z| = \left(\frac{56.0 - 53.2}{64.4} \right) \frac{(5(2.16))^2 \cdot 5}{(64.4 - 53.2)(64.4 - 56.0)(10.47)^2} (4500)$$

$$\Delta\omega_z = \pm 1.105 \text{ rad/sec} = \pm 10.55 \text{ rpm}$$

Thus the original assumptions are permitted since only a 10 1/2 percent change in ω_z is indicated.

In the $(M_z/C)\Delta t$ term, M_z will be caused by jet misalignments.

Consider the axial jet:

$$I_{\text{AXIAL}} = M\Delta V_A = 22,550 \text{ lb-sec}$$

$$\Delta\omega_{zA} = \frac{I_A \delta_A l}{C}$$

$$\Delta\omega_{zA} = \frac{22,550 (\delta_A) 2.16}{64.4} \left(\frac{60}{2\pi(57.3)} \right)$$

$$= 126.5 \text{ rpm/degree misalignment}$$

Consider the radial nozzle:

$$\Delta V_R = 155 \text{ fps}$$

$$I_{\text{RADIAL}} = \frac{702.5}{32.2} (155) = 3380 \text{ lb-sec}$$

$$\Delta\omega_{zR} = \frac{I_{\text{RADIAL}} (\delta_R) l}{C}$$

$$\Delta\omega_{zR} = \frac{3380 \delta_R 2.16}{64.4} \left(\frac{60}{2\pi 57.3} \right)$$

$$= 18.9 \text{ rpm/degree misalignment}$$

$$\Delta\omega_z = \frac{I_R \epsilon}{C}$$

$$\Delta\omega_{zC} = \frac{3380}{64.4(12)} \frac{60}{2\pi}$$

$$= 41.8 \text{ rpm/inch cg offset}$$

AXIAL JET INDUCED NUTATION

Since the axial jet is operated in a synchronously pulsed mode to generate precession torque pulses and in a continuous (long pulse) mode to generate a translational force, it is desirable to estimate the maximum nutation angle, $\theta_{\max}$, induced by either of the above modes. It is shown that for a given roll-to-pitch moment of inertia ratio I_z/I_x , spin speed ω_z , and torque level N_o , the value of $\theta_{\max}$ after continuous axial jet operation may be greater (seven to eight times) than the maximum nutation angle induced by a series of spin synchronous torque pulses. Continuous mode operation of the axial jet will result in the maximum nutation angles for nominal and limit parameters shown in Table 4-5.

$\theta_{\max}$ was computed using

$$\theta_{\max} = \frac{2N_o}{\omega_z^2 \left(\frac{I_z}{I_x} - 1 \right) I_z}$$

The nutation damper will asymptotically damp $\theta_{\max}$ to zero with an estimated damper time constant, $\tau_n \leq 5$ minutes.

TABLE 4-5. MAXIMUM NUTATION ANGLES FOR NOMINAL AND LIMIT PARAMETERS

Parameter	Nominal (50 percent) Propellant Depletion	Limit (100 percent) Propellant Depletion
ω_z , rpm	100	90
I_z/I_x	1.17	1.11
I_r , slug-ft ²	62.8	55.83
N_o , ft-lb	10	6.5 (~ 3-pound thrust)
$\theta_{\max}$, degrees	0.97	1.36

4. Launch and Orbit Considerations

Mathematical Model. Considering the spacecraft as an undamped, spinning symmetrical rigid body with constant spin speed ω_z , whose spin axis moment of inertia is greater than the transverse moment of inertia, $I_z > I_x = I_y$, the Euler equations of motion (in body coordinates x, y, z) resulting from synchronous axial jet pulsing (pulsing) may be represented in complex form by (note: $\dot{\omega} = (d/dt)\omega$)

$$\dot{\omega} + j\Omega\omega \approx n \quad ; j = \sqrt{-1} \quad (4-20)$$

$$\Omega = \left(1 - \frac{I_z}{I_x}\right) \omega_z \equiv \omega_z - \omega_n \equiv \left(1 - \frac{I_z}{I_x}\right) \frac{2\pi}{\tau_s} < 0 \quad (4-21)$$

$$n = \begin{cases} n_o = \frac{N_o}{I_x} ; q(m+k) \tau_s \leq t \leq [q(m+k) + k] \tau_s \\ 0 \quad ; [q(m+k) + k] \tau_s \leq t \leq (q+1)(m+k) \tau_s \end{cases} \quad (4-22)$$

$q = 0, 1, 2, \dots, \nu - 1$

where

- $\omega = \omega_x + j \omega_y$
= component of body angular velocity normal to spin axis, radians per second
- τ_s = spin period, seconds
= $2\pi/\omega_z \leq 0.8$ seconds (75 rpm)
- ω_n = nutation frequency, radians per second
= $\frac{I_z}{I_x} \omega_z$
- n = normalized torque, ft-lb/slug-ft²
= $N/I_x = \frac{1}{I_x} (N_x + j N_y)$
- k = fraction or number of "jet-on" spin periods per on-off cycle
> 0
- m = fraction or number of "jet-off" spin periods per on-off cycle
≥ 0

$$\frac{k}{m+k} = \text{duty cycle of axial jet}$$

$$q = \text{running index of on-off cycles} \\ = 0, 1, 2, \dots, \nu - 1$$

$$\nu = \text{total number of on-off cycles employed in a pulse series}$$

For synchronous pulse torquing at the rate of one pulse per spin period,

$$0 < k < 1$$

$$m + k = 1 \quad (4-23)$$

and ν , the total number of pulses, is completed in $\tau \equiv (\nu - m) \tau_s$ seconds. For the continuous operating mode, however, the thrust history of the axial jet is described by a single, long duration pulse of many spin periods, i.e.,

$$k > > 1$$

$$m = 0$$

$$(4-24)$$

and the action time of the axial jet $\tau_a = k \tau_s$ seconds.

A solution to Equation 4-20 for ω is given by

$$\omega = \omega_o e^{-j\Omega\tau} + \frac{2n}{\Omega} \frac{\sin\left(\frac{\Omega}{2} k \tau_s\right) \left[\sin\frac{\Omega}{2} \nu (m+k) \tau_s \right]}{\sin\left[\frac{\Omega}{2} (m+k) \tau_s\right]} e^{-j\frac{\Omega}{2} [\nu (m+k) - m] \tau_s} \quad (4-25)$$

where

$$\tau = \text{total time for completion of pulse series}$$

$$= [\nu (m+k) - m] \tau_s$$

$$\omega_o = \text{initial component of body angular velocity normal to spin axis (assumed zero)}$$

Now since the subsequent nutation angle θ is given by

$$\tan \theta = \frac{\omega}{\omega_n} = \frac{\omega}{\omega_z} \frac{I_x}{I_z} \approx \theta \quad (4-26)$$

then from Equations 4-21 and 4-25,

$$\begin{aligned} \tan \theta \approx \theta = \frac{\omega_o}{\omega_n} e^{-j\Omega\tau} \\ + \frac{2N \sin \left[\pi k \left(1 - \frac{I_z}{I_x} \right) \right] \sin \left[\pi \nu (m+k) 1 - \left(\frac{I_z}{I_x} \right) \right]}{\omega_z^2 I_z \left(1 - \frac{I_z}{I_x} \right) \sin \left[\pi (m+k) \left(1 - \frac{I_z}{I_x} \right) \right]} \\ \times \exp \left\{ -j \frac{\Omega}{2} \left[\nu (m+k) - m \right] \tau_s \right\} \quad (4-27) \end{aligned}$$

Pulsed Mode. For the case of one pulse per spin cycle, $(m+k=1)$, and $\omega_o=0$, Equation 4-27 becomes

$$\begin{aligned} \theta \approx \frac{2N \sin \left[\pi k \left(1 - \frac{I_z}{I_x} \right) \right] \sin \left[\pi \nu \left(1 - \frac{I_z}{I_x} \right) \right]}{\omega_z^2 I_z \left(1 - \frac{I_z}{I_x} \right) \sin \left[\pi \left(1 - \frac{I_z}{I_x} \right) \right]} \\ e^{-j \frac{\Omega}{2} (\nu-m) \tau_s} \quad (4-28) \end{aligned}$$

The above expression may be further reduced by examining the magnitude of

$$\sin \left(\frac{\Omega}{2} k \tau_s \right) = \sin \left[\pi k \left(1 - \frac{I_z}{I_x} \right) \right].$$

It is recalled that

$$\frac{\Omega}{2} k \tau_s = \frac{\omega_z}{2} \left(1 - \frac{I_z}{I_x} \right) k \tau_s = \pi k \left(1 - \frac{I_z}{I_x} \right)$$

For a 45-degree pulse width, $k = 1/8$, and since

$$\left| \left(1 - \frac{I_z}{I_x} \right) \right| \leq 0.3$$

$$\left| \frac{\Omega}{2} k \tau_s \right| \leq \frac{\pi}{8} (0.3) = 0.118 \text{ radian} \\ 6.75 \text{ degrees}$$

Thus

$$\sin \left(\frac{\Omega}{2} k \tau_s \right) \approx \frac{\Omega}{2} k \tau_s \cong \frac{\Omega}{2} \frac{\delta H}{N} = \frac{\omega_z}{2} \frac{\delta H}{N} \left(1 - \frac{I_z}{I_x} \right) \quad (4-29)$$

where δH is the incremental change in the direction of the angular momentum vector $H \cong I_z \omega_z$. Hence Equation 4-28 becomes (neglecting the phase angle)

$$\frac{|\theta|}{\delta H/H} \approx \frac{\left| \sin \left[\pi \nu \left(1 - \frac{I_z}{I_x} \right) \right] \right|}{\left| \sin \left[\pi \left(1 - \frac{I_z}{I_x} \right) \right] \right|} \quad (4-30)$$

Figure 4-9 shows a geometric construction of the history of θ during synchronous pulse torquing of a typical stable configuration ($I_z/I_x = 1.2$). When, in Equation 4-30

$$\nu = \frac{1-2r}{2 \left(1 - \frac{I_z}{I_x} \right)}; \quad r = 1, 2, 3, \dots; \quad (4-31)$$

$$\begin{aligned} |\theta| = |\theta|_{\max} &= \frac{\delta H}{H} \left| \frac{1}{\sin \left[\pi \left(1 - \frac{I_z}{I_x} \right) \right]} \right| \\ &= \frac{\pi N}{4 I_z \omega_z^2} \frac{1}{\sin \left[\pi \left(\frac{I_z}{I_x} - 1 \right) \right]} \quad (4-32) \end{aligned}$$

4. Launch and Orbit Considerations

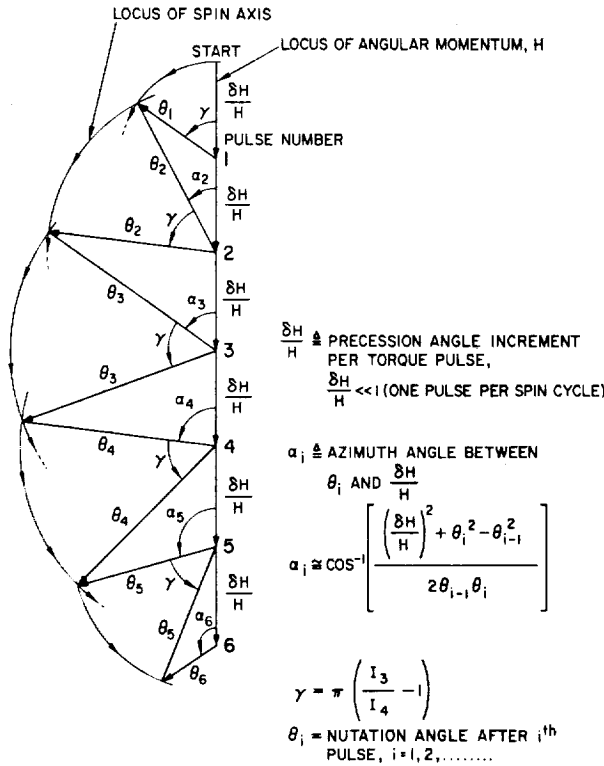


FIGURE 4-9. GEOMETRIC CONSTRUCTION OF NUTATION ANGLE HISTORY DURING SYNCHRONOUS PULSE TORQUING

where $k = 1/8$. Some values of $\theta_{\max}$ for $N = 10$ ft-lbs and $k = 1/8$ are given in Table 4-6 for typical and worst case moment-of-inertia ratios and spin speeds with $I_z = 56.75$ slug-ft² (bipropellant depletion).

Continuous Mode. For a single, long duration pulse, $m = 0$, $\nu = 1$, $\tau = \tau_a = k \tau_s$, so that Equations 4-25 and 4-26 reduce to

TABLE 4-6. MAXIMUM NUTATION ANGLE AFTER PULSED MODE

$\frac{I_z}{I_x}$	ω_s , rpm	$\theta_{\max}$, degree
1.1	75	0.42
1.2	75	0.217
1.1	100	0.238
1.2	100	0.112

$$\theta \cong \frac{\omega_0}{\omega_n} e^{-j\Omega\tau_a} + \frac{2n}{\Omega\omega_n} \sin\left(\frac{\Omega}{2}\tau_a\right) e^{-j\frac{\Omega}{2}\tau_a} \quad (4-33)$$

Since $\tau_a = k \tau_s \gg \tau_s$, the small angle approximation for $\Omega k \tau_s / 2$ can no longer be invoked. For example, if $I_z / I_x = 1.2$

$$\frac{\Omega k \tau_s}{2} = \pi k \left(1 - \frac{I_z}{I_x} \right) = \frac{\pi}{2} \quad (4-34)$$

when

$$k = \frac{2r - 1}{2 \left(\frac{I_z}{I_x} - 1 \right)} ; r = 0, 1, 2, \dots \quad (4-35)$$

$$\approx \frac{1}{0.4} = 2.5 ; r = 0$$

Thus for $\omega_0 = 0$ in Equation 4-33 the expression for $|\theta|_{\max}$ is approximated by

$$|\theta|_{\max} = \frac{2n}{\Omega\omega_n} = \frac{2N}{I_z \omega_z^2 \left(\frac{I_z}{I_x} - 1 \right)} \quad (4-36)$$

The ratio of Equation 4-36 to Equation 4-32 shows

$$\frac{|\theta_{\max}|_{\text{contin.}}}{|\theta_{\max}|_{\text{pulsed}}} = 8 \frac{\sin \left[\pi \left(\frac{I_z}{I_x} - 1 \right) \right]}{\pi \left(\frac{I_z}{I_x} - 1 \right)} \geq 6.9$$

≤ 8

for $1 < I_z / I_x \leq 1.3$. Thus it is seen that the worst case hot gas jet induced nutation angle occurs after continuous operation of the axial jet.

Table 4-7 contains some values of the maximum nutation angle $\theta_{\max}$ induced by the axial jet operated in the continuous mode. These values are listed for various combinations of spin speed ω_s , and roll-to-pitch moment of inertia I_z / I_x , with the additional conservative assumptions that the torque, $N = 10$ ft-lb (maximum value) and $I_z = 56.75$ slug-ft² (minimum value).

TABLE 4-7. MAXIMUM INDUCED NUTATION ANGLE, (θ_{max})*

ω_z , rpm	$\frac{I_z}{I_x}$	θ_{max} , degrees	N, ft-lb	I_z , slug-ft ²
75	1.10	3.25 (2.11)**	10 (6.5)**	56.75
75	1.13	2.50 (1.63)	10 (6.5)	56.75
100	1.10	1.82 (1.18)	10 (6.5)	56.75
75	1.22	1.48	10	56.75
100	1.13	1.40 (0.91)	10 (6.5)	56.75
75	1.30	1.09	10	56.75
100	1.22	0.83	10	56.75
100	1.30	0.61	10	56.75

*Continuous mode operation of axial jet for over 2½ spin periods.

**Parenthetical values correspond to final blow-down thrust level of approximately 3 pounds.

EMERGENCY PROCEDURES (ASCENT SEQUENCE)

Some preliminary considerations are discussed in the following paragraphs of correctable booster-spacecraft failures that may occur during the launch and ascent sequence beginning with Agena second burn cutoff and terminating when the spacecraft is initially placed on station with normal spin axis attitude.

Failures associated with AGE, computer programs, and ground-based portions of the telemetry and command subsystem will not be considered at this point. Failures incurred by Atlas-Agena equipment prior to Agena second burn cutoff that may be corrected (in part) by the spacecraft control system will be examined as more information is made available on the feasibility of such corrective procedures.

Table 4-8 is a tentative list of failures and possible causes associated with their respective nominal events in the order of their occurrence (except, possibly, events 7 and 8). The third column lists only the major components or subsystems involved in a particular failure although some intrasubsystem failure modes may occur without degrading the subsystem function. Fail-

ure of a subsystem in the spacecraft not listed in Table 4-8 may require another control mode but not a redundant subsystem. Qualitative fix methods are described below after each nominal event failure.

TABLE 4-8. CORRECTABLE FAILURES (TENTATIVE)—ASCENT SEQUENCE

Nominal Event	Failure Criteria	Possible Cause
Agena second burn cutoff	Fuel depletion (excess orbital energy)	Agena velocity meter
Agena yaw maneuver	Final attitude error > 3 degrees	Programmer/inertial reference package, Agena attitude jets, power supply
Spacecraft spinup	Spin speed error, $\delta\omega_z > 25$ rpm	Spin table rockets, (centrifugal switch), SS/D timer, power supply
Spacecraft separation	Final attitude error > 3 degrees	Tipoff
Apogee motor ignition	No change in orbital elements	Timer, telemetry and command subsystem igniter, power supply
Spin axis orientation	Little or no change in attitude	Axial valves, telemetry and command subsystem, jet control electronics, power supply
In-plane velocity correction	Little or no change in orbital elements (drift rate)	Radial valves, telemetry and command subsystem, jet control electronics, power supply
Inclination correction	Little or no change in orbital elements (inclination)	Axial valves, telemetry and command subsystem, jet control electronics, power supply

Qualitative Fix Procedures

- 1) Agena second burn cutoff (fuel depletion)
 - a) Change apogee boost attitude and ignition time to minimize orbital energy deviation from nominal, consistent with visibility requirements from telemetry and command stations.*

*More detailed information on visibility from ground stations will be forthcoming after more definitive fuel depletion injection parameters are made available.

4. Launch and Orbit Considerations

- b) Use spacecraft hot gas system to correct subsequent synchronous orbit period, eccentricity, and inclination (where possible). Use axial jet* operating in continuous (more efficient) mode if velocity requirement is greater than normal available velocity capability after allowance for longitude stationkeeping.
- 2) Agent yaw maneuver
 - a) Reorient spin axis to proper attitude before apogee boost using sun sensors and polarization measurements (pancake beam) to determine attitude.
 - 3) Spacecraft spinup
 - a) Activate axial jet (continuous and pulsed modes) after apogee motor burn if spin speed is high.
 - b) Activate axial jet (continuous mode) before apogee motor burn if spin speed is low, and adjust attitude and/or ignition time of apogee boost (if necessary) to minimize deviation of orbital energy from nominal.
 - 4) Spacecraft separation (same as procedure 2)
 - 5) Apogee motor ignition
 - a) Determine probable failure cause from telemetry data.
 - b) Select appropriate redundant mode or subsystem and command apogee ignition near first apogee consistent with

*With spin axis oriented in orbital plane.

telemetry and command station visibility.*

- 6) Spin axis orientation
 - a) Determine probable failure cause from telemetry data.
 - b) Select appropriate redundant subsystem or mode and command orientation maneuver at favorable time of day.
- 7) In-plane velocity correction (same as procedure 6 except radial jets are used)
- 8) Inclination correction (same as procedure 6 except axial jets are used in continuous mode)

The following list delineates the probable sources of failure indication associated with each nominal event in the general chronological order of their availability. Only those sources marked with a + are sufficient indications that a failure has occurred, while the other sources are merely necessary (though usually earlier) indications of failure. More quantitative descriptions of the failure indications will be forthcoming as the orbital injection, tracking station, and spacecraft design parameters become more definitive.**

Tentative Failure Indication Sources

- 1) Agena second burn cutoff (fuel depletion)
 - a) Agena guidance telemetry
 - b) FPS-16 data

*Usually, backup command firing can be initiated close to the first apogee in the event of a timer failure.

**The detailed description of failure mode procedures and data will generally follow the format of "Syncom I, A-26 Failure Mode Procedures and Data," D. D. Williams, Hughes Aircraft Company, IDC, Revision I June 1963.

- c) Range, range rate, and angle data (transponder ON)
- d) Minitrack data
- e) GSFC orbital elements⁺
- 2) Agena yaw maneuver
 - a) Agena guidance telemetry
 - + {
 - b) Spacecraft polarization (before spin-up)
 - c) Spacecraft sun angle, ϕ , determination (after spinup)
- 3) Spacecraft spinup
 - a) Agena spin rate detection/telemetry
 - b) Spacecraft sun sensor/telemetry (spin speed)⁺
 - c) Spacecraft radial accelerometer (if available)
- 4) Spacecraft separation (tipoff)
 - a) Agena guidance telemetry
 - + {
 - b) Spacecraft sun angle (ϕ) determination
 - c) Polarization angle measurement
- 5) Apogee motor ignition
 - a) Spacecraft axial accelerometer (if available)
 - b) Ground observer (weather and local night time permitting)
 - c) Sun sensor telemetry (spin speed change)
 - d) Solar panel power modulation (spin speed change)

- e) Spacecraft temperature
- f) GSFC orbital elements⁺
- 6) Spin axis orientation
 - a) Propellant tank pressure and solenoid current
 - b) Axial jet chamber and propellant tank temperature
 - c) Spin axis elements (ϕ angle and polarization angle)⁺
- 7) In-plane velocity correction
 - a) Propellant tank pressure and solenoid current
 - b) Radial jet chamber and propellant tank temperature
 - c) GSFC orbital elements (period and eccentricity)⁺
- 8) Inclination correction
 - a) Propellant tank pressure
 - b) Axial jet chamber and propellant tank temperature
 - c) GSFC orbital elements (inclination)⁺

In procedures 6, 7, and 8 a more detailed examination will be made of the sequence and type of valve failures (stuck open or closed) of the fuel and oxidizer valves of both radial and axial jets. This should lead to a more meaningful delineation of corrective procedures for each type of failure. For example, if during event 7, either a fuel or an oxidizer valve (but not both) sticks open then it is desirable to continue to operate with this radial jet and suffer the moderate loss in thrust effectiveness per pulse due to the low counteracting thrust component resulting from the fluid mass loss of the stuck-open valve; more-

over, the repeated on-off excitation of the valve solenoid may free the stuck valve.

However, if one of the fuel or oxidizer valves sticks closed then it is desirable to switch to the redundant radial jet and associated propellant supply. If, in addition, one of the valves in the redundant radial jet sticks closed then the spin axis is reoriented into the orbital plane (via pulsed axial jet operation) so that the thrust vector of the axial jet (operating in the continuous mode) is parallel or antiparallel to the orbital velocity vector depending on the desired longitude drift rate direction. Similar considerations will be given to axial jet valve failures.

Recent estimates of propellant depletion times* after a valve sticks open yield the following conservative values:

Propellant Load at Valve Failure	Depletion Time Oxidizer, hours	Fuel
100 percent	1.4	1.3
50 percent	0.75	0.70

Thus sufficient time exists to complete any initial ascent guidance correction function (in-plane or inclination) after a valve sticks open. During this time the failed jet thrust value decreases by less than 45 percent of initial value.

EARTH SENSOR FOR ADVANCED SYNCOM

This discussion is an extension and refinement of the concepts expressed in the section *Earth Sensor for Syncom* from the Advanced Syncom Monthly Progress Report for April 1963. Revised signal-to-noise-ratio calculations indicate the feasibility of reducing the aperture size,

which in turn allows for a single objective rather than the dual objective system considered previously. The earth sensor system in providing a reference for the phased array control electronics (PACE) of Advanced Syncom, allows for elimination of the associated vernier beam shifting circuits and the 2.81-minute clock as reliability alternatives to the circuit complexity required for earth sensor operation. Additionally, the earth sensor provides attitude information, and an earth centered reference for jet firing control.

The basic earth sensor, which is sensitive to infrared radiation in the 14 to 18 micron range, consists of a silicon objective, the optic axis of which is in the satellite equatorial plane and normal to the spin axis, and two thermistor detector flake elements in the focal plane, with an angular separation of ± 6.5 degrees from the optic axis, and coplanar with the spin axis (Figure 4-10).

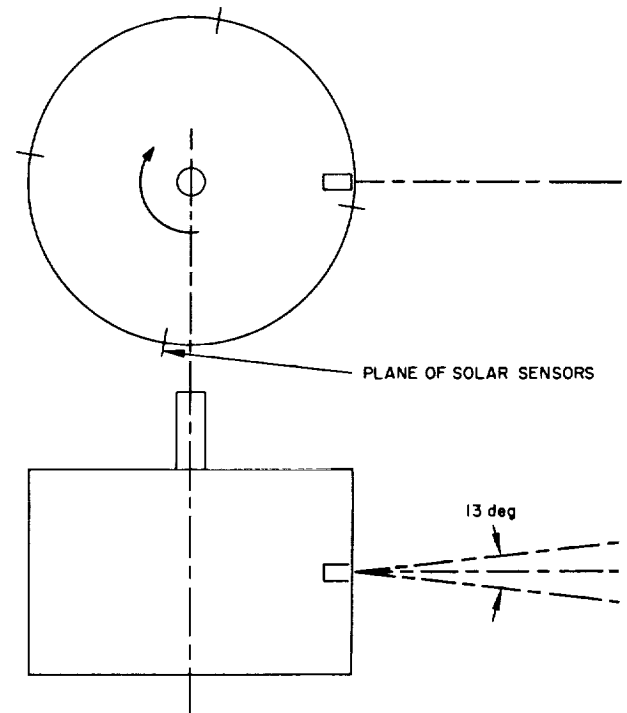


FIGURE 4-10. LOCATION OF EARTH SENSOR

*J. M. Ferry, "Failure Analysis Advanced Syncom RCS Injector Value," Hughes Aircraft Company, IDC, 26 September 1963.

As the satellite rotates, the focused infrared image of the earth sweeps across the detectors, generating pulses. If the optic axis of the earth sensor passes through the center of the earth, both pulses occur simultaneously.

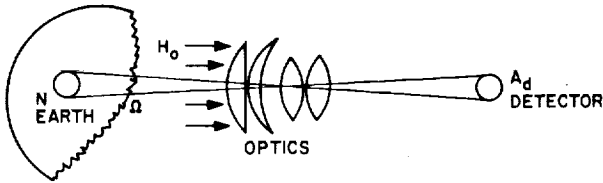


FIGURE 4-11. ILLUSTRATION FOR SENSITIVITY CALCULATIONS

Sensitivity Calculations. Assume that in the spectral band $\Delta\lambda$, the earth effectively radiates at a black body temperature T° Kelvin (see Figure 4-11). Let N equal the total black body radiance (watt/cm²Ω), and η equal the spectral utilization factor (the fraction of the total energy in $\Delta\lambda$). The effective irradiance at the aperture is

$$H' = N \eta \Omega \text{ (watts/cm}^2\text{)} \quad (4-37)$$

where Ω is the solid angle field of view in steradians. The effective flux transmitted through the objective is

$$F' = H' A_o \rho \quad (4-38)$$

where A_o is the area of the objective and ρ is the transmissivity of the optics. The noise equivalent power (the flux on the detector to produce a signal-to-noise ratio of unity) of the detector is related to the detectivity as

$$\text{NEP} = \frac{(A_d \Delta f)^{1/2}}{D^*} \text{ (watts)} \quad (4-39)$$

where A_d is the detector area, and Δf is the bandwidth of the detector amplifier. The maximum D^* for a thermistor-bolometer in the usual bridge circuit (active plus compensating flake) is

$$D^* = 0.8 \times 10^8 \sqrt{\tau} \text{ (cm cps)}^{1/2} \text{ / watt} \quad (4-40)$$

where τ is the detector time constant in milliseconds. The voltage signal-to-noise ratio is therefore

$$\frac{S}{N} = \frac{F'}{\text{NEP}} \quad (4-41)$$

Substituting from Equations 4-38 and 4-39,

$$\frac{S}{N} = \frac{N \eta \Omega A_o \rho D^*}{(A_d \Delta f)^{1/2}} \quad (4-42)$$

Assume the effective earth temperature as 210° K in the 14 to 18 micron range (the CO₂ band). Then $N = 3 \times 10^{-3}$ watts/cm²Ω, and $\eta = 0.18$. Let the field of view of each sensor be 1.3 degrees square. Then $\Omega = a^2 = (1.3 \times 1.75 \times 10^{-2})^2 = 5.17 \times 10^{-4}$ steradians. Assume the detector area is 10⁻² cm², and the detector time constant 2.5 milliseconds. Then $D^* = 1.26 \times 10^8$ (cm cps)^{1/2}/watt. Let $\rho = 0.3$ (including the bandpass filter), $\Delta f = 135$ cps, and the signal-to-noise ratio 30 db (31.6).

Solving for A_o ,

$$\begin{aligned} A_o &= \frac{S}{N} \frac{(A_d \Delta f)^{1/2}}{N \eta \Omega \rho D^*} \quad (4-43) \\ &= \frac{31.7 (10^{-2})^{1/2} (11.62)}{(3 \times 10^{-3}) (0.18) (5.17 \times 10^{-4}) (0.3) (1.26 \times 10^8)} \\ &= 0.35 \text{ cm}^2 \end{aligned}$$

which is the area of the objective. The corresponding diameter is $2\sqrt{A/\pi}$ or 0.667 cm (0.262 inches). The limiting optical resolution of the aperture at 18 microns is

$$\begin{aligned} d_r &= \frac{1.22\lambda}{D} \text{ radians} \quad (4-44) \\ &= \frac{1.22(18)}{6.67 \times 10^3} = 3.29 \times 10^{-3} \text{ radian} \\ &= 0.188 \text{ degree} \end{aligned}$$

which is greater than the angular resolution as limited by a 30-db signal-to-noise ratio.

The focal distance d_f is determined by the proportion

$$\frac{0.01}{d_f} = \tan 1.3 \text{ degrees} = 0.0227$$

$$d_f = 0.44 \text{ cm}$$

$$= 0.173 \text{ inches}$$

The focal ratio is $0.44/0.667 = f0.66$ which is not excessively low, and which is capable of producing a reasonably sharp image over the central 14.3-degree field of view of the objective.

From a knowledge of diameter and focal length of the objective, an estimation may be made of the size and weight of the earth sensor exclusive of the electronics (Figure 4-12). The optics and detectors are envisioned as being enclosed in a tube about 1.5 cm long, and 1 cm in diameter. The weight is less than 1 ounce.

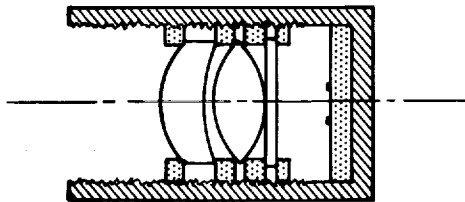


FIGURE 4-12. CROSS SECTION OF EARTH SENSOR OPTICAL UNIT

Analysis. If the spin axis is normal to the earth vehicle line, which is the desired orientation, both fields of view will simultaneously intersect a line of the earth. For small angular deviations from the desired orientation, there will be a time difference of this intersection. The geometry of the situation is illustrated in Figure 4-13. Assume that the fields of view of the sensors are of negligible angular dimension. From a consideration of Figure 4-13

$$d = r (1 - \cos \beta/2) \quad (4-45)$$

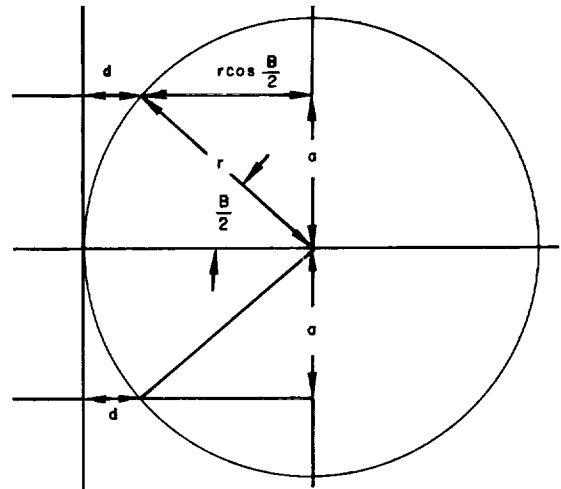


FIGURE 4-13. EARTH SENSOR SCAN GEOMETRY

$$\cos \beta/2 = \sqrt{1 - \left(\frac{a}{r}\right)^2} \quad (4-46)$$

$$d = r \left(1 - \sqrt{1 - \left(\frac{a}{r}\right)^2}\right) \quad (4-47)$$

$$\frac{\Delta a}{\Delta d} = \frac{r}{2a} \sqrt{1 - \left(\frac{a}{r}\right)^2} \quad (4-48)$$

which represents the relationship between time difference of pulses and pointing error.

Consider a square aperture of the above dimensions scanning across a uniform earth (see Figure 4-14). The relative sensor output is

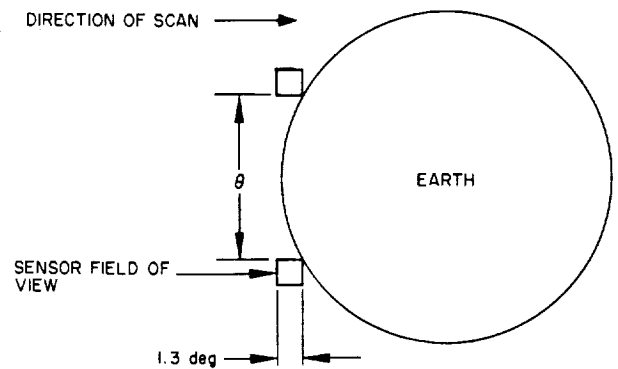


FIGURE 4-14. ILLUSTRATION FOR DEVELOPING STATIC OUTPUT OF DETECTOR-AMPLIFIER

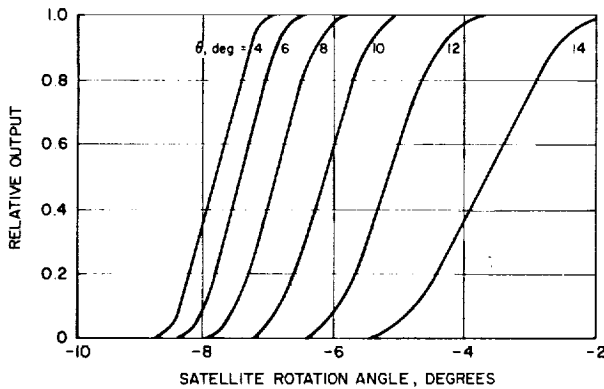


FIGURE 4-15. STATIC OUTPUT OF DETECTOR-AMPLIFIER

plotted as a function of angle for various chords (Figure 4-15) in the absence of time constants, in the detector or amplifier. These functions may be approximated by ramp functions.

$$G_{DA} = \frac{1}{\tau_1 \tau_3} \left[\frac{s}{\left(s + \frac{1}{\tau_1}\right) \left(s + \frac{1}{\tau_2}\right) \left(s + \frac{1}{\tau_3}\right)} \right] \quad (4-49)$$

where τ_1 is the cell time constant, and τ_2 and τ_3 are the lower and upper amplifier time constants. The values for these parameters will be taken as 2.5, 4.56, and 0.94 milliseconds, respectively (assuming an amplifier bandpass of 35 to 170 cps).

The response of the detector-amplifier to a ramp function of K volts per second slope, and duration τ_4 seconds is

$$E_{out} = \frac{K}{\tau_1 \tau_3} \left[\frac{1}{s \left(s + \frac{1}{\tau_1}\right) \left(s + \frac{1}{\tau_2}\right) \left(s + \frac{1}{\tau_3}\right)} \right] \left[1 - e^{-\tau_4 s} \right] \quad (4-50)$$

The corresponding time function is readily obtained by partial fraction expansion. It is

$$E_{out(t)} = \frac{K}{\tau_1 \tau_3} \left[\frac{e^{-t/\tau_1}}{\left(\frac{1}{\tau_2} - \frac{1}{\tau_1}\right) \left(\frac{1}{\tau_3} - \frac{1}{\tau_1}\right) \left(-\frac{1}{\tau_1}\right)} + \frac{e^{-t/\tau_2}}{\left(\frac{1}{\tau_1} - \frac{1}{\tau_2}\right) \left(\frac{1}{\tau_3} - \frac{1}{\tau_2}\right) \left(-\frac{1}{\tau_2}\right)} + \frac{e^{-t/\tau_3}}{\left(\frac{1}{\tau_1} - \frac{1}{\tau_3}\right) \left(\frac{1}{\tau_2} - \frac{1}{\tau_3}\right) \left(-\frac{1}{\tau_3}\right)} + \tau_1 \tau_2 \tau_3 \right] \quad (4-51)$$

$$- \frac{K}{(\tau_1 - \tau_3)} \left[\frac{e^{-\frac{t - \tau_4}{\tau_1}}}{\left(\frac{1}{\tau_2} - \frac{1}{\tau_1}\right) \left(\frac{1}{\tau_3} - \frac{1}{\tau_1}\right) \left(-\frac{1}{\tau_1}\right)} + \frac{e^{-\frac{t - \tau_4}{\tau_2}}}{\left(\frac{1}{\tau_1} - \frac{1}{\tau_2}\right) \left(\frac{1}{\tau_3} - \frac{1}{\tau_2}\right) \left(-\frac{1}{\tau_2}\right)} + \frac{e^{-\frac{t - \tau_4}{\tau_3}}}{\left(\frac{1}{\tau_1} - \frac{1}{\tau_3}\right) \left(\frac{1}{\tau_2} - \frac{1}{\tau_3}\right) \left(-\frac{1}{\tau_3}\right)} + \tau_1 \tau_2 \tau_3 \right] t \geq \tau_4$$

This time function is plotted in Figure 4-16 for ramp function 5 of Figure 4-15 at a nominal spin rate of 100 rpm. The leading edge of the pulse may be used to trigger a threshold circuit set at an appropriate level.

The basic signal processing applied to the detector-amplifier pulses is illustrated in Figure 4-17. Threshold levels are set in two monostable multivibrators triggered by the detector-amplifiers. The outputs of these multivibrators are equal and opposite in magnitude and may be determined by zener diodes. If the outputs of the two multivibrators are summed together, a rec-

4. Launch and Orbit Considerations

tangular pulse is produced, the polarity of which indicates the time difference between the two sensor outputs. After a predetermined time period (longer than the time it takes for the field of view to scan the whole earth) the first multivibrator to be triggered reverts back to its initial state and causes the other multivibrator to also revert back to its initial state.

The pulse train thus produced may be low-pass filtered and the subsequent dc level telemetered on a narrow-band channel.

When considering the effect of noise of the detector-amplifier on the signal processing scheme outlined above, it is necessary to consider noise triggering of the threshold detectors. Rice*

*S.O. Rice, "Mathematical Analysis of Random Noise," reprint by Dover Publications, 1954.

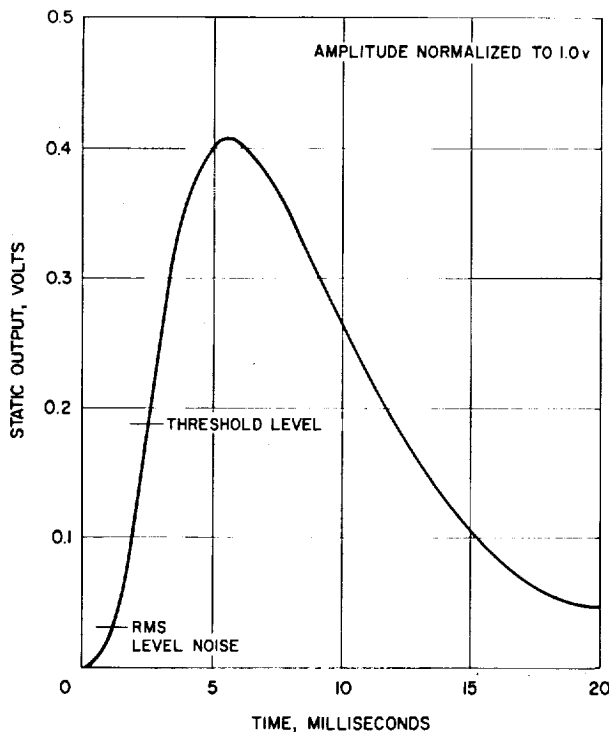


FIGURE 4-16. DYNAMIC OUTPUT OF DETECTOR-AMPLIFIER

gives the following expression for band-limited gaussian noise for the expected number of threshold crossings per second with positive slope. It is

$$\left[\frac{1}{3} \frac{fb^3 - fa^3}{fb - fa} \right]^{1/2} e^{-K^2/2} \quad (4-52)$$

where $K\sigma$ is the threshold level, and fb and fa define the noise bandwidth. For an amplifier bandwidth of 35 to 170 cycles and $K = 6$, the expected number of crossings per second is

$$\left[\frac{1}{3} \frac{170^3 - 35^3}{170 - 35} \right]^{1/2} e^{-18} \quad (4-53)$$

or 1.67×10^{-6} per second for a single detector amplifier, or 3.34×10^{-6} per second for two detector amplifiers, since either detector amplifier is capable of causing a false alarm. The false alarm rate is $3.34 \times 10^{-6} \times 3600$ or 12.02×10^{-3} per hour or a mean time of 83.2 hours between occurrences.

If the spin axis is misaligned such that the earth is not intersected by the sensors no output will occur. This may be distinguished from the null position when the spin axis is properly aligned by the fact that a small precession of the spin axis will not cause an output to occur. If the sun or moon is intersected, maximum out-

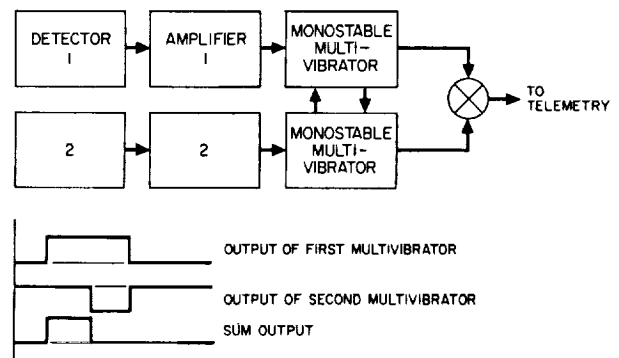


FIGURE 4-17. BASIC SIGNAL PROCESSING

put will occur, but only due to a single sensor since the subtended angle ($1/2$ degree) is less than the angular separation of the two sensors.

If only one sensor intersects the earth maximum output occurs, the polarity of which indicates which direction to precess the spin axis. Situations could arise where one sensor intersects the earth, the other the sun or moon, etc., but these conditions are predictable and none of them give the same as the desired output pattern near the null.

The sources of error affecting the pointing accuracy are cell and amplifier noise, telemetry link noise, non-uniformity of infrared emission from earth, and boresight misalignment of the optical axis. Since, as previously stated, in the 14 to 18 micron range of radiation the earth is a reasonably uniform source, that contribution of error may be taken as negligibly small, especially for a system utilizing edge detection.

A 30.0-db signal-to-noise ratio of the detector amplifier places the rms noise level at 0.0316 of the static (dc) output or about 0.0776 of the peak dynamic output of the detector amplifier. At the threshold level, the slope of the detector-amplifier pulse is 0.125 volts per millisecond. The uncertainty of when the threshold circuit is triggered is then $0.0316/0.125 = 0.253$ millisecond as the deviation or 0.358 millisecond as the deviation of the difference between the two detector-amplifiers. This deviation is reduced by two factors before entering the telemetry down link, low-pass filtering of the multivibrator pulse and by the geometrical factor developed previously.

The bandwidth of the detector-amplifier is approximately 100 cycles wide. Since the spacecraft axis is not precessing, a 1-cps low-pass filter may be used, giving a 10:1 reduction

of the time uncertainty. The time uncertainty is then 0.0358 millisecond or 0.0215 degree at a 100-rpm spin rate.

As the angular separation of the optic axes increases, the geometry further reduces the angular uncertainty, but this improvement is partially offset by decreasing attack angle of the limb by the sensor field of view which reduces the slope of the detector-amplifier pulse, correspondingly increasing the time uncertainty.

A reasonable separation is taken as 13 degrees. The improvement factor is then 0.434 as computed from Equation 4-48. The transmitted angular pointing error is then 0.00955 degree.

This very small error will be degraded by the telemetry down link, if transmitted over a single telemetry channel, because of the large dynamic range of the signal. The largest signal transmitted occurs when only a single sensor subtends the earth, equivalent to ± 18 degrees. If the peak signal-to-noise ratio of the telemetry channel is 30 db, the rms noise error is about 1.1 degree which is excessive. If a second telemetry channel is scaled to 35 times the sensitivity of the first, the three-sigma variance of the noise angle is 0.1 degree. Therefore the desirability of using two telemetry channels is evident for course and fine positioning of the spin axis.

Phased Array Antenna Control. The system of phased array antenna control by the earth sensor is illustrated in Figure 4-18. Pulses from either detector may generate a reference pulse for PACE, both of which occur simultaneously if the attitude of the vehicle is correct. Since there is a deviation of 0.358 millisecond on the time occurrence of each pulse, assume the first pulse provides the PACE reference. If P is the probability that one detection has occurred, then

$$P^2 = 2P - P^2$$

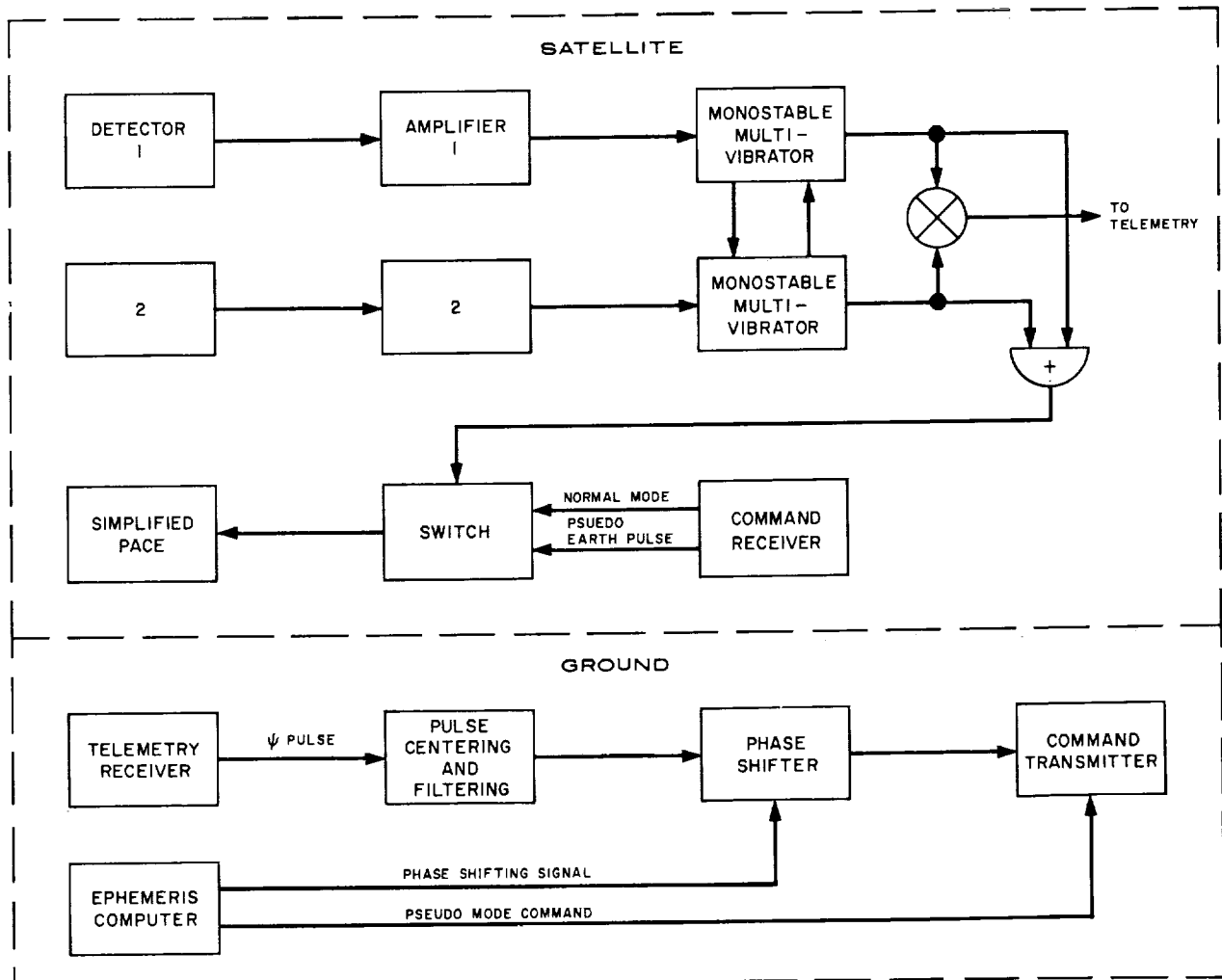


FIGURE 4-18. SIGNAL PROCESSING FOR EARTH SENSOR CONTROL OF PHASED ARRAY ANTENNA ELECTRONICS

is the probability that at least one of the two pulses has occurred. It follows that

$$P = 1 - \sqrt{1 - P'}$$

Let $P' = 0.25, 0.50$, and 0.75 . The corresponding values P are $0.133, 0.293$, and 0.500 . Relating these probabilities to the cumulative normal distribution, we find values of $1.1, 0.55$, and 0 . The mean time of occurrence of the first pulse is shifted ahead in time by $0.55/0.68$ sigma or 0.808 sigma (0.289 millisecond or 0.1735 degree). The deviation of the first pulse is $0.55/$

0.68 or 0.808 as large as the deviation of either pulse.

Assume the effective bandwidth of the PACE loop is 1 cps. Then the expected one sigma deviation of the beam is $0.1735/135$ or 0.01635 degree.

Inhibit Mode. If the solar or lunar image is incident upon either detector, an undesired pulse is generated. The lunar surface is more energetic in the 14 to 18 micron range than the earth, but subtends a smaller angle than the earth, so the

1.3 degree field of view is not completely filled. The net result is that the lunar and terrestrial images generate about the same amplitude pulse.

The declination of the sun from the satellite equatorial plane is

$$\delta = 23.45 \text{ degrees } \sin 2\pi t \quad (4-56)$$

where t is the fractional time of year measured from the vernal equinox. For a 1.3-degree field of view located 6.5 degrees from the equatorial plane, the sun will be in the field of view when its declination lies between ± 5.6 to ± 7.4 degrees. This means that the

Sun enters field of view	3 April
Sun leaves field of view	8 April
Sun enters field of view	3 September
Sun leaves field of view	8 September
Sun enters field of view	4 October
Sun leaves field of view	9 October
Sun enters field of view	3 March
Sun leaves field of view	8 March

Therefore, for about 20 days per year, at least a portion of the sun is in one of the fields of view. Allowing another 16 days per year as safety margin, the primary operation will be unusable for about 10 percent of the year due to the sun.

The orbit of the moon is inclined to ecliptic by 5 degrees, with the nodes regressing once around in 18.6 years. The moon will enter and leave the fields of view four times per month, each period being approximately 5/12 day, or 10 hours in extent. The earth sensor is affected only when the nodal crossings occur near the full moon, since the infrared illumination of the moon is considerably less at other times. Assuming the moon has sufficient irradiance for one

half its orbit, centered around full, to cause earth sensor response, the earth sensor will be inoperative 5 percent of the time because of the moon, allowing the same percentage safety margin as in the solar case. The total average time in which the sensor is affected by sun or moon is then $0.10 + 0.05 - (0.10)(0.05)$ or 14.5 percent of the time (53 days per year).

During this time, a ground generated "pseudo earth pulse" operates PACE through the command system. The ground station maintains an ephemeris of inhibit times for the sun and the moon, and switches to the pseudo earth pulse mode during those periods. Attitude information is unavailable during inhibit periods. The ground reference is the solar pulse.

It is of interest to estimate the beam jitter during the pseudo earth pulse mode. Conservatively assume a 30-db pulse signal-to-noise ratio received at the command receiver in a 300-cps bandwidth. For a pulse of approximately 6.6 milliseconds wide, let the slope of the leading edge be $31.6/3.3$ or 9.6 volts per millisecond as a point where the threshold device is triggered. The one sigma time deviation of the triggering time is $1/9.6$ or 0.104 millisecond. Assuming once again a 1-cps effective bandwidth for the PACE loop, the deviation is $0.104/\sqrt{300}$ or 0.006 millisecond which is 0.0036 degree as a one-sigma beam deviation. This compares with 0.01635 degree for direct operation from the earth sensor.

Alternative System — Strip Detector. The system outlined above provides attitude and antenna beam reference information for an average of 85.5 percent of the time. During the remainder of the time, the antenna beam is dependent on ground control. The alternative system described here reduces the percentage of the time the an-

4. Launch and Orbit Considerations

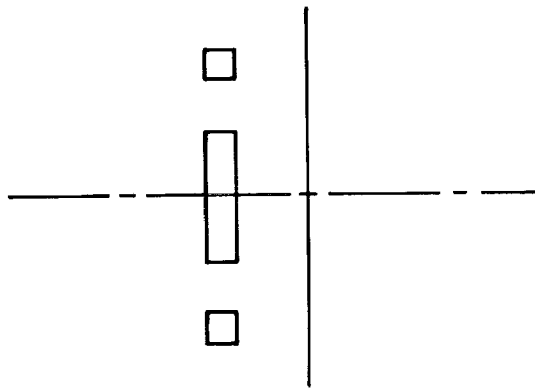


FIGURE 4-19. OUTLINE OF STRIP AND SQUARE DETECTOR ELEMENTS

tenna beam control is dependent on ground command. Assume we have a rectangular field of view about 1.3 degrees wide, and 5 degrees high, centered about the satellite equator, and coplanar with the spin axis (Figure 4-19). This calls for a thermistor detector element about 0.1×0.4 mm.

The lunar image will produce only one quarter as much signal as previously because of the larger field of view, and so will not trip the threshold device. When the declination of the sun is between ± 2.75 degrees, it will be in the field of view. Using Equation 4-56 it is found that the

Sun enters field of view 14 March

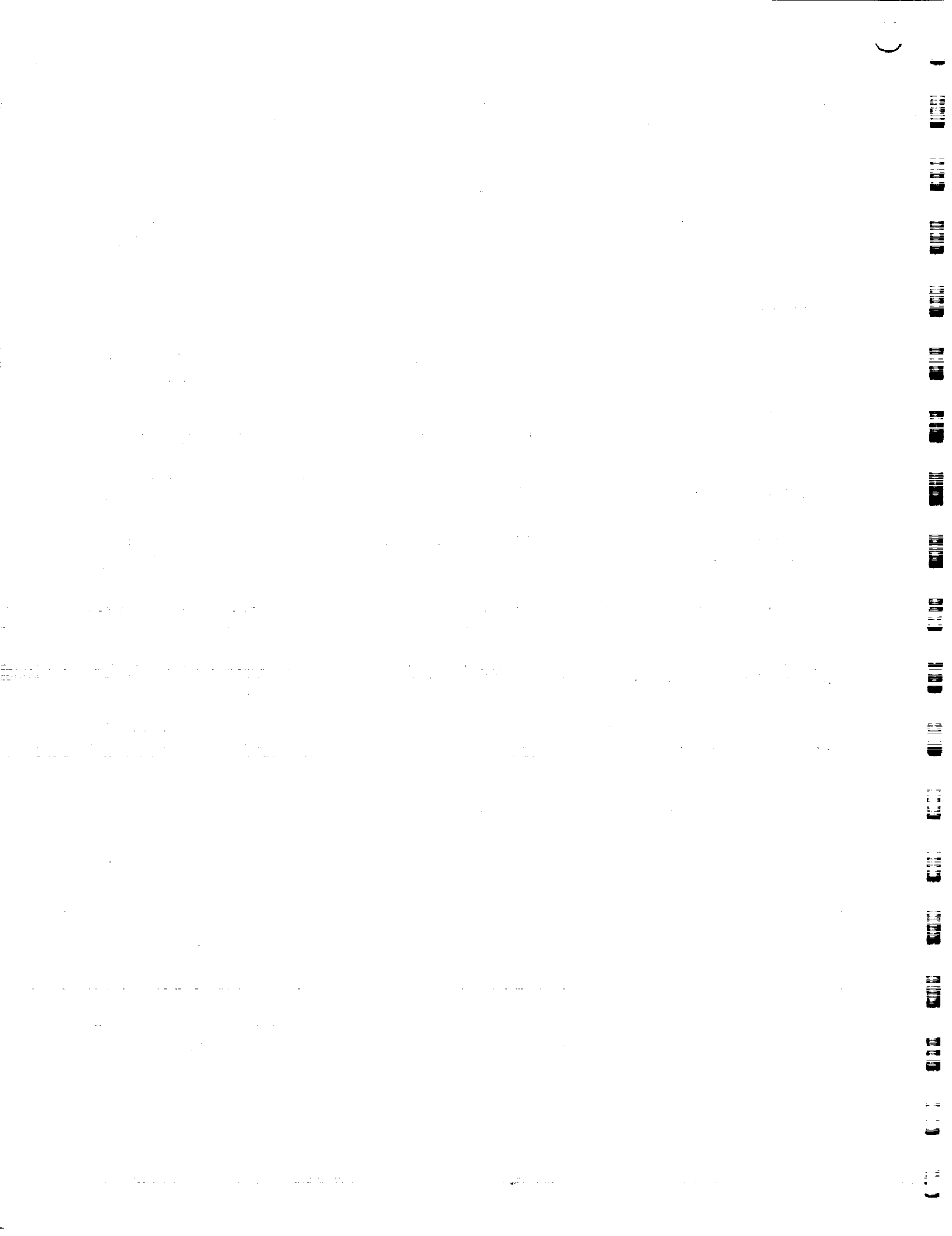
Sun leaves field of view 28 March

Sun enters field of view 14 September

Sun leaves field of view 28 September

or for about 28 days per year. Allowing another 22 days per year as safety margin, the sun will be in the field of view 40 days per year.

If the plane of acceptance of one of the sun sensors precedes the optic axis of the earth sensor by a small angle, its output may indicate the presence of the sun just prior to the solar image entering the earth sensor field of view. This indication may be used to inhibit the earth sensor pulse. Let this inhibit period be eight detector time constants, 20 milliseconds or $1/30$ of a satellite revolution. If the earth follows the sun by less than $1/30$ revolution, its signal will be inhibited. The total time therefore that the earth signal is unavailable is $40/30$ or 1.33 days per year. It is necessary to provide pseudo earth pulse signals from midnight to 12:48 a.m., local time at the subsatellite point, for 40 days per year.



5. SPACECRAFT SYSTEM DESIGN

SPACECRAFT BLOCK DIAGRAM

Figure 5-1 shows the latest configuration and interconnection of the various spacecraft subsystems. The diagram includes all nonquadrant equipment plus one set of quadrant equipment. All interconnections are indicated for a spacecraft containing four identical quadrants of communications and telemetry and command equipment.

COMMUNICATION TRANSPONDERS

The transponder accepts the incoming RF signal at approximately 6 gc and processes it in preparation for radiation by the transmitter at approximately 4 gc. Two modes of operation, either of which may be chosen by ground command, are available: a multiple access mode and a frequency translation mode. A block diagram including frequencies and power levels of the complete transponder is shown in Figure 5-2.

Although many of the building blocks are the same in the two modes, the operation of each is quite different. For either mode, a low frequency (UHF) oscillator is used to generate power required for frequency translation or other signal operations. In both modes the oscillator signal is multiplied up in frequency to provide the input mixer local oscillator signal. In both modes, intermediate frequency amplifiers are employed to increase the signal power after mixing, and in both modes regulators are provided to supply a constant -24 volt input. The rest of the blocks are unique to one mode because of the basic difference in operation and purpose of each mode.

The multiple access mode is designed to permit the interconnection of a large number of ground stations in a high channel capacity frequency division multiplex system. In this mode, the transponder serves as a telephone relay station, and the CCIR standards of intermodulation distortion plus maintaining each signal in an assigned channel are of paramount importance. An additional design goal is to share the spacecraft transmitted signal efficiently independent of the number of channels in use. These goals are met by utilizing a single-sideband signal from the ground transmitter, employing a very stable oscillator in the spacecraft, and converting to a phase modulated signal for transmission from the spacecraft.

The received signals, after frequency conversion by the input mixer, are amplified in a 5.3 mc bandwidth amplifier centered at approximately 35 mc. The amplitude of the composite signal fluctuates in the same manner as a band limited noise; hence this amplifier must be linear to prevent excessive intermodulation. The output power of this amplifier averages less than a microwatt, however, and the required linearity is achievable. It would be difficult to produce intermodulation levels of importance at such low signal levels even if the attempt were made to do so.

The amplified received signals and the transmitter master oscillator reference are the inputs to the IF phase modulator which performs the modulation conversion. The operation of this modulator is described with reference to Figure 5-3. The composite signal and reference carrier are introduced into a hybrid transformer ar-

ranged so that the signal produces in-phase vectors and the carrier out-of-phase vectors at output terminals A and B. The limiters restrict the resultant vectors to a constant amplitude, so that its modulation is low index phase modulation. This action of the limiters creates the second sideband characteristics of a phase-modulated carrier. The limited resultant vectors are added in the output summing network in the ratio $(\mu + 1)/(\mu - 1)$, causing the resultant vector's phase shift to be increased by the factor μ . This arrangement permits the ratio of carrier to signal to be high enough prior to limiting to keep intermodulation products low, while permitting the modulation index to be subsequently adjusted to the level corresponding to the communication system requirements. The output of the phase modulator is multiplied up in frequency and increased in power level to provide the drive for the communication transmitter.

The frequency translation mode is designed primarily for television or other wideband usage in which one ground transmitter utilizes the complete channel. In addition this mode provides a beacon signal for ground station tracking. The frequency translation mode has no inherent beacon as does the multiple access mode in which the carrier is always present. The primary design criteria for the frequency translation mode includes meeting the envelope delay distortion specifications for color TV and providing sufficient signal to the communication transmitter to allow it to operate saturated for a wide range of input signal levels.

In operation, the received signal is mixed and converted to an intermediate frequency of approximately 60 mc, amplified and limited and then mixed again for conversion to the transmit frequency. The limiter serves the additional function of introducing the beacon signal and con-

trolling its level. Wide-band circuitry is utilized throughout the signal chain so that the resulting usable bandwidth is 25 mc—sufficient to meet all CCIR television standards.

General Physical Arrangement of Transponder. The general arrangement of the transponder in the spacecraft, can be described with reference to the division of circuitry shown in Figure 5-2. The receiver mixers, bandpass filters, preamplifiers, and RF switches are located in the antenna electronics compartment just aft of the apogee motor. The antenna electronics package is 6.6 by 5.6 by 3.8 inches in size and weighs approximately 2.25 pounds.

Each of the dual mode transponders mounts in an assembly shown partially assembled in Figures 5-4, 5-5, and 5-6. This assembly measures 5.00 by 11.25 by 18.00 inches. The various transponder units are mounted on two aluminum bulkheads supported at the corners by aluminum angles. The entire package is fully enclosed in 0.016 inch aluminum sheet, which serves as RF shielding and provides structural support. Weight of the entire package is 15.50 pounds.

Minor Control Item Description

Circuits Common to Multiple Access and Frequency Translation Mode.

- **Input Mixer.** The input mixer is shown in Figure 5-7. Stripline circuitry is employed for all signal paths. A 3-db etched circuit ring hybrid introduces both the received signal and local oscillator signal. 1N832AM glass diodes develop the intermediate frequency difference signal which is coupled out through RF chokes to an OSM output connector. The mixer introduces approximately 6.5 db loss and operates with a noise figure of 9 db.

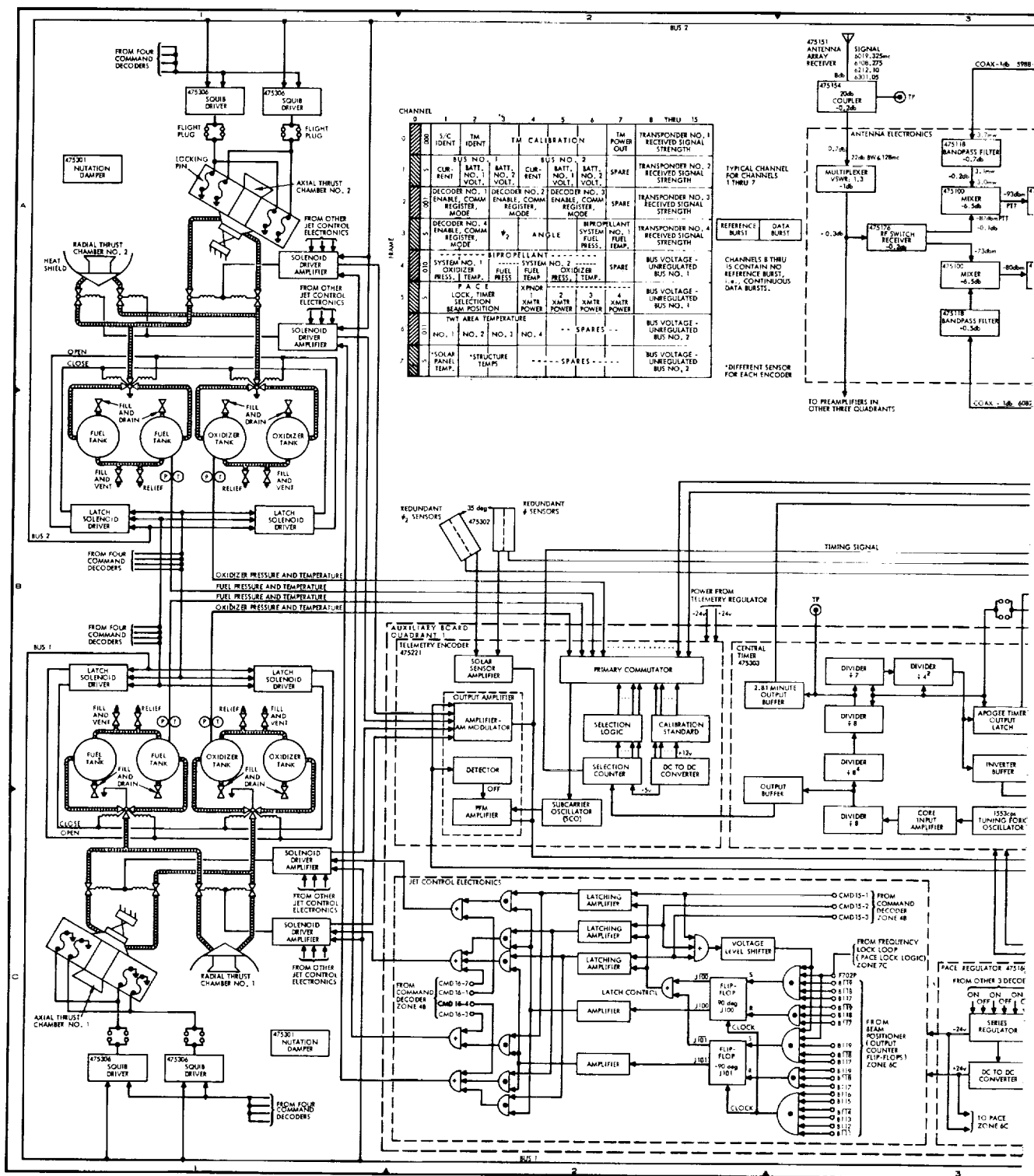
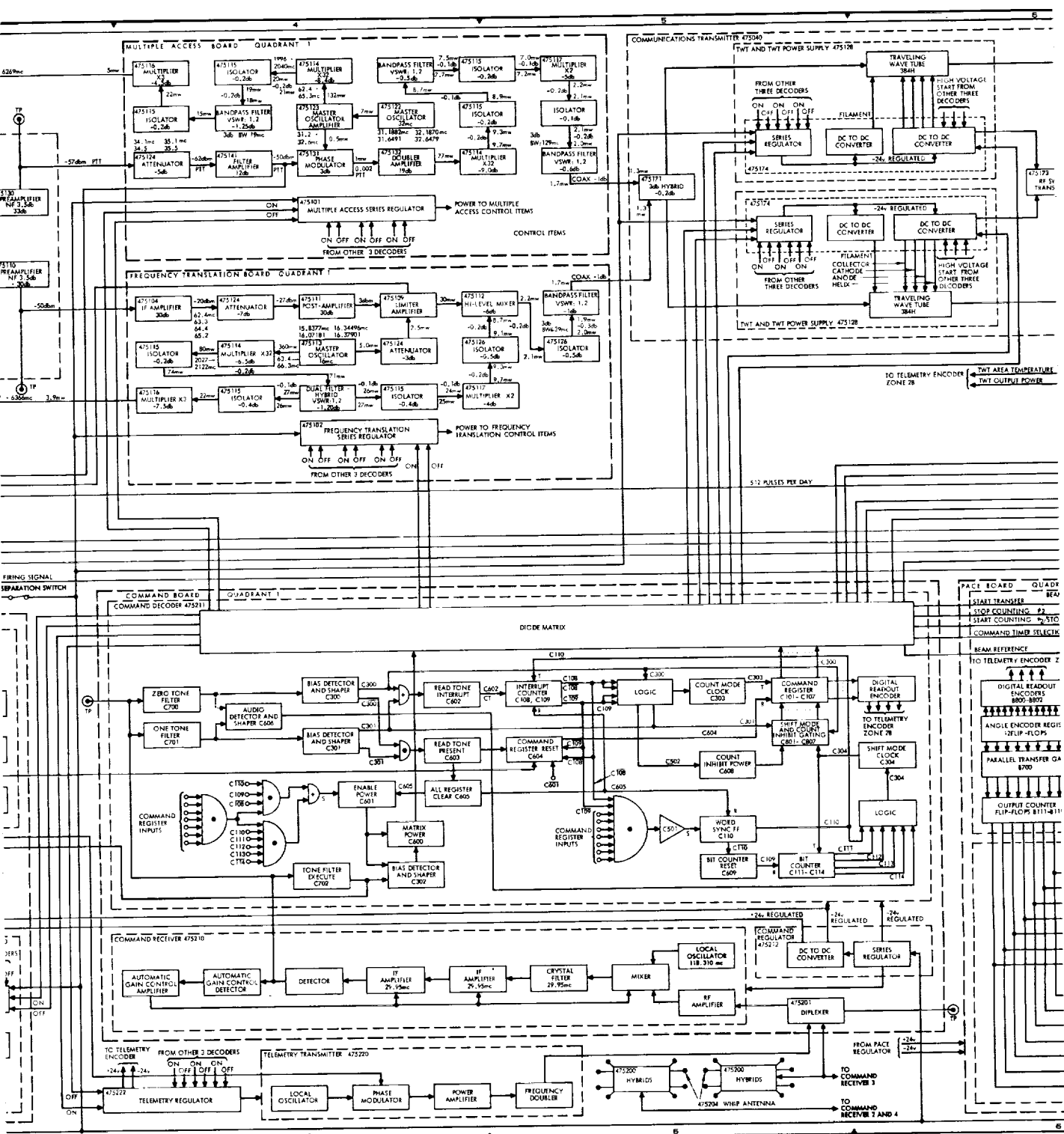


FIGURE 5-1. SPACECRAFT BLOCK DIAGRAM



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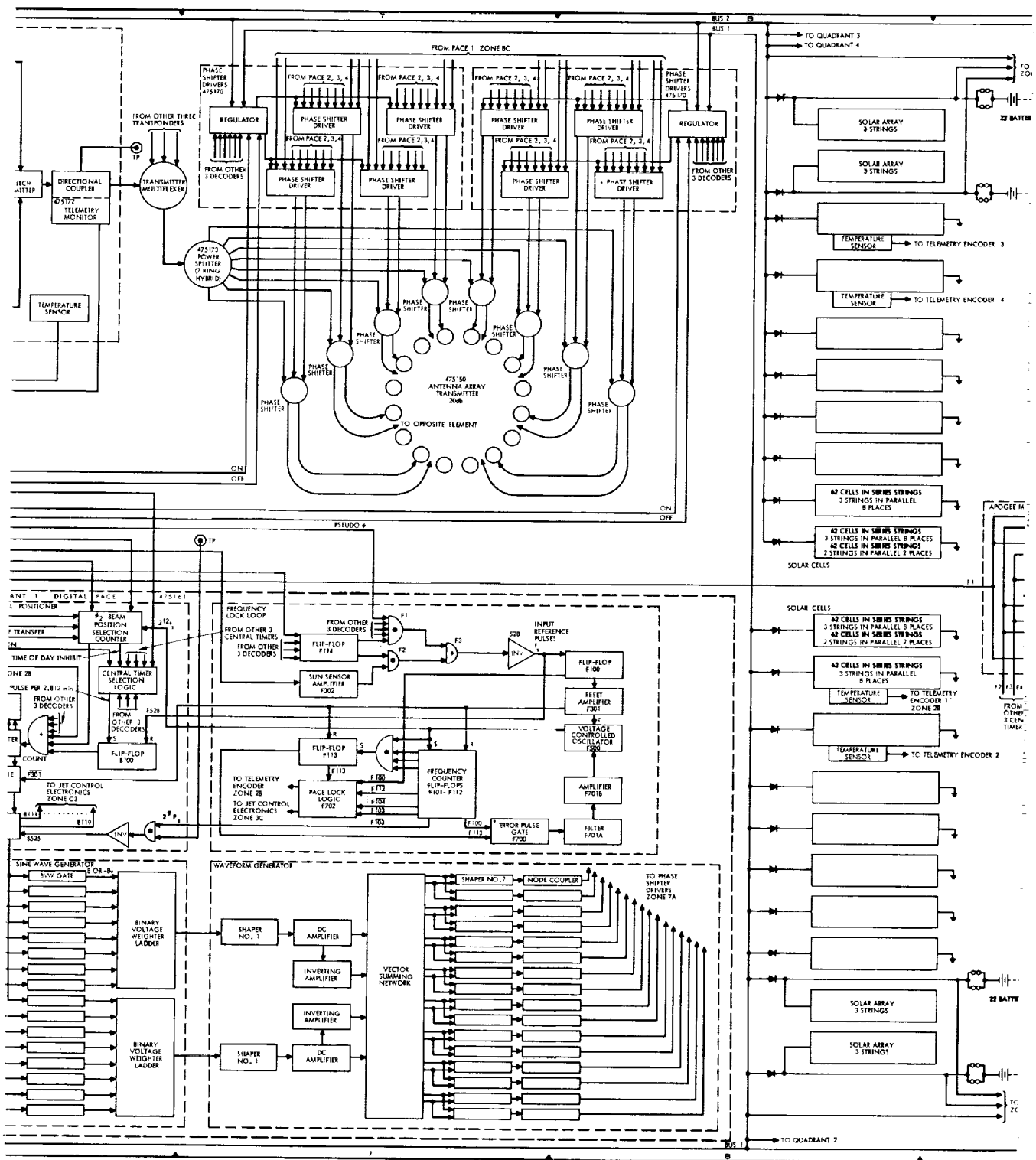
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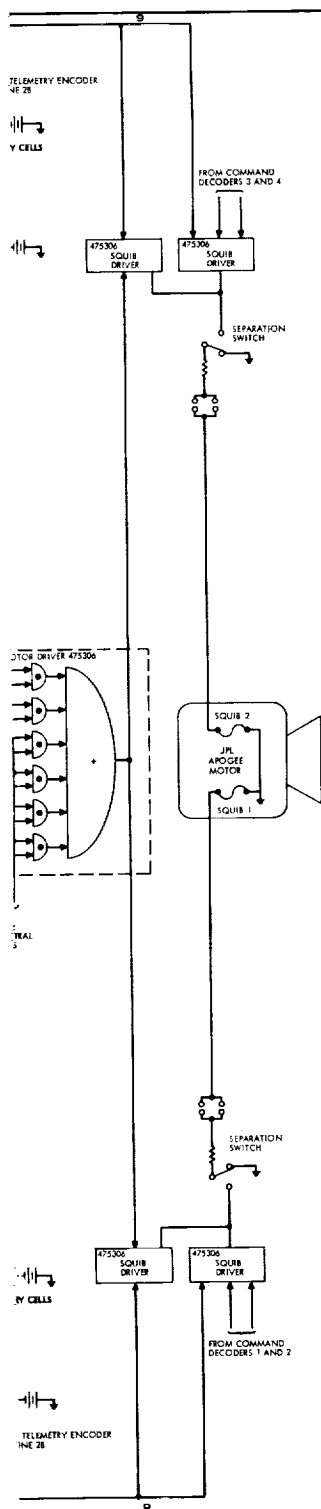
9

10

11



3



COMMAND ADDRESS WORD

SPACECRAFT NO./ QUADRANT NO.	MODIFIED OCTAL	6	5	4	3	2	1 BIT NO.
T2-1	00-1	0	0	0	0	0	1
2	00-2	0	0	0	0	1	0
3	01-3	0	0	0	1	1	1
4	01-4	0	0	0	1	0	0
Y1-1	02-1	0	0	1	1	0	1
2	02-2	0	0	1	1	1	0
3	02-3	0	0	1	0	1	1
4	02-4	0	0	1	0	0	0
F1-1	06-1	0	1	1	0	0	1
2	06-2	0	1	1	0	1	0
3	06-3	0	1	0	0	1	1
4	06-4	0	1	0	0	0	0
F2-1	4-1	1	1	0	0	0	1
2	4-2	1	1	0	0	1	0
3	4-3	1	0	0	0	1	1
4	4-4	1	0	0	0	0	0
F3-1	11-1	1	0	0	1	0	1
2	11-2	1	0	0	1	0	0
3	11-3	1	0	1	1	1	1
4	11-4	1	0	1	1	0	0
F4-1	12-1	1	0	1	0	0	1
2	12-2	1	0	1	0	1	0
3	16-3	1	1	1	0	1	1
4	16-4	1	1	1	0	0	0

COMMAND NUMBER

COMMAND FUNCTION	MODIFIED OCTAL	6	5	4	3	2	1
SPARES	00-1	0	0	0	0	0	1
	00-2	0	0	0	0	0	0
	00-3	0	0	0	0	1	1
TRANSPONDER	01-1	0	0	0	1	0	1
FREQUENCY TRANSLATION MODE ON, MULTIPLE ACCESS MODE OFF	01-2	0	0	0	1	1	0
	01-3	0	0	0	1	1	1
	01-4	0	0	0	1	0	0
MULTIPLE ACCESS MODE ON, FREQUENCY TRANSLATION MODE OFF	02-1	0	0	1	0	0	1
	02-2	0	0	1	0	1	0
	02-3	0	0	1	0	1	1
	02-4	0	0	1	0	0	0
SPARES	03-1	0	0	1	1	0	1
	03-2	0	0	1	1	1	0
	03-3	0	0	1	1	1	1
	03-4	0	0	1	1	0	0
TRANSPONDER OFF	04-1	0	1	0	0	0	1
	04-2	0	1	0	0	1	0
	04-3	0	1	0	0	1	1
	04-4	0	1	0	0	0	0
TRANSMITTER	05-1	0	1	0	1	0	1
TWT 1 FILAMENT ON, TWT 2 FILAMENT OFF	05-2	0	1	0	1	1	0
	05-3	0	1	0	1	1	1
	05-4	0	1	0	1	0	0
TWT 2 FILAMENT ON, TWT 1 FILAMENT OFF	06-1	0	1	1	0	0	1
	06-2	0	1	1	0	1	0
	06-3	0	1	1	0	1	1
	06-4	0	1	1	0	0	0
TWT HIGH VOLTAGE ON	07-1	0	1	1	1	0	1
	07-2	0	1	1	1	1	0
	07-3	0	1	1	1	1	1
	07-4	0	1	1	1	0	0
PAGE	10-1	1	0	0	0	0	1
PAGE ON FOR QUADRANT SPECIFIED, ALL OTHER PAGE OFF	10-2	1	0	0	0	1	0
	10-3	1	0	0	0	1	1
	10-4	1	0	0	0	0	0
ALL PAGE OFF	11-1	1	0	0	1	0	1
ACCEPT ANTENNA POSITION	11-2	1	0	0	1	1	0
PHASE SHIFTER DRIVER REGULATOR ON	11-3	1	0	0	1	1	1
PHASE SHIFTER DRIVER REGULATOR OFF	11-4	1	0	0	1	0	0
ECLIPSE MODE OPERATION ON	12-1	1	0	1	0	0	1
ECLIPSE MODE OPERATION OFF	12-2	1	0	1	0	1	0
TIMER SELECT	12-3	1	0	1	0	1	1
#2 COUNTER ON	12-4	1	0	1	0	0	0
TELEMETRY	13-1	1	0	1	1	0	1
TELEMETRY ON	13-2	1	0	1	1	1	0
	13-3	1	0	1	1	1	1
	13-4	1	0	1	1	0	0
TELEMETRY OFF	14-1	1	1	0	0	0	1
	14-2	1	1	0	0	1	0
	14-3	1	1	0	0	1	1
	14-4	1	1	0	0	0	0
BURNPELLANT SYSTEM	15-1	1	1	0	1	0	1
FIRE JETS 1 AND 2, - 90° AXIAL	15-2	1	1	0	1	1	0
FIRE JETS 1 AND 2, - 90° RADIAL	15-3	1	1	0	1	1	1
FIRE JETS 1 AND 2, - 90° RADIAL	15-4	1	1	0	1	0	0
SPARE	16-1	1	1	1	0	0	1
FIRE AXIAL JET, SYSTEM 1 (BACKUP MODE)	16-2	1	1	1	0	1	0
FIRE RADIAL JET, SYSTEM 1 (BACKUP MODE)	16-3	1	1	1	0	1	1
FIRE AXIAL JET, SYSTEM 2 (BACKUP MODE)	16-4	1	1	1	0	0	0
FIRE RADIAL JET, SYSTEM 2 (BACKUP MODE)	17-1	1	1	1	1	0	1
FIRING CIRCUITS	17-2	1	1	1	1	1	0
RELEASE CONTROL SYSTEM LOCKING PINS	17-3	1	1	1	1	1	1
FIRE AMOGEE ENGINE	17-4	1	1	1	1	0	0
SPARE							

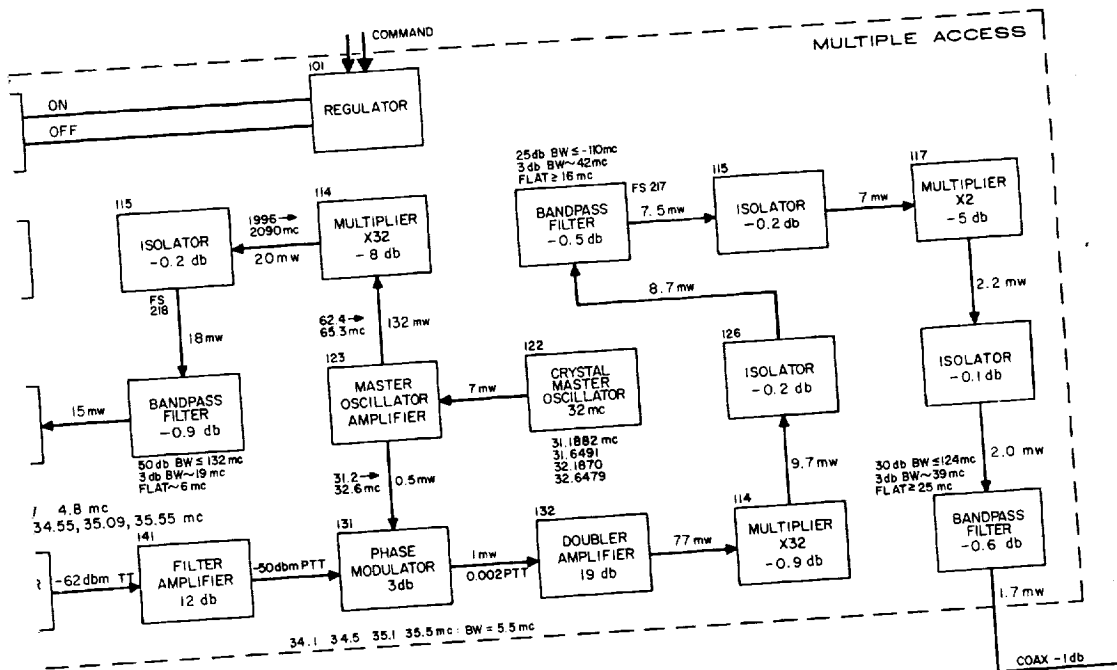
* BURCH PULSE GATE TRUTH TABLE

FLIP-FLOP 0	FLIP-FLOP 10	GATE OUTPUT
0	0	0
0	1	0
1	0	0
1	1	1

1 - FREQUENCY
TP - TEST ACCESS POINT

DIAGRAM

2

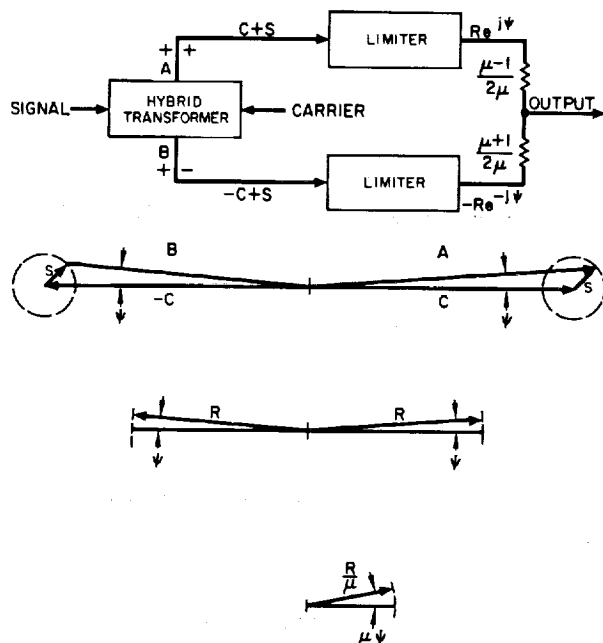


FIGURE 5-3. IF PHASE MODULATOR OPERATION

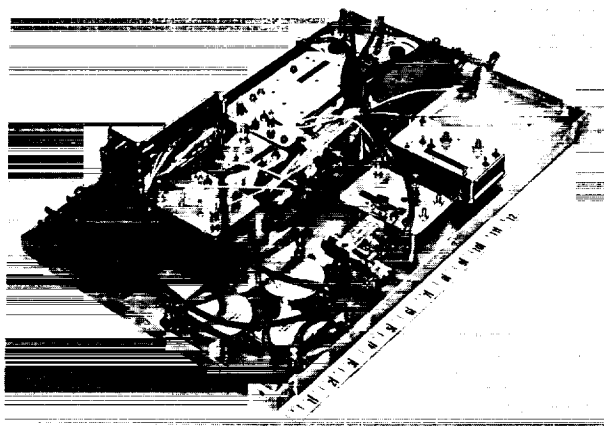
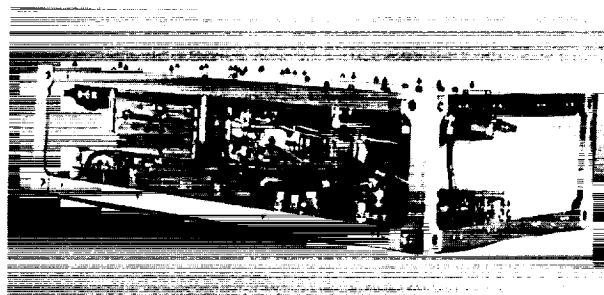
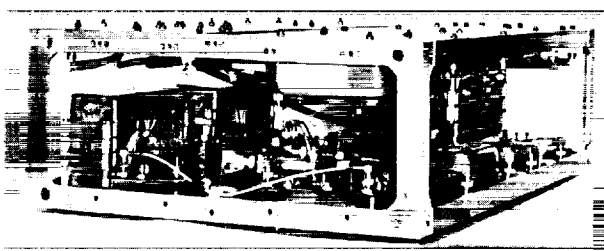


FIGURE 5-5. FREQUENCY TRANSLATION MODE — ASSEMBLED

• **X32 Multiplier.** The X32 multiplier used in both frequency translation and multiple access modes is shown in Figure 5-8. Multiplication from 63.4 mc to 2027 mc is performed in five varactor diode doublers. Each doubler is a push-push type, single ended at the input fre-



a) Front View



b) Side View

FIGURE 5-4. ASSEMBLED TRANSPONDER

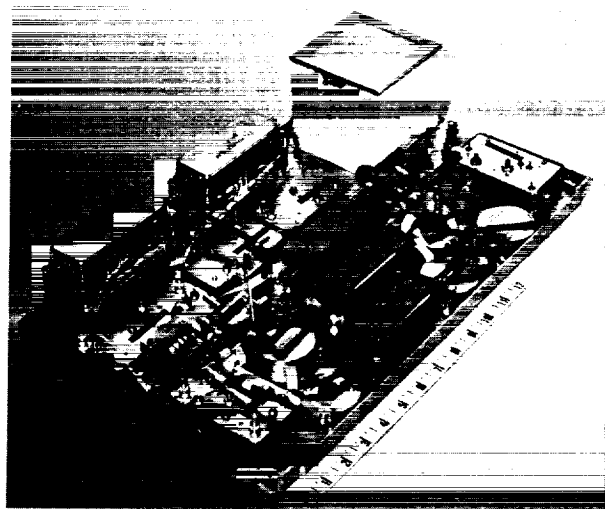


FIGURE 5-6. MULTIPLE ACCESS MODE — ASSEMBLED

quency and push-pull at the doubled output frequency. A pi-matching network couples directly to the driving transistor through a short coaxial section. Fixed bias is provided for the diodes. This bias unit is included in the X32 package and

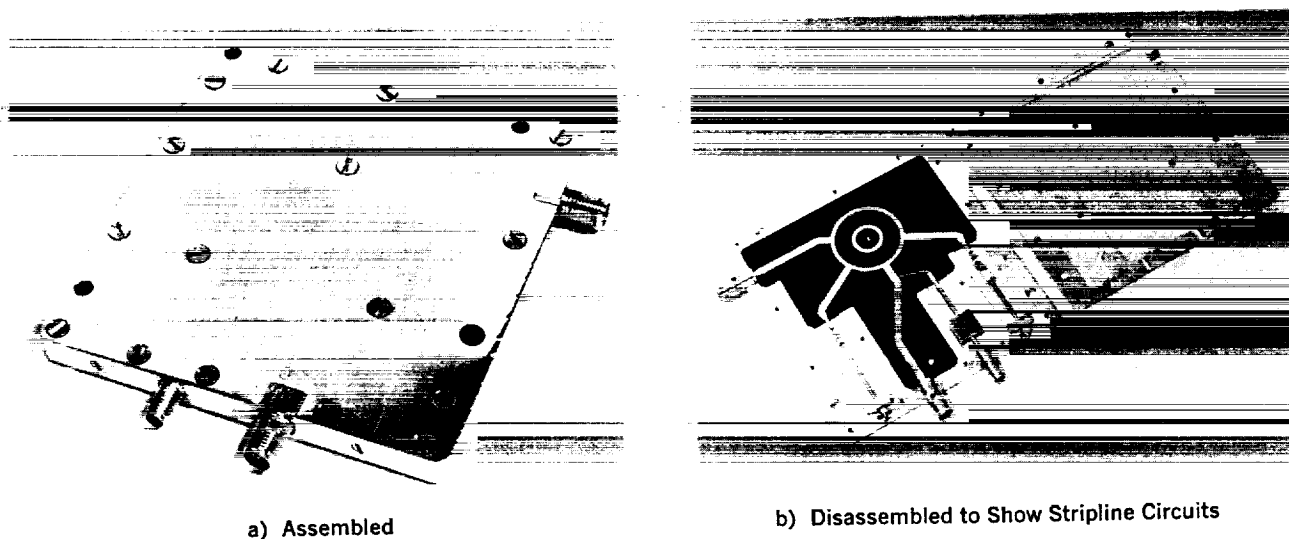


FIGURE 5-7. INPUT MIXER

consists of a dc-dc converter which provides ± 45 volts bias voltage for the diodes.

- **X3 Multiplier.** The X3 multiplier is shown in Figure 5-9. The multiplier is constructed in rigid large X-band waveguide. The input resonator employs a coaxial section for tuning. A varactor diode is utilized to generate the output signal which is coupled out from a waveguide cavity. Fixed bias, obtained from the X32 bias supply, is used to set the varactor operating point.

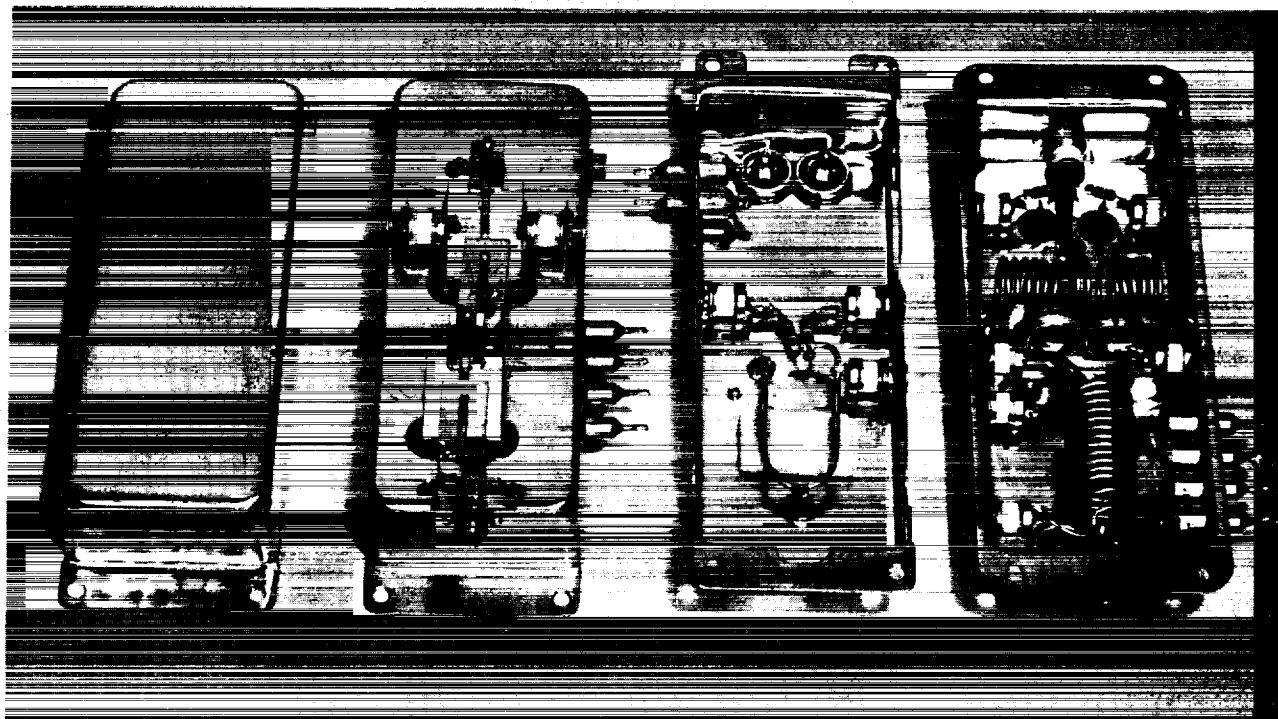
- **X2 Multiplier.** The X2 multiplier is shown in Figure 5-10. In the multiplier, adjustable capacitive input coupling is provided to a stripline resonator with varactor diodes at one end of the resonator. The output resonator is a ridged waveguide cavity. Two varactor diodes are series-connected across the output as a push-push doubler. The diodes are driven — one to cathode, one to anode — effectively in parallel with respect to the input signal and in series with respect to the output frequency. Fixed bias is provided to the varactor diodes from the X32 bias supply — RF chokes are placed on the bias

input lines to prevent interaction between the X2 and X32 units.

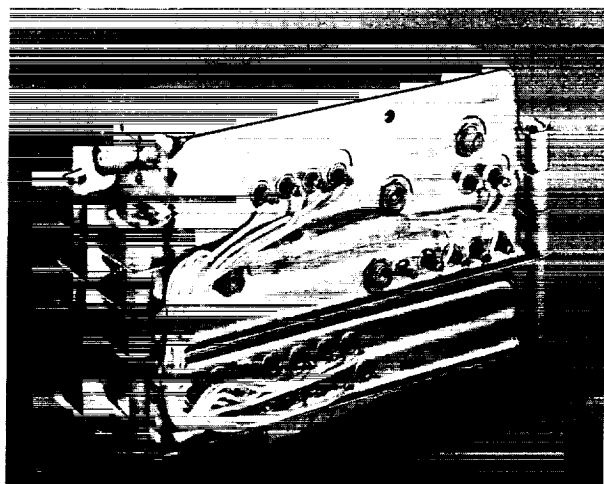
- **6 gc Bandpass Filter.** The 6 gc bandpass filters (Figure 5-11) are used in the receiver local oscillator chains of both the frequency translation and the multiple access transponders. The filter is a dual cavity bandpass filter designed to pass the local oscillator frequency and reject spurious signals outside the passband. It is constructed of two iris coupled cylindrical RF cavities with probe input and output coupling. Input and output impedance are 50 ohms. Bandwidth at 3 db down is approximately 18 mc.

- **Isolator.** The 2 kmc and 4 kmc isolators used in both transponder modes are shown in Figures 5-12 and 5-13. The isolators have stripline Y-junctions with a magnetized ferrite disc at the center to provide circulation action. Insertion loss in the forward direction is approximately 0.2 db and isolation at the center frequency exceeds 30 db.

- **IF Attenuator.** The IF attenuator is connected between the IF preamplifiers and the IF postamplifiers, and between the beacon fre-



a) Disassembled



b) Assembled

FIGURE 5-8. X32 MULTIPLIER

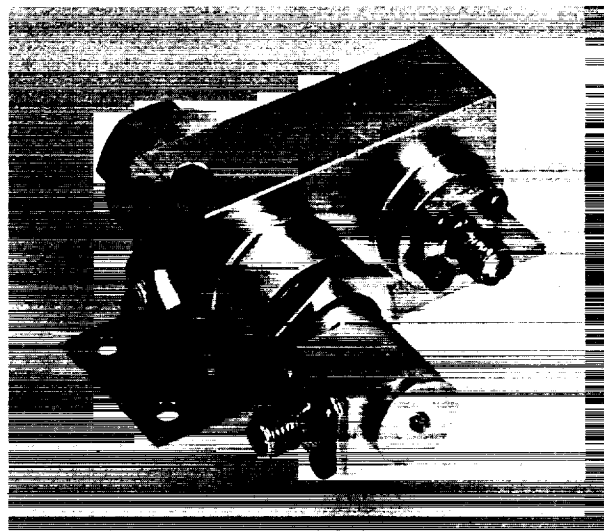


FIGURE 5-9. X3 MULTIPLIER

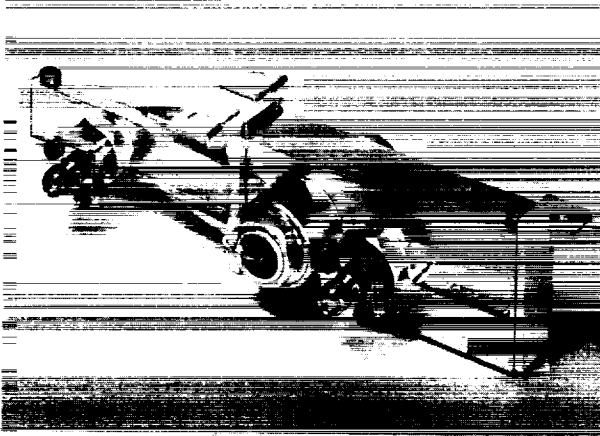


FIGURE 5-10. X2 MULTIPLIER

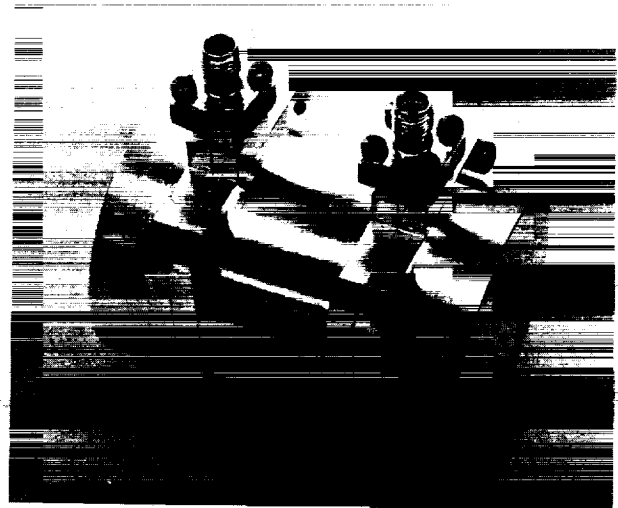


FIGURE 5-11. 6 GC BANDPASS FILTER

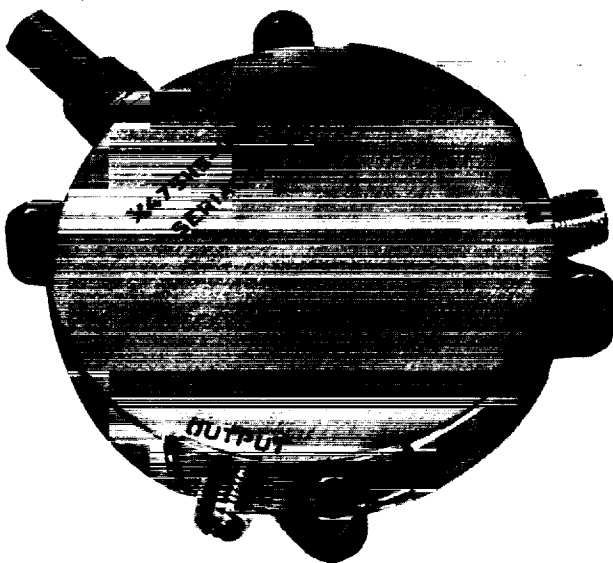


FIGURE 5-12. 2 KMC ISOLATOR

quency generator and the limiter amplifier. Its purpose is to provide a fixed squelch and amplitude threshold relative to the noise level by adjusting the overall signal amplification. Several different attenuations are available for use as system measurements dictate. The input and output impedance is 50 ohms.

- 4 gc Bandpass Filters. A bandpass filter is utilized at the output of each mode to reject spurious signals before they can reach the trans-

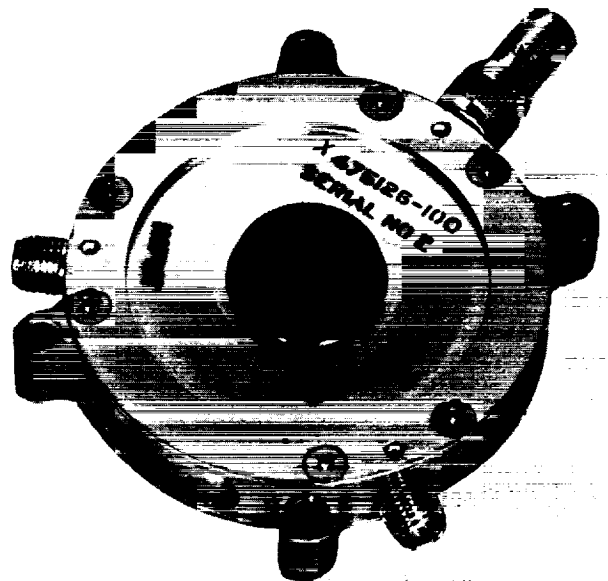


FIGURE 5-13. 4 KMC ISOLATOR

mitter. The filter has the following specifications:

3 db bandwidth	39 mc
30 db bandwidth	≤ 124 mc
Flat bandwidth	≥ 25 mc
Maximum loss	0.6 db

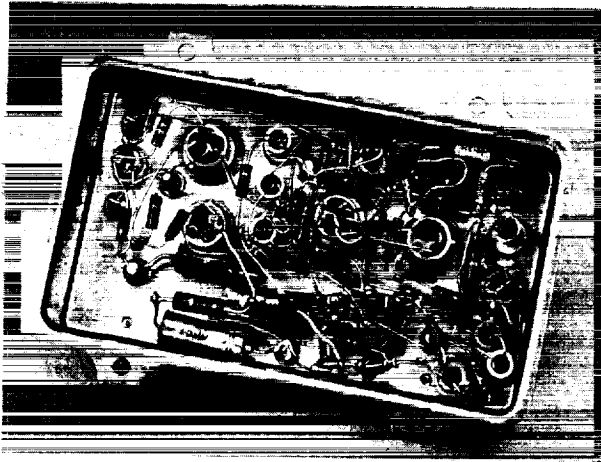


FIGURE 5-14. REGULATOR — FREQUENCY TRANSLATION TRANSPONDER

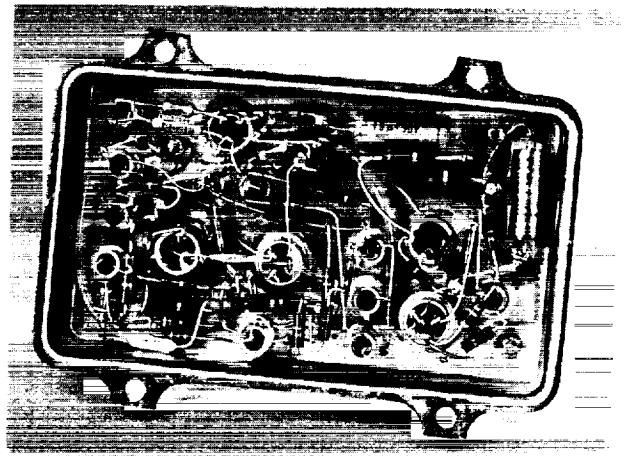


FIGURE 5-15. REGULATOR — MULTIPLE ACCESS TRANSPONDER

• **Regulators.** The regulators used for frequency translation (Figure 5-14) and multiple access (Figure 5-15) are similar series regulators having nominal output of -24 volts at 126 milliamperes (3.02 watts) for the frequency translation mode and 110 milliamperes (2.64 watts) for the multiple access mode. The regulators require an additional 4 milliamperes for operation. This current is independent of unregulated bus variations of -31 ± 5 volts. The power drawn from unregulated bus varies directly with input voltage as follows:

Frequency translation

$$P_{\text{MAX}} 36 \text{ volts} \times 114 \text{ milliamperes} = 2.47 \text{ watts}$$

$$P_{\text{MIN}} 26 \times 114 \text{ milliamperes} = 4.11 \text{ watts}$$

Multiple Access

$$36 \text{ volts} \times 130 \text{ milliamperes} = 4.68 \text{ watts}$$

$$26 \text{ volts} \times 130 \text{ milliamperes} = 3.38 \text{ watts}$$

For load and line variations from 30° to 90° F, the output voltage regulation should be ± 3 percent or better for the frequency translation transponder, and ± 2 percent for the multiple access transponder. (For details, see page 5-16.)

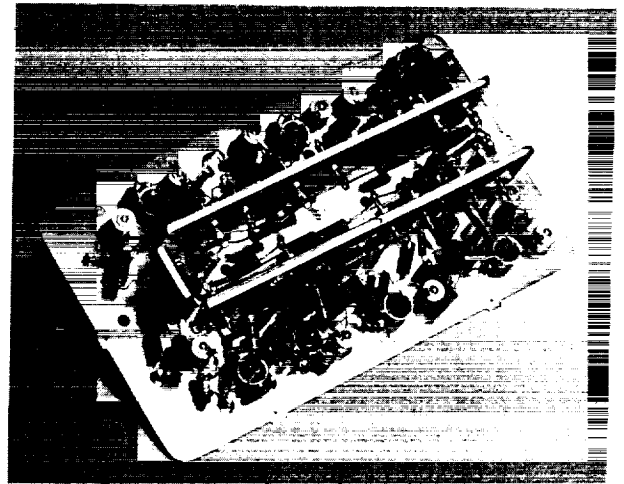


FIGURE 5-16. CRYSTAL MASTER OSCILLATOR

Frequency Translation Mode Circuits

• **Crystal Master Oscillator.** The master oscillator (Figure 5-16) is an eight-transistor stable RF source with outputs of approximately 65 and 48 mc. A crystal controlled transistor oscillator provides a basic frequency of approximately 16 mc. The basic frequency is multiplied by four and amplified to approximately 350 milliamperes to provide the 65 mc output to the X32 multiplier. The crystal master oscillator controls

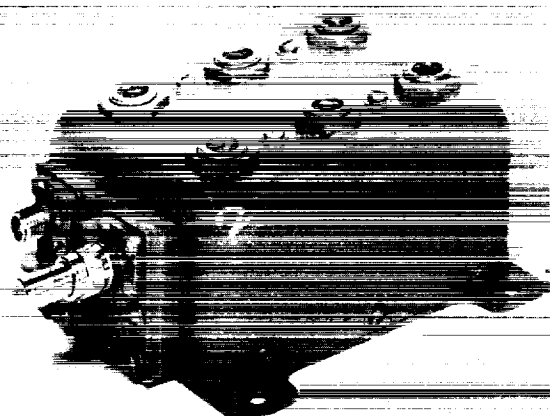


FIGURE 5-17. DUAL FILTER HYBRID

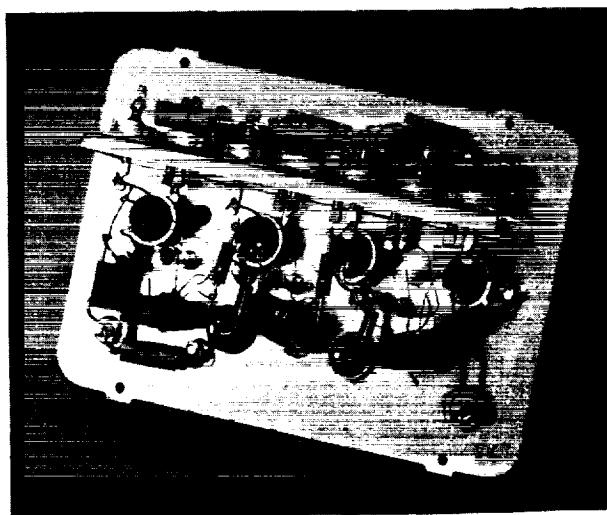


FIGURE 5-18. FREQUENCY TRANSLATION PREAMPLIFIER

both the local oscillator frequency and the output carrier frequency of the frequency translation transponder. The 48 mc output is generated by multiplying the basic frequency by three and amplifying the signal to 3.2 milliwatts. This signal is routed through an attenuator to the limiter amplifier to become ultimately the transponder beacon frequency. The power supply requirements of this unit are -24 volts dc ± 1 percent and -4 volts dc ± 1 percent.

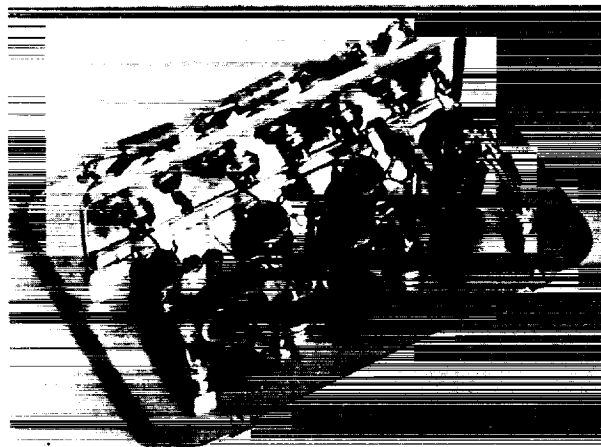


FIGURE 5-19. FREQUENCY TRANSLATION INTERMEDIATE AMPLIFIER

- **Dual Filter Hybrid.** The dual filter hybrid (Figure 5-17) divides the 2000 mc signal from the oscillator-multiplier chain into two equal parts and filters each half separately to remove undesired harmonics and provide separation between receiver and transmitter local oscillator chains. A quarter-wave stripline hybrid is used to split the power; two identical three-resonator bandpass filters provide the necessary filtering.

The specifications for this unit are as follows:

Center frequency (f_o)	2112 mc
Insertion loss (f_o)	1.25 db maximum
VSWR (f_o)	1.20:1 maximum
Rejection ($f_o \pm 40$ mc)	40.0 db minimum
Hybrid output ratio	6.0 db ± 0.5 db
Hybrid directivity (f_o)	20.0 db minimum
Isolation between filters ($f_o \pm 40$ mc)	90.0 db minimum
Weight	8 ounces maximum

- **Intermediate Frequency Amplifiers.** The frequency translation preamplifier (located with the antenna electronics), intermediate amplifier, and postamplifier (Figures 5-18, 5-19, and 5-20) are transistorized amplifiers each of 30 db gain and 25 mc bandwidth. The preampli-

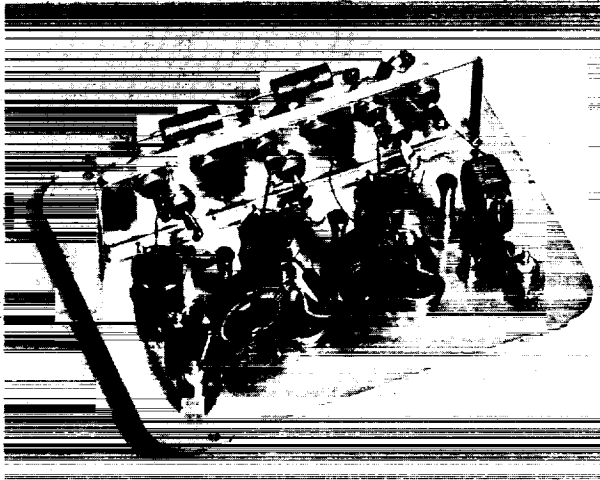


FIGURE 5-20. FREQUENCY TRANSLATION POSTAMPLIFIER

fier and intermediate amplifier have four transistor stages with emitter follower output stages. The postamplifier has three transistor stages with transformer output (50 ohms) to the limiter.

- **Limiter-Amplifier.** The limiter amplifier (Figure 5-21) includes a 2N1141 transistor emitter follower, a 2N1141 amplifier stage, a pair of IN3062 diodes for clipping, and another 2N1141 amplifier stage. The primary purpose of the limiter amplifier is to remove amplitude modulating noise on the communications signal.

The bandwidth of the limiter amplifier is about 25 mc. The input signal from the IF post-amplifier is amplified and amplitude limited. The limited signal is amplified in the third transistor stage, summed with a beacon input signal, and routed to the high level mixer. If the communications channel is unoccupied, the beacon signal provides about 25 percent of full output. If the communications channel is occupied by a limiting signal, the beacon signal is automatically reduced to about 5 percent of full power. The communications signal strength is measured by monitoring the average clamp-

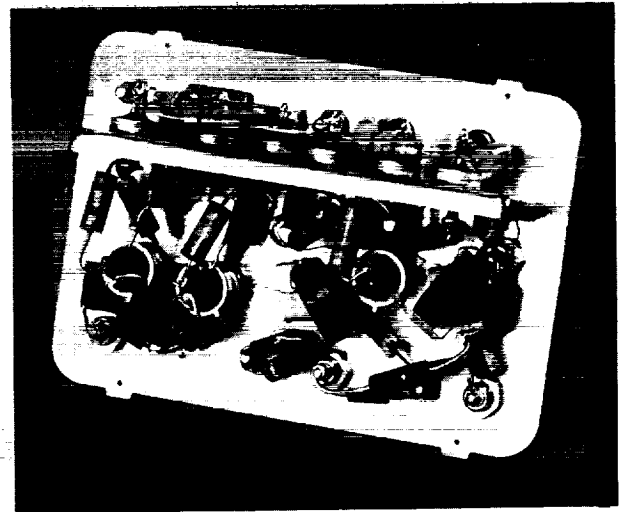


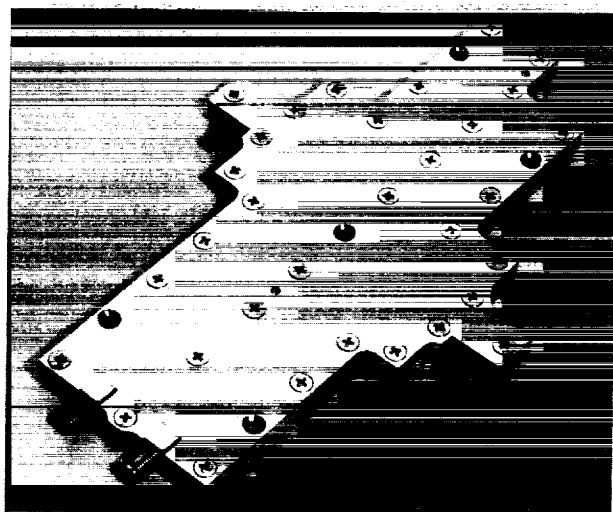
FIGURE 5-21. LIMITER AMPLIFIER

ing voltage to the limiter for analog transmission to the ground stations via the telemetry system. This voltage is relatively constant for signals above the limiting threshold.

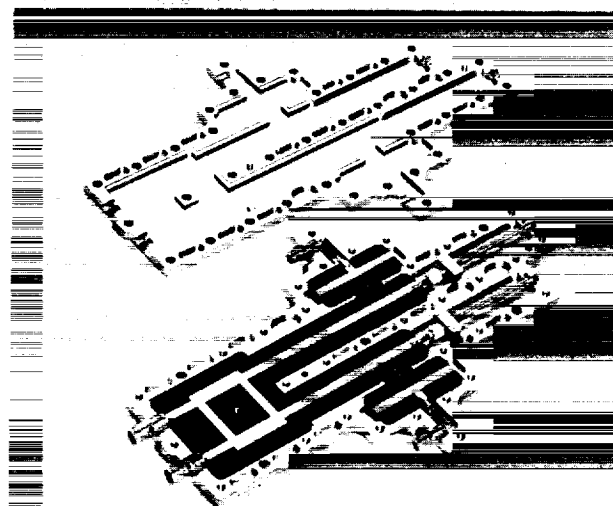
- **High Level Mixer.** The high level mixer (Figure 5-22) is a stripline balanced modulator employing a pair of varactor diodes as the non-linear heterodyning devices to mix the frequency modulated IF signal with the reference signal derived as the 256th harmonic of the master oscillator. The output consists of the two IF sidebands, each with modulation. In the transponder these sidebands are routed through a bandpass filter which rejects sidebands which are above the center frequency, the output of the filter becoming the transmitted output signal of the transponder.

Multiple Access Mode Circuits

- **34 mc Preamplifier.** The preamplifier (Figure 5-23), located in the antenna electronic package, is a three-stage 2N2415 transistorized amplifier of 36 db gain at 34 mc. Noise factor is approximately 3.5 db. Output is down less than 1 db for a bandwidth of 5 mc.



a) Assembled



b) Disassembled

FIGURE 5-22. HIGH-LEVEL MIXER

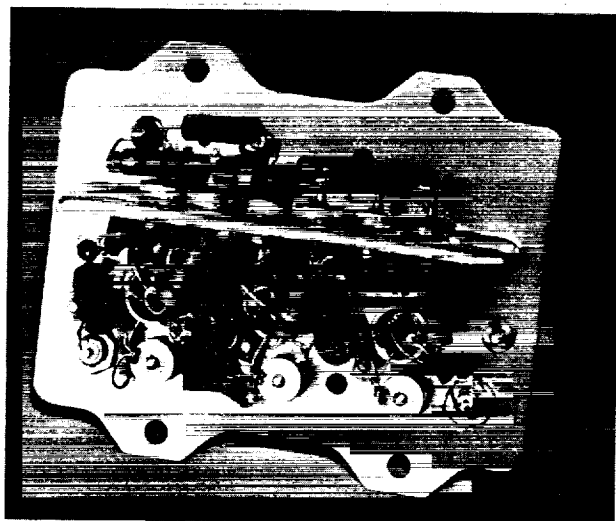


FIGURE 5-23. 34 MC PREAMPLIFIER

- **Filter Amplifier.** The filter amplifier, which operates at 34 mc, is designed to pass frequencies 250 kc above the carrier and upward and reject frequencies from 200 kc below the carrier and downward. The unit (Figure 5-24) includes three notch circuits and three transistor stages of amplification. Gain is 14 db.

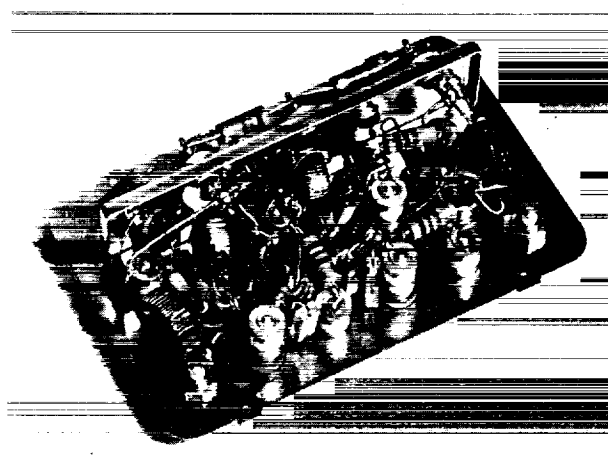


FIGURE 5-24. FILTER AMPLIFIER

- **Phase Modulator.** The phase modulator (Figure 5-25) is a four transistor stage device that converts the amplitude modulated IF signal to a phase modulated basic carrier signal that is subsequently multiplied by a factor of 128 and transmitted by the traveling-wave tube amplifier. The input stage has two 2N1141 transistors, both common base connected. Signals are applied to a hybrid transformer to provide in-phase output

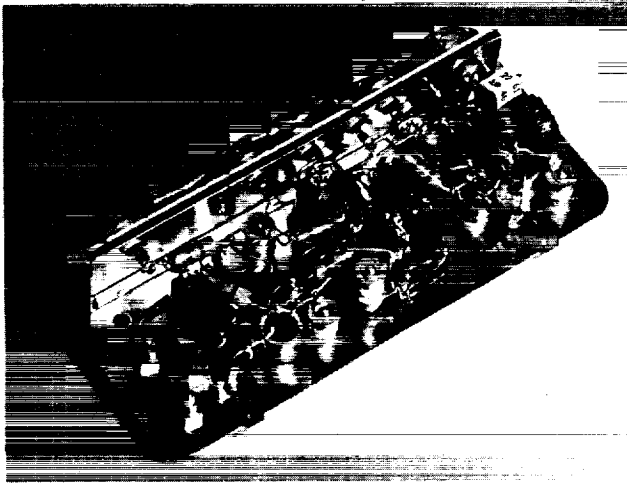


FIGURE 5-25. PHASE MODULATOR

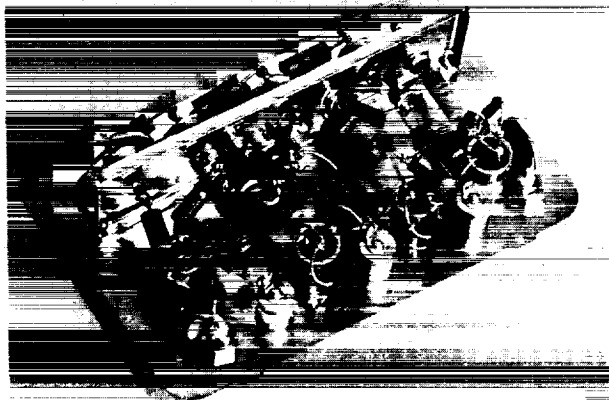


FIGURE 5-26. DOUBLER AMPLIFIER

signals from the IF input and 180 degrees out of phase output signals for the master oscillator input. The output of each transistor goes to a pair of IN3062 limiting diodes. Outputs of the two limiters are added in a resistive network and applied to the base of a 2N1408 grounded base output stage. The output level is 1 milliwatt.

- **Doubler Amplifier.** The doubler amplifier (Figure 5-26) includes a 2N1405 transistor X2 multiplier (32 mc to 64 mc) and three stages of amplification using two 2N1141 and one 2N

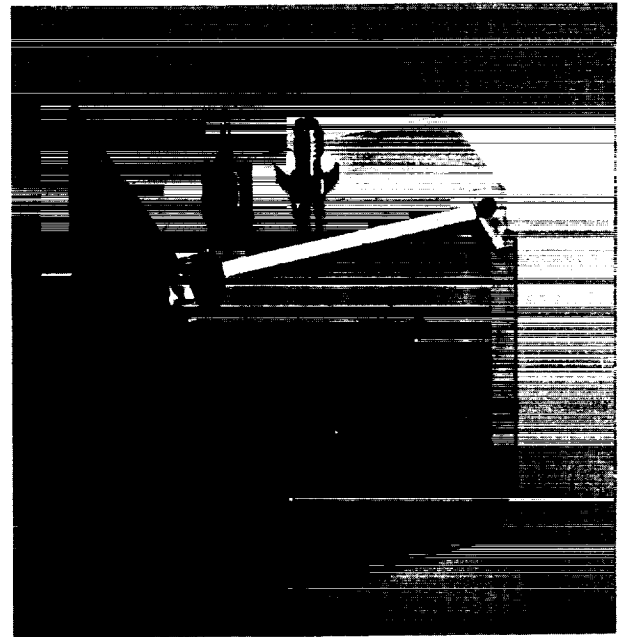


FIGURE 5-27. 32 MC MASTER OSCILLATOR

707A transistor. The signal input is approximately 32 mc phase-modulated up to 5 mc. Overall gain is 16 db.

- **32 mc Master Oscillator.** The multiple access mode oscillator is shown in Figure 5-27 in prefoam condition. In final form, the nonmagnetic stainless steel block is surrounded by 1 inch of foam to make a 4-inch cube. The 1.5 pound block provides a long thermal time constant with excellent short time frequency stability, preventing excess frequency drift during eclipse. The unit has been designed for a drift rate not exceeding 8 cps² at the receiver local oscillator frequency (6 kmc). The 32 mc master oscillator includes a crystal transistor oscillator and a transistor buffer stage. The oscillator drives the receiver local oscillator chain and also provides a 32 mc input to the phase modulator through the master oscillator amplifier.

- **Master Oscillator Amplifier.** The master oscillator amplifier (Figure 5-28) receives a sig-

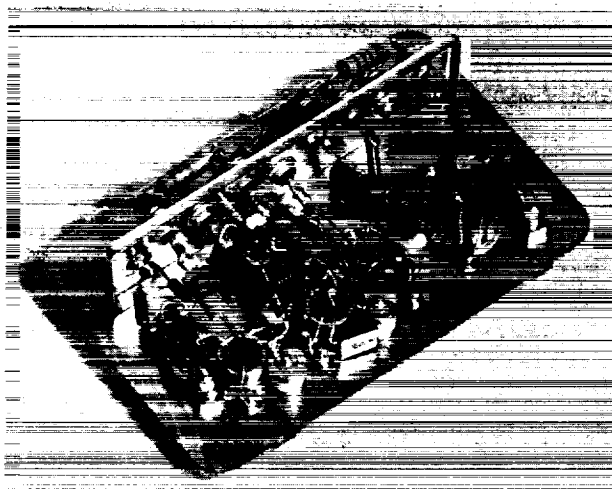


FIGURE 5-28. MASTER OSCILLATOR AMPLIFIER

nal of approximately 32 mc from the master oscillator and provides two output signals, one at 64 mc which is further multiplied to become the receiver local oscillator frequency and another at 32 mc which is provided to the phase modulator to develop the transmitted carrier frequency of the multiple access transponder. Two 2N1405, one 2N777A, and one 2N1506 transistors are employed in the multiplier-amplifier.

Regulators

Multiple Access Mode Regulator. The multiple access mode regulator is a series regulator supplying -24 volts from the -31 ± 5 volt unregulated bus. The output voltage tolerance is ± 2 percent for load, line, temperature, and initial adjustment. Full load is 126 milliamperes and is composed of the loads listed in Table 5-1. For testing convenience, full load has been defined in the test specification as a 200 ohm resistor. This enables the test procedure to be identical with the frequency translation mode regulator (102 unit, $I = 110$ milliamperes at full load).

TABLE 5-1. MULTIPLE ACCESS MODE REGULATORS LOAD ANALYSIS

Multiple Access Transponder Receiver	Current, milliamperes
130 Preamplifier	3.58
141 Filter amplifier	2.8
122 Master oscillator	3.65
123 Master oscillator amplifier	33.6
131 Phase modulator	28.0
132 Doubler amplifier	32.5
114 X32 multiplier (2 units at 11 milliamperes each)	22.0
	126.13

Test results on the first unit were comparable to the breadboard results reported in the June 1963 Monthly Progress Report.* Load, line, and temperature regulation on this unit are summarized in Table 5-2.

TABLE 5-2. MULTIPLE ACCESS MODE REGULATOR VOLTAGE REGULATION

Regulation	Conditions	Measured ΔV	
		Milli-volts	Percent of 24 Volts
Load	$200 < R < \infty$ ohms $V_{in} = -26$ volts, $T = 75^\circ F$	21	0.0875
Line	$-36 < V_{in} < -26$ volts $R = 200$ ohms, $T = 75^\circ F$	3	0.0125
Temperature	$+30 < T < +120^\circ F$ $R = 200$ ohms, $V_{in} = -26$ volts	63	0.262
Total	Load, line, and temperature	87	0.362

The voltage temperature coefficient of this unit was positive at $+29$ ppm/ $^\circ F$. However, this coefficient may be positive or negative, depending on differential amplifier tolerances. The maximum measured temperature coefficients, based on four differential amplifiers, were $-37.5 < K_t < +29$ ppm/ $^\circ F$.

*Syncom Monthly Progress Report, June 1963, p. 4-44 and 4-46 and Table 4-14.

The PARD was measured at 10 millivolts peak-to-peak at full load. The term PARD is an acronym standing for Periodic And Random Deviations, recently introduced to the electronics industry by the IEEE. Generally, it is the combination of ripple and noise chargeable to the supply. It does not include perturbations chargeable to a "noisy" atmosphere or a switching load. The latter would constitute dynamic load regulation.

Frequency Translation Mode Regulators (102 Unit). This unit is a series regulator supplying -24 volts from the -31 ± 5 volt unregulated bus. The output voltage tolerance is ± 3 percent for load, line, temperature, and initial adjustment. Full load is 110 milliamperes and is composed of the loads listed in Table 5-3. For testing convenience, full load has been defined in the test specification as a 200-ohm resistor. This enables the test procedure to be identical with the multiple access mode regulator (101 unit, $I = 120$ microamperes at full load).

Test results on the first two units were satisfactory. Regulation data are given in Table 5-4, as well as the average voltage temperature coefficient over the specified range and the maximum PARD.

COMMUNICATION TRANSMITTERS

384H Traveling-Wave Tube. During the report period the primary effort was directed to three general areas: additional tube development, quantity production of tubes and initiation of a cathode life improvement program. A higher gain traveling wave tube has been developed without sacrificing stability, reliability, high efficiency, or cathode life. All the design characteristics of the traveling-wave tube have been

TABLE 5-3. FREQUENCY TRANSLATION MODE REGULATORS LOAD ANALYSIS

Frequency Translation Transponder Receiver	Current, milliamperes
110 Preamplifier	7.34
104 Intermediate amplifier	7.34
111 Postamplifier (less zener current)	8.8
109 Limiter amplifier	28.0
113 Master oscillator	47.7
114 X32 multiplier	11.0
	110.18

TABLE 5-4. FREQUENCY TRANSLATION MODE REGULATOR PERFORMANCE

Regulation Conditions		Measured ΔV	
		S/N 1	S/N 2
		Per- cent- age Milli- volts of 24	Per- cent- age Milli- volts of 24
Load	$200 < R < \infty$ ohms $V_{in} = -26$ volts and $T = +75^\circ F$	12 0.050	8 0.033
Line	$-36 < V_{in} < -26$ volts $R = 200$ ohms and $T = +75^\circ F$	2 0.008	1 0.004
Temper- ature	$+30 < T < +120^\circ F$ $R = 200$ ohms and $V_{in} = -26$ volts	426 1.772	163 0.679
Total	$150 < R < \infty$ ohms $-36 < V_{in} < -26$ volts $+30 < T < +120^\circ F$	440 1.830	172 0.716
PARD	$150 < R < \infty$ ohms $-36 < V_{in} < -26$ volts $+30 < T < +120^\circ F$	9 millivolts peak-to-peak	6 millivolts peak-to-peak
Voltage temperature coefficient		-197 ppm/ $^\circ F$	-75.5 ppm/ $^\circ F$

tested and the results have been completely evaluated. Methods to further improve performance of the present traveling-wave tube design have been considered.

In the cathode life improvement program, the cathode operating temperature of the 384H has been accurately determined and is lower than initially anticipated. Results of the cathode processing study verify that the tube will not fail

due to loss of the zirconium activating agent. The initial diodes have been constructed and are operating in the diode test rack.

Development and Design. The present design of the 384H was based on following system specifications:

Frequency	3.96 to 4.12 gc
Duty,	CW
Total dc power consumption (including heater power)	14.5 watts (maximum)
Output power	4.0 watts (minimum) for all input power levels from 0.4 to 0.7 milliwatt
Stability	Short circuit stable at all frequencies and phases
Life	50,000 hours
Weight	20 ounces (maximum)

Length	10 inches (maximum)
Cooling	Via conduction to cold plate

Development was directed toward accommodating changes in system requirements and refining the design for production. The operating RF input power of the traveling-wave tube was changed from 1.0 milliwatt to 0.4 to 0.7 milliwatt, increasing the required saturated gain from 36 to 40 db and imposing an input drive range on the tube. The type of RF connectors were changed from TNC to OSM. To assure repeatability during the production run, the tube design was refined by optimizing the helix pitch and loss distribution and improving the impedance match.

Table 5-5 lists the tubes constructed for this development effort and summarizes the individual tube results. As indicated, tubes 32 and 33 were slightly longer than the previous tubes and both met all the performance specifications. Tube 32, with the same pitch as 384H-27, operated at a helix voltage (V_h) of 1310 volts and

TABLE 5-5. DEVELOPMENT TUBES

Tube number	Change from Previous Tube	Test Results	Status
31	Similar to tube 27 except for minor change in loss distribution.	Constructed for previous specifications of 36 db. Tube met all specifications except gain was low (35 db).	Storage
32	Same pitch as 27 except helix is longer and loss distribution modified.	Tube met all specifications, but, efficiency was not sufficient to provide margin of safety in production run.	Storage
33	Same as 32 except for 4 percent reduction in helix pitch and slight change in loss distribution.	Tube met performance specifications, but output power was too marginal for production run.	Storage
34	Same as 33 except for minor change in loss distribution.	Gain and power output low.	Storage
35	Same as 33 except for 2 percent increase in helix pitch and modification of impedance match.	Tube met all performance specifications with reasonable margin of safety for production run.	Scrapped (collector broke in packaging)
36	Same as 34.	Due to low insertion loss caused by distorted helix, tube oscillated.	Scrapped

an anode voltage (V_a) of 125 volts. While its efficiency was within the requirements, the margin of safety was not considered sufficient for a production design. Conversely, tube 33 had a slightly smaller pitch and operated at V_h of 1200 volts and V_a of 190 volts. The efficiency was increased by this change, but the output power was too marginal for this design to be considered suitable for a production run. Consequently, the pitch of tube 35 was selected to be between those of tubes 32 and 33.

The RF impedance match of tube 35 was improved by a change in the rate and length of the impedance tapered portions located on the ends of the helix. The loss distribution of tube 33 was repeated in tube 35. This particular distribution appears to produce greater stability and dynamic range than other distributions without decreasing gain or efficiency. The design of 384H-35 was selected for production.

To accommodate the OSM type connectors, several modifications were necessary: the external portion of the tube impedance match was modified, and, since the standard OSM connectors were not mechanically adaptable to the tube's pin match design, a modified version of the standard OSM connector was used. The design of tube 384H-35 has been used and performance tested on more than 20 tubes, and it meets or exceeds with excellent repeatability all the system requirements. The design parameters are summarized in Table 5-6. A typical 384H traveling-wave tube is shown in Figure 5-29.

Electrical Design and Performance. The system requirements of high efficiency, high reliability, and long life were the major factors in the electrical design. To ensure high reliability, it was desirable to take full advantage of construction techniques developed in the earlier Syn-

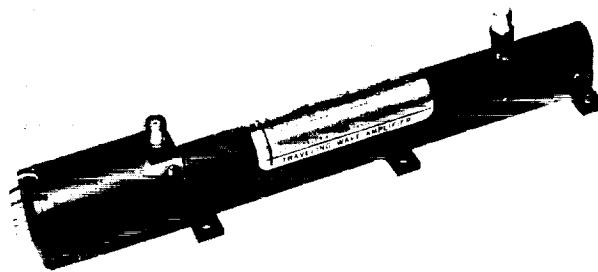


FIGURE 5-29. 384H TRAVELING-WAVE TUBE

com program and, if possible, to use the same parts. This was achieved by using a lower perveance gun. To attain the long life with a large safety factor, a higher convergence gun was used. High efficiency was achieved despite the anticipated drop in basic efficiency (efficiency before collector depression) due to using a lower perveance gun, by improving the optics of the gun and the focusing and collector design. The minimum weight and size requirements were met by using periodic permanent magnetic focusing (PPM).

The helix structure design is similar to the 314H but scaled up in frequency and voltage. Since the helix type traveling-wave tube is inherently a broad-band device with a peak output at the center of its natural bandwidth, gradually decreasing as the operating frequency moves away from this center frequency, the natural choice of an operating frequency for a narrow-band high efficiency application is at this center frequency. The 384H tube was designed in this manner. Figures 5-32 and 5-33 illustrate this "centering process" for this tube design. Figure 5-32 also indicates the variation of output power with frequency for the two required levels of input drive and Figure 5-33, the variation of saturated beam efficiency with frequency for a depressed collector voltage ($V_c = 700$) and non-depressed voltage ($V_c = 0$). The improved efficiency of depressed over nondepressed voltage

TABLE 5-6. SUMMARY OF 384H DESIGN

PERFORMANCE (typical values)		
Peak output power	4.4 to 5.0 watts	
Saturated gain	40 db	
Peak beam efficiency	35 to 40 percent	
VOLTAGES AND CURRENTS (typical values)		
See Figure 5-30		
Element	Voltage	Current, milliamperes
Cathode	1290	20
Anode	160	0.08
Collector	-700	19.0
Helix	Ground	1.0
Heater	4.5	0.260
DESIGN PARAMETERS (4.1 gc)		
γ_a	1.34	
C	0.075	
QC	0.24	
Dielectric loading factor	0.73	
Active helix wavelengths	25	
ELECTRON GUN		
Type	Converging Pierce gun utilizing stacked self-aligning construction	
Cathode type	Oxide coating	

ELECTRON GUN (continued)	
Cathode material	
Base	High purity nickel with 0.1 percent zirconium;
Coating	Double carbonate
Cathode current density	125 ma/cm ²
Estimated wearout life	10 years
Gun perveance	0.34 microperv
FOCUSING	
Type	Periodic permanent magnetic structure
Magnet material	Platinum cobalt
Period	0.350 inch
Peak axial field	900 gauss
RF STRUCTURE	
Type	Helix
Helix support	Pressure assembly, triangulated barrel with ceramic rods
Impedance match	Tapered helix with step transformer
MECHANICAL FEATURES	
Construction	Metal ceramic
Weight	16.5 ounces
Length	9.25 inches
Outline and mounting	See Figure 5-31

operation is obtained by allowing the collector power supply to recover a significant amount of beam power. The recovered power is equal to the product of collector current times the amount of voltage depression. The peak efficiency for production tubes is typically 35 to 40 percent with collector depression and 17 to 20 percent without depression. Typical beam transmission with full depression and at saturation is 94 to 97 percent.

Figure 5-34 shows the typical variation in gain with beam voltage, drive, and frequency. The curve for $V_h = 1175$ is the maximum small signal gain curve. As indicated, the maximum

small signal gain for this particular tube is about 54 db. The lower curves are for saturated and small signal operation at the normal operating beam voltages. The shift in frequency indicated in the figure is always present when a tube is overvoltage, to obtain maximum efficiency.

Figure 5-35 shows log-log plots of output power versus input power for the approximate end points of the required bandwidth. The required power levels are indicated by light broken lines. Peak efficiency and output power occur at different drive levels for the two frequencies. The heavy broken lines indicate the value of gain in the linear (small signal) region.

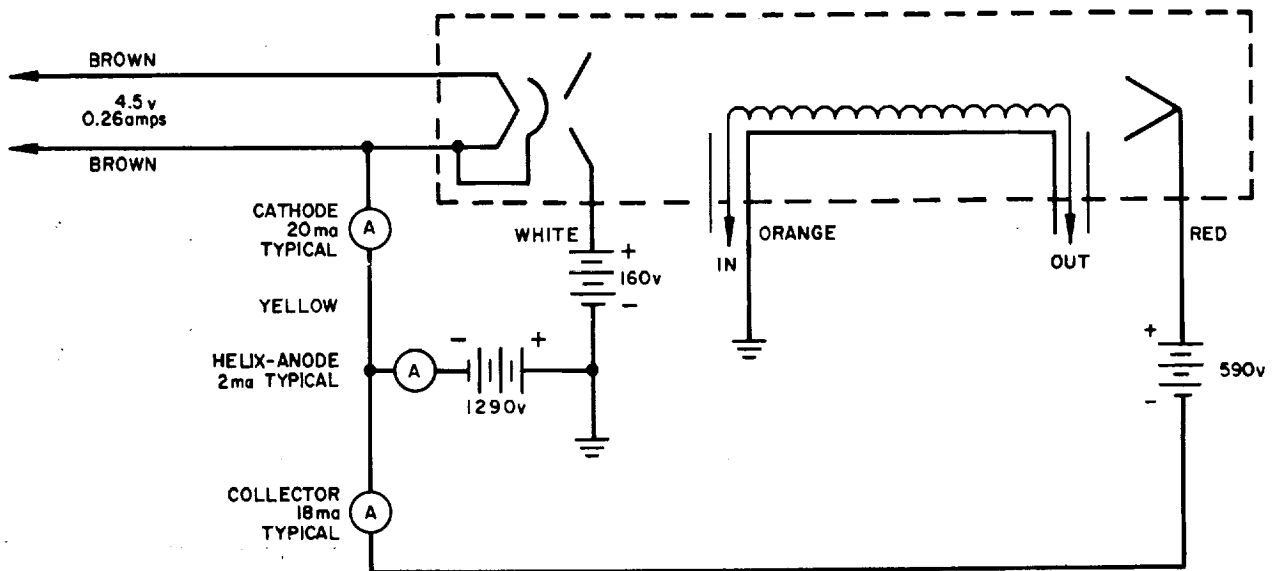


FIGURE 5-30. TRAVELING-WAVE TUBE WIRING DIAGRAM

Figure 5-36 indicates the effect of anode voltage variation on the output power. Since anode voltages are not large, small percentage variations are not critical. Conversely, variations in cathode (helix or beam) potential are very critical. To understand the problem, output performance over the frequency range and at the required input drive levels must be considered simultaneously.

Figure 5-37 illustrates the output power surface of the 384H as a function of two variables: frequency and input power. The surface of required performance is the 4.0 watt plane. The points of high and low overall efficiency (including heater power) are indicated on the surface. The change of drive required for saturated output power as the frequency varies can be noted. The lines of maximum power traverse the surface from approximately the line ($f = 3.95$ kmc, $P_{in} = 0.4$ milliwatt) to the line ($f = 4.20$ kmc, $P_{in} = 0.7$ milliwatt). To meet the minimum power at the corners ($f = 3.95$, $P_{in} = 0.7$ milliwatt) and ($f = 4.20$, $P_{in} = 0.4$ milliwatt), saturated power must be maintained between 4.4 to

4.8 watts. Figure 5-38 shows the effect of ± 1 percent beam voltage regulation on the output power surface. As illustrated, the output drops below the required value at the corners.

Since each traveling-wave tube will amplify only one channel and beacon and have a separate power supply, it was decided to split the frequency range in half, one tube covering the range 3979.5 to 4066.2 mi. and the other the higher frequency range 4119.9 to 4194.5 mi. No change in tube design is required; each tube is merely voltage tuned at two frequencies. Therefore, each has two sets of operating voltages: one for 4.0 kmc operation and the other for 4.15 kmc operation. This solution allows the beam voltage regulation to be increased to ± 1 percent without sacrificing gain, efficiency, or output power. Figures 5-39 and 5-40 illustrate this performance. Another advantage of the tuning operation is that power output is slightly increased; typical peak values range from 4.5 to 5.0 watts.

The 384H tube has an excellent impedance match from 3.5 to 5.0 kmc. The percent voltage reflection of the match is usually between 5 and

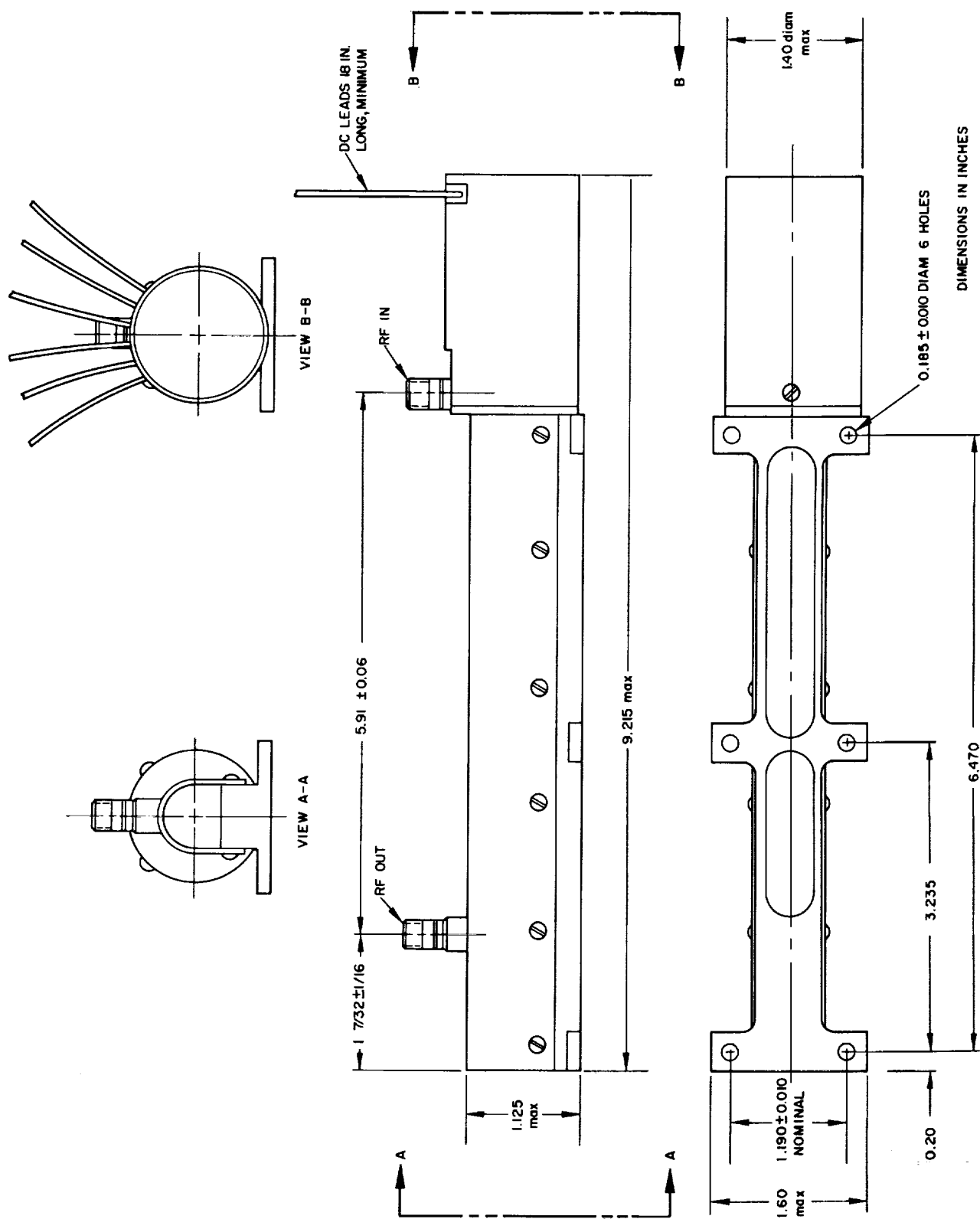


FIGURE 5-31. TRAVELING-WAVE TUBE OUTLINE

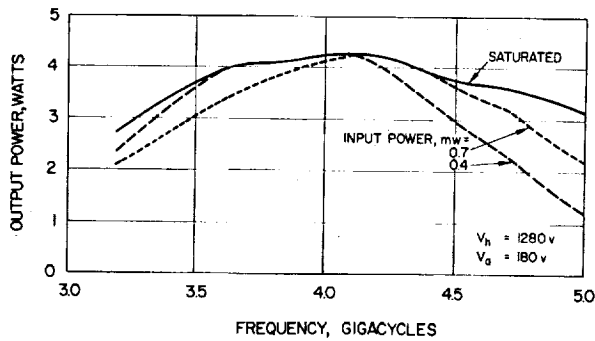


FIGURE 5-32. 384H OUTPUT POWER VERSUS FREQUENCY

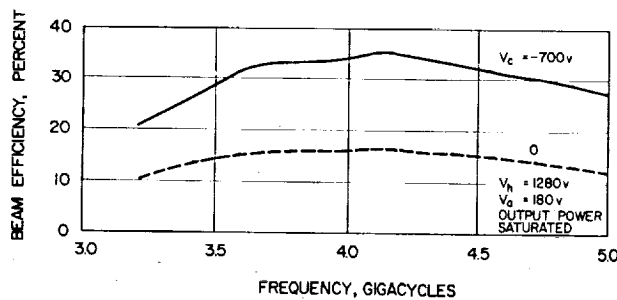


FIGURE 5-33. 384H BEAM EFFICIENCY VERSUS FREQUENCY

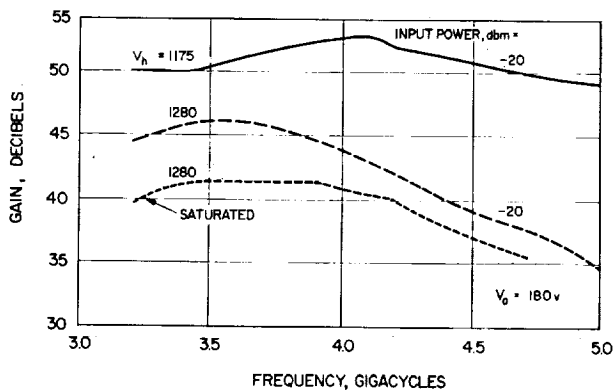
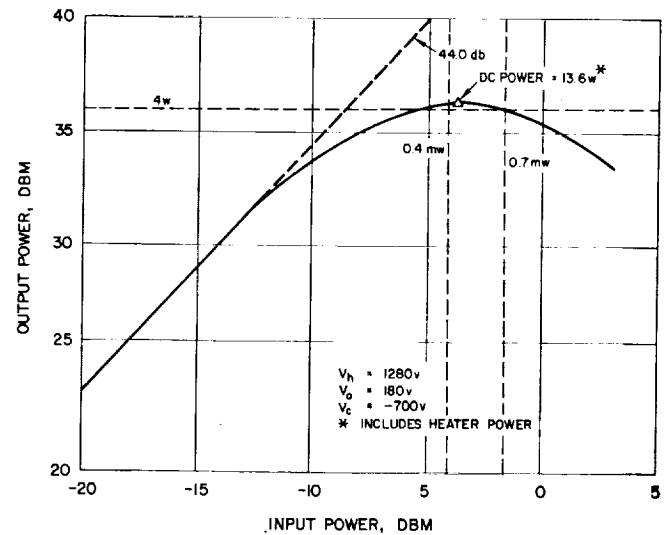
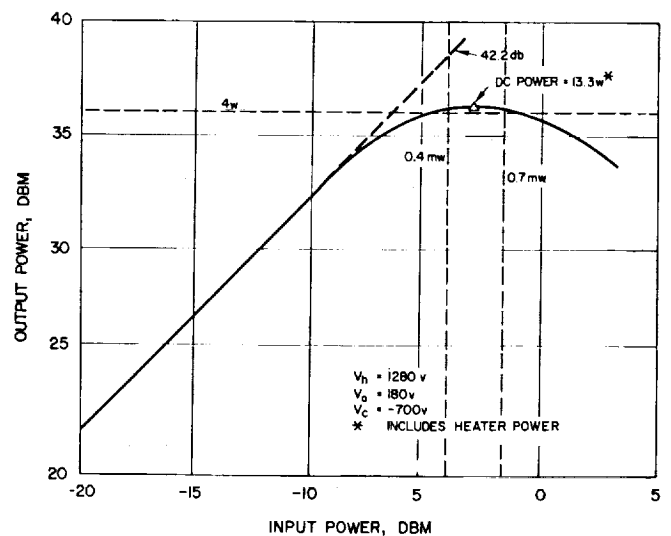


FIGURE 5-34. 384H GAIN VERSUS FREQUENCY

15 percent. Figure 5-41 shows the percent voltage reflection of the tube from 3.0 to 5.0 kmc. Stability is not a problem of the 384H. The 384H retains stability by maintaining very tight tolerances on the helix, excellent impedance match,



a) Frequency = 4.0 gc



b) Frequency = 4.2 gc

FIGURE 5-35. 384H OUTPUT POWER VERSUS INPUT POWER

and proper distribution and control of loss on the ceramic support rods. The stability is tested by measuring the "hot output match." With the RF input terminated in a matched load, a signal is fed into the RF output of the tube while the beam is operated at the small signal gain voltage. By the use of a ratio meter, the incident wave to the tube is compared to the reflected wave as the

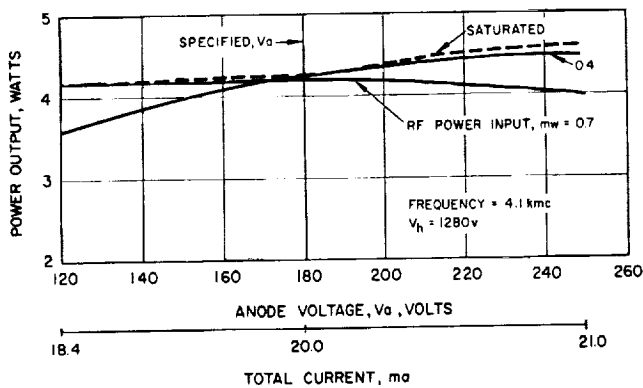
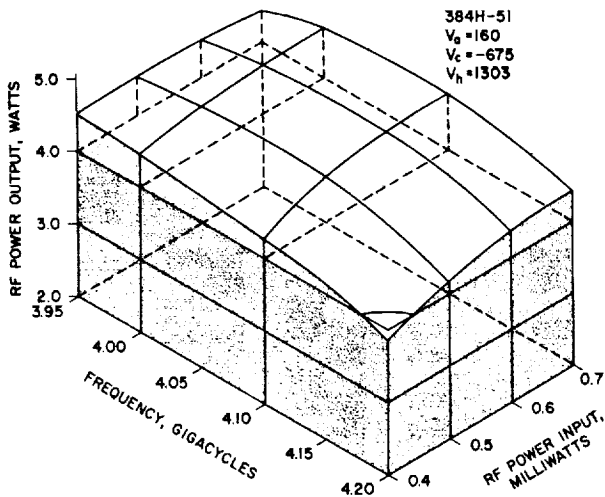
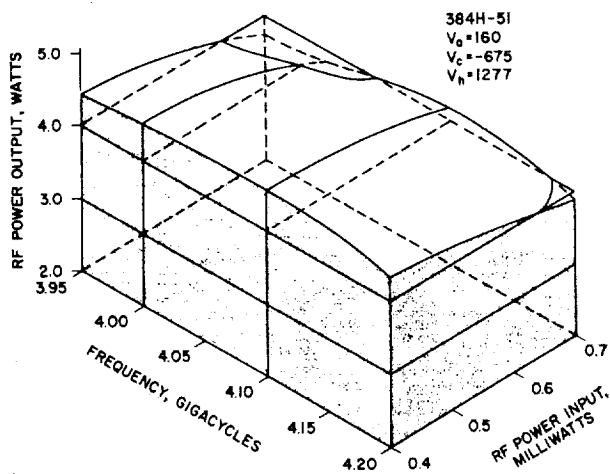


FIGURE 5-36. 384H OUTPUT POWER VERSUS ANODE VOLTAGE



a) 1303 Volts



b) 1277 Volts

FIGURE 5-38. 384H HELIX VOLTAGE AT ± 1 PERCENT REGULATION

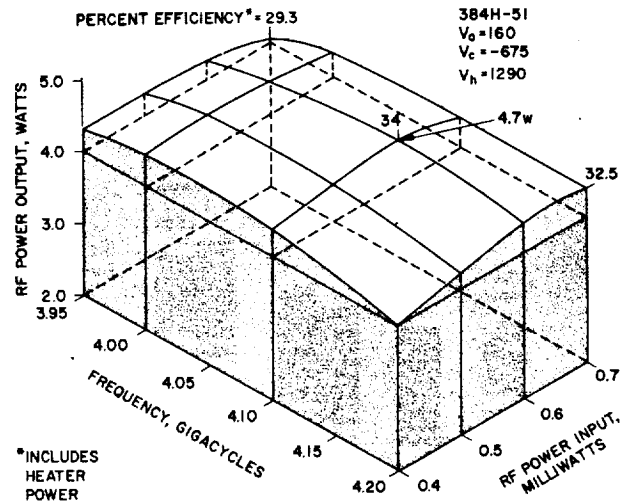


FIGURE 5-37. 384H HELIX VOLTAGE AT 1290 VOLTS

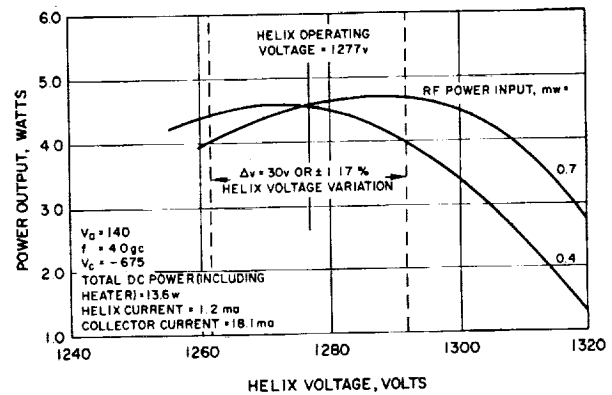


FIGURE 5-39. 384H OUTPUT POWER AS FUNCTION OF HELIX VOLTAGE VARIATION

frequency is swept from 3.0 to 5.0. A voltage reflection close to or greater than 100 percent automatically disqualifies any tube for shipment. Figure 5-42 shows a typical hot output match.

Fabrication Status. The production status as of 14 October 1963 was as follows:

	Number of Tubes
Stage I — Clean room assembly	2
Stage II — Bakeout process	4
Stage III — Magnet assembly	2

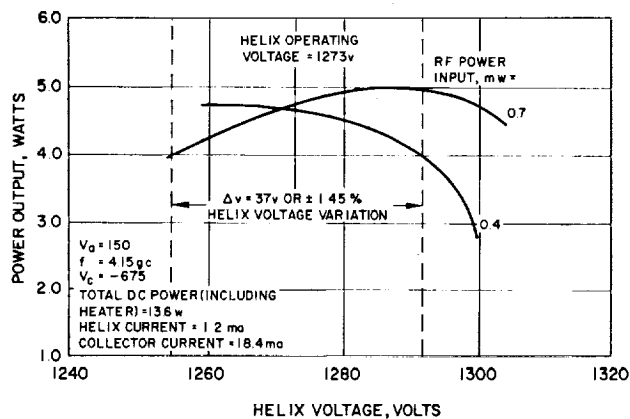


FIGURE 5-40. 384H OUTPUT POWER AS FUNCTION OF HELIX VOLTAGES ADJUSTED FOR OPERATION IN UPPER PART OF FREQUENCY BAND

Stage IV — Focusing and RF testing	
Stage V — Pre-aging	5
Stage VI — Refocusing and RF testing	1
Stage VII — Packaging	2
Stage VIII — RF testing after packaging	2
Stage IX — Environmental testing	4
Stage X — Shipped	7
Qualification test	1
Transponder	4
Life test	2

The list includes only tubes that are still in process and none that have dropped out because of substandard performance or construction defects.

Two interruptions have delayed the planned production schedule, but the consistent repeatability of tube performance indicates that all required tubes will be built and delivered on time. The first interruption occurred before any

of the tubes had been packaged. The required OSM RF connectors were not received on time. The mechanical design of the tube package is such that tubes cannot be packaged without the connectors being installed. Consequently, after several tubes had reached this stage production was curtailed until the new connectors arrived.

The second delay was a temporary halt of the production of standard tubes because a charging problem was discovered. The charging phenomenon caused a slight power output variation that was associated with a slow periodic helix current variation. It was found that tubes with this variation had been constructed with improperly cleaned ceramic windows, and the entire lot of ceramics was scrapped and a new window put into use. The magnitude of variation greatly decreased when tubes were pre-aged for 200 hours after the initial focusing and then refocused.

Several experimental tubes were constructed in an effort to learn the exact nature of the charging phenomenon. To date, no significant data has been obtained from these tubes. The change of ceramics coupled with the pre-aging stage of production appears to eliminate the problem. After a 3-week delay, tube production was resumed at an accelerated rate. During the delay period, it was determined that the magnitude of the output power variations would not affect performance of the communication system.

Life Improvement Program. In a properly designed vacuum tube, the ultimate or wearout life is determined by eventual reduction of electron emission from the cathode. An oxide cathode consists of a nickel pellet coated with a mixture of barium-strontium oxide (BaSr)O, usually abbreviated as simply BaO. The cathode must be heated to 650° to 800°C to act as a useful emitter. Pure BaO itself is a poor emitter; however, introduction of free barium atoms into the

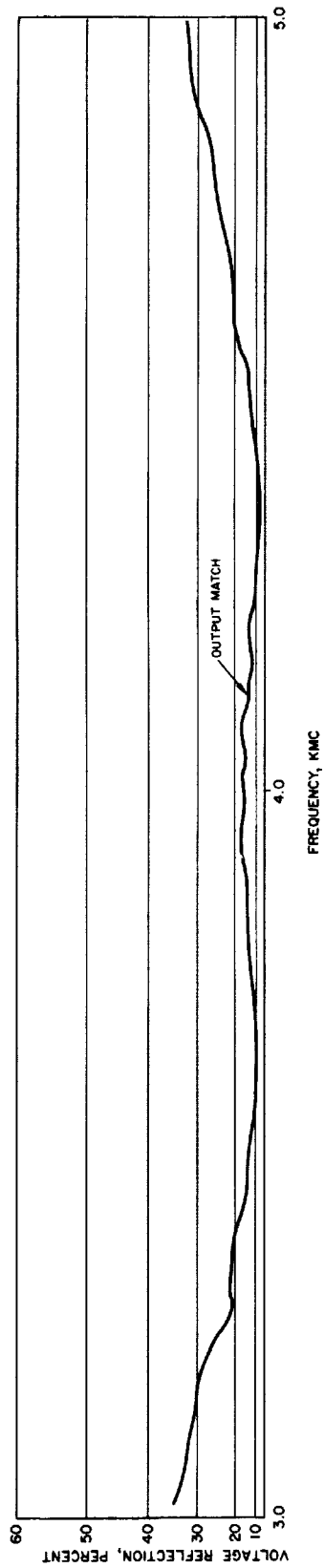


FIGURE 5-41. 384H PERCENT VOLTAGE REFLECTION OF COLD IMPEDANCE MATCH VERSUS FREQUENCY

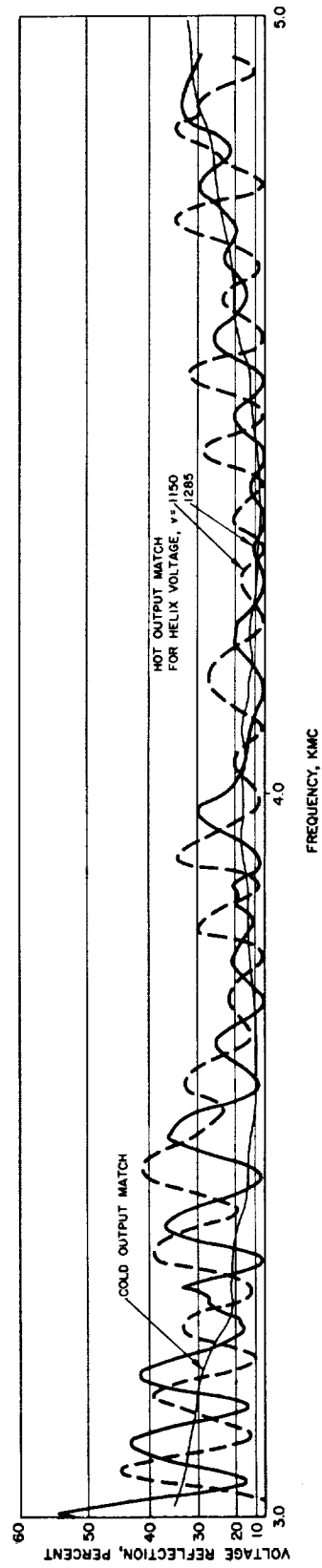


FIGURE 5-42. 384H PERCENT VOLTAGE REFLECTION OF HOT IMPEDANCE MATCH VERSUS FREQUENCY

BaO coating produces an active thermionic emitter. During emission, the BaO coating acts as an n-type, excess-impurity semiconductor, with Ba as the impurity donor. Thus, an adequate level of Ba donors must be maintained if emission is to be preserved. This level is believed to be that required to saturate the BaO coating, probably less than 0.01 percent (by weight) of free barium. Unfortunately, the Ba atoms are continuously being depleted through evaporation and through chemical reaction with residual gas molecules and ions. (Even at a vacuum as good as 10^{-9} Torr, there remain many billions of gas molecules in the electron gun portion of the tube alone.)

If emission is to be maintained, these lost Ba atoms must be continuously replaced at a rate equal to, or greater than, the sum of the evaporation rate plus the gas "poisoning" rate. The required replacement Ba atoms are produced by chemical reduction of the BaO coating molecules to produce free Ba plus an oxide of the reducing agent. The reducing agent consists of an impurity element purposely introduced into the nickel base-metal raw material before the cathode pellet is fabricated. When the cathode is heated during operation, this additive diffuses continuously through the nickel pellet to the BaO coating where it chemically reduces the coating to produce the required Ba atoms.

Based on this model of the oxide cathode, it is clear that end of life will occur when the rate of production of new Ba atoms falls below the rate of poisoning loss of Ba atoms. According to the diffusion dependency hypothesis,* the rate of BaO reduction to free Ba is determined by the rate at which the base metal additive arrives at

the BaO-nickel interface, and hence the Ba production rate is controlled by the diffusion rate of the additive through the base nickel. As the additive is used up during life this diffusion rate slowly decreases, until finally when the additive concentration in the base metal has dropped to 50 or 60 percent of its original value, the diffusion rate begins to drop quite rapidly. End of life occurs when the additive diffusion rate is too low to provide free barium at a rate which will overcome barium losses.

The cathode material used in the 384H is a very pure nickel alloy, containing 2 percent tungsten and 0.1 percent zirconium. The zirconium is the active reducing agent. This additive has been used for all Hughes space traveling-wave tubes to date as well as for the BTL Telstar traveling-wave tube. This choice is based on extensive experimental studies using various impurities which indicate that Zr produces the highest activity for the longest period of time.*

The diffusion rates for the 384H cathode were calculated at the inception of the 384H program with cathode temperature as the parameter. For convenience the rate at which Zr arrives at the BaO coating was plotted in terms of the rate of Ba atom production that would result from the availability of the arrived Zr. In order to estimate end of life, a required minimum arrival rate must be assumed equal to the expected rate at which Ba atoms are being lost. Based on various accelerated life studies, a conservative value is judged to be 10^{10} atoms/cm²/sec. Thus, based on the diffusion-dependent life theory and the initially determined range of cathode operating temperatures (720° to 760°C) the calculated life of the 384H cathode is 140,000 to 220,000 hours.

*See, for example, H. E. Kern, "Research on Oxide Cathodes," Bell Laboratories Record, December 1960.

*H. E. Kern and E. T. Graney, "Thermionic Emission and Diffusion Studies on Zirconium - Doped Nickel Cathodes" Report on Twenty-Second Annual Conference, Physical Electronics, March 1962.

In order for the diffusion-dependent life theory to apply, it must be certain that the amount of BaO coating available is sufficient that the BaO supply itself will not be depleted before the zirconium. Therefore the coating thickness and coating density are accurately measured on each cathode used and the data recorded in the permanent records kept for each tube. Calculations made for the presently used coating density and thickness indicate that more than adequate coating is available, so that this will not be the limiting factor.

The preceding discussion represents a much idealized model of the oxide coated cathode, however, and the diffusion-dependent life theory has not been completely verified on cathodes designed for extremely long life since the time involved is so great. However, accelerated life test data obtained at other laboratories by purposely designing cathodes with thinner base metal or with faster acting additives have tended to verify the life predicted by the theory. There is in fact some doubt about the failure mechanism itself since in at least one test the expected reduction in additive versus life was not observed.

The life theory described, which is the most commonly accepted one, does not indicate any dependence at all of cathode life on current density. Nevertheless, considerable empirical data from various other life test programs suggest that life is a function of current density. In fact, many estimates of cathode life are based on such data, because experience and actual life tests confirm that it is a reliable and conservative way to predict cathode life. However, there is some indication that accelerated life tests on tube types that had lower current densities showed a more favorable required arrival rate at end of life. In addition are various theories relating to a possible dependence of life on cathode current density which have to do with the electrical con-

ductivity of the coating. Some are reviewed in the introduction of a report by Hollister.*

Diode Program. In view of the design goal of 5 years for the Advanced Syncom spacecraft and discrepancies in the currently accepted cathode life theories, a life improvement program has been initiated to further understanding of potential failure mechanisms. Present theoretical understanding of the oxide coated cathode and empirical data obtained from space tube programs are sufficient to assure that design of the 384H cathode will not fail due to normal depletion of either the cathode coating or the zirconium activating agent in the cathode base material. However, as the part of the life improvement program it is desirable to establish that cleaning and processing of the tube parts, cathode coating application, and processing and activation schedules are adequate to ensure that full use of the zirconium activating agent and barium oxide coating is possible. It is also desirable to accelerate eventual failure of the cathode in controlled tests to gain early knowledge of the long life failures to be expected and determine whether indications of eventual failure might occur during the burn-in period of the 384H traveling-wave tubes.

The life improvement program utilizes a relatively inexpensive diode which is constructed and operated under the same conditions as the 384H traveling-wave tube. Except for the helix assembly, the diodes have the same electron gun, envelope, and parts as the 384H. Therefore, the diode cathodes have the same environment and operating conditions as the cathodes used in the actual tube, ensuring that the data obtained from the program will truly be representative of the 384H cathode.

*"Some Effects of Ion Bombardment on the Emitting Properties of Oxide-Coating Cathodes," Cornell University Research Report, EE 482.

The primary variables under investigation are cathode temperature, cathode thickness, and current density. To determine the relationship between these variables and long life performance, 50 diodes, a diode life test rack, and power supplies are being constructed.

- **Cathode Temperature.** The most critical variable involved is the cathode temperature. Of all the parameters measured in a tube, cathode temperature is often the least precisely determined. Therefore, it was necessary to control and reproduce the true cathode temperature within $\pm 5^\circ\text{C}$ and have a means of monitoring the cathode temperature during the entire duration of the life test program. To meet this requirement, platinum versus platinum —10 percent rhodium thermocouples are being installed in each diode. To minimize errors due to heat conduction losses, 2-mil-diameter thermocouple wire is being used. To ensure that the results obtained from the diode life improvement program will be representative of the 384H traveling-wave tube, the diodes are processed under exactly the same conditions as the 384H. During the bakeout, the vacuum envelope is maintained at 500°C .

A method was developed to bring the thermocouple wires out of the vacuum envelope without degrading the integrity of the vacuum seal necessary to obtain a "clean" environment. Two small hollow molybdenum pins were brazed into the ceramic wall of the diode, the thermocouple wire running through the pins and copper-brazed in place. Since the brazed joint introduces an extraneous junction in the thermocouple circuit, the first diode built had two extra thermocouples built into it. The first was placed at the brazed joint (i.e., the extraneous junction) and the other was a standard platinum versus platinum —10 percent rhodium thermocouple attached to the cathode alongside the regular thermocouple. The stand-

ard thermocouple was brought out of the vacuum envelope, sealed in place with an epoxy resin, and terminated in an ice bath reference junction. Using this first diode, all these parameters have been measured and the true cathode temperature obtained very accurately by the standard thermocouple, thus enabling calibration of the brazed-in thermocouple. The true cathode temperature of each of the diodes can now be determined within $\pm 5^\circ\text{C}$.

From temperature measurements on the first four diodes, it was determined that the 384H cathode operates at a lower temperature than was originally estimated (Figure 5-43). Since the operating temperature of the cathode is indicative of the poisoning atmosphere in the tube, the results of this program substantiate the fact that the poisoning atmosphere in the 384H is extremely low. This verifies the calculations that barium oxide coating is more than adequate and will not be depleted before the zirconium.

- **Construction of Diode Cathodes.** At the inception of the life improvement program, it was decided that one of the variables used to accelerate the failure of cathodes would be cathode thickness. The life of the cathode would be calculated by the diffusion-dependent life theory for

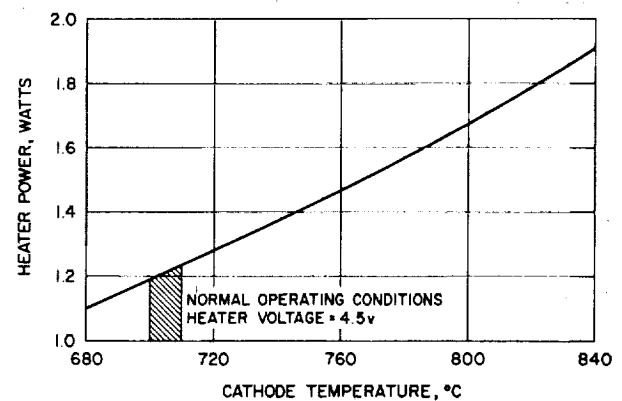


FIGURE 5-43. 384H HEATER POWER VERSUS CATHODE TEMPERATURE

three different values of cathode thickness: normal, one-half normal, and one-quarter normal. The cathode coating thickness and density will be the same for all diodes. As the cathode is made thinner it will contain less zirconium and produce a shorter life. Thus it will be possible to confirm the diffusion-dependent life theory.

A cross section of the cathode normally used in the 384H is shown in Figure 5-44. The entire cathode is machined from the 2 percent tungsten — 0.1 percent zirconium nickel alloy. The normal thickness of the cathode is 16 mils (region A in Figure 5-44). According to the diffusion theory, if the thickness is reduced to 10 mils, the life will be reduced from 26 years to 14 years. If the thickness is reduced to 3 mils, the life will be reduced to 5 years. All these calculations are for the normal operating cathode temperature. For the thicknesses specified above, it seemed reasonable to fabricate and build the diodes to these specifications. However, the original calculations ignored the effect of the side walls of the cathode (region B in Figure 5-44). More detailed diffusion calculations showed that as the temperature of the cathode was raised to further accelerate the failure of the cathode, the diffusion of zirconium out of region B will prevent the failure. For example, a 3 mil thick cathode running at the normal cathode operating temperature should

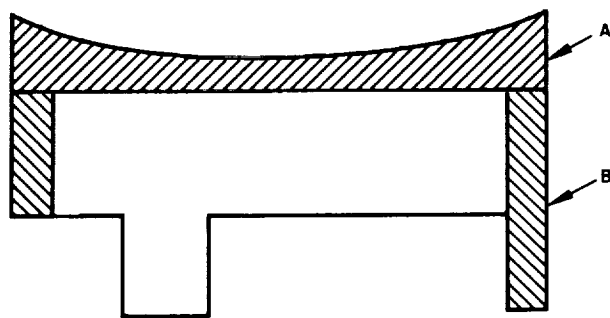


FIGURE 5-44. CROSS SECTION OF 384H CATHODE

fail after 5 years. When the operating temperature is increased to 480°C it should fail in 3 months. At the normal operating temperature, the effect of diffusion from region B would increase the life by only 4 months. At 840°C, the life of the cathode would be 2 years just by the diffusion of zirconium from region B. It is therefore necessary to redesign the thinner cathodes.

The 3 mil thick cathode and the 10 mil thick cathode are constructed as follows. Region B of Figure 5-44 is made of 270 grade nickel, a high purity nickel free from activating agents, with a disk of the 2 percent tungsten — 0.1 percent zirconium cathode alloy welded to it. This disk is then machined to the required dimensions. With the new design, it is felt that the diode program will result in the data needed to verify the diffusion-dependent theory of cathode behavior.

All of the normal cathodes have been made. Fifty one-half normal cathodes and fifty one-quarter normal cathodes have been ordered. (The 10 mil cathode is considered one-half normal cathode because its zirconium content will reduce the cathode life by a factor of 2; the 3 mil cathode is considered a one-quarter cathode because its zirconium content should reduce the cathode life by a factor of approximately 4.) Ten of each have been received, and the balance of the order is being machined to the final dimensions.

- Cathode Cleaning Study. At the beginning of the life improvement program, a thorough study of the cathode cleaning process was started. It is known that certain elements such as sulfur, silicon, and aluminum, when present in small trace quantities on the order of a few hundredths of a percent by weight, can "poison" a long life cathode. Therefore, after the cathode is machined to its final shape it must be cleaned. The cleaning process should remove all contaminants and

degas and lower the carbon content of the cathode with a minimum loss of the desired activating agent (zirconium). To assure that goals were being achieved, the cleaning process was divided into three stages:

- Stage I — chemical cleaning to remove grease, surface film, and contaminants
- Stage II — wet hydrogen firing to lower the carbon level
- Stage III — surface etch, air oxidation, dry hydrogen firing, and vacuum firing to remove embodied surface contaminants not removed in Stage I, prepare the cathode surface for maximum adhesion of the cathode coating, and degas the cathode

Thirty standard 384H cathodes were then sent through the cleaning process and a sample consisting of ten cathodes was removed after each stage. The three samples were analyzed for composition and gas content. The results of these analyses confirmed that stages I and III were effective, with no evidence of contamination pickup or zirconium depletion. In stage II there was evidence of some contamination pickup; the nitrogen and oxygen content increased, and there was a slight depletion of zirconium. However, the carbon and sulfur content was reduced to less than ten parts per million for both elements, which is an extremely low level. In stage III, the contamination pickup was reduced to a sufficiently low level to obviate any possible degradation of the expected life of the 384H normal cathode. Diffusion calculations also verified that the zirconium loss was so slight that it would have no effect on the life of the normal cathode.

Although the cleaning process was found to be adequate for the normal 384H cathode, the contaminant pickup and zirconium depletion in stage II is a function of the thickness of the material being cleaned. The 3 mil thick cathodes to be used in the diode program would be contaminated to an unacceptable level and the zirconium would be reduced so much that these cathodes would be useless. Therefore, immediate action was taken to eliminate this problem. The source of contamination has been traced to contaminants present in the ceramic muffle of the wet hydrogen firing furnace.

A different hydrogen furnace was taken apart, cleaned, and reserved exclusively for processing all space tube cathodes. This furnace was checked for contamination by firing some thin samples of 270 nickel, in which it is easier to detect contaminations than in the nickel alloy used for the cathodes. It was calculated that the firing temperature used in stage II was higher than necessary for the removal of the carbon and that at a lower temperature the zirconium depletion would be reduced to an insignificant level. Based on these considerations a more optimum firing temperature was chosen. Thin samples have been fired under these conditions and sent out for analysis. The results had not been received at the time of this report's publication. All of the normal cathodes required for the program have been processed and analyzed.

- Diode Life Test Procedure. In order to obtain the life failure data in a reasonably short period of time, the eventual failure of the cathode will be accelerated in a controlled manner. This will be accomplished by varying the cathode thickness, cathode temperature, and cathode loading in accordance with the following schedule:

- 1) Three normal cathodes to be run at normal cathode temperature and normal cathode

current. Calculated theoretical life is 26 years. These diodes will serve as a control to correlate diode results to the actual 384H tube.

- 2) Three normal cathodes to be run at an elevated cathode temperature of 840°C and normal cathode current. Calculated theoretical life is 2 years.
- 3) Three normal cathodes to be run at an elevated cathode temperature of 840°C and three times normal cathode current.
- 4) Five one-half normal cathodes to be run at an elevated cathode temperature of 840°C and normal cathode current. Calculated theoretical life is 1 year.
- 5) Five one-half normal cathodes to be run at an elevated cathode temperature of 840°C and three times normal cathode current.
- 6) Three one-quarter normal cathodes to be run at normal cathode temperature and normal cathode currents. Calculated theoretical life is 5 years.
- 7) Four one-quarter normal cathodes to be run at an elevated cathode temperature of 800°C and normal cathode current. Calculated theoretical life is 1 year.
- 8) Five one-quarter normal cathodes to be run at an elevated cathode temperature of 840°C and normal cathode current. Calculated theoretical life is 3 months.

The diffusion-dependent life theory was used to calculate the results. Because the diffusion dependent life theory is independent of current density, the calculated theoretical life was not stated for items 3 and 5.

The diode life test rack used for the 314 and 349 diodes is now being modified. The power

supplies are being converted so that they will provide the proper voltage for the 384H diodes. The voltage regulation is being improved to meet the standards of control necessary for the 384H diode program. When completed, sufficient life test racks will be available to run 35 diodes. The present capability is 25 diodes.

Traveling-Wave Tube Environmental Tests.

An Advanced Syncom traveling-wave tube, MTD 384H - No. 13, successfully passed qualification test requirements. All electrical parameters of the tube were checked before and after each test to verify that the proper characteristics of the tube were maintained. The tests were performed in the following order:

- 1) *Thermal vacuum.* The tube was attached to a mounting plate and the surface mounting temperature varied as required. The tube was operated continuously on dc power and the RF amplification was checked every 24 hours. Two tests, each of 72 hours duration, were made at 1×10^{-6} Torr and mounting surface temperatures of 25° to 35° F and 165° to 175° F. After the test, all parameters of the traveling-wave tube were checked and no changes were noted because of the test.
- 2) *Vibration.* The tube was subjected to the vibration environments in each of the three orthogonal axes indicated below. The tube was checked at the completion of each frequency sweep.

Sinusoidal Vibration (logarithmic sweep)

Frequency, cps	Thrust Axis, g, 0 to peak	Time, minutes
5-15	0.25 inch DA	0.75
15-60	0.5 inch DA to 24.2, 15 g to 60 cps	1.00

Frequency, cps	Thrust Axis, g, 0 to peak	Time, minutes
60-100	30	0.38
100-250	15	0.63
250-400	10	0.38
400-2000	7.5	1.20

	Acceleration, g	Duration, milliseconds
Forward	30	11
Lateral	15	11

The tube was subjected to the following acceleration environment:

	Transverse Axis, g, 0 to peak		Acceleration, g	Duration, minutes
5-15	0.25 inch DA	0.75	Thrust axis (Z-Z)	+30 -30 10 10
15-25	0.5 inch DA	0.38	Transverse axis (X-X)	+6 -6 4 4
25-60	0.5 inch DA to 35, then 30 g to 60 cps	0.63	Transverse axis (y-y)	+6 -6 4 4
60-250	15	1.00		
250-400	10	0.38		
400-2000	7.5	1.00		

After all these tests — shock and sustained acceleration — all parameters of the traveling-wave tube were checked and no changes were noted because of the test.

Random Vibration

Frequency, cps	All Axis, psd	Time, minutes
20-80	Flat 0.4 g ² /cps	6
80-1280	Increasing from 0.4 to 0.07 g ² /cps at 1.22 db/octave	6
1280-2000	0.97 g ² /cps	6

- 4) *Spin* — The tube was oriented on the spin fixtures in a position approximating the mounting angle on the spacecraft. Operated continuously on dc power, the tube was subjected to 12 g acceleration at a rotation of 150 rpm for 7 days.

Traveling-Wave Tube Power Supply. The power requirements of each traveling-wave tube (TWT) are supplied by a dc-to-dc converter and a dc-to-ac inverter. The input power to the converter and inverter is supplied by a series regulator. The inverter supplies filament power, while the converter supplies power to the tube cathode, anode, and collector (the tube helix is operated at ground potential). Both the inverter and converter are saturable core square wave oscillators. The converter has rectified outputs and the inverter supplies approximately constant power to the tube filament.

After the vibration tests, all parameters of the traveling-wave tube were checked, and no changes were noted because of test.

- 3) *Shock and sustained acceleration* — The tube was subjected to a sine acceleration pulse in each of the following orthogonal axes:

DC-to-DC Converter. The traveling-wave tube high voltage dc-to-dc converter is a saturable core square wave oscillator with three rectified outputs. It is similar to the converter used in Syncom 2 with the following exceptions:

- 1) An increase of approximately 2:1 in voltage level has required a larger transformer.
- 2) A larger output filter was required to meet a more stringent ripple requirement. This requirement is still 500 millivolts peak-to-peak, the same as Syncom 2. However, the 2:1 higher voltage level requires a 2:1 more stringent ripple specification. Also, a larger percentage of filter weight has been invested in inductors because of the physical size of 1000-volt capacitors.
- 3) A larger input filter was employed to reduce the amplitude of switching transients reaching the unregulated bus.
- 4) The diodes in the helix-collector and collector-cathode circuits were changed from 500-volt to 825-volt diodes. The 825-volt diodes have the additional advantage of a faster recovery time.
- 5) The base drive current limiting resistor was replaced by a sensistor which is a 7000 parts per million (ppm) per °C temperature sensitive resistor. The sensistor allows more base drive at low temperature, and vice versa, compensating for the change in switching transistor beta.

The first breadboard converter has been operated satisfactorily from -100° to 160°F . Figure 5-45 represents the effect of temperature on full load output voltage and efficiency. Since each TWT will require different voltages, the per unit, (where 1 per unit = 100 percent) change due to temperature was of greater sig-

nificance than the actual voltage change. Therefore, the V versus t curves of Figure 5-45 have been normalized with respect to the actual voltages at 75°F . Also, there are two efficiency curves in Figure 5-45. The curve indicating 86.1 percent efficiency at 75°F represents the basic dc-to-dc converter. However, the extremely light TWT anode circuit load will require a bleeder resistor to prevent the filter capacitors from charging up to the square wave leading edge spike. When the dissipation in a 100-kilohm bleeder resistor is charged to the supply (400 millivolts at 200 volts), the efficiency reduces to 83.8 percent.

The converter is not a regulated power supply, but derives its essentially constant output voltage from the regulated input voltage. Changes in TWT load current working against the inherent output resistance of the converter causes some change in output voltage. Figure 5-46 represents the V versus I characteristics of the dc-to-dc converter with constant input voltage. The three

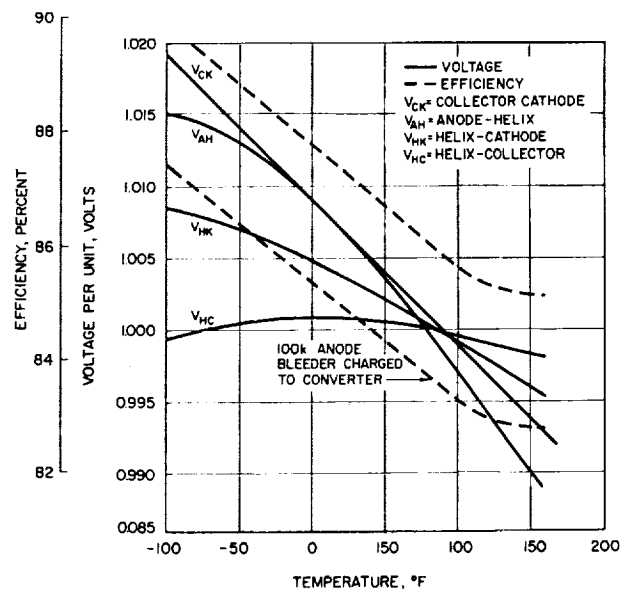


FIGURE 5-45. CONVERTER EFFICIENCY AND VOLTAGE AT FULL LOAD VERSUS TEMPERATURE

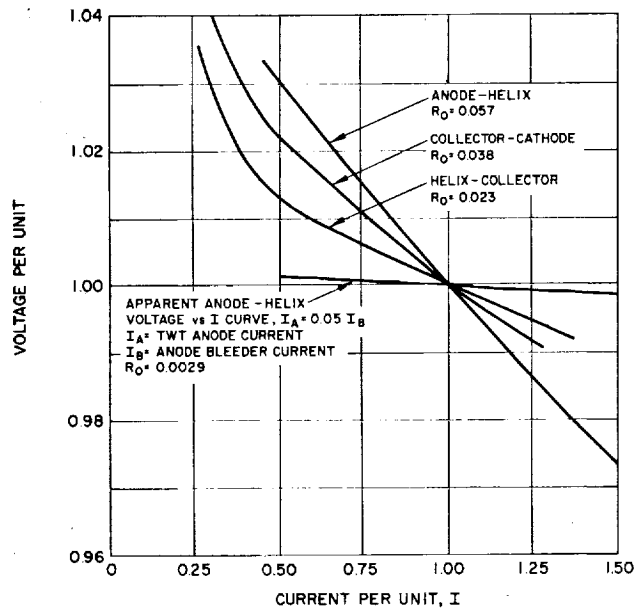


FIGURE 5-46. CONVERTER NORMALIZED V VERSUS I CURVES

TABLE 5-7. TYPICAL TRAVELING-WAVE TUBE POWER SUPPLY OUTPUT IMPEDANCE

Output	Output Resistance, kilohms per unit		Output Capacitance, microfarads
Anode-helix	5.8	0.0029	0.015
Helix-collector	5.0	0.023	0.068
Collector-cathode	1.2	0.038	0.068
Helix-collector	—	—	—

outputs of the converter were tested one at a time with the other two held at full load. Again the curves have been normalized since each TWT will require different voltages and currents. The slope of the Figure 5-46 curves at $V = 1.0$ per unit and $I = 1.0$ per unit represents the per unit output resistance R_0 for small dc changes in load. Dynamic load regulation will be governed primarily by the output filter frequency response. This filter is an inductive input three-section lattice for each output. At high frequency, the capacitors across the output terminals will predominate, and the output impedance will be

$Z = 1/j\omega c$. The dc output resistances and capacitances appear in Table 5-7.

The most sensitive voltage from the standpoint of TWT operation is the helix-cathode voltage, which is the sum of the helix-collector and collector-cathode output with the helix at ground potential. This voltage should be held to ± 1 percent of the rated value for initial adjustment plus unregulated bus and temperature changes. The TWT load changes during normal operation are considered negligible. Changes in the cathode-helix voltage due to load regulation will be much less than 1 percent. An initial adjustment to within 0.1 percent of the desired value should be possible for the helix-cathode output by adjusting the series regulator output voltage to compensate for all converter manufacturing tolerances.

Unregulated bus voltage changes may be considered negligible. These changes are attenuated on the order of 5000:1 by the series regulator. Since the converter acts essentially as a linear transformer, a change in the 24-volt input will result in an equal percentage change in each of the output voltages. An 8-volt unregulated bus variation can occur in normal operation, which would result in a voltage change of 68 ppm in each output.

The most significant factor affecting the TWT voltages will be temperature. Figure 5-45 demonstrates the converter inherent temperature regulation with constant input voltage. However, the total TWT supply temperature regulation will be the sum of the converter plus series regulator temperature regulation. Figure 5-45 shows the helix-cathode output to have a negative temperature coefficient of approximately -59 ppm/ $^{\circ}\text{F}$ for $30^{\circ}\text{F} \leq T \leq 120^{\circ}\text{F}$. If ± 0.8 percent of the total ± 1 percent tolerance is allowed

for temperature changes, this is a total voltage change of 1.6 percent (16,000 ppm), or an overall coefficient of 178 ppm/°F for 30°F < T < 120°F. Therefore, the regulator voltage temperature coefficient can be $+59 \pm 178$ ppm/°F. If the series regulator were designed to have a controlled positive temperature coefficient of +59 ppm/°F, then extremely close control of the TWT helix-cathode voltage could be achieved. One breadboard of the present series regulator design was measured and found to have a negative temperature coefficient of approximately -30 ppm/°F over the range 30°F ≤ T ≤ 120°F. However, there is some uncertainty associated with the regulator temperature coefficient. This is discussed at greater length in the series regulator paragraphs below.

DC-to-AC Inverter. The inverter, which provides approximately constant filament power, is a conventional saturable core square wave oscillator with a choke in series with the output. Without the choke, the inverter would serve as an ac square wave constant voltage source. The choke serves to increase the source impedance without dissipating power (neglecting the incidental dissipation of the choke). The circuit parameters are chosen such that, at the normal operating point, changes in load resistance will cause approximately equal but opposite changes in voltage and current. A true constant power source would have a hyperbolic volt-ampere characteristic ($P = EI$). The volt-ampere characteristic of this constant power source is made tangent to the desired power hyperbola at the intersection of the inverter E versus I characteristic and the hot filament load line.

A constant voltage source would allow a very large in-rush current to a cold filament since the cold filament resistance is usually much lower than the hot resistance. To ease the thermal

shock, the Syncom 2 filaments were supplied from a constant current source, however, this required a much longer warmup time. The constant power source is a compromise between this thermal shock and the long warmup time. In addition, the most desirable filament supply is one which keeps the filament temperature constant. A constant power source accomplishes this better than either a constant voltage source or a constant current source. This is particularly desirable during the latter life of the tube when the filament hot resistance tends to decrease.

Consider a normalized power system with 1.0 per unit voltage and 1.0 per unit current, yielding 1.0 per unit power dissipated in a 1.0 per unit ohm resistor. If the voltage or current source changes ± 1 percent for a fixed 1.0 per unit ohm resistor, the power will change approximately 2 percent, since $(1.01)^2 = 1.0201$ and $(0.99)^2 = 0.9801$. However, experimental data on the constant power source shows that a ± 1 percent change in input voltage will cause a change in output power of only ± 1.2 percent. The reason for this is that the switching frequency varies linearly with the input voltage, resulting in a linear change in the output choke inductive reactance. As the voltage increases (decreases), the ac current limiting of the choke simultaneously increases (decreases). Therefore, the output power is more a linear function than a square function of the input voltage.

Experimental data also show that for ± 16 percent resistance changes about the normal operating point, the product of rms voltage and current of the constant power source will remain within ± 1 percent. The combined effect of ± 1 percent voltage changes and ± 16 percent resistance changes is contrasted below for a constant voltage, constant current, and constant power source.

Source	Power Variation percent
Constant voltage	± 18
Constant current	± 18
Constant power	± 2.2

The first breadboard inverter has been operated satisfactorily from -100 to 160°F . Figure 5-47 represents the room temperature performance of this supply. Figure 5-48 is a suppressed origin blowup of the constant power region, showing the effect of temperature variations on performance. Figure 5-49 is derived from Figure 5-48 and shows the power versus temperature characteristics for various filament resistances. Figure 5-48 indicates that at room temperature the power regulation

due to load resistance will be ± 1 percent for a load between 73 and 110 percent of the rated hot filament resistance. This data also assumes a constant -24 volt input. It has been determined that a 1-percent increase (decrease) in output power will result from a 0.8-percent increase (decrease) in input voltage. Figure 5-49 shows the power regulation with temperature to be ± 2 percent (at 1.0 per unit load and 1.0 per unit input voltage) over the temperature range $+37 < T < +111^{\circ}\text{F}$. The power temperature coefficient is quite linear at -540 ppm/ $^{\circ}\text{F}$ over this range. The power supply has an efficiency at rated load of 80 percent at 30°F , 79 percent at 75°F , and 78 percent at 120°F . The present output choke is dissipating approximately 6 percent of the output power. By doubling the wire cross section (and increasing the core size as required) this choke incidental dis-

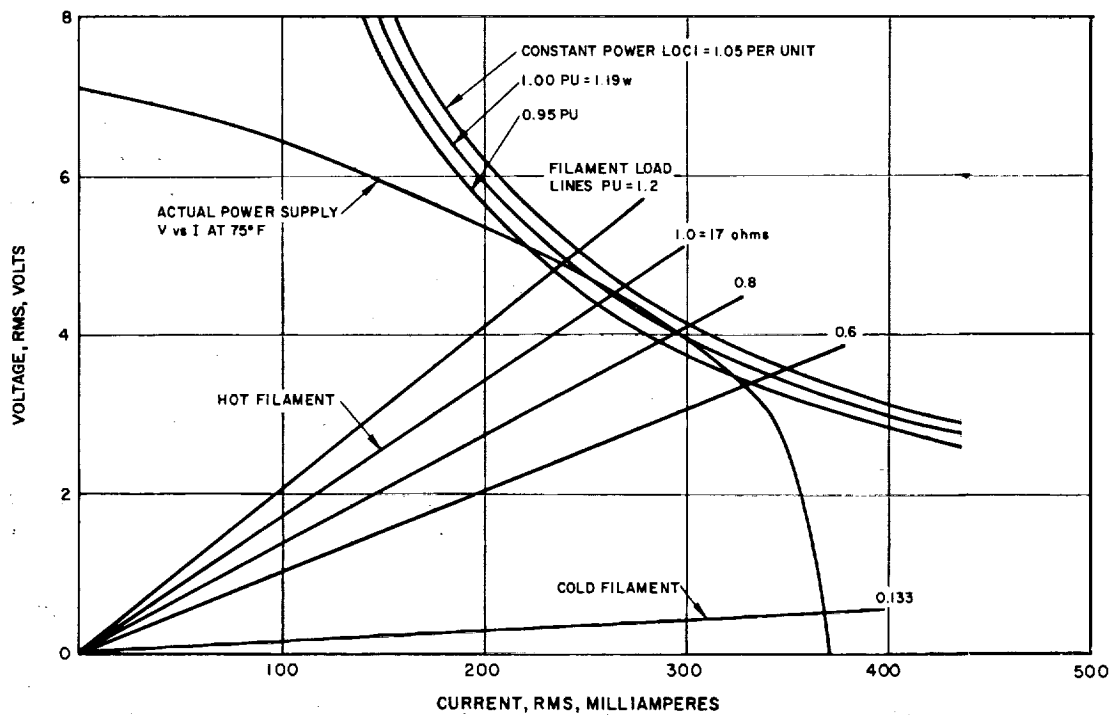


FIGURE 5-47. CONSTANT POWER FILAMENT SUPPLY PERFORMANCE

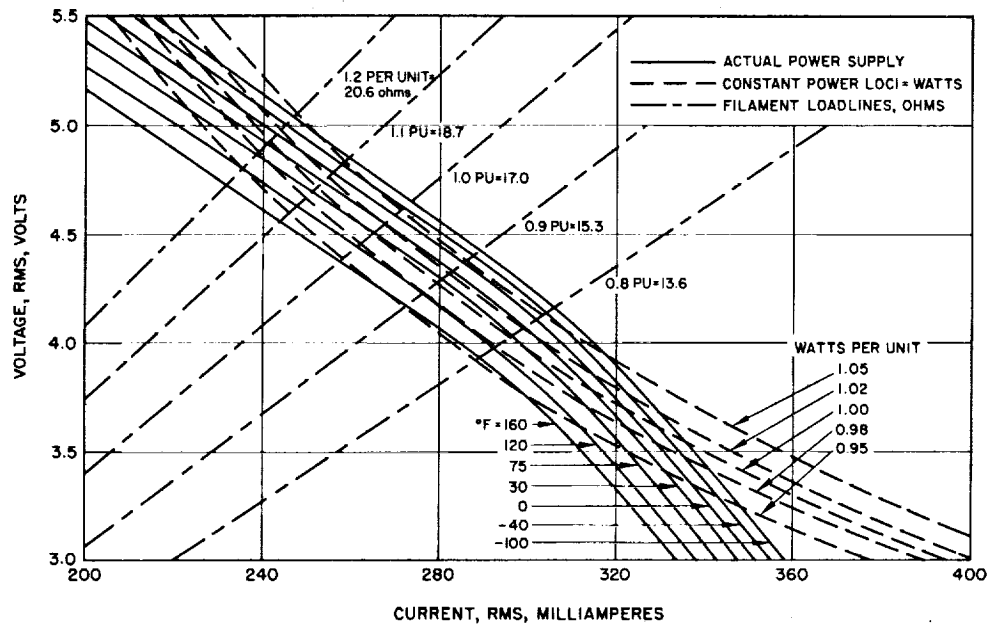


FIGURE 5-48. CONSTANT POWER FILAMENT SUPPLY TEMPERATURE EFFECTS

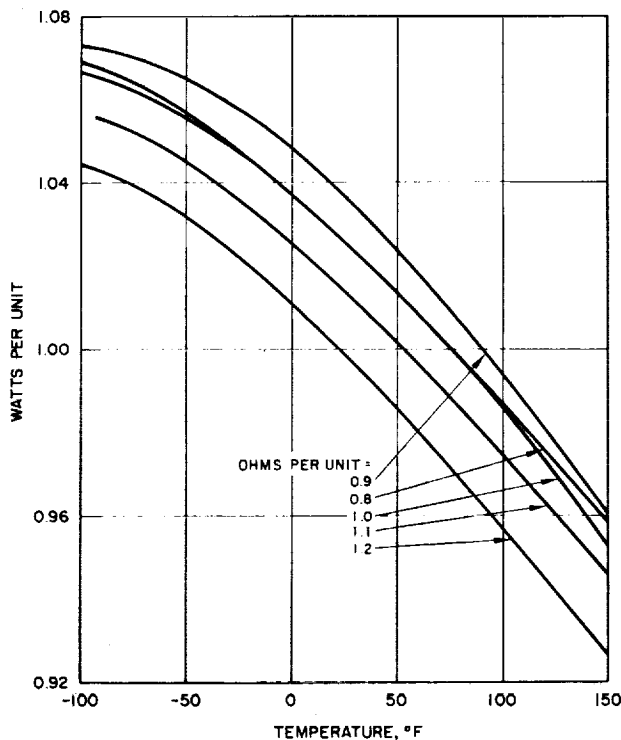


FIGURE 5-49. CONSTANT POWER FILAMENT SUPPLY POWER VERSUS TEMPERATURE FOR NOTED FILAMENT RESISTANCE

sipation could be reduced to approximately 2 percent of the output, increasing the efficiency 4 percent.

In comparing the constant power filament supply with the Syncom 2 constant current supply, the former has fewer parts, higher efficiency, and quicker warmup time. A quantitative comparison appears in Table 5-8. The lower parts count improves the overall reliability of the unit. A higher efficiency results from employing a reactor in the ac portion of the circuit for current limiting, instead of resistors and transistors in the dc section as is done in the constant current supply. A quicker warmup results from allowing more current to the cold filament. However, the inrush current is limited to 1.4 per unit for the cold filament resistance of 0.13 per unit.

Series Regulator. A series regulator is employed to supply -24 volts to the TWT high voltage converter and filament inverter. An aux-

TABLE 5-8. POWER SUPPLY COMPARISON

Supply	Constant Current	Constant Power
Total parts	22	14
Efficiency, percent	64	79
Warmup time, seconds*	46	28

*TIME REQUIRED TO REACH 90 PERCENT OF HOT FILAMENT VALUE.

iliary function is to operate the ferrite switch when the regulator is turned on. The regulator turns on and off in response to commands from the command decoder. The normal sequence of commands and the resulting response is as follows:

- 1) Filament ON: This command turns on the series regulator with the following results:
 - a) A current pulse of approximately 1 ampere for 5 milliseconds operates the ferrite switch.
 - b) The filament inverter is automatically turned on.
 - c) The converter remains off.
- 2) HV ON: This command starts the high voltage dc-to-dc converter. Sufficient time is allowed for the TWT filament to warm up before the converter is started.
- 3) TWT OFF: This command turns off the series regulator which stops both the converter and filament inverter.

This series regulator is quite similar to the one used in Syncom 2. Two features which have been added are:

- 1) An extra stage to the regulator loop by replacing the 2N2151 control element with a Darlington composed of a 2N1724 driven by a 2N1717. A 50-percent increase in steady state load current plus

TABLE 5-9. TRAVELING-WAVE TUBE POWER SUPPLY LOAD ANALYSIS

Supply	TWT, watts	Efficiency, percent	Series Regulators	
			Watts	Milli-amperes
Converter	13.3	84	15.84	661
Filament inverter	1.2	79	1.52	63
Series regulator total load			17.36	724

the 1-ampere pulse for the ferrite switch necessitated this change.

- 2) An improved overcurrent limit circuit which is independent of unregulated bus voltage. The Syncom 2 method of limiting output current by limiting the control element base drive would allow a 2:1 increase in current for an unregulated bus variation from -26 to -36 volts. The increased Advanced Syncom bus capacity can allow -34 volts with the TWT on if other loads are off. The control element dissipation would be excessive if the Syncom 2 method were employed.

The maximum allowable TWT input, including filament power, is 14.5 watts. With 1.2 watts for filament power the converter load can be 13.3 watts. Neglecting the ferrite switch, the steady state load on the series regulator is given in Table 5-9.

There will be approximately 6 milliamperes in the regulator control circuits, yielding a total load of 730 milliamperes on the unregulated bus. This current is independent of regulator input voltage. Table 5-10 represents input, loss, and efficiency data at various assumed unregulated bus voltages for both the series regulators and the total TWT power supply.

The first breadboard regulator exhibited a total output voltage change of 20 millivolts from

TABLE 5-10. INPUT, LOSS, AND EFFICIENCY DATA*

Input volts	-26	-30	-34
Output volts	-24	-24	-24
Control element, volts	2	6	10
Input watts at 730 milliamperes	19.0	21.9	24.8
Efficiency, percent	91.4	79.2	70.0
Losses, watts	1.64	4.54	7.44
TWT supply efficiency, percent	76.3	66.2	58.5
TWT supply loss, watts	4.5	7.4	10.3

*Series regulator only, unless TWT supply is specified.

no load to full load, or 0.083 percent. This corresponds to an output resistance of $R_o = 0.036$ ohms. With full load the output voltage varied 2 millivolts or 0.0083 percent, for an input voltage variation of from -26 to -36 volts. The above results were at 75°F. With a -26 volt input and full load, the output voltage changed 64 millivolts from 30 to 120°F with a linear negative temperature coefficient of -30 ppm/°F. Figure 5-50 represents the room temperature volt-ampere characteristic of this regulator, and Figure 5-51 represents the effect of temperature on no load and full load voltage, output impedance, and the overcurrent limit. Figure 5-50 illustrates the regulator performance with overload. The overcurrent feedback loop overrides the voltage feedback loop, and the device becomes a constant current source rather than a constant voltage source.

The PARD ((Periodic And Random Deviations) characteristics of the regulator are quite good. With a 1.0 per unit resistive load, the PARD was approximately 2 millivolts peak-to-peak. With the actual switching loads operating, the output voltage perturbations increased. The combination of PARD plus dynamic load regulation on the -24 volt output of the series

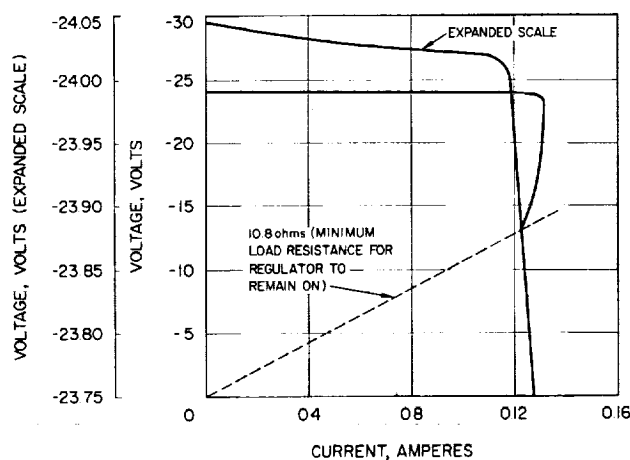


FIGURE 5-50. REGULATOR VOLT-AMPERE CHARACTERISTICS

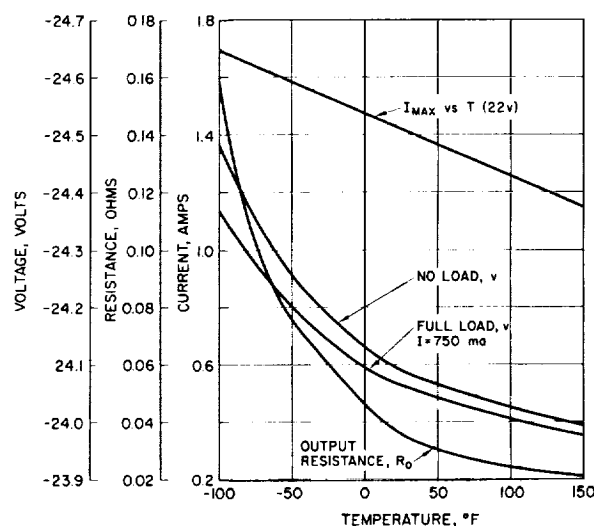


FIGURE 5-51. REGULATOR TEMPERATURE CHARACTERISTICS

regulator was 6 millivolts peak-to-peak for the filament inverter only and 30 millivolts peak-to-peak for both the filament inverter and dc-to-dc converter.

The regulator operated satisfactorily over the range $-100 < T < 160^\circ\text{F}$. As indicated above, it exhibited a voltage temperature coefficient of $-30 \text{ ppm}/^\circ\text{F}$ over the range $+30 < T < 120^\circ\text{F}$.

However, this overall regulator temperature coefficient is dependent primarily on the positive temperature coefficient of the zener reference cancelling the effect of the first stage transistor base-emitter junction negative temperature coefficient. The TWT helix-cathode voltage should be held to ± 1 percent for all variables except load changes (the load is quite constant). In the converter discussion above, it was determined that the series regulator should have a voltage temperature coefficient in the range $+59 \pm 178$ ppm/ $^{\circ}\text{F}$, and that a coefficient of $+59$ ppm/ $^{\circ}\text{F}$ would cancel the inherent temperature coefficient of the converter. Because of the uncertainties associated with the present reference circuit, a temperature compensated circuit consisting of a temperature compensating zener reference and a differential amplifier comparator should be used in this regulator. This decision was reached too late to be incorporated in the present design, however, the multiple access transponder regulator has demonstrated the capability of this circuit and no difficulties are anticipated. By the addition of one sensistor ($+7000$ ppm/ $^{\circ}\text{C}$ resistor) of appropriate size to the voltage divider, the desired positive temperature coefficient can be realized.

Ferrite RF Power Switch. The ferrite switch utilized to switch the selected traveling-wave tube to the multiplexer input is essentially a three-port circulator with provisions for reversing the magnetic field polarity (Figure 5-52). In this configuration, the circulator operates as a single-pole double-throw switch.

This switch requires a pulse of 21 watts of power for a measured microwave switching speed of about 2.5 milliseconds. The microwave characteristics over a 200 mc bandwidth are 0.2 db maximum insertion loss between transmission terminals and 20 db minimum isolation.

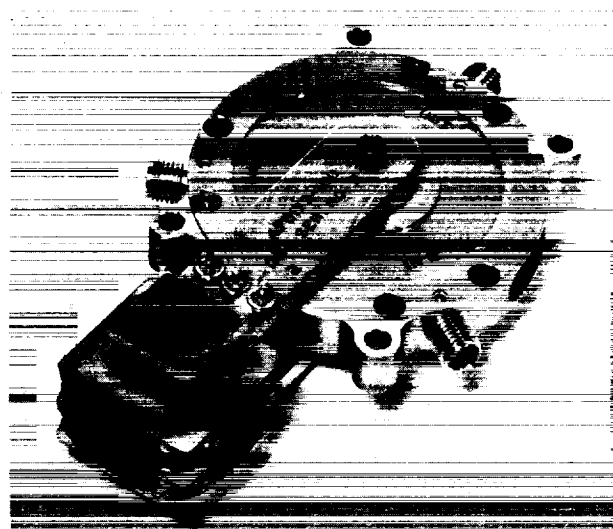


FIGURE 5-52. RF TRANSMITTER SWITCH

Power is needed only to reverse the ferrite magnet polarity; once this is accomplished, no further power is required.

Low insertion loss and switching power were the main design objectives. Assuming optimum circulator design, the limiting factor for lowest possible insertion loss is determined by the dielectric and magnetic losses of the garnet. For low switching power, the number of ampere turns per inch around the ferrite magnet must be a minimum, which requires a low coercive force ferrite magnet with good square loop properties. For the switch to operate with a low coercive force magnet, the internal garnet must have a low saturation magnetization and airgaps in the magnetic circuit must be minimized.

The airgap in the magnetic circuit was decreased to 0.010 inch by using soft iron wherever possible. The lowest saturation magnetization garnet was used consistent with low dielectric and magnetic losses and temperature stability. A coercive force of 275 oersteds and residual magnetism of 2040 gauss was specifi-

cally synthesized for use in this switch. The proper flux density was obtained by varying the cross-sectional area of the magnet and by winding 600 turns of No. 33 wire around the magnet. The switch requires 0.87 ampere at 24 volts.

Engineering models of the RF switch have been constructed. Insertion loss over the band through the switch was measured as 0.15 db for either switch position. Isolation between the two input ports was better than 23 db over the total transmitter band.

3 db Hybrid. The 3 db hybrid is a four-port stripline device connected between the outputs of the multiple access and frequency translation sections of the transponder and the traveling-wave tubes. Its purpose is to split the RF output from the operating transponder section and supply equal RF drive to both traveling-wave tubes, only one of which has both drive and dc power applied. All input and output impedances are 50 ohms.

The unit weighs 1 ounce including 0.5 ounce of OSM connectors. The insertion loss from either of the two input ports to each of the two output ports is 3.0 ± 0.5 db at 3980 mc and 4200 mc. The isolation between the two input ports or the two output ports is at least 20 db at 3980 mc and 4200 mc.

Telemetry and RF Coupler. The dual coupler is a passive RF unit between the traveling-wave tubes and the transmitter multiplexer. The dual coupler provides 1/100 of the transmitter output to a detector for telemetry monitoring purposes and also provides a means of testing the RF output of the communications transmitter. The detector is a coaxial diode that changes the RF output sample to a negative dc voltage for use by the telemetry encoder.

Units have been constructed for the engineering model. The measured insertion loss is 0.2 db.

The input VSWR is 1.18 and the coupling is -20.8 db to the telemetry detector and -23.8 db to the RF output connector. With 4 watts output from the traveling-wave tube, the detector output will be -2.8 volts.

COMMUNICATION ANTENNA SYSTEM

Phased Array Antenna System. The complete phased array antenna system is shown in block form in Figure 5-53. A power splitter divides the transmitter output signal into eight equal amplitude, equal phase parts. The outputs from the splitter are applied to eight phase shifters, each consisting of an input coupler, ferrite section, and output coupler. The ferrite sections consist of a tube of ferrite in a circular waveguide inside a four-pole, two-phase electromagnetic field coil. The output couplers convert the RF output of the ferrite sections into two equal amplitude signals with different phase shifts and feed the signals to diametrically opposite antenna elements. Sixteen antenna elements are symmetrically arranged about the spin axis on a circle of one wavelength radius. All of the elements are continuously driven, but are individually phased by the phase shifters in such a way that the radiation tends to reinforce in a given direction. The phase shifters are electronically controlled by the PACE unit.

To facilitate testing and to optimize the final antenna parameters, the 4-kmc transmitting antenna was first constructed in a breadboard version. The input power divider was made in stripline with TNC connectors. The input coupler was a coaxial transition from a TNC connector to a circular waveguide. The ferrite sections were located inside the circular coil windings. The output coupler used two TNC coaxial-to-waveguide transitions. Coaxial cables connect the output couplers to the antennas, with each

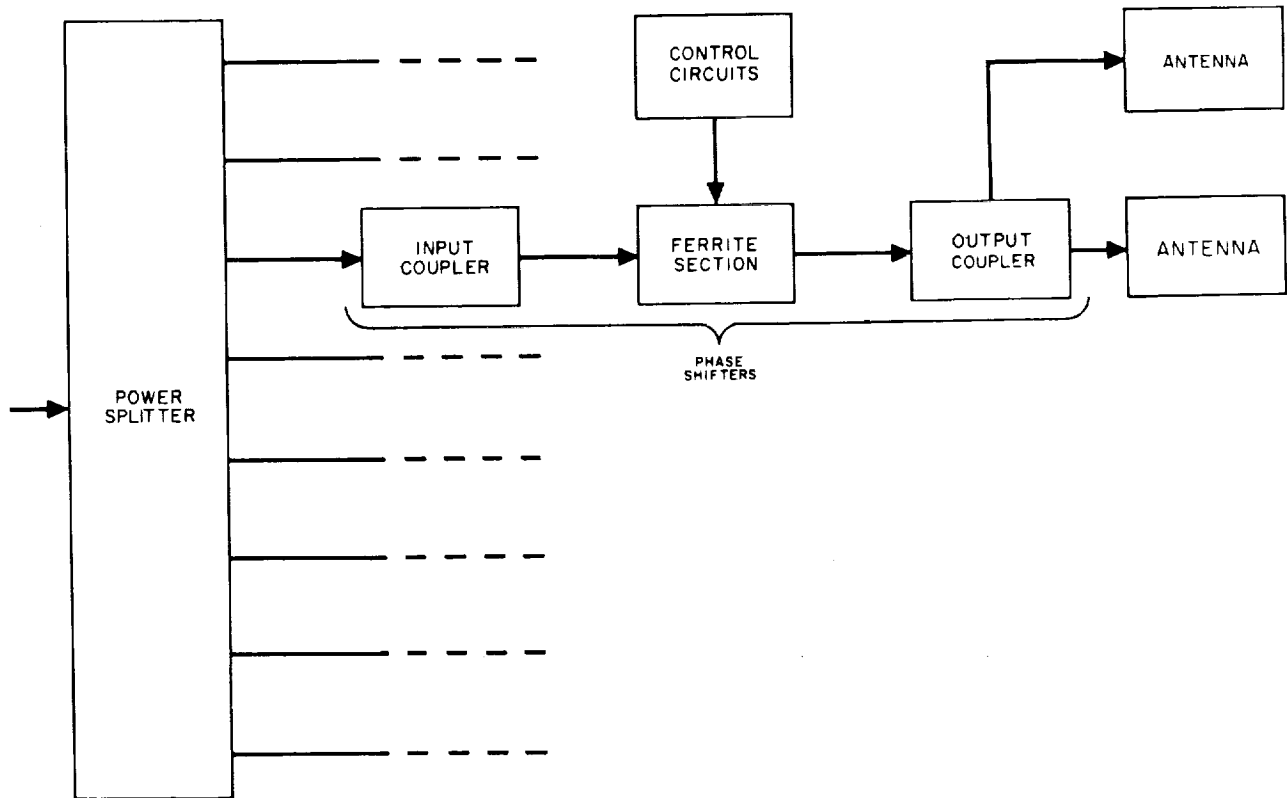


FIGURE 5-53. PHASED ARRAY ANTENNA SYSTEM BLOCK DIAGRAM

output coupler connected to two diametrically opposite elements. The final stripline version of the antenna is shown in Figure 5-54. This version uses stripline-to-waveguide couplers for both the input and output couplers and stripline rather than cables to feed the antenna elements. The striplines have the advantages over coaxial interconnections of fewer connectors, better coupling to and from the phase shifters, better control of line lengths, and lower losses. The hole in the center of the array is for installation of a receiving antenna, which is not shown.

The following paragraphs discuss the various parts of the antenna and status of each part.

Power Divider. The power divider splits the transmitter signal into eight equal amplitude, in-phase signals to drive the eight phase shifters.

The divider is fabricated in stripline and consists of seven identical hybrid rings, each ring being similar to a waveguide ring or "rat-race." The power divider, or splitter, breadboard configuration is shown in Figure 5-55 and the redesigned configuration in Figure 5-56. The stripline board shown in Figure 5-56 incorporates the input couplers to the ferrite sections through a direct stripline-probe to waveguide transition. Figure 5-57 is a close-up view of the coupler probes. The redesigned stripline dielectric is Duroid of $\frac{1}{32}$ inch thickness and the ground plane spacing is 0.147 inch. Line widths are chosen to give a characteristic impedance of 50 ohms. Figure 5-58 shows the divider before the drilling was completed and without the input coupler probes.

Assuming all arms of the rings "see" matched

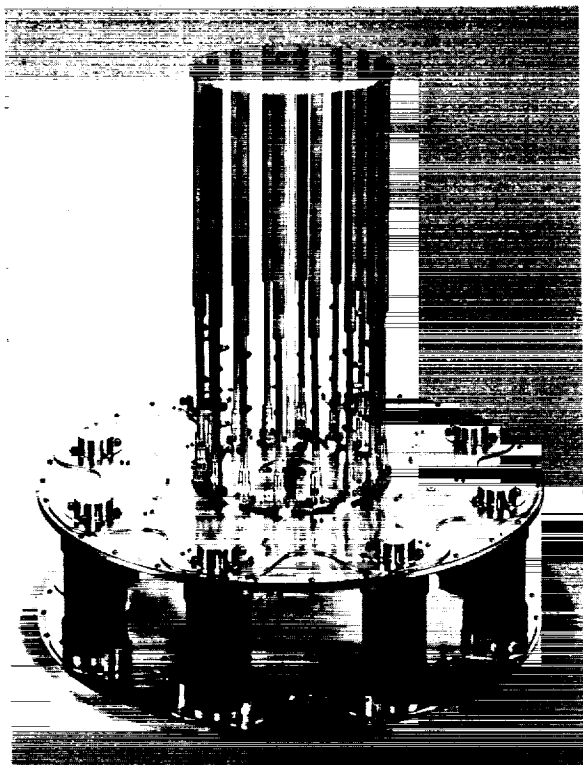
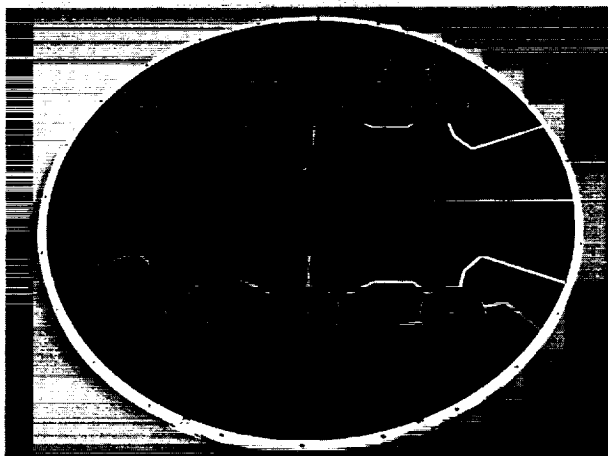
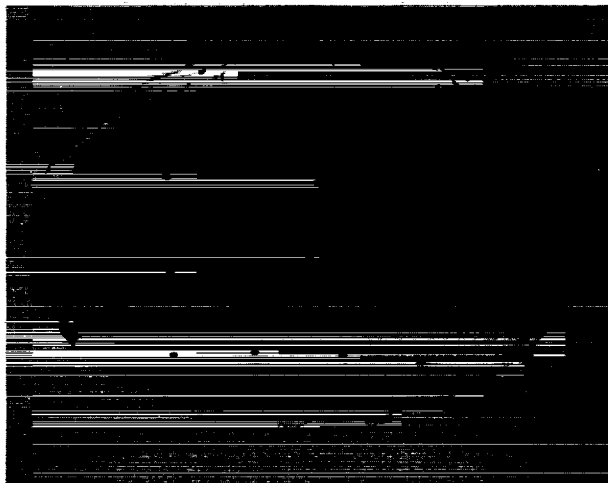


FIGURE 5-54. STRIPLINE VERSION OF PHASED ARRAY TRANSMITTING ANTENNA

loads, an input signal at arm 3 (or 2) splits into two equal amplitude, in-phase signals at arms 2 and 4 (or 1 and 3) with nothing appearing at arm 1 (or 4). An input to arm 4 (or 1) produces equal amplitude, out-of-phase signals at arms 1 and 3 (or 2 and 4) with nothing appearing at arm 2 (or 3). In Figure 5-56, the input at each ring is applied through arm 3. Inputs at arms 2 and 4 would be by reflection from the connecting arm and hybrid. The unused arms (1) of the hybrids are terminated in resistive loads to absorb any power which may be reflected back into that arm. To ensure equal phase shifts to all eight outputs of the power divider, symmetrical connections are used and the hybrids are cascaded so that each of the eight signals traverses the same path length from input to output.

A systematic approach began with constructing

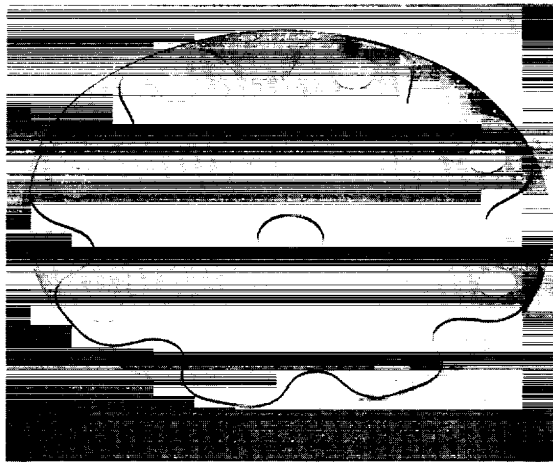


b) Complete Unit

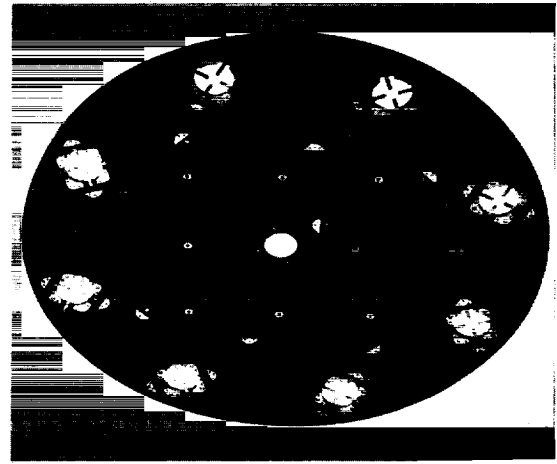
FIGURE 5-55. BREADBOARD POWER DIVIDER

and optimizing an individual stripline hybrid ring (Figure 5-59). The operation of the ring, which was used in the divider shown in Figure 5-56, is excellent over the normal operating band of about 3980 to 4200 mc.

Phase Shifters. As the antenna rotates, the angles θ_n vary, and therefore the required phase shifts vary. Diametrically opposite elements have opposite sense phase shifts. (They are not of opposite phase in the usual sense, i.e., 180 degrees apart.) For an electrical radius R equal

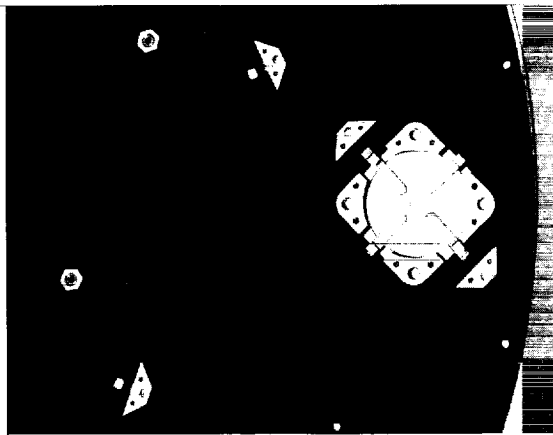


a) Top View of Stripline Board

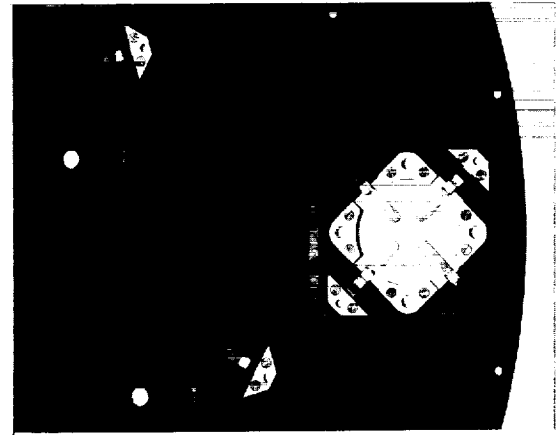


b) Top Ground Plane

FIGURE 5-56. REDESIGNED STRIPLINE POWER DIVIDER



a) Top View



b) Bottom View

FIGURE 5-57. INPUT COUPLER PROBES

to one wavelength, the resultant phase shift versus time is

$$\phi_n = -\phi_{n+8} = 2\pi \cos(\omega t + n \cdot 22.5 \text{ degrees})$$

where ω is the rotation rate of the antenna and 22.5 degrees is the angular separation of the antenna elements (equal to 360 degrees/16).

The ferrite phase shifters are designed to produce this required phase shift and provide the

two opposite polarity phase-shifted signals. The design is based on the adjustable phase changer originally described in the article "Adjustable Waveguide Phase Changer," by A. Gardner Fox, published in the December 1947 issue of the *Proceedings of the IRE*. The phase shift is made in circular waveguide, with the input coupler exciting a TE_{11} mode in the guide. This is essentially plane polarized in the plane of the input probe. The ferrite section, with magnetic field



FIGURE 5-58. POWER DIVIDER STRIPLINE
BEFORE DRILLING

applied transversely by means of the external "motor" winding, acts as a "half-wave plate" which rotates the plane of polarization. The angle of polarization rotation is equal to twice the angle of field rotation. Thus unlimited phase shift can be achieved with no moving parts.

The output coupler consists of a slow wave structure in one plane, acting as a quarter-wave plate, followed by two orthogonal coupling probes. The output coupler converts circular polarization to linear polarization. The plane-polarized wave from the ferrite section can be thought of as two counter-rotating circularly polarized waves. After passing through the slow wave structure, these components are converted to orthogonal plane-polarized signals which couple to the two output probes. Thus rotation of

the polarization in the ferrite section is equivalent to opposite polarity phase shifts of the two counter-rotating circular components and therefore to opposite polarity phase shifts at the two output probes. Thus, rotation of the applied magnetic field results in phase shifts of the RF outputs. The use of the two-phase motor stator as an electromagnetic field coil counteracts the factor of two between field rotation and polarization, so that output phase shift corresponds directly to the rotation of the voltages applied to the motor.

The dimensions of the ferrite section are the determining factor in the size of the phase shifters. After discussions with the ferrite manufacturer, Trans-Tech, Inc., a number of batches of ferrite which they felt would work at 4 kmc were made. These were tested in various configurations of inner and outer diameter and checked for loss over the band and polarization rotation sensitivity. Eventually a configuration was evolved in which the ferrite was 2.5 inches long, with an outer diameter of 1.15 inches and an inner diameter of 0.625 inch. The inner tube was filled with teflon, which helped to reduce the low cutoff frequency. With this diameter, the waveguide is cut off in air at 4 kmc and requires a dielectric constant of about 3 or more to transmit. The ferrite section had less than 1 db loss in the test fixture.

The motor stators used to provide the magnetic field were purchased from the Skurka Langdon Manufacturing Company. Initially, these required about 100 milliamperes at 6 volts to provide the 180-degree differential phase shift in the ferrite necessary to make it a pure polarization rotator. Subsequently, in the breadboard configuration, the motor was redesigned with more turns of finer wire, resulting in a requirement of about 40 milliamperes at 20 volts.

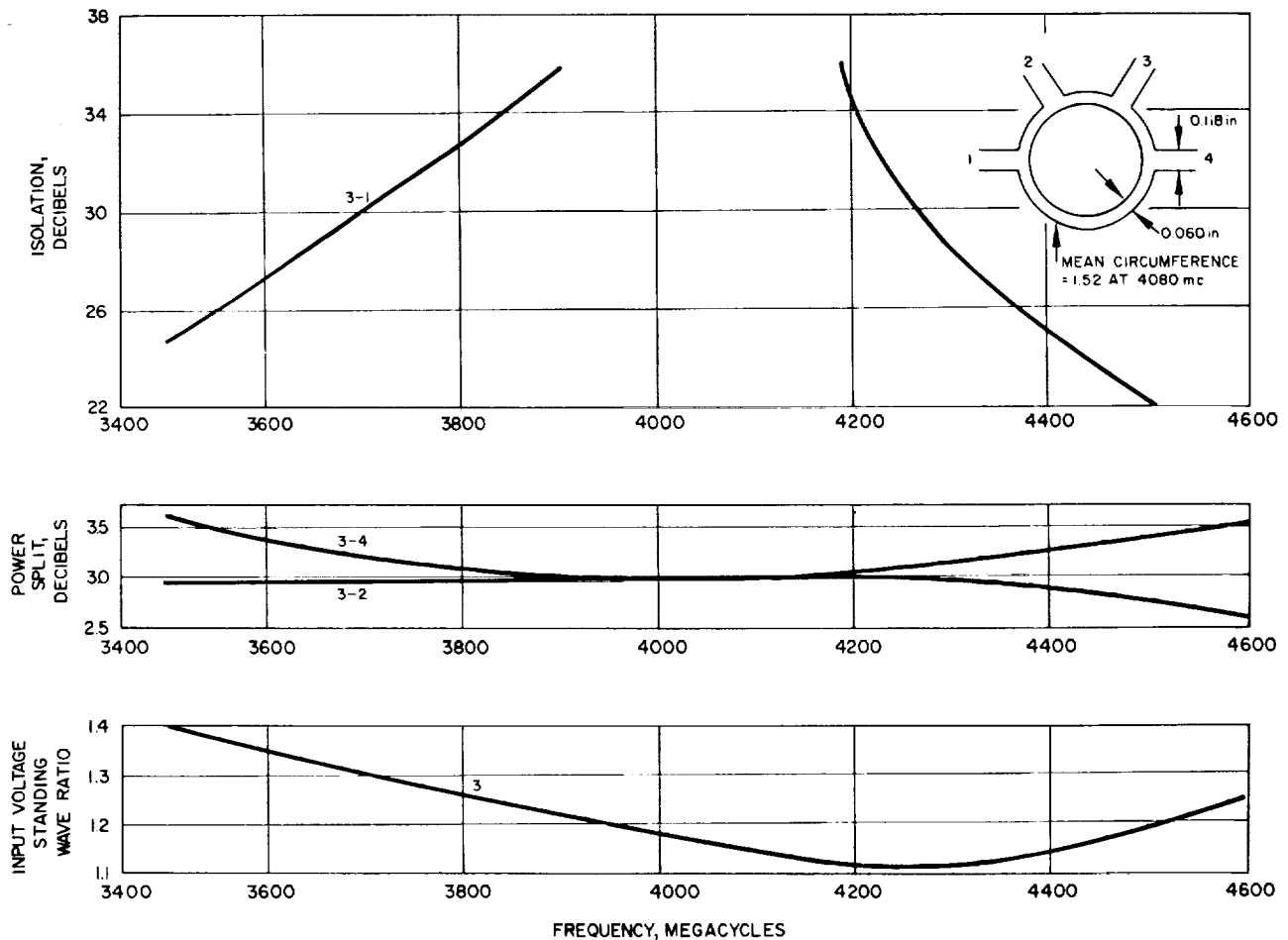


FIGURE 5-59. CHARACTERISTICS OF REDESIGNED HYBRID RING

A series of temperature tests was made on the phase shift properties of some of the breadboard phase shifters. The test setup is shown in Figure 5-60. A 12-position switch was used to generate the two phase voltages of the form $\sin \theta$ and $\cos \theta$, where θ varied from 0 to 360 degrees in 30-degree steps. A slotted line was used to measure phase, while isolators were employed to maintain a constant load on the phase shifter. The field winding was adjusted so that outputs 1 and 2 were in phase when $\theta = 0$ degrees. The input angle was then varied and the output phase, θ_0 , was measured. The difference between θ_0 and a linear

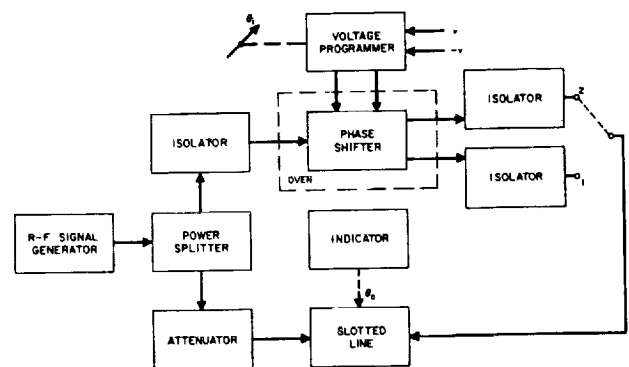


FIGURE 5-60. TEST SETUP FOR MEASURING BREADBOARD PHASE SHIFT VERSUS TEMPERATURE

output was calculated (see Figure 5-61). Five phase shifters were tested at 0°C, 25°C, and 50°C at 4170 mc. The phase difference (with arbitrary zero position) was plotted vertically against input angle. A number of factors were apparent from these curves. All the phase shifters varied about the same with temperature, increasing in length about 80 degrees when the temperature was raised from 25°C (room temperature) to 50°C and decreasing about 180 degrees when the temperature was lowered to 0°C. There was a variation in the deviation of output from the desired linear relationship amounting to about 10 degrees

rms. The reason for the variation is not known but could be nonsymmetry of field windings, errors in control voltage, error in the measuring procedure, residual magnetization in the ferrites, or misadjustment of the output couplers. It was also observed, but not recorded, that the amplitude of the phase shifters fluctuated as the input voltages were varied. It is expected that some of the problems will be alleviated by the stripline circuit. The general method of measurement appears to be satisfactory.

- Stripline Component Design. In the stripline version of the phase shifter, the input

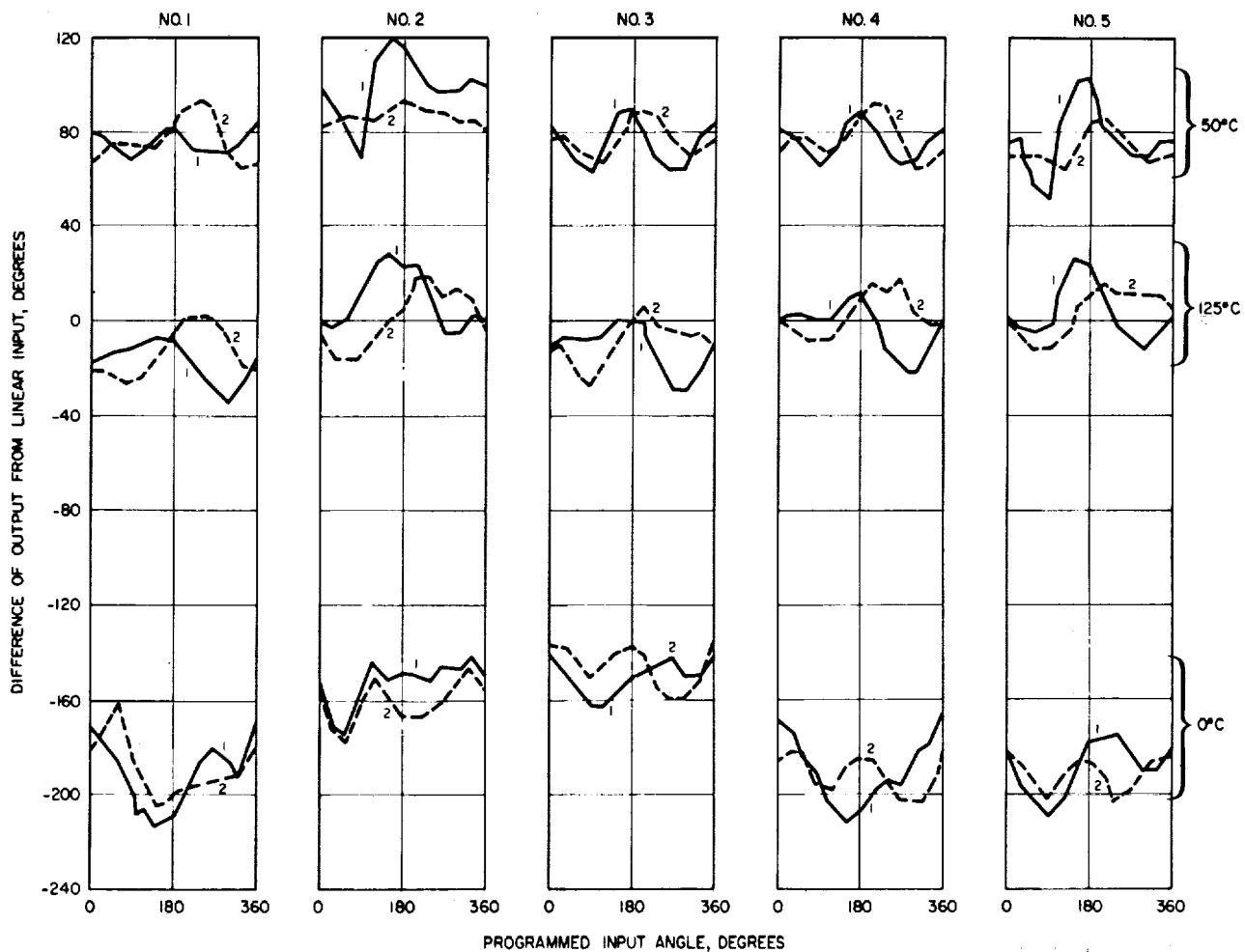


FIGURE 5-61. BREADBOARD PHASE SHIFTER PERFORMANCE VERSUS TEMPERATURE

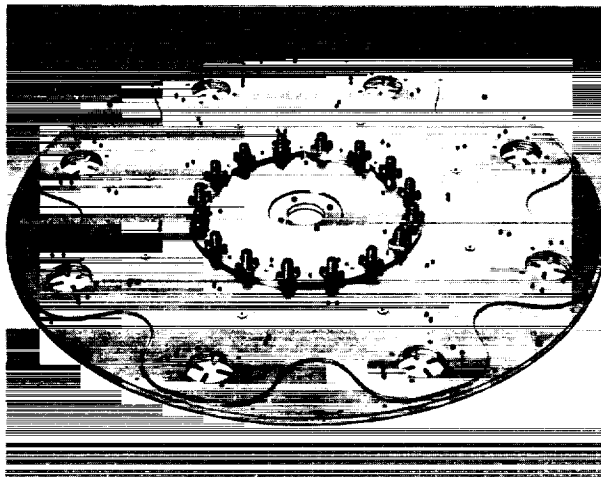
couplers are combined with the power divider stripline through a direct stripline-probe to waveguide transition (Figures 5-56 and 5-57). The output coupler is replaced by a four-probe coupler (or turnstile), a balun for coupling opposite probes 180 degrees apart, and a square 3-db 90-degree hybrid. The feed lines to the antennas are also in stripline. Because of the geometry involved, there are a number of points where two striplines cross over. The circuits are therefore made in two planes with vertical transitions connecting them where necessary. The dielectric used

is Duriod of $\frac{1}{32}$ inch thickness, while the ground plane spacing is 0.147 inch. Line widths are chosen to give a characteristic impedance of 50 ohms.

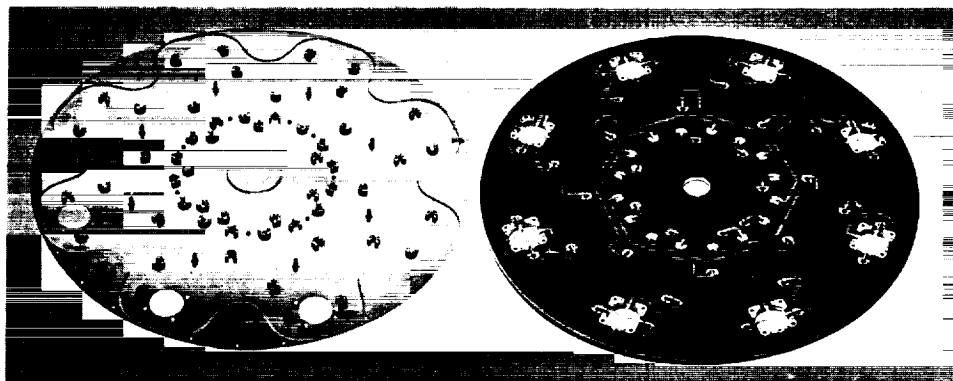
The completed stripline phase shifter output coupler and ground planes are shown in Figure 5-62. The ground planes are numbered consecutively from 1 to 4. The coupler is shown in various stages of disassembly. Starting with the complete assembly, components are consecutively removed from the right assembly (top views) and reassembled on the left (bottom views). A part of the bottom circuit board is shown in detail in Figure 5-63 and the top circuit board is shown in Figure 5-64.

Special test fixtures were fabricated to optimize the vertical stripline-to-stripline transition, the vertical TM-to-stripline transition, the 90-degree hybrid, and the four-probe coupler. An interim model of the hybrid was fabricated to optimize the dimensions.

In the 90-degree hybrid an input signal divides equally into two outputs 90 degrees apart. Thus, when such a hybrid is connected to the turnstile junction and an input signal is applied to the hybrid, a circularly polarized signal is set up in

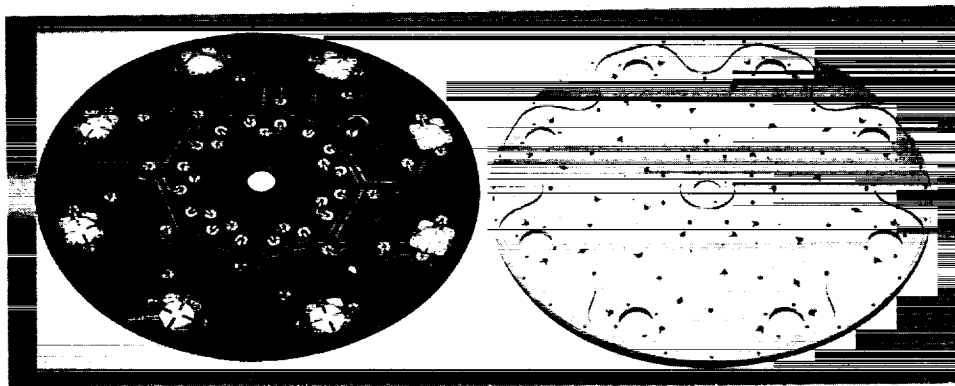


a) Complete Assembly

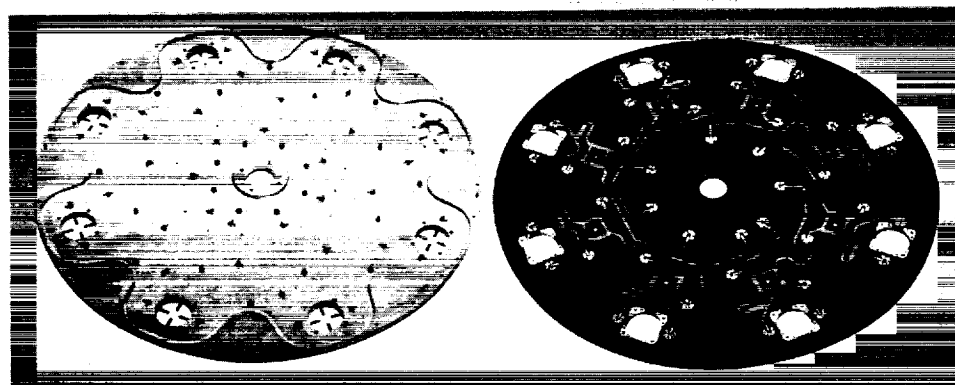


b) Left: Bottom View, Ground Plane 1
Right: Stripline Board, Top View

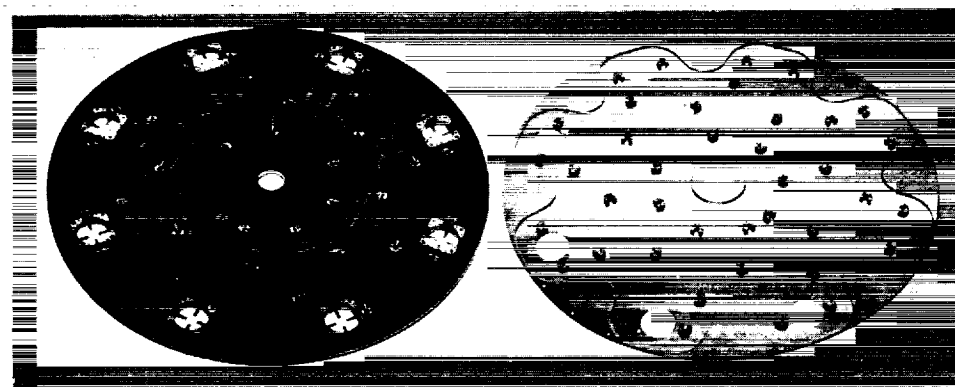
FIGURE 5-62. STRIPLINE OUTPUT COUPLER



c) Left: Top Stripline Board, Bottom View
Right: Ground Plane 2, Top View

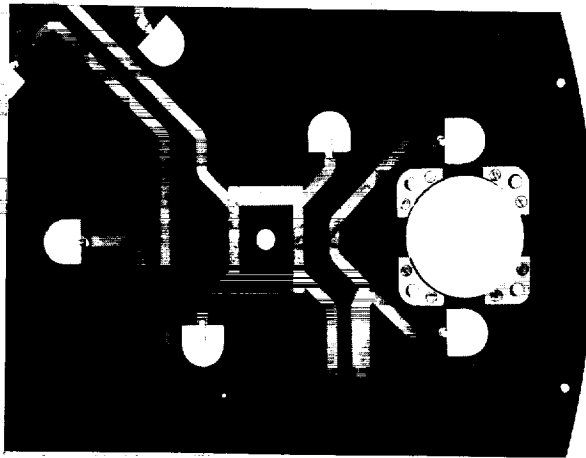


d) Left: Ground Plane 3, Bottom View
Right: Bottom Stripline Board, Top View

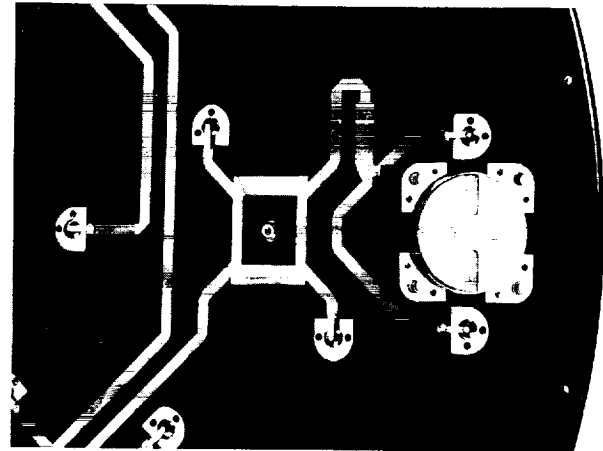


e) Left: Bottom Stripline Board, Bottom View
Right, Ground Plane 4, Top View

FIGURE 5-62 (CONTINUED). STRIPLINE OUTPUT COUPLER

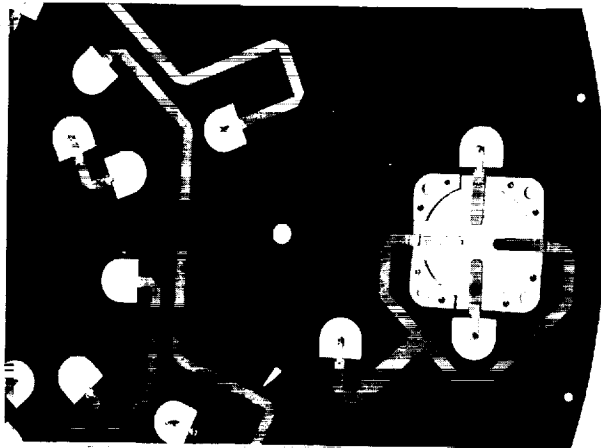


a) Top View, without Ground Plane

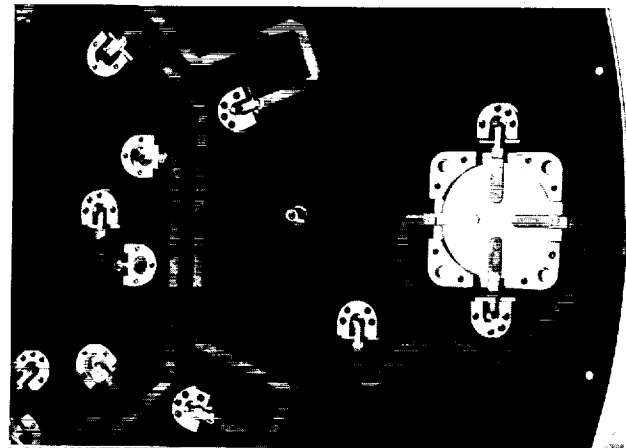


b) Bottom View, with Top Ground Plane and Top Board

FIGURE 5-63. BOTTOM STRIPLINE CIRCUIT BOARD



a) Bottom View



b) With Top Ground Plane

FIGURE 5-64. TOP STRIPLINE CIRCUIT BOARD

the waveguide. If the signal is applied to the other arm of the hybrid, a circularly polarized signal of opposite sense is set up. Since the operation is reciprocal, circularly polarized signals of opposite sense in the waveguide will couple separately to the two hybrid output terminals.

The four ground planes used in the output coupler are die-stamped identically to the ground planes for the power divider, although the hole patterns are different. To ensure proper registra-

tion, the ground planes and circuit boards are drilled identically with drill fixtures.

To match the control circuitry, the field coils for the ferrite sections were changed to coils with half the number of turns with twice the original wire diameter. Thus, the new coils operate at twice the current and half the voltage of the original coils. This permits design of the power amplifier circuits using nearly the optimum operating range for power transistors.

The assembled stripline circuits and ferrite sections are shown in Figure 5-65 and a simplified cross-section diagram is shown in Figure 5-66. The eight ferrite sections are mounted vertically between the stripline circuits. The bottom circuit contains the eight-way power divider and input couplers to the ferrite section, while the upper circuits include the output couplers from the ferrite section, the 90-degree hybrids, and the feed lines to the bases of the

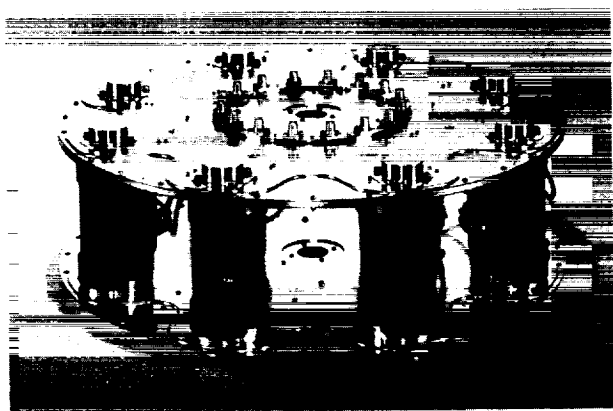


FIGURE 5-65. ASSEMBLED STRIPLINE CIRCUITS AND FERRITE SECTIONS

antennas. Two ground planes are used with the bottom circuit board and four with the upper circuit boards.

Preliminary Measurements — Stripline Version of Phased Array Antenna. Initial testing of the electrical characteristics of the stripline phased array indicates that the performance is generally as expected. Results obtained during the remaining contract period will be included in Volume 6 of this report. When connected to the breadboard demonstration PACE system, a well shaped beam was formed. A special voltage controller test box was fabricated to supply the required field coil voltages and control them by means of a manual selector switch. In this way the beam can be manually positioned in any of 32 different directions. This facilitates pattern and gain measurements. The fiberglass structure was also completed and installed on the array as shown in Figures 5-67 and 5-68. A limited number of patterns were taken on the roofhouse antenna range, with some of the results shown in Figure 5-69 which shows plots of the beam shape,

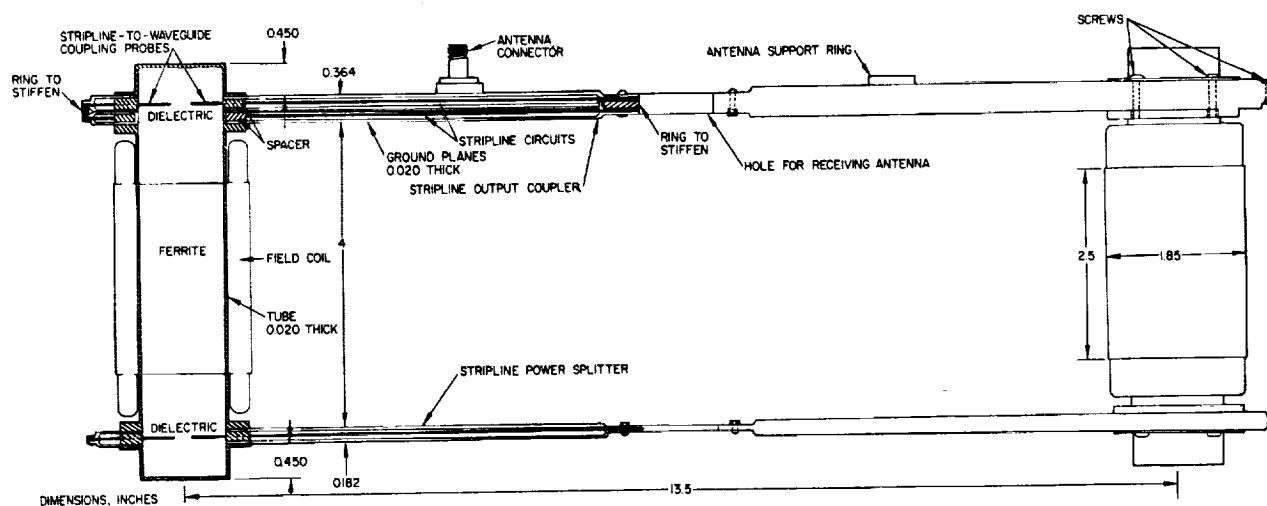


FIGURE 5-66. CROSS SECTION OF STRIPLINE PHASE SHIFTERS AND POWER DIVIDER ASSEMBLY



FIGURE 5-67. COMPLETE PHASED ARRAY ANTENNA

and Figure 5-70, which shows the relative gain variation with main beam direction.

Measurement of the gains of the individual antennas, also measured on the roofhouse range, are tabulated in Table 5-11. The theoretical gain at design center frequency is just under 8 db. They are all matched to better than 1.5 VSWR so the reduced gain of 1 to 2 db is believed to be due to loss in the matching sections in the antennas.

Measurement of the losses through the phase shifters and stripline indicates the loss to be between 1.5 and 2 db. Although the relative outputs of the 16 antenna connectors vary considerably with the magnitude of the voltage applied to the field windings, the measured main beam gain varied less than 1 db for over 20 percent variation in the supply voltages to the controller.

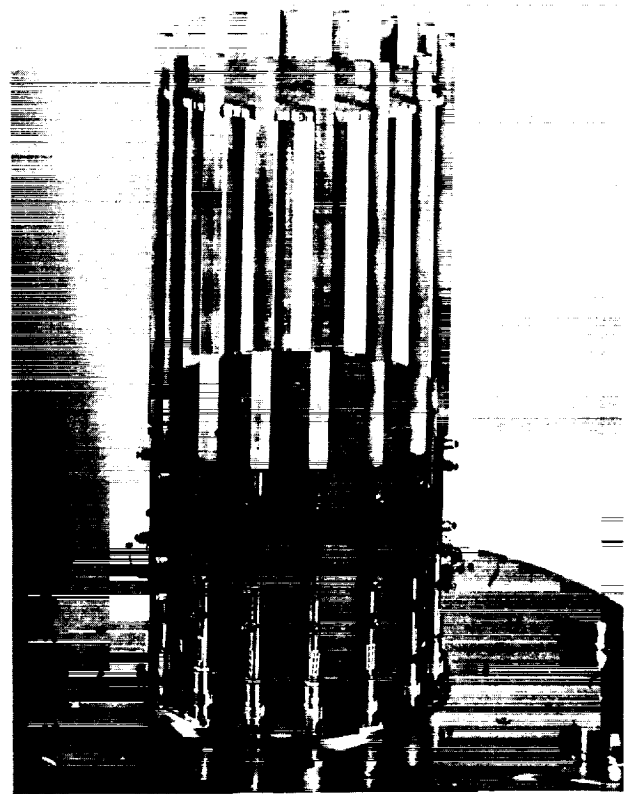
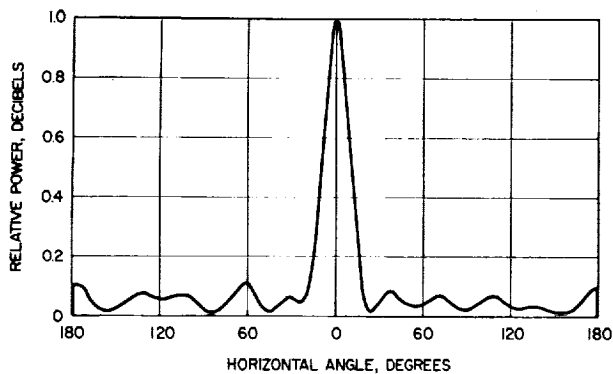
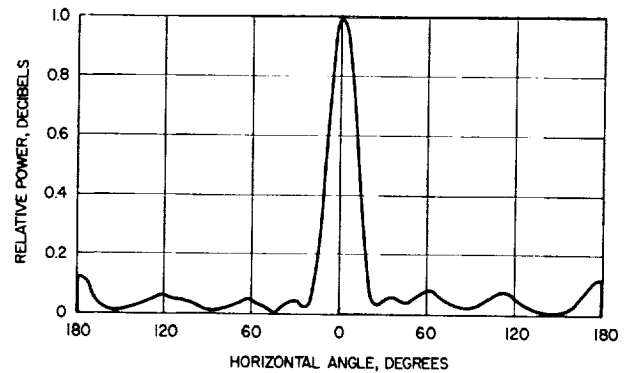


FIGURE 5-68. CLOSEUP OF PHASED ARRAY ANTENNA WITH FIBERGLASS STRUCTURE

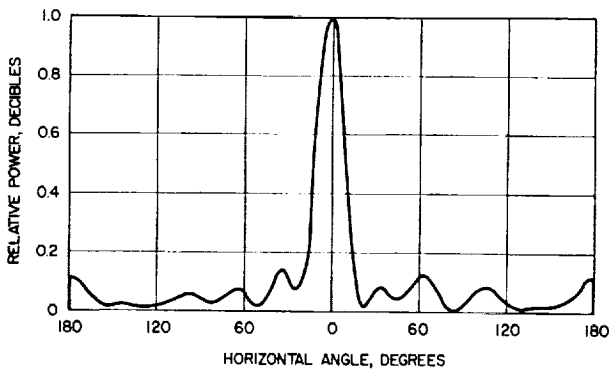
From losses listed above, the gain is between 14 and 15 db. There are additional losses due to the mutual interaction between antennas which shows up in two ways: 1) the input VSWR of an antenna changes in the presence of other unexcited antennas from 1.5 to 2.5 and more, and 2) preliminary measurements made by disconnecting one element of the array and measuring the intercepted power show that about 0.5 db of the power is absorbed by mutual coupling. Hence the gain of this initial all-stripline assembly should be between 13 and 14 db. The measured gain of the array at 4100 mc, compared to a standard horn, was just over 13 db. Thus, although the gain is lower than that attainable, the source of each of the losses is known and it is ex-



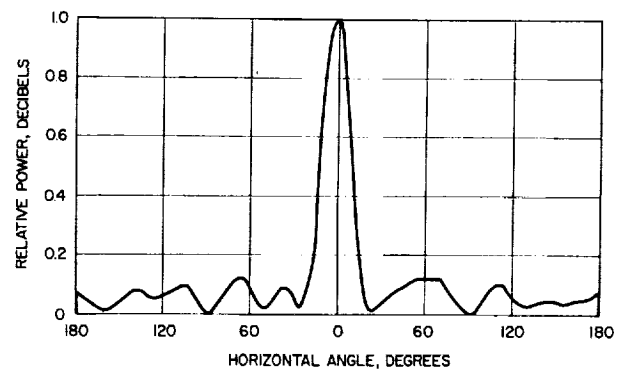
a) Beam Peak at Element No. 1



b) Beam Peak at Element No. 5



c) Beam Peak at Element No. 9



d) Beam Peak at Element No. 3

FIGURE 5-69. ANTENNA PATTERNS MEASURED AT 4100 MC

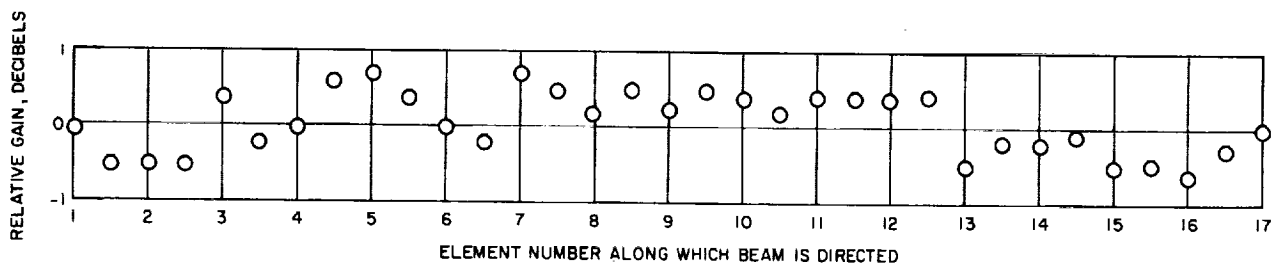


FIGURE 5-70. MAIN BEAM VARIATION VERSUS DIRECTION OF MAIN BEAM

pected that they can be reduced by at least 2 db to give a final net gain of nearly 16 db.

Phased Array Control Electronics (PACE)

PACE Development. The purpose of the PACE is to generate the set of 16 voltages

$$A \sin \left[2\pi \sin \left(\omega_s t + \theta + \frac{n\pi}{8} \right) \right]$$

$$A \cos \left[2\pi \sin \left(\omega_s t + \theta + \frac{n\pi}{8} \right) \right]$$

$$n = 0, 1, \dots, 7$$

$$\omega_s = \text{rotation frequency of satellite (rad/sec)}$$

which are required by the phase shifters. The first design problem was the choice of a suitable reference to supply the control electronics with an indication of spin speed. A sun sensor identical to the Syncom sun sensor was chosen because its use permitted the development of the control elec-

tronics with existing components. This choice imposes considerations of time-varying geometry, and if an infrared earth sensor is subsequently developed, it can be used to advantage in the control electronics.

TABLE 5-11. GAIN OF INDIVIDUAL ANTENNAS

Antenna Number	Gain, decibels		
	4000 mc	4100 mc	4200 mc
1	6.2	5.9	7.3
2	6.7	7.1	7.1
3	6.7	7.0	7.1
4	6.4	6.3	7.1
5	5.9	6.7	6.9
6	6.6	7.3	7.1
8	6.5	6.6	7.0
9	6.7	6.8	7.1
10	6.7	6.8	7.1
11	6.6	6.3	7.1
12	6.7	6.8	7.0
13	6.7	6.3	7.0
14	6.4	6.9	7.2
15	6.2	6.9	7.1
16	6.6	6.3	6.8

Figure 5-71 is a block diagram of the PACE. The frequency lock loop generates a pulse train at frequency $2^9 f_s$. The beam positioner counts this frequency down and corrects for the relative earth-sun rotation also experienced by the satellite. The sine wave generator converts the beam positioner output into the set of sinusoids

$$A \sin \left(\omega_s t + \frac{n\pi}{8} + \theta \right)$$

$$n = 0, 1, 2, \dots, 15.$$

The waveform generators convert these into the required output waveforms which are then amplified to the current power and voltage level by the phase shifter drivers. The loop generates a frequency source at a precise multiple of the input frequency of the sun sensor pulses. A frequency lock loop, as shown in Figure 5-72, was the original choice as the best means to achieve the frequency multiplication because of concern

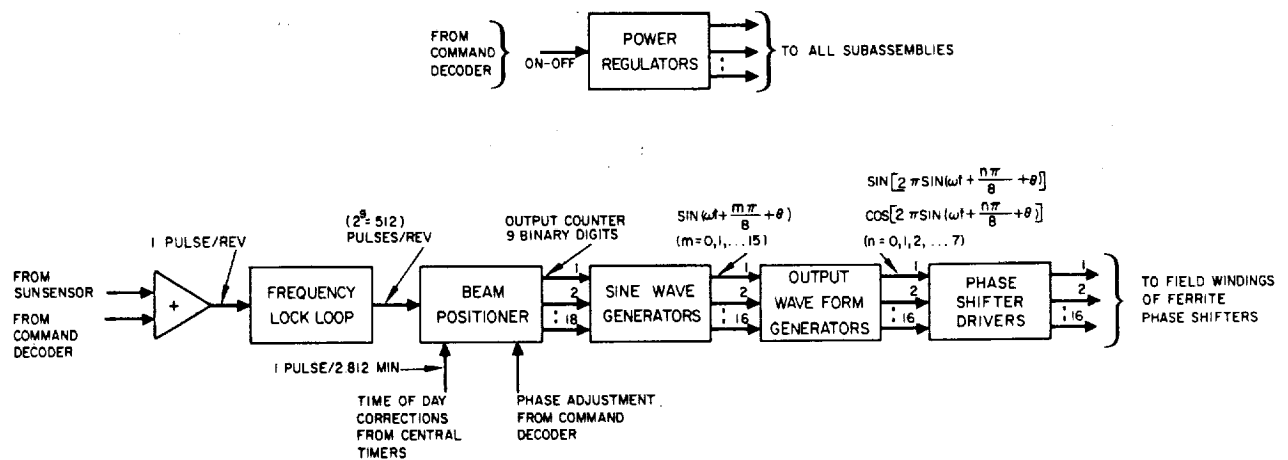
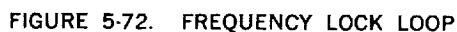


FIGURE 5-71. PACE BLOCK DIAGRAM.

the phase lock loop and its operation is explained in detail in the March 1963, Summary Report.

The frequency lock loop block diagram is given in Figure 5-73. The primary function of the loop is to generate pulses at a frequency $2^9 f_s$, where f_s is the spin frequency of the spacecraft. The input reference is one pulse per revolution, usually derived from the sun sensor but also derived from the command decoder during eclipses. Another output of the loop is $2^{12} f_s$, which



is used for on-board measurement of the solar aspect angle.

The input reference pulse triggers F100, causing it to change state with each pulse. When F100 goes from 1 to 0 the reset amplifier is triggered, which resets the voltage-controlled oscillator and flip-flops F101 through F113 to 0. Flip-flops F101 through F112 constitute a binary counter. At the end of the 2^{12} th voltage-controlled oscillator cycle after reset, F113 is set to 1. If the voltage-controlled oscillator frequency is exactly $2^{12}f_s$, F100 will also be set to 1 at the same time, and there will be 0 error. In general there will be some timing error between F100 going to 1 and F113 going to 1, generating an output from the error pulse gate. The logic of the error pulse gate is illustrated in the truth table of Table 5-12. When

TABLE 5-12. ERROR PULSE GATE TRUTH TABLE

F100	F113	Output
0	0	Decoupled
0	1	-V
1	0	+V
1	1	Decoupled

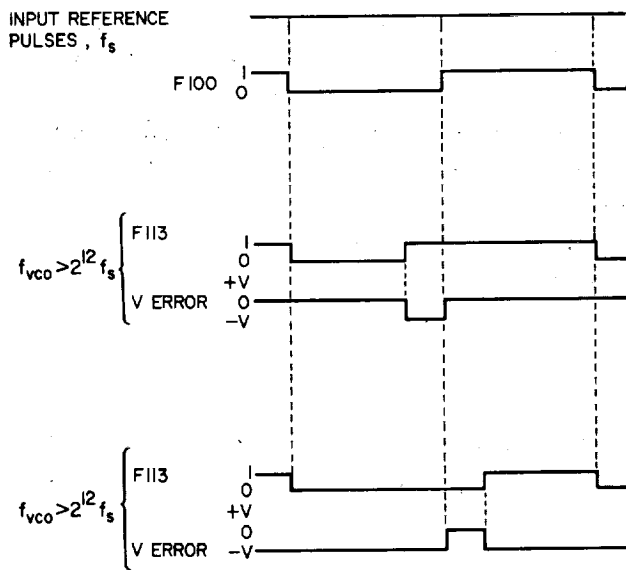


FIGURE 5-74. ERROR PULSE GATE WAVEFORMS

F100 and F113 are in the same state the output of the error pulse gate is decoupled, through very low leakage diodes, from the filter. When F100 and F113 are different, the output of the error pulse gate is either $-V$ or $+V$, depending on which of the two possible combinations occurs. Figure 5-74 shows the error pulse gate waveforms for both positive and negative errors.

The filter F701A has the configuration shown in Figure 5-75. The filter integrates the error pulses and provides loop compensation.

The amplifier F701B serves to unload the filter and drive the voltage-controlled oscillator. The schematic of amplifier F701B is shown in Figure 5-76. The very large input impedance of the field effect transistor is utilized to unload the filter. The amplifier F701B is essentially a source follower circuit. Transistors Q1 and Q2 and zener diode CR1 maintain a constant source-to-drain voltage which minimizes the gate current. Zener diode CR2 shifts the output level as required by the voltage-controlled oscillator circuit.

The lock logic samples flip-flops F103 through F113 shortly after F100 is set to 1. If F103 through F112 are all 0 and F113 is 1, the lock logic output is set to 1, indicating the loop error

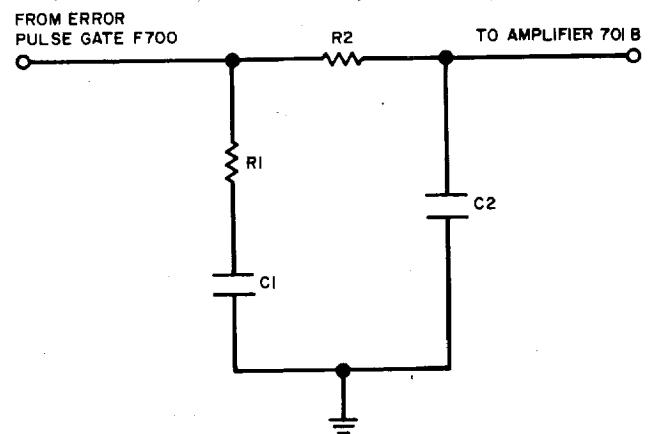


FIGURE 5-75. FILTER F701A

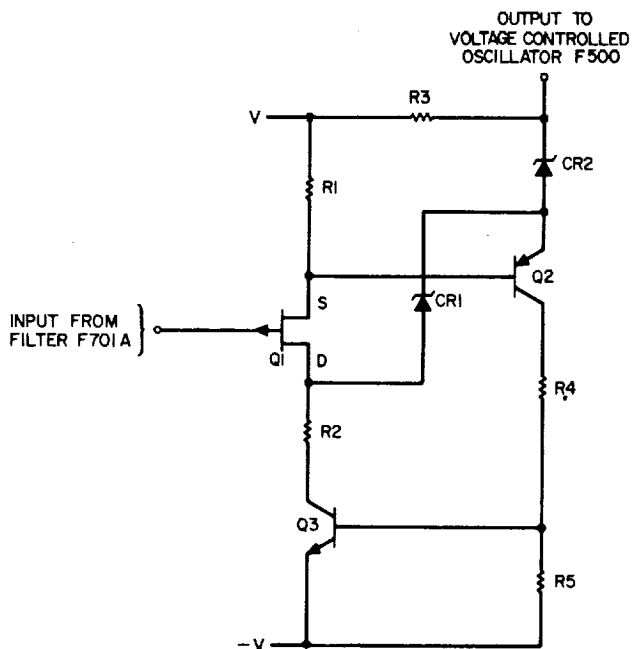


FIGURE 5-76. AMPLIFIER F701B

is less than one part in 2^9 . The lock logic output is telemetered in order to evaluate loop performance; it is also used to control the on-board orbital control system.

Figure 5-77 is a block diagram of the beam positioner. The function of the beam positioner is to generate in digital form the instantaneous rotational angle of the spacecraft with respect to a selected reference. The inputs are a pulse train from the frequency lock loop having 512 pulses per spacecraft revolution and a command input for phase control.

The output counter is a nine-flip-flop binary counter which counts the frequency lock loop output F103. The period of the output counter is equal to the spacecraft rotational period. The nine-flip-flop angle encoder register stores the phase reference of the output counter. The parallel transfer gate is triggered by F103, the reset amplifier of the frequency lock loop, which pulses

on alternate input reference pulses. Thus on every second spacecraft rotation the contents of the angle encoder register are parallel-transferred to the output counter. Small errors of the frequency lock loop are therefore not accumulated as they would be if the phase of the output counter were set less frequently.

The contents of the angle encoder register can be incrementally adjusted by means of the command system. To aid in operation of the system and to indicate performance the angle encoder register contents are telemetered. The digital readout encoders convert the contents of the angle encoder register into three analog voltages which are inputs to the telemetry encoder. Thus, to operate the system, the telemetered contents of the angle encoder register are noted and compared with the desired phase angle; the appropriate number of increments are commanded to obtain the desired phase. The resolution of the system is $360 \text{ degrees}/2^9$ or about 0.7 degree.

As the spacecraft orbits the earth, the angle between the spacecraft-earth line changes at the rate of about 360 degrees per 24 hours. Since the phase of the beam positioner is referenced to the spacecraft-sun line, maintaining the beam pointed toward the earth requires compensation for this time-varying geometry. Accordingly, the angle encoder register is stepped at the rate of 512 increments per 24 hours, or one pulse every 2.812 minutes. These "time-of-day" correction pulses are generated by the spacecraft central timer derived from a tuning fork reference oscillator.

The beam positioner output counter continuously drives two 8-bit, digital-to-analog (D/A) converters. The conversion is performed by weighted resistor ladders, with inputs controlled by the eight least significant bits of the output counter through low offset transistor

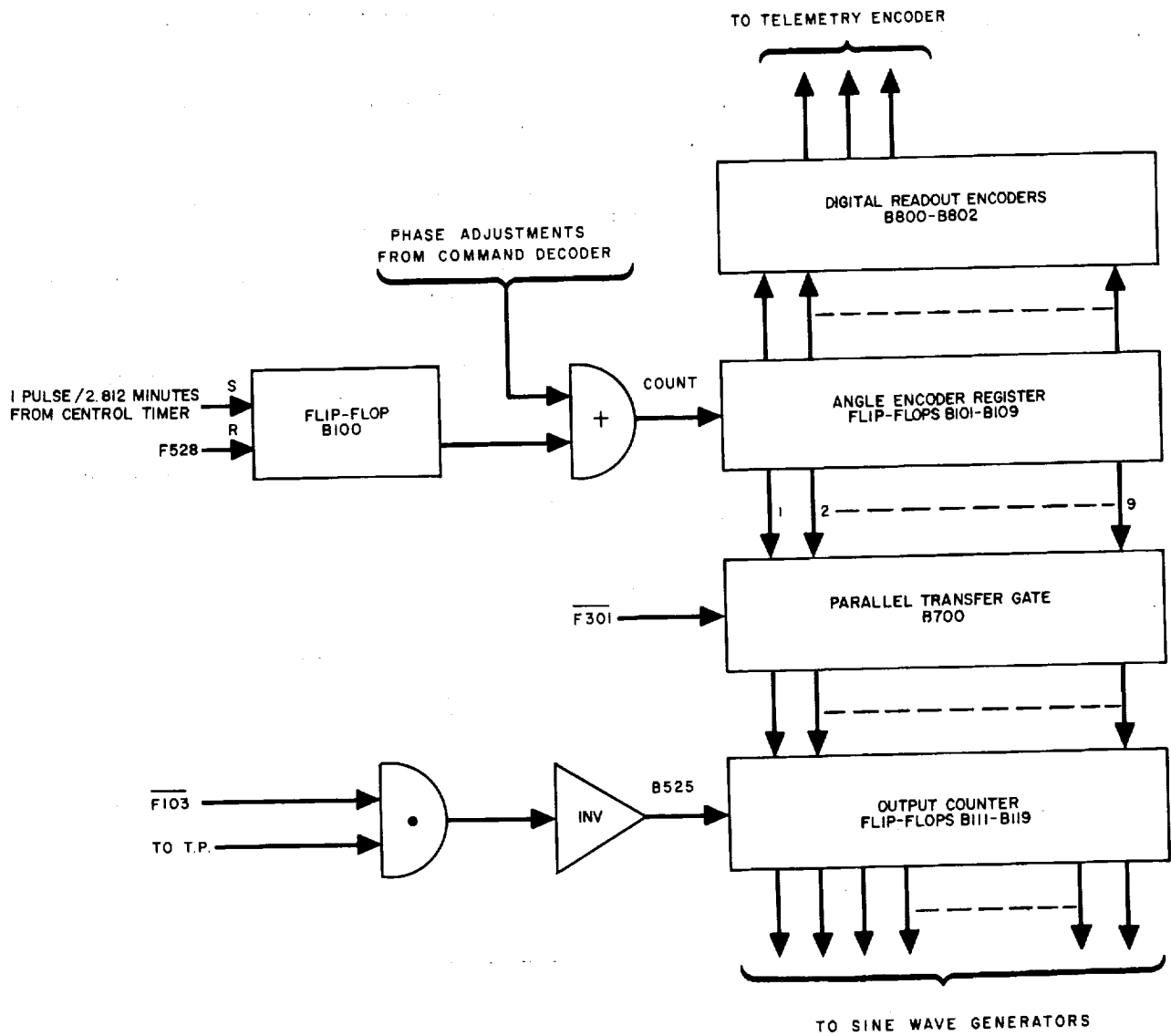


FIGURE 5-77. BEAM POSITIONER BLOCK DIAGRAM

switches. The first converter makes a normal conversion while the output counter counts from 0 to half of full scale. During this half-cycle, the output is a "staircase" which approaches a ramp, traversing from -14 to $+14$ volts in 256 equal voltage increments. When the most significant bit of the output counter switches from 0 to 1, the inputs to the ladder are switched from the normal outputs of the output counter flip-flops to the complementary outputs, the significance remaining the same. Since the complementary outputs of a counter counting up are the same as a counter counting down, the converter steps from $+14$ to -14 volts during the second half-cycle of the output counter. When the output counter recycles, a normal conversion is again made and the sequence described above is repeated. Figure 5-78 shows the output waveform and timing of the first converter. The most significant bit of the output counter is used as a gate to determine

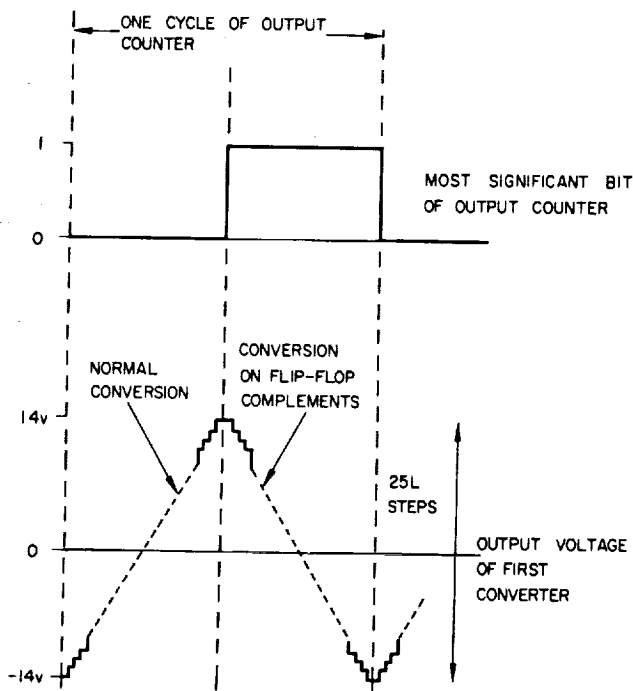


FIGURE 5-78. OUTPUT VOLTAGE AND TIMING OF FIRST DIGITAL-TO-ANALOG CONVERTER

whether a normal or complementary conversion is to be performed.

In order to generate the required output waveforms, a second signal must be derived from the output counter. This waveform must have the same form as the output of the first converter, but must lead it in phase by 90 degrees. A timing gate must be generated, which performs the selection as to whether a normal or complementary conversion is to be made. Figure 5-79 shows that this gate is the simple logic combination $8\bar{9} + \bar{8}9$, where 8 and 9 are the next-to-most and most significant bits of the output counter. Figure 5-79 also shows the output waveform and timing

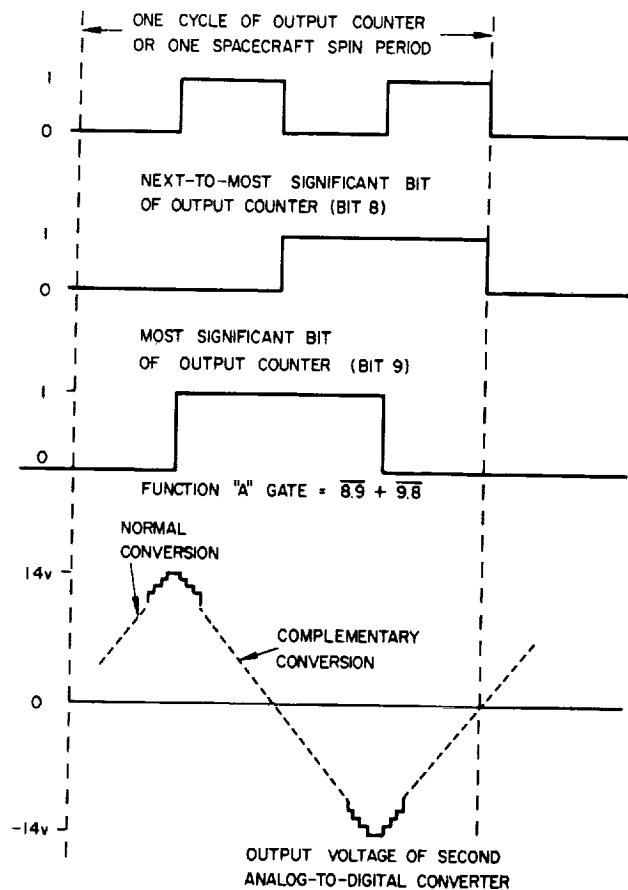


FIGURE 5-79. OUTPUT VOLTAGE AND TIMING OF SECOND DIGITAL-TO-ANALOG CONVERTER

of the second D/A converter. The normal and complementary outputs of bit 8 are reversed for this conversion; the other seven bits are connected as in the first converter.

The triangular-shaped outputs of the D/A converters are then shaped into sinusoids by a nonlinear, voltage variable resistor. Figure 5-80a shows an equivalent circuit for one of the shaping networks. V_g is the output voltage and R_g is the equivalent resistance of the resistor ladder of the D/A converter. Figure 5-80b shows the shaping that is performed. Figure 5-80c is the schematic of the circuit which creates the resistor $R = R(V_0)$. It is a diode function generator of the type commonly used in analog computers. Experimental data on the symmetrical six-diode shaper show that a sinusoid with total harmonic distortion of less than 1 percent is readily achievable with a triangular input.

The outputs of the shapers are thus very close approximations to sinusoids and the output of the shaper connected to the first D/A converter is defined as $A \sin \omega_s t$. With this phase reference, the second shaped output is $A \cos \omega_s t$. Both waves are amplified by dc amplifiers to form $B \sin \omega_s t$ and $B \cos \omega_s t$. These outputs are then inverted by two amplifiers with gains of -1.0 . The four sinusoids $B \sin \omega_s t$, $B \cos \omega_s t$, $-B \sin \omega_s t$, and $-B \cos \omega_s t$ have thus been generated.

The four sine waves are used to drive a network which produces the 16 output voltages

$$V_n = \frac{B}{\sqrt{2}} \sin \left(\omega_s t + \frac{n\pi}{8} \right), n = 0, 1, 2, \dots, 15$$

This network, called the vector summing network, implements the trigonometric identity

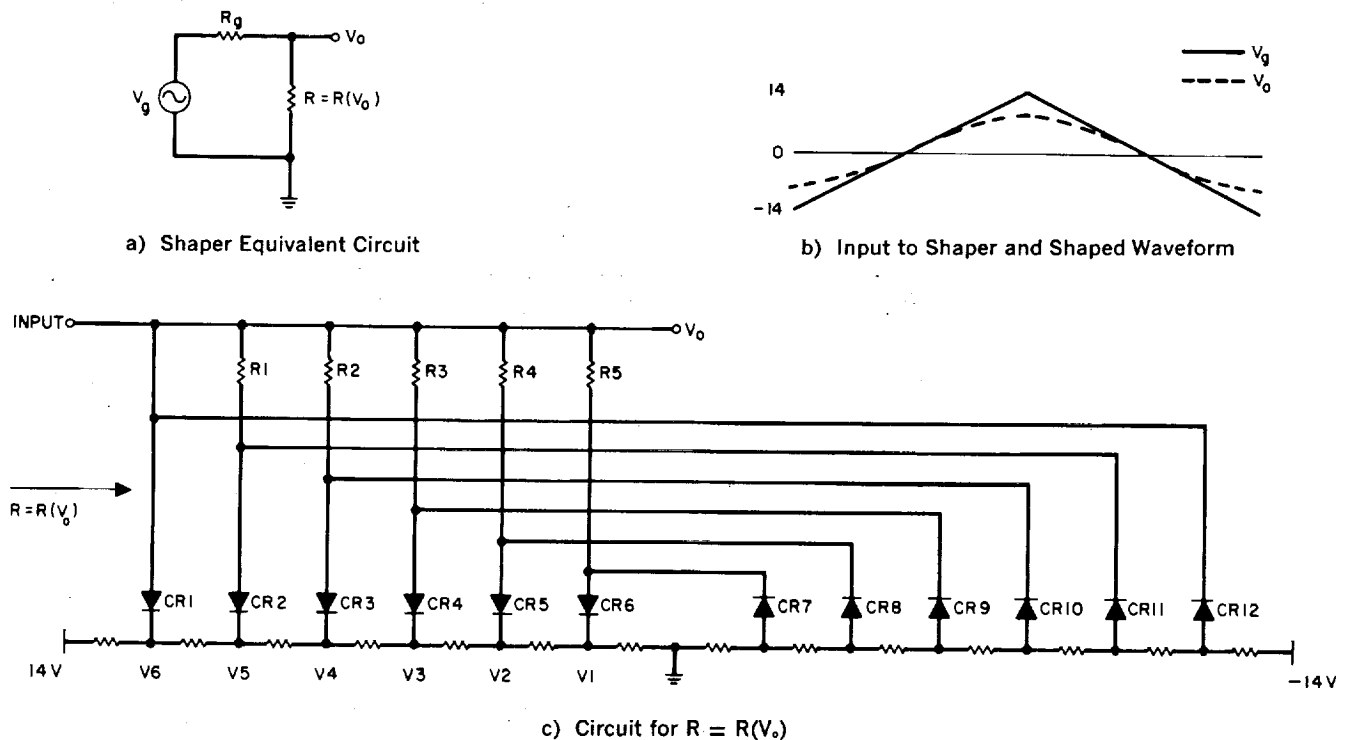


FIGURE 5-80. DEVELOPMENT OF SINE WAVES

$$A \sin \omega_s t + B \cos \omega_s t$$

$$= \sqrt{A^2 + B^2} \sin \left(\omega_s t + \tan^{-1} \frac{B}{A} \right)$$

An example showing the case for $n = 1$ is given in Figure 5-81. If the resistors are in the given ratios the output will be as shown. Transistors Q1 and Q2 form a complementary emitter follower which gives unity voltage gain and high current gain and presents a high input impedance to the input summing resistors. The 2 appears in the equation because each sine wave source loads the other. Each of the 16 phases is generated by the technique shown in Figure 5-81. The complete section from the low offset switches to the 16 sine wave outputs is called the sine wave generator subassembly and a block diagram is shown in Figure 5-82.

It has been shown that each of the eight phase shifters requires two signals of the form

$$D \sin \left[2\pi \sin \left(\omega_s t + \frac{n\pi}{8} + \theta \right) \right]$$

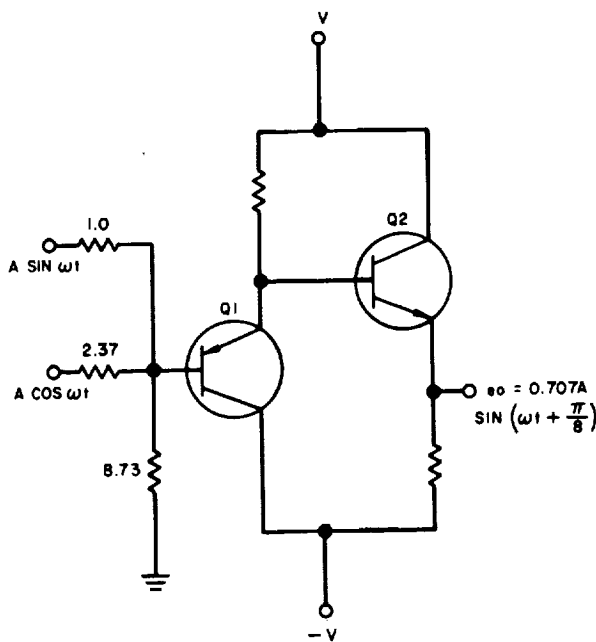


FIGURE 5-81. EXAMPLE OF VECTOR SUMMING NETWORK

$$n = 0, 1, \dots, 7$$

and

$$D \cos \left[2\pi \sin \left(\omega_s t + \frac{n\pi}{8} + \theta \right) \right]$$

$$n = 0, 1, \dots, 7$$

Normalized plots of these functions are shown in Figure 5-83 for $n = 0$. The eight phases of these waveforms are generated from the 16 sine waves by unique waveform generators. To understand the operation of these generators, consider the standard diode AND gate shown in Figure 5-84a. The output voltage is a diode drop above whichever input is at the more negative voltage; that is, it selects the more negative of the two inputs (lesser gate). Similarly, the output of the standard diode OR gate of Figure 5-84b is one diode drop below the higher of the two input voltages (greater gate).

The circuit which generates $D \cos [2\pi \sin \omega_s t]$ is shown in Figure 5-84c. Figure 5-84d shows the two input waveforms and Figure 5-84e the waveforms at A and B. The waveform at A is a positive full rectified sine wave. It is formed by taking the output from whichever input is higher (greater gate). The waveform at B is a negative full rectified sine wave and is formed by taking the output from whichever input is lower (lesser gate). The waveform at C is the waveform at A shifted down in voltage by $\frac{3}{4}$ of the input amplitude. The signal at D is that at B shifted up in voltage by $\frac{1}{4}$ of the input amplitude. These waveforms are shown in Figure 5-84f. The voltages at C and D are then combined in a greater gate to produce the voltage at E which is plotted in Figure 5-84g. Comparison of this figure with Figure 5-84b shows that if the waveform at E is shaped to remove the sharp edges, it is a close approximation to the required output. This shaping is performed by the nonlinear resis-

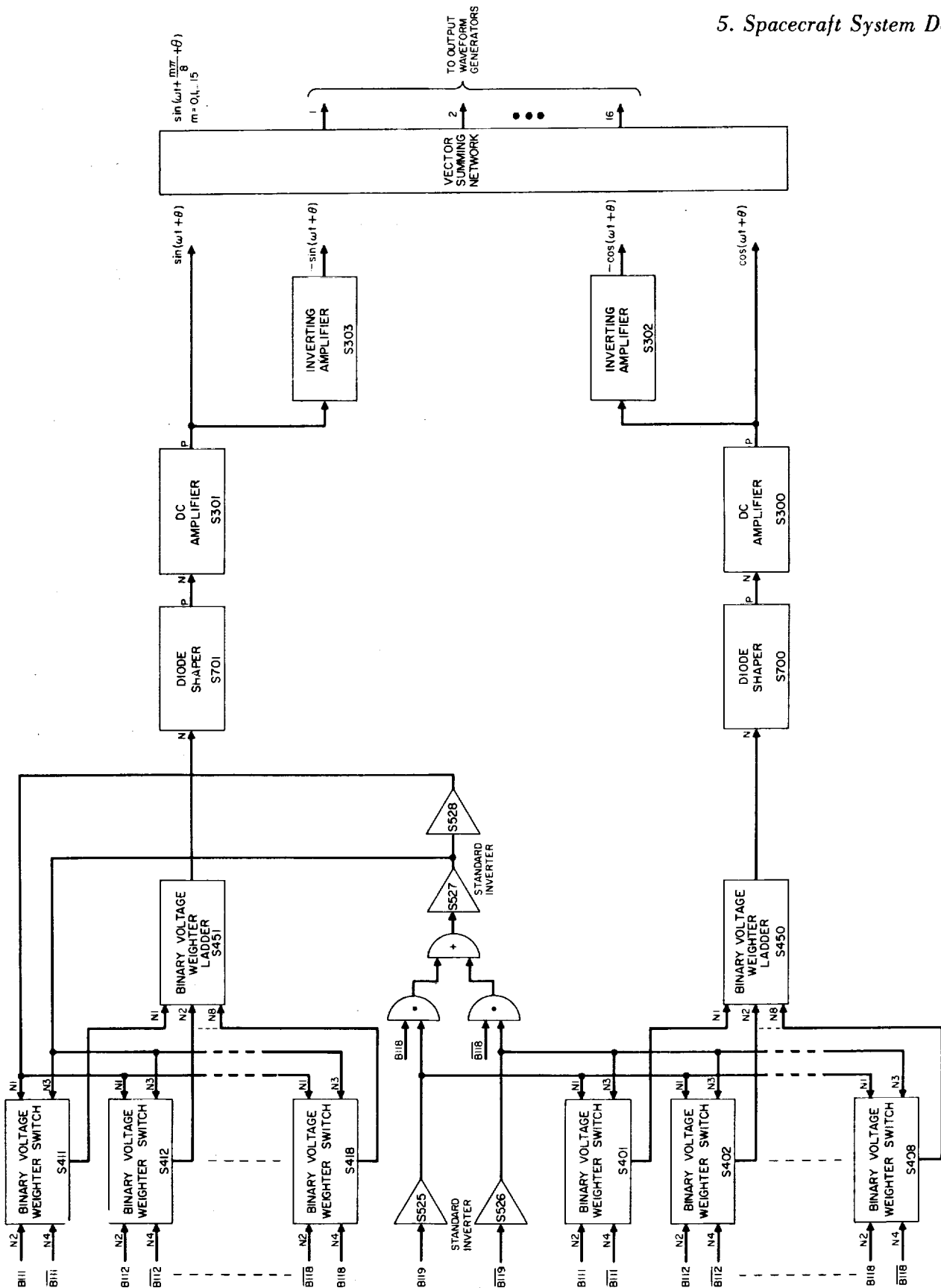


FIGURE 5-82. SINE WAVE GENERATOR

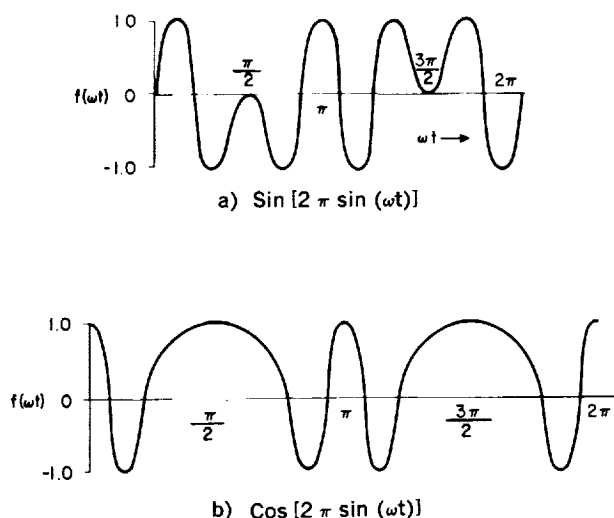


FIGURE 5-83. REQUIRED OUTPUT WAVEFORMS (NORMALIZED)

tor, $R(V_0)$. The other required waveform is generated in a similar manner.

Each of the signals created by the waveform generators is then amplified to a voltage and power level suitable for driving the phase shifters by a potentiometric type dc amplifier.

An advanced engineering model of this system was constructed on flat circuit cards. This model was evaluated under severe environmental extremes and in general performed very well. It was tested over a temperature range of -50 to 125°C . The input voltages of ± 24 volts ± 3 percent were varied from ± 20 to ± 30 volts, and noise spikes of 1 volt were introduced on the power inputs. In addition, the input frequency was varied from $\frac{5}{8}$ cps to $2\frac{1}{2}$ cps, corresponding to spacecraft spin speeds from 50 to 150 rpm.

For all of these conditions the advanced engineering model operated within specification over the temperature range from -50 to 100°C . At 125°C , the loop error was less than 0.7 degree only for spin speeds near 75 rpm. However, although the performance was degraded the system was still functioning. Table 5-13 is a summary

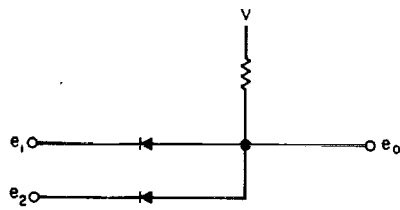
of the data taken. Lock is the time required for the loop to achieve an error of less than 0.7 degree from turnon; error is the steady-state time error after the loop has stabilized.

Since each pulse of the frequency lock loop represents $360/4096$ spacecraft degrees, it can be used to count the angle in spacecraft degrees between ψ and ψ_2 . A unit was developed for this purpose composed of a flip-flop counter, control circuitry, and the circuitry for converting the digital bits into analog form for telemetering. The ψ_2 counter counts the $\psi - \psi_2$ angle as magnitude (45 degrees) and sign.

Further improvements in the PACE circuitry appear feasible. Studies in the following areas are in progress.

- 1) Since the angle encoder register of the beam positioner is used to prevent the accumulation of errors over long periods of time, it could be used for short periods to count the ψ to ψ_2 angle upon ground command. This could save about 300 components per quadrant plus three telemetry channels. Care must be taken so that performance of this auxiliary function will not compromise the reliability of the antenna control.

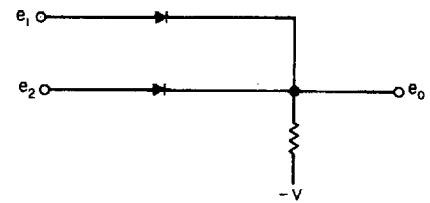
- 2) There is an alternate method of creating the ramp voltages in the sine wave generators. Referring to Figure 5-85, the precision one-shot enables both current generators. However, only the one selected by the flip-flop will supply current to the capacitor. The constant current applied to the capacitor for a fixed time supplies a fixed charge to the capacitor, thus generating a fixed ΔV each time the one-shot is triggered. The clamp circuit clamps the capacitor voltage at the extreme end of the ramp to prevent accumulation of errors. The flip-flop is controlled by the beam positioner logic. The high input impedance amplifier generates the output at a reasonable impedance level without loading the capacitor. A breadboard of this subassembly has been built



$$e_o = e_1 \text{ FOR } e_1 < e_2$$

$$e_o = e_2 \text{ FOR } e_2 < e_1$$

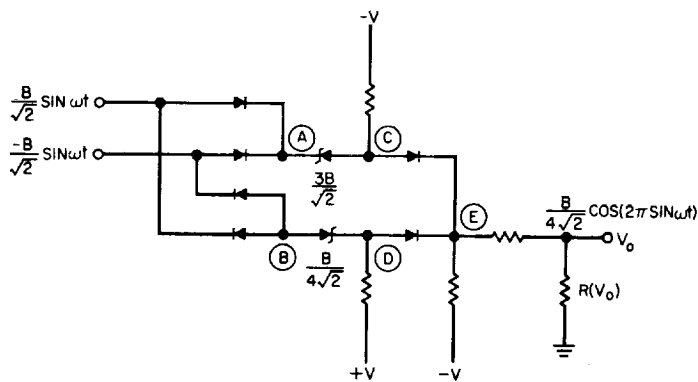
a) Lesser Gate



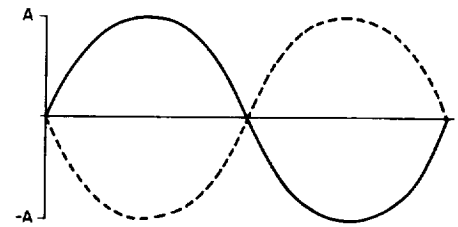
$$e_o = e_1 \text{ FOR } e_1 > e_2$$

$$e_o = e_2 \text{ FOR } e_2 > e_1$$

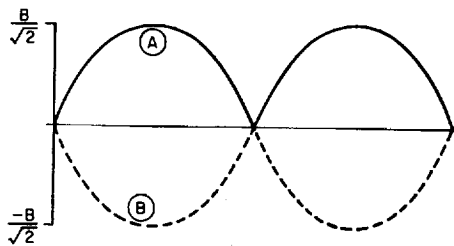
b) Greater Gate



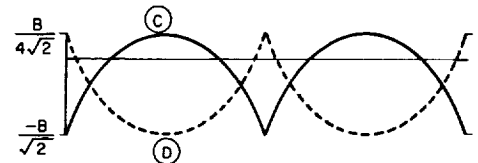
c) $\cos [2 \pi \sin (\omega t)]$ Generator



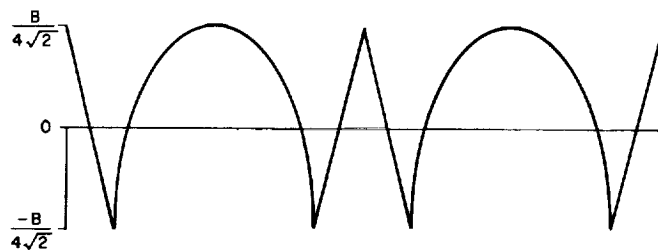
d) Input Waveforms



e) Waveforms at (A) and (B)



f) Waveforms at (C) and (D)



g) Waveform at (E)

FIGURE 5-84. DEVELOPMENT OF OUTPUT WAVEFORMS

TABLE 5-13. FREQUENCY LOCK LOOP PERFORMANCE

Spin Rate, msp	0° C			+50° C			Room Temperature 95° F			+100° C			+50° C			+100° C			-25° C		
	Lock- in Time	Steady- State Error		Lock- in Time	Steady- State Error		Lock- in Time	Steady- State Error		Lock- in Time	Steady- State Error		Lock- in Time	Steady- State Error		Lock- in Time	Steady- State Error		Lock- in Time	Steady- State Error	
400	2 min 15 sec	-10 μ sec		1 min 45 sec	+ 0.5ms		1 min	- 6 μ sec		7 min	+2.0 ms		8 min	+0.4 ms		3 min	+160.0 μ sec		2 min 10 sec	+50.0 μ sec	
500	2 min 31 sec	- 3 μ sec		55 sec	+ 0.7ms		2 min 3 sec	- 62 μ sec		No lock	+0.6ms		2 min	+0.6 ms		2 min	+0.6 ms		1 min 45 sec	+10.0 μ sec	
600	1 min 15 sec	- 2 μ sec		45 sec	+55.0 μ s		1 min 45 sec	- 28 μ sec		5 min	+0.8ms		2 min	+60.0 μ sec		2 min	+0.4 ms		1 min 20 sec	+ 2.5 μ sec	
800	1 min 35 sec	+12 μ sec		50 sec	+ 0.5ms		1 min 40 sec	+ 24 μ sec		6 min	+0.6ms		3 min	+0.5 ms		7 min	+0.8 ms		1 min 35 sec	-35.0 μ sec	
1200	1 min 29 sec	+55 μ sec		1 min 25 sec	+ 0.7ms		1 min 35 sec	+150 μ sec		No lock	—		7 min	+0.6 ms		8 min	+120.0 μ sec		1 min 55 sec	-75.0 μ sec	

Spin Rate, msp	Room Temperature 78° F			-25° C			-35° C			-40° C			-40° C			-45° C			-50° C		
	Lock- in Time	Steady- State Error		Lock- in Time	Steady- State Error		Lock- in Time	Steady- State Error		Lock- in Time	Steady- State Error		Lock- in Time	Steady- State Error		Lock- in Time	Steady- State Error		Lock- in Time	Steady- State Error	
400	2 min 15 sec	+225.0 μ sec		2 min 20 sec	+125.0 μ sec		1 min 20 sec	-56 μ sec		2 min 15 sec	-60 μ sec		2 min 10 sec	-35 μ sec		2 min 20 sec	-40 μ sec		1 min 50 sec	-90 μ sec	
500	35 sec	- 50.0 μ sec		45 sec	- 15.0 μ sec		45 sec	-40 μ sec		40 sec	-40 μ sec		45 sec	- 6 μ sec		40 sec	- 8 μ sec		45 sec	- 5 μ sec	
600	1 min 5 sec	- 2.5 μ sec		1 min 10 sec	- 2.5 μ sec		1 min	-42 μ sec		45 sec	-25 μ sec		1 min	- 5 μ sec		1 min 5 sec	-30 μ sec		1 min 10 sec	-10 μ sec	
800	45 sec	+ 42.0 μ sec		1 min 55 sec	+ 58.0 μ sec		50 sec	+ 60 μ sec		1 min	+60 μ sec		35 sec	+60 μ sec		55 sec	+60 μ sec		1 min 15 sec	+80 μ sec	
1200	55 sec	+360.0 μ sec		1 min 45 sec	- 75.0 μ sec		55 sec	+ 85 μ sec		40 sec	+65 μ sec		1 min 5 sec	+80 μ sec		1 min 20 sec	+90 μ sec		1 min 20 sec	+140 μ sec	

TABLE 5-13 (continued)

Spin Rate, msp	+100° C			+100° C			+115° C			+125° C		
	Lock-in Time	Count	Steady-State Error	Lock-in Time	Count	Steady-State Error	Lock-in Time	Count	Steady-State Error	Lock-in Time	Count	Steady-State Error
400	4 min	No lock 8155	- 2.2 ms	15 min	No lock 8147	-2.2 ms	15 min	No lock 8063	-6.0 ms	10 min	No lock 7934	-12.0 ms
500	14 min	No lock 8174	- 1.1 ms	14 min	No lock 8175	- 1.0 ms	14 min	No lock 8144	-3.0 ms	10 min	No lock 8094	-6.0 ms
600	1 min 55 sec	8191	-80.0 μ sec	6 min	8191	-80.0 μ sec	20 min	No lock 8171	-1.5 ms	10 min	No lock 8151	-3.0 ms
800	12 min	8195	+ 0.2 ms	12 min	8195	+0.2 ms	7 min	8185	-0.6 ms	4 min	8188	-0.4 ms
1200	18 min	8199	+ 1.0 ms	25 min	8199	+1.2 ms	14 min	No lock 8231	+5.5 ms	10 min	No lock 8286	+14.0 ms

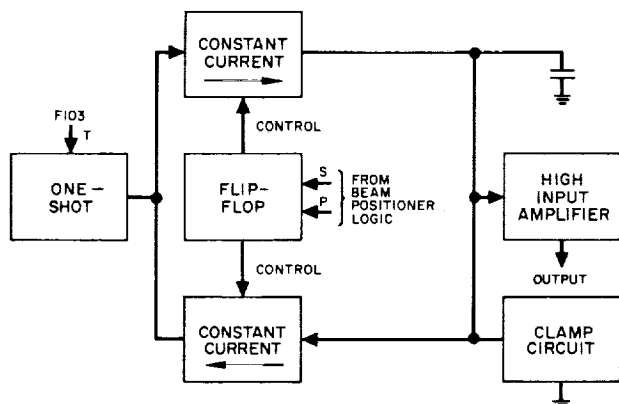


FIGURE 5-85. ALTERNATE RAMP GENERATOR

and is presently being evaluated. If the errors can be held to a reasonable level, this technique would reduce by about 150 the components per quadrant and the power by about 1 watt.

3) The present frequency lock loop can have a high loop gain even when in nominal lock. This means that in the event of high extraneous system noise or a lost solar pulse communications might be degraded for periods up to 90 seconds. To prevent this, circuitry will be included to limit the width of the error pulses once the loop has achieved lock.

4) In order to evaluate PACE performance during system test and flight, circuitry is being added to provide an indication of the frequency lock loop steady-state error. This will be in the form of an analog voltage proportional to steady-state error and will be telemetered.

PACE Product Design

• **Advanced Engineering Model.** The advanced engineering model consisting of the phased antenna control electronics (PACE), the jet control electronics (JCE), the ψ_2 angle counter, and the phase shifter drivers was designed on open-form wired etched circuit cards. The commercial components were designed onto 62

cards of 34 types resulting in a density of approximately 48 components per card. Fifty-eight of the cards were mounted in a two-section computer type unit of existing design. The four remaining cards, electrically simulating the quadrant interface with three other quadrants, were mounted in the unit designed for the Syncom 2 encoder and decoder simulator. The wire lists for the units were generated using an existing computer program which provided the error checked wire list and a signal usage list to obtain reductions in both wiring and testing time.

• **Spacecraft Module Construction.** The mass encapsulation technique of encapsulating welded modules, the natural next generation of Syncom techniques, was developed specifically for Advanced Syncom digital electronics. It is a technique whereby the individual welded modules are mounted and connected to an etched circuit card, electrically tested as a subassembly, and then encapsulated as a subassembly. This results in an environmentally sound unit requiring little additional structural support in a spacecraft environment and reduces the encapsulation costs.

The components are placed in a low density rigid polyurethane component holder (HMS 16-1317), which has been epoxy impregnated to increase strength and prevent flaking, and are interconnected by resistance welding (HP31-5, Grade A) nickel ribbon to the leads. The module is then placed on a clear header plate and covered with a vacuum formed, clear plastic protective cover and electrically tested (see Figures 5-86, 5-87, and 5-88). Any possible handling damage is thus prevented. After test the covers are removed and the modules are soldered to an etched circuit board (actual boards are shown in Figure 5-89); the entire assembly has a vacuum formed protective cover installed and is

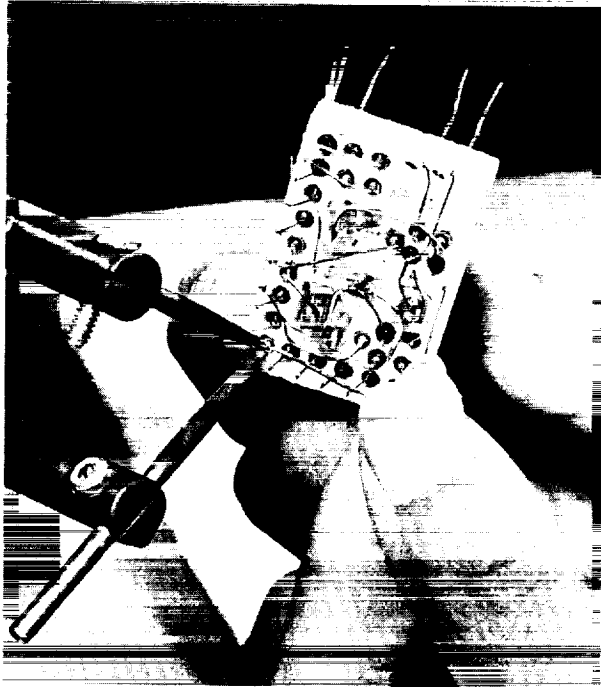


FIGURE 5-86. MODULE BEING WELDED

then electrically tested. After electrical testing the cover is removed and the assembly is placed in the tooling and encapsulated (HP 16-9) with a polyurethane foam (HMS 16-1287, Type IV).

There are several major advantages of this system over those previously used. By encapsulating at the board level the number of encapsulation processes required is reduced by as much as 15 to 1 and the number of tools required is proportionately reduced. By encapsulating later in the manufacturing cycle and having the ability to electrically test entire subassemblies prior to encapsulation, the probability of requiring repair after encapsulation is reduced. The combination of foam and etched board form a structurally sound beam, eliminating the need for honeycomb board supports required of previous systems and reducing the weight of structure required. The outline of each of the modules is stenciled on the encapsulating foam in addition to each module's

part and serial number. Should repair of a module become necessary, the foam surrounding the module is cut, the module leads are unsoldered, and the module is removed from the board. The module is repaired or replaced, mounted on the board, and the module area is refoamed and restenciled.

- Spacecraft Electronic Quadrant Product Design. The digital quadrant electronics equipment is packaged into three quadrant boards each 11.25 by 18 by 1.130 inches. These three quadrant boards, the command board, the PACE board, and the auxiliary board, are then integrated with the transponder subassembly to form the electronics quadrant.

The quadrant command board contains the command decoder, command receiver, diplexer, and command regulator. The quadrant PACE board contains the frequency lock loop, beam positioner, sine wave generator, and waveform generator. The quadrant auxiliary board contains the telemetry encoder, jet control electronics, ψ_2 angle counter, central timer, and PACE regulator.

Structure is very similar for all three quadrant boards with the only changes being those required by the different sizes of function blocks to be mounted. The structure consists of two 17.73-inch-long right angle extrusions with flat cross members 10.62 inches long bonded in place. A sheet of 0.005 inch thick epoxy glass laminate is bonded to the aluminum framework to provide insulation for the etched function blocks.

A vibration test was made in which an etched circuit board, similar in size to a Syncom 2 board but mass encapsulated, was tested and the responses recorded. With this data a stress analysis was performed which indicated a deflection of 0.160 inch maximum. This large amount of deflection was due to the ability of the angles to

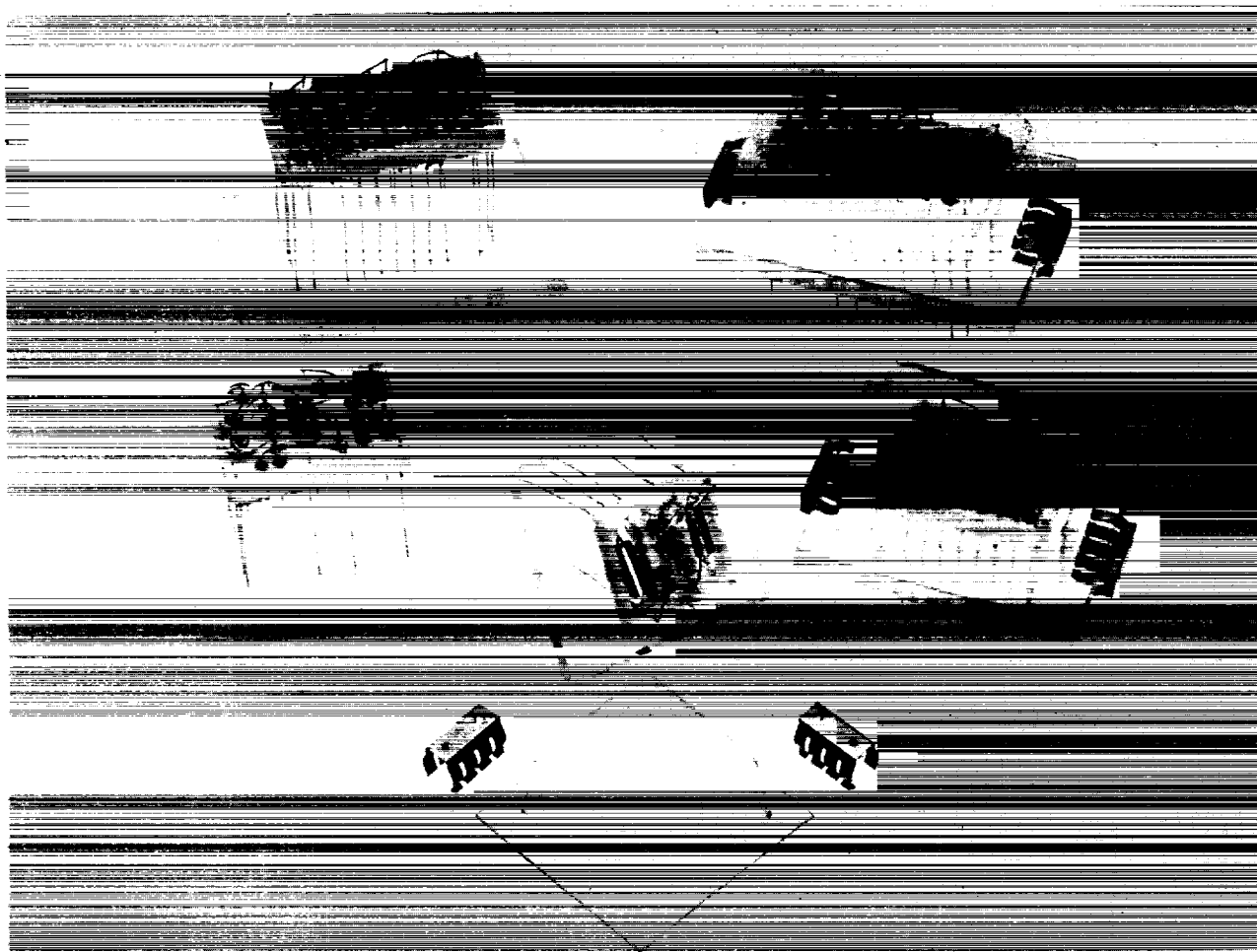


FIGURE 5-87. MODULE ASSEMBLY STEPS

pivot. When the ends were fixed the deflection was reduced to 0.032 inch. When a stress analysis was made of the quadrant it was found that the entire quadrant could flex from rectangular to form a parallelogram about the mounting. Two 0.020-inch-thick aluminum shear panels, one on the forward face and one on the outboard face of the quadrant, were provided which eliminated this condition. The weight of the three quadrant boards will be approximately 10 pounds.

- Spacecraft Design of PACE Electronics. The PACE circuits are designed with the fre-

quency lock loop, beam positioner, sine wave generator, and waveform generator placed on the PACE quadrant board and the ψ_2 angle counter on the auxiliary quadrant board. The components contained in the first four listed subsystems are divided into four separate function blocks. The PACE board contains 55 modules of 28 types with an average component density of 41 components per cubic inch at the module level.

The advanced development module (ADM) of the PACE quadrant board is now being welded in



FIGURE 5-88. MODULE READY FOR ENCAPSULATION

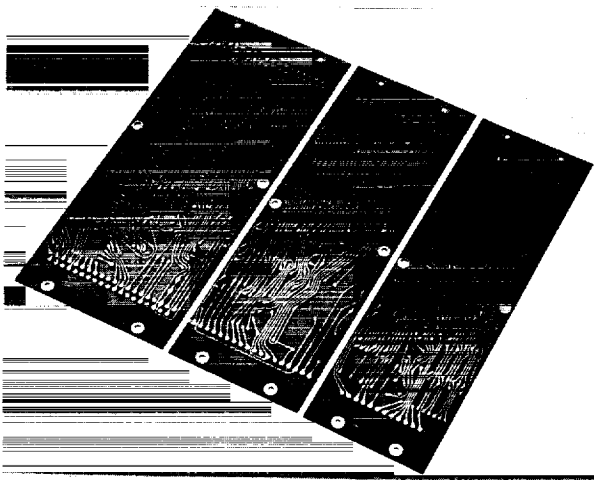


FIGURE 5-89. ETCHED CIRCUIT BOARDS FOR WELDED MODULES

module form and the fabrication is nearing completion. This model will allow evaluation of the mass encapsulation technique and will permit tests to ensure that the high impedance circuitry in the PACE is not degraded by encapsulation. Because of long delivery times, two of the modules were changed for this model so that readily available components could be substituted. Three weight simulation units have been fabricated which will be vibrated with the transponder to prove out the design.

All components used in the advanced development model were measured and tested prior to use. Approximately 3000 components were eval-

uated; the data is being analyzed and put into a form for design use. All failures have been identified and held for further study.

The complete set of fabrication drawings for the PACE unit are complete. The components for the ψ_2 angle counter have been designed into 13 welded modules of eight types. All module designs have been completed. The etched board design will not be accomplished until a decision has been reached whether or not the beam positioner counter will be time shared between the beam positioner and ψ_2 angle counter.

• **Phase Shifter Driver Product Design.** Two separate product designs of the phase shifter driver unit are being made. The first is an open-form, wired flat card breadboard unit to be used with the ADM PACE, and the second is the flight configuration spacecraft design.

The ADM design has 16 wired cards of four types contained in a modified Syncom 2 telemetry and command simulator frame with a 75-pin unit connector. The output transistors requiring a heat path whose thermal resistance is 33°C per watt or less were equipped with commercial heat sinks.

The spacecraft phase shifter drivers are packaged in two identical units each consisting of four identical etched circuit boards containing a power amplifier A(sin), a power amplifier B(cos), a dc-to-dc converter, plus a fifth etched circuit board containing the on-off drivers.

Welded module construction will be used throughout except for physically large components including one transformer, two inductors, and one capacitor which will be mounted directly to the circuit board. After each board is assembled and tested it will be mass encapsulated in a 4 lb/ft³ rigid polyurethane foam. The wiring of the five boards and the 37-pin unit connector is accomplished by point-to-point wiring and the unit is then assembled in a thin walled aluminum

frame. The solenoid driver heat sinks provide adequate heat sink of the power amplifier output transistors which dissipate 600 milliwatts.

Jet Control Electronics. Since the beam positioner has phase information as to the direction from the spacecraft to the earth, it can be utilized in the reaction control system to provide for pulsed operation of the jets at fixed angles with respect to the earth line. The reference is the center of the antenna beam which, it is assumed, has already been aligned to be coincident with the earth line.

The jet control electronics (JCE) provides the necessary electronic circuitry to operate the bipropellant propulsion control subsystem. Its modes of operation include operating the jets by sequentially timed control pulses generated in the spacecraft or permitting real time control of the jets with the use of the ground console synchronous controller.

With the logical use of the generated control pulses and with the phased array antenna beam centered on the earth, the radial jets can be pulsed through 45 spacecraft degrees such that the direction of the average thrust vector of the radial jets will be $+90$ or -90 degrees from the spacecraft-earth line and the axial jets can be pulsed (90 degrees $-B$) from the spacecraft-earth line. The angle B represents the angular displacement between the radial and axial jets (90 degrees). Both axial or both radial jets are pulsed during the same spacecraft revolution so that the average thrust direction of each coincides. For example, one radial jet is pulsed at 90 degrees and after one-half revolution the other radial jet will be pulsed at 90 degrees. The number of pulses is determined by the time duration of the command execute signal.

By rotating the antenna beam θ degrees in the direction of the spin sense, the radial and axial

jets can be sequentially pulsed ($+90 + \theta$ degrees) or ($-90 + \theta$ degrees) and ($+90$ degrees $-B + \theta$ degrees), respectively, with respect to the spacecraft-earth line.

The time reference for the sequential timing circuitry is a counter which is earth-oriented. That is, during normal operation the counter's characteristics permit selection of a reference point on the periphery of the spacecraft that will always be coincident with the antenna beam when the counter's content is 0. This counter is termed the output counter and is contained in the beam positioner electronics. It has a full scale count of 512 with each count representing $360/512$ spacecraft degrees.

It is necessary at this point to establish a reference radial jet to be termed radial jet I. It will be the first radial jet encountered while proceeding in a counter-clockwise sense from the reference point and viewing the spacecraft from the

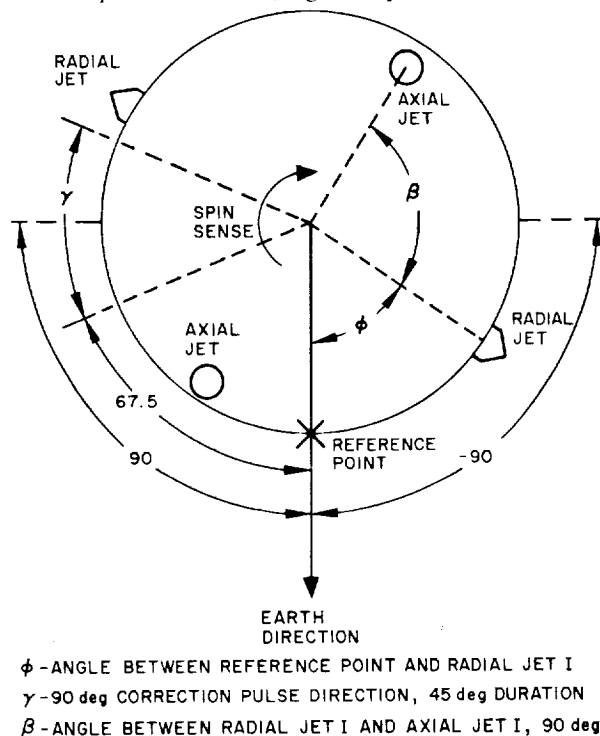


FIGURE 5-90. JET REFERENCE DIAGRAM

antenna end. The reference jet is shown in Figure 5-90 along with other pertinent items and angles.

Since radial jet I is displaced ϕ degrees from the reference point, it is necessary to detect a count of 67.5 degrees $+ \phi/360$ degrees times 512 to initiate the 90-degree pulse using radial jet I. Detection of a count corresponding to 45 degrees of subsequent spacecraft rotation is used to terminate the jet pulse. The jet pulse for a -90 degree correction is obtained by detecting a count of 67.5 degrees $+ \phi + 180$ degrees/360 degrees times 512 to initiate the pulse and then terminating the pulse after 45 degrees of subsequent spacecraft rotation. These operations are shown in Figure 5-91. Signals J100 and J101 are outputs of flip-flops and, as will be shown later in the derivations of the logic equation, represent the $+90$ and -90 -degree pulses for radial jet I. Also shown are the waveforms for signals B117, B118, and B119 which are the outputs of the three most significant flip-flops of the beam positioner output counter. Since all operations occur at multiples of 45 degrees after J100 is set, then only some function of these three signals need be used to control the set and reset input gates of J100 and J101. The remaining count is

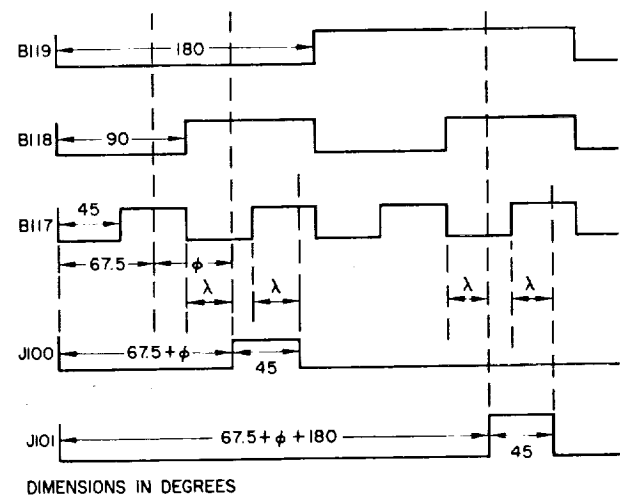


FIGURE 5-91. JET PULSE ENVELOPE

common to all four inputs and is used as a clock. Also since a flip-flop is set or reset in a 45-degree interval, the clock need only detect a fraction of 45 degrees or λ as shown in Figure 5-91. Analytically $\lambda = \phi - 22.5$ degrees. Also, a signal from the frequency lock loop indicating that the output counter is synchronous with spacecraft spin rate and the signal corresponding to that portion of the command word that selects the JCE are inputs to the set gates of J100 and J101. This is necessary to ensure that a pulse is generated only when the JCE has been selected and the reference counter's signals are correct.

With the use of ground console synchronous controller, a backup mode can be operated by real time to pulse any jet for any duration such that the direction of the average thrust vector is at any angle relative to the sun or earth lines. The circuitry for this backup mode is independent of the sequential timing circuitry.

Seven commands are necessary to control the JCE. The commands and the functions of each are tabulated in Table 5-14.

- **Pulse Mode Operation.** The block diagram of the jet control electronics is shown in Figure 5-92. The operation is as follows.

When a pulse mode command (15-1, 15-2, or 15-3) is given, gate J5 goes high which in turn enables gates J1 and J3. If F702P is high (frequency lock loop is in "lock"), flip-flop J100 will be set when the beam positioner output coun-

ter reaches the count $67.5 \text{ degrees} + \phi/360 \text{ degrees times } 512$ and will be reset 45 degrees later. Flip-flop J101 will be set when the beam positioner output counter reaches the count $67.5 \text{ degrees} + \phi + 180 \text{ degrees}/360 \text{ degrees times } 512$ and will be reset 45 degrees later. This generates the timing envelopes for J100 and J101 shown in Figure 5-91.

Also when a pulse mode command is given, one of the three latching amplifiers (J300, J301, J302) is turned on. The outputs of these amplifiers are gated together with the outputs of flip-flops J100 and J101 to fire a jet through a 45-degree angle. The logic is as follows:

$$\begin{aligned} \text{Axial No. 1 at } +90 \text{ degrees} - B &= J300 \cdot J100 \\ \text{Axial No. 2 at } +90 \text{ degrees} - B &= J300 \cdot J101 \\ \text{Radial No. 1 at } +90 \text{ degrees} &= J301 \cdot J100 \\ \text{Radial No. 1 at } -90 \text{ degrees} &= J302 \cdot J101 \\ \text{Radial No. 2 at } +90 \text{ degrees} &= J302 \cdot J100 \\ \text{Radial No. 2 at } -90 \text{ degrees} &= J301 \cdot J101 \end{aligned}$$

The latching amplifiers are used so that if the command execute is removed before a 45-degree pulse has been completed, the pulse will not be terminated early. The latch control enables the latching amplifiers only during the intervals when one of the flip-flops J100 or J101 is set.

If one of the flip-flops J100 or J101 should fail high, continuous firing of a jet could occur. The redundant latch control turns off the latching amplifiers at each clock pulse to prevent this failure mode.

- **Backup Mode Operation.** As seen in Figure 5-92, the command decoder backup mode command outputs are "ORed" together with the pulse mode signal. The backup commands are independent of the pulse mode timing circuitry. The selected jet fires for the duration of the command execute signal.

TABLE 5-14. JET CONTROL ELECTRONICS COMMANDS

Command	Function
15-1	Fire axial jets — pulse mode
15-2	Fire radial jets at +90 degrees — pulse mode
15-3	Fire radial jets at -90 degrees — pulse mode
16-1	Fire axial jet No. 1 — backup mode
16-2	Fire radial jet No. 1 — backup mode
16-3	Fire axial jet No. 2 — backup mode
16-4	Fire radial jet No. 2 — backup mode

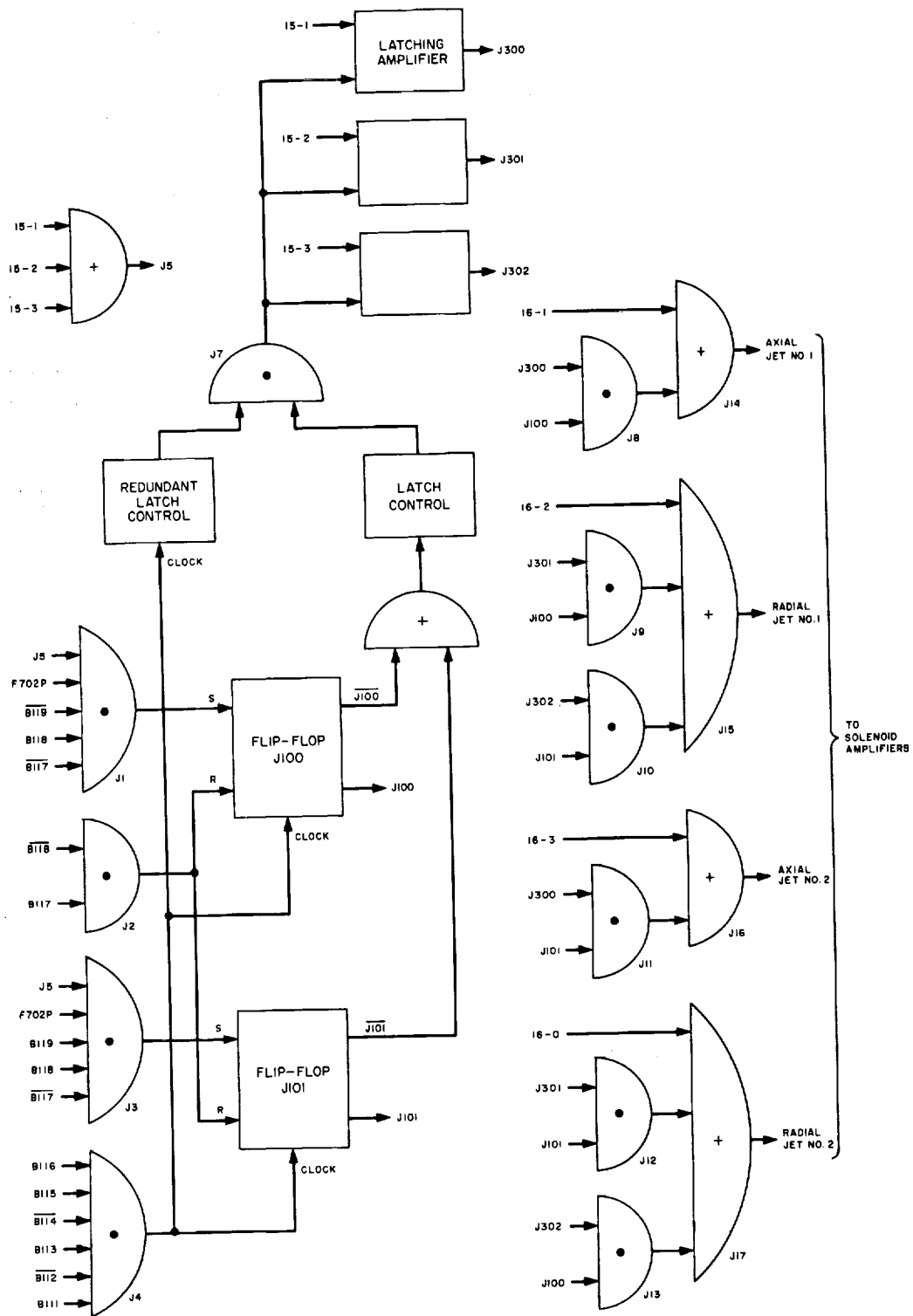


FIGURE 5-92. JET CONTROL ELECTRONICS

Jet Control Electronics Product Design. The jet control electronics consists of components packaged in three welded modules of the type described in the PACE product design section. The component density of the JCE is 32 components per cubic inch at the module level.

Solenoid Drivers. The function of the solenoid driver is to perform as an electronic switch to close the solenoid coil and unregulated bus circuit loop. Its design criteria requires that a sufficiently high current pulse be provided to saturate the solenoid and then supply a holding current for the remainder of the jet fire pulse. Also, component redundancy is such that failure of one component will not activate the jet continuously.

The solenoid driver consists of two parallel redundant preamplifiers, two redundant series power switches, a capacitor multiplier, a voltage limiter, and a current limiting resistor. The interconnection of these parts is shown in Figure 5-93. The operation is as follows.

When a signal is received from one of the jet control electronics outputs, both preamplifiers turn on and turn on both power switches. The output goes high until the effective capacitance of the capacitor multiplier has charged, at which time the capacitor multiplier changes to an open circuit. The output current then drops to a lower holding current which is determined by the coil resistance and the value of the series limiting resistor.

The voltage limiter is used to limit the snap-back voltage of the solenoid coils when the coils are deenergized; the purpose of this is to protect the transistors in the power switches. The output waveforms are shown in Figure 5-94.

Solenoid Driver Product Design. The four solenoid drivers will be designed into four identical units, each occupying a volume of 18 cubic inches and weighing approximately 0.5 pound each. Because of the high dissipation of this unit (20 watts peak) and possible high duty cycle, heat transfer was of prime concern in this unit.

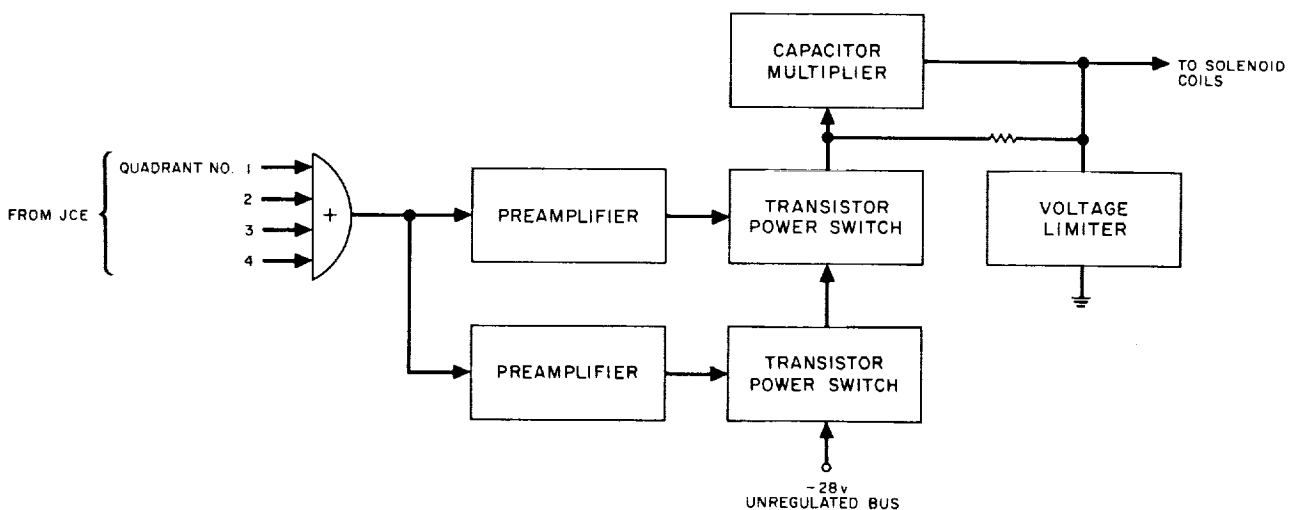


FIGURE 5-93. SOLENOID DRIVER BLOCK DIAGRAM

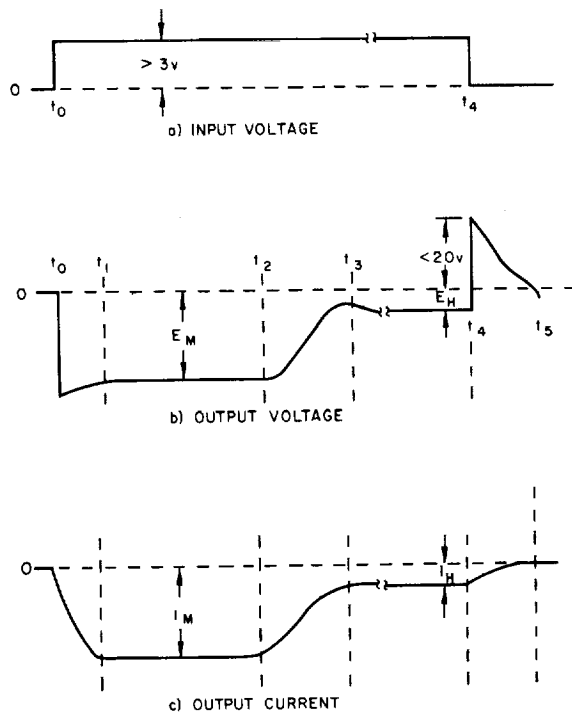


FIGURE 5-94. SOLENOID DRIVER WAVEFORMS

The mounting surface of the unit is attached directly to the structure of the spacecraft and the power transistors and diodes are mounted directly to the unit frame to provide the best possible heat path. The frame thickness of 0.040 was necessary to obtain the cross-sectional area required to dissipate the heat in the output transistors (6 watts average). The power resistors, whose characteristics are not as heat sensitive, are mounted on aluminum brackets which are then connected to the frame. The preamplifier circuit is mounted on an etched circuit card with the low dissipation components contained in a welded module. The output transistors of the preamplifiers, dissipating 300 milliwatts each, require heat sink; this is accomplished by bonding a hard anodized aluminum heat sink to the transistors with conductive epoxy and fastening it to the frame with a screw.

Interconnections between the etched circuit board, the other circuit elements, and the 9-pin unit connector are accomplished by point-to-point wiring. After assembly and test of the unit is accomplished, it is encapsulated in a 4 lb/ft³ rigid polyurethane foam, after which a cover is bonded in place.

Receiver Antenna Development. The receiver antenna is a six-element cloverleaf array. A UHF cloverleaf antenna array (*Proceedings of IRE*, December 1947, pp. 1556-1563) was scaled to obtain the desired omnidirectional pattern with horizontal polarization. A single cloverleaf test section was used in the development. The major considerations in scaling were 1) obtaining the correct resonant frequency, 2) obtaining the desired coupling from the single cloverleaf, and 3) eliminating the stray currents flowing along the outside of the coaxial line which give rise to end-fire beams of approximately the same amplitude as the desired broadside beam. Tentatively, the design problems were solved as follows:

- 1) The resonant frequency problem was solved first by decreasing the diameter of the four loops constituting the cloverleaf. However, this resulted in lower coupling.
- 2) The coupling from the cloverleaf was increased to the desired amount by attaching a strip of metal to the outer edges of the loops of the cloverleaf.
- 3) The stray currents flowing on the outside of the coaxial line were eliminated by attaching suppressor wires to the loops of the cloverleaf at a point near where they emerge from the coaxial line, and shorting these suppressor wires to the outer surface of the coaxial line $\lambda/4$ down the line in each direction from the cloverleaf. An

additional feature of the suppressor wire is that it acts as a vernier control on the frequency. This control is accomplished by varying the distance from the coaxial line to where the suppressor wire is connected to the loop.

A six-element cloverleaf array was fabricated and tested for operation at 6300 mc with a 200-mc bandwidth.

Early in the development period, the frequency and bandwidth requirements was changed to a 300-mc band centered at approximately 6165 mc. A scaled model of the 6300-mc clover-

leaf array, designed to operate at the new center frequency of 6165 mc (Figure 5-95), was fabricated. After preliminary tests were made on this array it was enclosed in a fiberglass sleeve having a wall thickness of 0.015 inch. The purpose of the sleeve is to add rigidity to the antenna and to support it in its proper operating position. Test results with this sleeve in place are shown in Table 5-15. Typical antenna patterns are presented in Figure 5-96.

As shown by the figures in Table 5-15, several problem areas still existed. A new array was constructed utilizing a larger spacing between

TABLE 5-15. MEASUREMENTS FOR 6165 MC CLOVERLEAF ARRAY WITH FIBERGLASS SLEEVE

Frequency, megacycles	VSWR	Omnidirectionality, decibels	Average Beamwidth, degrees	Beam Tilt, degrees	Measured Gain, decibels	Corrected Gain,* decibels
6015	1.90	0.5	18.8	3.0	8.0	7.9
6065	2.20	0.9	18.4	2.2	7.9	8.0
6115	2.80	0.6	18.1	2.4	6.7	7.3
6165	3.45	1.1	18.1	1.6	6.5	7.6
6215	4.1	0.4	17.9	2.4	5.9	7.4
6265	4.4	0.9	18.5	1.3	5.6	7.3
6315	4.8	0.4	17.4	1.9	4.5	6.4

*This is the measured gain corrected for mismatch loss for an assumed VSWR of 2.0 to 1 across the frequency band.

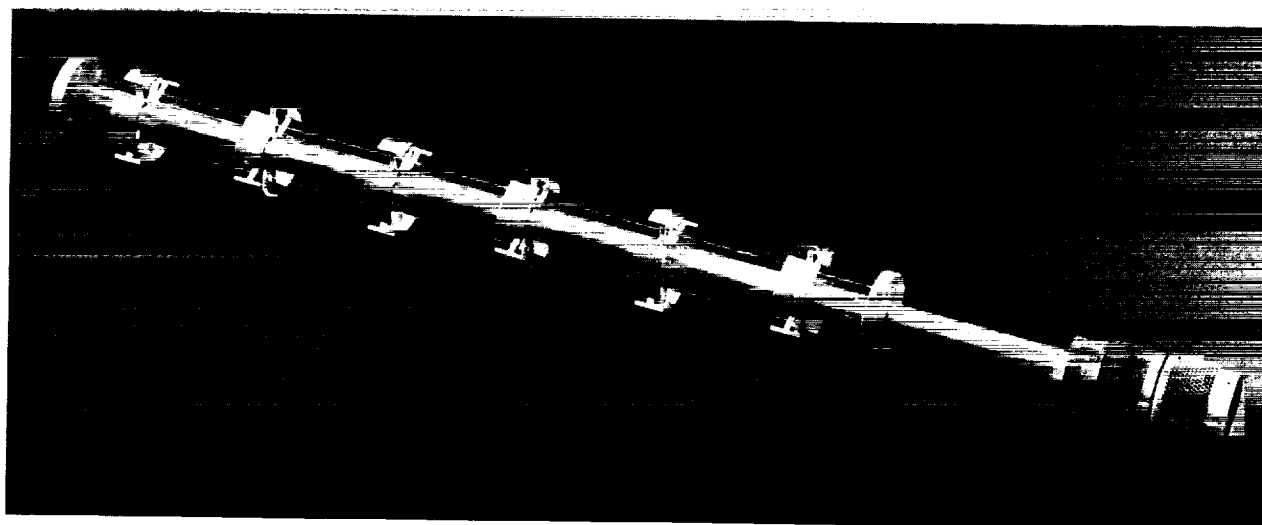
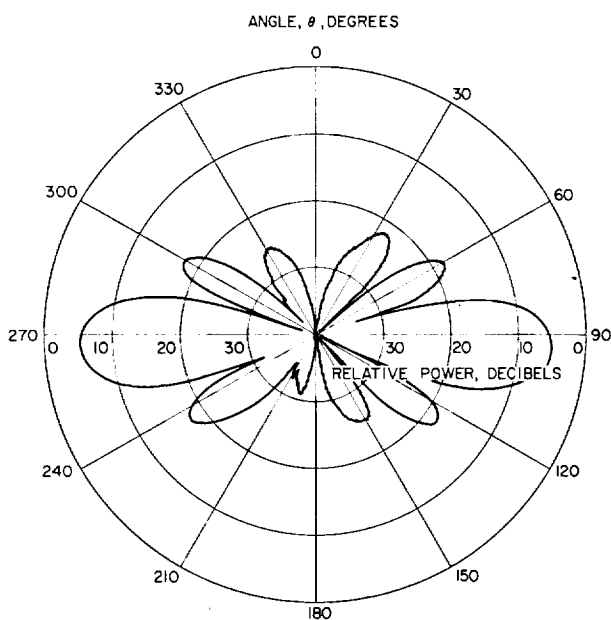
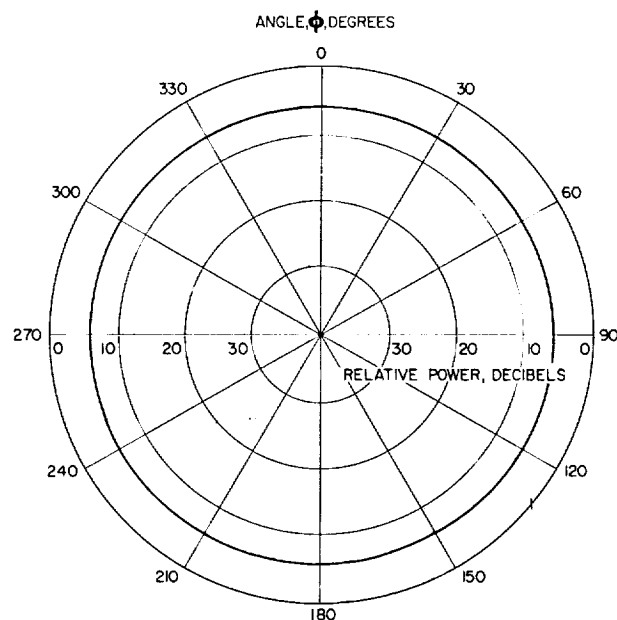


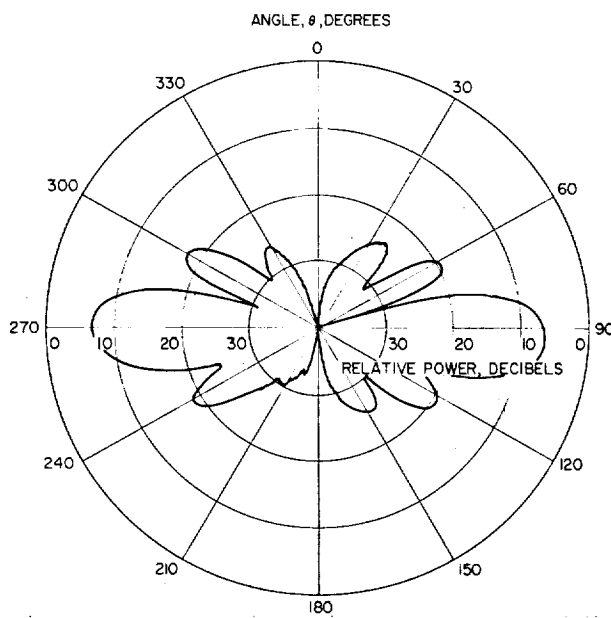
FIGURE 5-95. COLLINEAR ARRAY RECEIVING CLOVERLEAF ANTENNA



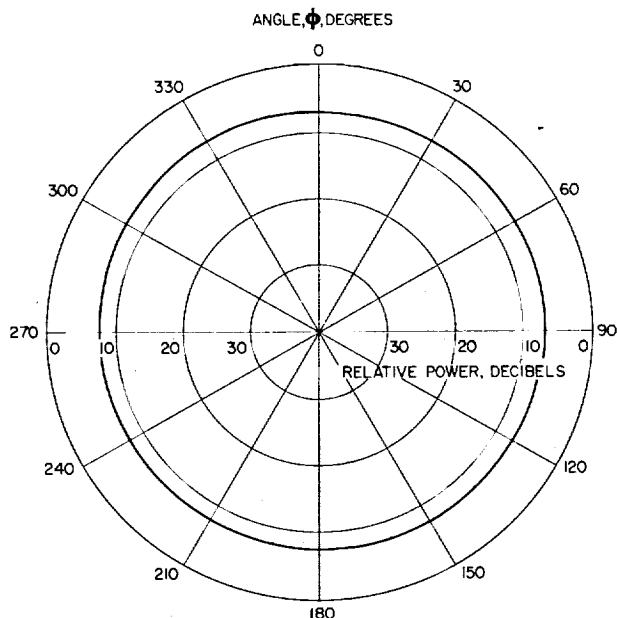
a) H-Plane, 6015 mc



b) E-Plane, 6015 mc



c) H-Plane, 6165 mc



d) E-Plane, 6165 mc

FIGURE 5-96. CLOVERLEAF ANTENNA PATTERNS

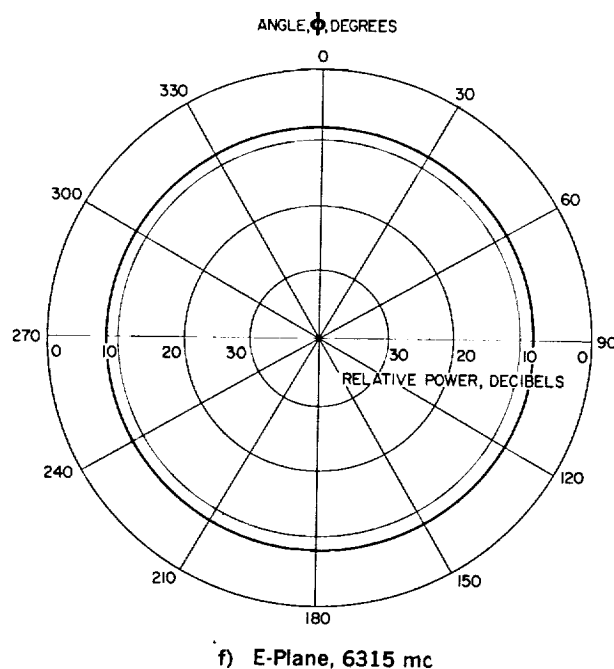
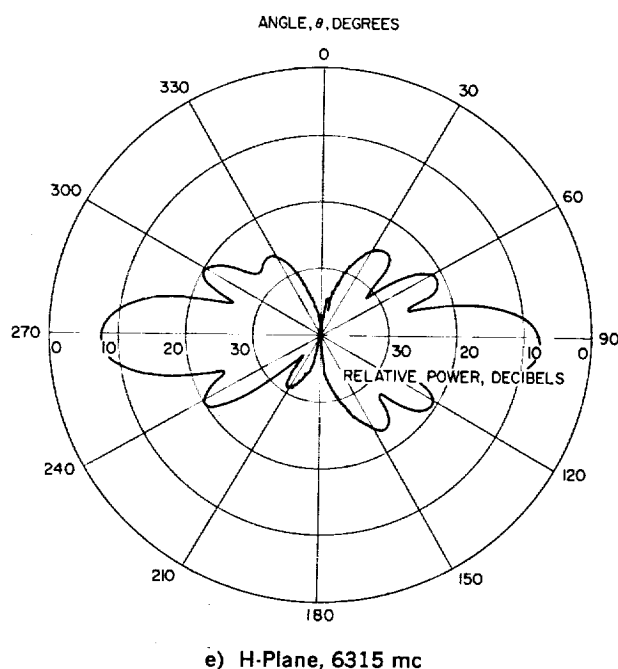


FIGURE 5-96. (Continued) CLOVERLEAF ANTENNA PATTERNS

elements to offset the sleeve effects. Care was taken in assembly of the cloverleaf elements to assure that the petals were along a cylindrical surface. The new array exhibited a beam tilt of less than 1 degree and ± 0.35 db omnidirectionality over the frequency band. Future antennas will be fabricated to these new spacing dimensions.

The VSWR of the new array indicated that the suppressor wire was probably not positioned properly. Rather than attempt refabrication at this time, the original antenna was utilized for VSWR reduction studies.

The VSWR has been reduced to less than 2.0 to 1 across the frequency band by placing a matching transformer around the center conductor at the proper position along the transmission line. It is believed that the VSWR can be reduced to about 1.8 to 1 by using a longer matching device, i.e., a two or three step transformer.

However, this will require adding a longer section of transmission line between the cloverleaf segments and the connector. At present the connector attaches above the phased array. If the longer section of transmission line is used it will be preferable to place this connector at the base of the phased array.

The gain has been improved by reducing the VSWR. A corrected gain has been calculated for a VSWR of 2.0 to 1 across the frequency band and is presented in Table 5-15. This is the gain that will be measured at the antenna terminal for a 2.0 to 1 impedance match.

Multiplexers. Receiver and transmitter multiplexers are passive RF devices which permit the use of one receiving and one transmitting antenna system for the reception and transmission of signals to and from all four communication channels. Figure 5-97 shows the multiplexer receiver. For each unit this task is accomplished by cascaded

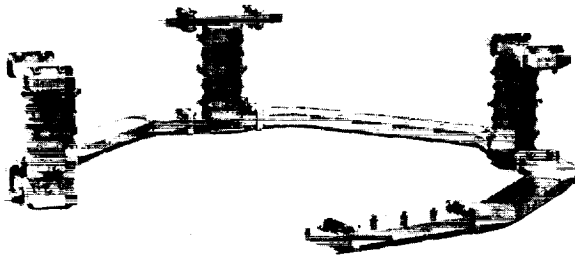


FIGURE 5-97. RECEIVER MULTIPLEXER

ing four filters along the path of a common rectangular waveguide feedline.

The first three of these four filters following the common antenna terminal are dual mode, circularly polarized, directional, three-section bandpass filters in circular waveguide which are coupled to the broadwall of two rectangular waveguides, one of which is the common feedline and the other the output.

The fourth filter is a three-section bandpass filter for the receiver multiplexer and a four-section filter for the transmitter multiplexer in rectangular waveguide which terminates the common feedline. All output terminals and the common antenna terminal are via coaxial to-rectangular waveguide transitions using OSM coaxial connectors. Specifications for the multiplexers are given in Table 5-16.

TELEMETRY AND COMMAND SYSTEM

Telemetry and Command Antenna System. The telemetry and command antenna system includes the whip antennas, hybrid baluns, and diplexers. For a four-quadrant spacecraft configuration, two complete antenna groups are required. Each group is composed of eight whip antennas, a dual hybrid balun, and two diplexers interconnected as shown in Figure 5-98.

Each set of eight whips — one set radial to the spacecraft, and the other 30 degrees off the

TABLE 5-16. MULTIPLEXER SPECIFICATIONS

	Receiver	Transmitter
Center frequency, mc		
Channel 1	6019	3992
Channel 2	6108	4051
Channel 3	6212	4120
Channel 4	6301	4179
Insertion loss, db (maximum)		
At f_0	1.0	1.0
At $f_0 \pm 17.5$ mc	1.4	
At $f_0 \pm 12.5$ mc		1.0
At $f_0 \pm 15$ mc		3.0
Bandwidth (1.3 VSWR), mc (minimum)	25	25
Rejection ($f_0 \pm 64$ mc), db (minimum)	22	25
Weight, pounds (maximum)	3.2	5.2

spin axis, as shown in Figures 5-99 and 5-100, operates independently. Each whip is driven 45 degrees out of phase with the adjacent whip in its set. The resulting patterns for each set at 136 mc and 148 mc are shown in Figure 5-101.

The hybrid balun configuration is shown schematically in Figure 5-102. A photograph of the hybrid balun is shown in Figure 5-103.

Each hybrid balun feed system has two input connectors coupling energy from the two diplexers and eight output connections for eight whip antennas. The hybrid balun delivers equal power from either of two telemetry transmitters to the eight whips with 45 degrees phase difference between adjacent whips. The phase sequence is in the opposite direction for the two transmitters. With this arrangement, power is not coupled from one transmitter to another transmitter. Extra one-eighth wavelength sections are included to prevent shorting of remaining whips if a whip is lost.

Telemetry and Command Diplexer. A frequency diplexer is required to connect the telemetry transmitter and command receiver to

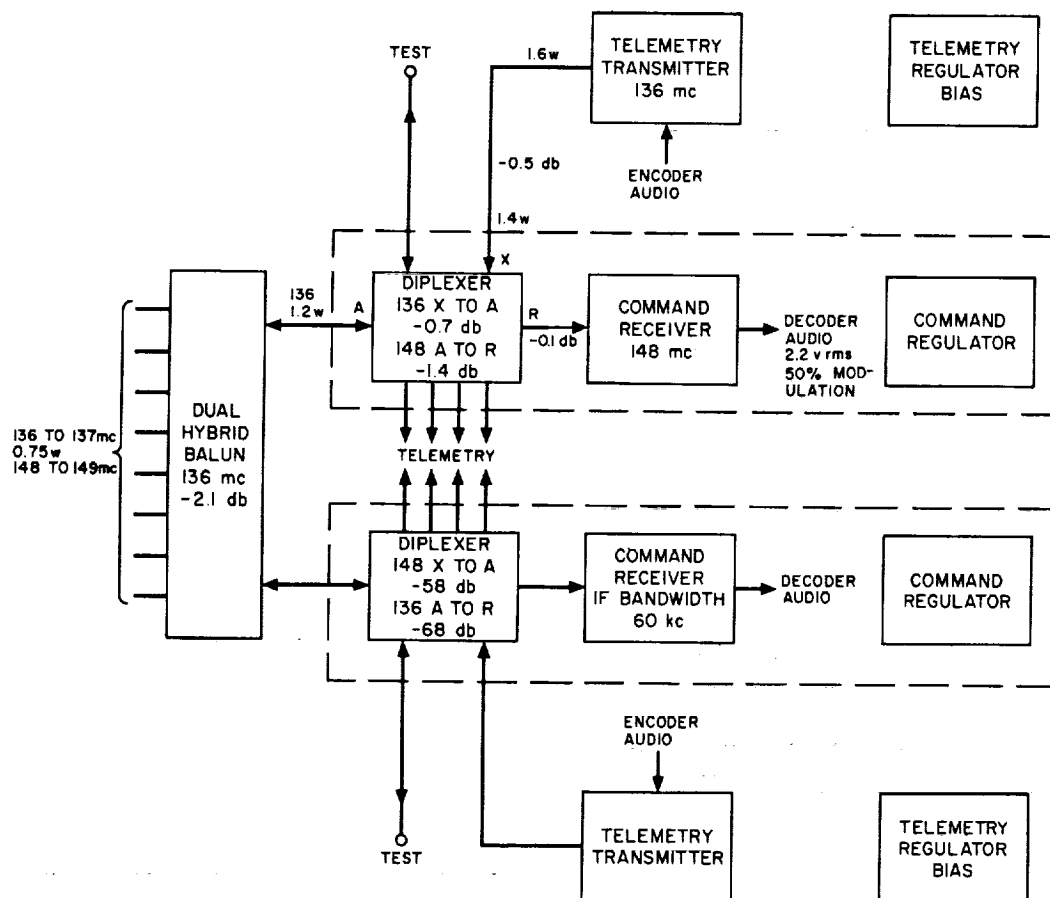


FIGURE 5-98. RECEIVER-TRANSMITTER SUBSYSTEM

the antenna without introducing excessive transmitter power into the receiver input. The rejection filter diplexer has the following characteristics:

Transmitter to RF output insertion loss at 136 mc	0.7 db
Receiver to RF input insertion loss at 148 mc	1.4 db
Transmitter to receiver isolation at 136 mc	68 db
Transmitter to receiver isolation at 148 mc	58 db

These characteristics are shown in Figure 5-104.

Telemetry Transmitter. The telemetry transmitter (Figures 5-105 and 5-106) is identical to the one utilized in Syncom 2 and no new development work has been required to meet Advanced Syncom objectives. The characteristics of the transmitter are as follows:

Frequency	136.470 or 136.980 mc
Output impedance	50 ohms
RF power output	1.6 watts
Modulation index	0 to 2 radians
Bandwidth	>24 kc

5. Spacecraft System Design

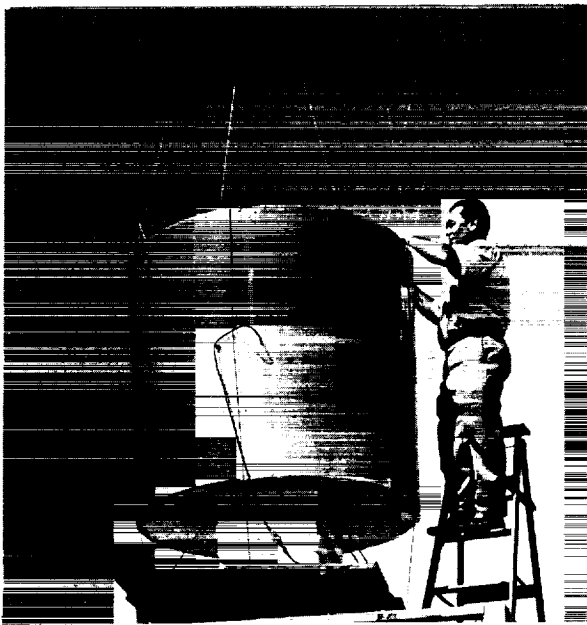
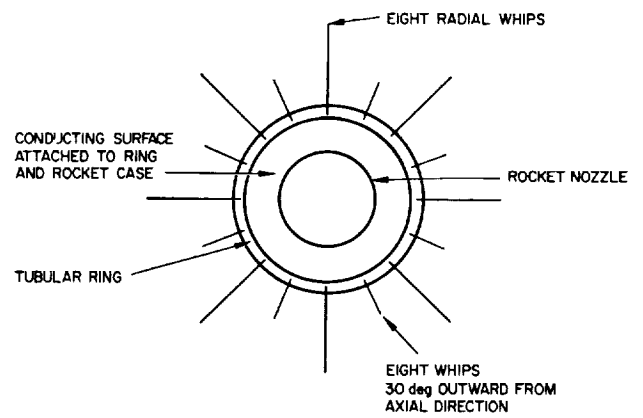
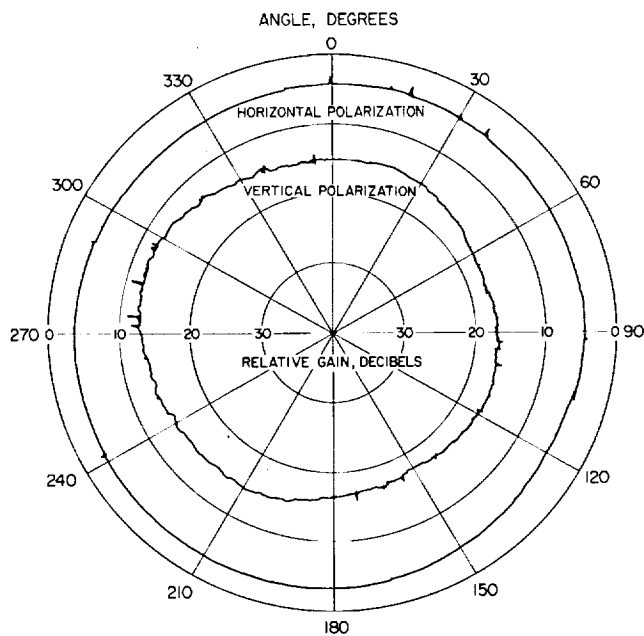


FIGURE 5-99. SPACECRAFT MOCKUP FOR TELEMETRY AND COMMAND ANTENNA TEST

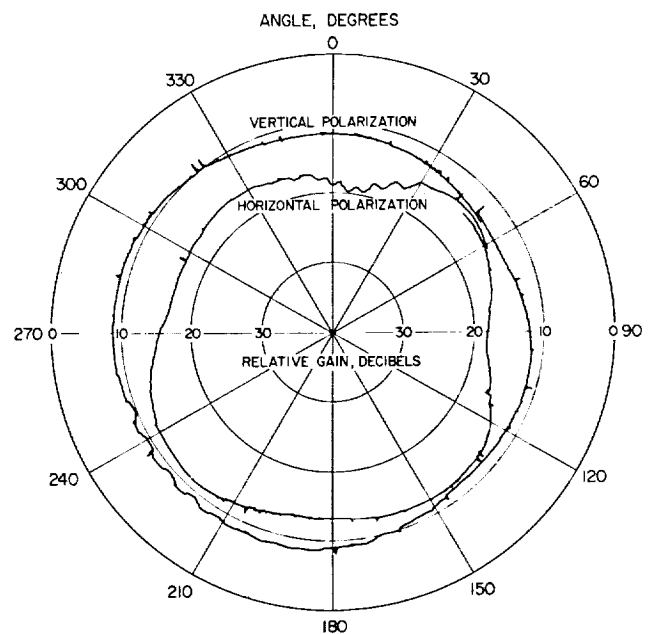


ALL WHIPS APPROXIMATELY 23 in.

FIGURE 5-100. NOZZLE END VIEW SHOWING WHIP ANTENNA CONFIGURATION

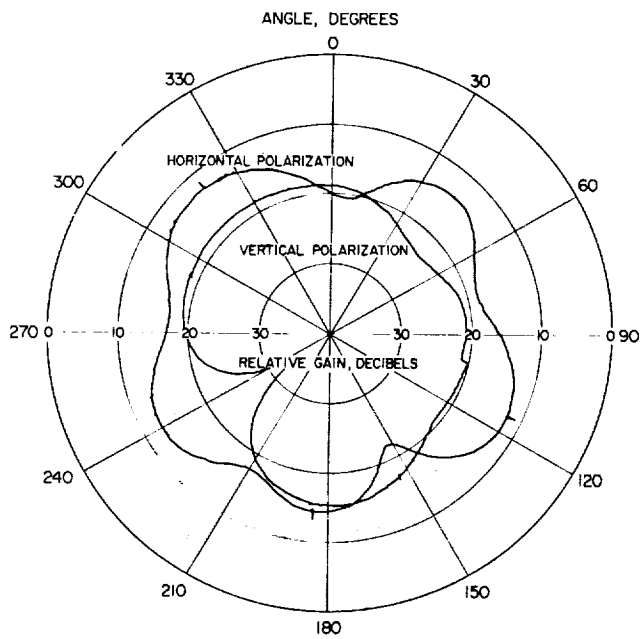


a) 8-Whip 30 Degrees to Spin Axis, 136 mc

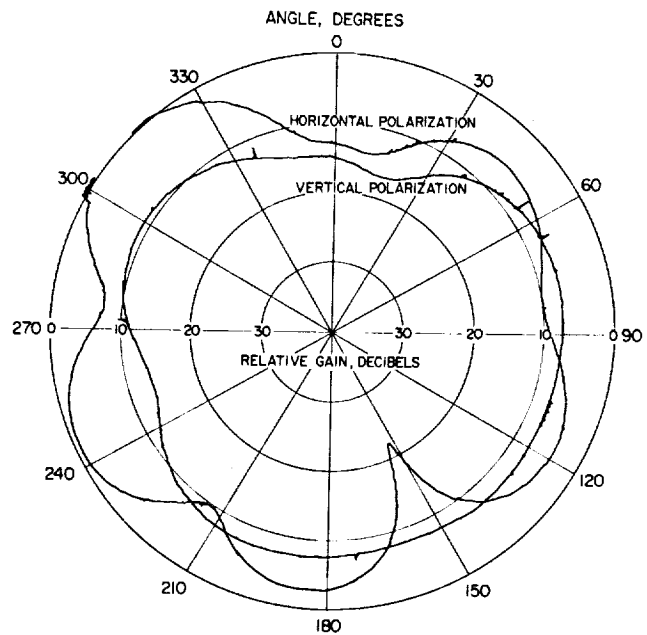


b) 8-Whip 30 Degrees to Spin Axis, 148 mc

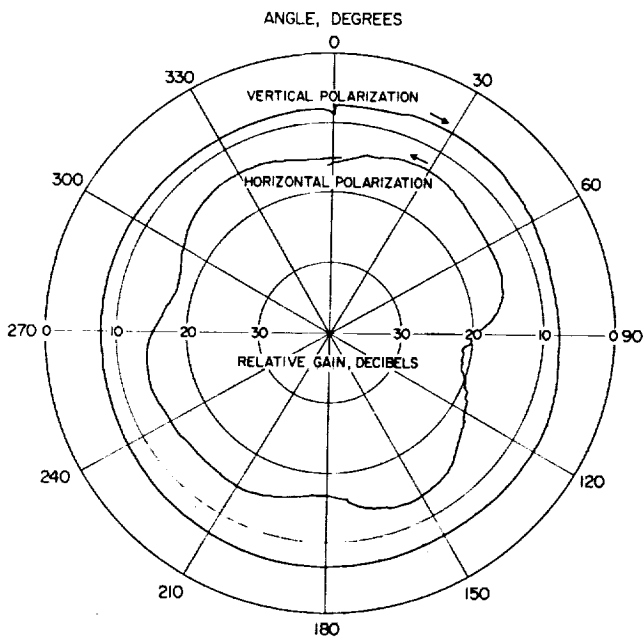
FIGURE 5-101. TELEMETRY AND COMMAND ANTENNA PATTERNS



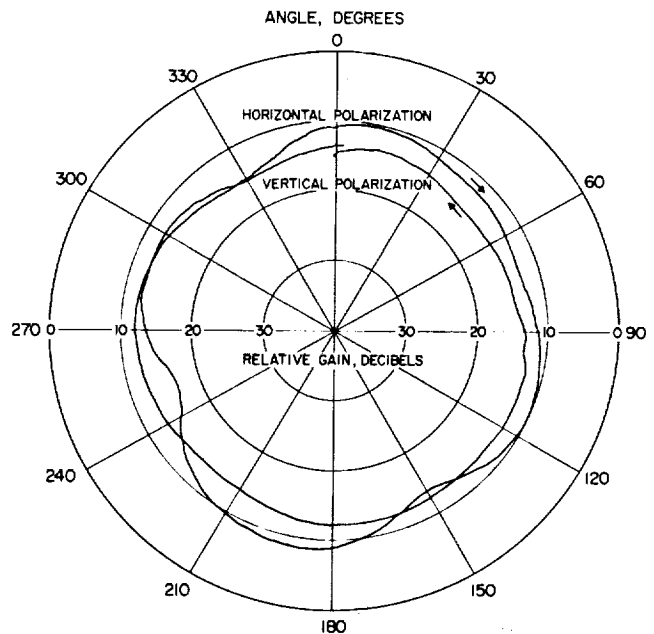
c) 8-Whip 30 Degrees to Spin Axis, 136 mc



d) 8-Whip 30 Degrees to Spin Axis, 148 mc

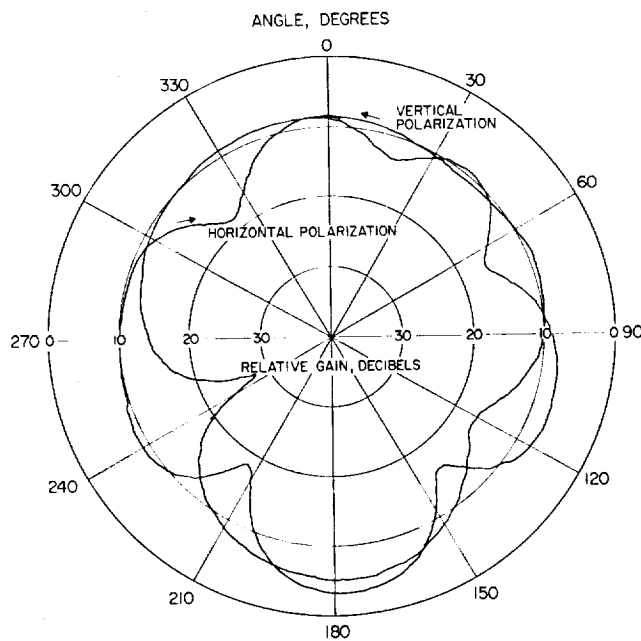


e) 8-Whip 90 Degrees to Spin Axis, 136 mc

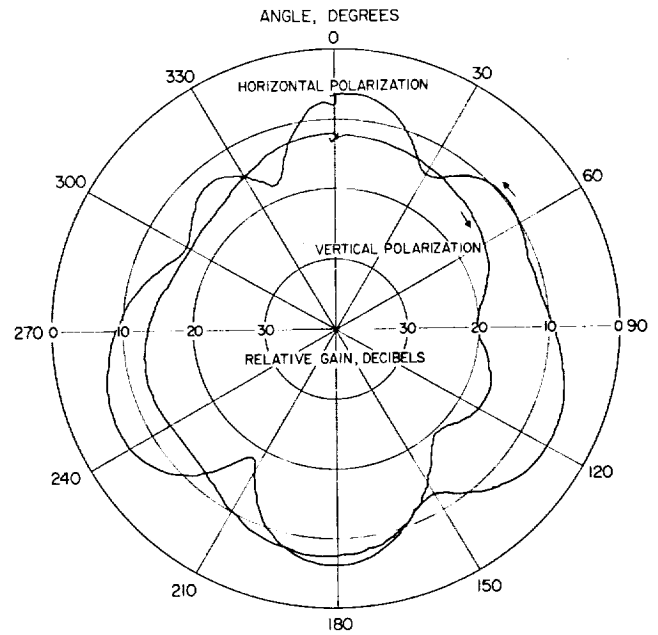


f) 8-Whip 90 Degrees to Spin Axis, 148 mc

FIGURE 5-101. (Continued) TELEMETRY AND COMMAND ANTENNA PATTERNS



g) 8-Whip 90 Degrees to Spin Axis, 136 mc



h) 8-Whip 90 Degrees to Spin Axis, 148 mc

FIGURE 5-101. (Continued) TELEMETRY AND COMMAND ANTENNA PATTERNS

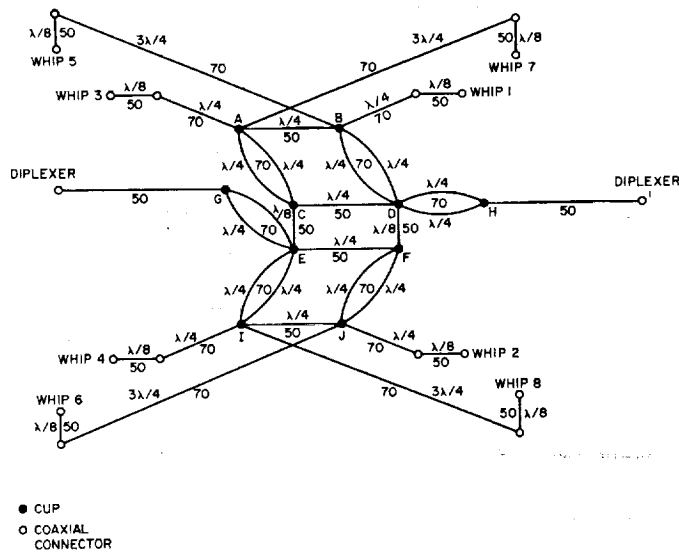


FIGURE 5-102. HYBRID BALUN CONFIGURATION

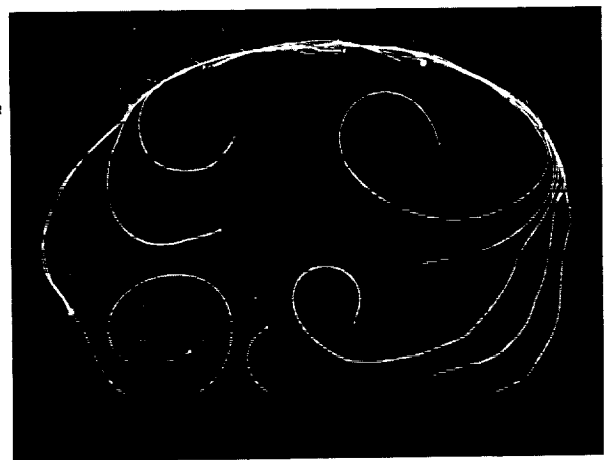


FIGURE 5-103. HYBRID BALUN

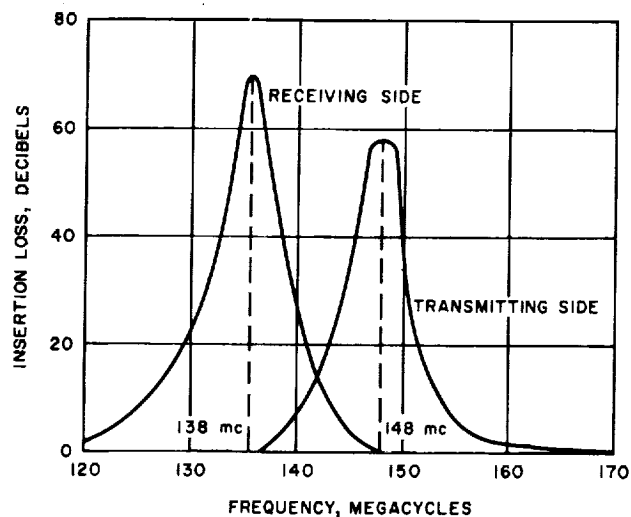


FIGURE 5-104. APPROXIMATE DIPLEXER CHARACTERISTICS

Spurious output	< -30 dbw
Oscillator stability	± 0.003 percent
Voltage	-24 volts
DC input power	6 watts
Weight	8 ounces each
Operating temperature	20 to 130° F

The transmitter employs six transistors. The crystal oscillator operates at 68 mc. The output of the oscillator is phase-modulated by the encoder information and then amplified through a four-stage RF amplifier. The final amplifier output drives a balanced varactor doubler stage. Fixed bias is applied to the doubler to correctly

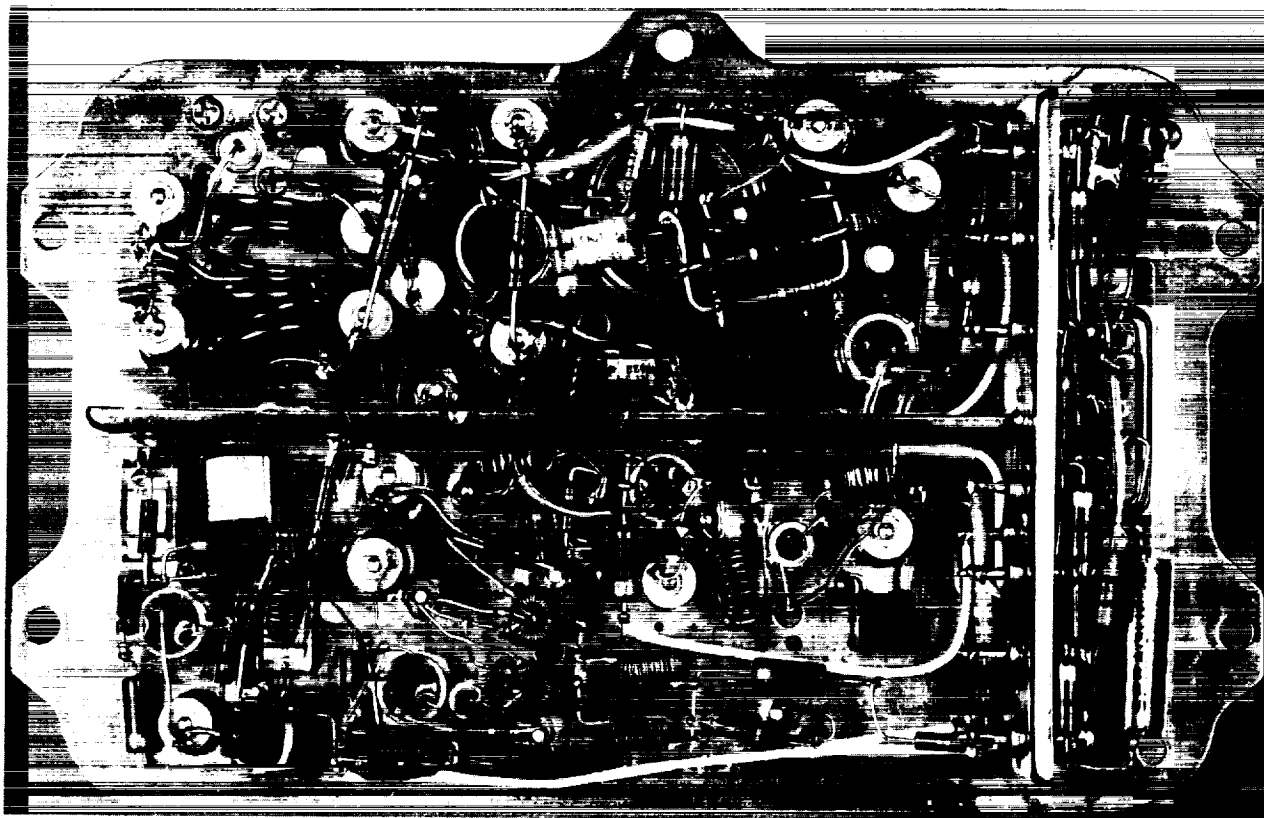


FIGURE 5-105. TRANSMITTER TELEMETRY

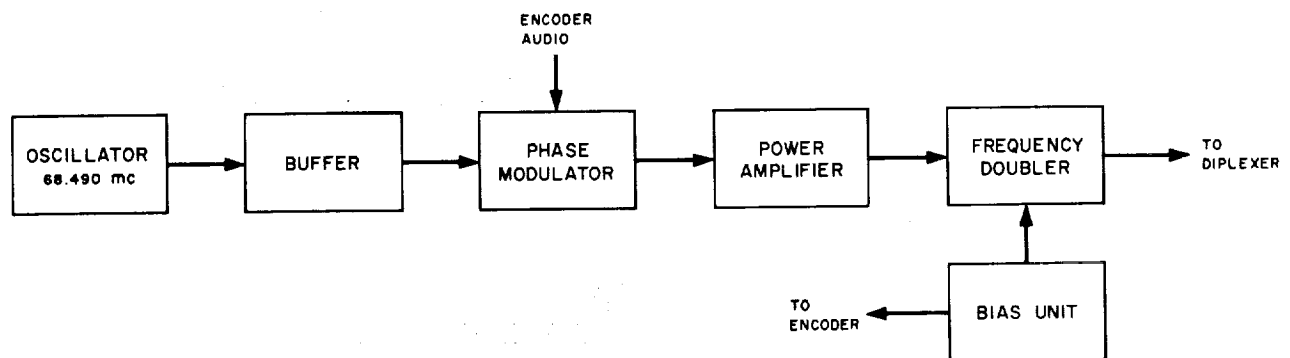


FIGURE 5-106. BLOCK DIAGRAM OF TELEMETRY TRANSMITTER

set the operating point. The output of the doubler provides 1.6 watts of RF signal.

Command Receiver. The command receiver (Figures 5-107 and 5-108) is almost identical to the Syncom 2 command receiver, and very little development work has been required to meet Advanced Syncom objectives. The transistors in the RF amplifiers have been changed to yield a lower noise figure than was previously obtained. The command receiver characteristics are as follows:

Carrier frequency	148 mc
RF input impedance	50 ohms
Noise figure	5 db
RF input overload	-70 dbw
IF bandwidth	60 kc
Image rejection	60 db
Oscillator stability	± 0.003 percent
DC voltage	-24 volts
DC power	500 mw
Weight	10 ounces
Operating temperature	20° to 120° F

The command receiver is an eleven-stage single conversion superheterodyne receiver. Two RF amplifier stages are employed prior to the mixer. The local oscillator input is generated by a crystal-controlled oscillator operating at approximately 118 mc. A 60 kc bandwidth crystal filter precedes the IF amplifier to effectively limit the noise bandwidth of the receiver. Five stages of IF amplification increase the signal to sufficient level for detection. The detected output is amplified and then fed to the decoder filters.

Telemetry Encoder. The Advanced Syncom telemetry encoder provides the telemetry transmitter with information to be sent to the ground receiving stations. The signal out of the encoder will be used to phase-modulate the transmitter carrier. Types of information that must be telemetered are spacecraft control data such as antenna position and spin speed and spacecraft engineering data such as structural temperatures and battery voltages.

Telemetry System. Telemetry requirements for Advanced Syncom involve real time processing and display of current engineering and control data on the spacecraft status to permit operation of the spacecraft. A NASA Goddard Space Flight Center standard pulsed frequency

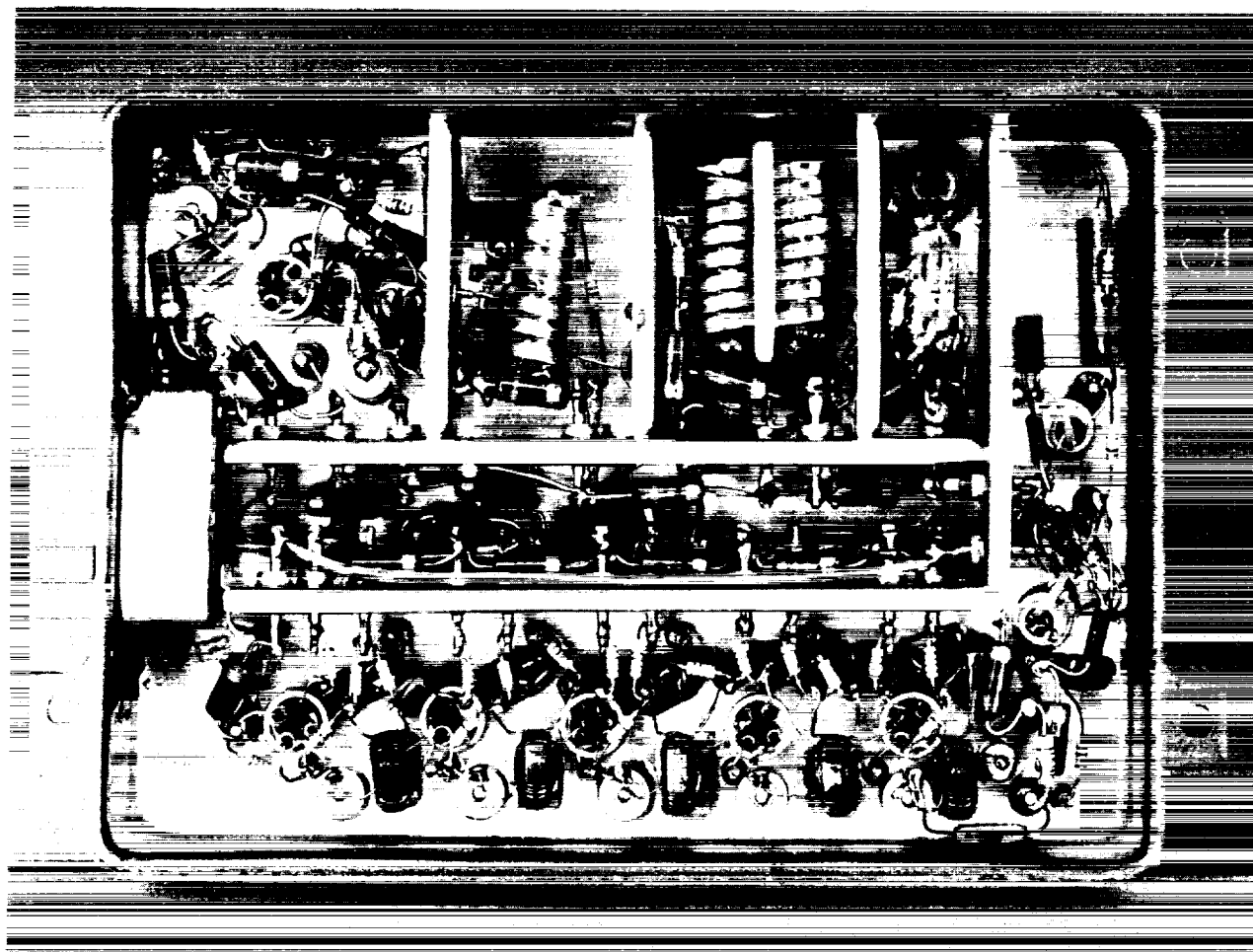


FIGURE 5-107. COMMAND RECEIVER

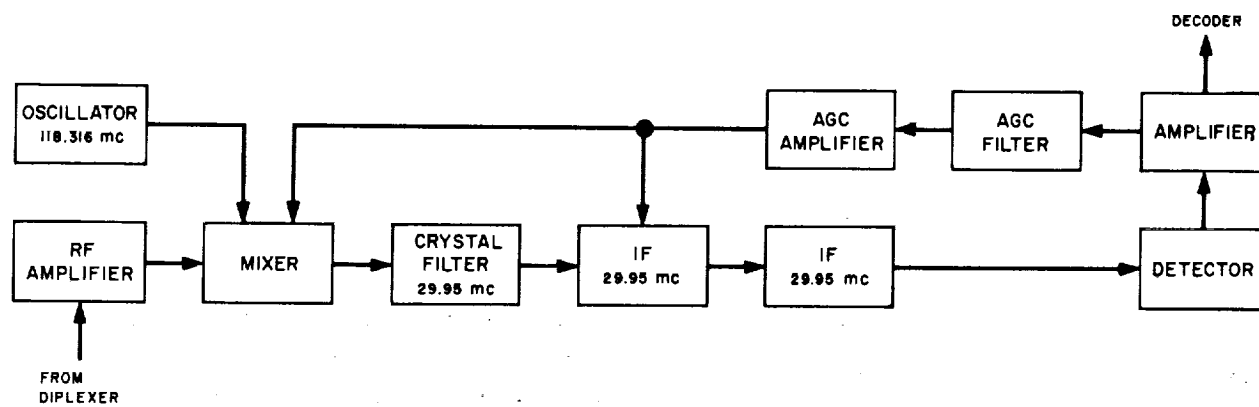


FIGURE 5-108. BLOCK DIAGRAM OF COMMAND RECEIVER

5. Spacecraft System Design

modulation (PFM) system is utilized for this data. In addition to the PFM data, other data must be transmitted in real time: the solar sensor pulses and the command execute tone with provisions for modulation of the tone by signals indicating that certain specific commands have been performed.

PFM System. In the PFM system, the information to be telemetered is encoded into subcarrier oscillator frequencies according to a specified format. The frequency of the oscillator is controlled by its input voltage which is the output of transducers and other analog signal sources. Time division multiplexing is used to accommodate the many inputs.

The format (Figure 5-109) consists of eight frames with 16 channels per frame assigned as follows:

Channel 0	Frame sync consisting of a T second reference burst followed by a 3T second sync burst
Channels 1 to 7	Series of 2T second bursts each preceded by a 2T second reference burst
Channels 8 to 15	Continuous data burst

During channels 8 to 15, the supercommutated solar panel output and transponder receiver signal strength data are not interrupted by a reference burst, to preclude possible loss of information.

The system frequencies are as follows:

Information rate	4.85 ± 5 percent channels/second
Data frequency band of subcarrier oscillator	5.0 kc to 15.0 kc

$$T = \frac{1}{4(48.5)} = 5.15 \text{ milliseconds}$$

Reference frequency 16.0 kc

Basic frame sync frequency 4.45 kc

To discriminate between frames, alternate sync bursts are stepped up in frequency from 4.45 kc. This stepped up frequency is the frame number encoded into a corresponding digital input level. The channel rate, which is a deviation from the NASA standard of a nominal 50 channels per second, was chosen because a stable 194 pps output is available from the spacecraft central timer.

The encoder is shown in Figure 5-110. Time selection of the data, reference, and sync bursts is performed by the selection counter and diode logic in conjunction with the commutator.

Logical combinations of the true and false outputs of the nine stage counter (E101-E109) determine which input is to be applied to the subcarrier oscillator. Figure 5-111 indicates the relation between the 9 bits of the selection counter and the PFM format. The basic driving frequency is the 194 pps central timer output. A detailed description of the selection scheme is presented in the following discussion.

The frequency of the subcarrier oscillator is linearly controlled by the level of the dc input voltage. Data band limits of 5 kc and 15 kc correspond to 0 and -5 volt control levels. During the data burst intervals, the data level limiter, E801, restricts the input of the subcarrier oscillator to the 0 to -5 volt range as required by the NASA/GSFC specification. The reference and sync are also generated by applying the appropriate voltage levels to the subcarrier oscillator. These are obtained from the voltage standard circuit, E703, which also provides the

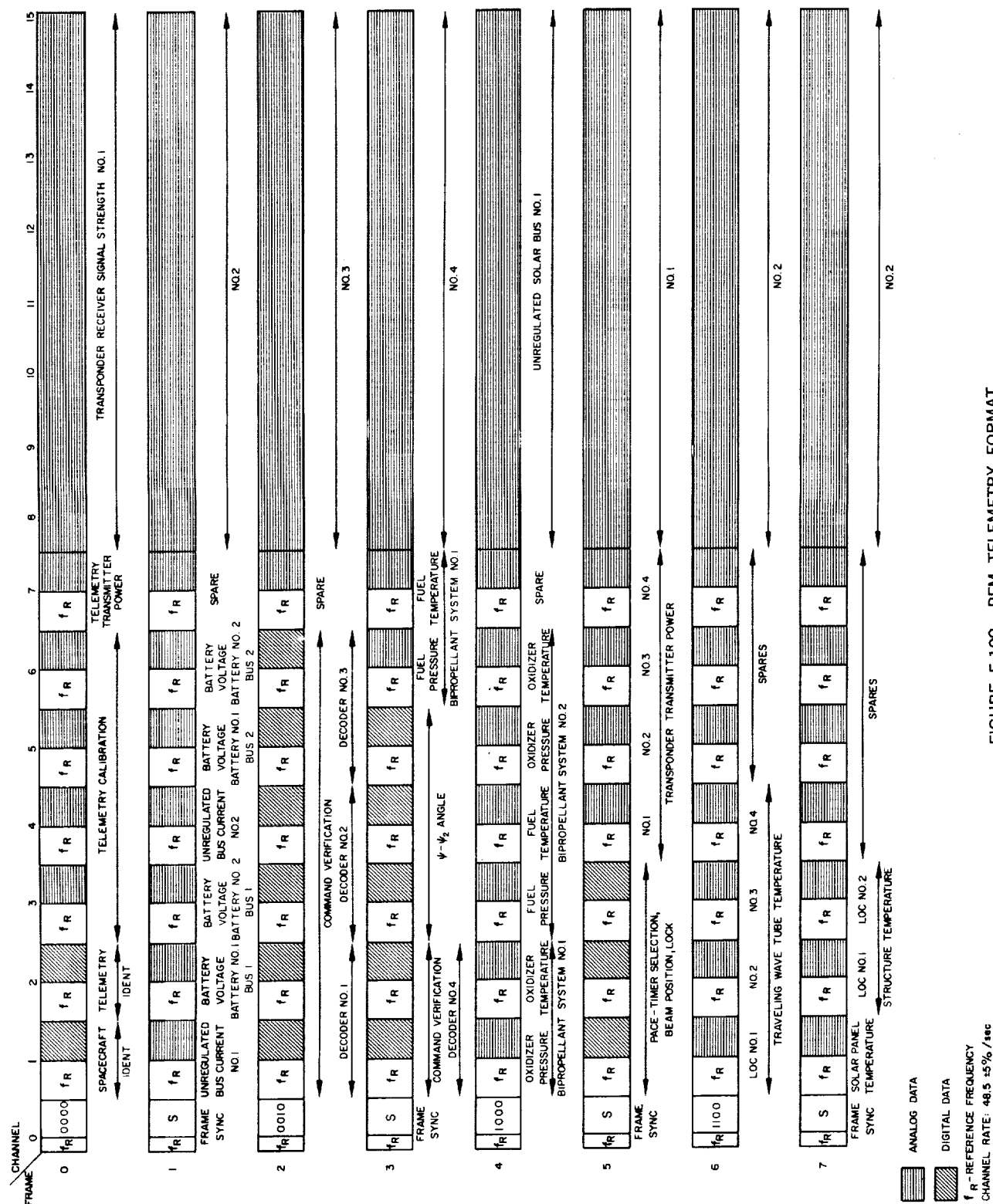
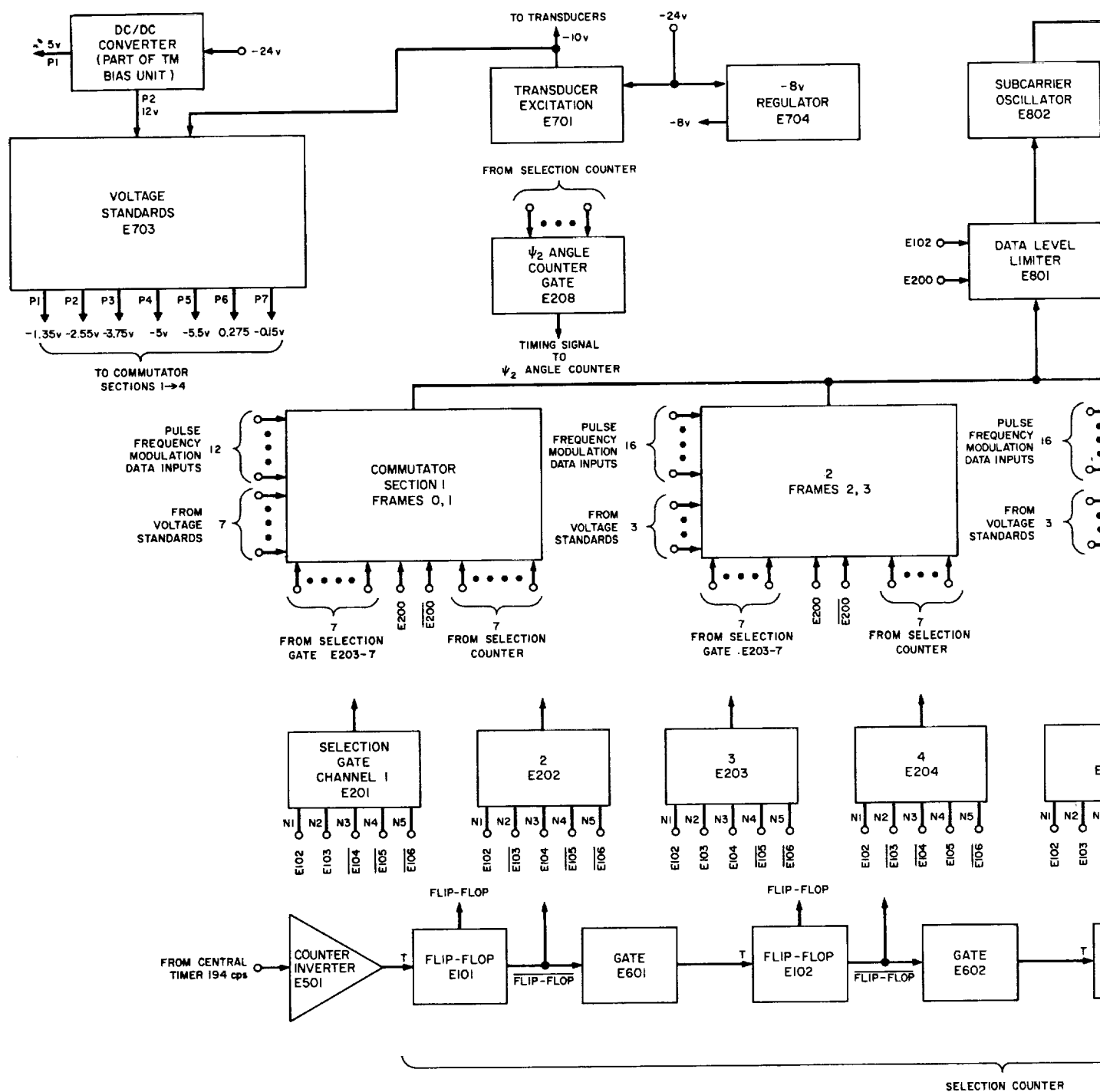
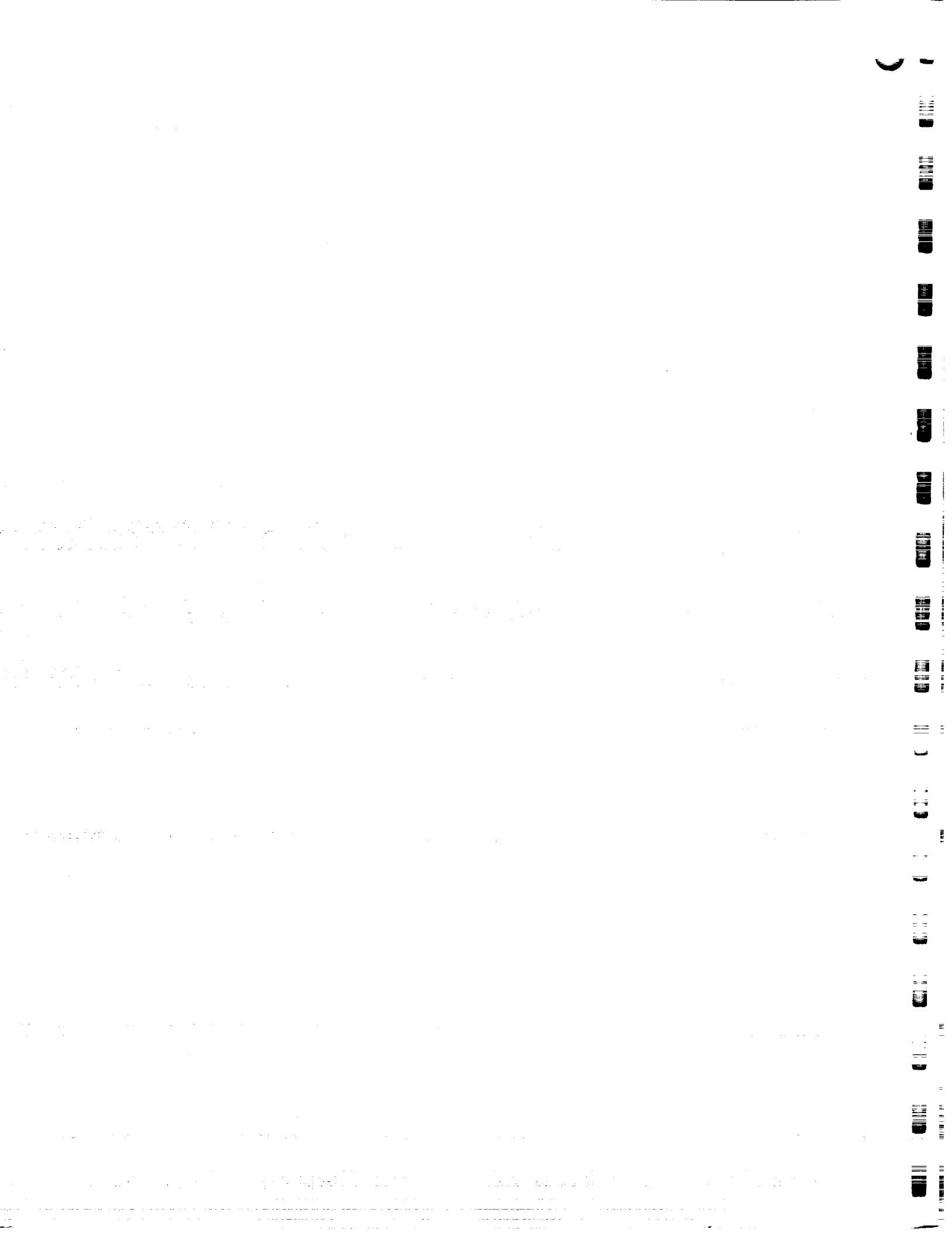


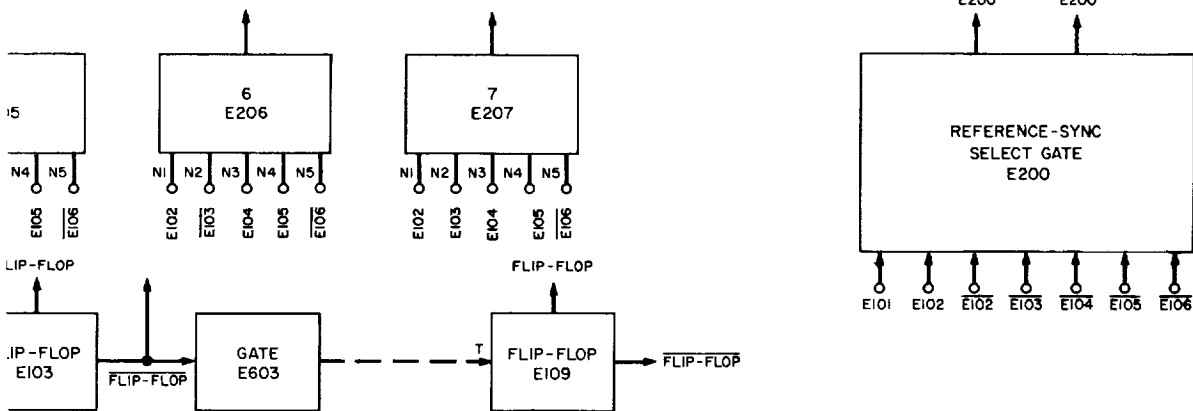
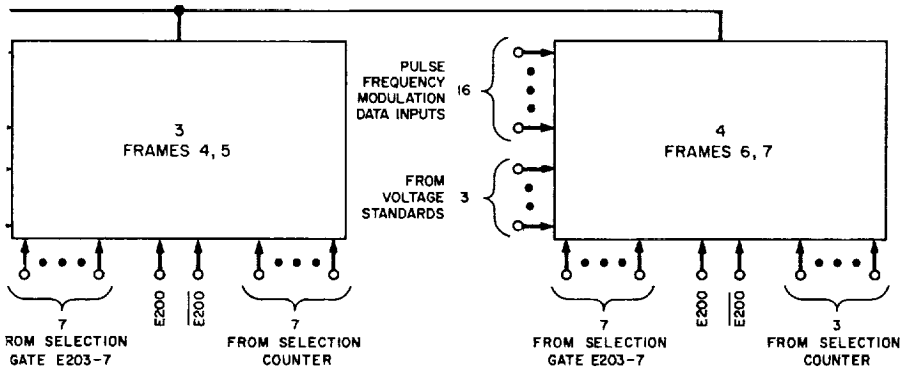
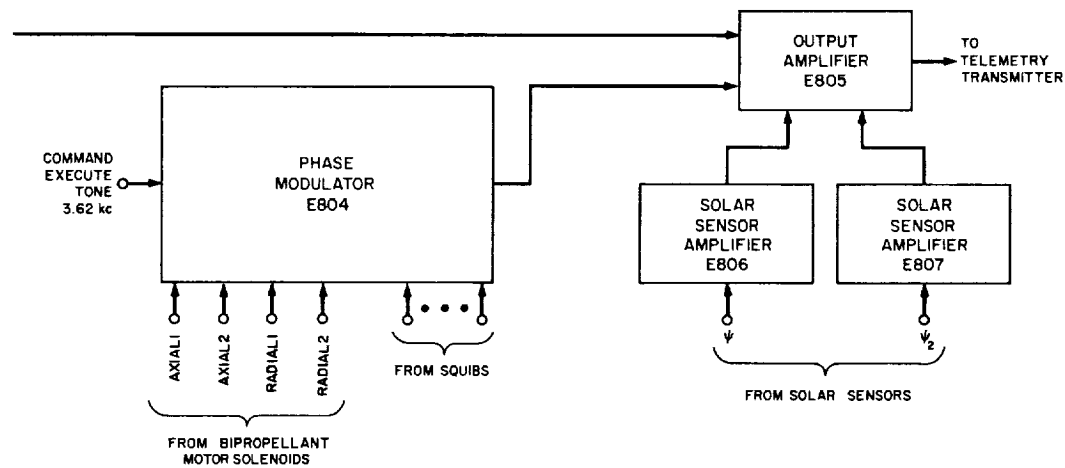
FIGURE 5-109. PFM TELEMETRY FORMAT



FLIP-FLOPS E101-9 ARE COUNTER FLIP-FLOPS
GATES E601-8 ARE COUNTER INTERSTAGE GATES

FIGURE 5-110. TELEMETRY ENCODER BLOCK DIAGRAM





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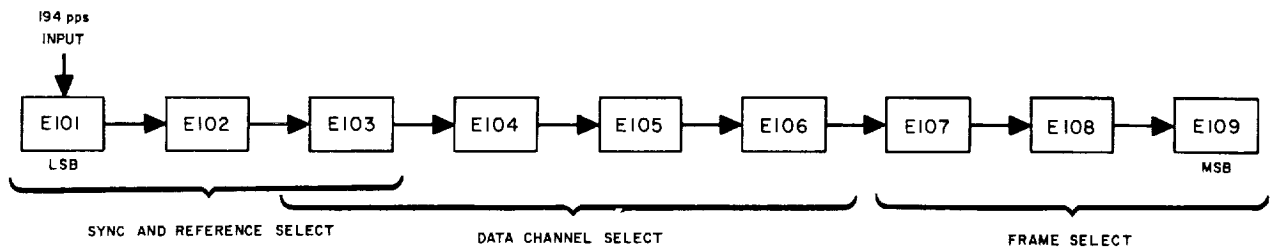


FIGURE 5-111. SELECTION COUNTER OUTPUT RELATION TO PFM FORMAT

TABLE 5-17. E703 OUTPUT AND DIGITAL LEVELS

E703 Output Voltage	Digital Level	Frequency, Megacycles	Use
+0.275	—	4,450	Odd frame sync
0	—	5,000	Calibration
-0.15	0000	5,300	Frame 0 sync
-1.35	0100	7,700	Frame 2 sync and calibration
-2.55	1000	10,100	Frame 4 sync and calibration
-3.75	1100	12,500	Frame 6 sync
-5.00	—	15,000	Calibration
-5.50	—	16,000	Reference

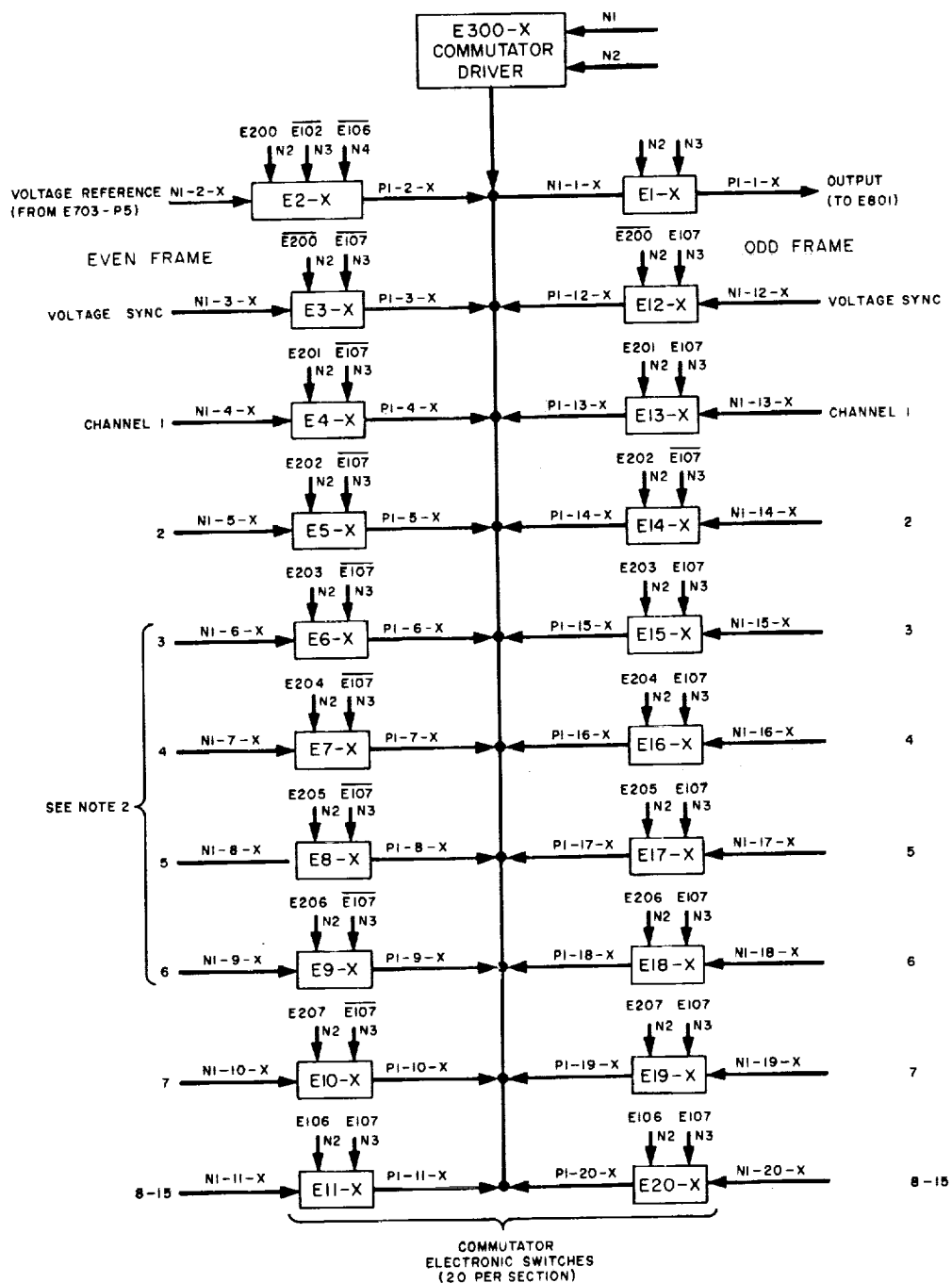
telemetry calibration voltages. Table 5-17 lists the dc output levels of E703 and the corresponding digital levels and uses.

The telemetry transmitter bias unit supplies the positive voltage (+12 volts) needed for the 4450 cps sync frequency. This voltage also permits considerable simplification of the commutator. The positive voltage required for the logical "zero" state level of the selection counter is also obtained from the transmitter bias unit. An accurate -10 volt transducer excitation, E701, supplies a reference for the temperature and pressure sensors. The ψ_2 angle counter gate generates a negative-going trigger which is used as the timing signal for resetting the angle counter. The output of the subcarrier oscillator goes to the output amplifier which provides an impedance

transformation before the signal is applied to the transmitter.

- **Selection Logic Description.** The commutator, consisting of four identical sections, performs time-division-multiplexing of the PFM inputs to generate the PFM format. The gating signals are obtained from logical combination of the counter outputs. Each commutator section is selected, one at a time, for a period of 2 frames. The block diagram of a commutator section is shown in Figure 5-112. Each block represents one solid state switch in series connection. There are two series switches between any input voltage and the subcarrier oscillator. The first-level switches are E2-E20, and the second-level switch is E1. When the commutator section is selected, switch E1 is on continuously. Only one of the first-level switches is on at any given time. The input to switch E2 is a fixed voltage corresponding to the reference frequency, 16.0 kc. The inputs to switches E3 and E12 are fixed voltages corresponding to the frame sync frequencies. The inputs to the other switches come from the various transducers and voltage sources. Figure 5-113 shows the series switch.

The commutator driver, E300, is a bootstrap circuit which provides the base drive current for switch E1 and the selected first-level switch. During the time for which the section is not selected, the current supplied by the driver is reduced by a factor of 2 since switch E1 is not



E1-X INPUTS (N1, N2)

SWITCH NO.	N2	N3
E1-1	E108	E109
E1-2	E108	E109
E1-3	E108	E109
E1-4	E108	E109

E300-X INPUTS

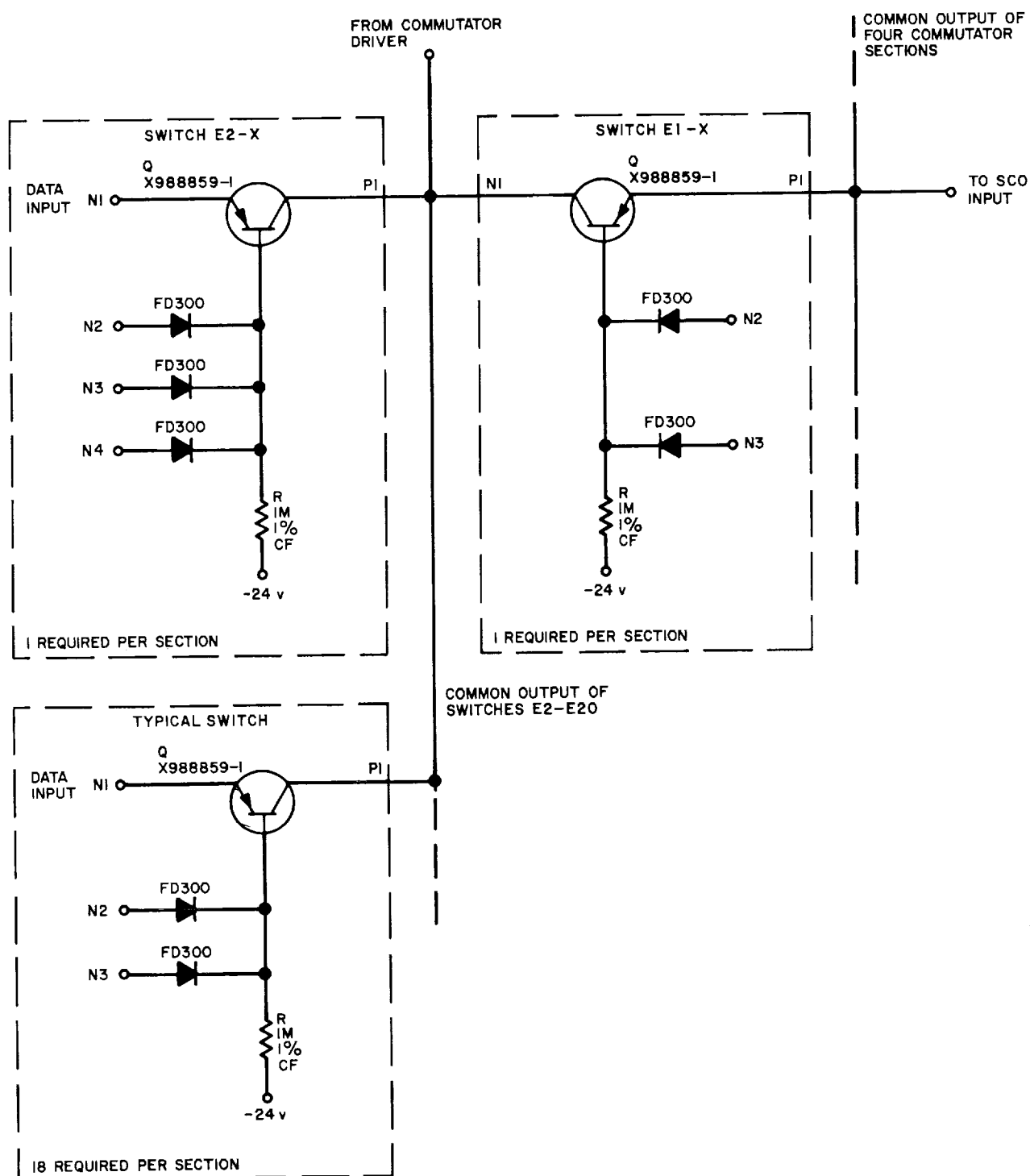
X	N1	N2
1	E108	E109
2	E108	E109
3	E108	E109
4	E108	E109

V SYNC INPUTS (N1)

SWITCH NO.	INPUT (FROM E703)
E3-1	P7
E3-2	P1
E3-3	P2
E3-4	P3
E12-1	P6
E12-2	P6
E12-3	P6
E12-4	P6

NOTES:
 1) X DENOTES SECTION NUMBER: ie SWITCH 18 OF SECTION 3 IS E-18-3
 2) N1 INPUTS TO E6-1 THROUGH E9-1 ARE GROUND, E703-P1, E703-P2, AND E703-P4 RESPECTIVELY (4 SECTIONS PER ENCODER)

FIGURE 5-112. COMMUTATOR SECTION



PRE-RELEASE

FIGURE 5-113. COMMUTATOR ELECTRONIC SWITCH

ON. The commutator nominally looks like an infinite resistance to the signal source since none of the base drive must be obtained from the commutator input.

This selection scheme has two advantages over the slightly simpler method of using a single series electronic switch for coupling each input to the subcarrier oscillator. One advantage is that if a failure occurs causing the output of a given switch to short to ground or to a power supply, only one-fourth of the inputs are disabled since the second level switch provides isolation between each set of two frames. A second advantage accrues from the offset-voltage compensation of the two series switches.

Commutator section and channel select logic is shown in Table 5-18. The -8 volt state of an input is a logical "one" and $+4.5$ volt state is a logical "zero." Switches corresponding to channels 1 through 15 have two logic inputs (N_2, N_3). One input ($E107, \overline{E107}$) selects between the odd and even frames and the other, except for channels 8 to 15, is the output of the selection gates E201-E207 as defined in Table 5-18. There is no selection gate for channels 8 to 15 since only one counter output is needed for channel selection. The use of the channel select gates reduces the number of logical inputs required at each switch, resulting in minimum diode logic.

The E200 and its complement $\overline{E200}$ are gating signals for sync and reference selection and are generated by gate E200. The logical equation is

$$\overline{E200} = (101 \cdot \overline{102} + 102) (\overline{103} \cdot \overline{104} \cdot \overline{105} \cdot \overline{106})$$

These logical functions are needed because of the unique format of channel 0.

Command Execute Tone. The method of the transmission of the command execute tone and the modulation information are as yet to be determined. The major factors are the type of com-

TABLE 5-18. COMMUTATOR SECTION AND CHANNEL SELECT LOGIC

FRAME SELECT LOGIC

Commutator Section	Frames	A	B
1	0,1	108	109
2	2,3	108	109
3	4,5	108	109
4	6,7	108	109

CHANNEL SELECT LOGIC

Selection Gate	Channel	D*	E	F	G	H
E201	1	102	$\overline{103}$	104	$\overline{105}$	$\overline{106}$
E202	2	102	$\overline{103}$	104	$\overline{105}$	$\overline{106}$
E203	3	102	103	104	$\overline{105}$	$\overline{106}$
E204	4	102	$\overline{103}$	$\overline{104}$	105	$\overline{106}$
E205	5	102	103	$\overline{104}$	105	$\overline{106}$
E206	6	102	$\overline{103}$	104	105	$\overline{106}$
E207	7	102	103	104	105	$\overline{106}$
	8-15					106

*E102 differentiates between the reference and data portion of channels 1 to 7.

mand execute tone modulation to be used and the number of discrete information levels which must be detected. The real time modulating information consists of the indications of bipropellant motor firings. The command execute tone will be modulated to at least five discrete levels; four of the levels will represent firing each of the four bipropellant motors and the fifth will represent all other cases. The method for telemetering squib firings is under investigation and may require additional discrete modulation levels. An investigation of the number of discrete levels that can be reliably detected at the ground stations for phase, frequency, and amplitude modulation was made, and the results are presented in the discussion of the encoding and squib firing later in this section.

Solar Sensor Pulses. The solar sensor pulses ψ and ψ_2 are superimposed on the audio signal and directly phase-modulate the 136 mc transmitter carrier. These pulses are telemetered for a backup ground control mode of the pulsed jet velocity and attitude control systems. The output pulses from the ψ and ψ_2 sensors are amplified and shaped by the solar sensor amplifiers E806, E807. The resulting output pulses are of the same polarity but different amplitudes.

Specifications and Circuit Descriptions. Schematics, descriptions, input requirements, and output specifications of the telemetry encoder circuits are presented in this discussion. All of the circuits represent either preliminary configurations or designs since the encoder breadboard fabrication has not yet been completed. No circuit is given for the modulator, E804, since its requirements are not yet defined. The data level limiter configuration is also not yet firm.

- *Reference-Sync Select Gate.* The reference sync select gate (Figure 5-114) generates the logical functions required by the unique format of the sync channel 0. The output E200 and its complement, $\overline{E200}$, are derived from the combination of the six least significant bits of the selection counter according to the logical equation;

$$E200 = (101 \cdot \overline{102} + 102) (\overline{103} \cdot \overline{104} \cdot \overline{105} \cdot \overline{106})$$

Q1, Q2 and Q3, Q4 are inverters with a capability of switching the capacitive loading within a few microseconds for either change in state.

When the quantity $(101 \cdot \overline{102} + 102) (\overline{103} \cdot \overline{104} \cdot \overline{105} \cdot \overline{106})$ is true, Q1 is off, so E200 is in the "zero" and $\overline{E200}$ is in the "one" state. When the above quantity is false, the input gate is decoupled from the inverter by the diodes CR12 and CR13, and R5 provides base drive to turn Q1 on. At this time, E200 is in the "one" state and $\overline{E200}$ is in the "zero" state.

The coupling from Q2 to Q3 through R7, R8 and C1 is designed to satisfy the constraints that not more than one commutator switch be selected at any given time and that the time delay between the selection of successive commutator input signals be less than 3 microseconds.

The purpose of Q2 and Q4 emitter follower output transistors is to provide fast output rise times without dissipating an excessive amount of power. The output specifications are as follows:

Rise time < 3 microseconds

Fall time < 3 microseconds

Current capability,
logical zero state 4.0 milliamperes minimum

- *Commutator Driver Circuit.* The purpose of the commutator driver (Figure 5-115) is to supply a fixed current to the collectors of the ON commutators as a function of the input signal voltage. The commutator then nominally looks like an infinite resistance to the signal source since none of the base drives must be obtained from the commutator input.

If the frame select commutator is ON, N1 and N2 are low and transistor Q3 will be ON; CR3 will then be back-biased. In this mode the variable load current demanded from P is generated by maintaining the current supplied through R1 constant and allowing current drawn by Q2 to vary as the signal voltage (V_s) varies, yielding a net output current which is dependent on V_s . If the frame select commutator is OFF, the driver output current should be reduced by one-half by turning Q3 OFF; the frame select commutator (E1-X) is then simulated by diode CR3 and resistor R7 equivalent to the commutator base resistor) is returned to the -24 volt supply.

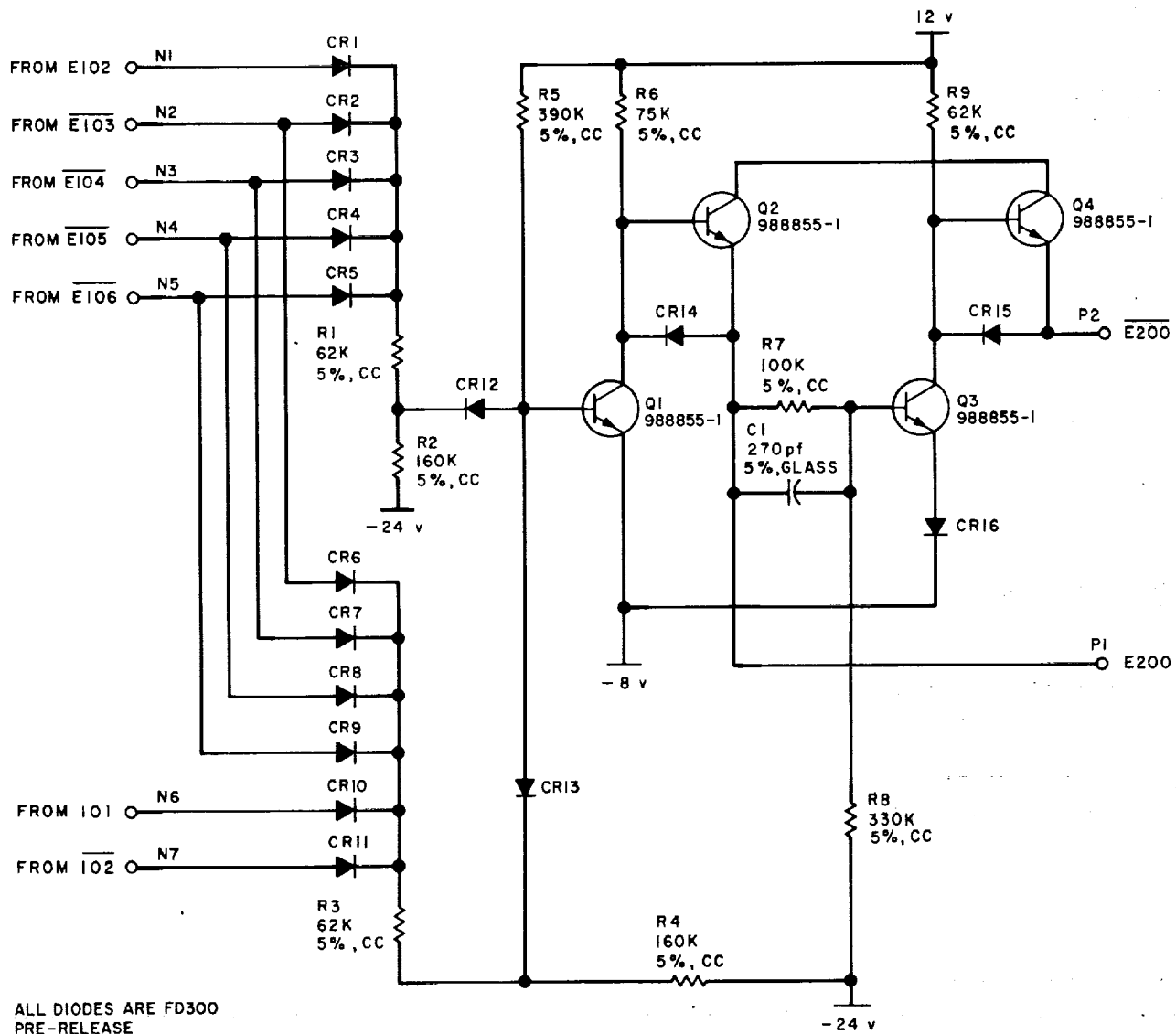


FIGURE 5-114. REFERENCE-SYNC SELECT GATE

Therefore, it appears to the basic commutator driver circuit as the same load when the frame select commutator is ON.

The divider network of R2, R3, and CR2 determines the base voltage of Q1. Diode CR2 adjusts for temperature variations in the base-emitter drop of Q1. CR6 compensates for temperature variations in CR1. R3 is returned to the -24 volt supply to compensate for variations in

commutator base current due to supply variations.

- Counter Flip-Flop. The counter flip-flop (Figure 5-116) is a four-transistor pedestal-gated n-p-n type. The output transistors, Q3 and Q4, provide the current drive for the capacitive load during the transition from the one to the zero state. Their collectors are returned to the +5 volt supply, clamping the output at this level.

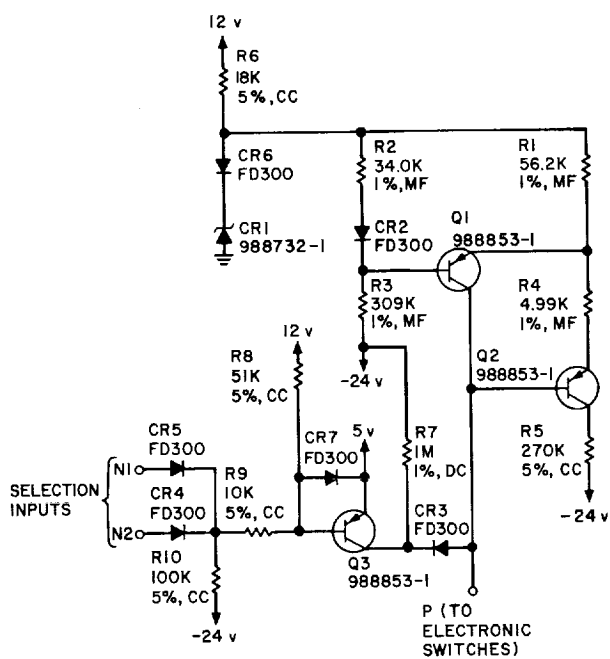


FIGURE 5-115. COMMUTATOR DRIVER

Instead of one, two steering diodes, CR3 and CR4 or CR5 and CR6, are used in series to increase the input noise rejection capability. The n-p-n configuration was chosen because it provides an optimum solution to the subsystem requirement that no two commutator switches be selected simultaneously. This is the result of the inherent time delay of the negative-going transistor during which the positive transition can be completed.

The output specifications are as follows:

Current capability	2.0 milliamperes out at zero state
Output voltage	4.2 volts minimum 5.0 volts maximum in zero state -5.5 volts maximum -8.7 volts minimum in one state

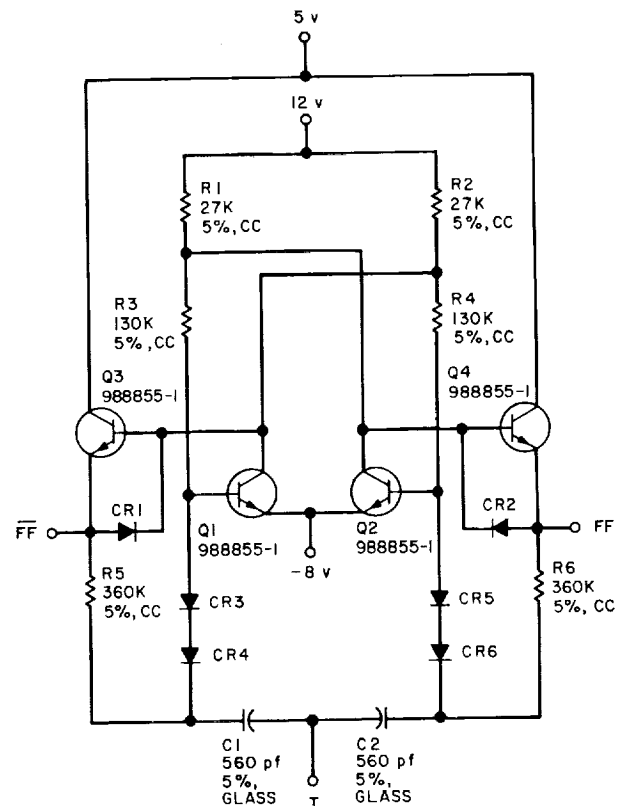


FIGURE 5-116. COUNTER FLIP-FLOP

Switching delay time	1.0 microsecond maximum
Voltage fall time	2.0 microsecond maximum
Voltage rise time	0.5 microsecond maximum

- Counter Interstage Gate. The counter interstage gate schematic is shown in Figure 5-117. This circuit is used between successive stages of the standard flip-flop. Its functions are clamping the flip-flop output transition as seen by the next stage input to about 7 volts and decoupling the flip-flop input when it is in the zero state.

- Counter Inverter. The counter inverter (Figure 5-118) is a common emitter amplifier

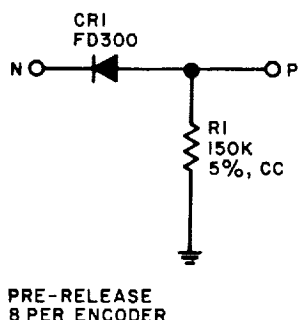


FIGURE 5-117. COUNTER INTERSTAGE GATE

which inverts the positive output pulses of the central timer. The negative output pulses of the inverter are used to trigger the first selection counter flip-flop. When the input to the counter inverter is low, Q1 is kept off by the bias developed by the voltage division of R1 and R2. The output of the inverter is then high. When the input goes high, R1 is decoupled from the circuit by the diode CR1, and R2 provides base drive to turn Q1 on. At this time the output goes low.

The output specifications are as follows:

Waveform	Negative pulses
Amplitude	7.1 volts $\pm$ 10 percent
Output capability	Will drive one count flip-flop

- **Output Amplifier.** The output amplifier provides impedance transformation and voltage amplitude adjustment for the PFM signal, the solar sensor pulses, and the execute tone. The circuit is a nonsaturating differential amplifier with feedback control of the voltage gain. The various inputs are resistively summed at the input of one side of the differential amplifier. Since this input is a virtual ground, mutual loading of the various inputs will be small. The output impedance of the amplifier will be less than 500 ohms. The amplified voltage swing of each input is determined by the ratio of the summing resistor and the amplifier feedback resistor.

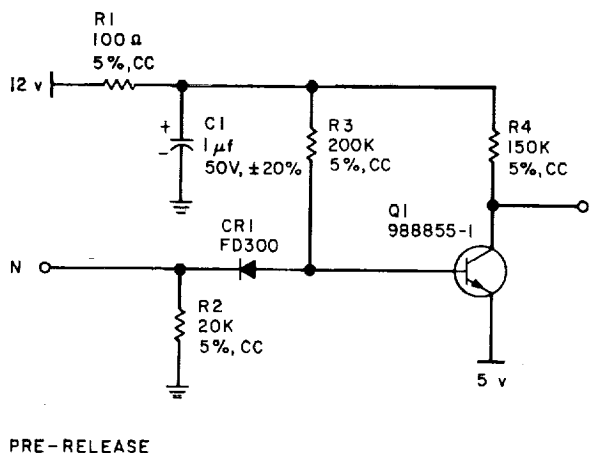


FIGURE 5-118. COUNTER INVERTER

The output specifications are as follows:

PFM signal	3 volts rms $\pm$ 10 percent
ψ pulse	-4.5 volts $\pm$ 10 percent
ψ_2 pulse	-2.25 volts $\pm$ 10 percent
Unmodulated command execute tone	3 volts rms $\pm$ 10 percent

- **Solar Sensor Amplifier.** There is a separate solar sensor amplifier for each of the ψ and ψ_2 pulses. The output of the amplifier is a current pulse which produces a voltage pulse at the output of the differential output amplifier.

The circuit, shown in Figure 5-119, consists of an amplifier stage and a shaper stage. Output transistor Q2 is normally off with its base emitter junction back-biased by about 5 volts. When the positive input pulse occurs, it is amplified and inverted and applied at the base of Q2, forward-biasing the base emitter junction and saturating Q2. The voltage pulse at the collector of Q1 is

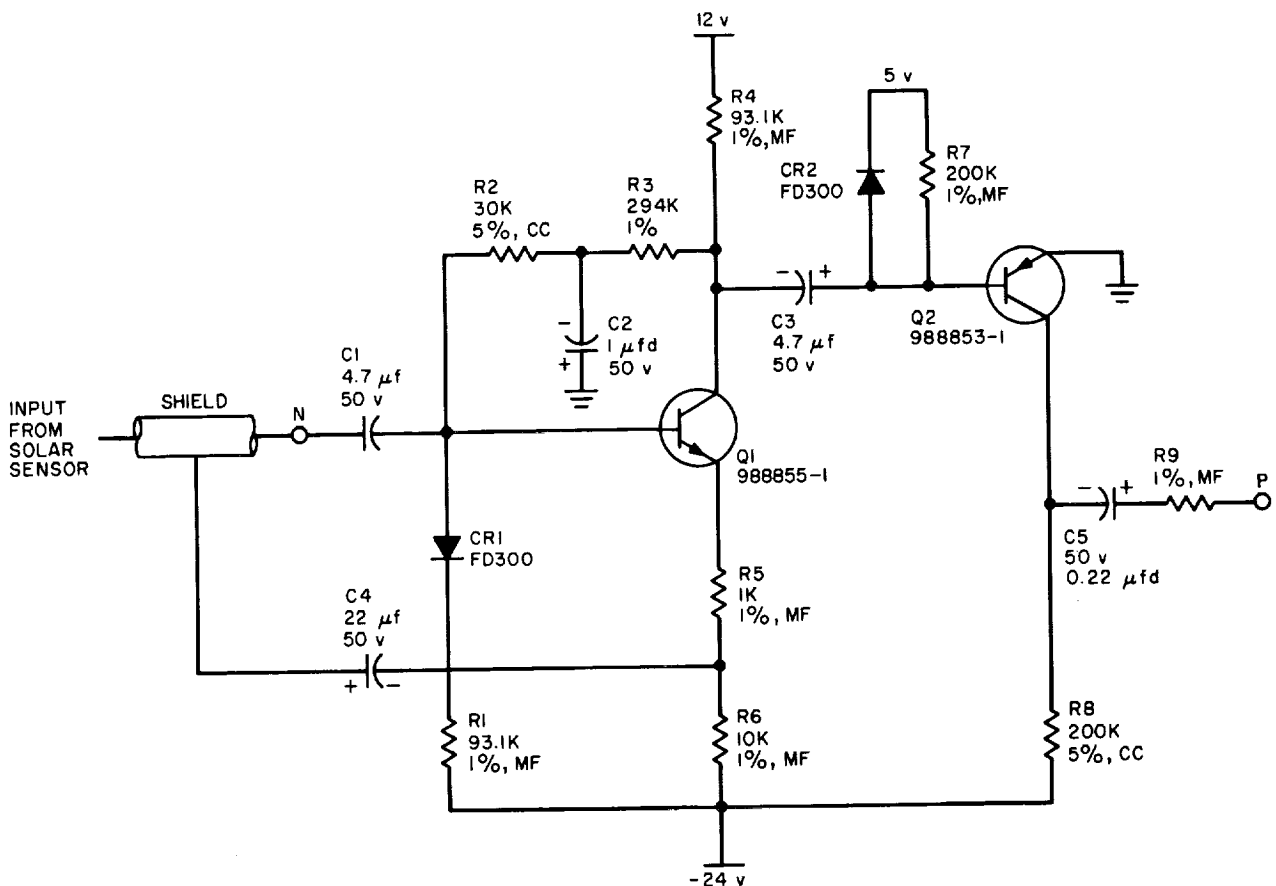


FIGURE 5-119. SOLAR SENSOR AMPLIFIER

clamped by the base emitter diode of Q2. Resistors R2 and R3 provided a feedback path for stabilizing the bias level of Q1. CR1 is used for temperature compensation of the base emitter drop of Q1. CR2 is for protection against base emitter breakdown of Q2.

The voltage pulse at the collector of Q2 is approximately 24 volts. This voltage is scaled by the ratio of the magnitude of R4 and the feedback resistor in the output amplifier, E805, such that the ψ and ψ_2 pulses occurring at the transmitter are nominally 4.5 and 2.25 volts respectively. The gain of the solar sensor amplifier is set so that the shaper stage is triggered (threshold level) at a nominal input voltage of 120

millivolts. The asymmetry of the output pulse with respect to the input results in errors of the ψ - ψ_2 angle of less than 0.1 percent. The asymmetry is due to the RC time constants of the circuit.

C4 is returned to the shield on the input wire so that most of the input noise will not be seen at the collector of Q1.

- Subcarrier Oscillator. Figure 5-120 shows the subcarrier oscillator. The signal input voltage is applied to the base of Q1. Transistors Q1 and Q2 provide current gain such that the signal input current is less than 1 microampere. Resistors R1, R2, R3, in conjunction with zener diode

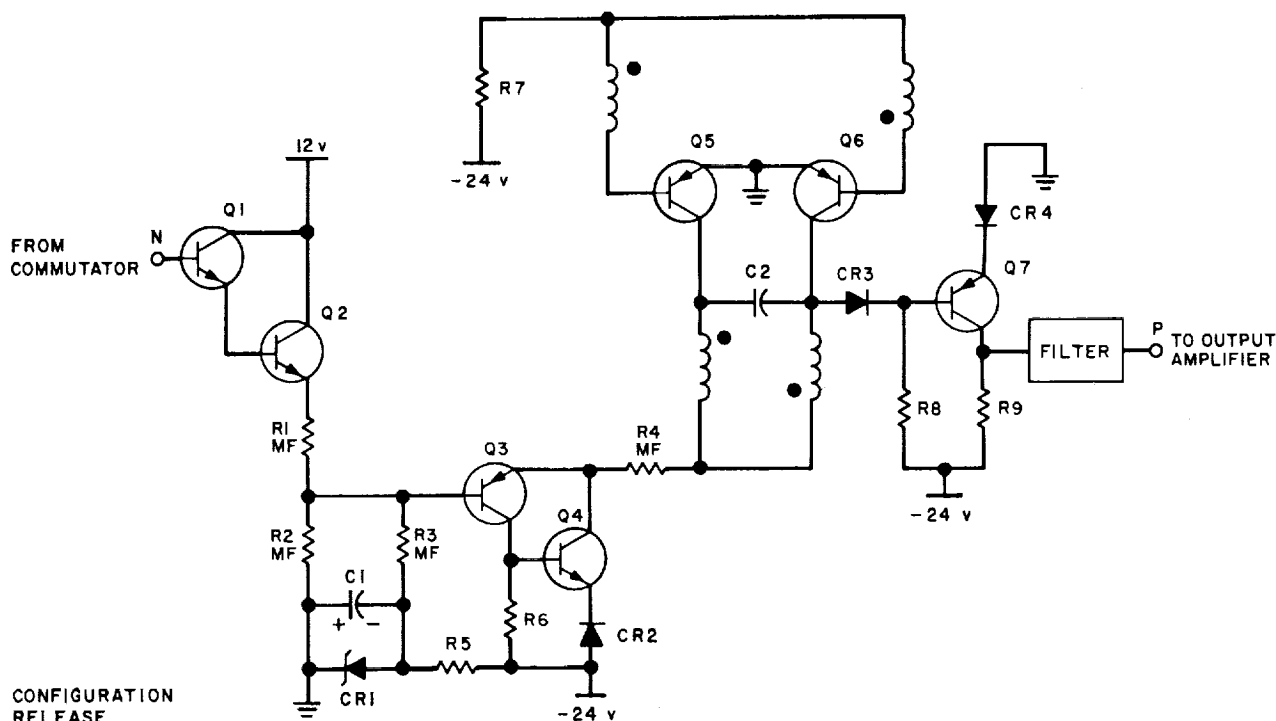


FIGURE 5-120. SUBCARRIER OSCILLATOR

CR1, constitute a voltage divider and shifter. Transistors Q3 and Q4 provide current gain.

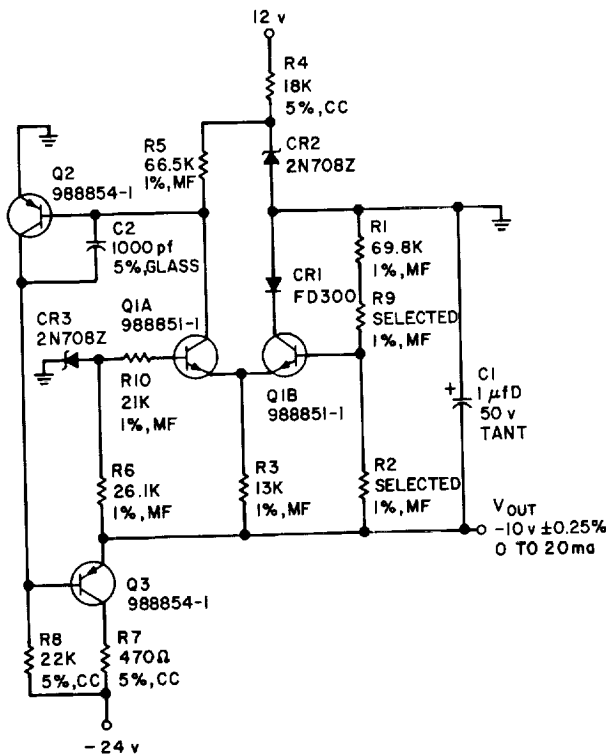
The output of the Q3-Q4 pair is the voltage seen by the basic oscillator circuit—transistors Q5 and Q6, resistors R4 and R7, capacitor C2, and the magnetic core. The magnetic core is a toroid constructed of square permalloy 80 (Magnetics, Inc.). This material was chosen to obtain low coercive force, and hence low current, and high squareness.

The basic oscillator output, at the collector of Q6, is applied to inverter Q7. The inverter provides voltage gain and produces a fixed amplitude over the frequency band. The inverter output is applied to a low-pass filter, then to the output amplifier. Temperature-compensating components are not shown in the schematic.

Input-output specifications for the subcarrier oscillator are as follows:

Frequency range	4450 to 16,000 cps
Transfer characteristics	$f = 5000 (\pm 0.5 \text{ percent})$ $- 2000 (\pm 2 \text{ percent}) V_{in}$ cps
Linearity	Deviation from best straight line less than 100 cps
Signal input current	Less than 1 microampere
Output waveform	"Square wave" filtered by low-pass filter with rolloff beginning above 16 kc, down by at least 3 db at 100 kc, continuing at minimum rate of 2 db

• **Transducer Excitation.** The -10 volt transducer excitation (Figure 5-121) is designed to accept from 0 to 20 milliamperes current. Q1A and Q1B form a differential amplifier. The base of Q1A is at a fixed voltage set by temperature compensated zener diode CR3. R1, R9, and R2 form a voltage divider which establishes a voltage at the base of Q1B. Any variation in output voltage is sensed at the base of Q1B by the divider and amplified by Q1B and Q1A. Q2 further amplifies and inverts the signal, thus driving Q3, the series element, in proper phase to reduce the output voltage variation. Zener diode CR2 establishes a regulated collector voltage for Q1A. CR1 maintains the collector of Q1B at almost the same potential as the collector of Q1A over the temperature range. C1 is a filter capacitor for fast charging loads, and C2 reduces


FIGURE 5-121. -10 VOLT TRANSDUCER EXCITATION

the loop gain of the regulator at high frequencies to prevent oscillations.

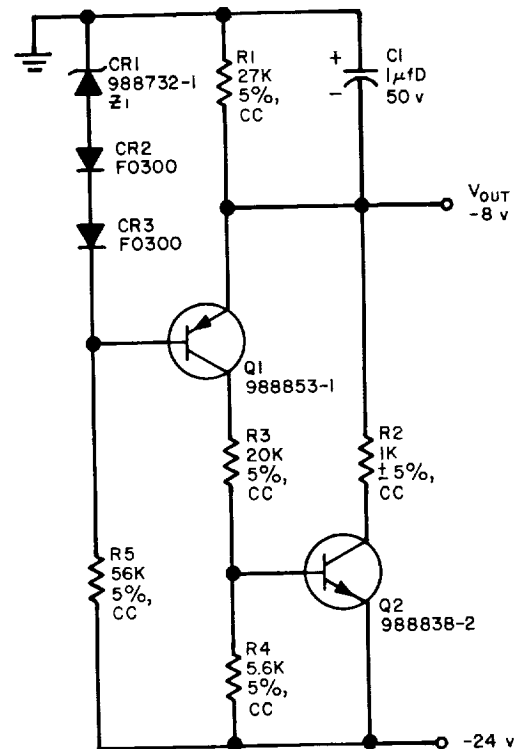
Q1A and Q1B are a pair of 988855-1 transistors closely matched for V_{BE} and B . They are both in a single TO-5 can, thus providing tracking of the critical parameters over temperature.

The nominal output voltage can be varied by selecting the value for R9.

Output specifications are as follows:

Voltage	-10 volts ± 0.52 percent
Current	0 to 20 milliamperes

• **-8 Volt Regulator.** The -8 volt regulator (Figure 5-122) is designed to accept from 0 to 15 milliamperes current. Q1 is an emitter follower with the output voltage established by



PRE-RELEASE

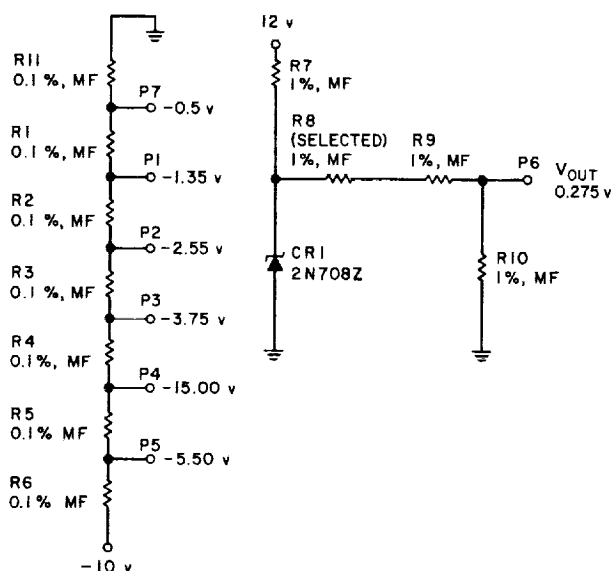
FIGURE 5-122. -8 VOLT REGULATOR

the series combination of a zener diode and two forward-biased diodes connected to the base of Q1. The two forward-biased diodes have temperature coefficients of opposite sign from that of the zener diode and are used for temperature compensation and to increase the voltage output level to that required for the encoder logical one state. Q2 provides additional gain and tends to reduce the output impedance. R2 and R3 provide short protection. C1 is a filter capacitor and also prevents the regulator from oscillating.

Output specifications are as follows:

Voltage	-7.6 volts $\pm$ 5 percent
Current	15 milliamperes maximum

• **Voltage Standards.** The schematic of the voltage standards circuit is given in Figure 5-123. The positive voltage used for odd frame sync is generated at output P6 from a voltage divider network. The divided reference voltage is established by the temperature compensated zener diode CR1. Initial tolerances of the zener diode



CONFIGURATION RELEASE

FIGURE 5-123. VOLTAGE STANDARDS

and resistors are cancelled by selection of the value for R8.

The negative voltages used for the reference, even frame sync, and calibration are generated from the voltage divider string consisting of R1-R6 and R11. The -10 volt reference voltage is the accurate transducer excitation.

• **ψ_2 Angle Counter Gate.** The ψ_2 angle counter gate (Figure 5-124) generates a negative-going timing signal, E208, which is used for recycling the ψ_2 angle counter. The combination of the four most significant bits of the output counter according to the logical equation

$$E208 = E106 + \overline{E107} + \overline{E108} + E109$$

generates a positive-going pulse during the first half of frame 3. The ψ_2 angle counter timing signal which is the trailing edge of the gate output pulse, occurs two channels after the ψ_2 angle data has been telemetered. At the cost of three additional diodes, the timing signal could be generated immediately after the ψ_2 angle channels; however, this is not necessary.

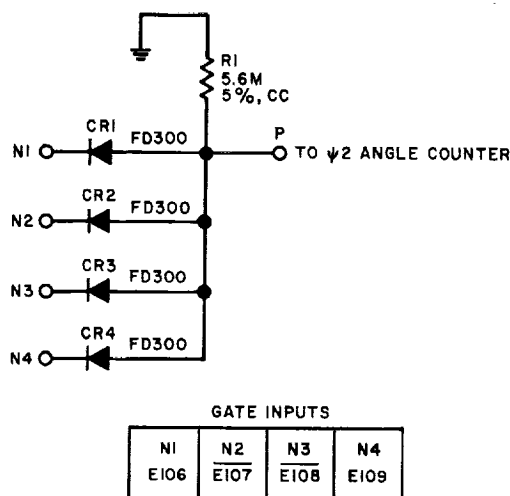


FIGURE 5-124. ψ_2 ANGLE COUNTER GATE

The value of the gate resistor, R1, is sufficiently large that positive voltage spikes on the output due to the finite delay and switching times of the logical inputs will not falsely trigger the ac-coupled input flip-flop of the ψ_2 angle counter. That is, the RC time constant of the gate resistor and the coupling capacitor of the flip-flop is very long compared to the duration of these voltage spikes.

Output specifications are as follows:

Amplitude	5.4 volts minimum 7.2 volts maximum
Fall time	2.0 microseconds maximum
Current capacity	1 milliampere minimum dynamic 1 milliampere minimum static

- **Temperature Sensor.** The temperature sensor (Figure 5-125) is a transducer used to indicate the temperatures of various spacecraft system components such as the structure, bipropellant fuel, and traveling-wave tubes. The circuit consists of the actual sensing device, a thermistor, connected in parallel with one section of the voltage divider composed of the fixed resistors

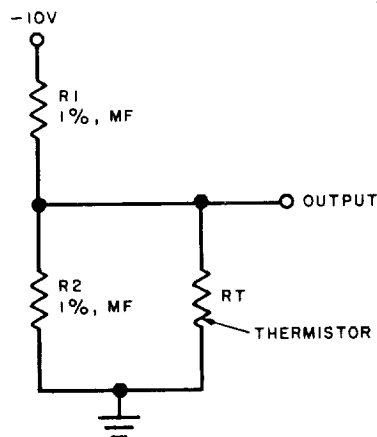


FIGURE 5-125. TEMPERATURE SENSOR

R1 and R2. Through selection of the values for R1, R2 and the thermistor value at $+25^{\circ}\text{C}$, the output voltage of the sensor can be adjusted to the 0 to -5 volt limits over the range of sensed temperatures.

In the spacecraft system, each temperature sensor receives its voltage excitation from the encoder(s) which is sampling its output. Most of the sensed temperature data must be capable of being telemetered by any one of the quadrant encoders. If a single sensor is used, the -10 volt excitation of each telemetry encoder must be decoupled from the others at the sensor circuit. The temperature coefficient of the decoupling network would require temperature compensation in the sensor circuitry, resulting in a more complex configuration. For this reason and for reasons of reliability and accuracy, four separate sensors will be used instead of one for the areas in which temperature data is common to all encoders. For temperatures that are unique to a given encoder, only one sensor will be used. The four sensors will be mounted on a plated brass plate with the grounded ends common to the plate. The four divider networks R1 and R2 will be packaged with the encoder.

Power Summary. A power summary for the encoder is given in Table 5-19. The total power dissipated in the encoder is 1.18 watts for the worst case conditions indicated in the table. This value includes the power dissipated in the temperature sensor divider circuits which will be packaged with the encoder and the power drawn from the -24 volt supply by the telemetry bias unit due to the encoder loads on the $+12$ and $+5$ volt outputs and the intervals in the bias unit itself.

Component Summary. The quantities of components are summarized by type in Table 5-20.

TABLE 5-19. POWER SUMMARY (PER UNIT)

Supply Tolerance		+12 Volt +12.0 ± 5 percent		+5 Volt +4.75 ± 5 percent		-10 Volt -10.0 ± 0.25 percent		-8 Volt -7.6 ± 5 percent		-24 Volt -24 ± 3 percent	
Circuit	Number per Unit	Current, ma	Power, mw	Current, ma	Power, mw	Current, ma	Power, mw	Current, ma	Power, mw	Current, ma	Power, mw
Counter inverter	1	0.06	0.7								
Counter flip-flops	9	8.80	105.8	-1.35	-6.4			-7.42	56.5		
Counter interstage gates	8							-0.37	1.4		
Selection gates	7			2.76	13.1			-2.76	21.0		
Reference-sync select gate	1	0.45	5.4	0.08	0.4			-0.35	2.7	-0.35	8.4
Commutator electronic switch	80			2.26	10.7					-2.26	54.2
Commutator driver	4	1.76	22.0	-0.39	-1.9					-1.56	37.5
Solar sensor amplifier	2	0.47	5.7							-0.47	11.3
Subcarrier oscillator	1	0.50	6.0							-4.0	96.0
Output amplifier	1	0.50	6.0							-1.0	24.0
Modulator	1									-2.0	48.0
Pressure transducers	4					-2.00	20.0				
Temperature sensors	11					-5.40	54.0				
Voltage standards	1	1.0	12.0			4.0	40.0				
Telemetry bias unit	1										
Losses										-1.80	43.2
Load										-7.17	172.1
-8 volt regulator	1									-10.95	262.0
Transducer excitation	1	0.28	3.3							-12.4	298.0
Data level limiter										-0.5	12.0
Total for nominal conditions (Note 2)		13.85	166.1	3.36	16.0	-11.40	114.0	-10.95	83.1	-44.5	107.0
Total for worst case conditions (Note 2)		14.50	182.5	3.52	17.6	-11.43	114.6	-11.43	91.5	-47.7	117.5

Maximum power dissipated per encoder (including temperature sensors and pressure transducers) = 1.18 watts

NOTES:

- 1) Temperature sensor power varies with sensed temperature. Power indicated assumes that sensed temperatures are such as to cause maximum circuit power dissipation. Minimum dissipation is approximately 1/2 that indicated.
- 2) Conditions for computing power are:
 Nominal — Nominal resistance, nominal voltage
 Worst case — Maximum resistance, maximum voltage for low usage circuits
 Nominal resistance, maximum voltage for high usage circuits

5. Spacecraft System Design

TABLE 5-20. COMPONENT SUMMARY

Quantity Required	Component	Description
47	Transistor	988855-1
20	Transistor	988853-1
2	Transistor	988854-1
2	Transistor	988838-2
2	Transistor	988851-1
86	Transistor	988859-1
323	Diode	FD300
3	Diode	Zener 988732-1
2	Diode	Zener 988732-2
3	Diode	Zener 988902-1
121	Resistor	5 percent, carbon composition
93	Resistor	1 percent, deposited carbon
73	Resistor	1 percent, metal film
7	Resistor	0.1 percent, metal film
20	Capacitor	Glass or mica
18	Capacitor	Solid tantalum
1	Core	
11	Thermistor	
832		

• Doppler Shift. In considering coherent PSK modulation, the expected doppler shift is of importance. An approximate expression has been derived for the doppler shift for a synchronous vehicle:*

$$\Delta f = \frac{v_0 e}{c} \times 4.3 \times 10^3$$

where e is the orbital eccentricity and c the velocity of light, in feet per second. Just after apogee engine firing, a three sigma estimation of e is 0.033, which includes the effect of orbital inclination errors. For a subcarrier tone of 3.6 kc, this is $(3.6 \times 10^3) (33 \times 10^{-3}) (4.3 \times 10^3) / 0.9843 \times 10^9$ or 0.52×10^{-3} cps. Additional doppler shift is caused by orbital period error, but the magnitude is negligible in comparison to that

*Advanced Syncom Monthly Progress Report, February 1963.

caused by inclination and eccentricity. A doppler shift of 0.52×10^{-3} cps is equivalent to 360 degrees phase shift in 1925 seconds or 0.187 deg/sec (11.2 deg/min).

• Coherent PSK. The received subcarrier tone will be phase-shifted by discrete angles and coherently retransmitted to the ground. The expected subcarrier tone signal to noise ratio on the ground is 16.0 db in a 2.2 kc bandwidth.* Assuming a 1-millisecond rise time filter is used (160 cycles), the signal-to-noise ratio is 27.39 db. The phase deviation is approximately $(N/S)_{rms}$ radians or 2.44 degrees. The response of the phase detector is from 0 to 80 cycles.

Assuming the subcarrier signal is to be phase-shifted at 60-degree intervals (allowing five inputs and one zero reference input), the output level of the phase detector must reach ± 12.3 sigma for an adjacent sector to be falsely identified. This is clearly improbable.

Rice** has given the following expression, relative to a band limited gaussian process, for the number of threshold crossing with positive slope per second:

$$N_c = \frac{1}{\sqrt{3}} \left[\frac{f_b^3 - f_a^3}{f_b - f_a} \right]^{1/2} e^{-K^2/2}$$

where f_b and f_a define the upper and lower filter cutoff frequencies, and K is the threshold factor.

For a low pass filter, the expression reduces to

$$N_c = 0.577 f_b e^{-K^2/2}$$

An arbitrary false alarm rate of 10^{-3} per second (mean time between false alarms of 1000

*Syncom Mark I Telemetry and Command Systems Analysis, January 1963.

**S.O. Rice, *Mathematical Analysis of Random Noise*, reprint by Dover Publications, 1954.

seconds) may now be assumed. The number of times per second the upper or lower threshold is crossed is

$$N_c = 2(0.577)(80) e^{-K^2/2} = 10^{-3}$$

Solving for K, 4.785 is obtained or a full sector width of $4.785 \times 2.44 \times 2$, or 23.3 degrees. Therefore, 15 references may be transmitted, including the zero reference.

By decreasing the bandpass filter bandwidth, a greater number of references may be transmitted. Following the same line of reasoning, a 16 cycle bandwidth allows for 54 references, and a 1.6 cycle bandwidth allows 199 references to be transmitted. If 29 references are to be transmitted, a filter bandwidth of 50.6 cycles is required.

A block diagram of this system is illustrated in Figure 5-126. The necessity of frequent zero references of the ground receiver due to changing doppler shift may render a coherent PSK system impractical. It is important to recognize that

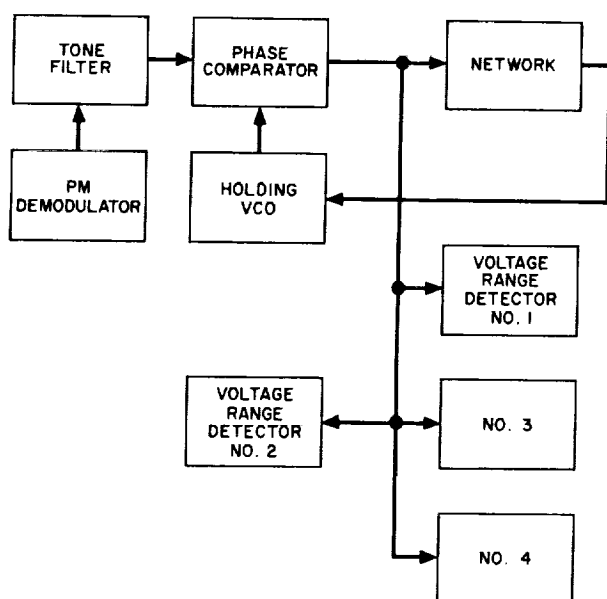


FIGURE 5-126. PSK ENCODE SYSTEM

only changes in the magnitude of the doppler shift, not the doppler itself, is potentially capable of causing trouble.

- FSK. The received subcarrier tone may be frequency translated for a voltage-controlled oscillator operated at various frequencies. It is assumed, as previously, that the signals are passed through 160 cycle filters and envelope detected as to their presence or absence (Figure 5-127). Since the tone signal to noise ratio is 27.39 db, the tones may be detected with negligible error probabilities. The stability of the voltage-controlled oscillator is the primary determining factor as to the number of references which can be transmitted. The effect of doppler shift (0.52×10^{-3} cps) is negligible.

- AM. The subcarrier tone is phase-modulated on the satellite carrier with various deviations, corresponding to reference inputs. A block diagram of this system is shown in Figure 5-128. Assuming the same parameters as previously, the number of references which may be transmitted is to be determined. The output of the subcarrier demodulator is Rayleigh-distributed, which be-

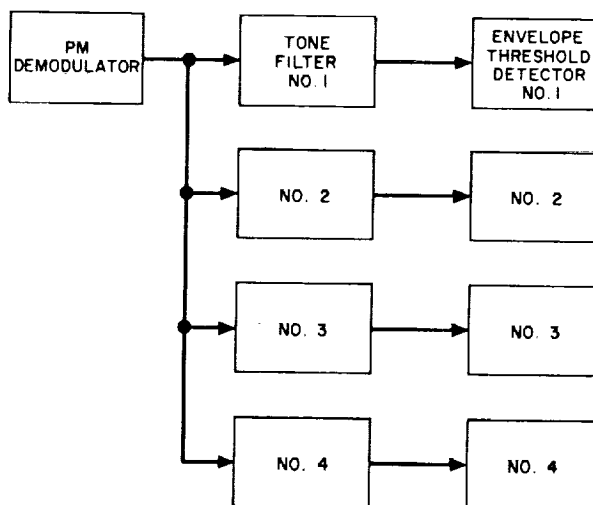


FIGURE 5-127. FSK ENCODE SYSTEM

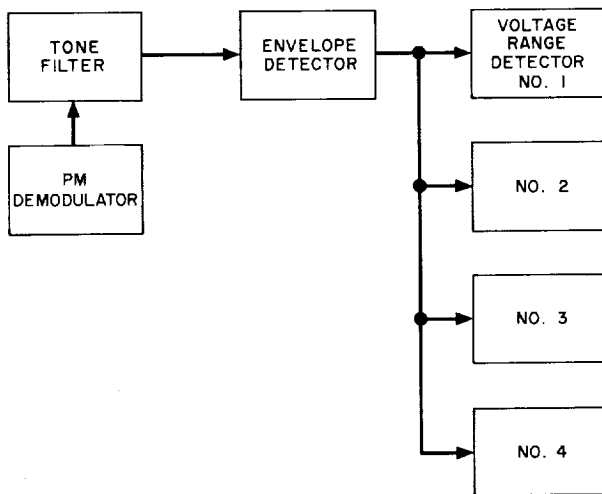


FIGURE 5-128. COMMUTATOR SECTION BLOCK DIAGRAM

haves as the normal distribution for moderately large threshold levels. Solution of the equation

$$(0.577)(80)(2)e^{-K^2} = 10^{-3}$$

yields the threshold factor, as previously, where $K = 4.785$. Assuming N references are to be transmitted (including the calibrating reference), it may be shown that

$$\frac{\text{peak signal}}{\text{rms noise}} = 2K(N-1)$$

which yields $N = 4.46$, or four levels (including a calibration reference). For a 16 cps and a 1.6 cps bandwidth, 13 and 42 references may be

transmitted. The system is thus relatively inefficient as compared to coherent PSK.

The modulation indices must be chosen to give equal incremental level changes at the receiver demodulator. As an example, assume the maximum deviation of the satellite carrier is 1.2 radians. Considering only the first sideband pair, $J_1(1.2) = 0.4983$. The first reference corresponds to zero (no signal). The second reference should be $0.4983 \times 1/3 = 0.1661 = J_1(0.35)$. The third reference should be $0.4983 \times 2/3 = 0.3322 = J_1(0.70)$. The deviations for the four levels should therefore be 0, 0.35, 0.70, and 1.20 radians.

- Adjustment of Data for Finite Signal-to-Noise Ratio. The received subcarrier tone signal-to-noise ratio at the satellite is 32.2 db in a 200 cycle bandwidth. This depresses the received tone signal-to-noise ratio by 1.02 db. Noiseless transmitted tones will be assumed for FSK. Table 5-21 summarizes the results for the three systems considered.

- Degradation by Solar Pulse Transmission During the period of solar pulse transmission, which is short relative to the satellite spin period, the subcarrier tone signal-to-noise ratio will be degraded. Gating at the receiver may be employed during reception of the solar pulse to prevent false alarm triggering during that period.

TABLE 5-21. RESULTS

Satellite Modulation of Subcarrier	Relative Spectral Occupancy	Total Levels Transmitted With 0.001 False Alarms for Ground Received Tone Noise Bandwidths of:			Satellite Subcarrier Modulator	Receiver Subcarrier Demodulator
		160 Cycles	16 Cycles	1.6 Cycles		
Coherent PSK	1	13	48	177	Stable phase-shifting network	Tone filter plus phase lock loop
FSK	N	N, limited only by oscillator stability			Stable voltage-controlled oscillator	Tone filters plus diode detectors
AM	1	4	11	40	Linear amplitude modulator	Linear amplitude demodulator

Command Decoder. Under normal operating conditions the Advanced Syncom command decoder responds in the primary mode (Figure 5-129) to an input which is frequency-shift-keyed (FSK) modulated. The essence of this type of modulation is that the message is represented by tones, one to signify a logical "one," the other a logical "zero." The message is shifted into the command register with a bit sync clock signal which amplitude modulates both signal frequencies. Thus, while only one tone is present at any one time, the clock is always present.

In the secondary mode (Figure 5-130), the decoder responds to a 100 percent pulse amplitude modulated input which corresponds to interrupting the message tone at regular intervals. This mode requires that only one tone channel be used to send the entire message. The command register counts the interrupts and so accumulates the desired message.

Figure 5-31 shows the command format for both modes of operation. In the primary mode the command word is divided into six parts: introduction, word sync, enable address, command, conclusion, and execution (not shown). Time before execution is not defined. The secondary mode command format is divided into seven parts: introduction, first complete tone interrupt, enable address, second complete tone interrupt, command, conclusion, and execution. Six of these are shown in Figure 5-131. Again the conclusion time is not defined.

The dual mode decoder contains 32 circuits — flip-flops, inverters, detectors, reset circuits, and clock amplifiers responsible for the control operation in one mode or the other. A single or even multiple component failure in any one of 16 of these circuits will not impair subsystem operation. Multiple failures in several circuits could even be tolerated if they occurred in circuits common to the same mode of operation.

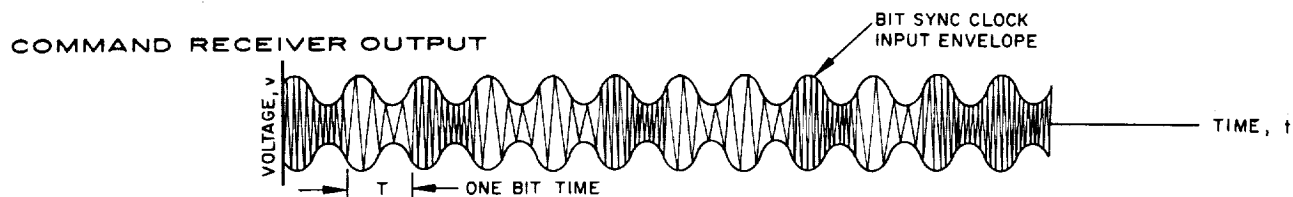


FIGURE 5-129. PRIMARY MODE: FREQUENCY-SHIFT-KEY MODULATED INPUT TO COMMAND DECODER

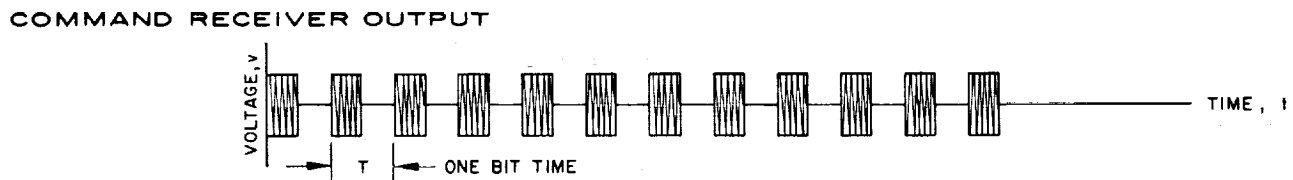


FIGURE 5-130. SECONDARY MODE: PULSE AMPLITUDE MODULATED INPUT TO COMMAND DECODER

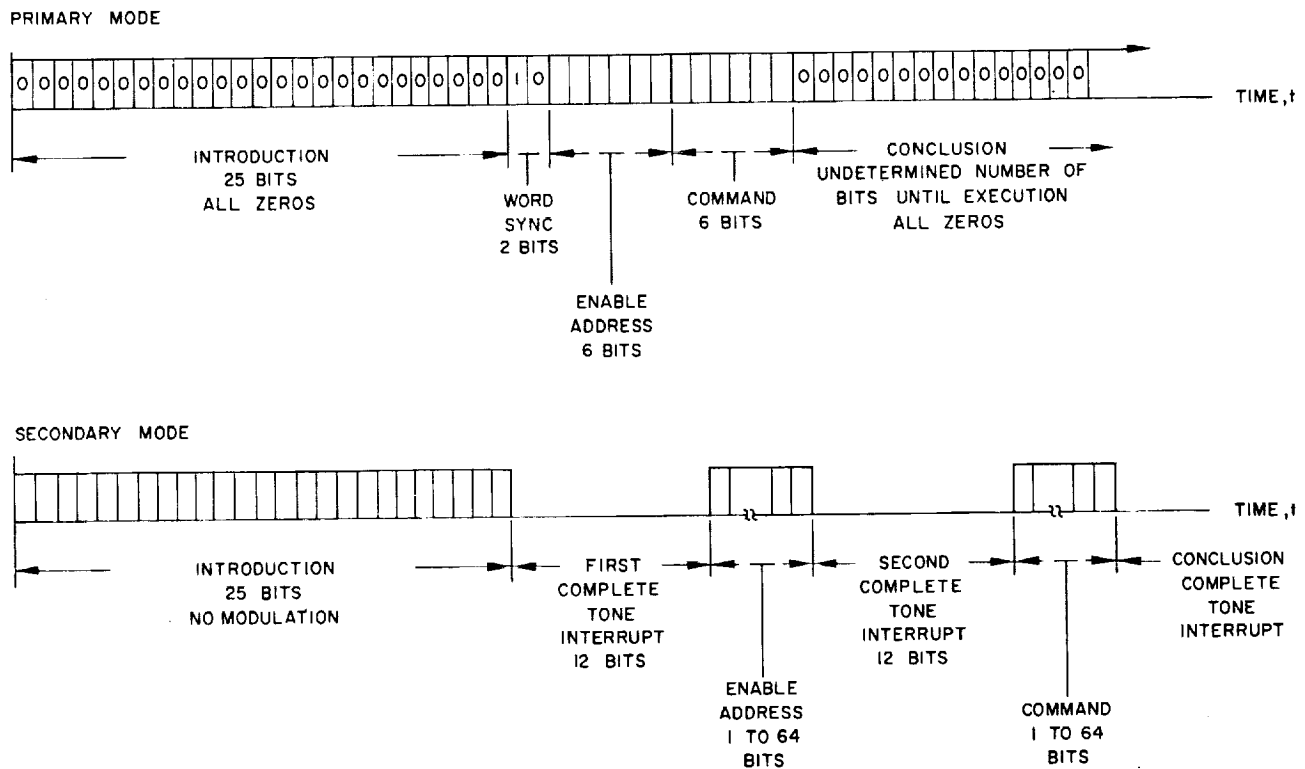


FIGURE 5-131. COMMAND DECODER COMMAND FORMAT

In the 16 remaining control circuits, certain components could fail without limiting operation capability.

Figure 5-132 shows the input circuitry of the decoder, the filters, and detectors, called C300, C301, and C606 in the block diagram of Figure 5-133. Part a of Figure 5-132 shows the circuitry necessary to synthesize the waveforms of part b. The filters are double tuned and yield a Butterworth response with a bandwidth of 400 cycles. The tone detectors provide minimum filter loading and at the same time allow two independent thresholds to be set. The first of these thresholds is a voltage threshold, the second a period threshold. The pull-in and dropout times of the detector can be varied in a nearly independent fashion. The tone detector requires that a signal first exceed a given voltage threshold and then

be present for a specified period of time before the output will change state. The output stages of the tone detectors provide the voltage levels and gain necessary to ensure response of the digital circuits which follow.

The audio detector, like the tone detectors, minimizes the loading of the filters. The first stage of this detector amplifies the input voltage linearly; the second detects the bit sync clock envelope shown in Figure 5-129. Stage three filters the detected envelope, and the last stage shapes the resultant sine wave (at bit sync frequency) in a slightly unsymmetrical square wave.

Output waveforms for these circuits in both modes of operation are shown in Figure 5-132b. These waveforms are the inputs to the digital circuits of the decoder.

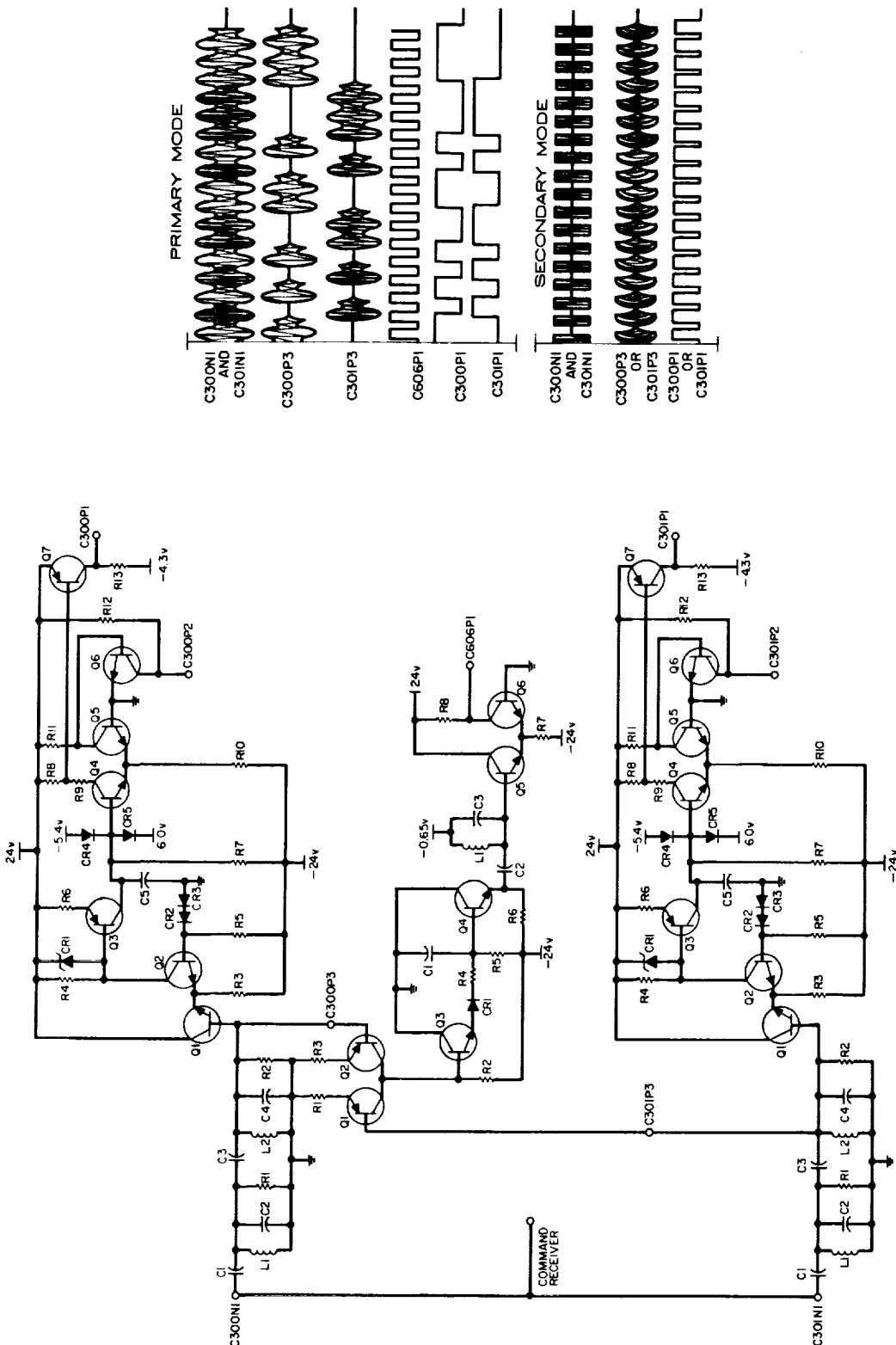


FIGURE 5-132. MESSAGE TONE FILTERS AND DETECTORS C300 AND C301 AND AUDIO DETECTOR C606



The digital output waveforms of the tone and audio detectors are repeated in Figures 5-134 and 5-135, along with the most important digital waveforms of the primary and secondary modes, respectively.

Primary Mode Operation. Before the initiation of a command, it is assumed that the decoder sees nothing at the input; no words are being sent. Initially, the outputs of the tone and audio detectors are in the logical zero state. A command is initiated from the ground, and it enters the decoder according to the command word format discussed above and shown in Figure 5-131.

The purpose of the introduction is to prepare the decoder for operation. When the zero tone is detected by C300, C300P1 will go to the logical one state (the complement of C300P1, C300P2, will go to the zero state). The input AND gate to the tone present detector (C603, shown in Figure 5-136 and in Figure 5-133), which is controlled by the complements of C300 and C301, is caused to go to low, and the first part of C603, a ramp circuit, will begin its rundown. The rundown to the threshold of the second part of C603, a Schmitt trigger, requires 100 milliseconds nominally. When the threshold of the Schmitt is crossed, the output of C603 makes a sharp negative transition. This transition, which occurs at the input to the mode register clear and the command register reset circuits (C605 and C604 respectively, shown in Figures 5-137 and 5-138), causes reset pulses to be generated which clear the six command register flip-flops (C106 to C101, Figure 5-139), the interrupt counter flip-flops (C108 and C109, Figure 5-140), the word sync flip-flop (C110, Figure 5-141), and the enable power circuit (C601, Figure 5-142). The reset pulse is nominally 100 microseconds long; it resets the flip-flops by pulling down the respec-

tive collectors to a sufficiently low voltage to cause the flip-flop outputs to go into the zero state.

When the reset pulses end, the decoder is completely clear and is ready to receive the desired message. The timing of the tone present detector and the reset circuits is such that the resetting operation is complete 6 bit times before the end of the introduction. As a result, zero will be accumulated in the command register.

This accumulation is accomplished by shifting into the command register, with the shift mode clock (C304, Figure 5-143), the outputs of the tone detectors. The input gating to the shift clock is in the proper state to allow the shifting to occur once the decoder has been cleared.

Following the introduction, the word sync enters the decoder. The word sync is 2 bits long and consists of a one followed by a zero. These 2 bits are shifted into command register flip-flops C106 and C105. When C105 goes high, the output of inverter C501 goes low causing the word sync flip-flop to be set to the one state.

When C110 goes high, the secondary mode of operation is completely disabled, and the negative transition of C110 causes the bit counter reset (C609, Figure 5-137) to clear the bit counter (C111 to C114), resetting it to zero.

As mentioned above, the secondary mode of operation is disabled when C110 goes high. This is done by inhibiting the input to the interrupt counter so that it will not register an interrupt in the input waveforms. Since C108 and C109 were cleared during the introduction, the gate which controls the output of the count mode clock (C303, Figure 5-143) will ensure that it remains in the zero state.

During the time that the message is being put into the decoder in the primary mode, there will never be a complete interrupt in the outputs of

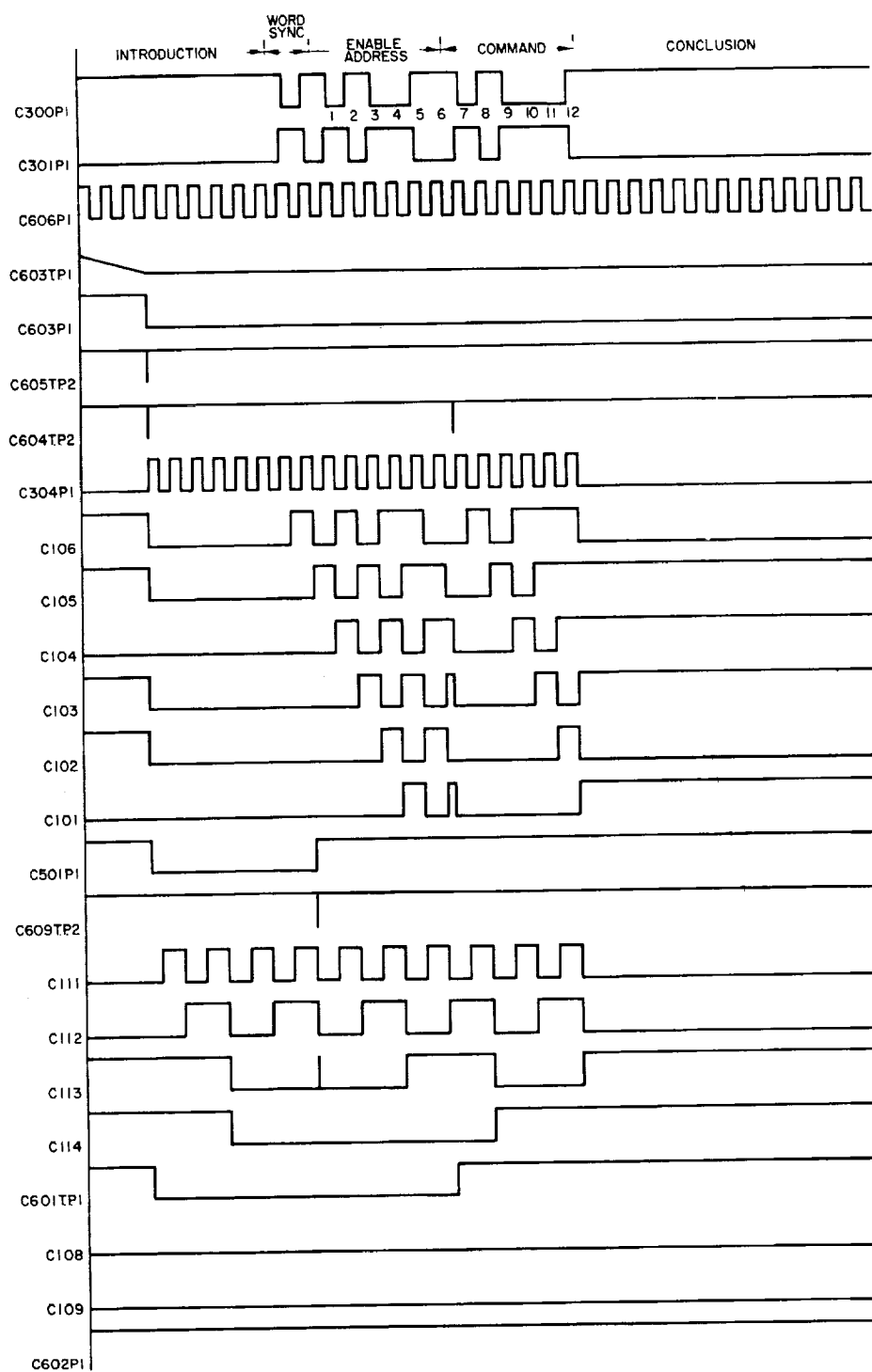


FIGURE 5-134. PRIMARY MODE WAVEFORMS

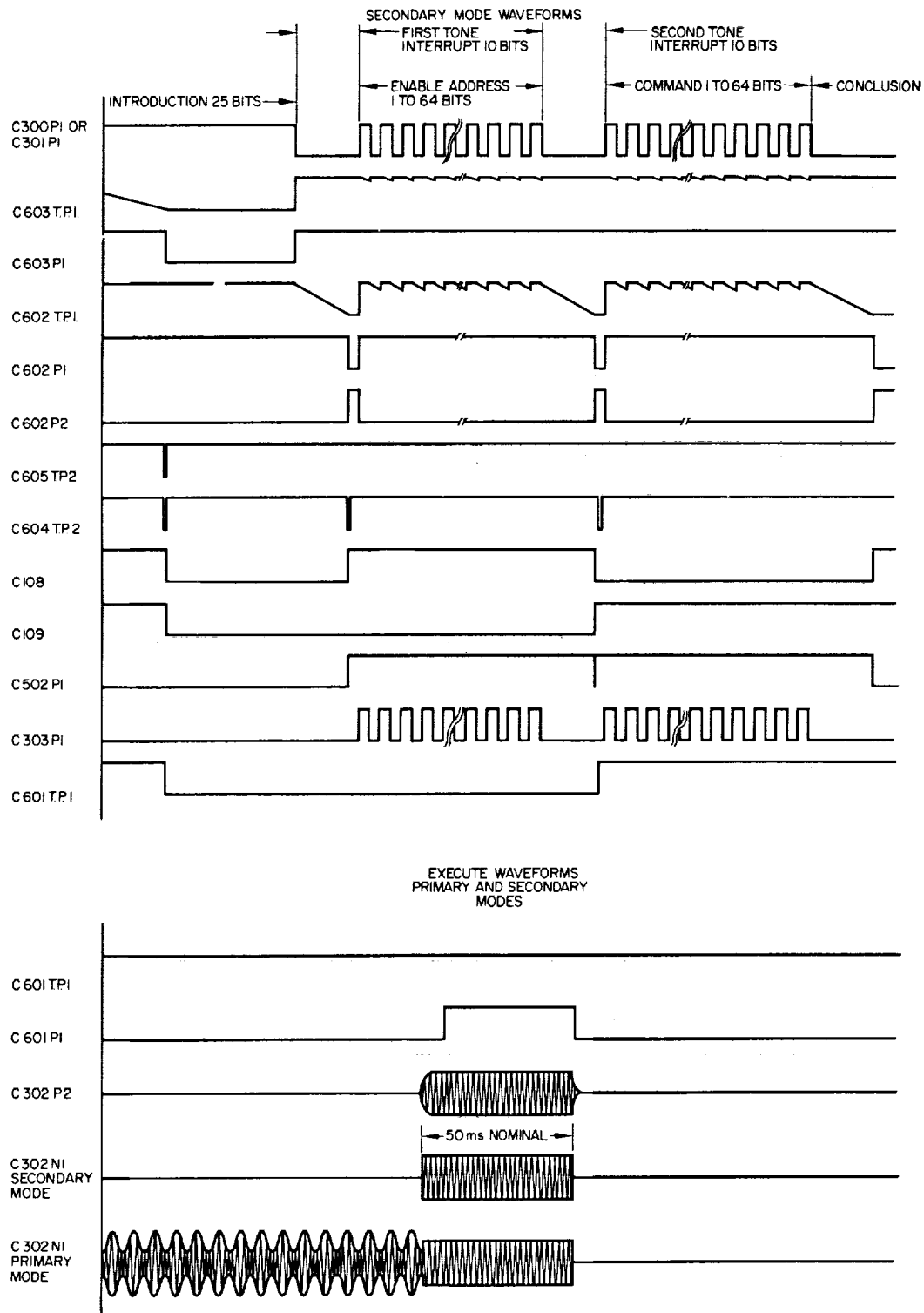
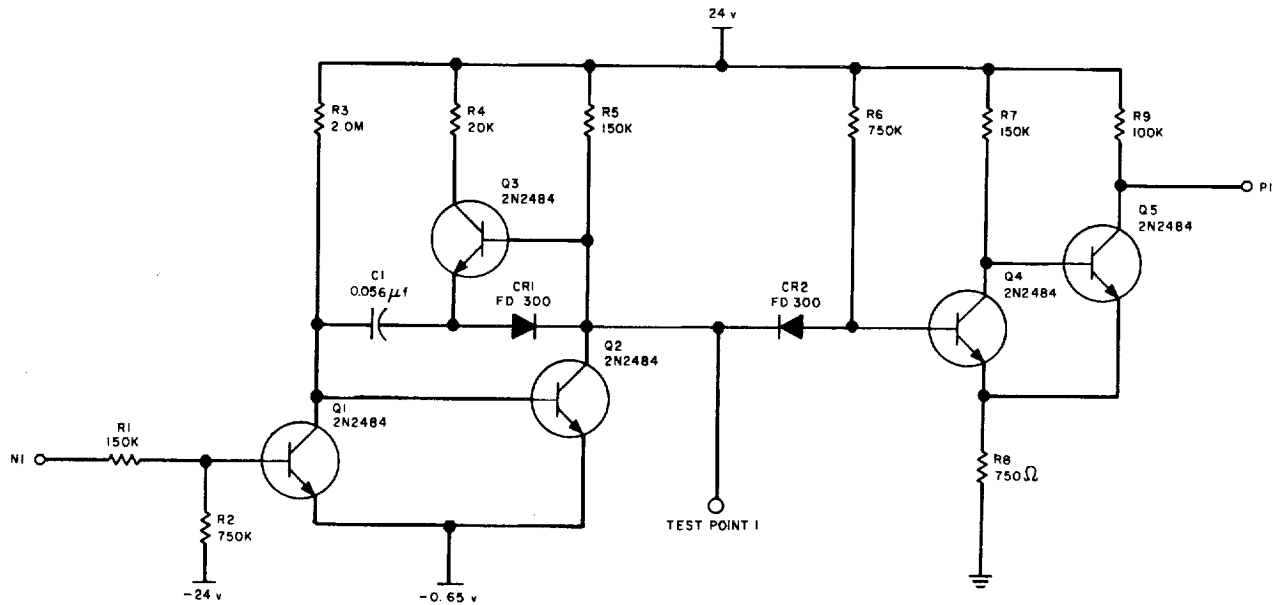
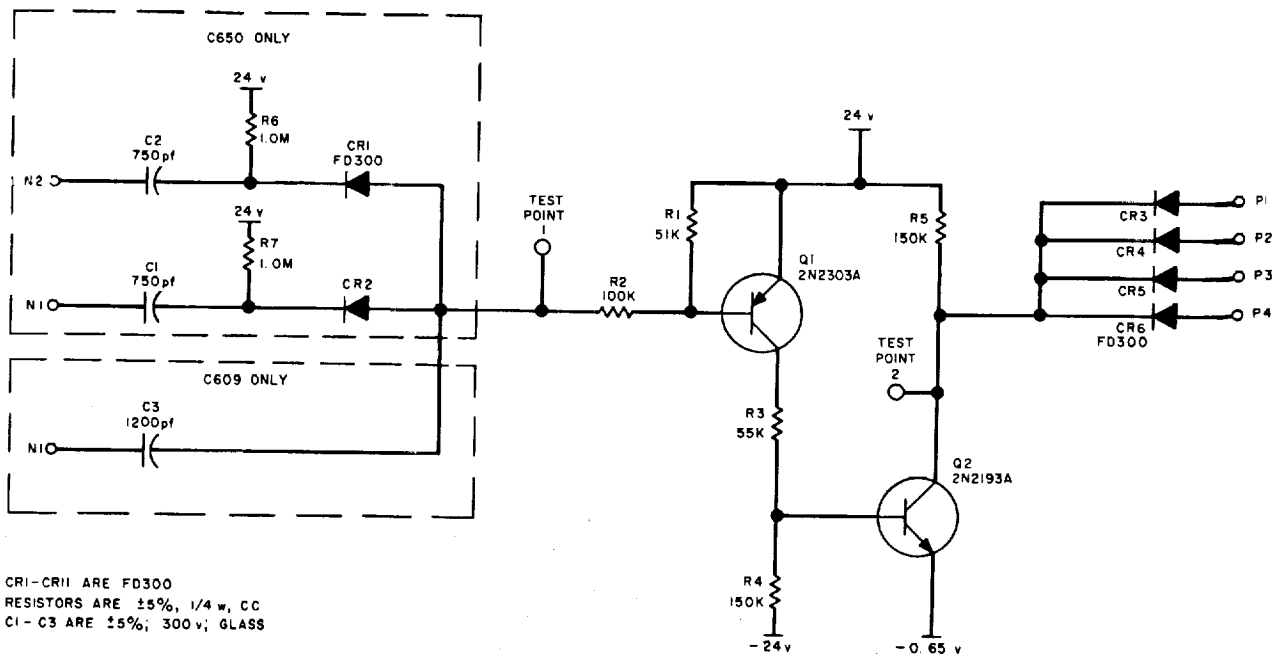


FIGURE 5-135. DECODER WAVEFORMS



C1 IS A $\pm 5\%$, PAPER
 INPUT GATING 68k TO 24 v
 RESISTORS ARE $\pm 5\%$, 1/4 w,
 CC UNLESS OTHERWISE SPECIFIED

FIGURE 5-136. TONE PRESENT DETECTOR C603



CR1-CR6 ARE FD300
 RESISTORS ARE $\pm 5\%$, 1/4 w, CC
 C1-C3 ARE $\pm 5\%$, 300 v, GLASS

FIGURE 5-137. MODE REGISTER CLEAR (C605) AND BIT COUNTER RESET (C609)

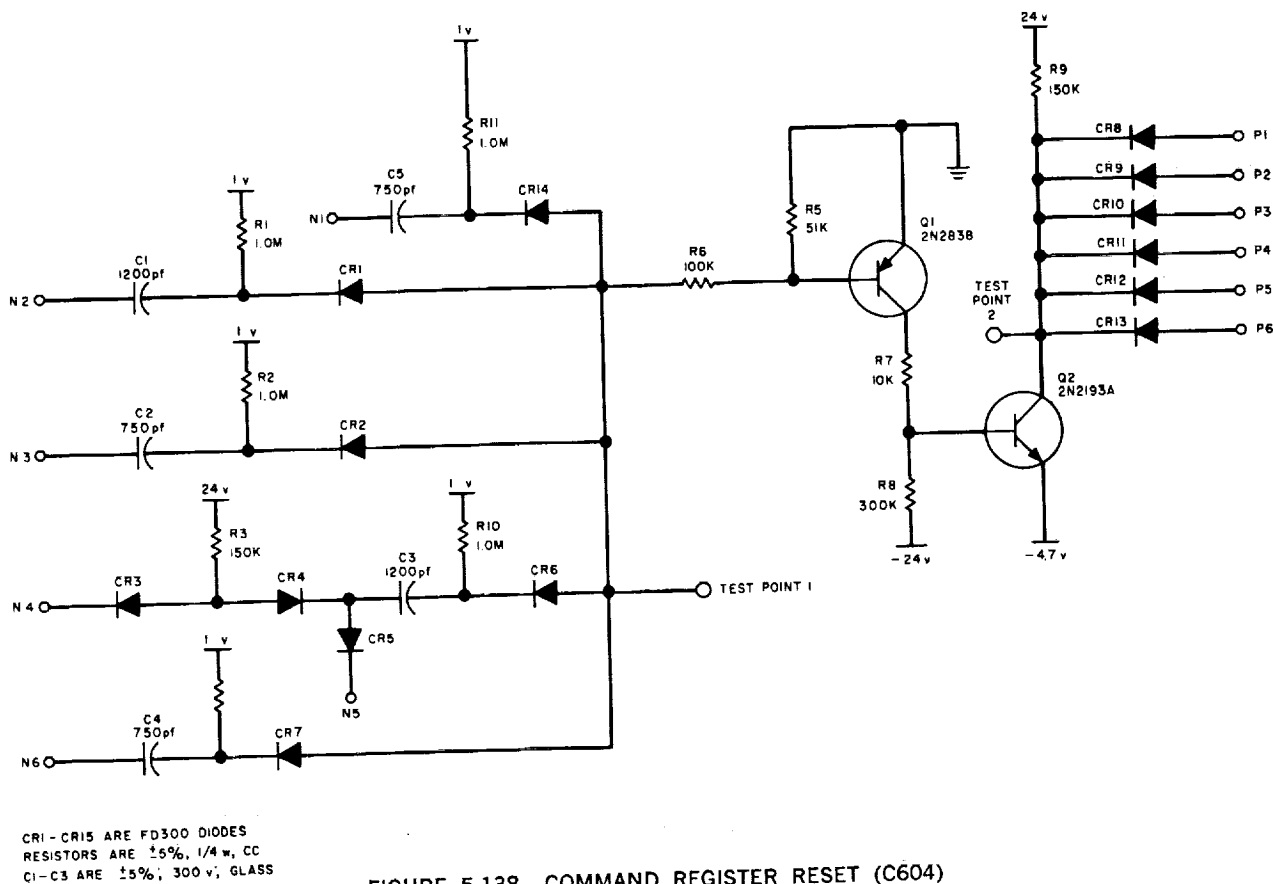


FIGURE 5-138. COMMAND REGISTER RESET (C604)

the tone detectors. When execution occurs, however, the input will be interrupted. Disabling the count mode clock ensures that a spurious change in the detector output will not alter the contents of the command register when multiple executions are necessary.

The detection of word sync establishes the mode of operation and prepares the bit counter to start counting from zero the number of bits shifted into the command register. The bit counter counts the bits by counting the negative transitions of the shift mode clock.

Following the detection of word sync, the enable address is shifted into the command register. When the enable address is in the com-

mand register, the bit counter will have counted these 6 bits, and the gating that controls the enable power circuit will allow it to sample the contents of the command register. If the required command register inputs are high, the enable power circuit will come on, causing the command register to be cleared by the command register reset and the execute circuitry to be enabled so that the presence of an execute tone will cause execution of the contents of the command register.

The command follows the enable address into the command register; and at its conclusion, denoted by the occurrence of the twelfth bit following word sync detection, the shift mode input gating will be disabled by the bit counter.

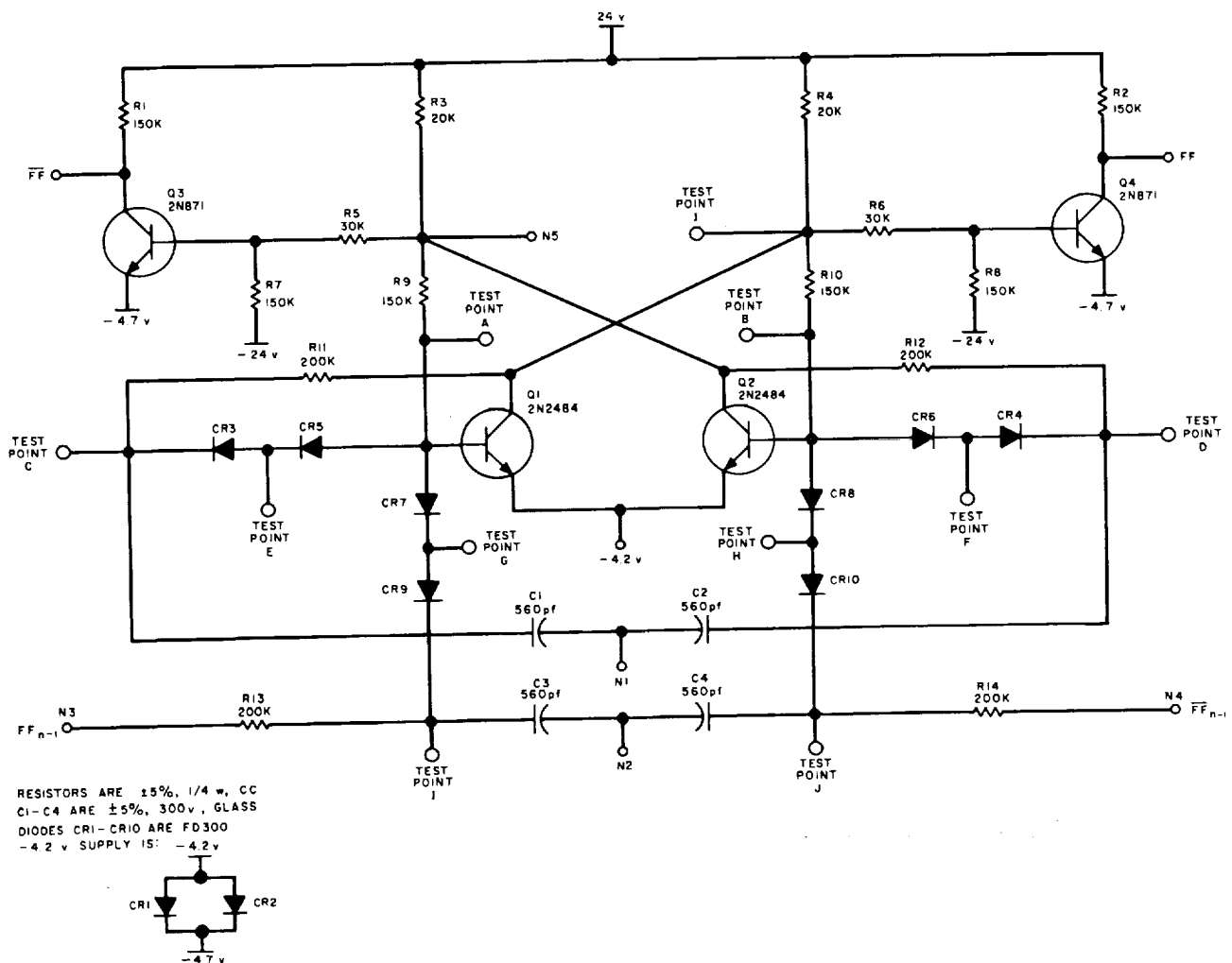


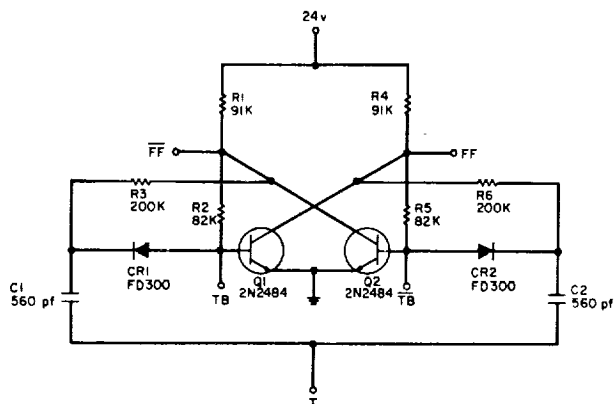
FIGURE 5-139. COMMAND REGISTER FLIP-FLOP ELEMENT (C106-C101)

The conclusion follows the command and consists of the continuous presence of a zero tone. The presence of this tone causes the entire command system to be insensitive to any changes caused by outside interference or noise.

During the conclusion, the contents of the command register will be telemetered to earth for verification. If the command is correct, an execute tone will be sent in place of the zero tone. When this tone is detected by the execute detector (C302, similar to C300 and C301), the

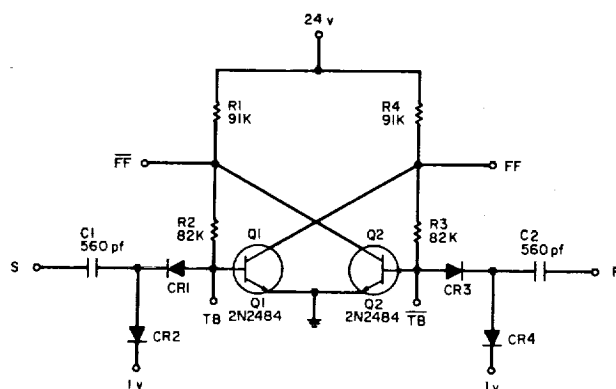
enable power circuit applies power to the diode matrix (Figure 5-144), thus causing execution of the contents of the command register. This completes the command cycle in the primary mode of operation. Once the zero tone is interrupted, reapplication of either tone for 25 bit times will cause the command decoder to be cleared.

Secondary Mode Operation. Operation in the secondary mode begins in the same manner as in the primary mode. The introduction serves



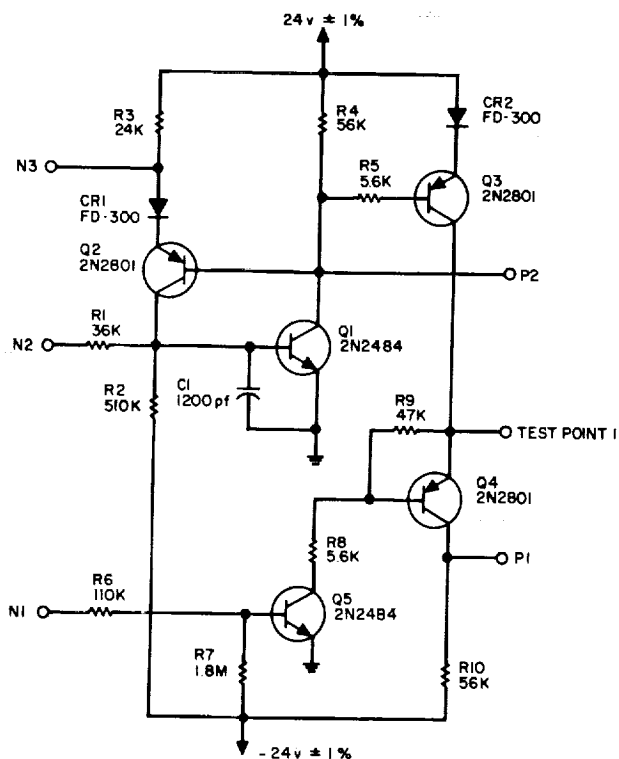
RESISTORS ARE CARBON COMPOSITION $\pm 5\%$, 1/4 w
ALL CAPACITORS ARE GLASS, 300v, 5%

FIGURE 5-140. STANDARD FLIP-FLOP (34/PACE)



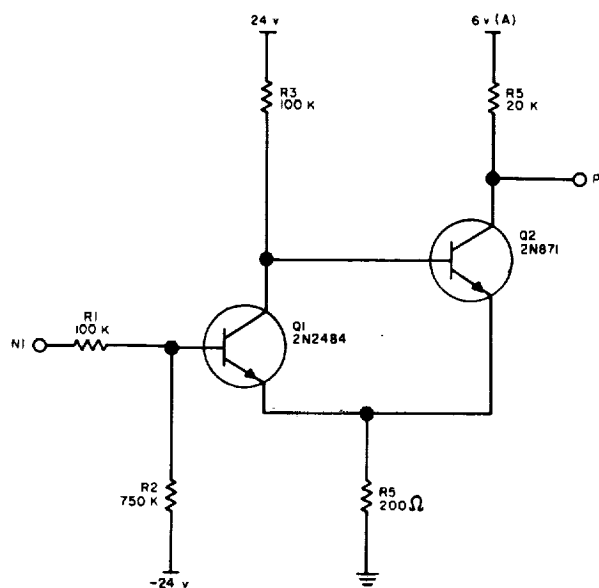
RESISTORS ARE CARBON COMPOSITION $\pm 5\%$, 1/4 w
ALL CAPACITORS ARE GLASS, 5%
ALL DIODES ARE FD300'S
THIS FLIP-FLOP WILL BE USED FOR A110, A113, B100, J100 AND J101.
ADDITIONAL INPUT LOGIC WILL BE RELEASED SEPARATELY WHERE NEEDED.

FIGURE 5-141. R-S FLIP-FLOP (5/PACE)



RESISTORS ARE $\pm 5\%$, 1/4 w
N3 INPUT "OR" TERMS MUST HAVE
A 68K, 5%, 1/4 w "AND" RESISTOR
C1 = GLASS, $\pm 5\%$, 988503-68

FIGURE 5-142. ENABLE POWER CIRCUIT (C601)



RESISTORS $\pm 5\%$, 1/4, CC

FIGURE 5-143. COUNT MODE CLOCK (C303) AND
SHIFT MODE CLOCK (C304)

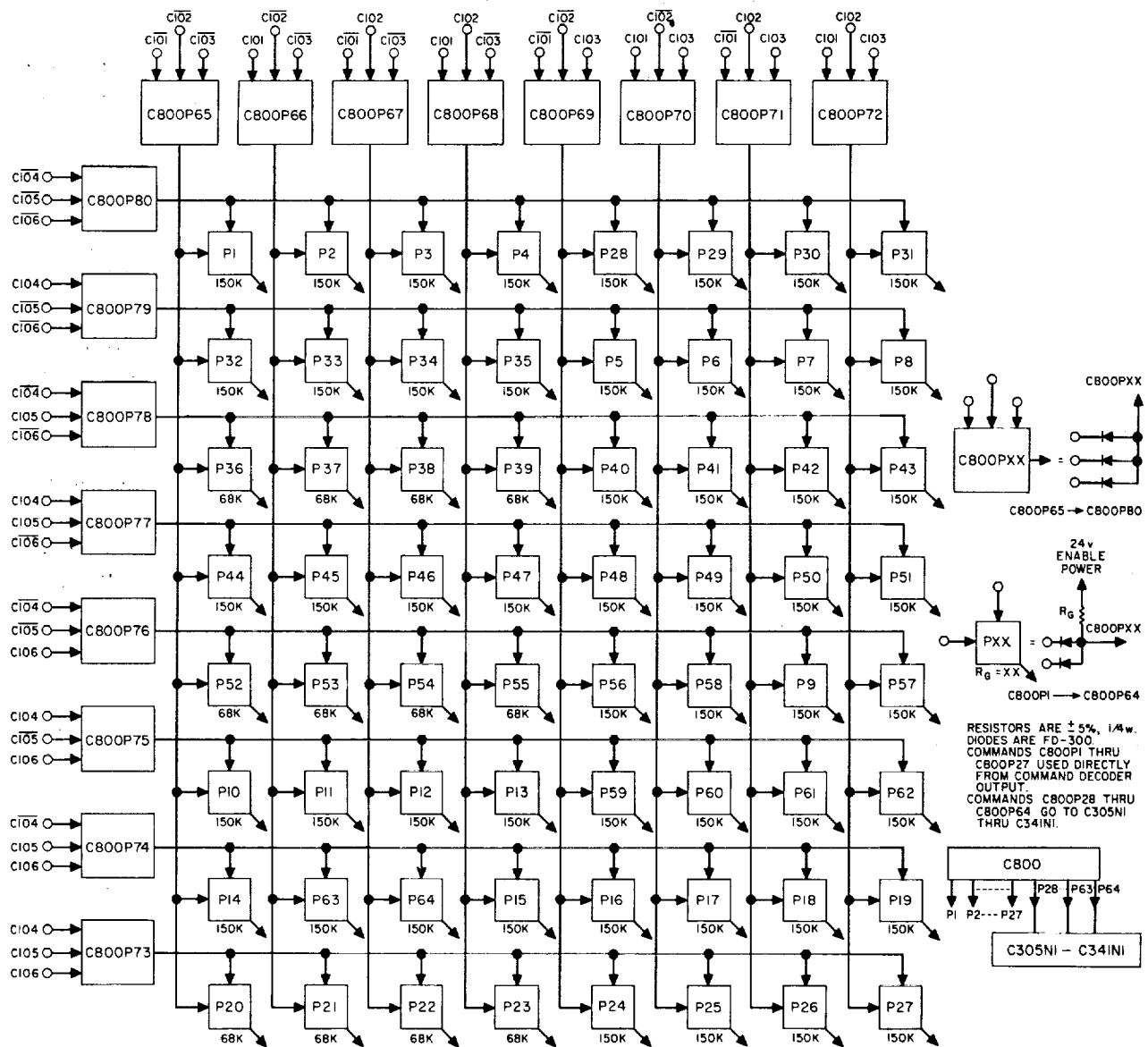


FIGURE 5-144. DIODE MATRIX (C800) PRELIMINARY CIRCUIT

the same purpose and accomplishes the same thing in the same way. The only difference between the introduction for the two modes is in the form of the output from the command receiver. In the secondary mode, the introduction may be accomplished using either tone and the input will be unmodulated.

As with the introduction, the message may be sent with either tone since the secondary mode requires only message tone for operation. At the end of the introduction, the input tone is interrupted completely for 12 bit times. With this interrupt, all similarity to primary mode operation ends. This interrupt corresponds to the

first complete tone interrupt shown in Figure 5-131.

When this tone interrupt occurs, the input gating to the tone interrupt detector (C602, Figure 5-145) will assume a zero state. The tone interrupt detector operates similarly to the tone present detector. The absence of an input causes the first part, a ramp circuit, to run down and to trigger a Schmitt, if the interrupt is sufficiently long (60 milliseconds nominally). Then, provided that both interrupt counter flip-flops are not in the one state and the word sync flip-flop is in the zero state, the negative transition at the output of the tone interrupt detector will cause the interrupt counter to change state.

The first complete tone interrupt will cause C108 to go high enabling the count mode clock input gating, and disabling the word sync flip-flop input gating, the shift mode clock input gating, and the count inhibit power circuit (C608, Figure 5-146). When this occurs, the primary mode is completely disabled.

Following the first complete tone interrupt, the enable address is counted into the command reg-

ister by the count mode clock, which follows the output of the tone detector in operation. Since the shift mode clock is gated into the zero state, it cannot interfere with the count process. When the desired number of counts has been accumulated in the command register, the input is again completely interrupted.

The second complete tone interrupt, like the first, is 12 bits long. During this time the interrupt detector will cause the interrupt counter to change state, with C108 assuming the zero state and C109 the one state. From the time that this change occurs and before the resumption of the pulse amplitude modulated input, the enable power gating will be able to sample the contents of the command register. If the contents correspond to the correct enable address, the enable power circuit will come on, clearing the command register and preparing the execute circuitry for the execution of the command.

At the conclusion of the second complete tone interrupt, the pulse amplitude modulated input is resumed and the command is counted into the command register. When the required number of counts have been entered, the input is com-

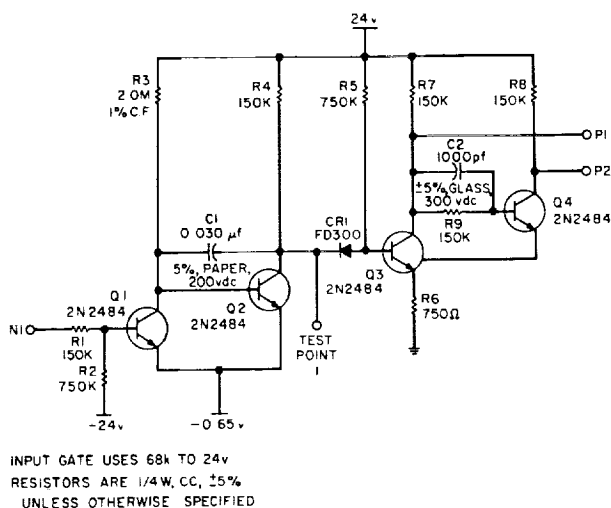


FIGURE 5-145. TONE INTERRUPT DETECTOR (C602)

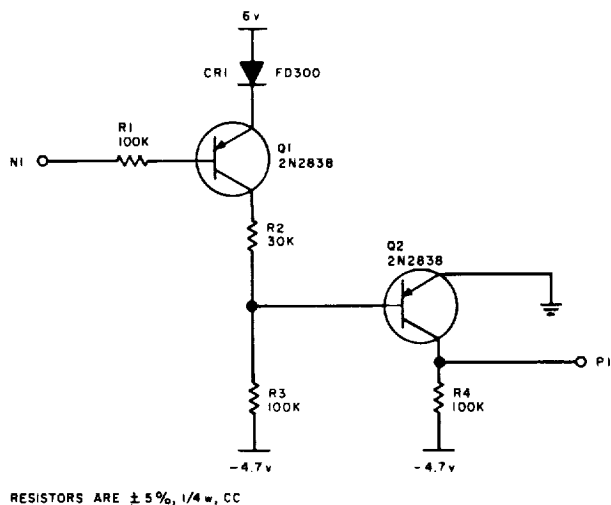


FIGURE 5-146. COUNT INHIBIT POWER (C608)

pletely interrupted again. The detection of this interrupt will cause C108 to change to the one state, disabling the count mode clock. The output of inverter C505 will also go low, disabling the input to the interrupt counter.

After the third interrupt, neither message tone is sent until the decoder is to be cleared. The contents of the command register are verified via telemetry, and if the command contained therein is correct, the execute tone is sent, causing execution of the command. With the completion of the command cycle, the decoder is cleared by inserting the zero or one tone for 25 bit times.

General Information

- **Circuits.** The count inhibit power circuit, (C608, Figure 5-146) is used to prevent the command register from counting when operating in the primary mode. The count mode clock, as stated above, is disabled; but, since the command register must be able to shift or count, there are count connections between the command register flip-flops which can cause problems when operating in the primary mode. For this reason, the flip-flops are connected in a count configuration through interstage gates between the flip-flops. When operating in the primary mode, these gates are disabled by the count inhibit power circuit.

The inverter, C504, is used to prevent commands intended for other satellites from interfering with Advanced Syncom. When the command is in the command register and the shift mode clock is disabled by the bit counter, a continuous zero tone will be present in the Advanced Syncom command format. If this zero tone is not present and C301 goes high, the command register will be cleared. Differences between the OGO format and the Syncom format ensure that a command intended for an OGO satellite will not be executed in an Advanced Syncom satellite.

Two digital readout encoders are included in the block diagram which encode the required digital signal to a form suitable for use by the telemetry encoder. The auxiliary power supplies are necessary to optimize the design of the digital circuits, to increase power utilization efficiency, and to minimize components. The execute amplifier, C342, is used to couple the execute tone directly to the telemetry encoder for direct verification.

Not included in the block diagram, or as a part of the actual command decoder circuit releases, are several circuits which complete the interface between the decoder and the rest of the satellite. These circuits cause commands to be executed in the subsystems affected when a signal is received from the decoder. There are two types of interface coupling, ac and dc, as would be expected. For the most part, the ac coupled circuits (Figure 5-147 through 5-150) control power regulators which supply subsystem power, and which otherwise (if dc coupling were used) would require voltage transformation to maintain control.

The improvement in reliability gained by using ac coupling is great, considering that inter-quadrant wire short circuits to ground will not end control of the regulators. Where the outputs of the decoder are dc coupled, there is no need for voltage transformation and there is no problem with interquadrant shorts.

Detailed attention has been given to the design of the drive circuits and output circuitry of the decoder to ensure that noise generated in the harness will not cause seemingly random executions.

- **Parts and Power.** A total of 1217 parts are used in the Advanced Syncom dual mode command decoder. This parts count may vary

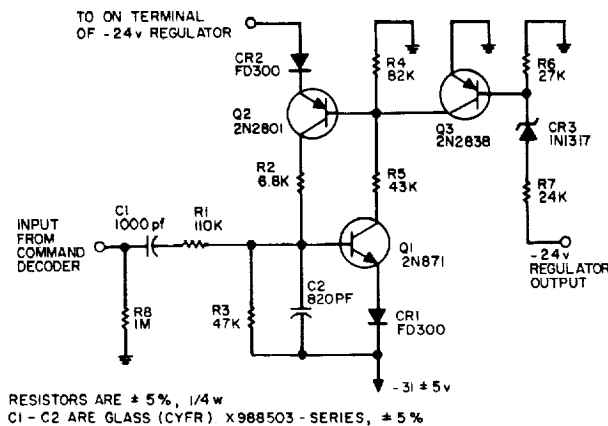


FIGURE 5-147. REGULATOR ON DRIVER

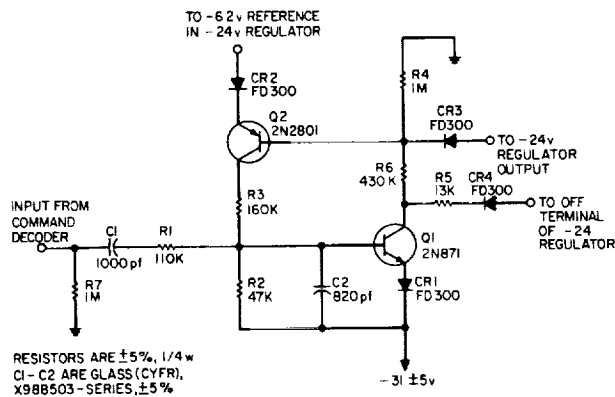


FIGURE 5-148. REGULATOR OFF DRIVER

± 5 percent when final designs are complete. At present these parts are broken down as follows:

Transistors	150
Diodes	
Switching	471
Zener	6
Resistors	
5 percent, 1/4 watt, C.C.	461
1 percent, 1/8 watt, C.F.	35
Capacitors:	
5 percent glass or mica	72
5 percent paper	10
10 percent tantalum	5
Inductors, 1 percent toroid	7
Total	1217

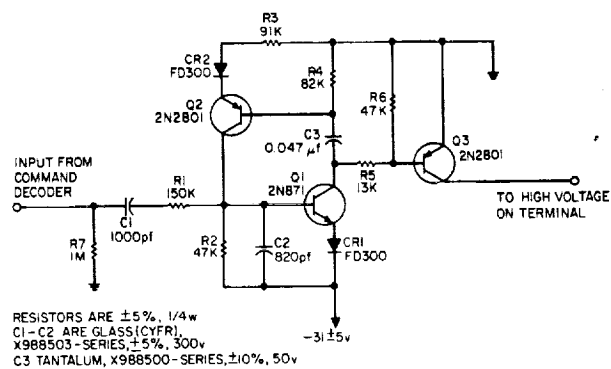


FIGURE 5-149. HIGH VOLTAGE ON DRIVER

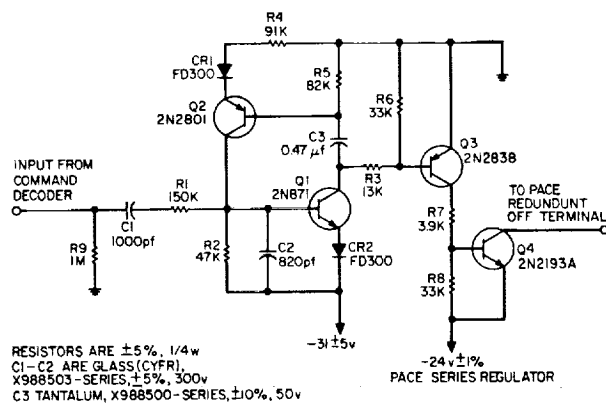


FIGURE 5-150. REDUNDANT PACE OFF DRIVER

The total maximum stand-by power dissipated in the decoder is 1.3 watts. The total maximum power dissipated during the time that a command is being executed is 3.5 watts.

Summation of Progress on Decoder. During the design and development of the Advanced Syncom command decoder, there was question regarding the sensitivity of the decoder to false-command cycles. It was felt, for instance, that an OGO spacecraft itself would present a problem by virtue of its presence and the control of it through the same general command format and console as that used for Advanced Syncom. In addition to this problem, it became obvious that it would be desirable to have some means of preventing an enable power circuit from

coming on randomly because of a single bit error in an enable address sent to another decoder. Finally, the problem of noise was considered with regard to its effect on either mode of operation.

OGO has the greatest tendency to interfere with Syncom only in the primary mode of operation. The output of the command receiver will look essentially the same for either a Syncom or an OGO command. The difference between the two commands, however, is in word length. The OGO command consists of a series of zero bits followed by a single one, which is an OGO word sync, and then by three decoder address bits. The command is 10 bits long and is followed by the command complement and a minimum of 11 zeros. The total number of bits to the end of the command complement from word sync is 24. Since Advanced Syncom, for a similar cycle, requires only 14 bits, the last 10 bits, the command complement, would occur after Syncom's cycle was concluded and the shift mode clock was disabled. At this time, to conclude a command, Advanced Syncom requires that a continuous zero tone be present. If this does not occur, the Advanced Syncom decoder command register will be cleared by the first one bit which occurs after the disabling of the shift mode clock. This is done by C504 in conjunction with the command register reset, C604.

The advantage of detecting a single bit error in the enable address is questionable because of the number of parts required if a formal check of some type is made on the command register at the enable time. If the address is coded properly at the outset, a simple parity check can be made by including all bits in the register in the enable input gate and requiring that all addresses contain an odd number of ones; with no increase in parts, a simple check can be made by

sacrificing half of the available addresses. This sacrifice of addresses appears reasonable since 32 addresses should be adequate.

The final problem considered was that of noise interference. The effects of random noise on the Advanced Syncom decoder were considered in two parts. The first of these was the effect of noise on the decoder during the time that an input is present. It was found that the signal-to-noise ratio of the command receiver in conjunction with the decoder filters is adequate to ensure that noise is not a significant problem.

The second part of the problem involves the decoder when no input is present. It was found that reasonable choice of thresholds in the detectors would eliminate the major portion of the problem. In general, the quantitative analyses made on the decoder consider the receiver characteristics, the decoder filter bandwidths, and the detector design. They yield a quantitative probability that a detector will produce an erroneous output. Design has been directed toward reducing this probability to a minimum without impairing operation. It should also be noted that the probability of occurrence of individual false outputs from the detector has no relationship to the sequence of outputs required before a false command could accumulate in the command register.

As a final, simple means of ensuring that errors will not accumulate in the decoder, the command register is cleared every 315 minutes by an input from the central timer.

The preliminary design of the Advanced Syncom decoder is complete. The design has been tested on a subsystem level, and in all respects it performs as it was designed. The last step in the design is the evaluation of the breadboard. This evaluation will be based on operation at temperature extremes, in extremely noisy sur-

roundings, and with inserted noise at each subsystem input. The latter test should yield valuable information regarding decoder sensitivity. The decoder will also be evaluated with regard to necessary input signal levels and modulation index.

The command decoder has been mated with the OGO console and been found to operate properly in the primary mode. The only exception to this statement is that the real time execute has not, at this time, been implemented and tested. Figure 5-151 shows the OGO console mated with the command decoder.

The OGO standard has an NRZ, frequency-shift-keyed modulated word format. The message tones are amplitude modulated with a low frequency sine wave which, when detected, serves as a bit sync clock.

Real time execution was needed for the following functions:

- a) Implementation of the backup mode of the jet control electronics.
- b) Implementation of beam adjustment by command.
- c) Implementation of beam control while operating in an eclipse.
- d) Verification of command prior to execution.

Telemetry Channel Allocations. During the report period, the essential operational characteristics of the Advanced Syncom telemetry system were defined. The salient points are as follows:

- 1) The telemetry system will be the GSFC standard PFM system with approved modifications as listed below.
- 2) The format will be comprised of eight frames.

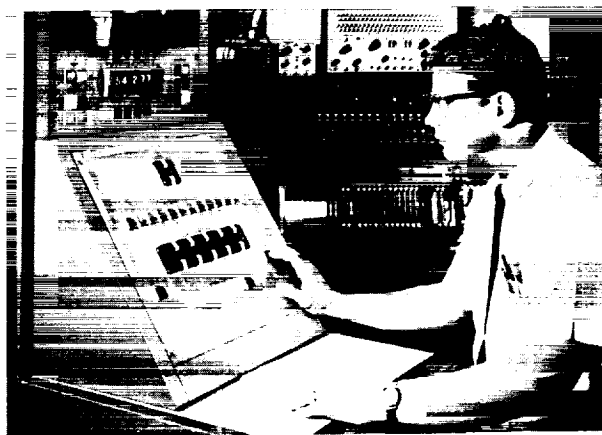


FIGURE 5-151. OGO CONSOLE

- 3) The reference frequency will be generated by driving the voltage-controlled oscillator above 15.5 kc rather than by use of a separate oscillator.
- 4) The bit rate will be 48.5 ± 5 percent bits per second.
- 5) The telemetry system will use two frequencies — 136.470 and 136.980 mc — with at least one transmitter operating at each frequency.
- 6) Provisions will be made in the telemetry system for indicating actuation of bipropellant solenoids and all squib firings.
- 7) Provisions will be made for indicating ψ , ψ_2 , and the execute pulse.
- 8) Each frame will be composed of 16 channels as follows:

Channel 0 will be a frame sync.

Channels 1 through 7 will contain a series of seven data bursts, each preceded by a reference burst.

Channels 8 through 15 will be a continuous data burst.

- 9) The information format will be as shown in Figure 5-152.

5. Spacecraft System Design

CHANNEL	0	1	2	3	4	5	6	7	8 THRU 15
0	000	SPACECRAFT IDENT	TELEMETRY IDENT	TELEMETRY CALIBRATION	TELEMETRY POWER OUT	TRANSPONDER NO. 1 RECEIVED SIGNAL STRENGTH			
1	5	CURRENT	BUS NO. 1 BATTERY NO. 1 VOLTAGE	BATTERY NO. 2 VOLTAGE	SPARE	TRANSPONDER NO. 2 RECEIVED SIGNAL STRENGTH			
2	41	DECODER NO. 1 ENABLE, COMM REGISTER, MODE	DECODER NO. 2 ENABLE, COMM REGISTER, MODE	DECODER NO. 3 ENABLE, COMM REGISTER, MODE	SPARE	TRANSPONDER NO. 3 RECEIVED SIGNAL STRENGTH			
3	5	DECODER NO. 4 ENABLE, COMM REGISTER, MODE	ψ_2	ANGLE	BIPROPELLANT SYSTEM NO. 1 FUEL PRESSURE	TRANSPONDER NO. 4 RECEIVED SIGNAL STRENGTH			
4	010	SYSTEM NO. 1 OXIDIZER TEMPERATURE	FUEL PRESSURE	FUEL TEMPERATURE	SPARE	BUS VOLTAGE - UNREGULATED BUS NO. 1			
5	5	PACE LOCK, TIMER SELECTION BEAM POSITION	TRANSPONDER 1 TRANSMITTER POWER	TRANSPONDER 2 TRANSMITTER POWER	TRANSPONDER 3 TRANSMITTER POWER	BUS VOLTAGE - UNREGULATED BUS NO. 1			
6	011	TWT AREA TEMPERATURE	NO. 1	NO. 2	NO. 3	BUS VOLTAGE - UNREGULATED BUS NO. 2			
7	5	*SOLAR PANEL TEMPERATURE	*STRUCTURE TEMPERATURES	SPARES	SPARES	BUS VOLTAGE - UNREGULATED BUS NO. 2			

TYPICAL CHANNEL FOR CHANNELS 1 THRU 7

REFERENCE BURST	DATA BURST
-----------------	------------

CHANNELS 8 THRU 15 CONTAIN NO REFERENCE BURST, i.e., CONTINUOUS DATA BURSTS.

*DIFFERENT SENSOR FOR EACH ENCODER

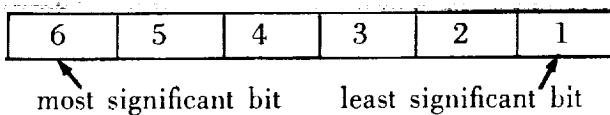
FIGURE 5-152. TELEMETRY FORMAT

- 10) The location of temperature sensors has tentatively been decided upon, as shown in Table 5-22. A total of 16 sensors will be included on each spacecraft. The fuel and oxidizer tank sensors and traveling-wave tube sensors are redundant for each of the two telemetry encoder frequency assignments.

Command Allocation. The format for the Advanced Syncom commands consists of three major subdivisions: 1) word sync, 2) command address, and 3) command number. A 6-bit system will be used for both the address and the command. The message format is described in the following paragraphs.

Word Sync. The word sync is used to define the start time for the command which follows. Word sync in the shift mode will be a series of zeroes followed by a one and a zero. In the count mode, a series of zeroes or ones followed by a 10-bit time interrupt will designate the word sync.

Command Address. The second 6 bits of the command message will consist of a spacecraft number and a quadrant number for a specific command system. The format is defined as follows:



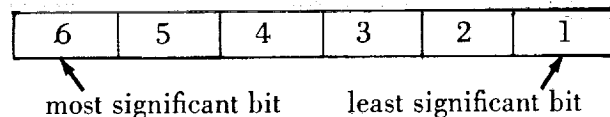
where bits 6, 5, 4, 3 are the spacecraft number, and bits 2, 1 are the quadrant number. This format makes it possible to address a maximum of 64 quadrants.

Command Number. The last group of six bits of the command message will consist of the command number. The format is defined as follows:

TABLE 5-22. TEMPERATURE SENSORS

Sensor Description and Location	Number of Sensors	Telemetry Encoder Number*
Reaction control system — fuel and oxidizer tanks	4	I and II
Near traveling-wave tube collector — ribs 3 and 9	2	I and II
Solar panels		
Near radial jet 1	1	II
Near either end of vehicle	1	I
Antenna cruciform	1	I
Sun sensor	1	II
Aft bulkhead	1	I
Battery	1	I
Apogee motor mounting point	1	II
Radial jet 1 mounting surface	1	II
Forward shelf near telemetry transmitter	1	II
Axial jet 1 mounting surface	1	I

*Telemetry encoder transmitter I operates at 136.470 mc; telemetry encoder transmitter II operates at 136.980 mc.



Whenever applicable, bits 6, 5, 4, 3 will designate the command type, and bits 2, 1 will designate the appropriate quadrant. The command function must include capability to control each quadrant where applicable. The command format is shown in Table 5-23. For ease of discussion, a modified octal representation of the binary command is defined as follows:

- 1) The first digit is zero or one and is identical to the most significant binary bit in the register.
- 2) The second digit goes from zero to seven and is the octal representation of the next three most significant bits in the register.
- 3) The third digit goes from zero to four and is the modified octal representation of the

TABLE 5-23. COMMAND FORMAT

Command Function	Modified Octal Command	Binary Word					
		6	5	4	3	2	1
Spares	00-1 00-2 00-3 00-4	0 0 0 0	0 0 0 0	0 0 0 0	0 0 0 0	0 1 1 0	1 0 1 0
Transponder							
Frequency translation mode on, multiple access mode off	01-1 01-2 01-3 01-4	0 0 0 0	0 0 0 0	1 0 0 0	1 1 1 1	0 1 1 0	1 0 1 0
Multiple access mode on, frequency translation mode off	02-1 02-2 02-3 02-4	0 0 0 0	0 0 0 0	1 1 1 1	0 0 0 0	0 1 1 0	1 0 1 0
Transponder off	04-1 04-2 04-3 04-4	0 0 0 0	1 1 1 1	0 0 0 0	0 1 1 0	0 1 0 0	1 0 1 0
Spares	03-1 03-2 03-3 03-4	0 0 0 0	0 0 0 0	1 1 1 1	1 1 1 0	0 1 1 0	1 0 1 0
Transmitter							
Traveling-wave tube 1 filament on, traveling-wave tube 2 filament off	05-1 05-2 05-3 05-4	0 0 0 0	1 1 1 1	0 0 0 0	1 1 1 1	0 1 1 0	1 0 1 0
Traveling-wave tube 2 filament on, traveling-wave tube 1 filament off	06-1 06-2 06-3 06-4	0 0 0 0	1 1 1 1	1 1 1 1	0 0 0 0	0 1 0 0	1 0 1 0
Traveling-wave tube high voltage on	07-1 07-2 07-3 07-4	0 0 0 0	1 1 1 1	1 1 1 1	1 1 1 0	0 1 1 0	1 0 1 0
PACE							
PACE on for quadrant specified, all other PACE off	10-1 10-2 10-3 10-4	1 1 1 1	0 0 0 0	0 0 0 0	0 1 1 0	0 1 1 0	1 0 1 0
Command Function	Modified Octal Command	Binary Word					
		6	5	4	3	2	1
PACE							
All PACE off	11-1	1	0	0	1	0	1
Accept antenna position	11-2	1	0	0	1	1	0
Phase shifter driver regulator on	11-3	1	0	0	1	1	1
Phase shifter driver regulator off	11-4	1	0	0	1	0	0
Eclipse mode operation on	12-1	1	0	1	0	0	1
Eclipse mode operation off	12-2	1	0	1	0	1	0
Central timer select	12-3	1	0	1	0	1	1
Spare	12-4	1	0	1	0	0	0
Telemetry							
Telemetry on	13-1 13-2 13-3 13-4	1 1 1 1	0 0 0 0	1 1 1 1	1 1 1 0	0 1 1 0	1 0 1 0
Telemetry off	14-1 14-2 14-3 14-4	1 1 1 1	1 1 1 1	0 0 0 0	0 1 0 0	0 1 1 0	1 0 1 0
Bipropellant system							
Fire jets 1 and 2, +90 degrees axial	15-1	1	1	0	1	0	1
Fire jets 1 and 2, +90 degrees radial	15-2	1	1	0	1	1	0
Fire jets 1 and 2, -90 degrees radial	15-3	1	1	0	1	1	1
Spare	15-4	1	1	0	1	0	0
Bipropellant system							
Fire axial jet, system 1 (backup mode)	16-1	1	1	1	0	0	1
Fire radial jet, system 1 (backup mode)	16-2	1	1	1	0	1	0
Fire axial jet, system 2 (backup mode)	16-3	1	1	1	0	1	1
Fire radial jet, system 2 (backup mode)	16-4	1	1	1	0	0	0
Firing circuits							
Fire control system squibs	17-1	1	1	1	1	0	1
Release control system locking pins	17-2	1	1	1	1	1	0
Fire apogee engine	17-3	1	1	1	1	1	1
Spare	17-4	1	1	1	1	0	0

last two or least significant bits in the register. This digit defines the quadrant, where applicable, quadrant 4 being represented by octal four but binary zero.

Examples:

Modified Octal to Binary

0	5	2	(Quadrant 2 traveling-
0	101	10	wave tube filament on)

Binary to Modified Octal

1	110	01	
1	6	1	(Fire axial jet, system 1)

CENTRAL TIMER

The circuit design of the central timer is complete and final circuits have been released. A flat card engineering model of the central timer was constructed and has operated successfully over the temperature range, -20 to 65°C ; the

design temperature range is 0 to 50°C . (See Figures 5-153, 5-154, and 5-155.)

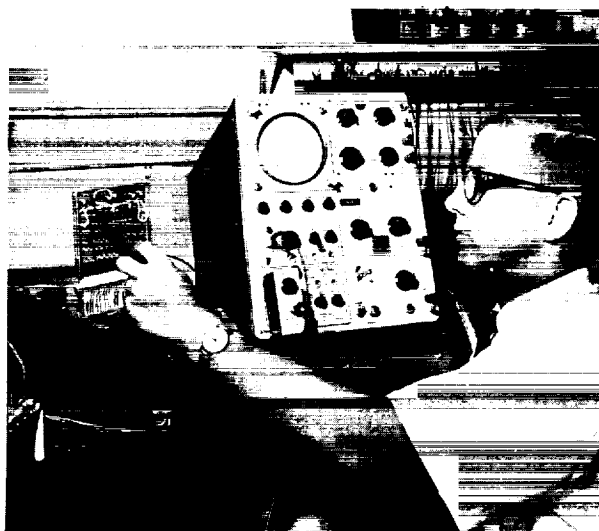


FIGURE 5-154. CIRCUIT CARD CONTAINING THREE COUNT-OF-EIGHT STAGES UNDERGOING CHECKOUT

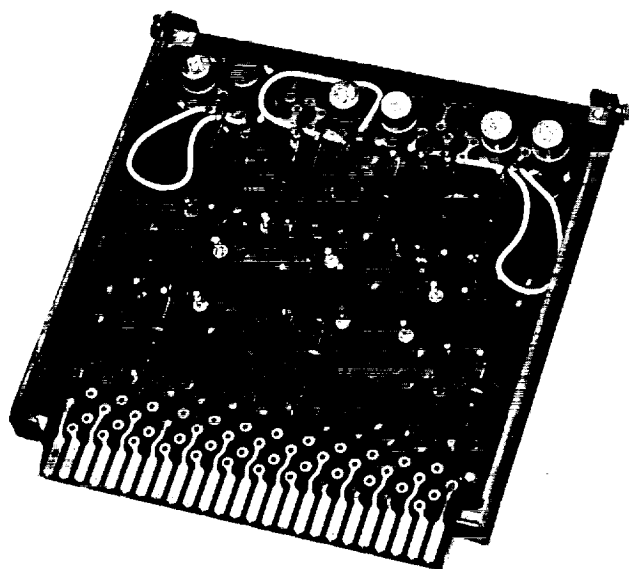


FIGURE 5-153. CIRCUIT CARD CONTAINING THREE COUNT-OF-EIGHT STAGES

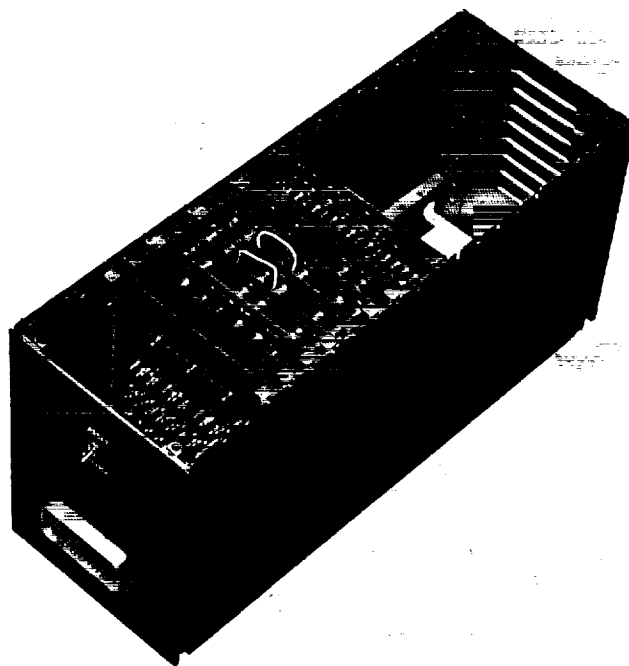


FIGURE 5-155. CENTRAL TIMER ADVANCED ENGINEERING MODEL CHASSIS

Other environmental tests (vibration and non-operating temperature cycling) of the tuning fork oscillator and scaler stages have also been initiated.

Tests to determine the long-term stability of intermediate counts within a core are also being conducted. Although testing should continue for many months, preliminary results indicate a high level of core stability.

The method of packaging the central timer is under investigation. It is very desirable to have the size of a replacement core module as small as possible, but because of the inherent variability of magnetic characteristics between cores (even from the same lot) and because of the critical coupling required between scaler stages, measures must be taken which provide tight control of scaler stage-to-stage compatibility. The present design offers two packaging alternatives:

- 1) Make the replacement core module small (only two or three scaler stages) and require a very stringent core selection and adjustment procedure to ensure standard module inputs and outputs.
- 2) Put all cores (ten) of a timer in a single module and make this the replacement core module. Each core is then adjusted to drive only the following core. With this method, no selection or standardizing procedure is necessary.

Alternative 1 results in a small replacement module, but at the expense of a very costly selection and adjustment procedure. Alternative 2 results in a relatively costly replacement module but the adjustment procedure is quite simple. A strong point for alternative 2 would be the

higher reliability of the ten-core module as compared to an ordinary module, in that the core module has only ten parts and few internal connections. Such a module should also be relatively insensitive to failures by input-output shorting provided a current-limiting fuze is provided on any external -12 volt supply used to test the scaler modules.

A third alternative for packaging the timer is to provide relatively small core modules (three to four stages), each with a shaper stage at its input. This approach would give both a small replacement module and a simple adjustment procedure, at the expense of the seven additional parts for the shaper stage. A new design for this shaper stage is required but the effort should be minor. No selection between alternatives has yet been made.

Progress Items. During the report period the following functional timer definitions have been specified:

- 1) The timer outputs will be those indicated above.
- 2) Prior to separation, the last four scaler stages will be held reset. This will then result in an apogee timer error of +0, -21 seconds.
- 3) A flight plug will be incorporated in series with the reset line to allow complete system checkout of the central timer.
- 4) A test point will be provided at the input to the seventh scaler stage. This point will be driven externally to provide a rapid checkout of the last three scaler stages, the output latch, the apogee driver input gate, and the apogee squib driver.

Tests have been initiated to determine the long-term stability of intermediate counts within a core. The plan has been to incrementally increase the time interval between full scale counts and note whether there is a maximum time interval beyond which unreliable operation results. To date, a sample of five count-of-eight stages has been run to 80 minutes between full scale counts. Each core has been cycled ten times at that time interval. No miscounts (or variations of the last pulse from its optimum position on the ϕ -H curve) have been observed. Tests at 240 minutes between full-scale counts are now being conducted. The data is not yet complete, but again, no miscounts or variations have been observed. Three count-of-eight stages have been set to a count of four on a Friday and power turned off for the weekend. On the following Monday morning power was turned on and each core was incremented to full scale. In all cases, the proper count was observed with the last pulse remaining optimally positioned on the ϕ -H curve.

Functional Description. The purpose of the central timer in the Advanced Syncom spacecraft is to provide an accurate timing source within the vehicle which serves as a real time reference for many spacecraft operations.

Four timers are provided per spacecraft, one in each of the vehicle's four quadrants. Each timer uses the command regulator +24 and -24 volt power supplies of its local quadrant. The timers are complete within themselves, and thus operate independently of one another.

The central timer generates the following timing functions:

- 1) An output is provided at 194.18 ± 0.02 percent pps, which serves as the telemetry encoder commutation clock frequency for the local quadrant.

- 2) Output pulses at 2.8126 ± 0.02 percent minutes are provided as time-of-day correction inputs to the beam positioner of each PACE. To assure that a timer failure will not disable the time-of-day correction inputs for the beam positioner in that quadrant, each timer's 2.81-minute output is provided to all four PACEs. Thus any of the four timers can be selected to drive any of the four PACE beam positioners.
- 3) The third output of the central timer is the apogee motor enable signal which is generated 315.01 ± 0.06 , -0.41 minutes after spacecraft separation. Redundancy against false triggering of the apogee motor due to a premature timer output is provided, in that a minimum of two timers must generate an enable output before the apogee motor will be fired. Additionally, up to two timers may fail completely and proper apogee firing will still be maintained.
- 4) The final output of the central timer is a command register reset signal which is generated every 315.01 ± 0.06 minutes. Its function is to reset the decoder command register every 315 minutes to ensure against the generation of erroneous commands due to noise.

Block Diagram Description. The block diagram of the central timer is shown in Figure 5-156. All power is derived from command +24 volts and command -24 volts which are on continuously. The -12 volts is generated internally and serves as the prime voltage supply for the core scaler stages.

The real time reference for each timer is a 1553-cps tuning fork oscillator. The oscillator output feeds into the core input amplifier which squares the input sine wave and provides the

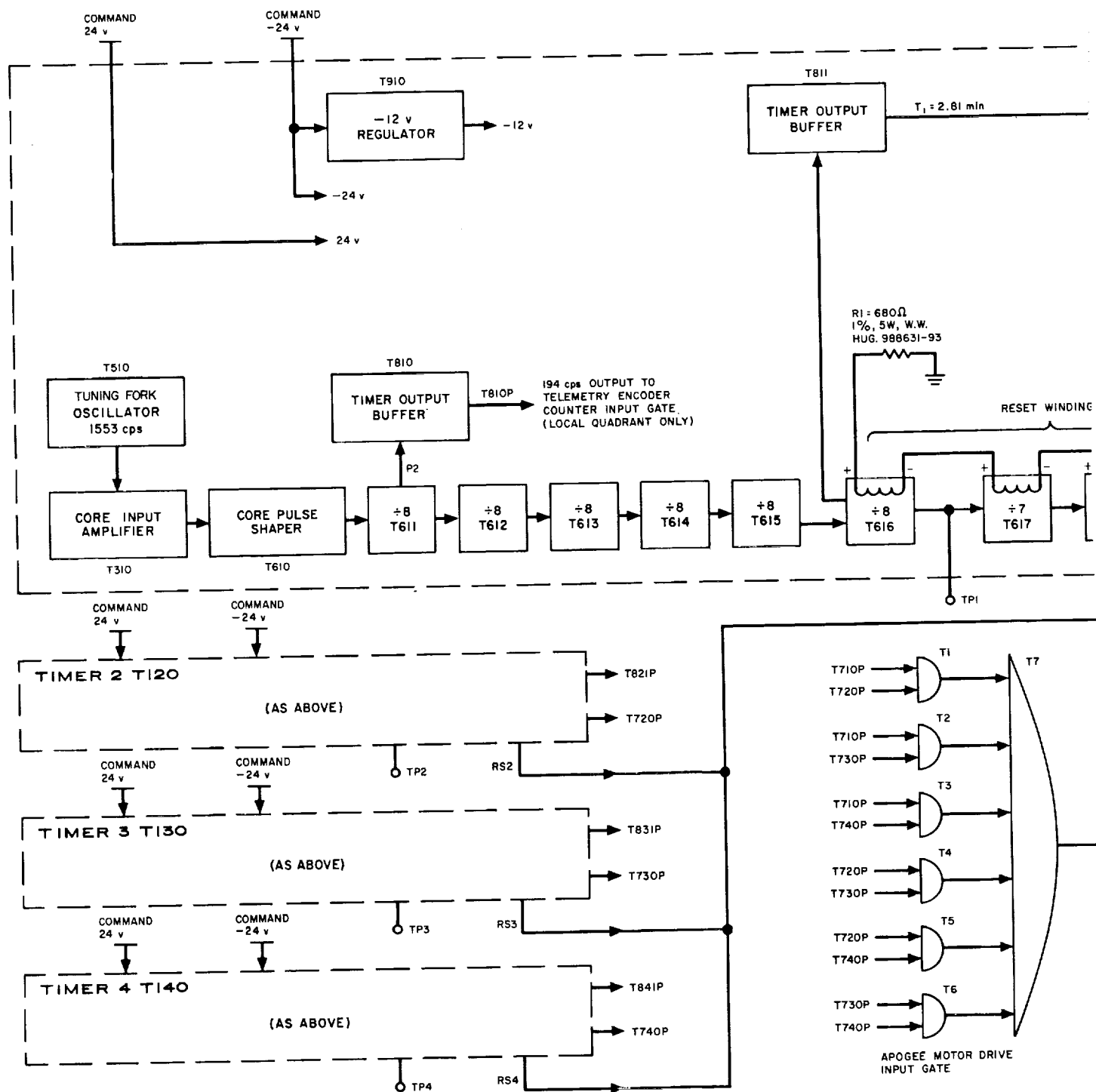
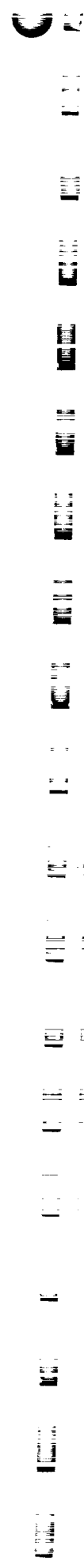
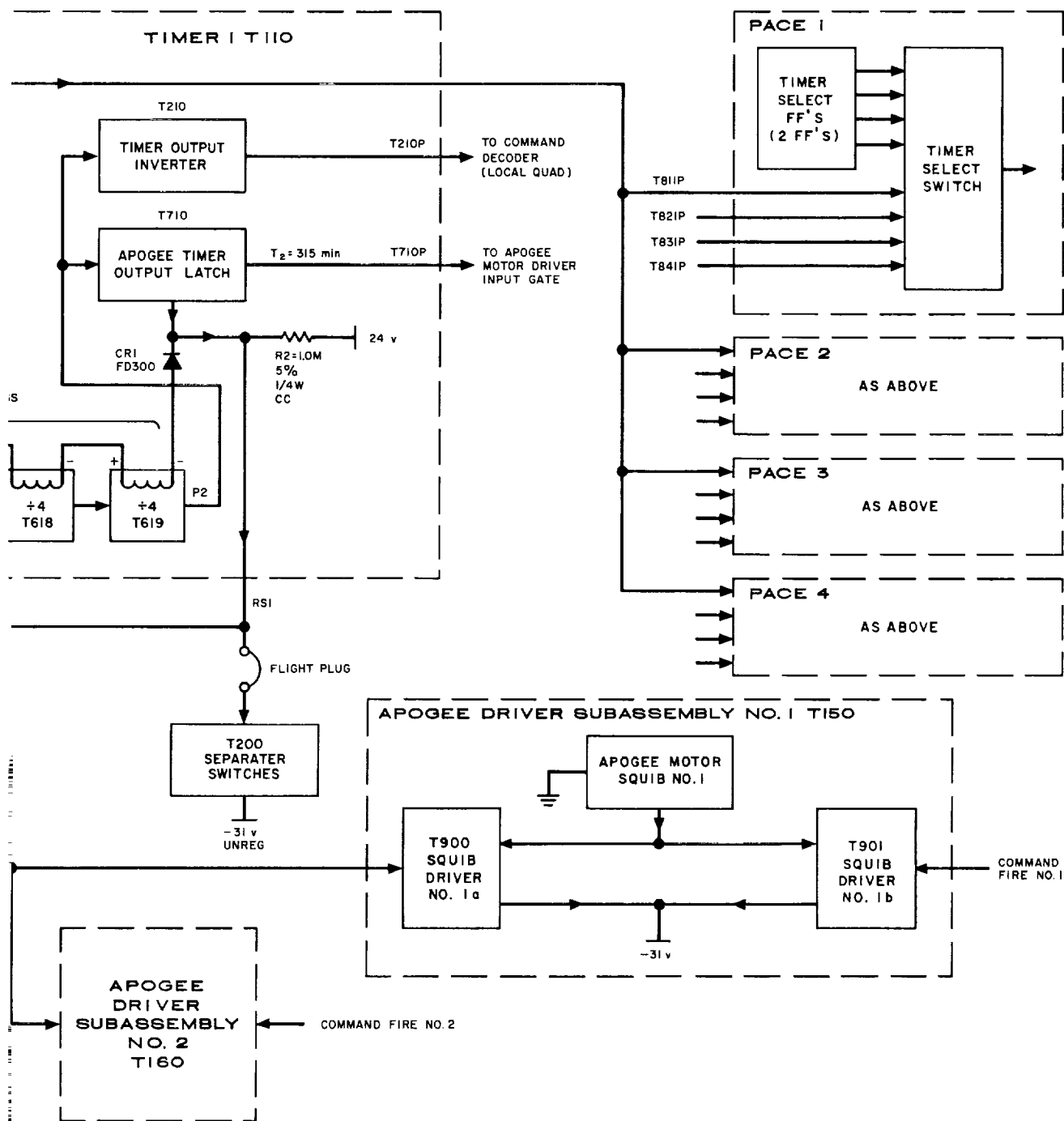


FIGURE 5-156. CENTRAL TIMER





2

required current gain. This output then drives the core pulse shaper which provides a fixed volt-second area pulse into the first count-of-eight core frequency scaler. Each scaler stage in turn drives the following stage. The outputs of the first and sixth count-of-eight stages drive both the following stage and a timer output buffer. The output of the last scaler stage drives both the apogee motor output latch and the timer output inverter. Two latches going high enable the apogee squib drivers via the apogee motor driver input gate.

The separation switches shown are required to inhibit operation of the last four frequency scalers until the Advanced Syncom spacecraft separates from the Agena launch vehicle. The actual number of separation switches to be used has not yet been determined.

Core Frequency Scalars. The Advanced Syncom central timer uses incrementally saturating magnetic cores to perform the frequency scaling operation. By using one shaper stage and nine count stages, a frequency division of $8^6 \times 7^1 \times 4^2 = 29.36(10)^6$ is obtained. The total number of parts required is 70.

Product Design. The central timer, consisting of components including a tuning fork and ten tape-wound toroidal transformers, has been designed into six welded encapsulated modules of the type described in the PACE product design section. The component density of the six welded modules is 19.5 components per cubic inch and the weight of the timer will be approximately $\frac{1}{3}$ pound.

The tuning fork is a vibrating device and is affected by some portions of the Advanced Syncom environment, the most severe of which are spin acceleration and lateral vibration. The tuning fork is oriented such that the response of the

tynes to spin acceleration (10 g) and lateral vibration is minimized. The tuning fork, which will be mounted as close as possible to the quadrant attachments, will see an amplification of twelve at the resonant frequency of the unit (approximately 200 cps) resulting in a maximum response of 30 g. As a result of these precautions the tuning fork will operate within its required limits with adequate margins. The Hughes specification for the tuning fork is now being written.

The countdown of the tuning fork frequency is accomplished by scalars, utilizing tape-wound orthonol cores with count reductions as high as 8 to 1. The wound core is encapsulated as a component to ensure low handling losses and facilitate welded module fabrication. Three scaler stages are packaged in a module with the capability of fine tuning each core (a total of three per module) with a two-turn winding in the unencapsulated module stage. Hughes specifications are now in progress for both the unwound core and the wound core package. The product design of the timer will be completed during this program.

Circuit Descriptions

Tuning Fork Oscillator. The tuning fork oscillator circuit is shown in Figure 5-157. Transistors Q3 and Q4 form a two-stage common emitter amplifier. There is positive feedback from the collector of Q4 to the base of Q3 through the tuning fork. The voltage transfer ratio of the tuning fork, defined as the ratio of pickup coil voltage to drive coil voltage, is about 1:30 at the fork resonant frequency, 1553.446 cps. The voltage transfer ratio decreases very rapidly as the frequency is changed from the resonant frequency. The frequency characteristic is similar to that of a tuned circuit

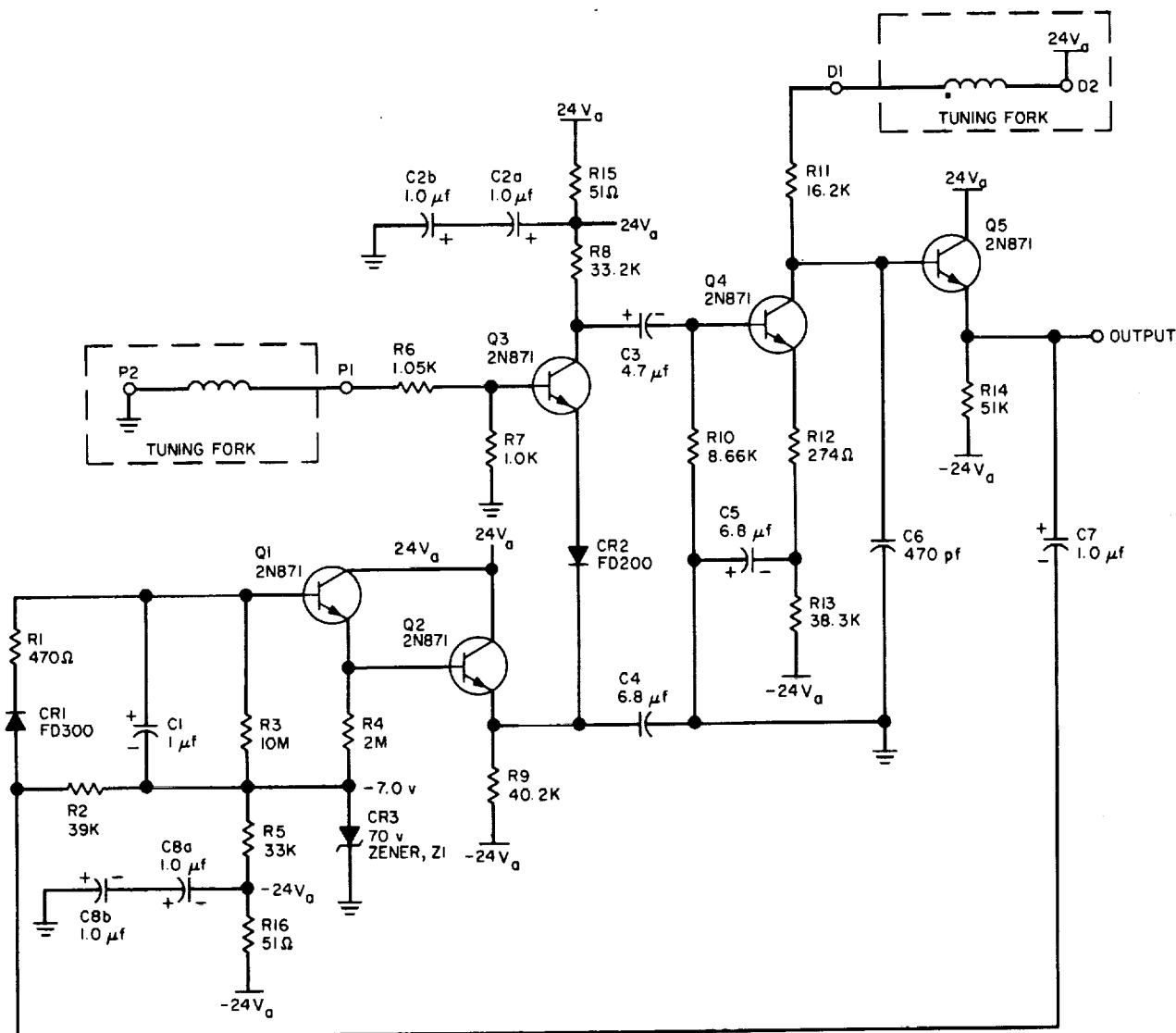


FIGURE 5-157. TUNING FORK OSCILLATOR

with a Q of about 5000. When the power is turned on, the loop gain at the fork frequency is greater than unit (about 3) and oscillations build up. The magnitude of the oscillations increases until the loop gain is reduced to exactly unity by the automatic gain control (AGC) circuitry. The oscillations build up to their steady state value within several seconds after the power is turned on.

The AGC depends on the fact that the voltage

gain of Q3 is approximately proportional to its emitter current, because the unbypassed emitter resistance, consisting of the small signal resistance of the emitter junction and of the diode CR2, is inversely proportional to the emitter current.

The output voltage is about 14 volts peak-to-peak and is essentially sinusoidal. The voltage at the anode of the zener diode CR3 is about -7 volts dc. The sinusoidal output voltage ap-

pears at the anode of CR1 referenced about -7 volts. When the power is first turned on, CR1 is not conducting. As oscillations build up to a sufficient level, CR1 conducts for a small portion of each period. This rectified current is filtered by C1 and amplified by Q1 and Q2. A small increase in the output voltage beyond the point necessary to make CR1 conduct causes a relatively large increase in the emitter current in Q2, a decrease in the emitter current in Q3, and a corresponding decrease in the loop gain. The output voltage stabilizes at the point where the loop gain is exactly unity.

Power requirements: +24 volts, -24 volts

Input requirements: None

Output specifications: Approximately 14 volts peak-to-peak; essentially sinusoidal; frequency $1553.446 \text{ cps} \pm 0.010 \text{ percent}$.

Core Input Amplifier. The core input amplifier (Figure 5-158) provides a square wave

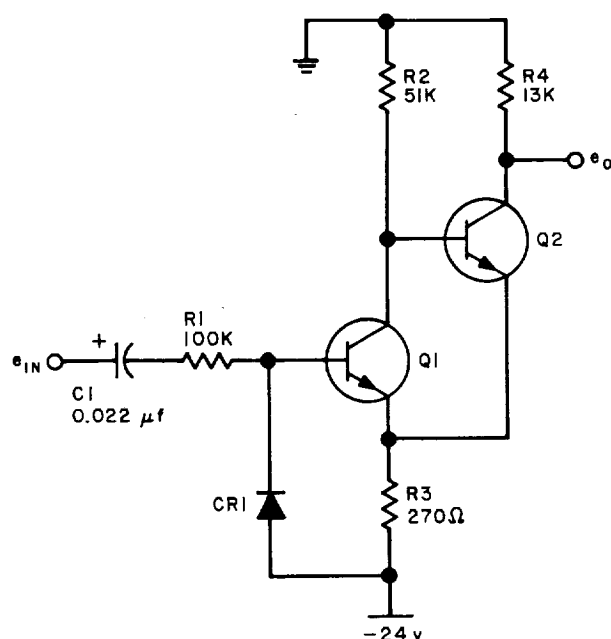


FIGURE 5-158. CORE INPUT AMPLIFIER

input to the core pulse shaper from the sine wave output of the tuning fork oscillator. A Schmitt trigger configuration of the amplifier is necessary to provide both a rapidly falling output edge and the required current gain.

The capacitively coupled input, C1, provides dc isolation between the tuning fork oscillator and the core input amplifier. Resistor R1 serves as the load of the tuning fork oscillator as well as the Q1 base-drive resistor. Diode CR1 allows the tuning fork oscillator to be symmetrically loaded for both positive and negative-going half cycles of the input.

Transistors Q1 and Q2 in conjunction with resistors R2 and R3 form the Schmitt trigger. Resistor R4 provides the output current to the core pulse shaper.

Input requirements:

Voltage	14 ± 5 volts peak-to-peak sine wave
Frequency	$1553 \text{ cps}, \pm 1 \text{ percent (for test)}$ $1553.446 \text{ cps}, \pm 0.01 \text{ percent (for timer operation)}$

Output specifications: Must be met for 0 to 50°C ambient.

Wave shape	Square wave
Asymmetry	$\frac{\text{on time}}{\text{off time}} \leq \frac{1.5}{1.0}$
Voltage levels —	
Low state	$-24 + 1.3, -0$ volts
High state	-10.5 ± 1.5 volts when loaded
Load	One core pulse shaper

Output voltage ≤ 0.5 microsecond
fall time (10 to 90 percent points)

Core Pulse Shaper. A core pulse shaper stage (Figure 5-159) is required to precede any string of scaler stages to ensure that the initial scaler stage receives properly shaped input pulses. It is in effect a count-of-one scaler stage.

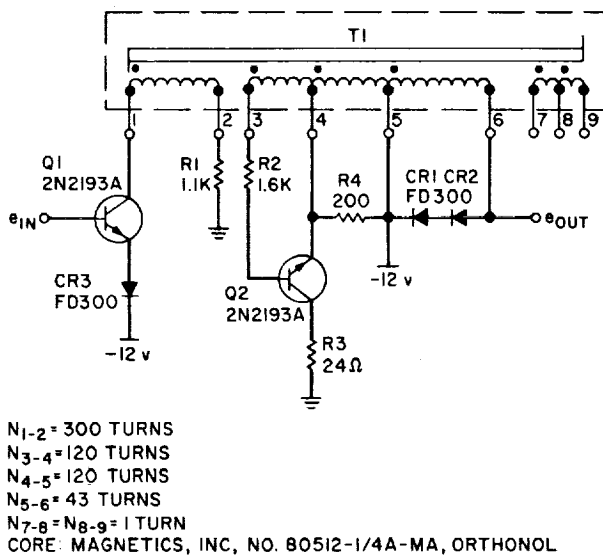


FIGURE 5-159. CORE PULSE SHAPER

The blocking oscillator of the shaper stage is essentially the same as that of the scaler stages. The main difference is the input configuration. In operation, a positive voltage is impressed at e_{in} which turns Q1 on and drives the core to saturation. When this input voltage is removed and Q1 turns off, the snap-back voltage generated across winding N_{3-4} triggers the blocking oscillator which generates a negative output pulse of fixed volt-second area to the first scaler stage (via winding N_{5-6}). Diodes CR1 and CR2 are required to limit the positive voltage spike which appears at pin 8 on completion of the blocking

oscillator cycle. Limiting is necessary to prevent this positive spike from coupling into the first scaler stage.

Input requirements: Requires output of the core input amplifier.

Waveform 1553 cps ± 1 percent square wave (-12 to -24 volts)

Input high time ≥ 150 microseconds

Input current ≥ 0.67 milliampere

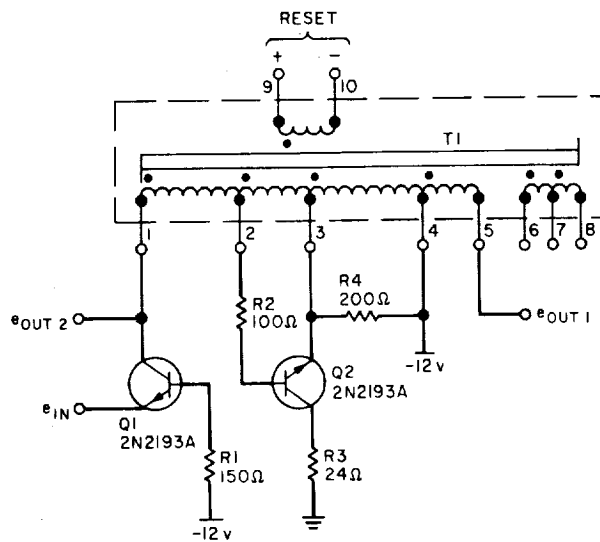
Input fall time ≤ 0.5 microsecond (voltage) (10 to 90 percent points)

Output specifications: Must be met for 0 to 50°C ambient.

Waveform Negative 3-volt pulse of approximately 13 microseconds duration from quiescent level of -12 volts

Volt-second area 40 volt-microseconds

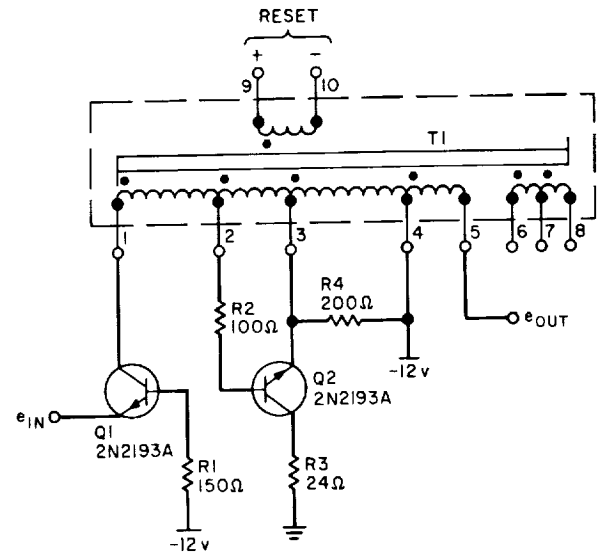
Core Frequency Scalers. Operation of the core frequency scalars is based on the very square ϕ -H curve of orthonol-type, tape-wound cores. Because of this characteristic, the ϕ -H curve of a core can be incrementally stepped up, generating an output pulse only after a sufficient number of input pulses have been applied to cause core saturation. At this time, the core resets itself to the opposite saturation state and simultaneously generates an output pulse. The frequency scaling factor for each stage is then defined as the number of input pulses required to saturate the core. For reliability considerations, a count-of-eight is the maximum scaling factor used for any one stage in the central timer.



$N_{1-2} = 162$ TURNS
 $N_{2-3} = 18$ TURNS
 $N_{3-4} = 120$ TURNS
 $N_{4-5} = 42$ TURNS
 $N_{6-7} = N_{7-8} = 1$ TURN
 $N_{9-10} = 50$ TURNS

CORE: MAGNETICS, INC., NO. 80512-1/4A-MA, ORTHONOL

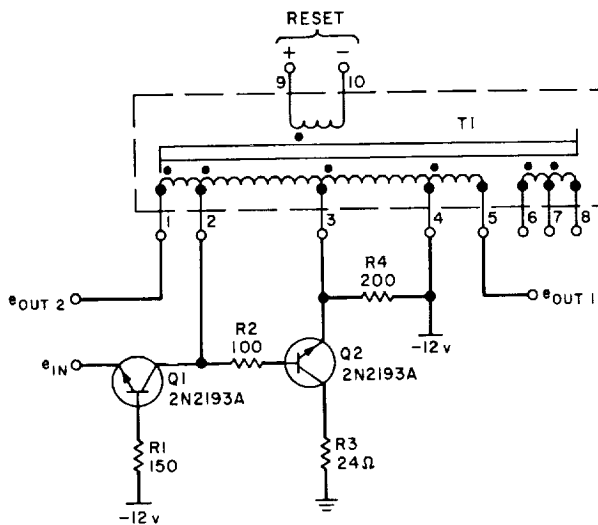
FIGURE 5-160. CORE FREQUENCY SCALER COUNT-OF-EIGHT STAGE



$N_{1-2} = 120$ TURNS
 $N_{2-3} = 18$ TURNS
 $N_{3-4} = 120$ TURNS
 $N_{4-5} = 42$ TURNS
 $N_{6-7} = N_{7-8} = 1$ TURN
 $N_{9-10} = 50$ TURNS

CORE: MAGNETICS, INC., NO. 80512-1/4A-MA, ORTHONOL

FIGURE 5-161. CORE FREQUENCY SCALER COUNT-OF-SEVEN STAGE



$N_{1-2} = 162$ TURNS
 $N_{2-3} = 18$ TURNS
 $N_{3-4} = 120$ TURNS
 $N_{4-5} = 42$ TURNS
 $N_{6-7} = N_{7-8} = 1$ TURN
 $N_{9-10} = 50$ TURNS

CORE: MAGNETICS, INC., NO. 80512-1/4A-MA, ORTHONOL

FIGURE 5-162. CORE FREQUENCY SCALER COUNT-OF-FOUR STAGE

The various stages used are shown in Figures 5-160, 5-161, and 5-162.

This discussion applies to all count stages, but specifically refers to the count-of-eight stage shown in Figure 5-160.

The input voltage, e_{in} , and output voltage $e_{out 1}$, quiesce at -12 volts with a pulse level of approximately -15 volts. The pulse width is about 13 microseconds and the area of the pulse is exactly 40 volt-microseconds. The input 40-volt-microsecond pulse is applied across windings N_{1-4} which increments the core up its ϕ -H curve. After the eighth input pulse the blocking oscillator fires, which resets the core to negative saturation and simultaneously generates an output 40-volt-microsecond pulse to drive the next count stage. That a volt-second area is proportional to an increment of flux, $\Delta\phi$, is apparent from the following equations.

From Faraday's Law,

$$e = N \frac{d\phi}{dt} 10^{-8}$$

where

ϕ = flux, maxwells

N = turns

t = time, seconds

e = generated or applied voltage, volts

Transposing and integrating:

$$\Delta\phi = \frac{10^{+8}}{N} \int_0^t e dt$$

volt-second area

Application of such an input pulse steps the core up its ϕ -H curve as shown in Figure 5-163.

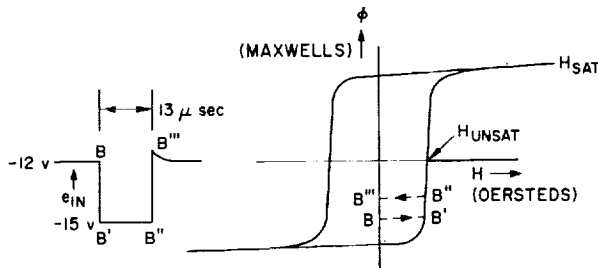


FIGURE 5-163. CORE FREQUENCY ϕ -H CURVE

As the input pulse terminates, a small snap-back voltage is generated across windings N_{1-4} of opposite polarity to the applied pulse. This is shown as the positive spike on the voltage waveform of Figure 5-163, and is due to the fact there is a small reversible flux when going from point b'' to b''' as the input pulse terminates. It is this snap-back voltage that detects full-count and triggers the blocking oscillator which resets the core and simultaneously generates an output pulse. The snap-back voltage must be small enough to ensure not-triggering for intermediate pulses and large enough on the last pulse to

ensure positive triggering of the blocking oscillator. The square hysteresis loop of orthonol provides this assurance by the large ratio of H_{sat} to H_{unsat} . For a toroid, magnetizing current, I_m , is proportional to magnetizing force, H , as shown:

$$H = \frac{0.4\pi NI_m}{l} = KI_m$$

where

H = magnetizing force, oersteds

N = turns

I_m = magnetizing current, amperes

l = core mean length, centimeters

Because the generated snap-back voltage results from the core saturation current flowing through R_4 , the ratio of the snap-back voltage at triggering to that at not-triggering can be made about as large as the ratio of H_{sat} to H_{unsat} .

Input transistor, Q_1 , is necessary to provide interstage isolation between the count stages. A low $V_{CE sat}$ for Q_1 is required to assure that it does not absorb a significant portion of the input volt-second area; this assurance is provided by the very low forced-beta at which Q_1 operates for intermediate pulses.

Transistor Q_2 , in conjunction with resistors R_2 , R_3 , and R_4 and windings N_{2-3} and N_{3-4} , forms the core reset blocking oscillator. R_2 provides the Q_2 base current, R_3 limits the Q_2 collector current, and R_4 provides the required damping at the termination of the blocking oscillator cycle. R_4 also appears as the load to input pulses and as such determines the level of the generated snap-back voltage.

Winding N_{1-2} is selected to provide the correct number of turns from pin 1 to pin 4 so that the proper $\Delta\phi$ is traversed for each input pulse.

Winding N_{4-5} is the output winding and is selected to give the proper output volt-second area.

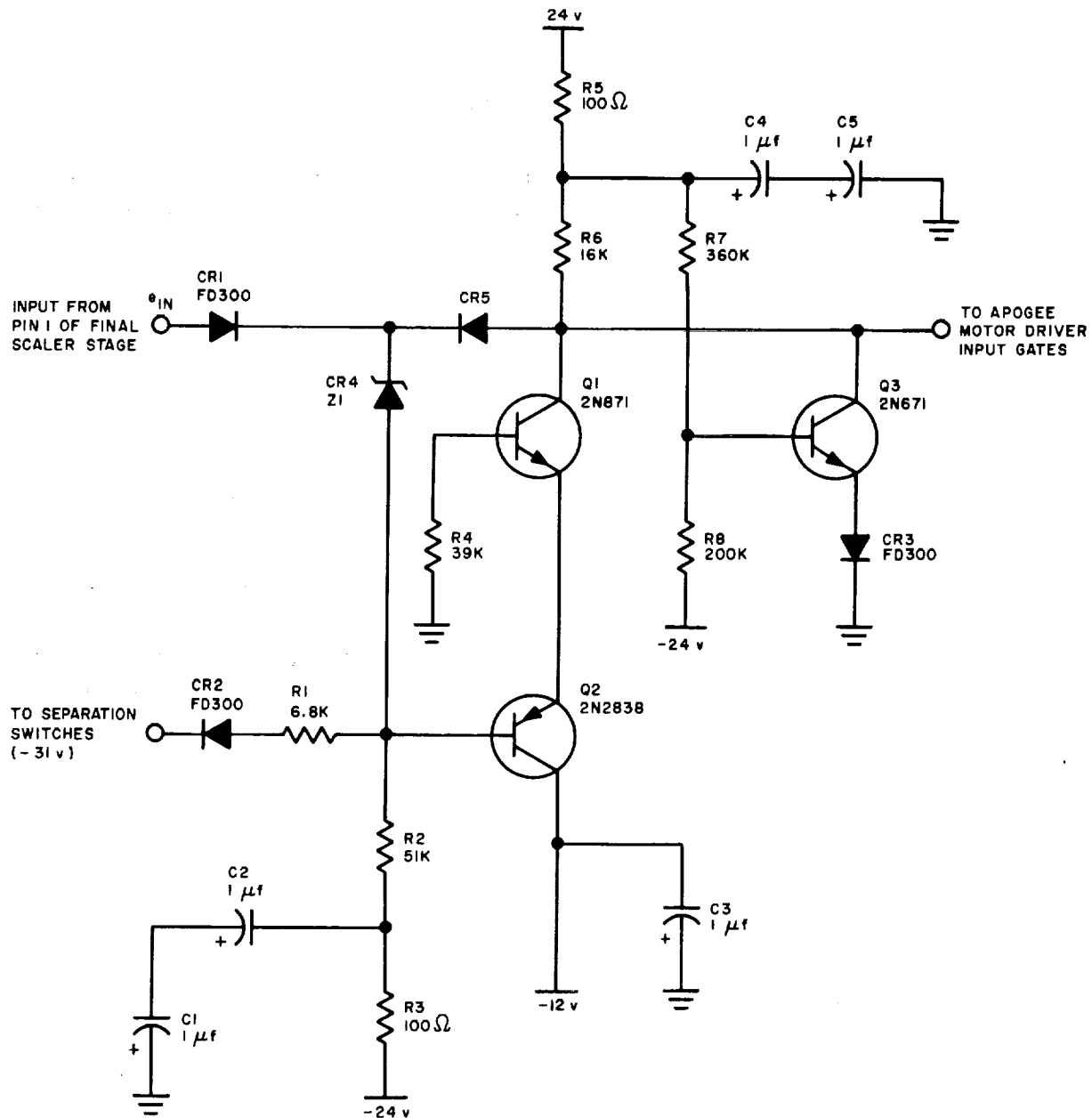


FIGURE 5-164. APOGEE TIMER OUTPUT LATCH

Windings N_{6-7} and N_{7-8} are one-turn windings which are used to fine-adjust the output pulse. By proper interconnection, they can effectively alter N_{4-5} by plus or minus two turns.

Reset winding, N_{9-10} , is necessary to provide positive reset before separation of the Advanced

Syncom vehicle from the booster second stage.

Apogee Timer Output Latch. The apogee timer output latch shown in Figures 5-164 and 5-165 is a bistable circuit utilizing complementary transistors in series to provide low power dissipation in one of its stable states (the high

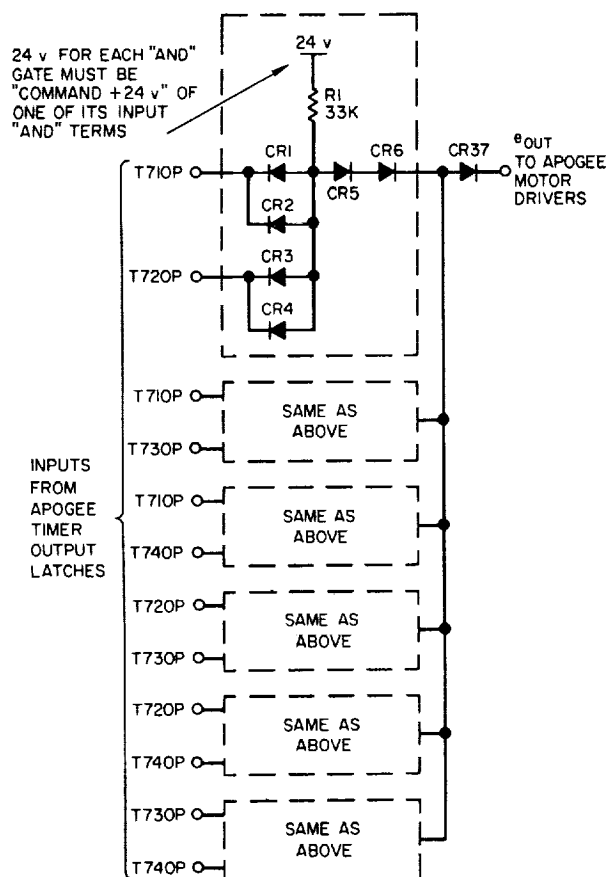


FIGURE 5-165. APOGEE MOTOR DRIVER INPUT GATE

state). In the low power state, both transistors are off, while in the high power state, both transistors are on.

The function of the apogee timer output latch is to provide a positive output voltage at a pre-determined time after spacecraft separation. This signal, in conjunction with the output of one other latch, provides the enabling signal to the apogee motor squib driver via the apogee motor driver input gate. The input to the latch is the output pulse, $e_{OUT 2}$, of the last count-of-four stage. The input signal is a 13-microsecond pulse of approximately 12 volts magnitude. The quiescent input voltage level is -12 volts.

Before separation, the latch is forced to the low position by the unregulated -31 volt supply.

After separation, -24 volts holds the latch in the low state until the input pulse appears. At that time the latch switches to the high (low power) state. Since the latch is in the low state for only a small fraction of the spacecraft lifetime, it is felt that this provides a highly efficient circuit.

In the low state, Q1 and Q2 base current drive is provided by I_{R4} flowing through Q1 and Q2 and into R2. In the high state, off bias is provided across the series connected base-emitter junctions of Q1 and Q2 by the R2, R6, CR4 divider voltage developed at the base of Q2.

$$V_{B2 \text{ off}} \geq +2.0 \text{ volts}$$

Transistor Q3, in conjunction with R7, R8, and CR3, is provided as a safeguard against the possibility of the latch going high should there be a power failure in which -24 volts is lost before +24 volts. If this does occur, Q3 turns on and clamps the output to approximately +1 volt (which is still a logical zero). In normal operation, the R7-R8 resistor divider keeps Q3 off. Diode CR3 is necessary to prevent emitter-base breakdown of Q3 during normal operation.

Input requirements: Requires output, $e_{OUT 2}$, of the last count-of-four stage.

Pulse voltage level	$\geq +8$ volts
Quiescent voltage level	-12 volts
Pulse duration	10 micro-seconds minimum and 16 micro-seconds maximum between 50 percent points

5. Spacecraft System Design

Output specifications: Must be met for 0 to 50°C ambient.

Low state

Output voltage

— 9.5 volts
minimum
— 13 volts
maximum

Current capability

3 milli-
amperes
maximum
not including
preload

Loads

Two apogee
motor driver
input gates

High state

Output voltage

+ 17.0 volts
maximum
+ 11.0 volts
minimum

Current

0

Capacitive loading

≤ 300 pf
total

Output lead length

≤ 10 feet
10 × 20 pf
= 200 pf

Number of connectors

≤ 4

between latch output and
input gate

4 × 20 pf
= 80 pf

Gating diodes

3 × 5 pf
= 15 pf
≤ 300 pf
total

Timer Output Buffer. The timer output buffer, shown in Figures 5-166 and 5-167, is a switching amplifier which generates a + 24 volt output pulse of approximately 15 microseconds duration for every input pulse from a count-of-eight scaler stage. Its purpose is three-fold:

- 1) Provides interface isolation (an output short will not affect timer operation)
- 2) Minimizes noise on interconnecting lines (intermediate scaler pulses are not seen)
- 3) Provides a very fast rise-time pulse, suitable for clocking flip-flops

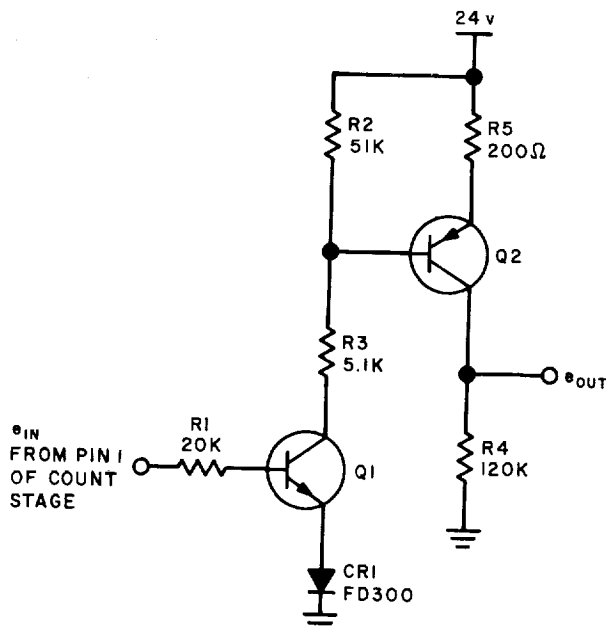


FIGURE 5-166. TIMER OUTPUT BUFFER

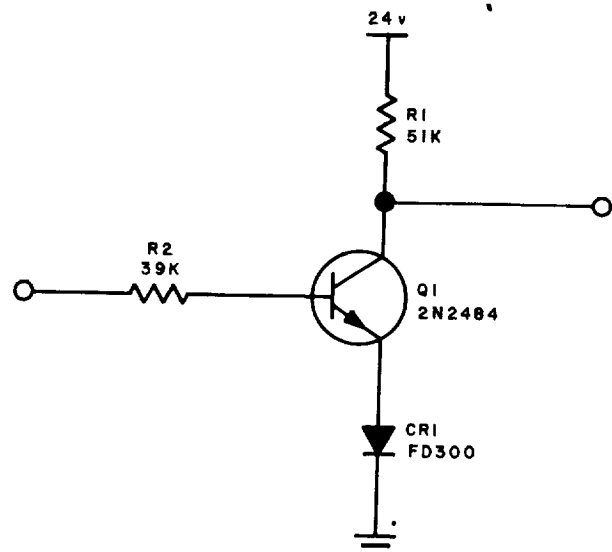


FIGURE 5-167. TIMER OUTPUT INVERTER

Transistors, Q1 and Q2, in conjunction with resistors, R1-R4, and diode CR1, form a two-stage switching amplifier. The npn-pnp configuration was chosen because of the very low power dissipation that could be attained for the quiescent state. Both stages are off in the quiescent state and both are on only for the duration of the input pulse. The power dissipation in both the 194-cps and 2.81-minute output applications is essentially zero. R5 provides current limiting in the event of an output short.

Input requirements: Requires output, e_{OUT} , of a count-of-eight stage.

Pulse voltage level	≥ 5 volts
Quiescent voltage level	-12 volts
Pulse duration	10 micro-seconds minimum and 16 micro-seconds maximum between 50 percent points

Output specifications: Must be met for 0 to 50°C ambient (Figure 5-168).

Voltages	
Pulsed	24 ± 0.7 volts
Quiescent	0.0 ± 0.1 , -0 volts

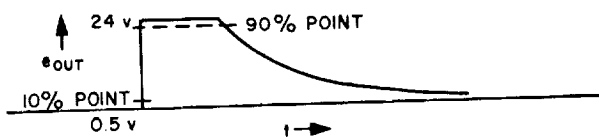


FIGURE 5-168. TIMER OUTPUT BUFFER OUTPUT SPECIFICATION

Rise time	≤ 0.5 micro-second (10 to 90 percent points)
Pulse width	$10 \leq \tau \leq 30$ micro-seconds (between 90 percent points)
Output current	up to 4 milli-amperes (pulsed dc)
Output capacitance	≤ 300 pf total

— 12 Volt Regulator. The -12 volt regulator shown in Figure 5-169 is designed as a current source. Zener diode CR1 establishes a 14-volt reference. R2 and R3 divides the zener voltage down to -12 volts. Q1 is an emitter follower with additional current gain provided by Q2. When a load is applied at the output, the emitter of Q1 tends to go positive with respect to its base. This causes Q1 to conduct harder which supplies more base drive current to Q2; due to the increased base drive, Q2 conducts and absorbs the load current through its collector. R5 and R7 protect the regulator against accidental shorts. R6 allows Q1 to conduct a minimum collector current under no-load conditions assuring that the beta of Q1 will not go below a certain minimum. Filter capacitor C1 prevents oscillations and provides additional transient load capability.

Input requirements: Voltage: -24 ± 0.7

Output specifications:

Output voltage:	-12 ± 1 volts
Output current:	0 to 25 milliamperes dc; 60 milliamperes for 130 microseconds; 0 duty cycle pulse

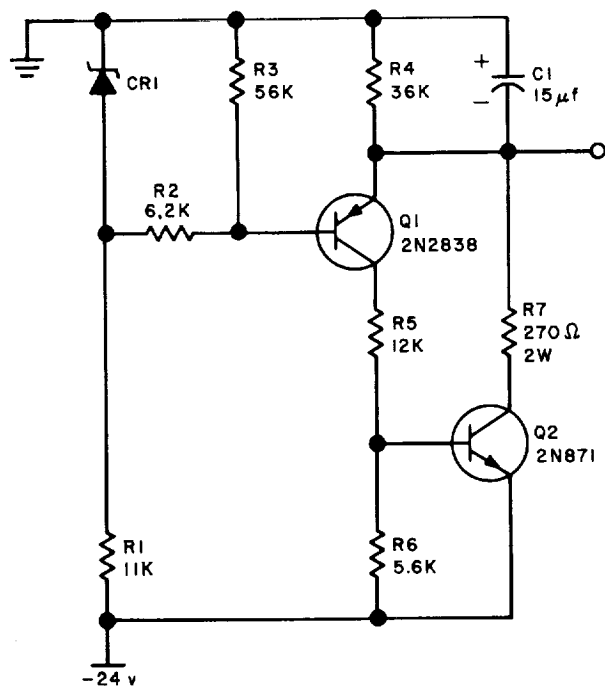


FIGURE 5-169. -12 VOLT REGULATOR

Squib Drivers. The squib drivers are used to ignite three types of squibs: apogee motor; bipropellant reaction control system pin puller, and bipropellant reaction control system valve squibs. For these functions a squib driver circuit was designed which can be driven by either the output of the central timer or the output of the command decoder. The output of each squib driver is capable of driving one or two bridges (two bridges per squib).

A functional diagram of the squib driver circuit is shown in Figure 5-170. When the input signal goes high the switch turns on. When the switched input signal has been present for more than 1 second, the output of the filter triggers the shaping circuit. The timing circuit is a resis-

tor-capacitor network which drives the current amplifier with a pulse of sufficient width to fire the squib. The current is amplified to a sufficient level to fire the squib.

A breadboard of the squib driver circuit has been tested over a temperature range of -25° to 75°C . There was no degradation of performance in these tests. An engineering model of the squib driver circuit has been built and it is planned to mate it with the engineering models of the command decoder and central timer and perform further tests.

Squib Driver Product Design. The conceptual product design of the squib driver unit has been accomplished; however, the detail design will not be completed during this phase of the program. The unit will contain twelve squib drivers in a sheet metal frame occupying a volume of approximately 100 cubic inches. The low power components of the driver circuit will be designed in welded modules with the high dissipation components mounted separately. Four drivers will be mounted to a circuit board which will then be mass —encapsulated. Three identical encapsulated circuit boards will then be enclosed in a thin sheet metal frame. Because of the short duty cycle of the driver circuit, the thermal capacity of the components provide enough heat handling capability so that little external heat sinking is required.

ELECTRICAL POWER

Power Supply System Design. The power system selected for Advanced Syncom uses radiation-resistant, moderately high efficiency n on p



FIGURE 5-170. FUNCTIONAL DIAGRAM OF SQUIB DRIVER CIRCUIT

TABLE 5-24. CHARACTERISTICS OF ELECTRICAL POWER SYSTEM

Solar cell	
Silicon n-p type	
Air mass zero sunlight	
Efficiency at 140 mw/cm ² and 77° F with 0.030 covers installed	9 percent
Cell active area	1.8 cm ²
Total number of cells	23,436
Total number of series strings	378
Battery	
Type	Nickel-cadmium
Number of batteries	Four
Number of cells per battery	22
Nominal cell rating	6 amp-hr
Maximum depth of discharge	20 percent
Average depth of discharge	12.7 percent

silicon solar cells as the primary energy source. Satellite operation during eclipse periods is maintained with electrical power from hermetically sealed nickel-cadmium storage batteries. The use of proved and tested components assures high confidence in satisfactory power system component life for 5 years in orbit.

The major characteristics of the electrical power system are summarized in Table 5-24. The power system, as shown in Figure 5-171, consists of the following:

- 1) A solar array composed of flat-mounted n on p solar cells on fiberglass-faced, aluminum honeycomb substrate panels. Sixteen panels are used to form the outer cylindrical surface of the spacecraft. Parallel interconnections at the cell level are employed to enhance solar array reliability. The physical wiring of the individual cell groups has been segregated by connection of alternate cell groups to provide maximum system reliability and flexibility.
- 2) Energy storage is provided for normal spacecraft operation while in complete eclipse. The storage system consists of four, 22-cell nickel-cadmium batteries, each connected to the spacecraft bus

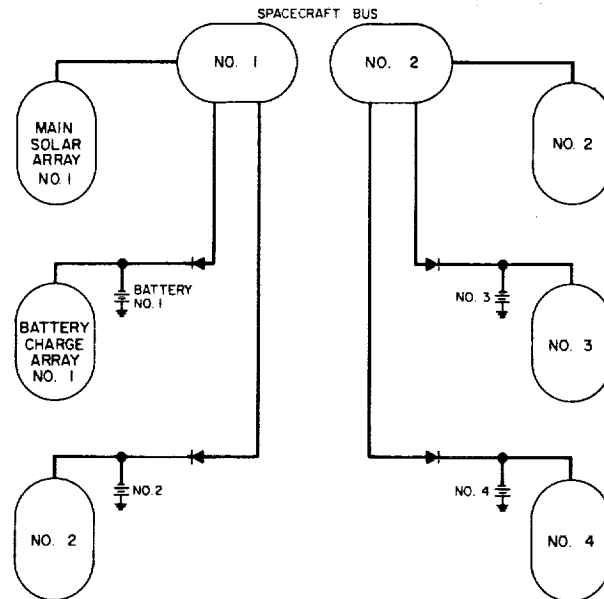


FIGURE 5-171. ELECTRICAL POWER SYSTEM SCHEMATIC

through a diode. These batteries, in addition to supplying spacecraft power during boost and eclipse periods, also provide energy for pulse loads such as control system valve solenoids and apogee motor igniter.

- 3) Battery charging energy is supplied to each battery directly from an isolated section of each solar panel. Alternate sections are connected to each battery to minimize the battery charging current ripple. Battery charging during ground operation will be accomplished by means of umbilical connections to ground charging controls.
- 4) Disconnects are provided to prevent inadvertent battery discharging by the command receivers prior to installation on the booster and external power connection.
- 5) Each major electronic unit has a separate voltage regulator or voltage regulator-switch combination. The nominal voltage at the equipment will be maintained at - 24 volts. Each regulator unit input cir-

5. Spacecraft System Design

cuit will be current limiting to protect the unregulated voltage supply bus from excessive current requirements.

- 6) Each traveling-wave tube operates from separate, regulated, high-voltage dc-dc converter and filament power supplies.

A summary of power demands for the spacecraft is given in Table 5-25.

To evaluate the 3- to 5-year operational capability of the Advanced Syncom, the following criteria and assumptions were considered:

- 1) Solar eclipse will occur twice annually for a period of 48.5 days each eclipse. The maximum time of eclipse for any 24-hour period will be 1.18 hours (Figure 5-172).
 - a) Eclipse season (umbra) $2\theta = 2K \text{ arc sin } [\sin S / \sin i]$
 - b) Eclipse season (umbra and penumbra) $2\theta = 2K \text{ arc sin } [\sin (S + \alpha) / \sin i]$
 - c) Maximum time in eclipse, $E = 2S/V$
 - d) Maximum time in penumbra, $E_p = 2K \text{ arc cos } [\cos (S + \alpha)] \times \cos S^{-1}$
- 2) Maximum battery depth of discharge will be limited to 20 percent of nominal battery capacity.
- 3) Leakage current of subsystem postregulators — ON mode, 3.5 milliamperes, OFF mode, negligible
- 4) Battery maximum charging rate and charging efficiency based on:

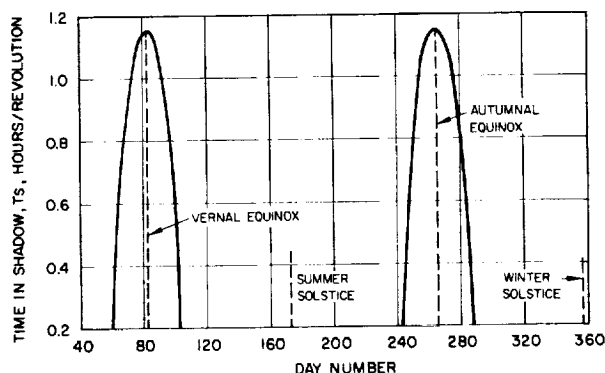


FIGURE 5-172 SHADOW TIME FOR SATELLITE IN 24-HOUR EQUATORIAL ORBIT

TABLE 5-25. POWER DISTRIBUTION PER BUS

Item	Number of Units per Quadrant	Milli-amperes per Unit	Maximum Number of Units Operating per Bus	Bus 1		Bus 2	
				Units Operating	Load, ma	Units Operating	Load, ma
Telemetry transmitter	1	245	2	1	245		
Encoder	1	27	2	1	27		
Command receiver	1	27	2	2	54	2	54
Digital electronics (PACE)	1	30	1	1	30		
Phase shifters and power amplifiers	One unit common to both buses	620	1, 2	1, 2	310	1, 2	310
Communication transponders	1	75	2	2	150	2	150
Traveling-wave tube (4-watt)	2	693	2	2	1386	2	1386
Total					2202*		1900*

*Typical totals. Either bus can carry up to 2512 milliamperes.

Four batteries necessary to supply complete spacecraft electrical load during maximum eclipse orbits.

Available battery charging time, 22.82 hours per orbit minimum.

Battery discharging time, 1.18 hours per orbit maximum.

- 5) Three ohm-cm n on p silicon solar cells will be used.
- 6) Bus voltage is greater than -25.5 volts at the input to the sub-system postregulators.
- 7) Solar panel temperature.
Normal incidence, $\beta = 0$ degree: $75 + 5/-10^{\circ}\text{F}$
Oblique incidence, $\beta = 25$ degrees: $60 + 5/-10^{\circ}\text{F}$
- 8) Battery temperature range: 30 to 120°F .

Solar Array. The solar array is composed of 23,436, 1 by 2 cm n-p silicon solar cells connected in series-parallel groups (Table 5-26). Each series-parallel group is connected to the spacecraft electrical bus with blocking diodes. Series-parallel solar cell interconnections have been used to increase the electrical power system reliability.

Within the physical constraints of the spacecraft envelope, it is possible to install the solar cells on each panel in a manner to provide the designed 27.2-volt output with a single row of solar cells extending from the top to the bottom of each panel. Only by developing the entire bus voltage in this manner is it possible to provide sufficient paralleling of cells to enhance system reliability.

Low resistivity n-p cells are approximately ten times more resistant to radiation damage than p-n type cells. Although high-resistivity n-p cells

have even greater radiation resistance than low-resistivity types, their output voltage is significantly lower, and their use would necessitate a 10 percent increase in spacecraft length to accomplish the desired series-paralleling connection of cells.

The solar cells are bonded to 16 lightweight, fiberglass-faced, aluminum honeycomb panels. Each panel is 22.2 by 26.25 inches and subtends an arc of 0.79 radian. Four of the 16 panels used have a slightly different cell layout to accommodate protrusion of the velocity control jet through the panel. The quantity of series-connected solar cells is determined by the individual cell characteristics and the output voltage requirements.

TABLE 5-26. SOLAR ARRAY CHARACTERISTICS

Item	Bus 1	Bus 2	Battery Charge Array (four per spacecraft)
Total number of cells per string	62	62	62
Total number of cells per group	186	186	186
Total group per bus	54	56	4
Total cells	10,044	10,416	744 ($\times 4 = 23,436$)
Characteristics	See Figure 5.7-3		See Figure 5.7-3
Weight of Solar Panels and Components			
Solar cells	0.257 gram		
Coverglass	0.300 gram		
Adhesive (panel)	0.020 gram		
Adhesive (coverglass)	0.005 gram		
Z-tab	0.014 gram		
Total	0.596 gram per solar cell		
Diodes (248)	7.44 grams per spacecraft		
Feedthroughs	33.6 grams per spacecraft		
Wiring harness	2 pounds at 32 ft/lb		
1 ohm/1000 feet			
Solar panel weight	23.7 pounds per spacecraft		

Solar Array Output. The power output of the solar panels is based on air mass zero, 77°F temperature, and solar intensity of 140 mw/cm². Figure 5-173 shows only normal sun incidence performance. Operation during the remaining portions of the year (non-eclipse orbit) where the sun incidence angle can increase to approximately 25 degrees is essentially the same as the array operation shown with the normal incidence angle, since the battery charge arrays then will furnish power to the unregulated bus on demand through the battery discharge diode.

Solar Array Reliability. The system reliability studies have included a detailed formulation of the expected solar array degradation due to random failures occurring during the orbital lifetime of the spacecraft. A technique for determining the distribution of failures in a multi-redundant solar cell configuration is described and applied to the Advanced Syncom solar array.

In general, a solar array may be described as consisting of solar cell strings arranged in series-parallel groups to provide the required current output and voltage level for the spacecraft equipments. In a multiredundant solar array the strings are interconnected to form alternate series-parallel current paths so that each individual cell failure would not cause the loss of the output of a complete string. Thus, the multi-redundant configuration may be defined as consisting of a number of solar cell groups each with m rows of cells and n cells per row with each cell interconnected in a series-parallel grid arrangement. Individual cells are assumed to exhibit the exponential failure distribution with failures independently and randomly distributed about the solar array.

The approach to the analysis will be to determine the expected failure distribution occurring

in an individual cell group and then to apply the result to the array.

The probability of survival (reliability of an individual cell) is defined as

$$P_{\text{cell}} = e^{-\lambda_c t} \quad (5-1)$$

where λ_c is the cell failure rate and t is the orbital mission time. The probability of failure of an individual cell (q_{cell}) is one minus the probability of survival, or

$$q_{\text{cell}} = 1 - P_{\text{cell}} = 1 - e^{-\lambda_c t} \quad (5-2)$$

The probability of failure of r cells within a row of the group may be expressed as the binomial probability distribution:

$$P(r) = \left(\frac{n}{r}\right) q^r p^{n-r} \quad (5-3)$$

The probability of s rows with exactly r failures is the binomial probability distribution

$$P(s) = \left(\frac{m}{s}\right) P(r)^s Q(r)^{m-s} \quad (5-4)$$

which may be approximated by the Poisson probability distribution

$$P(s) \approx \frac{[mp(r)]^s e^{-mP(r)}}{s!} \quad (5-5)$$

when $mp(r)$ is large relative to $P(r)$ and m is large relative to $mP(r)$. Note: $Q(r) \approx 1 - P(r)$.

The probability that at least one cell row will have exactly r failures is one minus the probability of having no cell row with r failures:

$$\begin{aligned} P(\text{at least one cell row with exactly } r \text{ failures}) \\ &= 1 - P(s = 0) \\ &= 1 - Q(r)^m \approx 1 - e^{-mP(r)} \end{aligned} \quad (5-6)$$

This becomes the desired expression for the probability that a multiredundant series-parallel cell group will have at least one cell row with exactly r failures.

Multiplying this expression by N for the number of multiredundant cell groups which com-

prise the solar array and varying r yields the expected number of cell groups $E(g)$ having at least one cell row with exactly r failures for any orbital time t and cell failure rate λ_c .

$$E(g) = [N 1 - Q(r)^m] \approx N [1 - e^{-mP(r)}] \quad (5-7)$$

The technique described may be applied to the multiredundant Advanced Syncom solar array configuration and used to determine the optimum number of solar cells required to provide sufficient power over the orbital lifetime of the spacecraft. Figure 5-174 presents the results of such an analysis for the Advanced Syncom solar array configuration consisting of 64 groups of cells each with three multiredundant series-parallel strings with three cells per row and 62 rows per

cell group. This figure shows the expected number of series-parallel string groups having r random failures as a function of orbital lifetime.

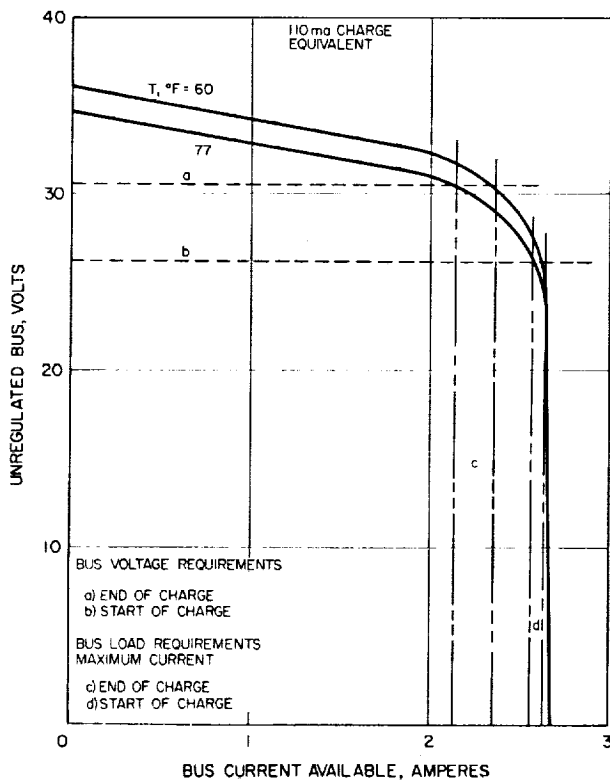
The derivation of the figure may be illustrated by the following calculations for an orbital time of 5 years and individual cell failure rate $\lambda_c = 0.01$ percent per thousand hours. The reliability of an individual cell as expressed by Equation 5-1 is:

$$P_{cell} = 0.995618$$

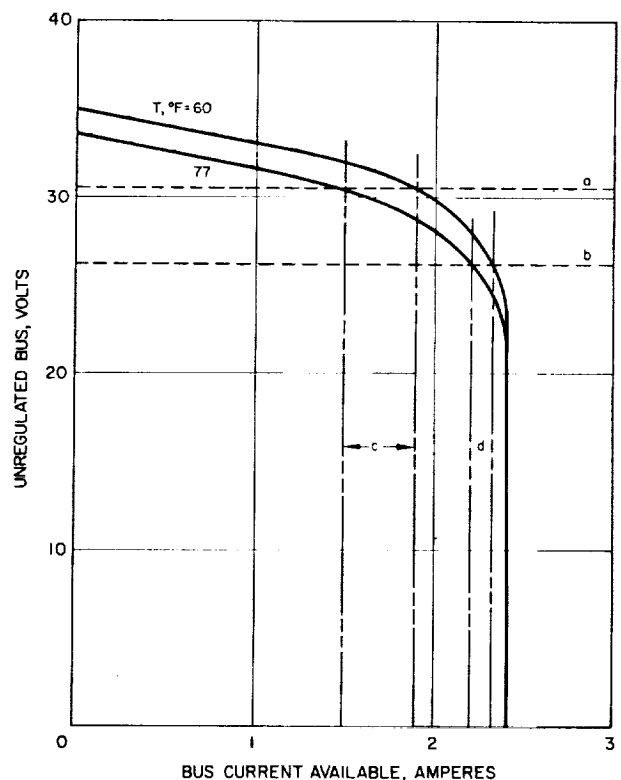
and the unreliability being 1 minus the reliability becomes

$$q_{cell} = 1 - 0.995618 = 0.004382.$$

The probability of failure of (r) cells, as expressed by Equation 5-3, becomes $(r = 0,1,2,3)$:



a) Initial Design



b) 1-Year Operation

FIGURE 5-173. BUS NO. 1 PERFORMANCE

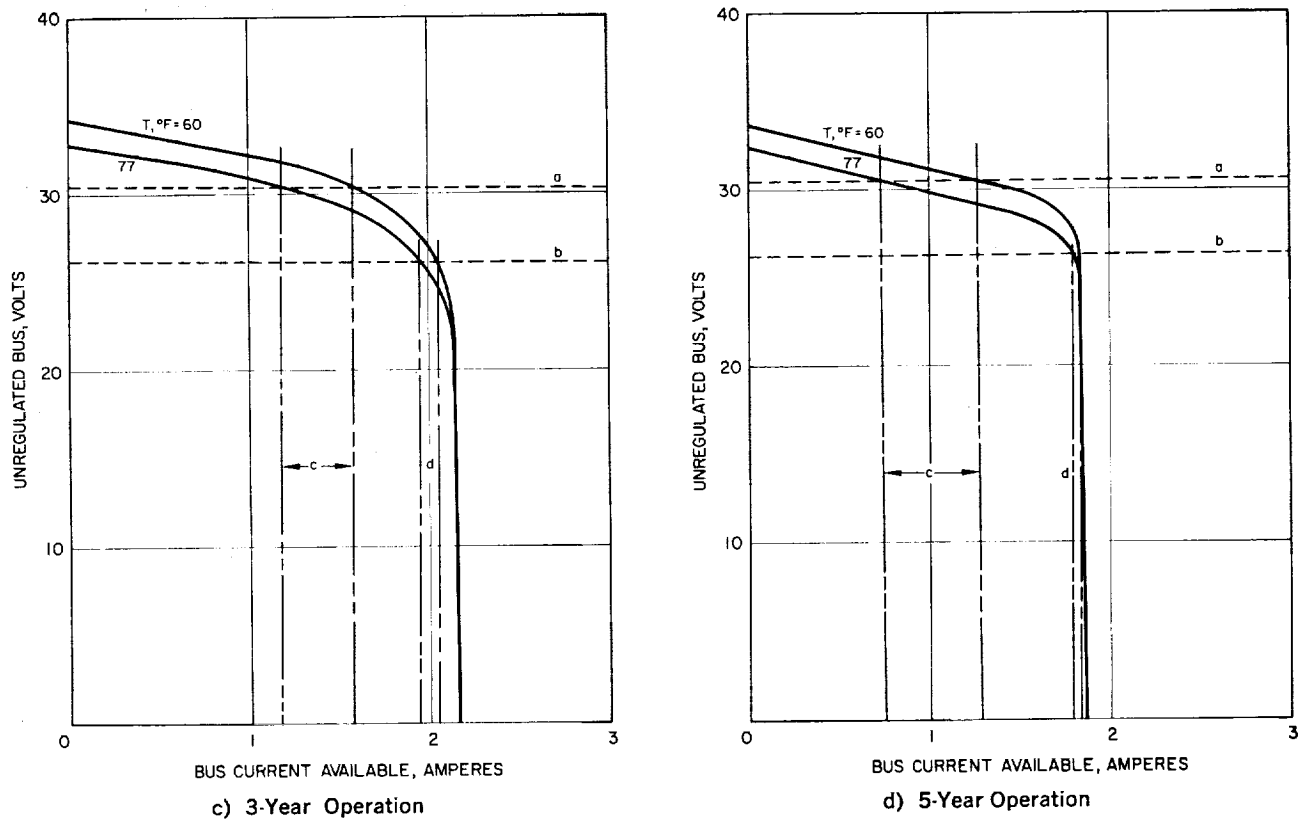


FIGURE 5-173 (Continued). BUS NO. 1 PERFORMANCE

$$P(0) = p^3 = 0.9869115 \text{ exactly zero failure}$$

$$P(1) = 3p^2q = 0.0130310 \text{ exactly one failure}$$

$$P(2) = 3pq^2 = 0.0000574 \text{ exactly two failures}$$

$$P(3) = q^3 = 0.0000001 \text{ exactly three failures}$$

The probability that at least one row will have exactly r failures may be written as Equation 5-6:

$$P(r=0) = 1 - [p(r+1) + p(r-3)] = 0.4422$$

$$P(r=1) = 1 - e^{-(62)(0.013031)} = 0.5542$$

$$P(r=2) = 1 - e^{-(62)(0.0000574)} = 0.0036$$

$$P(r=3) = 1 - e^{-(62)(0.0000001)} = 0.0000$$

The expected number of string groups having r failures from Equation 5-7 becomes ($N = 64$):

$$E(0) = (64)(0.4422) = 28.30 \approx 28$$

$$E(1) = (64)(0.5542) = 35.47 \approx 35$$

$$E(2) = (64)(0.0036) = 0.23 \approx 1$$

$$E(3) = (64)(0.0000) \approx 0$$

This analysis shows that random failures will be distributed so the probable occurrence of two or three failures within any row of a string group would be infrequent. Therefore, the solar array output will "degrade gracefully" toward approximately 0.67 maximum power, the loss of any one cell per row in any group being approximately equal to the loss of 0.33 power output of that group.

The 64 groups of cells used in this analysis have been selected as representative of one of two Advanced Syncom power supplies. The com-

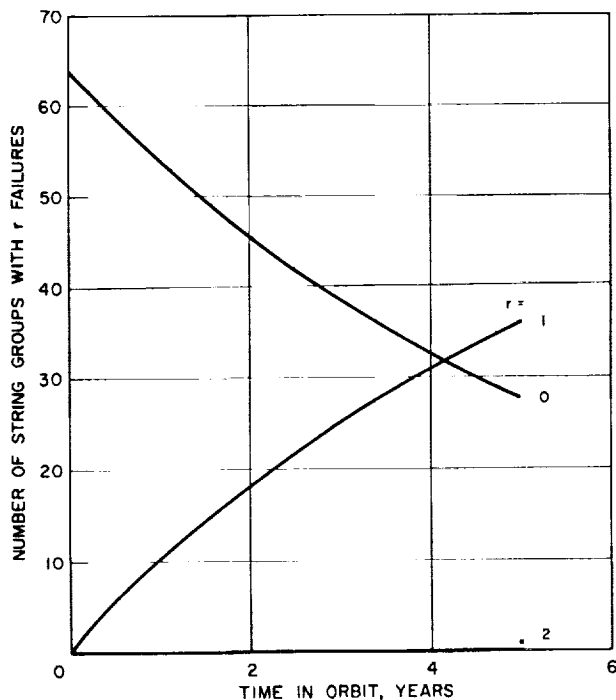


FIGURE 5-174. SOLAR ARRAY DEGRADATION DUE TO RANDOM CELL FAILURES

bination of the results presented for the occurrence of random failures with the effects on solar cell lifetime due to radiation would result in a characteristic plot of the expected solar array output during the orbital lifetime and the optimization of the number (N) of cell groups. This optimization would be based on the spacecraft power demand during the orbital mission and the expected survival of some spacecraft beyond the average lifetime or mean-time-to-failure.

In addition to the assumption of an independent and random distribution of solar cell failures, a second assumption in applying this model is that the primary failure mode of the solar cell is an open circuit which occurs at random with constant failure rate ($\lambda_c = 0.01 \times 10^{-5}$ failures per hour). The probability of an individual solar cell failing in a short circuit is neg-

ligible, and shorts to ground within the array are assumed not to occur.

In the Advanced Syncom configuration, each series-parallel string group will be connected to the bus through an isolating diode whose primary failure mode is a short circuit. Thus, the failure of any diode will only increase the reverse current leakage but will not cause the loss of power from a complete series-parallel string group. Therefore, these diodes are not included in the model. The expected diode failure rate is less than 10^{-7} failure per hour, which indicates that the probability of all diodes surviving the 5-year orbital mission time is extremely high and does not significantly affect the model presented.

Procurement Schedule

- Solar Cells. 2400 low efficiency cells have been ordered for the development panel. The panel with solar cells applied is now being assembled.

- Array Assembler. The assembler has been received and is being checked out.

- Components and Materials. Most of the components and materials listed are in-house. The honeycomb (0.125-inch hex by 0.5-inch thick curved) for the solar panels will be delivered 15 October 1963. Solar panel substrate fabrication is complete, and the solar cells have been received.

Fabrication Schedule. A panel using glass to simulate solar cells has been fabricated for vibration test.

A developmental panel with low efficiency solar cells applied is to be completed 31 October 1963. The cells will be assembled using the new solar array assembler.

5. Spacecraft System Design

Battery Power System. Battery cells of the nickel-cadmium construction are used for energy storage. Each battery has 22 series-connected cells (Table 5-27).

Battery design depends on the bus voltage needed, eclipse power load, depth of discharge, solar array recharging rate, time available for recharge, and reliability requirements. Twenty-two nickel-cadmium cells are used as a minimum number of cells to provide 25.5 volts to the electronic systems at the maximum depth of discharge of 20 percent. A battery specification has been written for the required design.

The current required from any one battery will be a maximum of 1.03 amperes. The cell utilizes flat-plate construction with both the input and output terminals electrically insulated from the cell case. A group of 12 cells interconnected is shown in Figure 5-175.



FIGURE 5-175. GROUP OF 12 CELLS

The battery array charging capability is based on the following conditions:

- 1) Battery cell size, 6 amp-hr, nickel-cadmium
- 2) Twenty-two series-connected cells per battery

TABLE 5-27. DESIGN SPECIFICATIONS FOR NICKEL-CADMIUM BATTERY CELL

Type	Sealed sintered plate
Weight	Less than 0.62 pound
Terminals	Ceramic bushing suitable for 5-year usage
Capacity	6.0 amp-hr at 75° F at 1.2 amperes 4.8 amp-hr at 100° F at 1.2 amperes 4.8 amp-hr at 30° F at 1.2 amperes
Maximum charge rate (with cutoff control)	5 amperes
Overcharge capability	0.6 ampere continuous
Maximum charge voltage	1.48 volts
Cycle life	10,000 minimum at 25 percent depth
Environment	30 to 120° F sustained operation Space vacuum Operation in any orientation Vibration, shock, and acceleration typical of spacecraft launchings

- 3) Depth of discharge during maximum eclipse, 20 percent
- 4) Battery voltage (20 percent depth of discharge) 26.4 volts
- 5) Minimum voltage to spacecraft regulators: 25.5 volts for proper regulation
- 6) Minimum charge current to batteries: 100 milliamperes to replace 100 percent of the energy taken out during a maximum eclipse (50 percent efficiency)
- 7) On-charge voltage at 100 milliamperes charge rate
- 8) The battery array must be capable of supplying 100 milliamperes at a voltage greater than the on-charge voltage requirements
- 9) Two batteries per bus required for full system operation

Battery charging is accomplished by using a separate portion of the solar cell array. Power will be supplied from the battery for 485 eclipses

periods in a 5-year span. Current requirements during each of these days will vary as a function of the time in the eclipse.

Three basic types of batteries are available for space application; however, only one type has the recharging cycle life necessary for the requirement of 465 days in a 5-year period. Some of the characteristics of the three basic types are as follows:

Cell	Recharge Efficiency	W-hr/lb.	Cycle Life
Nickel-cadmium	0.65	10	9500
Silver-cadmium	0.70	13	100 to 1500
Silver-zinc	0.80	50	200

Battery Reliability. Recently reported results of the most extensive life cycling tests ever conducted on nickel-cadmium cells indicates that: 1) cell cycle life with shallow discharges is considerably longer than cycle life at deep discharges, and 2) cycle life is reduced by high and low ambient temperatures. The data show that for long life, depths of discharge should be kept below 25 percent and battery temperatures should be maintained near 75° F. Advanced Syncom batteries will be maintained near this temperature by proper thermal design. At 20 percent depth-of-discharge, the 500 charge-discharge cycles will be achieved with a high degree of reliability.

Cycling tests have been conducted on Syncom 2 batteries and General Electric 12 amp-hr batteries. These tests simulated the charge-discharge cycle of typical orbital conditions. Figures 5-176 and 5-177 show the results of the tests on the Syncom 2 batteries. Figures 5-178 and 5-179 show similar data on the General Electric cell.

The end of discharge voltage on these cells should remain above 1.2 volts per cell at the end of 500 cycles.

Solar Cell Assembler. A solar cell assembler has been built by Hughes (Figure 5-180). The machine automatically positions solar cells on an assembly table and by induction soldering connects a die-formed "zee" tab to the cells. After the solder connections are completed, the solar cell group is advanced one position and the operation repeated. This process continues until a three-parallel, 62-series group of solar cells has been completed. In addition, a solar panel jig has been designed to accept one Advanced Syncom solar panel. Groups of solar cells are positioned on the panel during the solar cell-to-panel bonding operation.

Electrical Power System Analog. The array performance characteristics, as shown in the July progress report, were mathematically constructed and therefore may contain minor errors in the slope of the I-V curve. In an effort to determine exact system voltage margins (for battery charging), a system analog was designed and fabricated. The analog block diagram is shown in Figure 5-181. The heart of the analog is an amplifier designed to synthesize the output of an

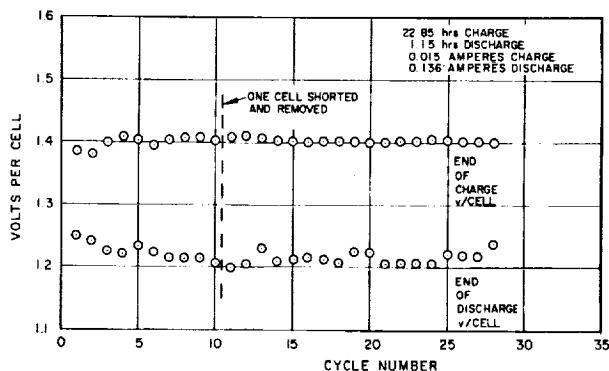


FIGURE 5-176. SYNCOM 2 BATTERY TEST

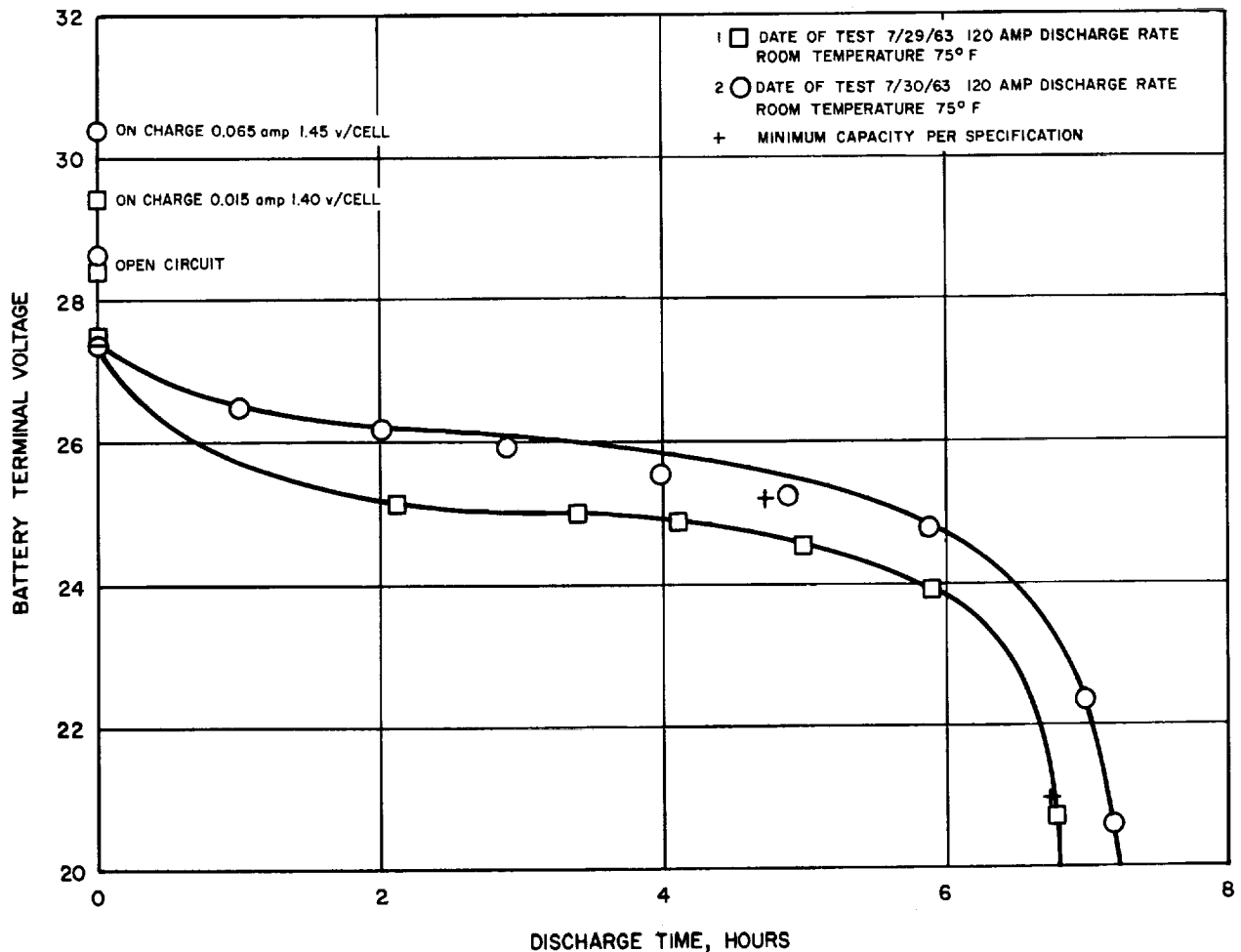


FIGURE 5-177. NICKEL-CADMIUM BATTERY DISCHARGE CURVE

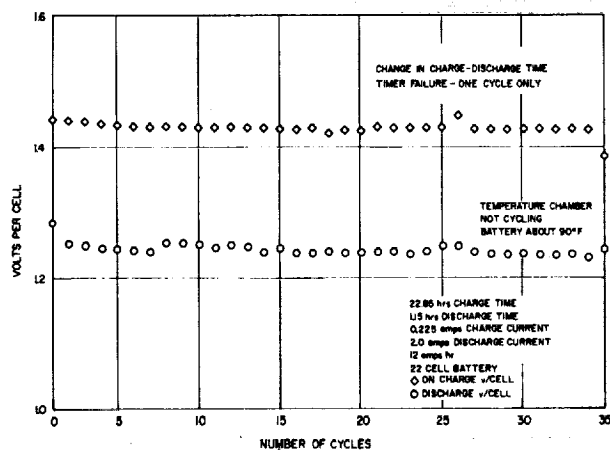
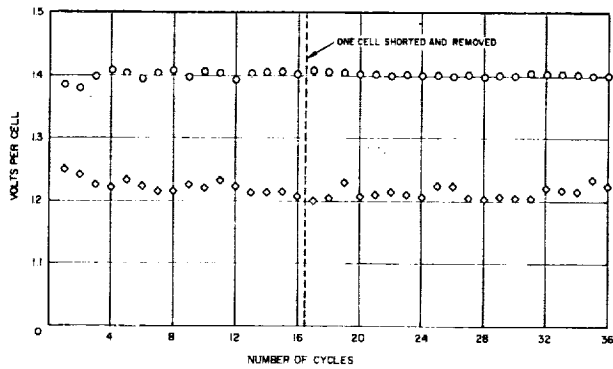


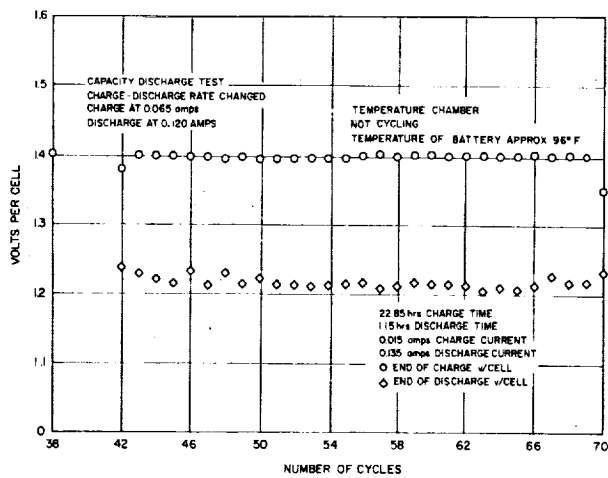
FIGURE 5-178. GENERAL ELECTRIC BATTERIES TEST

illuminated solar cell array of any number of series and parallel connection to yield a desired voltage and current. A portion (dependent on the gain setting) of the amplifier output current is fed back to load the individual input solar cell. In this manner, the amplifier output characteristics can be made to exactly duplicate a large solar array. The amplifier is shown in Figure 5-182.

Two such amplifiers have been constructed together with loads duplicating system loads such as command receivers, PACE, traveling-wave



a) Cells 1 to 36



b) Cells 38 to 70

FIGURE 5-179. SYNCOM BATTERY TEST

tube, etc. The system is now assembled and is in final electrical checkout. The amplifier output representing the battery charging array can be modulated to simulate the actual output variation with spacecraft rotation of the spacecraft array. (The battery charge array segments are located 90 degrees apart; therefore, the current available varies at four times the spacecraft spin rate.)

The system analog will make possible rapid determination of system performance levels under any combination of the following variables:

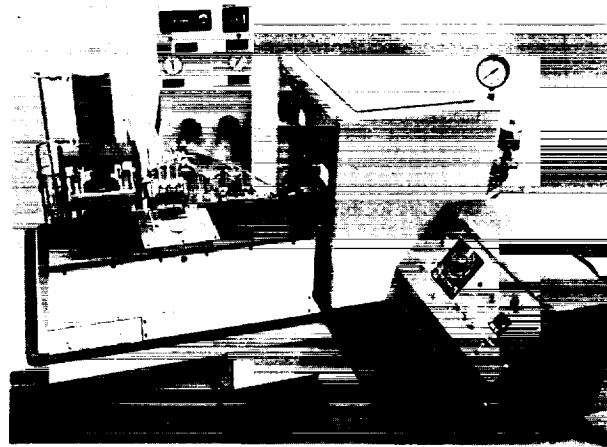


FIGURE 5-180. SOLAR CELL ASSEMBLY MACHINE

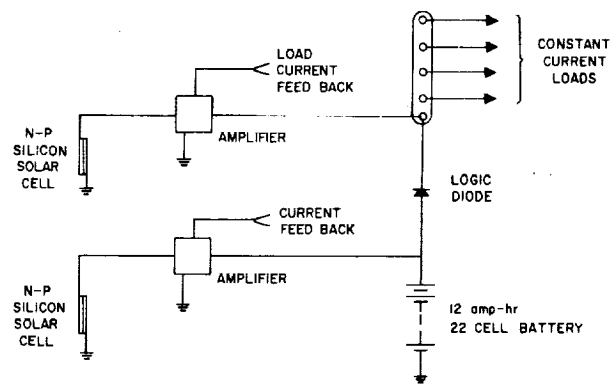


FIGURE 5-181. ELECTRICAL POWER SYSTEMS ANALOG BLOCK DIAGRAM

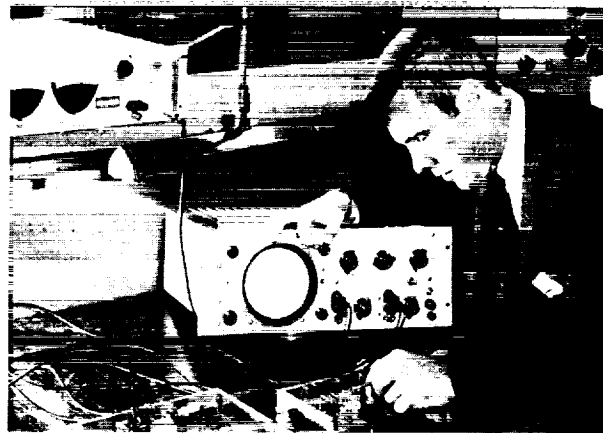


FIGURE 5-182. AMPLIFIER FOR SIMULATING SOLAR SENSOR ARRAYS

5. Spacecraft System Design

- 1) Number of series solar cells
- 2) Solar array temperature
- 3) Solar array performance degradation (random cell failures, particulate radiation, etc.)
- 4) Electronic load change

Particulate Radiation Damage. Radiation damage theory (Reference 5-1) shows that high-energy charged particles displace atoms in semiconductor crystals. These displacements then act as recombination centers for electron-hole pairs. In a photovoltaic cell, light creates holes and electrons in the cell material which then diffuse to the junction. The contact potential at the junction accelerates the holes and electrons to produce a current. If holes and electrons recombine at the recombination centers, they are not available to produce useful current. A characteristic of semiconductor material is its lifetime, the time it takes for the number of excess carriers (e.g., holes and electrons created by the light) to decay to $1/e$ of its maximum value by recombination. The lifetime of a material is clearly shortened by introducing recombination centers (Reference 5-2). In addition, the longer the lifetime, the more current produced. These principles are detailed below for protons.

The short circuit current, I_{sc} , is related to the lifetime, τ , as follows:

$$I_{sc} = K_1 G e \sqrt{D \tau} = K_1 G e L \quad 5-8$$

K_1 = constant for silicon

G = generation rate of carriers

e = electronic charge

D = diffusion constant for silicon

L = diffusion length for silicon

The number of displacements, N_d , produced by a flux of ϕ particles/cm² with energy, E , is

$$N_d = K_2 \phi f(E) \quad 5-9$$

K_2 = constant for silicon

Although there are analytical expressions for $f(E)$ for protons and electrons, it will be more useful to use empirical results in this report.

The lifetime is related to the number of displacements as:

$$\frac{1}{\tau} = \frac{1}{\tau_0} + K_3 N_d \quad 5-10$$

τ_0 = lifetime of unirradiated material

K_3 = constant for silicon

Therefore,

$$I_{sc} = \frac{K_1 G e \sqrt{D}}{\sqrt{\frac{1}{\tau_0} + K_3 N_d}} \quad 5-11$$

and

$$I_{sc} = K_1 G e \sqrt{D \tau_0} \left[1 + K_2 K_3 \tau_0 \phi f(E) \right]^{-1/2} \quad (5-12)$$

Equation 5-12 shows that the short circuit current is reduced as the particle flux increases.

Equation 5-12 may be written in another form.

Let

$$I_{sc0} = K_1 G e \sqrt{D \tau_0}$$

Then:

$$\left(\frac{I_{sc0}}{I_{sc}} \right)^2 - 1 = K_3 \tau_0 [K_2 \phi f(E)] = K_3 \tau_0 N_d \quad (5-13)$$

If the flux required to produce 25 percent degradation is considered, then:

$$\phi_{25} = \frac{7/9}{K_2 K_3 \tau_0} \frac{1}{f(E)} = K_{25} F(E) \quad (5-14)$$

Values of ϕ_{25} have been found for different energies of protons and electrons (References 5-3

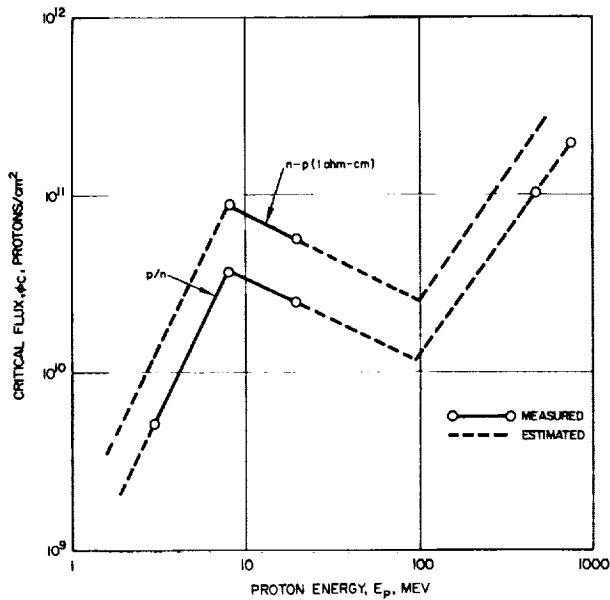


FIGURE 5-183. CRITICAL PROTON FLUX

through 5-14). These values are summarized in Figures 5-183 and 5-184. Although most of the experiments have been on silicon p-n cells, some have been on n-p cells. Figures 5-183 and 5-184 show them to be three times more radiation resistant than their p-n counterparts. The difference is due to the fact that most of the damage is done in the bulk of the cell, not the diffused layer; this is n-type silicon in p-n cells and p-type in n-p cells. Since there are different impurities in the two types, it is expected that different recombination centers would be created, which would not have equal effects on the carriers.

Both theory and experiment show that particles with energies below a threshold energy will cause no displacements. For protons, this threshold energy will cause no displacements. For protons, this threshold is about 100 ev, but for electrons, the threshold is 145 kev for n-type silicon and about 200 kev for p-type silicon. Therefore, there is no damage to p-n cells by electrons with energies below 145 kev, and none

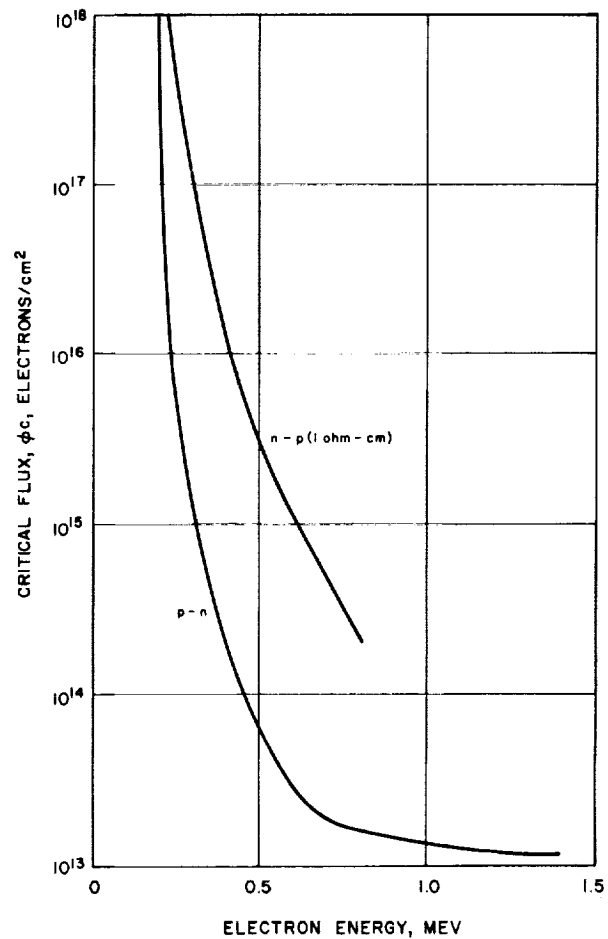


FIGURE 5-184. CRITICAL ELECTRON FLUX

to n-p cells by electrons with energies below 200 kev (References 5-9 and 5-11).

Solar Cell Degradation from Non-Monoenergetic Fluxes in Space. When a particle flux has a distribution of energies, Equation 5-9 must be modified.

$$N_d = \int_a^\beta K_2 f(E) \frac{d\phi}{dE} dE \quad (5-9a)$$

α, β are energy limits of flux and

$$\left[\left(\frac{I_{sc0}}{I_{sc}} \right)^2 - 1 \right] = C \int_a^\beta \frac{1}{F(E)} \frac{d\phi}{dE} dE$$

C = constant.

For convenience, let

$$\left[\left(\frac{I_{so}}{I_s} \right)^2 - 1 \right] = \Delta$$

From Equation 5-13,

$$\Delta_{25}(E) = C \frac{1}{F(E)} \phi_{25}(E)$$

Since

$$\Delta_{25}(E) = \Delta_{25} = \left(\frac{7}{9} \right)$$

then

$$\frac{\Delta}{\Delta_{25}} = \int_a^\beta \frac{d\phi}{dE} \frac{dE}{\phi_{25}(E)} \quad (5-15)$$

Equation 5-15 can be evaluated for trapped and solar flare particles since $\phi_{25}(E)$ can be empirically derived from Figures 5-183 and 5-184.

A distribution spectrum is assumed of the form:

$$\phi(>E) = A E^{-n} \quad \alpha \leq E \leq \beta$$

Then

$$d\phi/dE = -n A E^{-n-1}$$

and the total flux is

$$\phi_T = A [\alpha^{-n} - \beta^{-n}]$$

Therefore, the following can be written:

$$d\phi/dE = -n \phi_T \frac{E^{-n-1}}{\alpha^{-n} - \beta^{-n}}$$

Hence,

$$\frac{\Delta}{\Delta_{25}} = \frac{-n \phi_T}{\alpha^{-n} - \beta^{-n}} \int \frac{E^{-n-1}}{\phi_{25}(E)} dE \quad (5-16)$$

Equation 5-16 may be written:

$$\frac{\Delta}{\Delta_{25}} = \frac{\phi_T}{\langle \phi_{25} \rangle} \quad (5-16a)$$

where

$$\langle \phi_{25} \rangle = \frac{\alpha^{-n} - \beta^{-n}}{-n \int_a^\beta \frac{E^{-n-1}}{\phi_{25}(E)} dE}$$

to arrive at the formula for $Q' = 1 - I_{sc}/I_{sc0}$, the fractional degradation of short circuit due to the flux ϕ as seen in solar light.

$$Q' = 1 - \left[1 + \frac{7}{9} \left(\frac{\phi}{\langle \phi_{25} \rangle} \right) \right]^{-1/2} \quad (5-17)$$

The formula for Q' in Equation 5-17 is for short circuit current degradation under sunlight. The fractional degradation of efficiency cannot be determined theoretically since the variation of open circuit voltage with flux cannot be simply determined. It is necessary to empirically derive the efficiency degradation from laboratory experiments (References 5-3, 5-5, and 5-15). It is known from experiments that the degradation in power is related to the short circuit current degradation. Initially, using Equation 5-17, it was determined that

$$Q = 1 - \left[\frac{7}{9} \left(\frac{\phi}{\langle \phi_{25} \rangle} \right)^{1/2} + 1 \right]^{-1/2} \quad (5-18)$$

where Q , the fractional degradation in efficiency, fits all the data in References 5-3, 5-5, and 5-15.

Figures 5-185 and 5-186 were derived from Equation 5-18 using various values of $\langle \phi_{25} \rangle$ for protons and electrons.

Experimental and Theoretical Comparison. W. R. Cherry (Reference 5-18), using 1-mev electrons and 4.6-mev protons, performed radiation damage studies on solar cells. There is a definite correlation between these experimental results and results predicted in Figure 5-185 and 5-186. Table 5-28 shows the correlation between experiment and theory. The ϕ_c used was

TABLE 5-28. EXPERIMENT AND THEORY CORRELATION

Figure from Reference 5-18	Flux, Total	Percent Degradation			
		Experimental		Theoretical	
		Protons	Electrons	Protons	Electrons
3	2×10^{11}	33		31	
3	1×10^{11}		54		53
4	2×10^{11}	40		38	
4	1×10^{11}		58		57
5	3×10^{11}	42		40	
5	1×10^{11}		34		36
6	1×10^{10}	8		9	
6	1×10^{11}		50		49
7	1×10^{11}		43		46

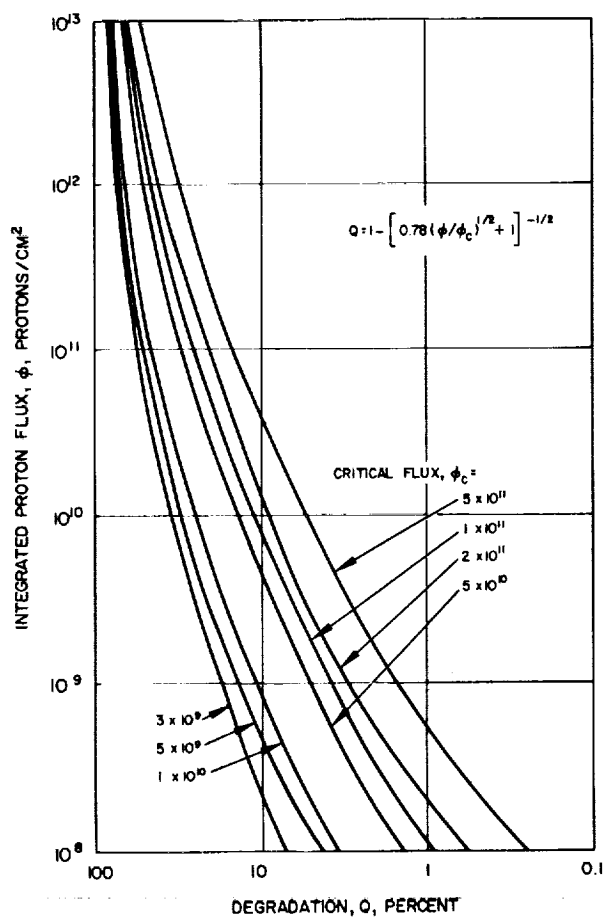


FIGURE 5-185. PROTON DEGRADATION EFFECTS

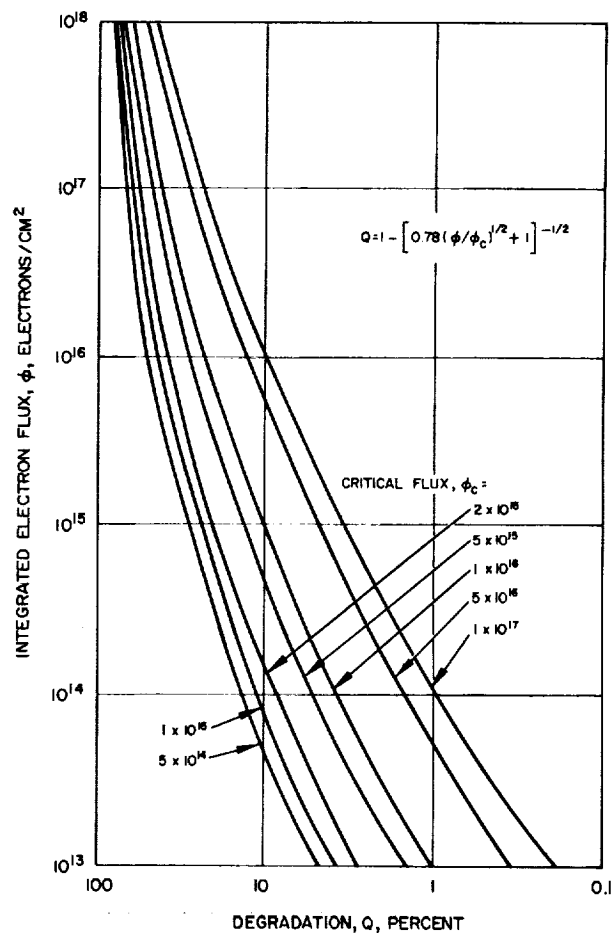


FIGURE 5-186. ELECTRON DEGRADATION EFFECTS

5. Spacecraft System Design

taken from the curves of Reference 5-18 and used as a basis for determining degradation from the theoretical curves.

Syncom Radiation Damage Study. The analysis of the space environment radiation damage to Advanced Syncom solar cells was performed under the following conditions:

- 1) Synchronous satellite orbit, altitude $\sim$ 22,000 miles
- 2) Satellite life, 6 years
- 3) Three n-p solar cells, 3 ohm-cm resistivity
- 4) Quartz shield parameter study, 6-, 30- and 45-mil

For a radiation damage analysis of solar cells, critical fluxes for both current and power degradations for the various types of radiations should be known. The lack of available experimental data on 3 ohm-cm solar cells necessitated the use of the following approximations:

- 1) Critical integrated flux for protons = $\phi_c^p = 2 \times 10^{11}$ protons/cm²
- 2) Critical integrated flux for electrons = $\phi_c^e = 2 \times 10^{15}$ electrons/cm²

Degradations Due to Solar Flares. Solar flare data obtained from Reference 5-1 and representing an average flux of protons for the period 1947 to 1953 is as follows:

Proton Energy, mev	Intensity, Protons/cm ² per year
5	4×10^{10}
10	1×10^{10}
20	2×10^9
30	1×10^9
100	1×10^8

The number of flares considered was not given, and relativistic flares were ignored.

- 1) A 6-mil quartz shield stops $E < 3$ -mev protons (Table 5-29)
 $\phi_{sp}(E > 3 \text{ mev}) \approx 6 \times 10^{10}$ protons/cm² per year
 $(\phi = \sim 1 \times 10^{10} + 4 \times 10^{10} + 1 \times 10^{10} + \dots \approx 6 \times 10^{10}, \text{ using solar flare data})$
- 2) A 30-mil quartz shield stops $E < 15$ -mev protons (Table 5-30)
 $\phi_{sp}(E > 15\text{-mev}) \approx 1 \times 10^{10}$ protons/cm² per year
 $(\phi = \sim 7 \times 10^9 + 2 \times 10^9 + 10^9 + \dots \approx 1 \times 10^{10})$

TABLE 5-29. SOLAR FLARE DEGRADATION OF 6-MIL SHIELD

Time, years	$\phi_{sp} \times 10^{11}$	Q, per cent	Q', per cent	Q'', per cent
1	0.6	19	9.9	6
2	1.2	24	17.4	8
3	1.8	27	23	10
4	2.4	29	28	11
5	3.0	30	30*	11.5
6	3.6	31	31*	11.8

*Formula used is not good for $\phi \rightarrow \phi_c$, so the degradation was assumed to be the same as the power degradation.

TABLE 5-30. SOLAR FLARE DEGRADATION OF 30-MIL SHIELD

Time, years	$\phi_{sp} \times 10^{11}$	Q, per cent	Q', per cent	Q'', per cent
1	1	9	1.9	1.5
2	2	12	3.8	3
3	3	14	5.2	4
4	4	16	6.9	4.5
5	5	18	8.6	5
6	6	19	9.7	5.5

TABLE 5-31. SOLAR FLARE DEGRADATION OF 45-MIL SHIELD

Time, years	$\phi_{sp} \times 10^{11}$	Q, per cent	Q', per cent	Q'', per cent
1	0.8	2.6	1.5	1.2
2	1.6	3.8	2.8	2.2
3	2.4	4.6	4.4	3.1
4	3.2	5.2	5.6	3.8
5	4.0	5.8	6.9	4.5
6	4.8	6.3	8.2	4.9

3) A 45-mil quartz shield stops $E < 17$ mev protons Table 5-31)

$\phi(E > 17 \text{ mev}) \approx 8 \times 10^9$ protons/cm² per year

$(\phi_{sp} = \sim 5 \times 10^9 + 2 \times 10^9 + 10^9 + \dots \approx 8 \times 10^9)$

Figure 5-185, with an estimated curve for $\phi_e = 2 \times 10^{11}$, was used to supply the power degradation, Q. The fractional degradation of short circuit current, Q', was computed using the formula

$$Q' = 1 - \frac{I_{sc}}{I_{so}} = 1 - \left[1 + \frac{7}{9} \left(\frac{\phi}{\langle \phi_{25} \rangle} \right) \right]^{-1/2}$$

The fractional degradation of open-circuit voltage, Q'', was found using Q' and Figure 5-187, with a plot of experimentally determined current degradations against voltage degradations.

Results of the solar flare degradations are summarized in Figures 5-188 through 5-190.

Degradations Due to Trapped Radiations. The data on the trapped or Van Allen radiations were obtained from Reference 5-3. Data included the following:

- Electrons

1) The 6-mil shield was assumed to stop electrons of energy below 200 kev and the electron flux used was

$\phi_e = 1 \times 10^8$ electrons/cm² per second

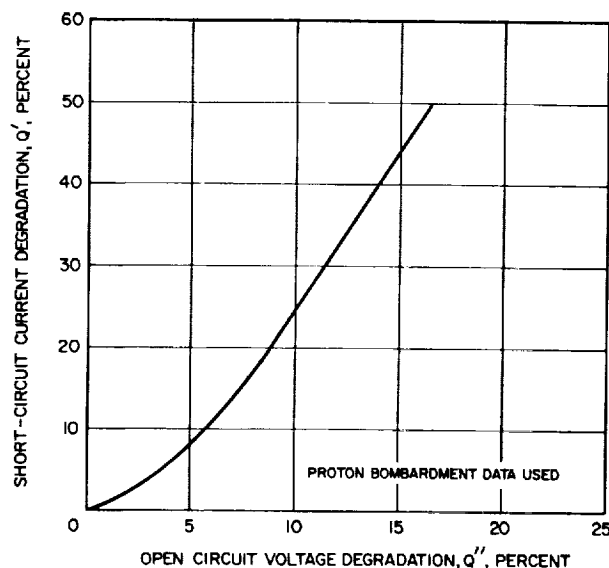


FIGURE 5-187. ESTIMATED RELATIONSHIP BETWEEN OPEN CIRCUIT VOLTAGE DEGRADATION AND SHORT CIRCUIT CURRENT DEGRADATION FOR THREE OHM-CM N-P SOLAR CELLS

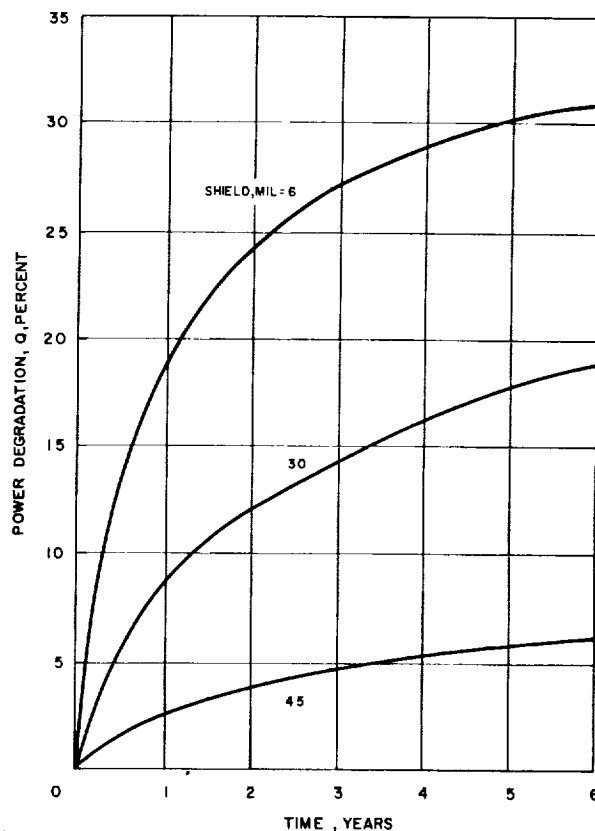


FIGURE 5-188. POWER DEGRADATION DUE TO SOLAR FLARES

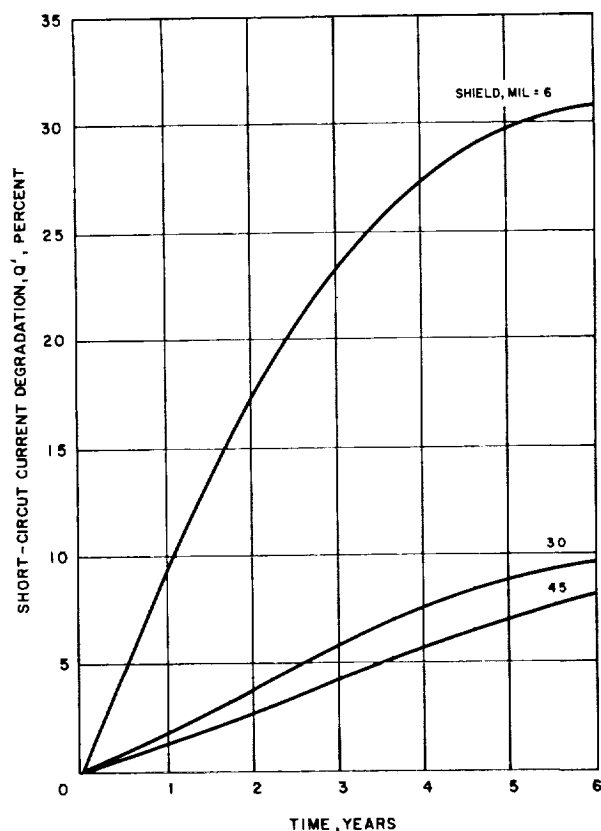


FIGURE 5-189. CURRENT DEGRADATION DUE TO SOLAR FLARES

- 2) The 30- and 45-mil shields stop electrons $E < 1$ mev with electron fluxes of $\phi_e = 2 \times 10^6$ electrons/cm² per second

• Protons. It was assumed that the various shields under consideration stop all the Van Allen protons at the synchronous altitude. The degradations due to the trapped electrons were computed in a manner similar to that for protons. Figure 5-184 was used to find the power degradation using the previously assumed critical flux for electrons of 2×10^{15} electrons/cm². Only the power effects were determined because the formula for the current degradation is invalid for fluxes approaching the critical flux (Conclusion). The results are shown in Tables 5-32 and 5-33 and Figure 5-191.

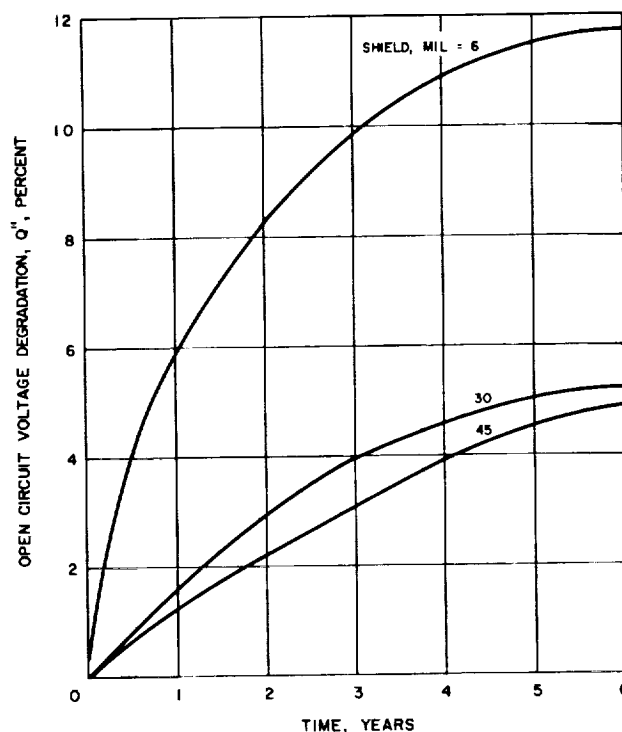


FIGURE 5-190. VOLTAGE DEGRADATION DUE TO SOLAR FLARES

TABLE 5-32. TRAPPED RADIATION DEGRADATION OF 6-MIL SHIELD

Time, years	$\phi \times 10^{15}$	Q, percent
1	3	30
2	6	36
3	9	40
4	12	43
5	15	45
6	18	46

TABLE 5-33. TRAPPED RADIATION DEGRADATION OF 30- AND 45-MIL SHIELDS

Time, years	$\phi \times 10^{15}$	Q, percent
1	7	7.5
2	14	10
3	21	11.5
4	28	13
5	35	14
6	42	15

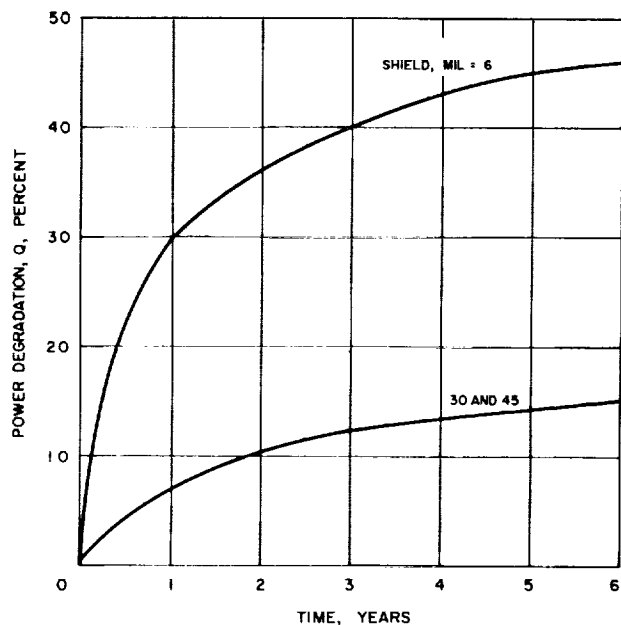


FIGURE 5-191. POWER DEGRADATION DUE TO TRAPPED ELECTRONS

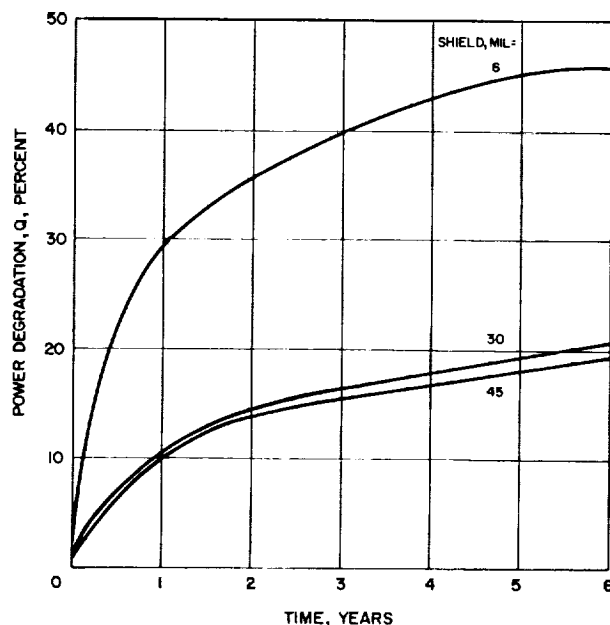


FIGURE 5-192. POWER DEGRADATION IN SPACE ENVIRONMENT

Degradations Due to Total Radiations. The proton damage of solar flares and the trapped electron damage are not directly additive. The proton flux was converted to an equivalent electron flux (for solar cell damage) so the combined effects could be correctly considered. These fluxes were then added and the appropriate degradations determined from the total integrated flux as in previous sections discussing solar flare and trapped radiation degradations (Tables 5-34 through 5-36). The results are shown in Figures 5-192 through 5-194.

The conversion factor for changing the proton flux to an equivalent electron flux was 10^4 .

$$(\phi_{sp \rightarrow e} = \phi_{sp} \times 10^4)$$

Conclusion. The final results of the radiation damage analysis show the 6-mil shield to be unacceptable due to inordinate degradations. The differences between the 30- and 45-mil covers

appear slight, so the 30-mil shield was recommended. The final degradations expected are presented in Figure 5-195.

The analysis performed was largely based on estimated quantities. As more knowledge is gained on the space environment and the critical fluxes applicable to the cells in question, the uncertainty of the results can be greatly reduced.

TABLE 5-34. TOTAL RADIATION DEGRADATION OF 6-MIL SHIELD

Time, years	$\phi_{sp} \times 10^{10}$	$\phi_{sp \rightarrow e} \times 10^{10}$	$\phi_e \times 10^{10}$	$\phi_{total} \times 10^{10}$	Q, per-cent	Q', per-cent	Q'', per-cent
1	6	0.06	3	3.06	30	*	*
2	12	0.12	6	6.12	36		
3	18	0.18	9	9.18	40		
4	24	0.24	12	12.2	43		
5	30	0.30	15	15.3	45		
6	36	0.36	18	18.4	46		

*These were not computed because ϕ_e becomes much less than ϕ_{total} and the critical flux for power is not really the same as the critical flux for current.

TABLE 5-35. TOTAL RADIATION DEGRADATION OF 30-MIL SHIELD

Time, years	$\phi_{sp} \times 10^{16}$	$\phi_{sp} \rightarrow \phi_e \times 10^{14}$	$\phi_e \times 10^{14}$	$\phi_{total} \times 10^{14}$	Q, per cent	Q', per cent	Q'', per cent
1	1	1	0.7	1.7	10.5	3.4	3
2	2	2	1.4	3.4	13.5	6.1	4.3
3	3	3	2.1	5.1	16	8.6	5.2
4	4	4	2.8	6.8	18	11.1	6.0
5	5	5	3.5	8.5	19.5	13.5	6.7
6	6	6	4.2	10.2	21	16.0	7.5

TABLE 5-36. TOTAL RADIATION DEGRADATION OF 45-MIL SHIELD

Time, years	$\phi_{sp} \times 10^{16}$	$\phi_{sp} \rightarrow \phi_e \times 10^{14}$	$\phi_e \times 10^{14}$	$\phi_{total} \times 10^{14}$	Q, per cent	Q', per cent	Q'', per cent
1	0	0.8	0.7	1.5	10	2.8	2.5
2	1.6	1.6	1.4	3	13	5.2	4
3	2.4	2.4	2.1	4.5	15	7.8	5
4	3.2	3.2	2.8	6	17	9.9	5.6
5	4.0	4.0	3.5	7.5	18.5	12.2	6.2
6	4.8	4.8	4.2	9	19.5	13.8	6.7

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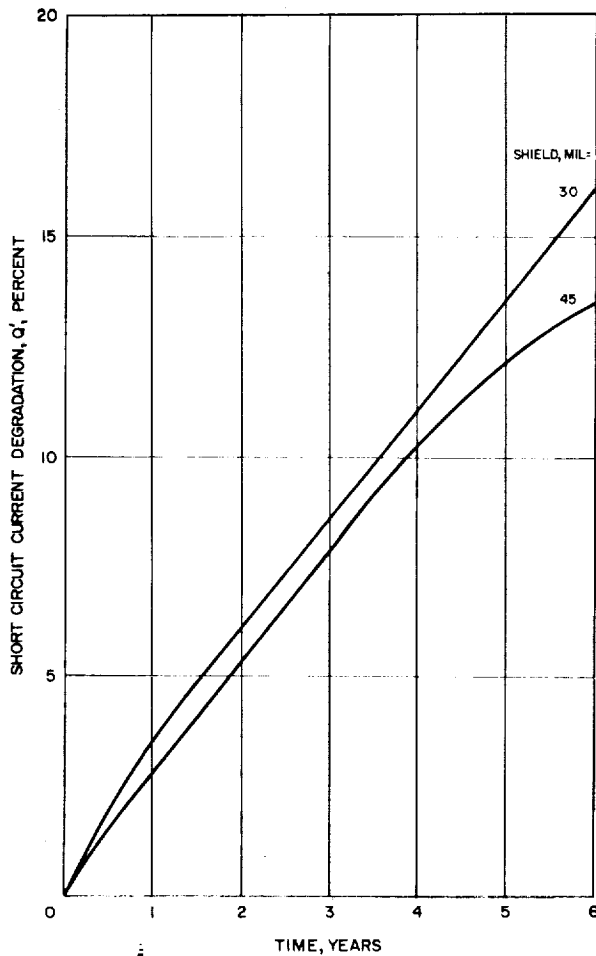


FIGURE 5-193. CURRENT DEGRADATION IN SPACE ENVIRONMENT

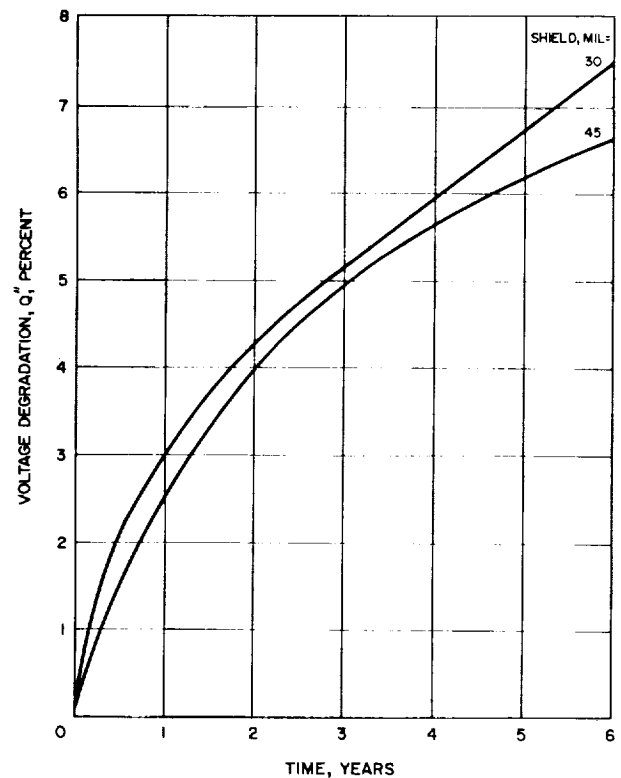


FIGURE 5-194. VOLTAGE DEGRADATION IN SPACE ENVIRONMENT

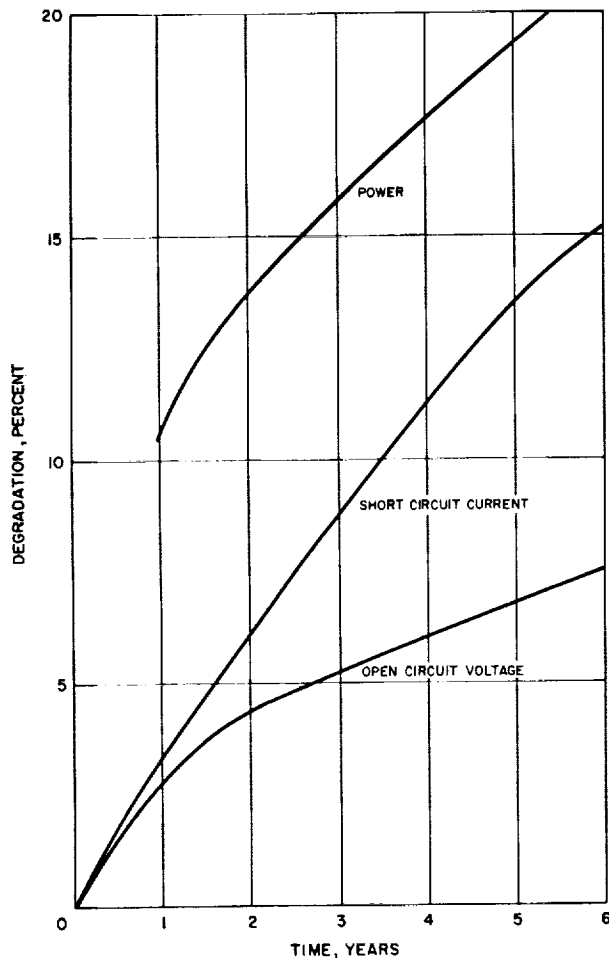


FIGURE 5-195. PARTICULATE RADIATION DEGRADATION

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STRUCTURE

General Arrangement and Structural Design.

The equipment and structures have been rearranged from the T-1 model to improve the present vehicle and provide the maximum of alternate payload flexibility. The principal objectives of this effort were reduction of structure weight, maintenance of structure stiffness, increase in space available for electronics components, accommodation of the revised apogee motor, increase in thermal conductivity of the structure to reduce temperature of high heat loss components, reduction of manufacturing costs, increase in the ratio of roll axis moment of inertia to pitch axis moment of inertia, and improvement of installation and maintenance access to all components.

The primary structure for the vehicle is subdivided into three main sections: aft thrust tube assembly, central thrust tube assembly, and forward equipment support structure assembly. The aft and central thrust tube assemblies are joined in a single tube assembly. The forward equipment support structure assembly is attached to the tube assembly by bolted connections so that it can be readily attached and removed. This arrangement facilitates fabrication, assembly, and maintenance of the spacecraft. The arrangement of equipment and structure is shown in Figure 5-196. The bipropellant control system and its relation to the spacecraft structure is illustrated in Figure 5-197.

The aft face of the aft thrust tube assembly attaches to the Agena-spacecraft interconnect structure by a V-band clamp. The interface control dimensions are shown in Figure 5-198. Radial ribs of aluminum alloy are attached to the aft thrust tube and are part of this assembly. The radial ribs support the aft electronic components and provide thermal paths for heat distribution. The radial ribs also provide a structure to sup-

port the components that are not supported directly from the central tube structure. These components include the control system fuel and oxidizer tanks, battery assembly packages, honeycomb solar panels, and aft hoist support fittings. Ring frames are used forward and aft of the radial ribs to provide balancing reaction to the longitudinal loads in the ribs and distribute lateral rib loads into the central tube structure. The forward frame has additional stiffness to avoid introducing deflection loads into the apogee motor case. The antenna package and multiplexers are supported inside the thrust tube by a two-beam cruciform structure forward of the separation plane. To attain accurate alignment of the apogee motor thrust vector, the aft thrust tube assembly is arranged so that the attach interface and the apogee motor alignment face will be machined in a single machine setup of the aft assembly.

The central structure is a thrust tube of aluminum alloy sheet with external longitudinal stiffeners. This semimonocoque structure provides support for the forward structure and the equipment components adjacent to this structure which include the electronic quadrant packages, support for the forward ends of the fuel and oxidizer tanks for the control system, and forward support for the battery packages. The longitudinal stiffeners are magnesium and aluminum alloy, depending on the magnitude of the load to be distributed into the thrust tube sheet. The forward ends of the longitudinal stiffeners provide for attachment of the removable forward equipment support structure and the forward hoist attach fittings. The diameter of the sheet thrust tube provides clearance about the apogee motor for a motor installation protective sleeve and for thermal insulation between the motor case and the surrounding structure. The space envelope and interface control drawing for the apogee motor is shown in Figure 5-199.

The forward equipment support structure assembly has an upper torus ring, an interconnect tubular truss structure, and a ring frame at the tie to the central thrust tube structure. The upper torus ring is a sheet metal box section ring frame to support the forward mounted equipment and provide partial support for the solar panels. Intercostals are installed within the box section at all concentrated load points. Flanged lightening holes are located in the box structure to provide for inspection, maintenance, and repair during test and qualification. No blind fasteners are used in this section. The intermediate truss structure is of tube members with riveted end fittings and is fabricated in paired tube sections to provide accurate alignment of the forward structure with minimum tooling.

A study was conducted to determine the optimum structure arrangement for this interconnect structure. The alternate arrangements included a beaded sheet structure and a formed sheet Vierendiel truss. The tubular truss arrangement was selected on the basis of minimum weight, minimum cost, and minimum deflection. The forward equipment support structure is bolted to the central thrust tube structure so that it is removable. With the forward support structure removed, the two unitized control systems may be installed or replaced in the spacecraft as a complete assembly without excessive deflection of the interconnecting lines (Figure 5-200).

Each individual solar cell support panel is supported by three attachment points spaced on 120 degree radial lines from the center of gravity of the panel. Each attach point provides restraint normal to the plane of the panel and, in the plane of the panel, restraint perpendicular to the radial line from the center of gravity. Two attachments are ball and socket and the third is a slotted groove so that no other restraint is applied to

the solar panel. This support arrangement (Figure 5-201) prevents the possible introduction of loads into the panel from deflections of the spacecraft primary structure. It also provides a quick-disconnect means of solar panel installation or removal without the use of tools.

The 16 solar panels are $\frac{1}{2}$ inch thick aluminum alloy honeycomb faced with fiberglass and epoxy resin. Suitable panels are provided with an opening to allow the velocity jets to project through the face of the panel. The panel configuration is located relative to the spacecraft so as to place the velocity jets at the intersection of adjacent panels, providing greater flexibility in the longitudinal location of the velocity jets, which must be located in the plane of the longitudinal center of gravity of the spacecraft. If subsequent design progress requires relocation of the velocity jets, it can be accomplished with minimum effort.

Most of the details of the spacecraft dynamic envelope in the modified Agena Nimbus shroud have been finalized. They are illustrated in Figure 5-202.

Structural Dynamic Test and Analysis. Structural dynamics effort in this reporting period has included testing of the T-1 structural model, dynamic analyses of the T-1 and T-2 configurations, and initiation of the solar panel development vibration test.

Vibration tests were conducted on the T-1 structural model from period 25 April through 2 July 1963 at the Hughes Space Environmental Simulation Laboratory using a Ling L-249 electromagnetic shaker. Test results are described in Volume 7, *Advanced Syncom Structural Vehicle (T-1) Vibration Test Results*.

The spacecraft qualification level tests were conducted in accordance with the NASA test

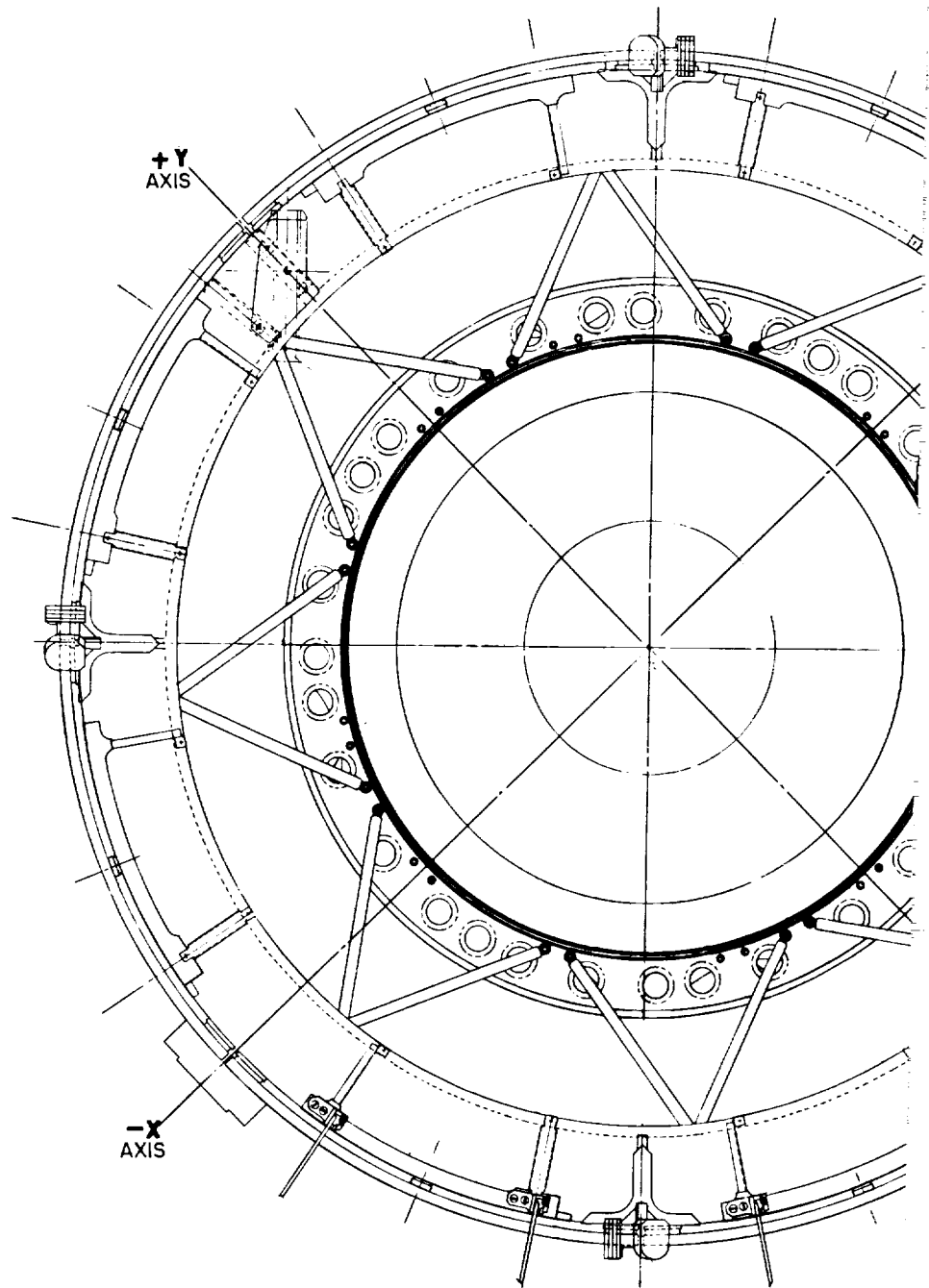
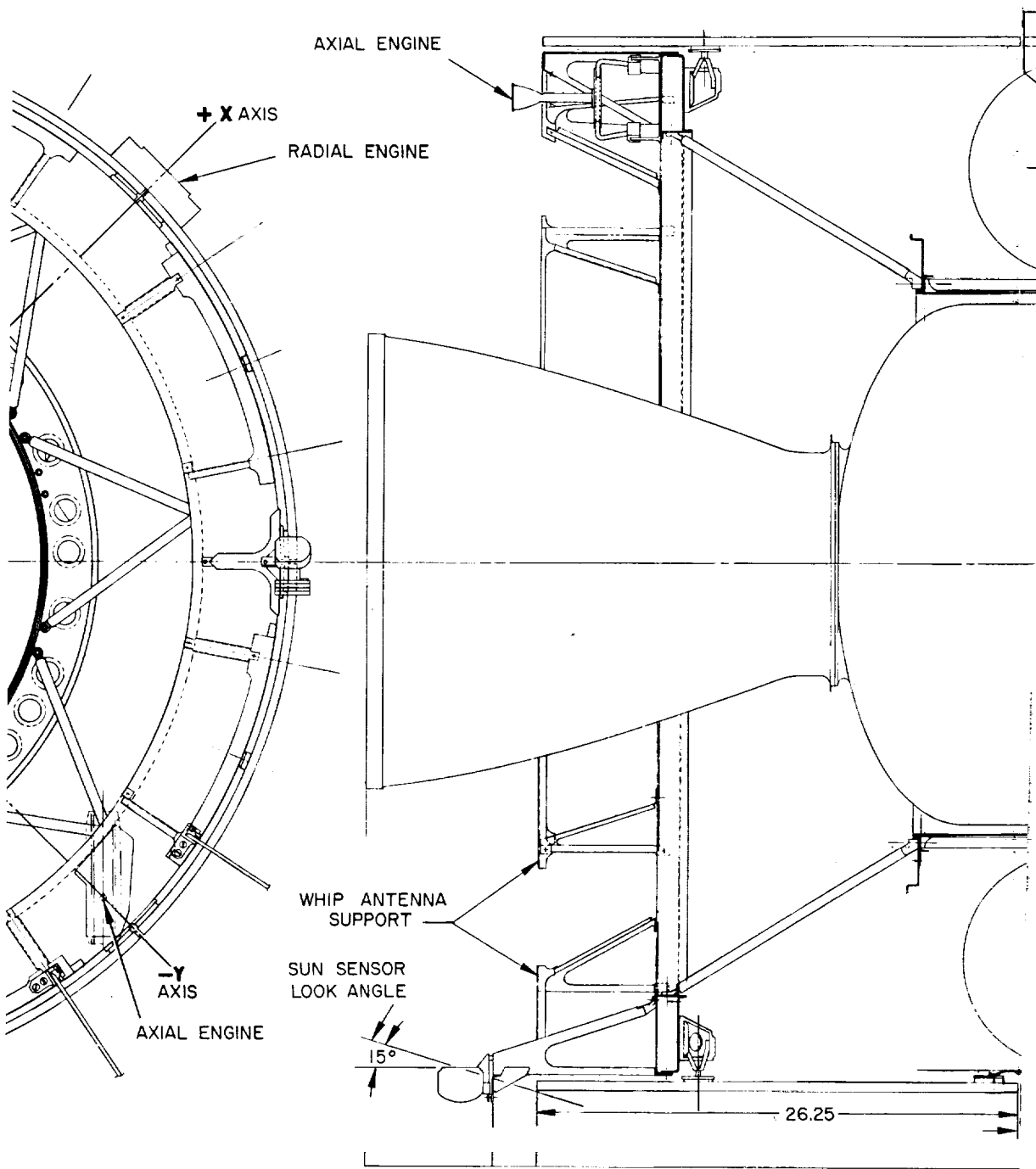


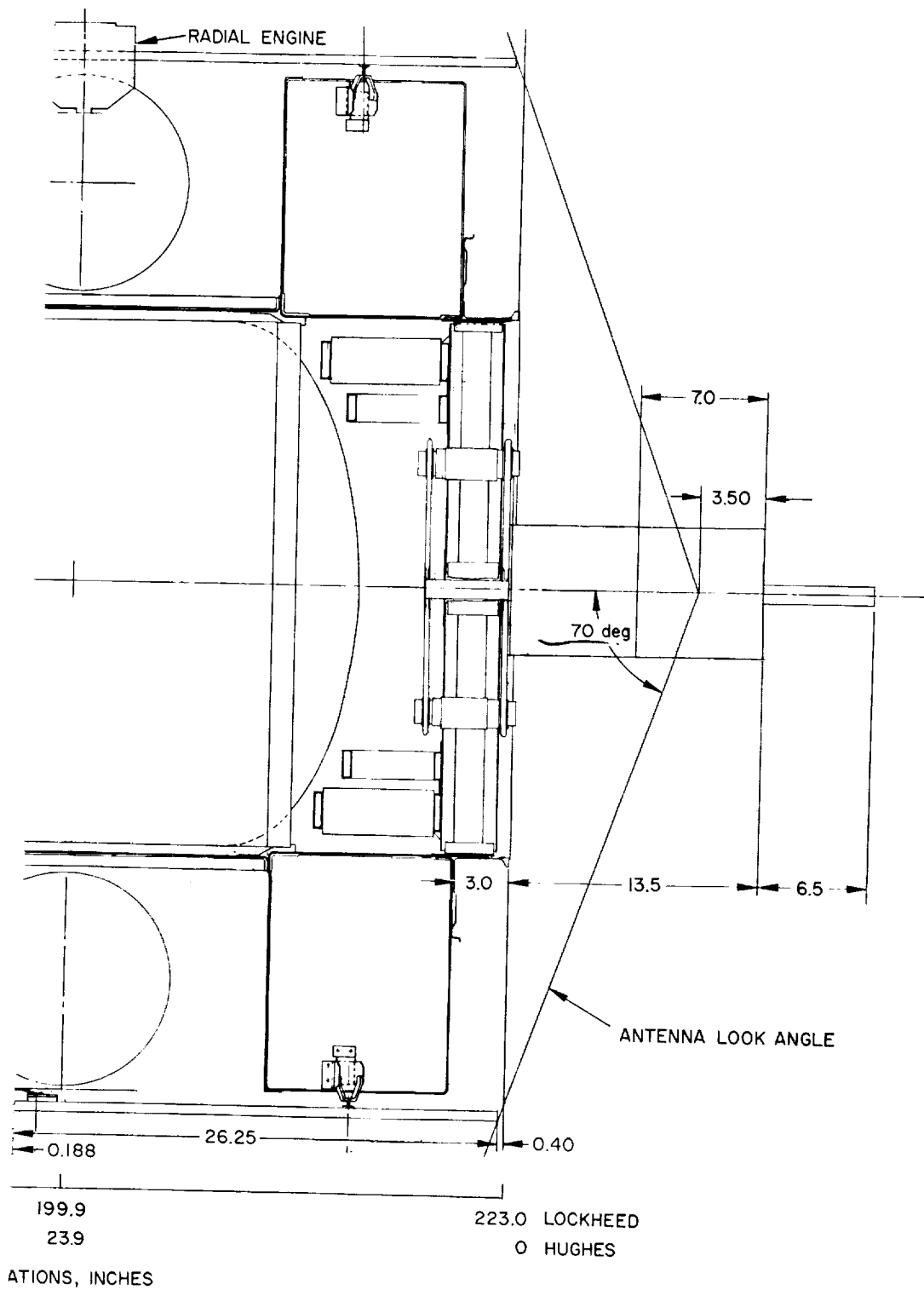
FIGURE 5-196. GENERAL ARRANGEMENT OF SPACECRAFT



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62.5	55.5	53.088

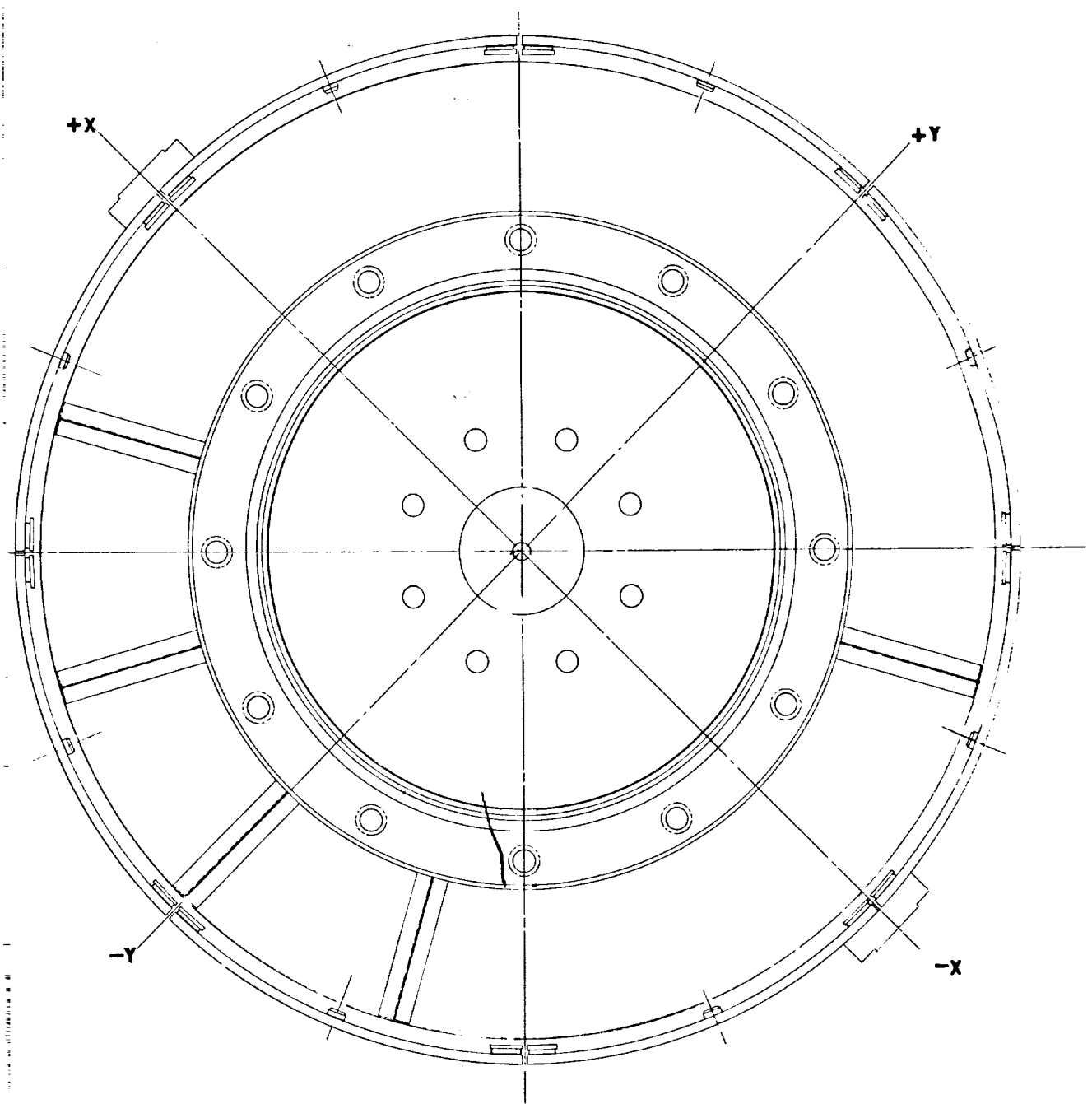
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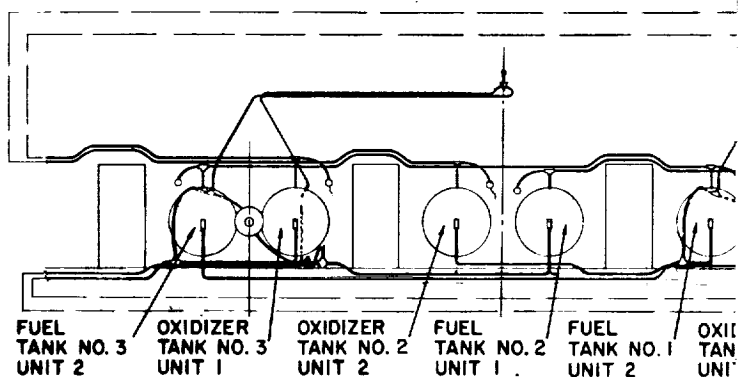
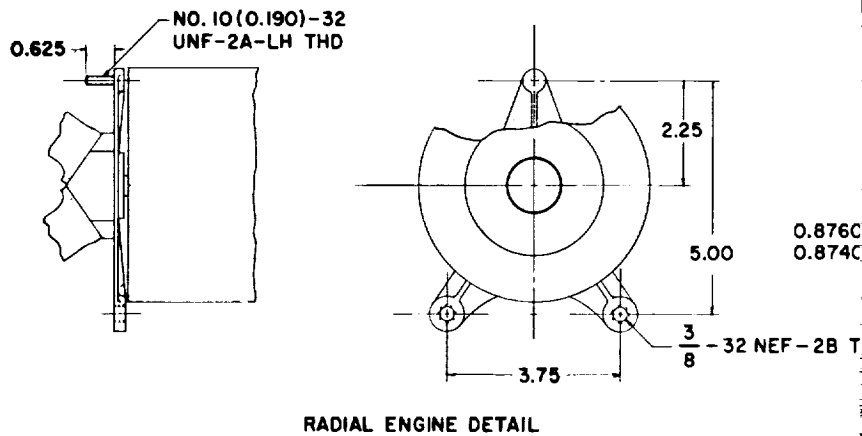
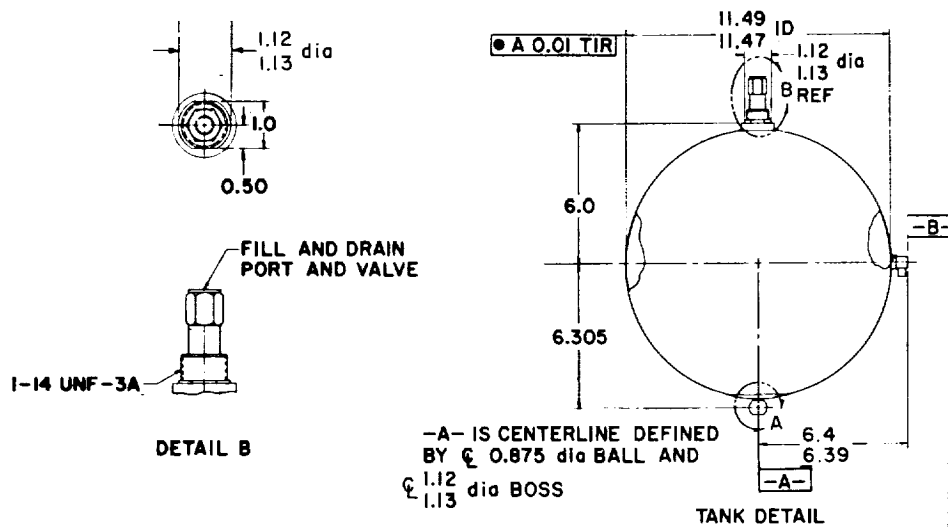
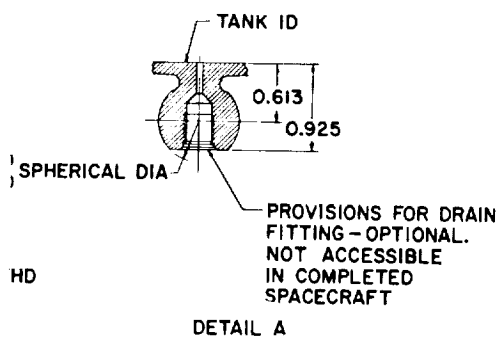


FIGURE 5-197. BIPOPELLANT REACTION CONTROL SYSTEM IN



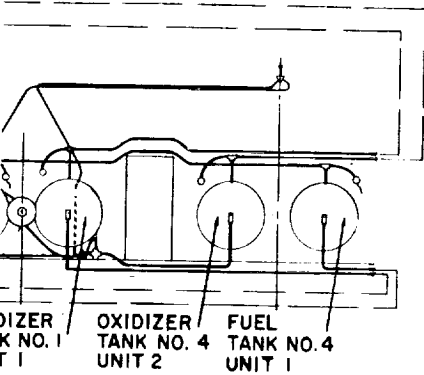
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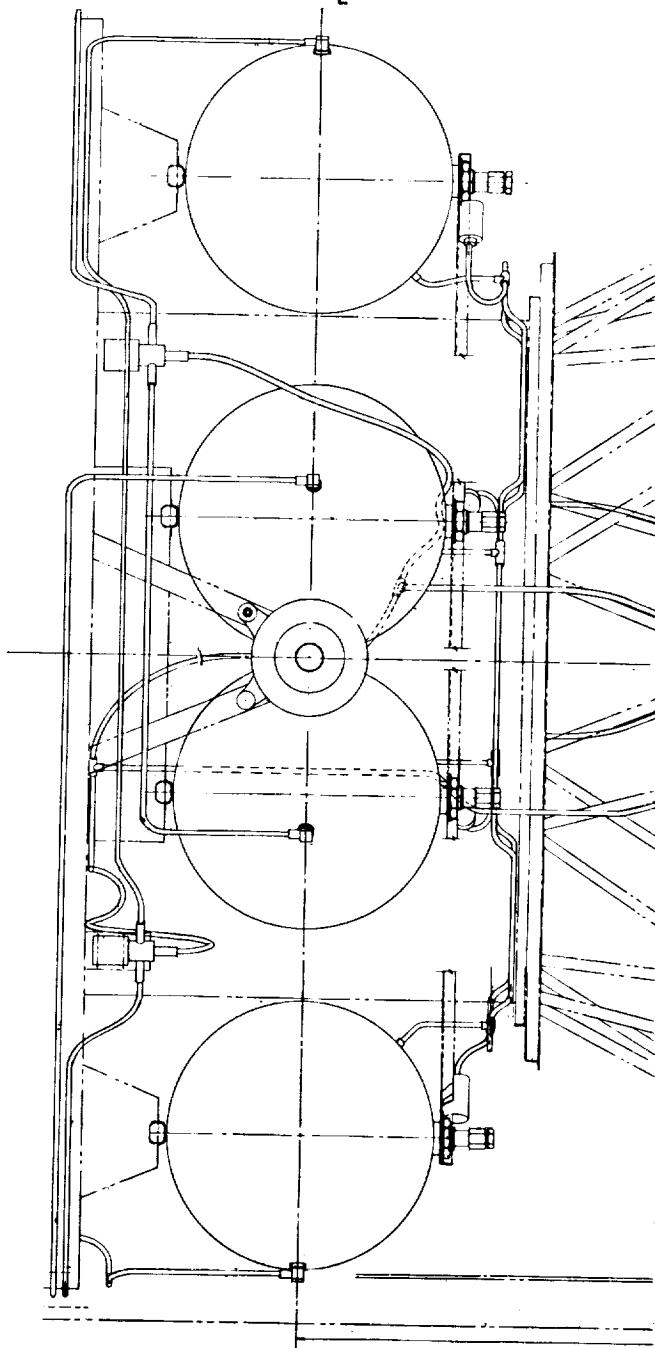
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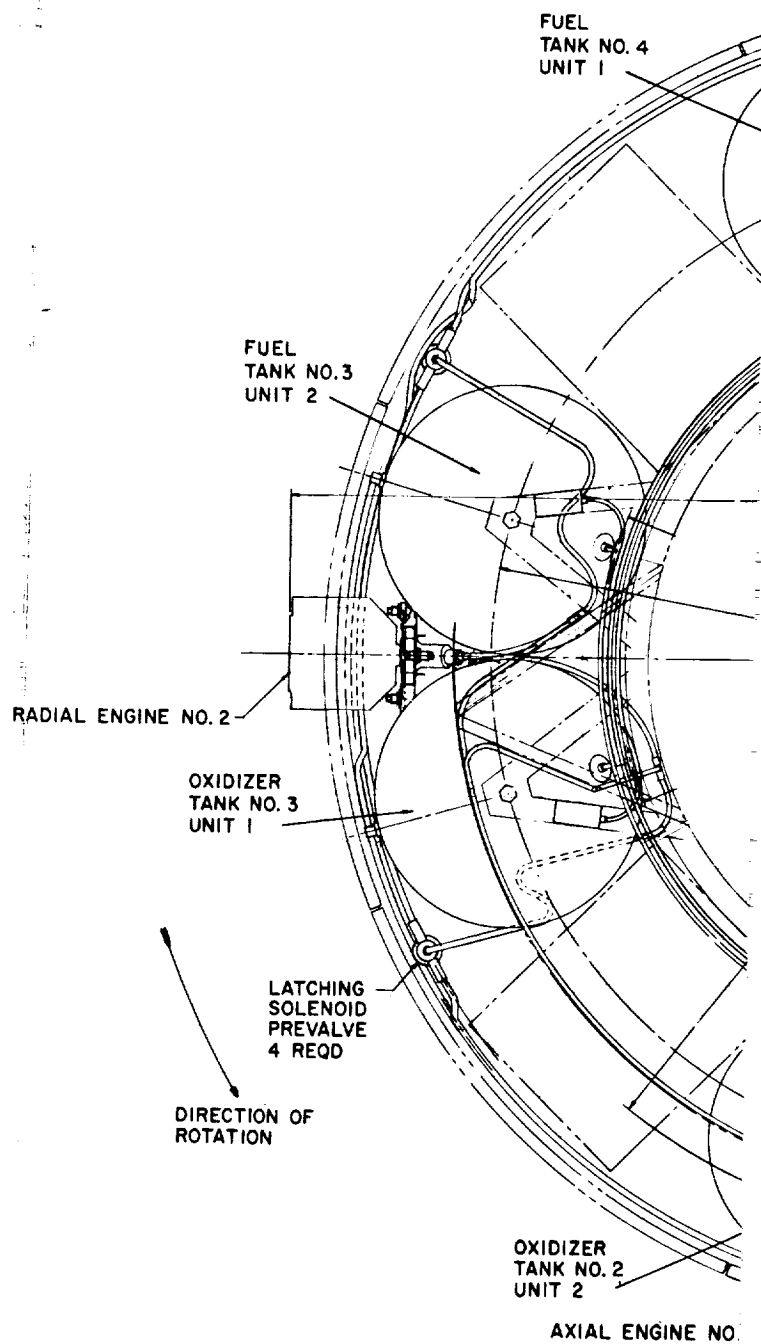
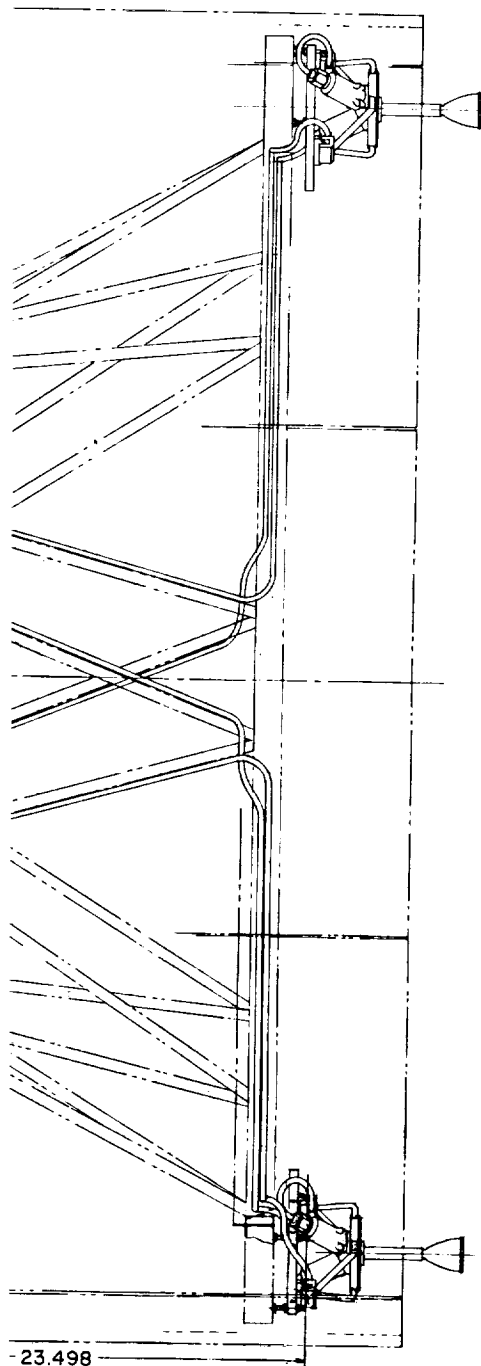
INSTALLED IN SPACECRAFT

TANK AND RADIAL
ENGINE 6



2

5. Spacecraft System Design

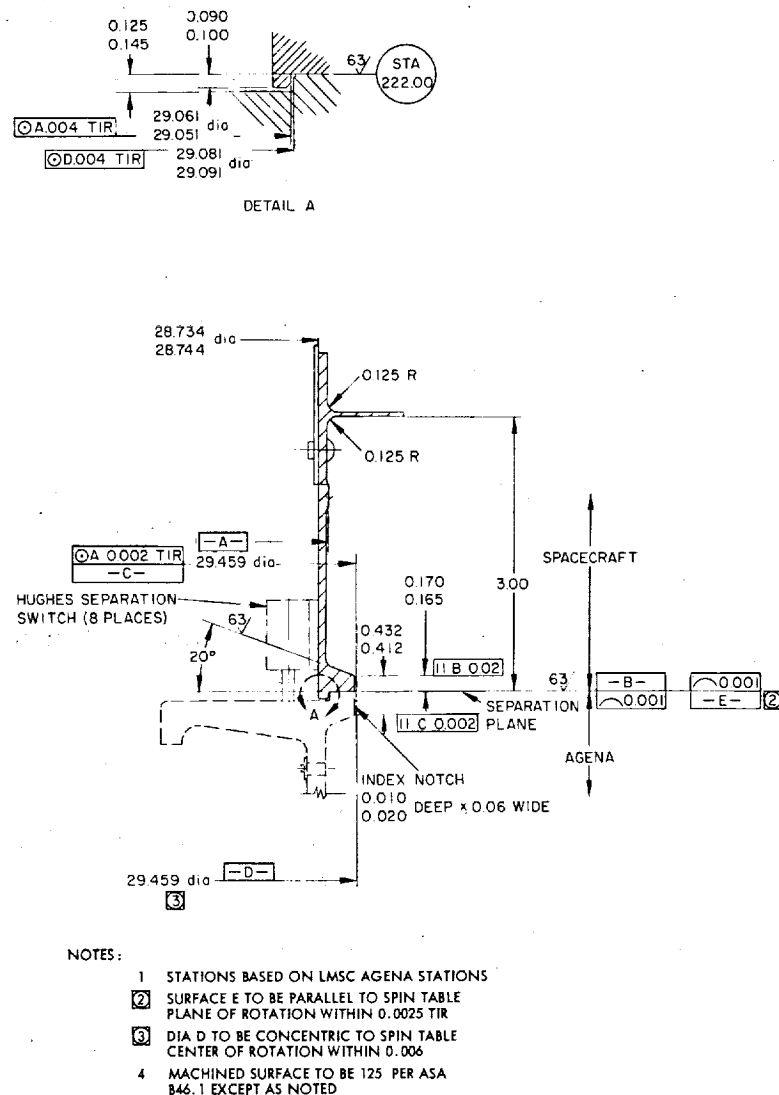


FIGURE 5-198. SEPARATION PLANE INTERFACE

specifications given in Advanced Syncom Project Notice 9, Revision B, repeated in Table 5-37. Random qualification tests were not conducted (with two exceptions), and the spacecraft response to random inputs has been calculated from measured sinusoidal data. The test objectives were accomplished. The strength and stiffness adequacy of the spaceframe was verified, and data necessary to predict structural design

loads, component design loads, and qualification vibration test levels were acquired.

There was no damage to the primary structure during the test. Failures which occurred on the secondary structure and components are listed in Table 5-38. Design modifications have been suggested to eliminate these failures on subsequent vehicles.

TABLE 5-37. VIBRATION LEVELS FOR EVALUATION OF ADVANCED SYNCOM STRUCTURE.
(Levels to be applied at separation interface)

Sinusoidal excitation,* three axes, logarithmic sweep at 2 octaves per minute, 4.35 minutes duration.	
5 to 15 cps	0.25 inch double amplitude
15 to 250 cps	3 g peak
250 to 400 cps	5 g peak
400 to 2000 cps	7.5 g peak
Random excitation,* three axes, 6 minutes duration per axis.	
20 to 80 cps	0.04 g ² /cps
80 to 1280 cps	Increasing from 0.04 g ² /cps at 0.61 db per octave
1280 to 2000 cps	0.07 g ² /cps
Torsional vibration, applied at separation plane and corrected to interface diameter assuming rigid body motion, sinusoidal excitation, logarithmic sweep at 1 octave per minute.	
50 to 80 cps	7.5 g peak tangential or 5-foot diameter

*Exception may be taken where the predominant longitudinal and lateral frequencies are sufficiently decoupled from those of the Atlas/Agena with the spacecraft attached. In that case, the spacecraft response at the center of gravity in the bandwidth of the predominant frequency may be limited to that of the separation plane input in the same predominant frequency bandwidth.

TABLE 5-38. T-1 QUALIFICATION LEVEL VIBRATION TEST FAILURES

Test	Run	Description of Failure
2	1	Innerface sheet buckling of solar panels S/N 2 and S/N 12, along restrained edges.
3	2	Failure of orientation rocket platform truss located over quadrant 3 electronics.
4	1	Failure of orientation rocket platform truss located over quadrant 1 electronics.
6	1	Failure of orientation rocket platform truss located over quadrant 4 electronics. Orientation rocket truss is one that failed during test 3, run 2. It had been repaired and installed in new position prior to test 5, run 2.
6	1	Innerface sheet buckling of solar panels S/N 2 along restrained edge.
6	4	Failure of orientation rocket platform truss located over quadrant 2 electronics. This truss failed during test 4, run 1. It had been repaired and installed in new position on spacecraft prior to test 5, run 1.
6	4	Sun sensor bracket (Hughes Drawing X209711, item 15) cracked.
10	2,3	Failure of several solar panels as described in test report.

TABLE 5-39. SPACECRAFT FUNDAMENTAL FREQUENCY COMPARISON
(CYCLES PER SECOND)

Vibration	Test Results	T-1 (Analytical)	T-2 (Analytical)
Longitudinal	130	135	139
Lateral	49	48	50

Development vibration test levels for components have been derived from T-1 test data. The test spectra are given in Figures 5-203 through 5-207.

The fundamental frequencies found experimentally during the vibration tests compare closely with those derived analytically, as shown in Table 5-39. The analytical studies were based on a 10 degree of freedom lateral model and a 5 degree of freedom longitudinal model. In both cases, the model was cantilevered from the Agena interface.

The expected change in the fundamental lateral frequency of T-2 for changes in thrust tube gage is shown in Figure 5-208. Analytical studies also yielded a torsional fundamental frequency of 74 cps based on a 3 degree of freedom model mounted at the Agena interface.

The response of the T-2 analytical model to sinusoidal base excitation has been obtained. Lateral response curves are presented in Figure 5-209 for the rib and apogee motor mass center.

T-2 Solar Panel Vibration Test. Developmental vibration tests are currently in progress on the T-2 solar panels. The first three plate modes of the panel have been determined, and strain gages are being installed at points of maximum bending moment in these modes. Sinusoidal frequency sweeps of 1 octave per minute were performed from 5 to 2000 cps at 1/2 g input to establish resonant frequencies of the panel. Figure 5-210 shows the vibration fixture and

Z IS ϕ OF NOZZLE AS DEFINED
BY THROAT AND EXIT DATUM
DIAMETERS TO BE ESTABLISHED
BY MOTOR MANUFACTURER

25.0 dia MAX

NAS 1068C4M PLATE NUT OR
EQUIVALENT 24 PLACES

ϕ 0.02 dia

0.29 MIN HOLE

ϕ 0.02 dia

28.134
28.129 dia (REF)

0.500
BSC (REF)

0.05 x 45 deg C

DETAIL - A

FIGURE 5-199. APOGEE MOTOR SPACE ENVELOPE AND INTERFACE

100

100

100

100

100

100

100

100

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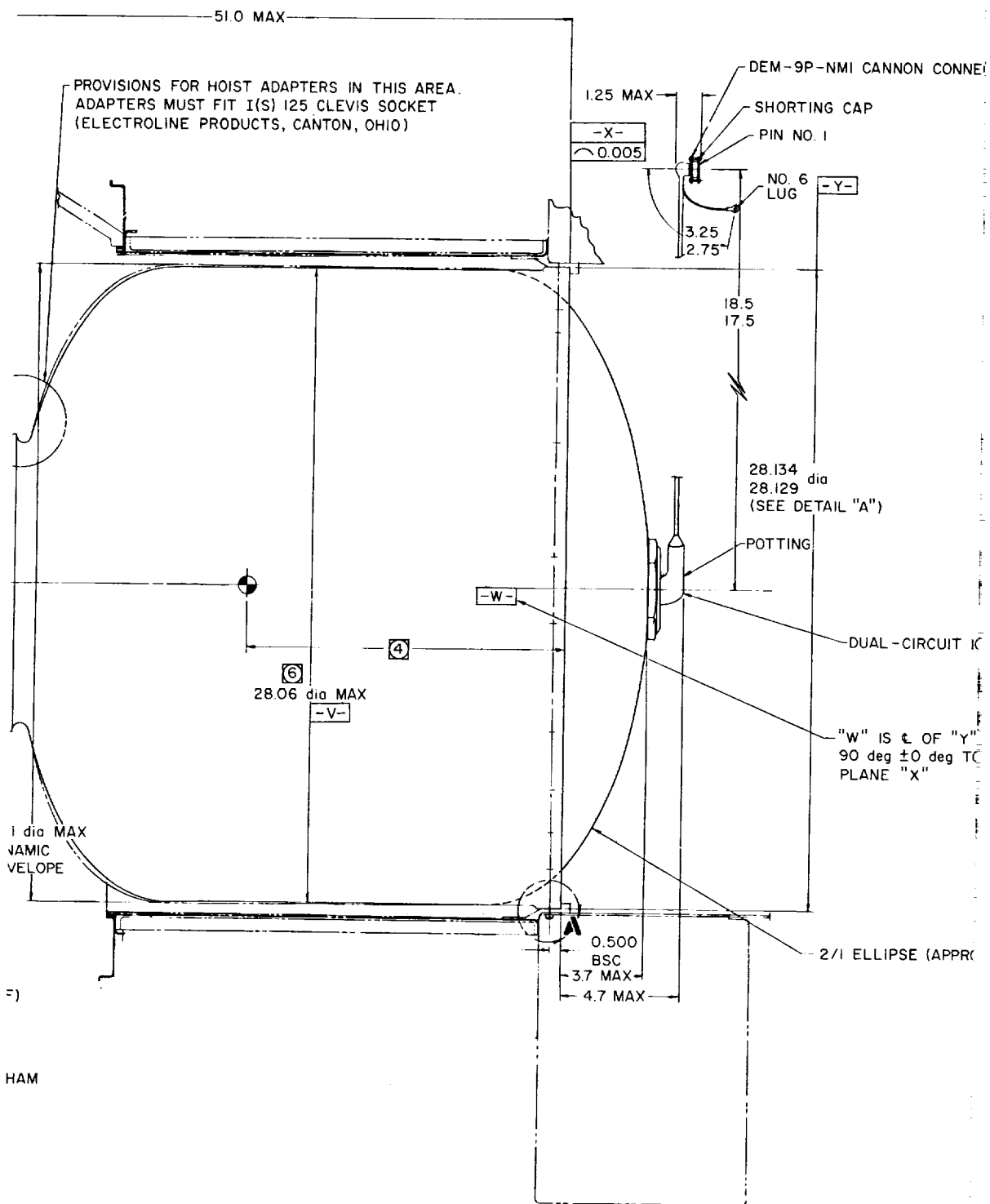
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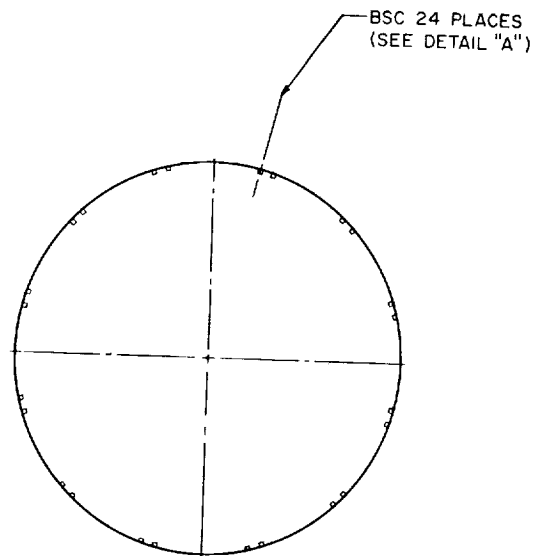
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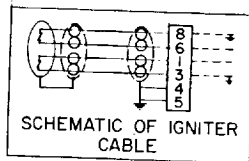


OTOR



IGNITER

dia AT
)



MOMENT OF INERTIA lb-in ²	MAXIMUM PITCH	MINIMUM ROLL
LOADED	112,500	90,000
B.O.	25,000	11,000

DX)

1. ϵ "Z" SHALL PASS THRU PLANE "X" WITHIN 0.03 OF ϵ "W"
2. ϵ "Z" SHALL BE PERPENDICULAR TO PLANE "X" WITHIN 0.001 PER in
3. FULLY LOADED MOTOR WEIGHT SHALL NOT EXCEED 874.5 lbs
4. STATIC MOMENT OF BURNED-OUT MOTOR ABOUT PLANE "X" SHALL NOT EXCEED 2011 lb-in.
5. SEE TABLE FOR MOMENTS OF INERTIA
6. DIA "V" INDICATES MAX ALLOWABLE EXCURSIONS OF CASE INCLUDING ECCENTRICITY AND OVALITY WITH RESPECT TO ϵ "W"

3

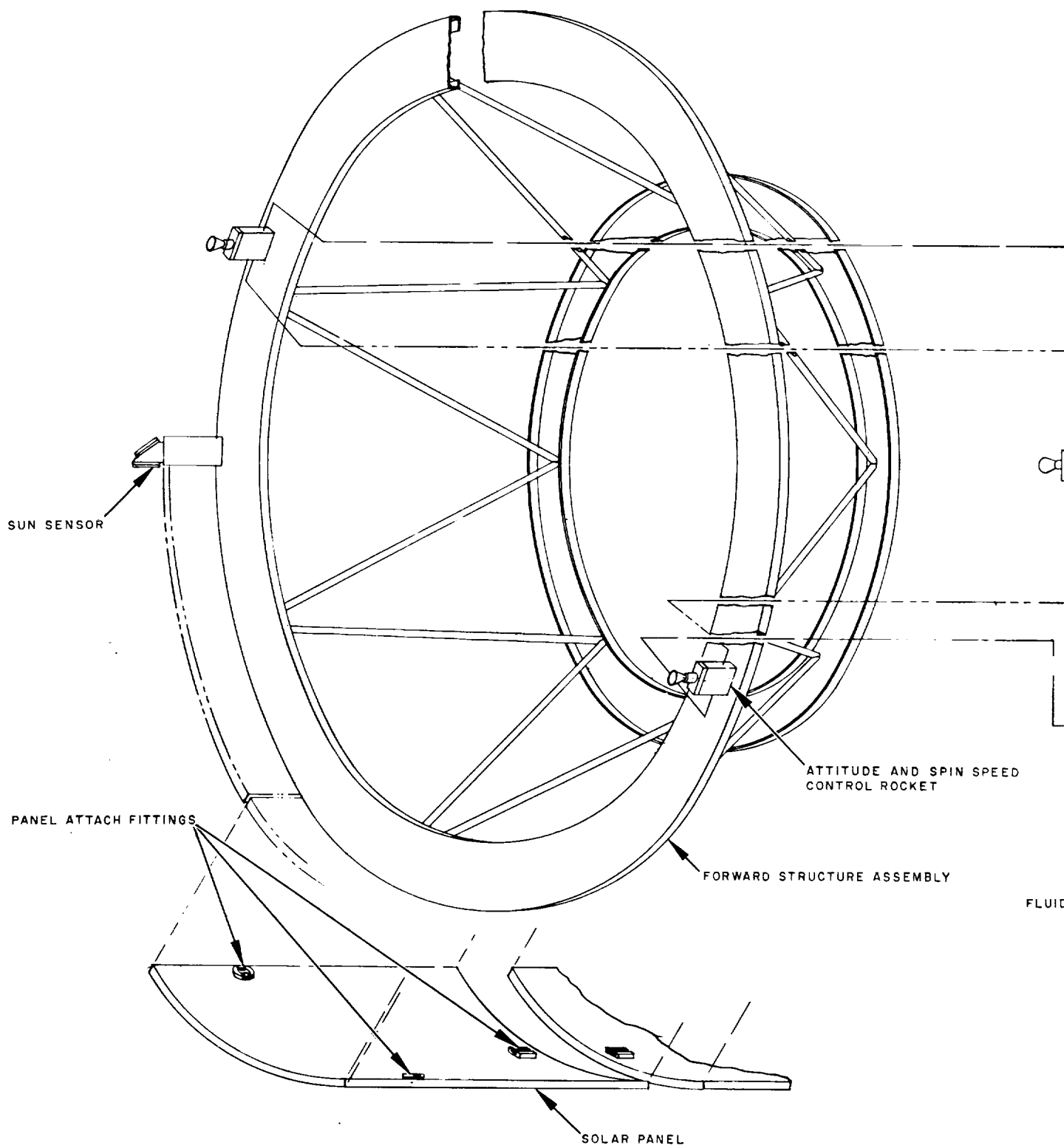
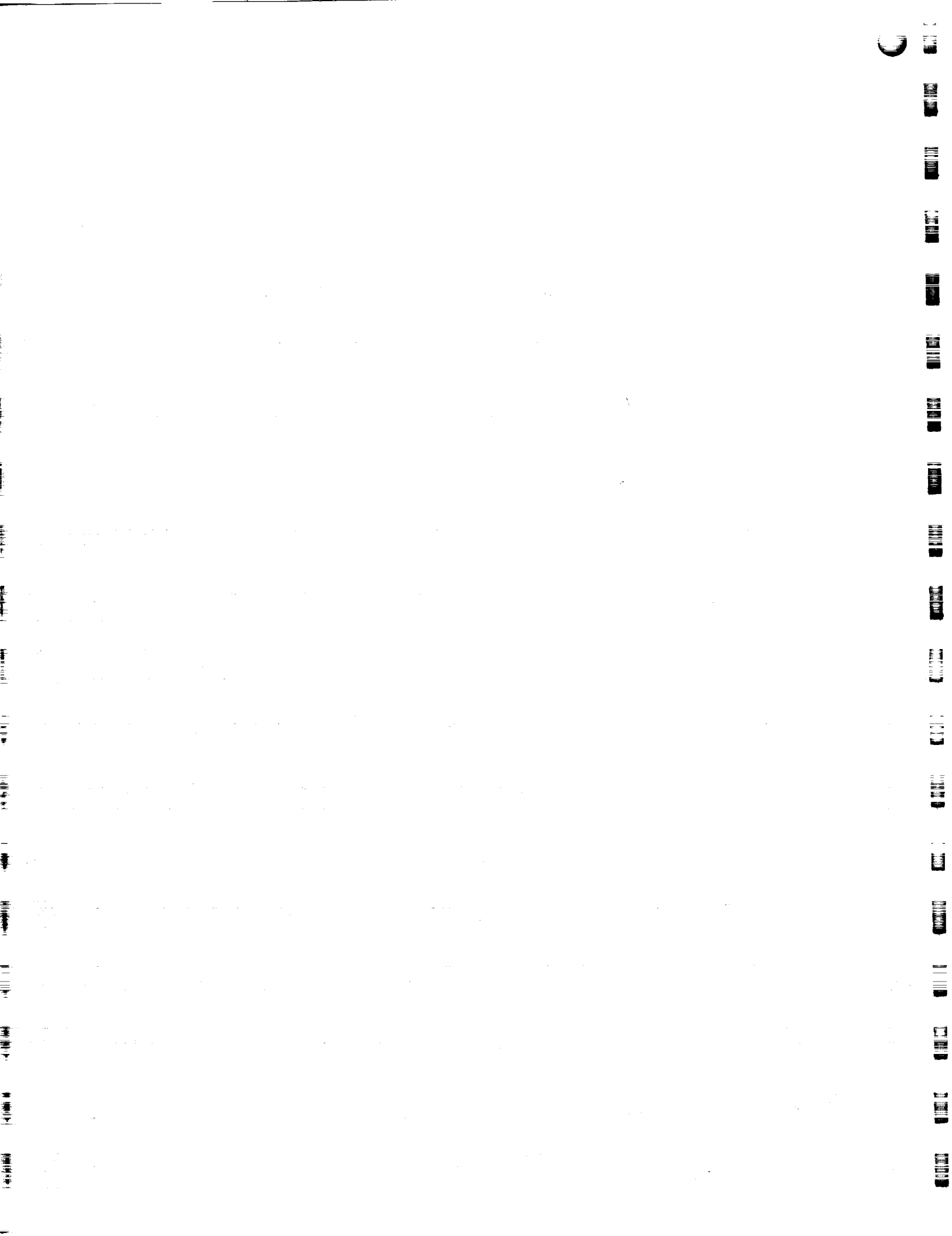
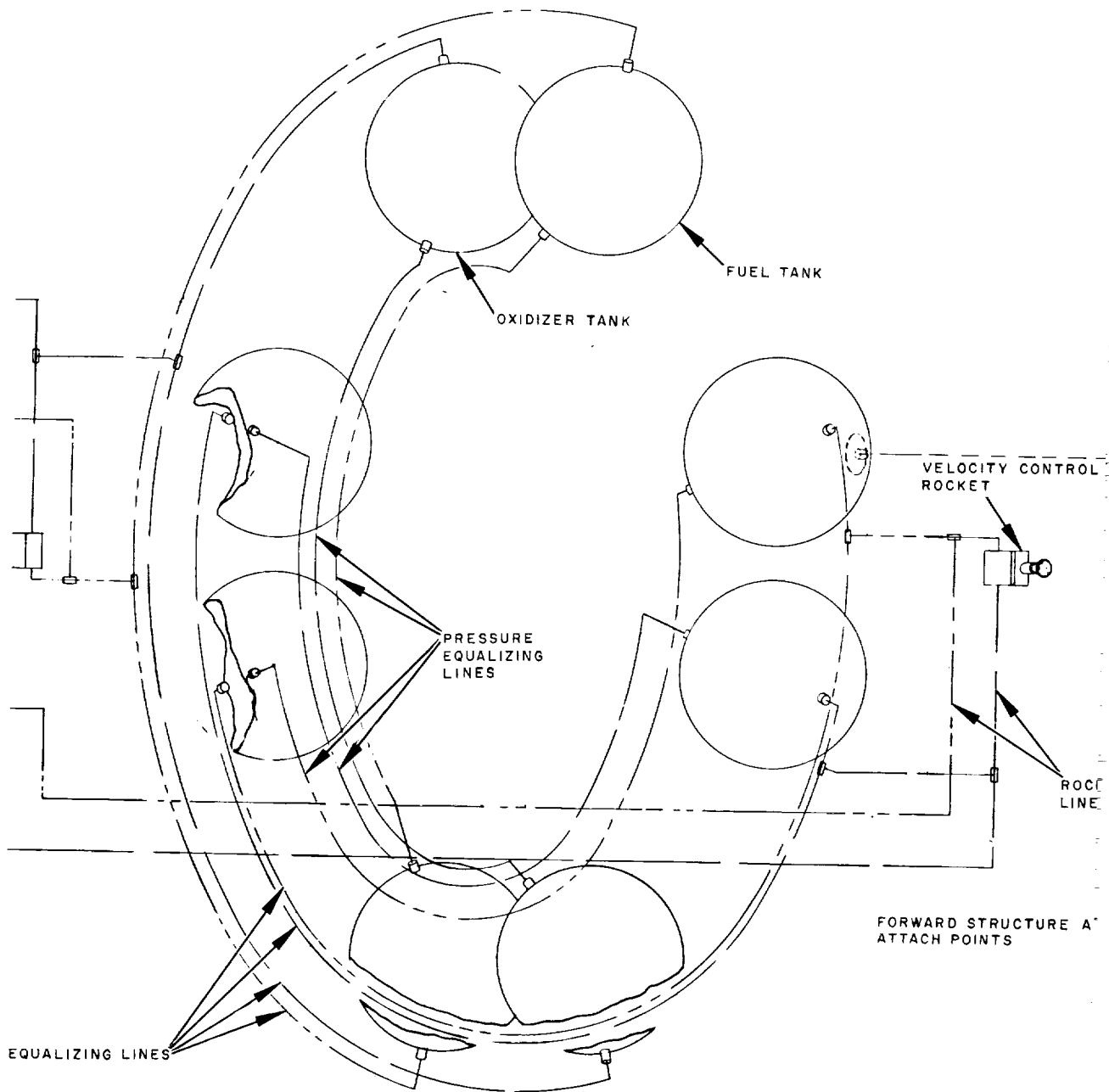


FIGURE 5-200. EXPLODED VIEW ILLUSTRATING INSTALLATION OF BI-PROPELLANT REACTION CONTROL SYSTEM





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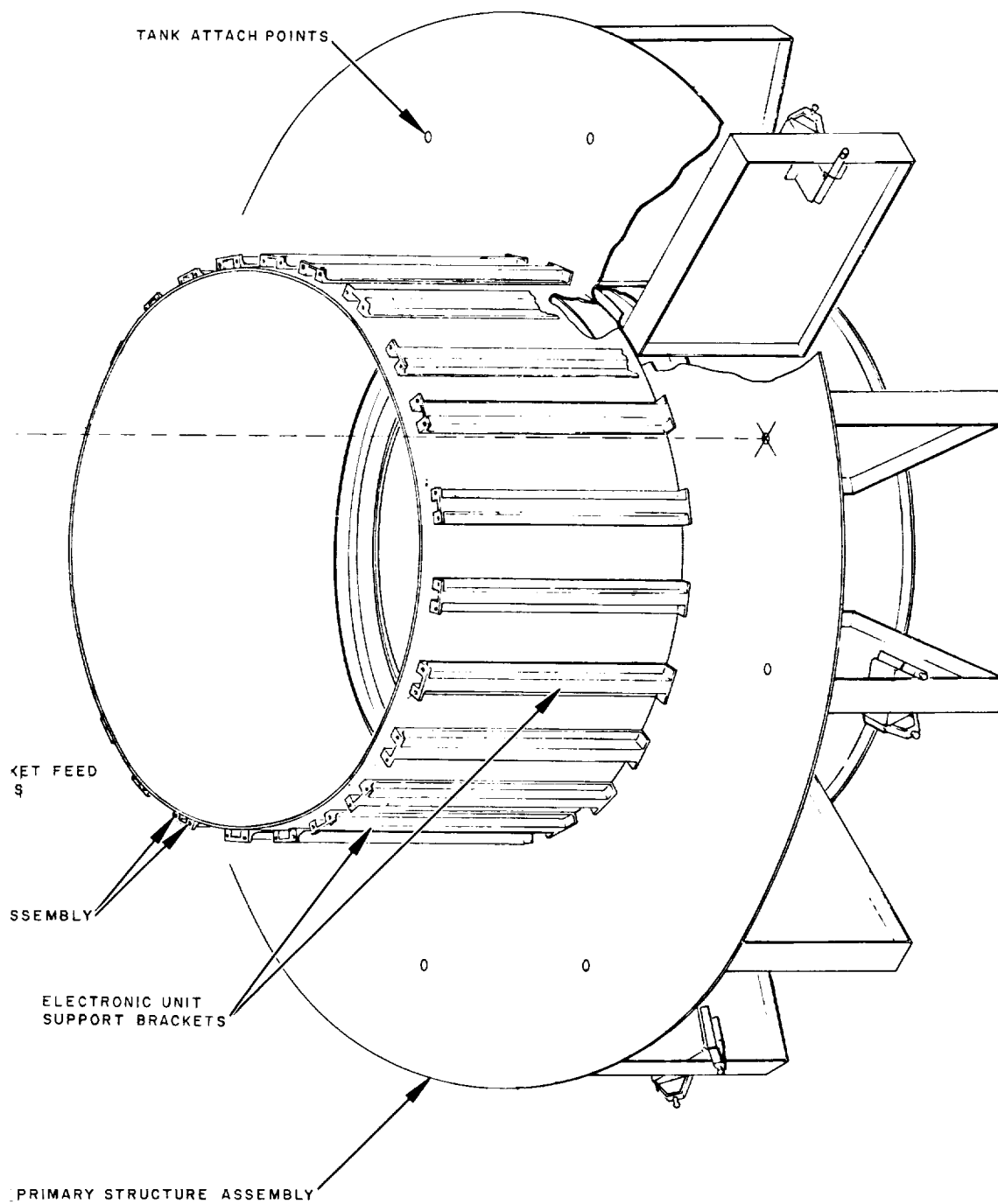
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102. 103. 104. 105. 106. 107. 108. 109. 110. 111. 112. 113. 114. 115. 116. 117. 118. 119. 120. 121. 122. 123. 124. 125. 126. 127. 128. 129. 130. 131. 132. 133. 134. 135. 136. 137. 138. 139. 140. 141. 142. 143. 144. 145. 146. 147. 148. 149. 150. 151. 152. 153. 154. 155. 156. 157. 158. 159. 160. 161. 162. 163. 164. 165. 166. 167. 168. 169. 170. 171. 172. 173. 174. 175. 176. 177. 178. 179. 180. 181. 182. 183. 184. 185. 186. 187. 188. 189. 190. 191. 192. 193. 194. 195. 196. 197. 198. 199. 200.

201. 202. 203. 204. 205. 206. 207. 208. 209. 210. 211. 212. 213. 214. 215. 216. 217. 218. 219. 220. 221. 222. 223. 224. 225. 226. 227. 228. 229. 230. 231. 232. 233. 234. 235. 236. 237. 238. 239. 240. 241. 242. 243. 244. 245. 246. 247. 248. 249. 250. 251. 252. 253. 254. 255. 256. 257. 258. 259. 260. 261. 262. 263. 264. 265. 266. 267. 268. 269. 270. 271. 272. 273. 274. 275. 276. 277. 278. 279. 280. 281. 282. 283. 284. 285. 286. 287. 288. 289. 290. 291. 292. 293. 294. 295. 296. 297. 298. 299. 300.

301.



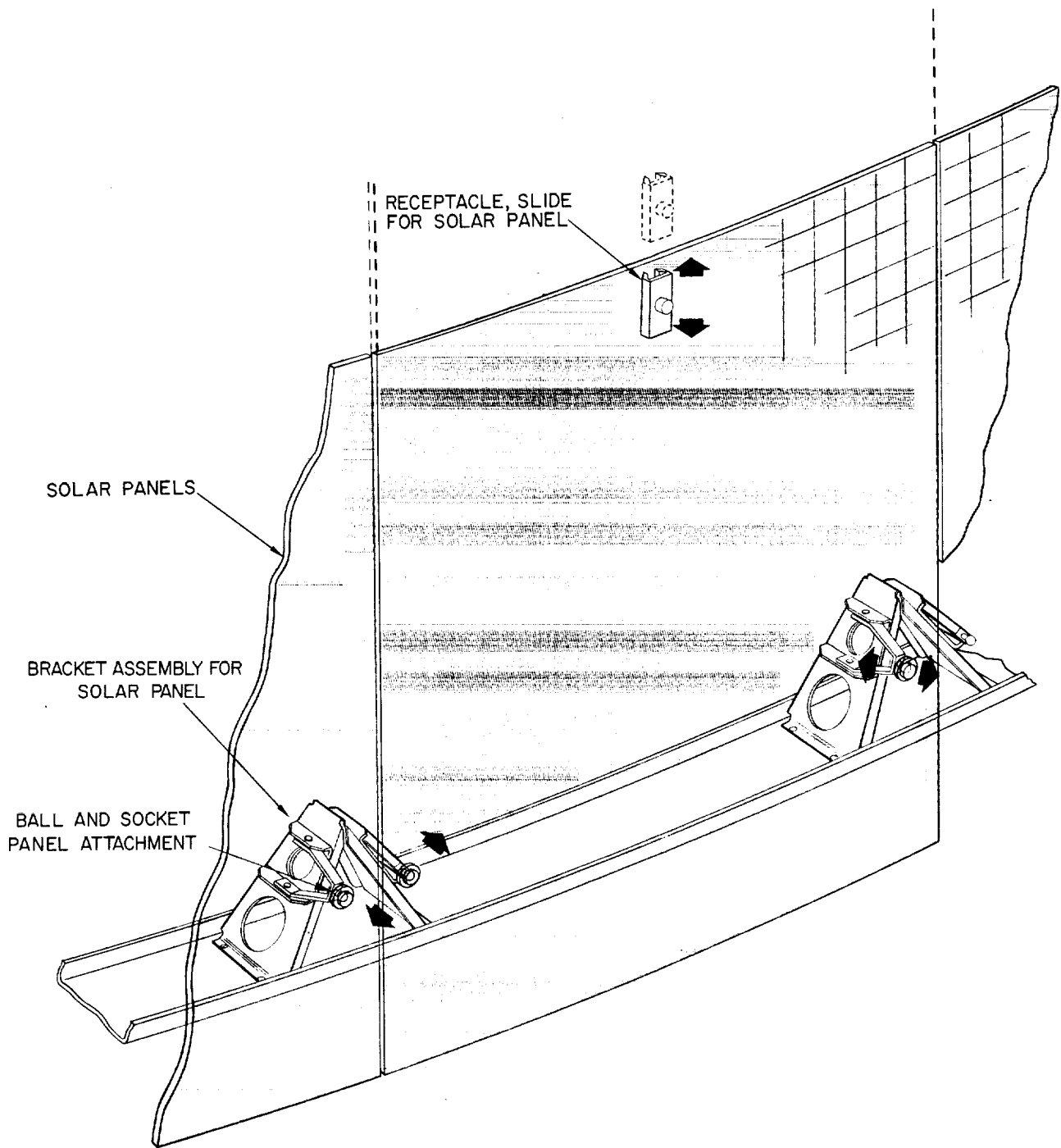


FIGURE 5-201. SOLAR PANEL SUPPORT SYSTEM

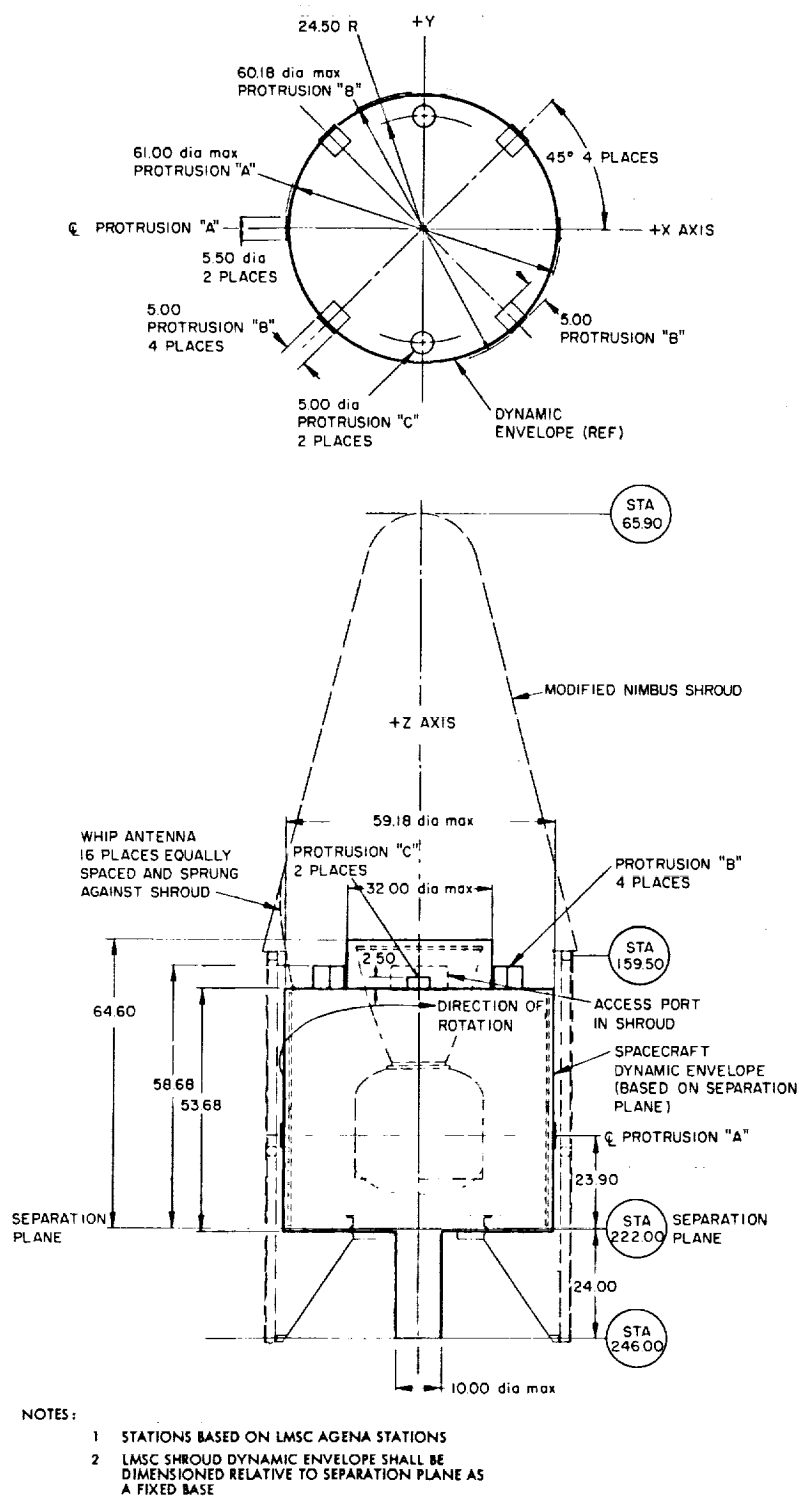


FIGURE 5-202. AGENA SHROUD INTERFACE AND DYNAMIC ENVELOPE

5. Spacecraft System Design

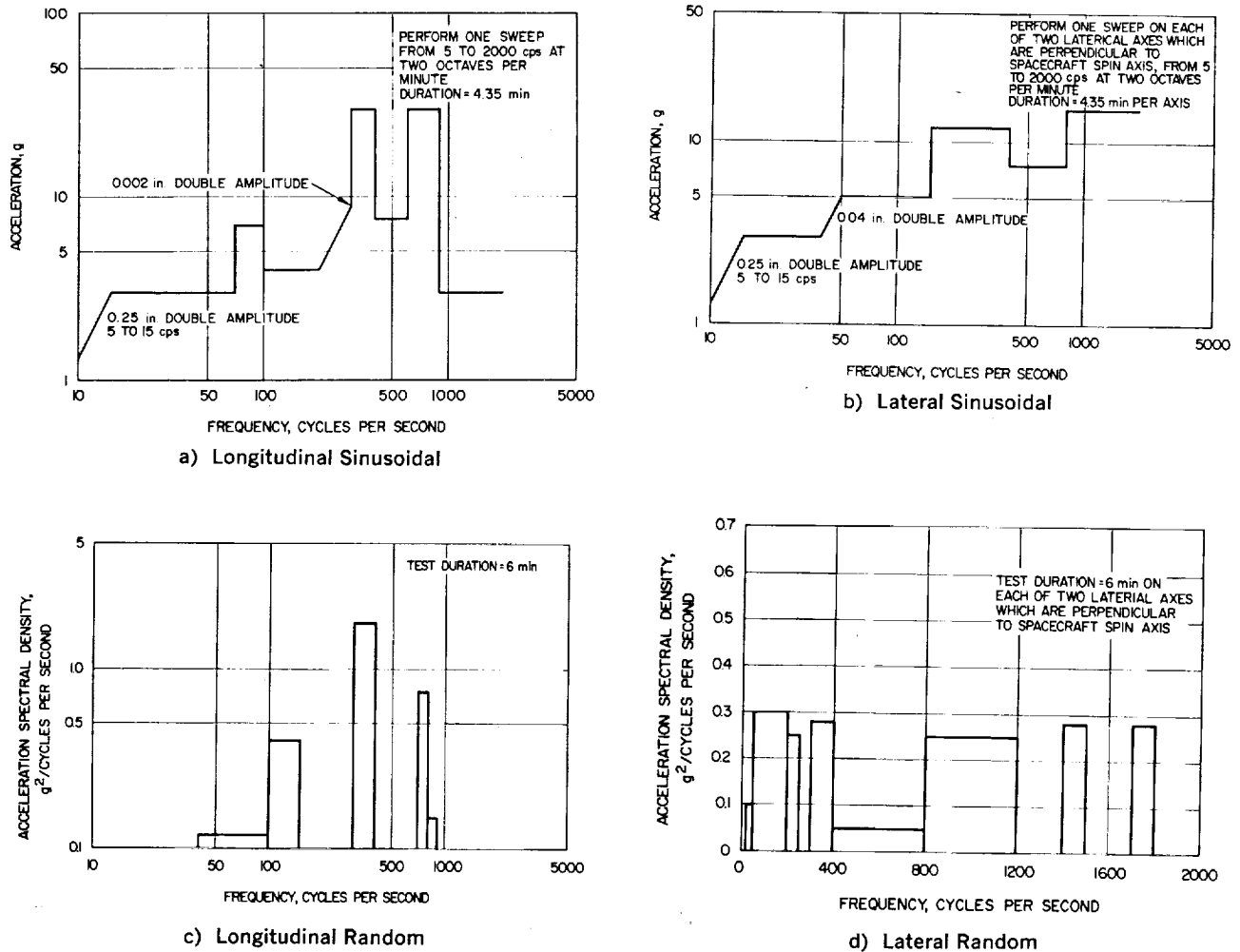


FIGURE 5-203. QUALIFICATION TEST SPECTRA
Antenna component

Figure 5-211 shows the panel installed on the shaker. The panel is attached to the test fixture at the same location as on the spacecraft using proper supports. The mode shapes were determined for the first three resonances by monitoring the output of an accelerometer mounted successively at various points on the concave side of the panel. Figure 5-212 shows the node line patterns for the first three frequencies.

Optimum locations for strain gages and accelerometers have been determined for the forth-

coming qualification tests based on the results of these modal surveys. The strain gage and accelerometer locations which will be used in the qualification tests are shown in Figure 5-212. Qualification level tests will be performed in the lateral (perpendicular to plane of panel) direction only, using the input levels derived from T-1 vibration test data. The test spectra are given in Figure 5-207.

Solar Panel Acoustic Test. An acoustic test of the solar panel is planned in which the panel

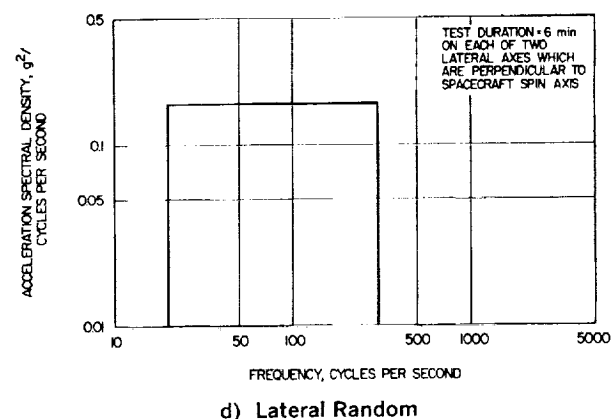
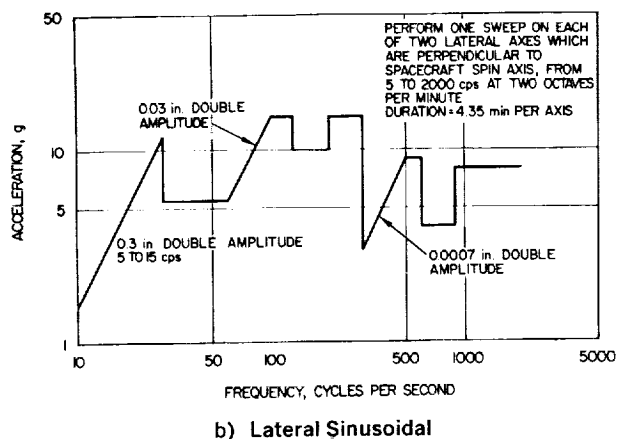
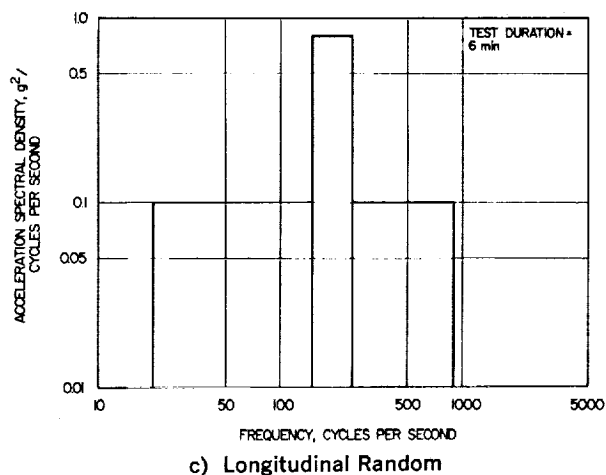
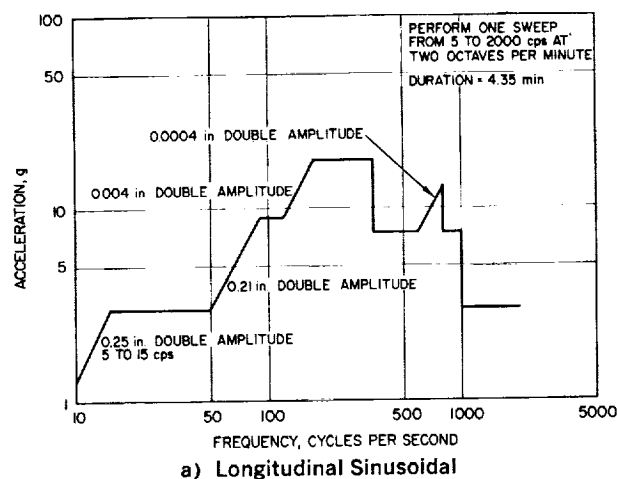


FIGURE 204. QUALIFICATION TEST SPECTRA
Reaction control system — axial jet

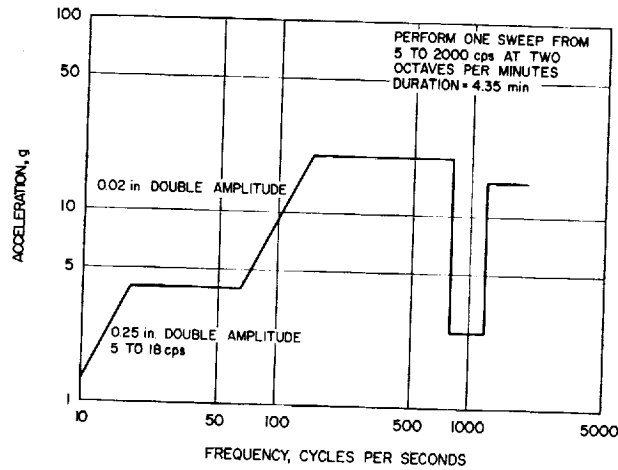
will be subjected to 150 db (re 0.0002 dynes/cm²) for 13.5 minutes. The environment will consist of sinusoidal tones, swept logarithmically at 0.6 octave per minute from 40 cps to 10,000 cps directed at normal incidence to the panel. Arrangements are being completed with North American Aviation in El Segundo for the use of their acoustic test facility.

Structural Analysis. Structural analysis effort in this report period has included prelim-

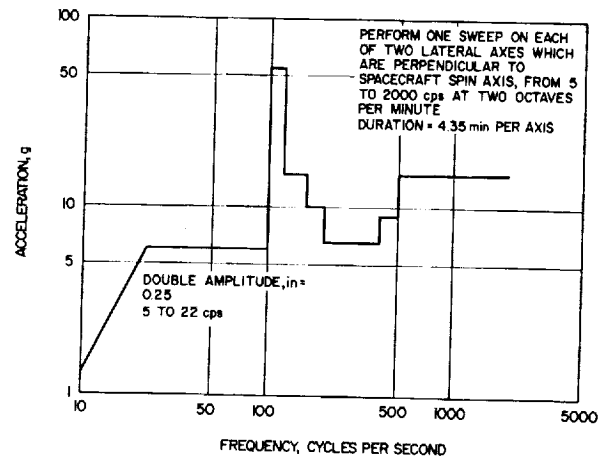
inary stiffness comparisons of the T-1 and T-3 thrust tubes*, analysis of the new solar panel configuration and its associated hardware in preparation for developmental vibration tests, and the stress analysis required for support of the structural design effort. Loads used for the analysis are based on results of the T-1 developmental

*Advanced Syncom Summary Report, August 1963, p. 4-35.

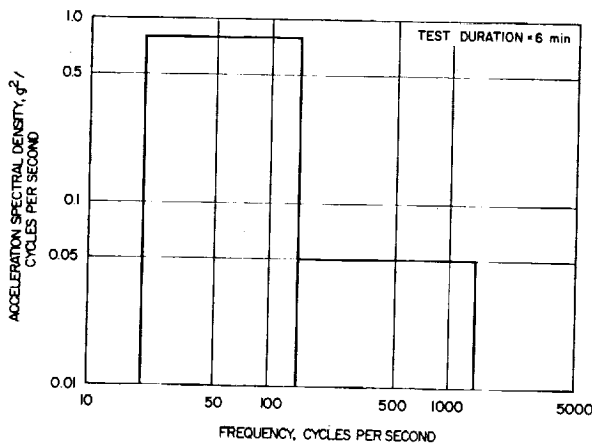
5. Spacecraft System Design



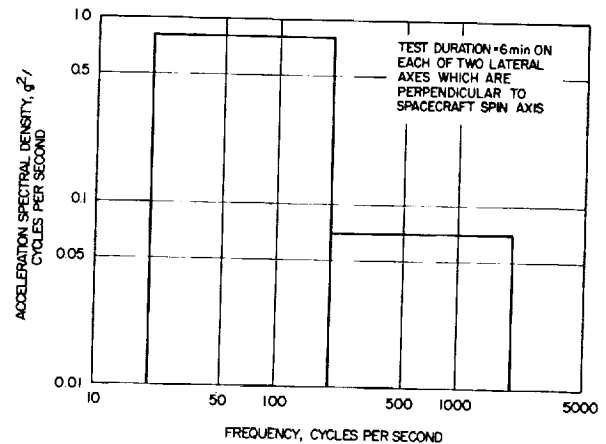
a) Longitudinal Sinusoidal



b) Lateral Sinusoidal



c) Longitudinal Random



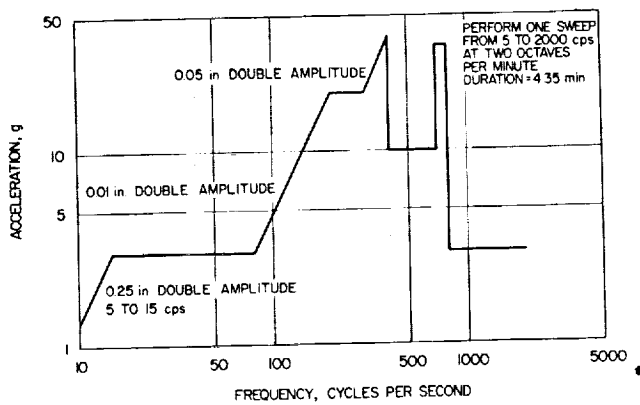
d) Lateral Random

FIGURE 5-205. QUALIFICATION TEST SPECTRA
Fueled bipropellant tank

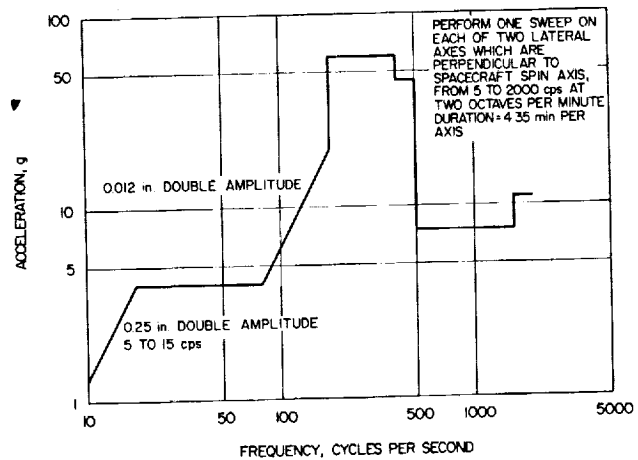
TABLE 5-40. MARGINS OF SAFETY

| Drawing Number | Critical Stress | Margin of Safety |
|----------------|--------------------------|------------------|
| X209875 | Local crippling cylinder | +0.09 |
| X209813 | Buckling flat sheet | +0.36 |
| X209788 | Shear fastener | +1.45 |
| X209788 | Shear tube | +0.61 |
| X209851 | Buckling | +0.26 |
| X209849 | Compression | High |
| X209848 | Shear | +3.39 |
| X209871 | Local crippling | +0.25 |
| X209868 | Buckling | +0.05 |

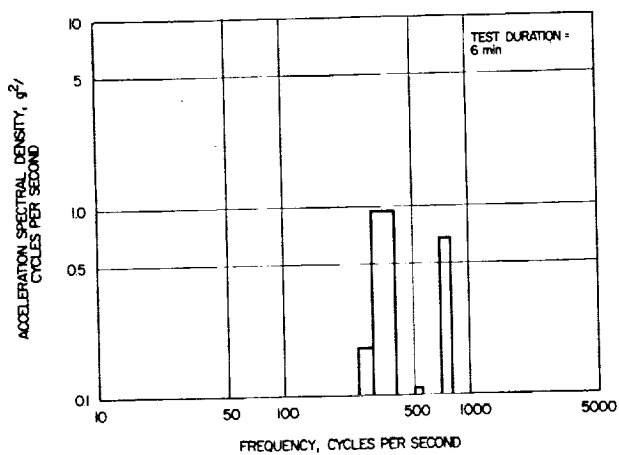
vibration test. An additional constraint considered in the sizing of members was stiffness. Wherever possible, support members were sized such that the component fundamental frequency was not coincident with that of the spacecraft. A summary of margins of safety is presented in Table 5-40. A view showing the structural analysis model is shown on the following page.



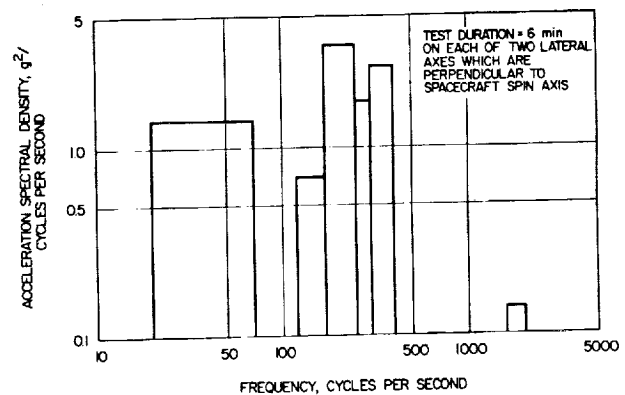
a) Longitudinal Sinusoidal



b) Lateral Sinusoidal

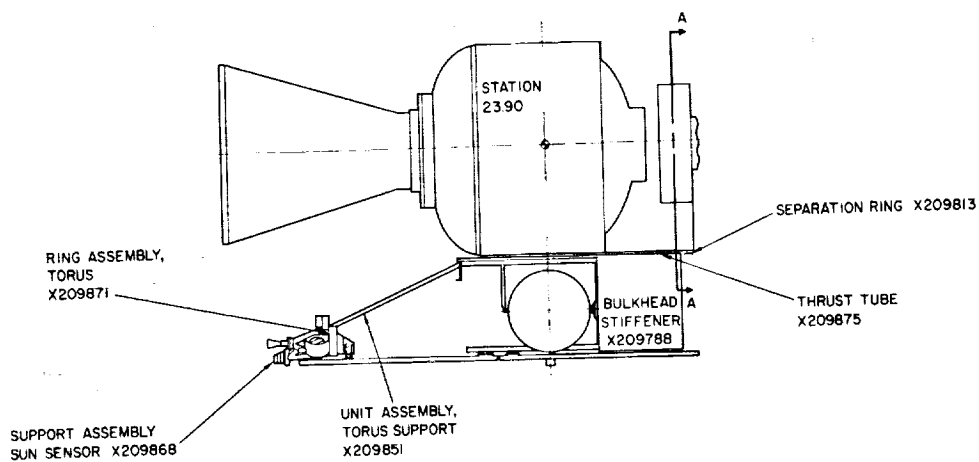


c) Longitudinal Random

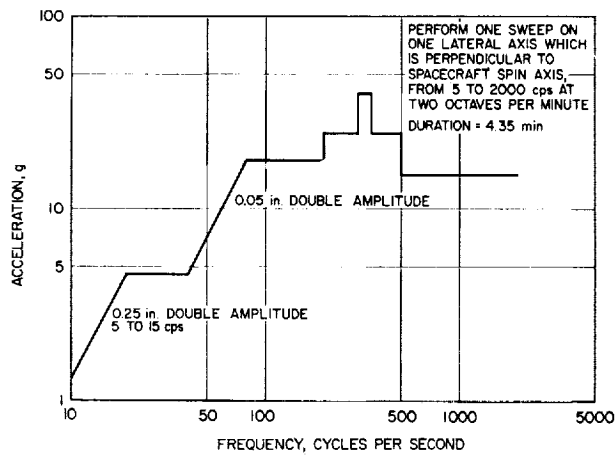


d) Lateral Random

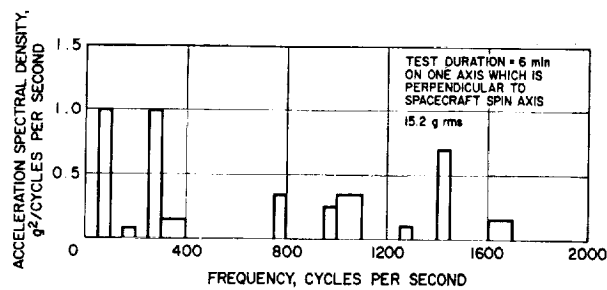
FIGURE 5-206. QUALIFICATION TEST SPECTRA
Rib-mounted components



5. Spacecraft System Design



a) Lateral Sinusoidal



b) Lateral Random

FIGURE 5-207. QUALIFICATION TEST SPECTRA
Solar panel

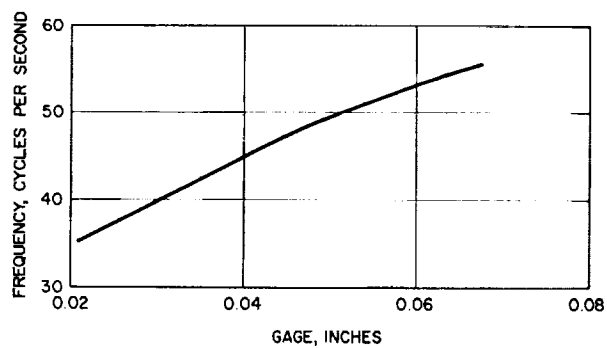


FIGURE 5-208. FUNDAMENTAL LATERAL FREQUENCY
VERSUS GAGE OF THRUST TUBE

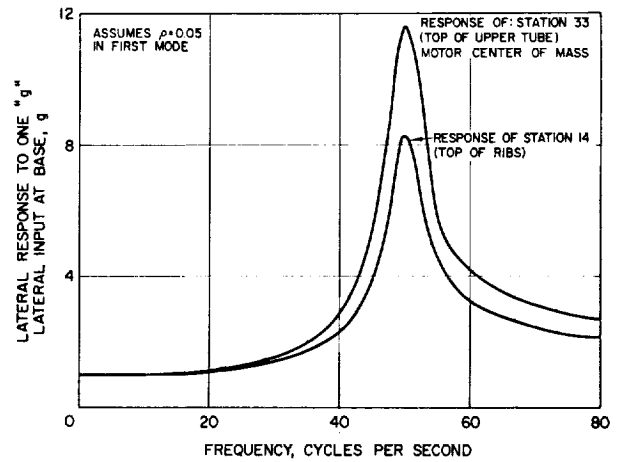


FIGURE 5-209. LATERAL RESPONSE OF T-2
ANALYTICAL MODEL

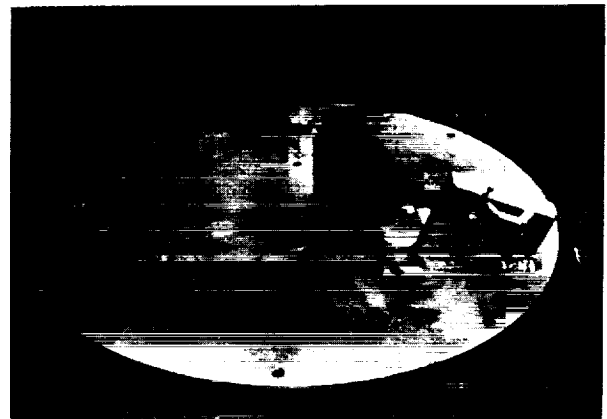


FIGURE 5-210. SOLAR PANEL VIBRATION
TEST FIXTURE



FIGURE 5-211. SOLAR PANEL VIBRATION
TEST SETUP

• Thrust Tube, Aft (Drawing No. X209-875)

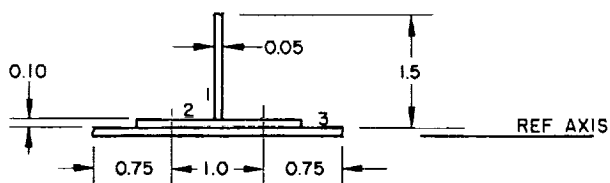
Material

7075-T6 aluminum alloy 2014-T651

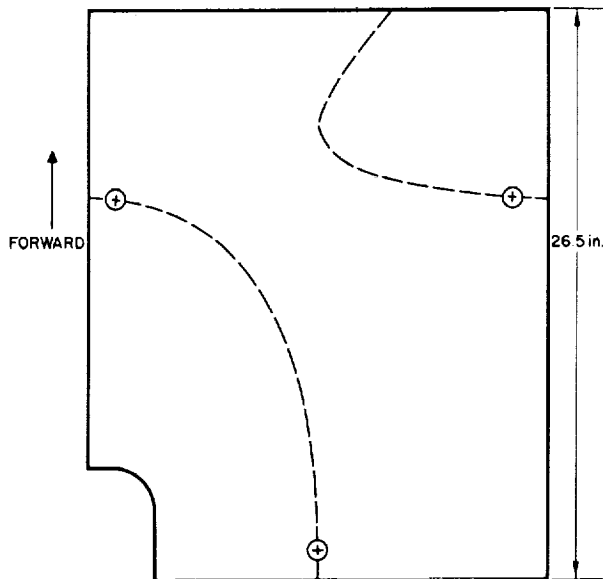
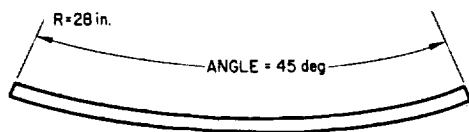
QQ-A-287 QQ-A-255

Section Properties (section through thrust tube and rib)

Assume 30t effective skin in compression



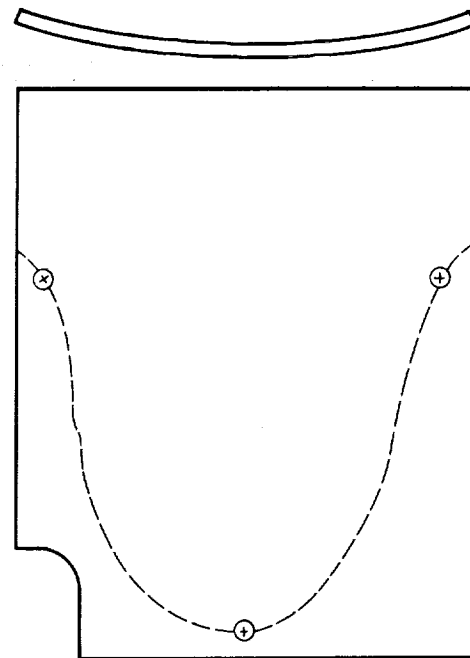
DIMENSIONS IN INCHES



- ⊕ PANEL MOUNT POINTS
- NODE LINES
- ⊙ ACCELERATION
- ▬ GAGE AND DIRECTION

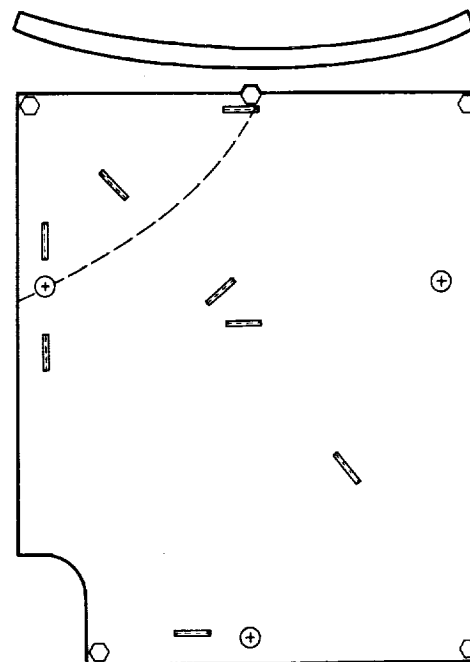
a) Fundamental Pattern; $f_1 = 67.5$ cps

FORWARD



b) Second Resonance Pattern; $f_2 = 86.5$ cps

FORWARD



c) Third Resonance Pattern; $f_3 = 123.7$ cps

FIGURE 5-212. NODE PATTERNS

5. Spacecraft System Design

| Item | | A | y | Ay | Ay ² | I _{ox} |
|------|--------------|--------------|-------|---------------|-----------------|-----------------|
| 1 | 1.4 × .050 | 0.070 | 0.85 | 0.0595 | 0.0505 | 0.0114 |
| 2 | 1.812 × .100 | 0.181 | 0.100 | 0.0181 | 0.0018 | — |
| 3 | 2.5 × .050 | 0.125 | 0.025 | 0.0031 | — | — |
| | | <u>0.376</u> | | <u>0.0807</u> | <u>0.0523</u> | <u>0.0114</u> |

$$\bar{y} = 0.214 \quad A\bar{y}^2 = 0.0173 \quad 0.0523$$

$$+ 0.0114$$

$$0.0637$$

$$- 0.0173$$

$$I_x = Ay^2 + I_{ox} - A\bar{y}^2 = 0.0464 \text{ in}^4$$

$$\rho = \sqrt{I/A}$$

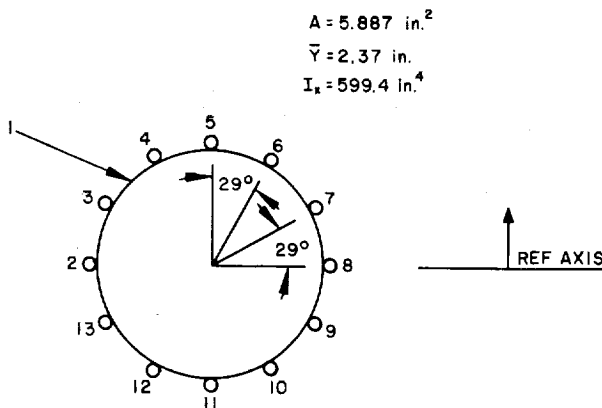
$$= \sqrt{0.0464/0.376}$$

$$= 0.35$$

Total Effective Compression Area

$$A = 12 \times 0.376 = 4.5 \text{ in}^2$$

Effective Bending Section



Assumptions

- 1) Entire tension side is effective
rib area = 0.251 in^2
- 2) Effective area in compression
 $A = 0.376 \text{ in}^2$
 $\bar{y} = 0.214 \text{ inch}$

| Item | | A | y | Ay | Ay ² | I _{ox} |
|------|---------------------------------|--------------|--------|--------------|-----------------|-----------------|
| 1 | $\pi \times 14.34 \times 0.050$ | 2.250 | 9.14 | 20.6 | 188.0 | 43.5 |
| 2 | | 0.251 | | | | |
| 3 | | 0.251 | 7.11 | 1.785 | 12.7 | |
| 4 | | 0.251 | 12.80 | 3.22 | 41.2 | |
| 5 | | 0.251 | 14.67 | 3.68 | 54.0 | |
| 6 | | 0.251 | 12.80 | 3.22 | 41.2 | |
| 7 | | 0.251 | 7.11 | 1.785 | 12.7 | |
| 8 | | 0.251 | | | | |
| 9 | | 0.376 | -7.05 | -2.65 | 18.7 | |
| 10 | | 0.376 | -12.71 | -4.79 | 60.9 | |
| 11 | | 0.376 | -14.58 | -5.48 | 79.9 | |
| 12 | | 0.376 | -12.71 | -4.79 | 60.9 | |
| 13 | | 0.376 | -7.05 | -2.65 | 18.7 | |
| | | <u>5.887</u> | | <u>13.93</u> | <u>588.9</u> | <u>43.5</u> |

$$\bar{y} = 2.37 \quad A\bar{y}^2 = 33.0 \quad I_x = 588.9$$

$$43.5$$

$$632.4$$

$$33.0$$

$$599.4 \text{ in}^4$$

Applied Loads

Assume 30 g lateral or longitudinal load at spacecraft cg

$$P = 1518 \times 30 = 45,500 \text{ lb}$$

Moment due to lateral load

$$M = 45,500 \times (23.90 - 3.5)$$

$$= 927,000 \text{ in-lb}$$

Calculated Stresses

$$f_c = \frac{P}{A} = \frac{45,500}{4.5} = 10,100 \text{ psi}$$

$$f_b = \frac{Mc}{I} = \frac{927,000 \times (14.34 + 2.37)}{599.4}$$

$$= 25,900 \text{ psi}$$

Allowable Stresses

$$F_{\text{COLUMN}} = \frac{\pi^2 E}{(L/C)^2} = \frac{\pi^2 \times 10.5 \times 10^6}{(10/.35)^2} > F_{cy}$$

$$\frac{b_{\text{avg}}}{t_{\text{avg}}} > \frac{1.203}{0.075} = 16.1$$

$$F_{cc} = 34,000 \times \sqrt{\frac{61,000}{67,000}}$$

$$= 28,200 \text{ psi (see Figure 5-213)}$$

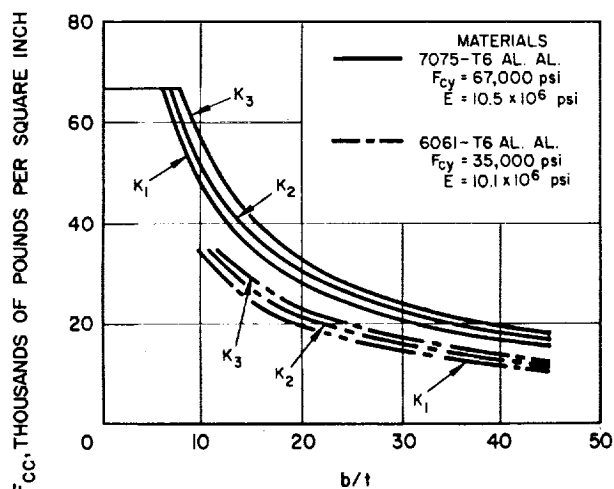


FIGURE 5-213. ALLOWABLE CRIPPLING STRESS
VERSUS b/t

$$F_{cc} = \frac{K}{b \cdot 0.75} \sqrt{F_{cy} E}$$

Edge conditions:

$K_1 = 0.316$ free-free
 $K_2 = 0.342$ fixed-free
 $K_3 = 0.366$ fixed-fixed

Margin of Safety

$$MS = \frac{28,200}{25,900} - 1 = +0.09$$

- Ring, Separation (Drawing No. X209813)

Material

2014-T652 aluminum alloy

QQ-A-367

Section Properties

$$\begin{aligned} A &= 2\pi R t. \\ &= 2 \times \pi \times 14.41 \times 0.085 = 7.69 \text{ in}^2 \\ I &= \pi R^3 t \\ &= \pi \times 14.41^3 \times 0.085 = 800 \text{ in}^4 \end{aligned}$$

Applied Loads

Assume 30g lateral or longitudinal load at spacecraft cg

$$P = 1518 \times 30 = 45,500 \text{ pounds}$$

Moment Due to Lateral Load

$$M = 45,500 \times 23.9 = 1,090,000 \text{ in-lb}$$

Calculated Stresses

$$f_c = \frac{P}{A} = \frac{45,500}{7.69} = 5920 \text{ psi}$$

$$f_B = \frac{M_C}{I} = \frac{1,090,000 \times 14.41}{800}$$
$$= 19,600 \text{ psi}$$

Allowable Stresses

$$\begin{aligned} \frac{F_C}{n} &= \frac{0.871 E}{(R/t)^m (L/R)^n} \\ &= \frac{0.871 \times 10.5 \times 10^6}{(14.41/0.085)^{1.275} (3.9/14.41)^{.55}} \\ &= 26,800 \text{ psi} \end{aligned}$$

(For 99 percent Probability)

Margin of Safety

$$MS = \frac{26,800}{19,600} - 1 = +0.36$$

- Rib Bulkhead Stiffener (Drawing No. X209788)

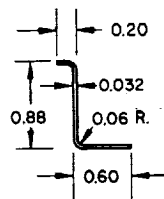
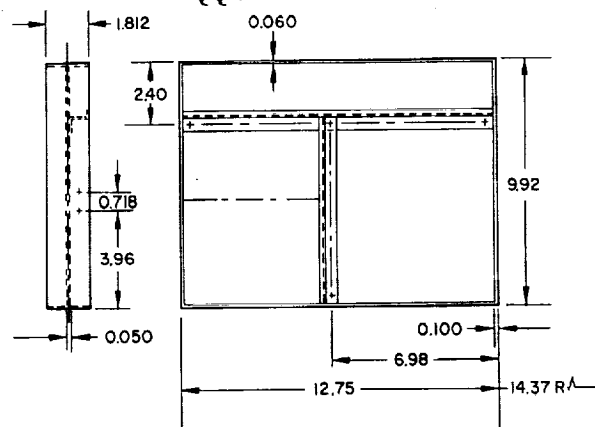
Material

Stiffeners: 7075-T6 AL AL

QQ-A-287

Basic rib: 2014-T651 AL AL

QQ-A-255



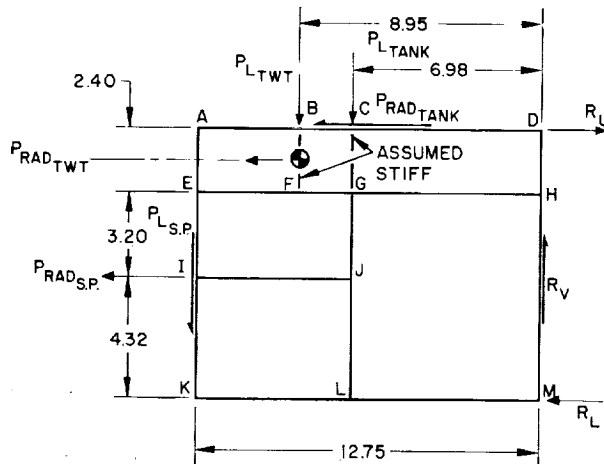
DIMENSIONS IN INCHES

5. Spacecraft System Design

Applied loads, longitudinal

Load factor $n_z = \text{dynamic} + \text{zero quasi-static}$

Load factor $n_{RAD} = 6.7R$ (feet)



DIMENSIONS IN INCHES

$$P_{L \text{ TANK}} = 28.5 \text{ lb.} \times 27.0 = 769 \text{ pounds}$$

$$P_{RAD. \text{ TANK}} = (28.5 \text{ lb.} \times 6.7 \text{ g/feet} \times 1.78 \text{ feet}) \frac{1}{2} = 170 \text{ pounds (spin)}$$

$$P_{L \text{ TWT}} = 2.50 \text{ lb.} \times 20.0 = 50 \text{ pounds}$$

$$P_{RAD. \text{ TWT}} = (2.50 \text{ lb.} \times 6.7 \text{ g/feet} \times 1.94 \text{ feet}) = 32.50 \text{ pounds (spin)}$$

$$P_{L \text{ SOL. PAN}} = (3.60 \text{ lb} \times 14.0) \frac{1}{2} = 25.2 \text{ pounds}$$

$$P_{RAD. \text{ S.P.}} = (6.70 \text{ g/feet} \times 2.24 \text{ feet} \times 3.60 \text{ pounds}) \frac{1}{3} = 18.0 \text{ pounds (spin)}$$

$$R_V = 25.2 + 50.0 + 769 = 844.2 \text{ pounds}$$

$$\Sigma M_M = 9.92R_U - 12.75 \times 25.2 - 8.95 \times 50.0 - 6.98 \times 769$$

$$- 8.72 \times 32.50 - 4.32 \times 18.0 - 9.92 \times 170 = 0$$

$$= 9.92R_U - 321.0 - 448.0 - 5370 - 285.0$$

$$- 78.0 - 1685.0 = 9.92R_U - 8247.0 = 0$$

$$R_U = 832.0 \text{ pounds}$$

$$\Sigma M_D = 9.92R_L - 8.95 \times 50.0 - 6.98 \times 769 - 12.75 \times 25.2$$

$$+ 1.20 \times 32.50 + 5.60 \times 18.0 = 0$$

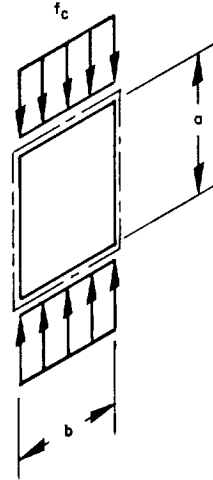
$$= 9.92R_L - 448.0 - 5370.0 - 321.0 + 39.0$$

$$+ 101.0 = 9.92R_L - 5797.0 = 0$$

$$R_L = 585.0 \text{ pounds}$$

Assumed Stiffener Check

Assume pressure tank loads are reacted by a plate $2.0 \times 2.40 \times 0.050$, simply supported each edge.



$$a = 2.40 \text{ inches, } b = 2.0 \text{ inches, } t = 0.050 \text{ inches.}$$

$$f_c = \frac{P_{L \text{ tank}}}{b \times t} = \frac{769}{2.0 \times 0.05} = 7690 \text{ psi}$$

Allowable Stress

$$\frac{a}{b} = \frac{2.40}{2.0} = 1.20$$

$$K = 3.62 \text{ (Reference 5-1, page 372)}$$

$$F_{CCR} = K E \left(\frac{t}{b} \right)^2$$

$$F_{tu} = 65,000 \text{ psi for 2014 = T6}$$

$$F_{ty} = 59,000 \text{ psi}$$

$$F_{cy} = 61,000 \text{ psi}$$

$$E = 10.5 \times 10^6 \text{ psi}$$

$$E_c = 10.7 \times 10^6 \text{ psi}$$

$$F_{CCR} = 3.62 \times 1.07 \times 6.25 \times 10^8 = 24,200 \text{ psi}$$

(Reference 5-2, page 3.2.1.0(b))

Margin of Safety

$$MS = \frac{24.2}{7.69} - 1 = 2.15$$

Shear Panel Check

$$q_{MH} = q_{MD} = \frac{R_v}{h} = \frac{844.2}{9.92} = 85.0 \text{ lb/in}$$

$$f_s = \frac{85.0}{0.050} = 1700 \text{ psi}$$

Allowable Stress

Panel: MLGH $7.52 \times 6.98 \times 0.050$
(see figure above)

$$\frac{a}{b} = \frac{7.52}{6.98} = 1.075$$

$$K_s = 7.6$$

$$F_{sCR} = K_s E \left(\frac{t}{b} \right)^2 \quad E_c = 1.07 \times 10^7 \text{ psi}$$

$$F_{sCR} = 7.6 \times 1.07 \times 5.13 \times 10^2 = 4170 \text{ psi}$$

(Reference 5-1, page 394)

Margin of Safety

$$MS = \frac{4170}{1700} - 1 = 1.45$$

Flange-Crippling Check

Assume R_U as a compressive load.

$$A_{FLG} = 1.812 \times 0.060 = 0.1086 \text{ in}^2$$

$$R_U = 834.0 \text{ pounds}$$

$$f_c = \frac{834.0}{0.1086} = 7770.0 \text{ psi}$$

Allowable Stress

$$\frac{b}{t} = \frac{0.876}{0.060} = 14.6 \text{ for } K_2$$

$$F_{CC} = 38,500 \text{ psi for 7075-T6}$$

$$F_{CY} = 67,000 \text{ psi 7075-T6}$$

$$F_{CY} = 61,000 \text{ psi 2014-T6}$$

$$F_{CC2} = F_{CC1} \sqrt{\frac{F_{CY2}}{F_{CY1}}} = 38,500 \sqrt{0.910}$$

$$= 38,500 \times 0.954$$

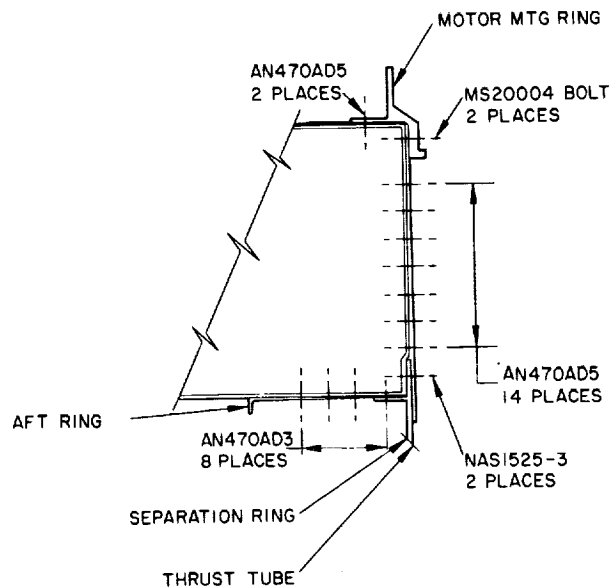
$$F_{CC2} = 36,700 \text{ psi}$$

(Figure 5-213)

Margin of Safety

$$MS = \frac{36.7}{7.77} - 1 = 3.72$$

Rib-Thrust Tube Attachment



Assume upper cap load (R_U) is reacted by mounting ring through the two AD5 rivets.

Assume lower cap load (R_L) is reacted by aft ring and separation ring by eight AD3 rivets.

Shear (R_V) reacted by thrust tube by 14 AD5 rivets.

Upper Cap

$$R_U = 832 \text{ pounds}$$

$$\frac{\text{Shear}}{\text{Rivet}} = \frac{832}{2} = 416 \frac{\text{Pounds}}{\text{Rivet}}$$

Lower Cap

$$\text{Assume } R_L = R_U = 832 \text{ pounds}$$

$$\frac{\text{Shear}}{\text{Rivet}} = \frac{832}{8} = 104.0 \frac{\text{pounds}}{\text{rivet}}$$

Thrust Tube

$$R_V = 844.2 \text{ pounds}$$

$$\frac{\text{Shear}}{\text{Rivet}} = \frac{844.2}{14} = 60.3 \frac{\text{pounds}}{\text{rivet}}$$

5. Spacecraft System Design

Allowables

AN470AD3

$$\text{Shear} = 247 \frac{\text{pound}}{\text{rivet}}$$

Bearing 0.020 7075-T6 sheet

(Reference 5-2, page 8.1.1.1.1(c))

$$\text{Bearing} = 192 \times 1.44 = 277 \frac{\text{pound}}{\text{rivet}}$$

(Reference 5-2, page 8.1.1.1.1(e))

Bearing 0.050 2014-T6

$$\text{Bearing} = 480 \times 1.29 = 619 \frac{\text{pound}}{\text{rivet}}$$

AN470AD5

$$\text{Shear} = 675 \frac{\text{pound}}{\text{rivet}}$$

Bearing 0.050 7075-T6 sheet

$$\text{Bearing} = 795 \times 1.44 = 1145 \frac{\text{pound}}{\text{rivet}}$$

Margin of Safety

Upper Cap

$$\text{MS} = \frac{675}{416} - 0.1 = 0.61$$

Lower Cap

$$\text{MS} = \frac{247}{104.0} - 1 = 1.36$$

Thrust Tube

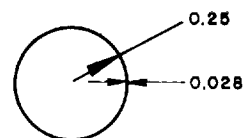
$$\text{MS} = \frac{675}{60.3} - 1 > 10$$

- Unit Assembly, Torus Support, Torus Ring
(Drawing No. X209851)

Material

2024-T4 aluminum alloy
QQ-A-268

Section Properties



DIMENSIONS IN INCHES

$$A = 2 \pi R t$$

$$= 2 \times \pi \times 0.236 \times 0.028 = 0.0415 \text{ in}^2$$

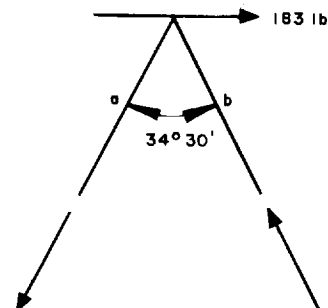
$$I = \pi R^3 t$$

$$= \pi \times 0.236^3 \times 0.028 = 0.001155 \text{ in}^4$$

$$\rho = \sqrt{\frac{I}{A}} = \sqrt{\frac{0.001155}{0.0415}} = 0.167$$

$$L/\rho = 17.5/0.167 = 105.0$$

Applied Loads



Assume 30 percent of total lateral load is resisted at one bi-pod

$$P = 0.30 \times 610.2 = 183 \text{ pounds}$$

$$a = b$$

$$a \sin 17^\circ 15' + b \sin 17^\circ 15' = 183$$

$$2 a \sin 17^\circ 15' = 183$$

$$a = 310 \text{ pounds}$$

Calculated Stresses

$$f_c = \frac{P}{A} = \frac{310}{0.0415} = 7460 \text{ psi}$$

$$P/\text{Attach} = 310/2 = 155 \text{ pounds}$$

Allowable Stresses

$$F_C = \frac{\pi^2 E}{(L/\rho)^2} = 9400 \text{ psi}$$

P_s for AD3 rivet = 217 pounds

(Reference 5-2, page 8.1.1.1.c)

Margins of Safety

$$MS_{\text{TUBE}} = \frac{9400}{7460} - 1 = + 0.26$$

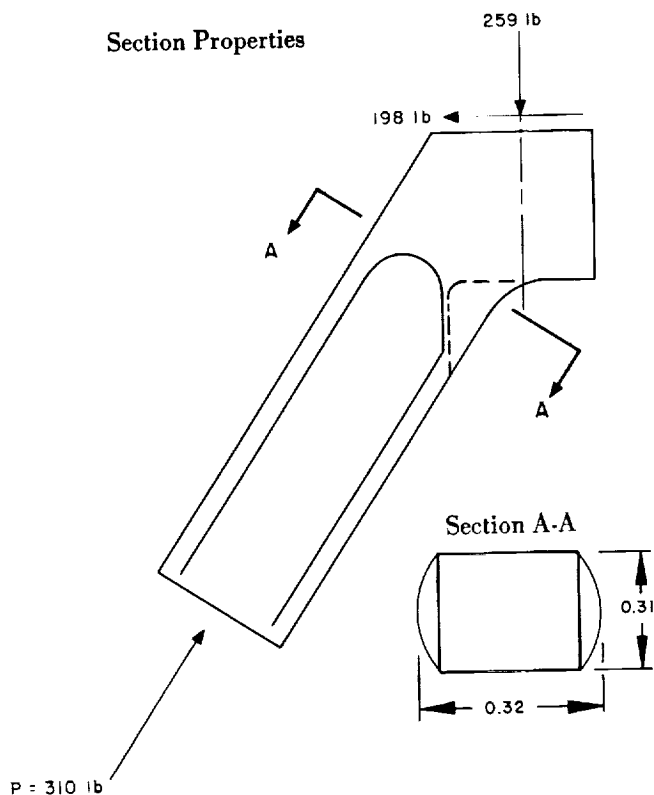
$$MS_{\text{RIVET}} = \frac{217}{155} - 1 = + 0.40$$

- Fitting Lower Truss Subassembly, Torus Ring
(Drawing No. X209849)

Material

AZ60-A magnesium alloy
QQ-M-31

Section Properties



$$A = 0.32 \times 0.31 = 0.099 \text{ inch}^2$$

Calculated Stress

$$f_C = \frac{P}{A} = \frac{310}{0.099} = 3130 \text{ psi}$$

Allowable Stress

$$F_{CY} = 30,000 \text{ psi} \quad (\text{Reference 5-3})$$

Margin of Safety

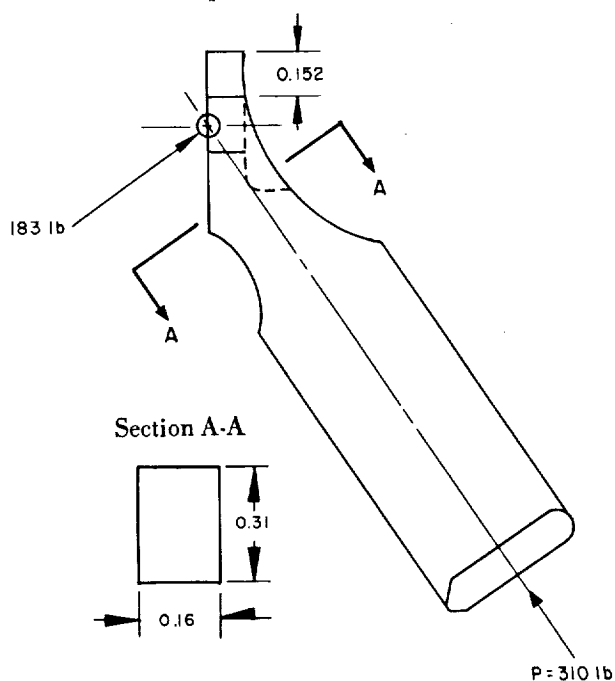
$$MS = \frac{30,000}{3130} - 1 = \text{high}$$

- Fitting, Upper Truss Subassembly
(Drawing No. X209848)

Material

AZ60-A magnesium alloy
QQ-M-31

Section Properties



DIMENSIONS IN INCHES

5. Spacecraft System Design

Internal loads obtained by method presented in Reference 5-4, page 319.

Internal loads presented in Table 5-42.

$$A = 0.31 \times 0.16 = 0.0495 \text{ in}^2$$

Calculated Stresses

$$f_c = \frac{P}{A_L} = \frac{310}{0.0495} = 6260 \text{ psi}$$

$$f_s = \frac{P}{A_s} = \frac{*183}{0.2 \times 0.152 \times 0.120} = 5020 \text{ psi}$$

Allowable Stresses

$$F_{CY} = 0.39000 \text{ psi}$$

$$F_{SU} = 32000 \text{ psi} \quad (\text{Reference 5-3})$$

Margins of Safety

$$MS_{COMP} = \frac{30,000}{6260} - 1 = +3.8$$

$$MS_{SHEAR} = \frac{22,000}{5020} - 1 = +3.39$$

- Ring Assembly, Torus

Material

2024-T4 sheet aluminum alloy
QQ-A-355

Loads

Longitudinal loading condition.

Applied loads presented in Table 5-41.

*Use of this load for shear tearout of lug is conservative.

TABLE 5-41. APPLIED LOADS
All loads applied act forward longitudinally

| Load | Point of Application | Load Magnitude, pounds | Load | Point of Application | Load Magnitude, pounds | Notation |
|-----------------|----------------------|------------------------|-----------------|----------------------|------------------------|--|
| P ₁ | t ₁ | 33.6 | P ₁₂ | t ₅ | 33.6 | t _N Solar panels to torus tie |
| P ₂ | A ₁ | 13.2 | P ₁₃ | A ₅ | 13.2 | |
| P ₃ | t ₂ | 33.6 | P ₁₄ | t ₆ | 33.6 | |
| P ₄ | S ₂ | 20.3 | P ₁₅ | S ₄ | 20.3 | A _N Antenna to torus tie |
| P ₅ | A ₂ | 13.2 | P ₁₆ | A ₆ | 13.2 | |
| P ₆ | t ₃ | 33.6 | P ₁₇ | t ₇ | 33.6 | S _N Sun sensor to torus tie |
| P ₇ | E ₁ | 77.3 | P ₁₈ | E ₂ | 77.3 | |
| P ₈ | A ₃ | 13.2 | P ₁₉ | A ₇ | 13.2 | E _N Rocket motor to torus tie |
| P ₉ | t ₄ | 33.6 | P ₂₀ | t ₈ | 33.6 | |
| P ₁₀ | S ₃ | 20.3 | P ₂₁ | S ₁ | 20.3 | |
| P ₁₁ | A ₄ | 13.2 | P ₂₂ | A ₈ | 13.2 | |

TABLE 5-42. INTERNAL LOADS

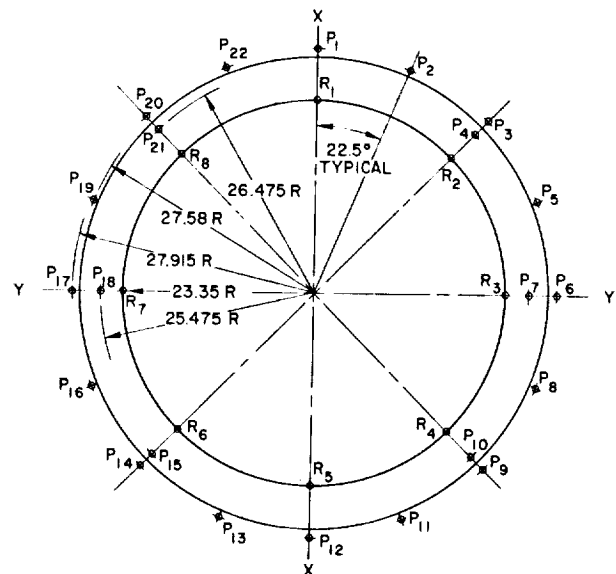
| Structure Section | Shear, pounds | Bending Moment, in-lb | Torsion, in-lb |
|-------------------|---------------|-----------------------|----------------|
| A-A | 17.00 | 161.50 | 131.28 |
| B-B | 31.00 | 1145.00 | 200.43 |

TABLE 5-43. LOADS REACTION
All reactions act aft longitudinally

| Reaction | Reaction Magnitude, pounds | Reaction | Reaction Magnitude, pounds |
|----------------|----------------------------|----------------|----------------------------|
| R ₁ | -76.25 | R ₅ | -76.25 |
| R ₂ | -76.25 | R ₆ | -76.25 |
| R ₃ | -76.25 | R ₇ | -76.25 |
| R ₄ | -76.25 | R ₈ | -76.25 |

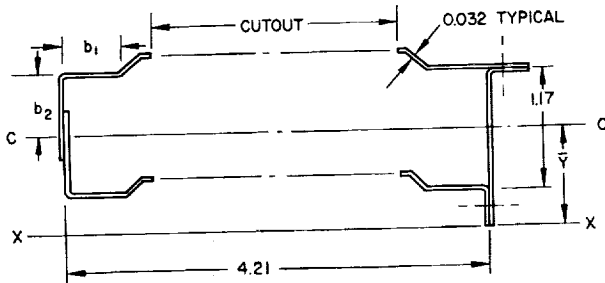
LOADS DIAGRAM

Load and Reaction Points Shown
View Looking Aft



DIMENSIONS IN INCHES

Section Properties



DIMENSIONS IN INCHES

Section A-A (Drawing No. X209871)

$$b = \frac{b_1 + b_2}{2} = \frac{0.621 + 0.60}{2} = 0.6105 \text{ inch}$$

$$t = 0.032 \text{ inch}$$

$$\frac{b}{t} = \frac{0.6105}{0.032} = 19.10$$

(Reference Section A-A, Drawing No. X209871;
Section A-A and B-B are identical)

| Element | b | h | A | Y | Ay | Ay ² | I _o |
|---------|-------|-------|--------|-------|---------|-----------------|----------------|
| 1 | 0.60 | 0.032 | 0.0192 | 0.416 | 0.00798 | 0.00332 | 0.0000016 |
| 2 | 0.032 | 0.844 | 0.027 | 0.822 | 0.02219 | 0.0182 | 0.0016 |
| 3 | 0.032 | 0.844 | 0.027 | 1.208 | 0.0326 | 0.0343 | 0.0016 |
| 4 | 0.60 | 0.032 | 0.0192 | 1.626 | 0.0312 | 0.051 | 0.0000016 |
| 5 | 0.60 | 0.032 | 0.0192 | 1.626 | 0.0342 | 0.051 | 0.0000016 |
| 6 | 0.40 | 0.064 | 0.0256 | 1.598 | 0.041 | 0.0655 | 0.0000087 |
| 7 | 0.032 | 1.20 | 0.038 | 1.03 | 0.0391 | 0.402 | 0.0046 |
| 8 | 0.064 | 0.40 | 0.0256 | 0.20 | 0.00512 | 0.001 | 0.0004 |
| 9 | 0.60 | 0.032 | 0.0192 | 0.416 | 0.00798 | 0.0033 | 0.0000016 |
| | | | 0.220 | 0.570 | 0.218 | 0.273 | 0.0082 |

$$\bar{Y} = \frac{\sum AY}{A} = \frac{0.218}{0.22} = 0.99$$

$$\bar{Y}^2 = 0.98$$

$$I_x = \sum AY^2 + I_o = 0.273 + 0.0082 = 0.281$$

$$I_c = I_x - (\sum A) \bar{Y}^2 = 0.281 - 0.22 (0.98) = 0.0654 \text{ in}^4$$

Calculated Bending Stress at Section B-B

$$M_{B-B} = 1145 \text{ in-lb}$$

$$I_o = 0.0654 \text{ in}^4$$

$$\bar{Y} = 0.99 \text{ inch}$$

$$f_b = \frac{M \bar{Y}}{I} = \frac{1145 (0.99)}{0.0654} = 17,320 \text{ psi}$$

Calculated Shear Stress at Section B-B

Item 1 and Item 2 Drawing No. X209871

$$q = \frac{T}{2A} = \frac{200.43}{2 (4.92)} = 21.00 \text{ lb/in}$$

$$P = Lq = 4.21 (21.00) = 88.00 \text{ pounds}$$

$$A_s = \text{shear area} = 1.20 (0.032) = 0.0384$$

$$f_s = \frac{P}{A_s} = \frac{88.00}{0.0384} = 2290 \text{ psi}$$

Calculated Shear Stress at Section B-B

Item 1 and Item 2 Drawing No. X209871

Assuming no Lightening Hole

$$P = 88.00 \text{ pounds}$$

$$A_s = \text{shear area} = 4.17 (0.032) = 0.133 \text{ in}^2$$

$$f_s = \frac{P}{A_s} = \frac{88.00}{0.133} = 662 \text{ psi}$$

Allowable Stresses

Local buckling of Item 1 and Item 2 Drawing No. X209871

$$\frac{b}{t} = 19.10 \quad (\text{see Figure 5-213})$$

$$F_{CC} = 21779$$

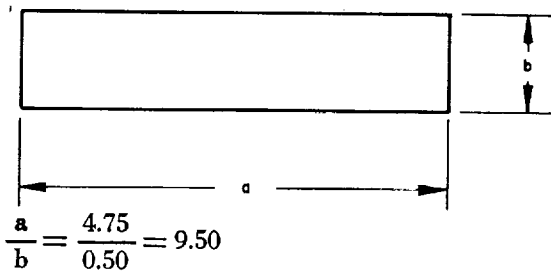
Critical Shear Stress

$$F_{SCR} = KE \left(\frac{t}{b} \right)^2$$

Element of sheet around cutout

(Item 1, Drawing No. X209871)

5. Spacecraft System Design



From Reference 5-1, page 394

$$K = 4.80$$

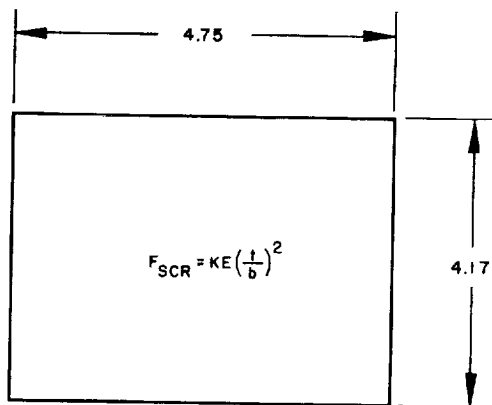
$$F_{SCR} = 4.80 (10.5) \times 10^6 \left(\frac{0.032}{0.50} \right)^2$$

$$F_{SCR} = 20 \times 10^4 \text{ psi}$$

From Reference 5-2, page 3.2.3.0(b)

$$F_s = 0.6 F_{ty} = 28800 \text{ psi}$$

Critical shear stress Item 1 and Item 2,
Drawing No. X209871 assuming no cutout.



DIMENSIONS IN INCHES

$$\frac{a}{b} = \frac{4.75}{4.17} = 1.14$$

From Reference 5-1, page 394

$$K = 7.75$$

$$F_{SCR} = 7.75 (10.5) \times 10^6 \left(\frac{0.032}{4.17} \right)^2$$

$$F_{SCR} = 4800 \text{ psi}$$

Margins of Safety

Bending

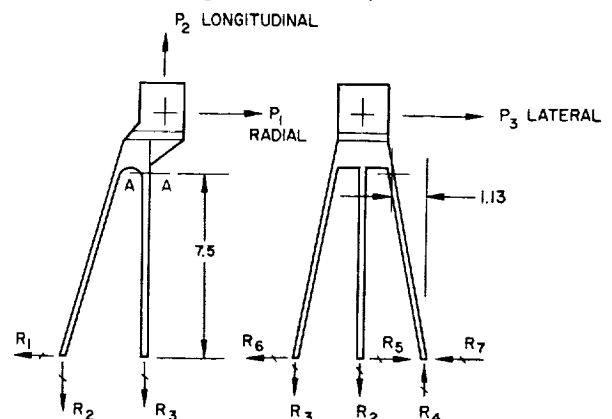
$$MS = \frac{21779}{17320} - 1 = + 0.25$$

Shear

$$MS = \frac{28800}{2290} - 1 = + 11.50$$

$$MS = \frac{4800}{662} - 1 = + 6.25$$

- Support Assembly Sun Sensor
(Drawing No. X209868)



Weight of sensor assembly only = 0.26 pounds

Sign convention — positive shown

Design Conditions

I — Radial $g = + 84.8$
Longitudinal $g = 0$

II — Radial $g = + 14.8$
Longitudinal $g = \pm 70.0$

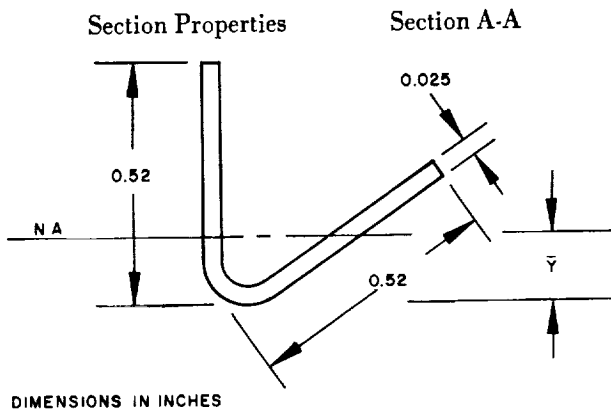
III — Lateral $g = \pm 70$
Longitudinal $g = 0$

IV — Longitudinal $g = \pm 73$
Lateral $g = 0$

V — Radial $g = 14.8$
Lateral $g = \pm 70$

Material

7075-TN AL AL sheet $t = 0.025$
QQ-A-283



Critical Load Condition

Condition III

$$R_3 = R_4 = 39.2 \text{ pounds}$$

$$R_6 = R_7 = 10.8 \text{ pounds}$$

Moment due to above loads at Section A-A

$$M = 10.8 (7.5) - 39.2 (1.13) = 36.7 \text{ in-lbs}$$

Calculated Stresses

$$f_c = \frac{P}{A} = \frac{40}{0.026} = 1540 \text{ psi}$$

$$f_B = \frac{MC}{I} = \frac{36.7 (.32)}{4.7 (10^{-4})} = 25,000 \text{ psi}$$

Allowable Stress

$$F_{CC} = 23,000$$

for b/t = 20 7075-T6 AL AL sheet (Figure 5-213)

Margin of Safety

$$f_c = 26,540 \text{ psi}$$

$$MS = \frac{23,000}{26,540} - 1 = 0.05$$

• REFERENCES

5-1. Peery, D. J., *Aircraft Structures*, McGraw-Hill Book Company, New York, New York, 1949.

5-2. Armed Forces Supply Support Center, "Strength of Metal Aircraft Elements," MIL-HDBK-5, Washington 25, DC, August 1962.

5-3. Magnesium Design Data Book, Dow Metal Products 1957.

5-4. Den Hartog, J.P., *Advanced Strength of Materials*, McGraw-Hill Book Company, New York, New York, 1952.

Mass Property Analysis. The majority of the mass properties engineering effort during the Interim Contract period was expended to maintain the weight and roll-to-pitch (R/P) moment of inertia ratio desired for the test model Advanced Syncom, HSX-302-T2. Configuration changes were extensive, and the corresponding weight values associated with these changes were described in each of the Monthly Progress Reports.

Configuration changes and updating of electronic component weights did not cause the spacecraft weight to exceed the specified 1518-pound maximum weight of separation from the Altas/Agna booster. The weight objective can be achieved and maintained. The minimum R/P objective of 1.10 for the final orbit condition (after bipropellant depletion — nominally seven years) has been achieved but is difficult to maintain. Continued effort toward mass properties control is necessary to prevent exceeding the maximum spacecraft weight and to provide mass distribution analyses that will ensure acceptable R/P ratios.

A degradation of the R/P to a low of 1.05 for the final orbit condition was noted in the July Monthly Progress Report. Design changes that contributed to the R/P degradation were:

- 1) Increasing length and thickness of solar panels.
- 2) Increasing thickness of protective cover glass of solar cells.
- 3) Increasing electronic subsystem weights.

5. Spacecraft System Design

- 4) Changing structure subassembly at the forward and aft extremities.
- 5) Relocating ballast and balance weights from the center-of-gravity plane.

Various design changes to basically the same configuration were analyzed to increase this R/P. The roll moment of inertia was considered as optimum since the various components are located as far outboard as possible; therefore, studies were made to decrease the pitch moment of inertia and its transfer distance to the spacecraft center of gravity. Resulting design changes included:

- 1) Shortening the solar panel length by removing 1.50 inches from the forward end. Solar cell strings were shortened by 2 cells in each row of panels to accomplish this.
- 2) Shortening the aft ribs 0.8 inch.
- 3) Moving apogee motor and mount ring 0.8 inch aft.
- 4) Moving the torus ring 1.25 inches aft and adjusting associated components mounted to it accordingly.
- 5) Mounting the telemetry transmitter and regulator in forward area.
- 6) Mounting the wire harness in forward area.
- 7) Reducing weight.
- 8) Adjusting the spacecraft center of gravity.

These design changes accounted for the R/P of 1.11 given in the August report.

The September Progress Report included Hughes calculated values of weight, balance, and moment of inertia for the JPL apogee motor in the prelaunch and burned-out conditions in lieu of JPL estimates. Inclusion of the Hughes-calculated values for the JPL apogee motor would show a lower R/P ratio. Additional changes that

were made in the structural configuration to maintain the R/P ratio are as follows:

- 1) Moving torus 2.5 inches aft and adjusting associated mounted components.
- 2) Mounting batteries ± 2.0 inches from spacecraft center of gravity.
- 3) Re-evaluating the wire harness for weight and moving it forward of the center subassembly.
- 4) Updating the electronic subsystem weights.
- 5) Increasing the thermal shield weight estimate.
- 6) Redesigning the solar panel supports because of rotation of $22\frac{1}{2}$ degrees in panels about spacecraft.
- 7) Making numerous design changes in structure.

The final orbit condition roll-to-pitch ratio given in the September report was 1.10, which is still being maintained.

A special study aimed at increasing the R/P ratio was undertaken. It includes redesign of the spacecraft center section and elimination of the ribs in the aft area. Rib-mounted electronic units were relocated to the aft face of the aft bulkhead; a forward bulkhead was added to the center section; the batteries were mounted to the bulkheads in the center subassembly area; and the quadrant packages were designed to incorporate individual or sets of electronic boards. The wiring harness was located aft of the quadrants to prevent the center of gravity from moving into the forward row of solar panels. A final orbit R/P of 1.132 was obtained with the configuration used in this special study. An increase to 1.145 was noted if the batteries were moved to within ± 2.0 inches from center of gravity and a lighter gage torus assembly was used. It was apparent

from these studies that the R/P of 1.20 for the final orbit condition could not be achieved with the current design parameters for the GFE apogee motor.

The current mass property data for the Advanced Syncom Model HSX-302-T2 is listed in Tables 5-44, 5-45 and 5-46.

TABLE 5-44. DETAILED DESCRIPTION OF WEIGHT CHANGES SINCE THE MONTHLY PROGRESS REPORT FOR SEPTEMBER 1963

| Description of Weight Change | Weight Change, pounds |
|---|-----------------------|
| Electronic subsystem | (+5.6) |
| Revised estimate, electronic quadrants | +1.8 |
| Revised estimate, repackaged rib-mounted components | +2.0 |
| Revised estimate, communications antenna | +1.8 |
| Wire harness subassembly | (-1.1) |
| Redesigned main wire harness | +3.7 |
| Incorporate quadrant harnesses into main harness | -4.8 |
| Power supply subsystem | (-0.1) |
| Battery cell weight based on specification | -0.5 |
| Revised estimate, edge treatment of substrate | -1.4 |
| Increase quantity of diodes | +1.8 |
| Control subsystem | (-6.1) |
| Amended Marquardt estimate to reflect latest flight spacecraft estimates | -6.1 |
| Structure subsystem | (+4.1) |
| Aft subassembly, miscellaneous changes | +0.3 |
| Redesign in solar panel supports | +1.3 |
| New estimate quadrant package mounting brackets | +2.2 |
| Revised estimate, reaction control system tank supports | +0.5 |
| Revised estimate, battery supports | +3.9 |
| Center subassembly, miscellaneous changes | -0.5 |
| Design changes, truss assembly | +0.3 |
| Decrease in gage in torus assembly plus miscellaneous design changes | -3.2 |
| Redesign of whip antenna mount | -0.7 |
| Miscellaneous subsystem | (+20.0) |
| Estimate for weight necessary to maintain spacecraft center of gravity, roll-to-pitch ratio, and principal axis | +20.0 |
| Ballast subsystem | (-22.4) |
| Decreased in order that the maximum spacecraft weight may be maintained | -22.4 |
| Net weight change in payload separation | 0.0 |

TABLE 5-45. CURRENT WEIGHT STATUS OF HSX 302-T2

| Subsystem | Δw^* , pounds | Current Weight, pounds | Target Weight, pounds | ϕ^{**} | ϕ^{***} |
|-----------------------|-----------------------|------------------------|-----------------------|-------------|--------------|
| Electronics | + 5.6 | 184.1 | 157.4 | 0.302 | 0.121 |
| Wire harness | - 1.1 | 14.9 | 16.4 | 0.024 | 0.010 |
| Power supply | - 0.1 | 112.6 | 112.5 | 0.185 | 0.074 |
| Controls (dry) | - 6.1 | 47.1 | 49.9 | 0.077 | 0.031 |
| Propulsion | 0 | 102.6 | 102.0 | 0.168 | 0.068 |
| Structure | + 4.1 | 108.1 | 94.2 | 0.177 | 0.071 |
| Miscellaneous | +20.0 | 37.8 | 14.5 | 0.062 | 0.025 |
| Ballast | -22.4 | 2.8 | 62.5 | 0.005 | 0.002 |
| Final orbit condition | 0.0 | 610.0 | 609.4 | 1.000 | 0.402 |

| | Current Weight, pounds | Z-Z, inches | I_{x-x} , slug-ft ² | I_{y-y} , slug-ft ² | Roll-to-Pitch Ratio |
|---|------------------------|-------------|----------------------------------|----------------------------------|---------------------|
| Final orbit condition | 610.0 | 23.90 | 57.71 | 52.50 | 1.10 |
| N ₂ pressurant | 2.6 | — | — | — | — |
| N ₂ H ₄ -CH ₄ fuel | 51.6 | — | — | — | — |
| N ₂ O ₄ oxidizer | 81.9 | — | — | — | — |
| Total at apogee burnout | (746.1) | 23.90 | 71.61 | 59.45 | 1.20 |
| Apogee motor propellant | 752.3 | — | — | — | — |
| Apogee motor expendables | 19.6 | — | — | — | — |
| Total Payload at separation | 1518.0 | 23.03 | 88.42 | 76.46 | 1.16 |

*Change in subsystem weight since last report.

**Ratio of subsystem weight to final orbit condition weight.

***Ratio of subsystem weight to total payload at separation.

Thermal Control. The main effort of the thermal control task during this report period has been to:

- 1) Specify a vehicle thermal control system compatible with Syncom temperature requirements, mission objectives, and all modes of spacecraft operation

TABLE 5-46. DETAILED WEIGHT BREAKDOWN FOR HSX-302-T2

| Component | | Weight,
pounds | Component | | Weight,
pounds |
|------------------------|---|-------------------|-------------------------|-------------------------------|-------------------|
| Electronics subsystem | | (184.12) | Apogee motor subsystem | | (102.59) |
| 475025 | Electronic quadrant | 117.49* | X209749 | Apogee motor | 102.59* |
| 475220 | Telemetry transmitter | 2.00* | Structure subsystem | | (108.07) |
| 475222 | Regulator — telemetry | 1.12* | X209862 | Aft subassembly | 4.09** |
| | rib-mounted units | 8.30 | X209813 | Separation ring | 4.71** |
| | antenna electronics | 8.75* | X209875 | Aft thrust tube | 5.34* |
| 475125 | Traveling-wave tube (TWT) | 10.00* | X209812 | Motor mount ring | 16.20* |
| 475174 | TWT power supply | 0.25* | X209896 | Ribs | 0.94** |
| 475171 | 3 db hybrid | 1.08* | X209816 | Frame ring | 6.53* |
| 475173 | RF switch | 4.00* | X209882 | Aft bulkhead | 0.87** |
| 475170 | Phase shift driver | 0.52* | X209862 | Miscellaneous hardware | 2.78** |
| 475172 | Telemetry and test coupler,
antenna compartment | 19.51* | X209793 | | 1.40* |
| 475030 | Communication antenna | 5.25* | + 879 | Solar panel attachments | 5.65* |
| 250203 | Transmitter multiplexer | 2.86* | X209863 | Miscellaneous | 5.68* |
| 250202 | Receiver multiplexer | 0.06* | X209883 | Center subassembly | 0.32* |
| 475103 | Directional receiver coupler
coaxial cable,
antenna subsystem | 0.15* | X209874 | Center thrust tube | 2.24* |
| | | 0.64* | X209884 | Tube stiffener | 0.78* |
| 475204 | Whip antenna | 0.64* | X209857 | Zee support ring | 3.92* |
| 475204 | Whip antenna | 1.50* | X209885 | Quadrant support | 0.51** |
| 475200 | Balun hybrid | (14.86) | X209883 | Miscellaneous hardware | 1.80* |
| Wire harness subsystem | | (14.86) | X209855 | Solar panel retainer support | 1.68* |
| 475300 | Main harness | 14.86* | X209785 | Solar panel retainer | 1.56* |
| Power supply subsystem | | (112.65) | X209854 | Forward channel support | 5.64* |
| X209799 | Forward battery | 27.28* | X209908 | Radial engine support | 14.64* |
| X209799 | Aft battery | 27.28* | X209876 | Forward tank support | 3.10* |
| X209781 | Substrate-solar cell | 23.27** | X209890 | Aft tank support | |
| X209888 | Solar cell | 34.82** | | Battery support | |
| Control subsystem | | (47.13) | X209863 | Miscellaneous hardware | |
| | Axial engine | 1.20* | X209850 | Forward subassembly | |
| | Axial engine | 1.20* | X209851-1 | Truss tube | 1.07** |
| | Swivel mount | 1.66* | X209848 | Forward truss fittings | 0.39** |
| | Swivel mount | 1.66* | X209850 | Forward fittings hardware | 0.07* |
| | Radial engine | 1.20* | X209849 | Aft truss fittings | 0.66** |
| | Radial engine | 1.20* | X209850 | Aft fittings hardware | 0.10* |
| | Radiation shield | 0.68* | X209853 | Support channel | 0.74* |
| | Radiation shield | 0.68* | X209871 | Torus assembly | 4.49** |
| | Thrust alignment motor | 0.35* | X209880 | Solar panel support | 2.60** |
| | Thrust alignment motor | 0.35* | X209863 | Miscellaneous hardware | 0.50* |
| | Tank | 26.96* | | Wire harness support | 2.00* |
| | Relief valve | 0.76* | | Rib-mounted units support | 1.43* |
| | Relief valve | 0.76* | X209856 | Communication antenna support | 1.64** |
| | Fill propellant valve | 1.92* | X209902 | Whip antenna support | 0.65* |
| | Solenoid latch valve | 0.80* | | Control subsystem support | 1.35* |
| | Tubing and fittings | 3.30* | Miscellaneous subsystem | | (37.79) |
| | Pressure transducer | 1.40* | X209866 | Sun sensor | 1.15* |
| | Temperature transducer | 0.05* | | Paint | 3.00* |
| | Radial restrictor (variable) | 0.23* | | Nutation damper | 2.00* |
| | Axial restrictor (variable) | 0.23* | X209909 | Ground plane | 0.87* |
| | Radial filter | 0.03* | X209111 | Ground plane | 1.77* |
| | Axial filter | 0.03* | X209910 | Thermal barrier | 4.00* |
| | Miscellaneous | 0.48* | | Center-of-gravity control | 0.40* |
| | | | | Center-of-gravity control | 12.60* |

TABLE 5-46. (continued)

| Component | | Weight, pounds | | | |
|---|------------------------|----------------|---|---|---------------------|
| Static-dynamic balance | | 2.69* | | | |
| Static-dynamic balance | | 2.31* | | | |
| Principal axis control | | 0.87* | | | |
| Principal axis control | | 1.13* | | | |
| R/P control | | 2.50* | | | |
| R/P control | | 2.50* | | | |
| Ballast subsystem | | (2.76) | | | |
| Aft ballast | | 1.20* | | | |
| Forward ballast | | 1.56* | | | |
| | Current Weight, pounds | Z-Z, inches | I _{x-x} , slug-ft ² | I _{y-y} , slug-ft ² | Roll-to-Pitch Ratio |
| Final orbit condition | 609.97 | 23.90 | 57.71 | 52.50 | 1.10 |
| N ₂ pressurization | — | 2.60* | — | — | — |
| N ₂ H ₄ -CH ₄ fuel | — | 51.61* | — | — | — |
| N ₂ O ₄ oxidizer | — | — | — | — | — |
| Total at apogee burnout | 746.09 | 23.90 | 71.61 | 59.45 | 1.20 |
| Apogee motor propellant | 753.30* | — | — | — | — |
| Apogee motor expendables | 19.61* | — | — | — | — |
| Payload at separation | 1518.00 | 23.03 | 88.46 | 76.46 | 1.16 |
| Final orbit condition | Weight | | Percent/100 | | |
| Actual | 0. | | 0. | | |
| Calculated | 82.84 | | 0.055 | | |
| Estimated | 1435.16 | | 0.945 | | |
| | 1518.00 | | 1.000 | | |

*Estimated.

**Calculated.

***Actual.

- 2) Determine subsystem temperature limits during normal steady state orbital mode
- 3) Investigate vehicle transient response to firing of the apogee and reaction control motors

The thermal control system concept has been specified and is based purely on the results of analytical findings. A detailed nodal analytical

model has been developed to a point of approximately 75 percent completion and is expected to be completed and used extensively during the early months of the Advanced Syncom development and launch program. A single node vehicle parametric analysis has been completed in which the effects of solar orientation, vehicle power dissipation, vehicle outside surface properties, etc., on spacecraft temperatures has been studied. A vehicle thermal system analysis has been made from which a preferred thermal control system has been specified.

Subsystem thermal analytical studies have been completed on generally an individual basis. Reaction control system thermal analyses have been performed for both the radial and axial jets. Solar panels have been investigated for both the steady state and eclipse modes of operation. The apogee motor effect on the spacecraft has been studied from the point of view of vehicle warmup due to motor firing, motor plume effects on the forward end of the spacecraft, and thermal energy leakage rates from the vehicle through the apogee motor during the normal orbital phase of the spacecraft. The communications antenna has been thermally analyzed to determine normal orbital maximum and minimum temperatures and antenna component heating during apogee motor firing. The Advanced Syncom temperature sensor locations and telemetry format have also been established.

Vehicle Thermal Control System Analysis.

The Advanced Syncom spacecraft detailed nodal thermal analysis has been partially completed. A quarter section of the vehicle has been simulated by a thermal network consisting of approximately 60 nodes. The analytical model is shown in Figure 5-214, indicating nodal locations and nodal numbers. Much of the input data required has been calculated and tabulated for future

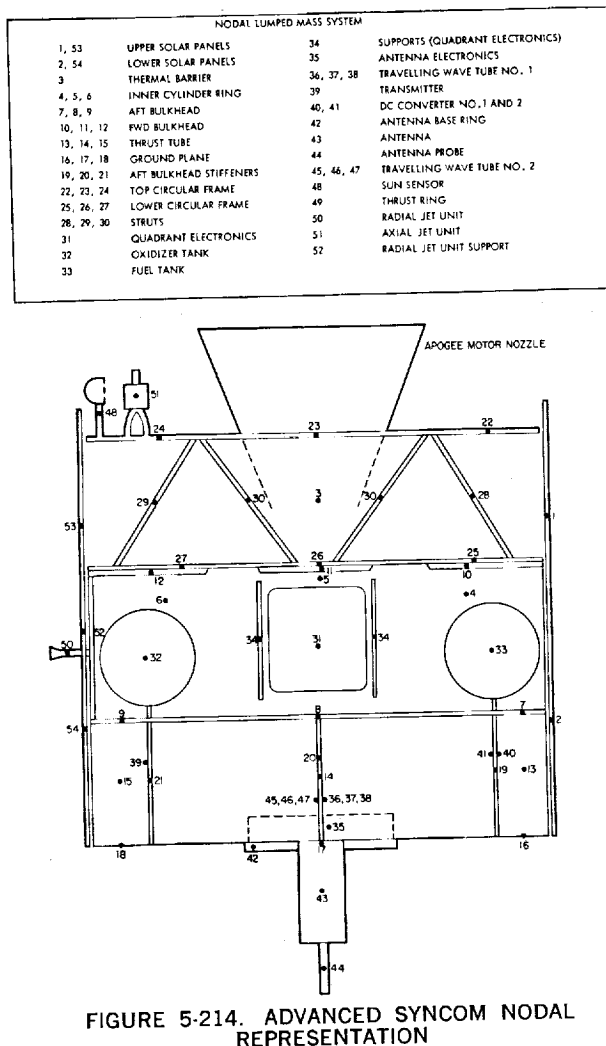


FIGURE 5-214. ADVANCED SYNCOM NODAL REPRESENTATION

computation on the IBM 7090 computer. The data includes thermal capacity, internal heat generation, thermal conductance, thermal radiation coefficients, and spacecraft geometry.

A preliminary bulk thermal analysis of Advanced Syncom has been completed in an attempt to establish the significance of various parameters in their effects on the basic thermal design. To investigate all significant parameters, a simplified analytical model of the vehicle was utilized in which several compromises were made. The

actual Advanced Syncom vehicle consists of basically five external surfaces: solar panels, ground plane, thermal barrier, apogee motor nozzle, and phased array antenna. These surfaces, in addition to the internal power dissipation of the vehicle, determine the spacecraft bulk temperature in the deep space environment. The parametric survey was made based on a simplified Advanced Syncom model, not including the phased array antenna and apogee motor nozzle in the bulk thermal analysis. The model assumed a right circular cylindrical geometry consisting of solar panels, thermal barrier, and ground plane. Parameters varied were 1) solar angle, 2) internal power, 3) end plane surface properties, 4) solar panel surface properties, and 5) earth orbital variations.

The changes in vehicle bulk temperature for given perturbations in the parameters are shown in Figure 5-215. The vehicle bulk temperature is seen to be extremely sensitive to such factors as solar panel absorptance and solar panel emittance but only slightly affected by ground plane and thermal barrier emittance.

The parametric analysis was extended to include an error analysis of the degree of uncertainty in the detailed temperature predictability of a Syncom type of vehicle. In addition to basic uncertainties in the parameters mentioned above, other factors contributing to the overall tolerance on a specified temperature such as nodal power dissipation and modeling inaccuracies were considered. All the parameters investigated along with known or expected reasonable perturbations to the nominal values are listed in Table 5-47. The corresponding fluctuation in vehicle bulk temperature is shown for each parameter. The root sum square deviation from a calculated nominal temperature is calculated in Table 5-47. On the basis of a bulk temperature estimate for

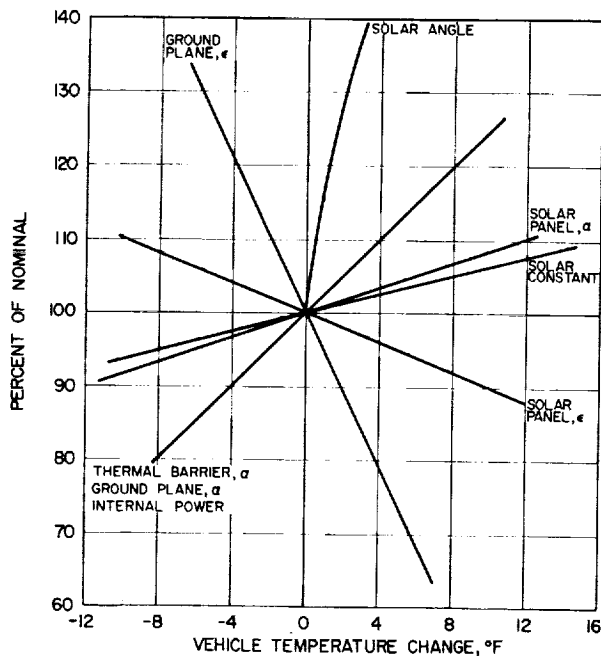


FIGURE 5-215. THERMAL EFFECTS OF VEHICLE PARAMETERS

a Syncom type of vehicle, a tolerance of $\pm 11^\circ\text{F}$ must be applied; for a nodal temperature estimate (i.e., at a specific location) the tolerance is $\pm 14^\circ\text{F}$. However, for a vehicle of known power dissipation (to within 5 percent), known solar angle (i.e., time of year), and known earth orbital position, the root sum square calculation as shown in Table 5-48 results in a tolerance of $\pm 12^\circ\text{F}$ to be applied to a nodal temperature calculation.

Conclusions derived from this analysis are as follows:

- 1) The thermal barrier surface must be included in the vehicle design for satisfactory thermal control. If no barrier exists, the vehicle temperature cannot be raised above 55°F when the sun is on the antenna end of the spacecraft.
- 2) A tolerance of $\pm 12^\circ\text{F}$ must be assumed on any computed nodal temperature even

TABLE 5-47. ADVANCED SYNCOM TEMPERATURE RESPONSE TO PARAMETRIC VARIATIONS

| Parameter | Possible Deviation from Nominal | $+\Delta T, ^\circ\text{F}$ | $-\Delta T, ^\circ\text{F}$ | $+\Delta T^2$ | $-\Delta T^2$ |
|--------------------|---------------------------------|-----------------------------|-----------------------------|---------------|---------------|
| Bulk | | | | | |
| Solar angle | ± 25 degrees | 2 | 0 | 4 | 0 |
| Internal power | ± 5 percent | 2 | 2 | 4 | 4 |
| Solar absorptance | | | | | |
| Ground plane | ± 5 percent | 2 | 2 | 4 | 4 |
| Thermal barrier | ± 5 percent | 2 | 2 | 4 | 4 |
| Solar panel | ± 5 percent | 6 | 6 | 36 | 36 |
| Emittance | | | | | |
| Ground plane | ± 5 percent | 1 | 1 | 1 | 1 |
| Thermal barrier | ± 5 percent | 0 | 0 | 0 | 0 |
| Solar panel | ± 5 percent | 5 | 5 | 25 | 25 |
| Orbital variations | ± 4.4 percent | 7 | 7 | 49 | 49 |
| Subtotal | | ± 27 | -25 | 127 | 123 |
| Nodal | | | | | |
| Power | ± 5 percent | 1 | 1 | 1 | 1 |
| Conductance | ± 20 percent | 8 | 8 | 64 | 64 |
| Radiation coupling | ± 20 percent | 3 | 3 | 9 | 9 |
| Total | | 39 | 37 | 201 | 197 |

$$\text{Bulk temperature variation (rss): } T = T_{\text{nominal}} \pm \sqrt{\frac{127}{123}} = T_{\text{nominal}} \pm 11^\circ\text{F}$$

$$\text{Nodal temperature variation (rss): } T = T_{\text{nominal}} \pm \sqrt{\frac{201}{196}} = T_{\text{nominal}} \pm 14^\circ\text{F}$$

TABLE 5-48. ADVANCED SYNCOM VARIATIONAL TEMPERATURE TOLERANCES

| Parameter | $\pm \Delta T^2$ | $-\Delta T^2$ |
|---|------------------------------|---------------|
| Power dissipation | 5 | 5 |
| Solar absorptance | 44 | 44 |
| Emittance | 26 | 26 |
| Modeling inaccuracies | 73 | 73 |
| Total | 148 | 148 |
| Nodal temperature | $T_{\text{nominal}} \pm 148$ | |
| $T = T_{\text{nominal}} \pm 12^\circ\text{F}$ | | |

after best engineering practice has been applied both to analytical methods and surface property measurement. No provision has been made in the tolerance for potential surface property degradation due to space effects or apogee motor firing effects.

Although the space effects on thermal properties of proposed Advanced Syncom outside surfaces (ground plane, thermal barrier, solar cells, etc.) are not known, an attempt has been made to evaluate vehicle sensitivity to these possible effects. A simplified analytical model of the Advanced Syncom spacecraft has been analyzed to determine the effect on bulk temperature of possible changes in the surface nominal thermal properties due to ultraviolet exposure, micrometeorite impingement, apogee motor blast, etc. The analytical model was parametrically evaluated for changes in solar absorptance and IR emittance of the thermal barrier, ground plane, and solar panels. Results for various degrees of degradation are shown in Figure 5-216 on a root sum square temperature deviation basis. It can be seen that significant temperature variations can exist for relatively small surface degradation

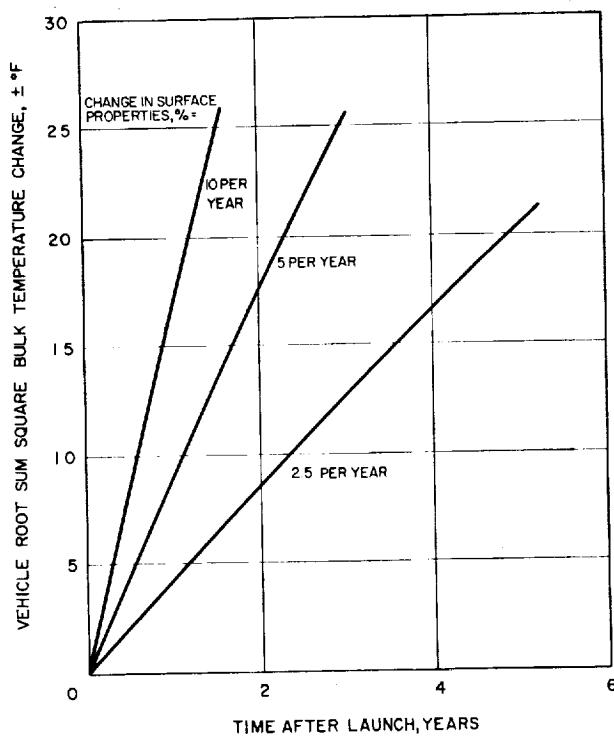


FIGURE 5-216. SURFACE DEGRADATION EFFECTS ON VEHICLE TEMPERATURES

rates; i.e., at only 5 percent per year degradation to all surfaces, a temperature variation of approximately 25° F can be expected to exist after 3 years.

In an attempt to determine the optimum thermal control system for Advanced Syncom, a two-surface or two-node analytical model of the spacecraft has been utilized. The model approximates the vehicle as consisting of solar panels and vehicle bulk structure with only radiation exchange occurring between the two. The analysis has been performed with emphasis on determining 1) spacecraft orbital bulk temperature variations, 2) vehicle axial gradients at extreme sun positions, 3) potential need for active thermal control system, and 4) a general overall thermal design concept.

Results have yielded vehicle bulk and solar panel temperatures at high and low power for three spacecraft configurations varying from the Syncom 2 thermal control concept to the concept of complete isolation of the spacecraft ends from deep space. A preferred thermal control system has resulted and the recommendation has been made to incorporate a completely passive thermal control technique in the Advanced Syncom vehicle design. Limiting conditions for the recommended design have been specified, and major areas of potential problems and uncertainties have been discussed.

The analytical model considered a cylindrical vehicle as one node and cylindrical solar panels as the other. The resulting equations were solved for vehicle bulk and solar panel temperatures for high and low power modes, solar angles of 90 ± 25 degrees, and three configurations varying from the Syncom 2 concept of enclosed ends to the idealized concept of perfectly insulated ends. Results are shown in Figure 5-217 for the three configurations at the high power mode condition.

An active thermal control concept has been considered for use on Syncom in the event that reasonable temperature swings of system components cannot be obtained passively or that external surface thermal property degradation is so great over the long time period of the spacecraft as to seriously perturb temperatures. In the investigation, it was determined that for spacecraft geometry such as Syncom (in which the sides of the cylinder comprise the solar array and the ends are required to isolate the vehicle from space to achieve reasonable bulk temperature) an active thermal control system could be useful only if it were physically located on the same surface as the solar cells. Therefore, only if the heat absorbing or heat radiating ability of the sides of the cylinder were variable could the controllable swing in temperature be significant and even then only at the expense of significant increase in length (and consequently weight) of the spacecraft. It was finally determined that the main objectives of an active thermal control system could be established, at least in part, with a properly designed and significantly lighter, passive thermal control concept.

The major objectives of a thermal control system for the Advanced Syncom Spacecraft have been stated as:

- 1) To provide an average vehicle bulk steady state orbital temperature of 70°F during the life of the spacecraft.
- 2) To minimize changes from this average temperature during orbital operation.
- 3) To minimize temperature gradients within the vehicle induced by changes in solar angle within the range of 90 ± 25 degrees.

From Figure 5-217 it is seen that the vehicle bulk temperature can be made to vary from 90° to 64°F, depending on the degree of isolation of the end planes from space and on the solar angle. As seen in Figure 5-218 based on a Syncom 2 type of thermal concept in which the vehicle ends are enclosed but not necessarily isolated to a high degree from outer space, the vehicle axial gradient may be as high as 115°F. For the idealized concept of perfectly isolated ends, however, the axial gradient in temperature is non-existent. Although the effects of space conditions on surface properties of materials are relatively unknown, consideration has been given to this factor in selecting a thermal design. The internal thermal coupling of the vehicle components has also been included, at least in general terms, in specifying a recommended thermal design concept.

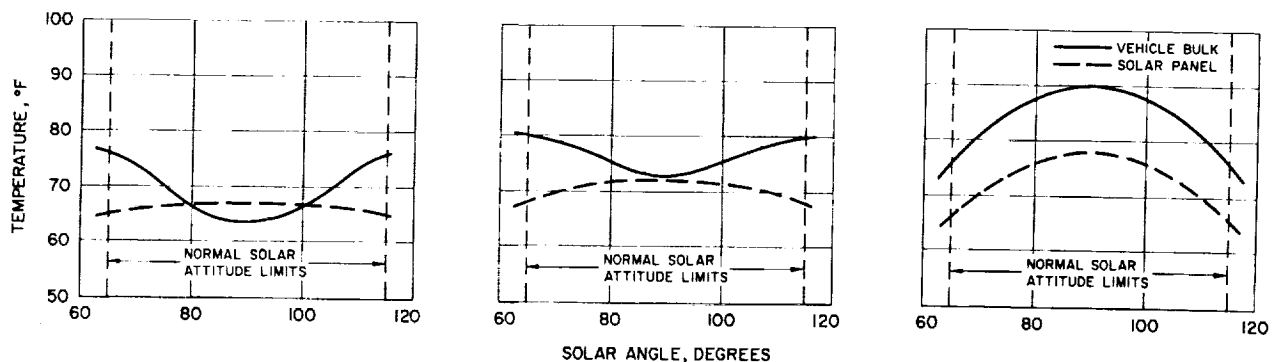


FIGURE 5-217. VEHICLE END GEOMETRY VERSUS TEMPERATURE

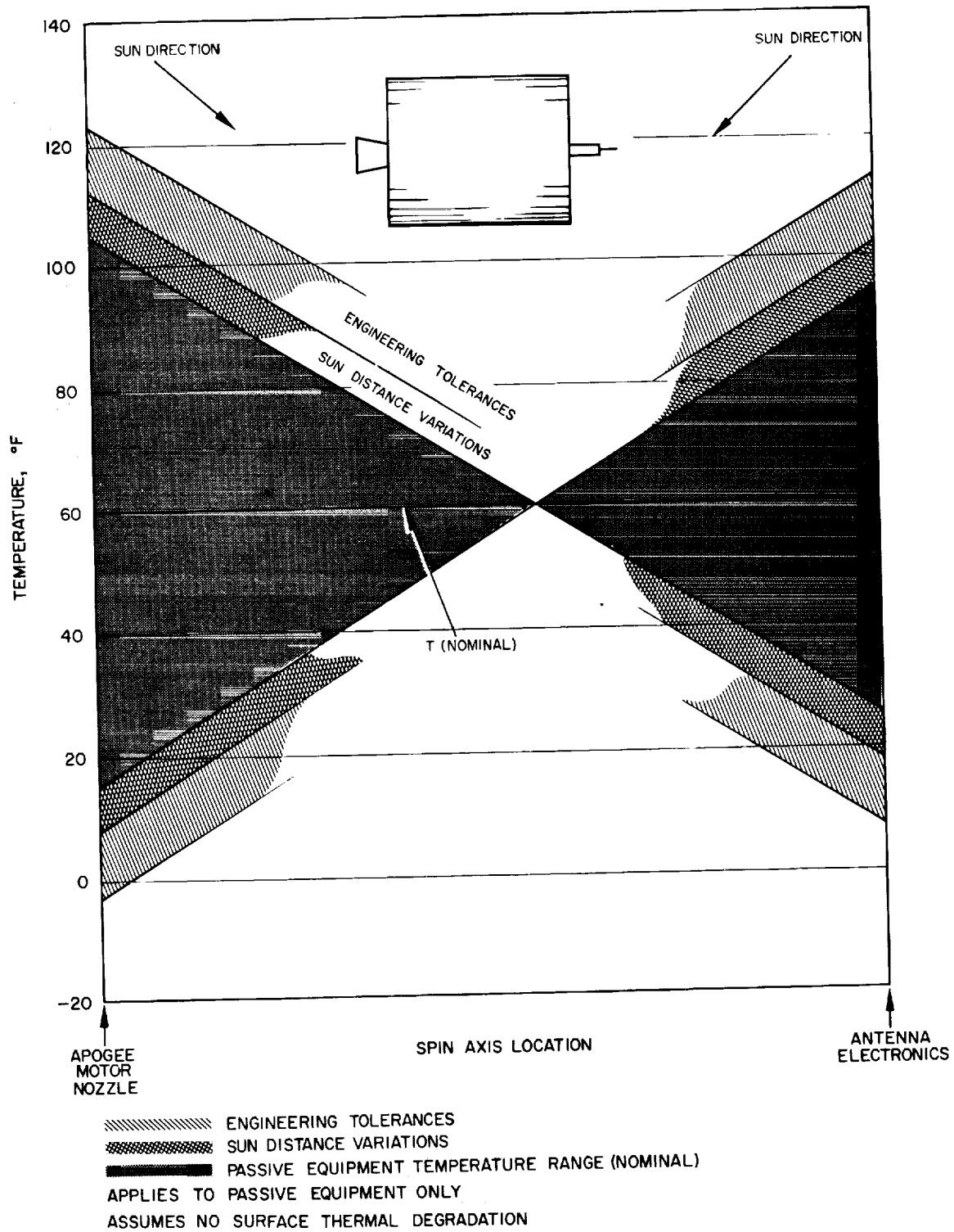


FIGURE 5-218. ADVANCED SYNCOM VEHICLE ORBITAL TEMPERATURES WITH SYNCOM 2 TYPE THERMAL CONTROL SYSTEM

The resultant recommended spacecraft passive thermal design consists of the following:

- 1) Semi-insulated ends — This concept requires that the end planes of the vehicle (i.e., the ground plane and thermal barrier) be approximately normal to the solar panels, possibly insulated from the vehicle structure through the use of several layers of thermal radiation shielding, and containing the minimum of discontinuities (cracks, cavities, etc.). This concept will minimize the loss of energy from the vehicle at all times and reduce the absorption of solar energy at the illuminated end plane, resulting in an average bulk and solar panel temperature similar to Figure 5-217 (curve B). In addition, the axial temperature gradients within the vehicle will be reduced considerably from those previously reported for the Syncom 2 design concept. The proposed design will also reduce the vehicle temperature dependence on the external surface properties of the end planes to a negligible amount, thus isolating the surface property degradation problem to only the solar panel properties.
- 2) Soft mounting for passive components — To reduce to a minimum the temperature fluctuation of passive (non-heat-generating) components such as tanks, and batteries during short-time spacecraft transients caused by eclipses, power transients, or motor firings, it has been recommended that insulative types of mounts be considered. In active components, hard mounts are required so that vehicle structure can remove dissipated energy rapidly from local heat generating sources.

- 3) Localized radiation shielding where required — In the areas of the spacecraft where high heat fluxes over short periods of time are experienced, such as in the vicinity of the apogee motor or reaction control jets, it may be necessary to protect surrounding equipment with radiation shielding to prevent momentary overheating of any vehicle components or systems.

The selection of a completely passive thermal control system for the Advanced Syncom spacecraft includes several potential problem areas and system limitations which warrant further consideration:

- 1) Quoted vehicle bulk temperatures apply to average structural mounting surfaces and passive equipment. Active heat dissipating elements must be mounted in such a way as to minimize temperature rises at local hot spots.
- 2) The exterior surface property degradation attributable to space effects is not well known. Although the proposed thermal design isolates this potential problem to the solar panels, a significant thermal effect might be experienced. However, since the solar panels are nearly black in radiation properties, it is not felt that the vehicle bulk temperature could rise more than approximately 20°F above design values. The most severe imaginable drop in vehicle bulk temperature caused by surface damage is in the 30° to 40°F range. Neither of these potential perturbations seems catastrophic in itself. Further assurance is provided by the thicker cover glass for the solar cells.
- 3) Since no active thermal devices are specified for the Advanced Syncom design, the

quoted bulk temperature range applies only in the region of solar angles between 65 and 115 degrees in reference to the positive spin axis. No provision is made for thermal control outside this solar region.

- 4) Vehicle internal temperatures during eclipse transients are unknown at this time. The degree of thermal coupling between the vehicle and solar panel will determine the rate of bulk temperature drop with time.
- 5) External motor firing effects are as yet unknown. Plume expansion for both the apogee motor and the reaction control jets has been partially investigated, but present state-of-the-art knowledge of the total effects is incomplete. Further analysis and test is required to establish thermal protection requirements for the thermal barrier and solar panels.

Subsystem Thermal Studies

- Reaction Control System. Preliminary thermal analysis of the Advanced Syncom axial reaction control nozzles has indicated maximum and minimum steady state non-operating temperatures of 0 and 90°F respectively. Calculations have been based on steady state orbit temperature conditions at solar angles of 115 and 65 degrees respectively, and boundary temperatures, where applicable, have been taken from Syncom 2 analytical data.

Preliminary thermal analysis has been made to determine radial engine temperatures and insulation requirements. In accordance with specifications, the radial engine external package temperature is not to exceed 200°F in order to limit radiative heat transfer to the vehicle. No attempt was made to include vehicle heating

effects caused by the engine exhaust products during firing. The only heat contribution was assumed to be from the hot engine exhaust nozzle.

Preliminary tests of the radial reaction jet performed at Marquardt indicated that steady state temperature for the radiation shield facing the engine ranged from 1200°F at a point opposite the nozzle exit to 1400°F at a point opposite the throat. An analytical model was assembled and steady state heat balances made to determine engine assembly external temperatures for various insulation configurations. With the original opening of approximately 3 inches in the solar panel, Marquardt tests have indicated temperatures of approximately 900°F at the external shroud of the engine assembly. The high external temperatures probably resulted from radiation or conduction from the nozzle tip to the external shroud, across the uninsulated gap. Since the opening in the solar panel has been increased to over 5 inches, the insulation can be extended to the nozzle tip, and maximum temperatures in the region of 200°F at the external shroud are expected. This assumes that thermal conduction from the inner radiation shield to the external shroud can be eliminated or reduced to a negligible quantity.

- Solar Panels. Calculations have been performed on the Advanced Syncom solar panels to determine the range of expected temperature. It is desirable to reduce the temperature level of the panel to increase panel electrical output. Since the orbital temperature level of the Advanced Syncom solar panel is basically a function of the solar absorptance and IR emittance of the panels and the degree of thermal coupling to the vehicle, investigations have been made to determine the effect of varying the surface properties. Although the solar cell properties themselves are the major factor in determining the

panel temperature, the effect of the surface properties of the material between and around the individual cells is sometimes significant.

The surface substrate for the Syncom 2 solar panels is a black paint (PT-401), of surface properties $\alpha_{\text{solar}} = 0.92$ and $\epsilon_{\text{IR}} = 0.85$. If an inorganic white paint such as the Hughes white is used for the Advanced Syncom solar panel painted regions (which account for nearly 10 percent of the total surface area), the temperature of the panels can be reduced nominally by 10° to 15°F based on the surface properties of $\alpha_{\text{solar}} = 0.18$ and $\epsilon_{\text{IR}} = 0.85$. As the thickness of the solar cell cover glass increases, however, the significance of the painted areas diminishes to a negligible amount.

Preliminary estimates have been made of the eclipse cooldown rates for the Advanced Syncom solar panel and battery. Figure 5-219 indicates maximum expected cooling rates based on a 70 minute eclipse and previously calculated Syncom 2 eclipse transient temperatures.

- Apogee Motor. A thermal analysis of the energy transfer mechanism between the Advanced Syncom vehicle and apogee motor has

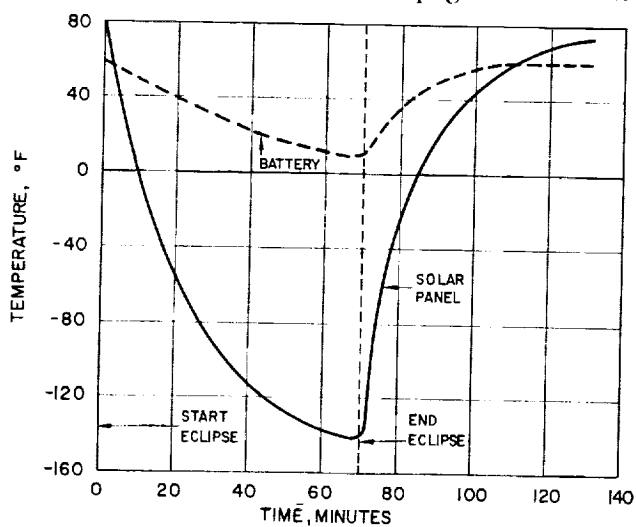


FIGURE 5-219. POWER SYSTEMS ECLIPSE TEMPERATURES

been made with emphasis toward determining: 1) the need for radiation insulation between the apogee motor and the vehicle thrust tube and 2) the need for a conduction insulation ring at the apogee motor mount point. Both types of insulation have been investigated during two modes of spacecraft operation: apogee motor phase and orbital phase. The analysis utilized a ten-node model shown schematically in Figure 5-220. The apogee motor, mounting ring, and vehicle are simulated in an attempt to determine effectiveness of both radiation and conduction insulation in the system. The figure indicates the vehicle temperature response to the given apogee motor firing transient input. Although the analysis was only approximate in that the thermal network was somewhat coarse and the nodal points did not all reach their maximum temperatures in the chosen time period, it is felt that there is insufficient justification for the insulation ring requirement.

The analysis also investigated the need for the insulative motor mounting ring. It was determined that the heat loss from the vehicle through the apogee motor is low (on the order of 1 to 2 watts) without the insulating ring; hence again, it is felt that no justification exists for the ring from a thermal standpoint.

The effect of the exhaust of the apogee motor on the forward end of the Advanced Syncom spacecraft has been partially investigated. The total thermal effect from a solid rocket engine is a combination of convective heat flux from the hot gases and radiative heat flux from the hot metal oxide particles in the exhaust plume of the engine. A preliminary investigation of the convective effects of the Advanced Syncom apogee engine has indicated that convective heating of the forward end of the vehicle will be of low magnitude.

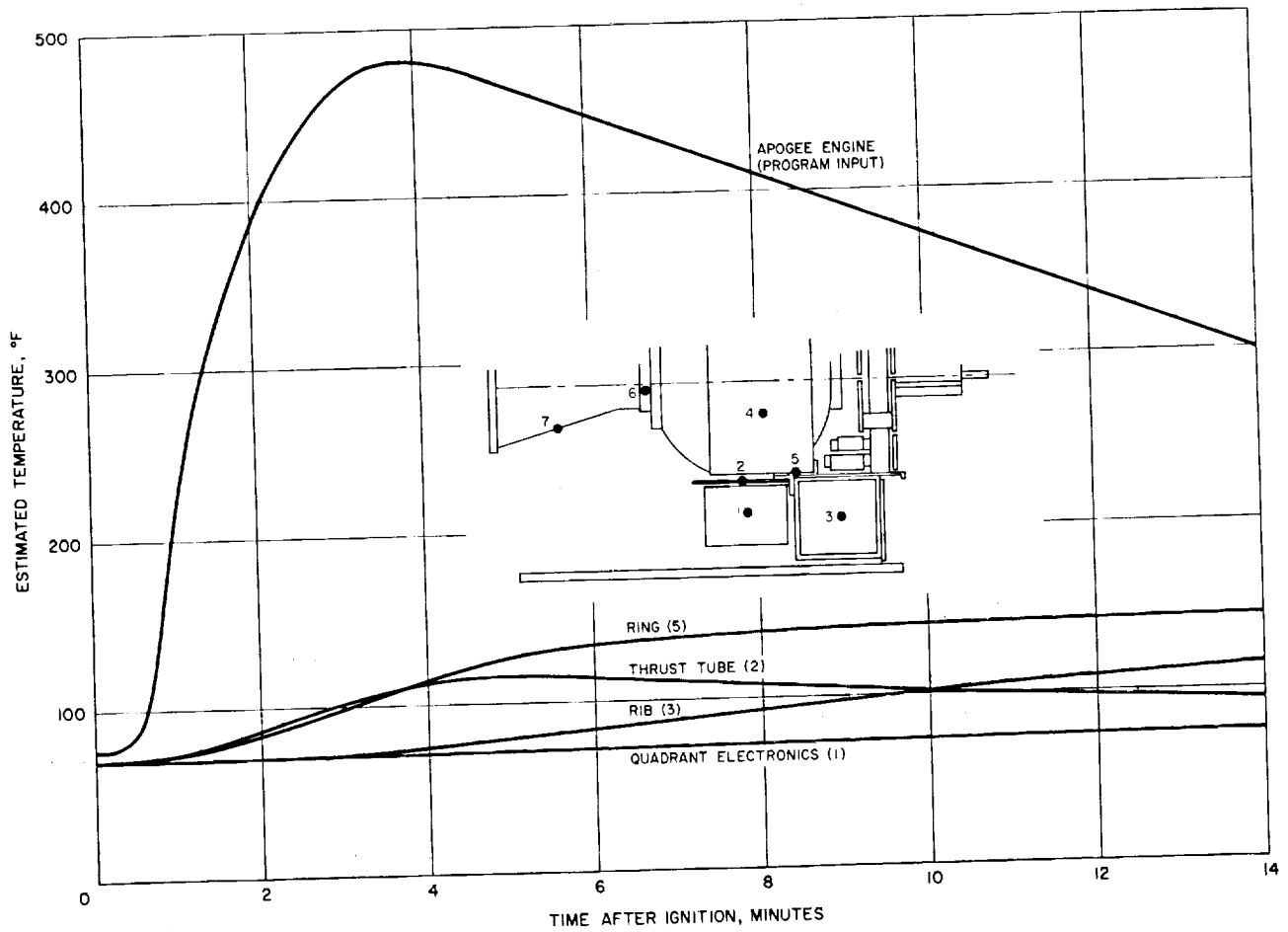


FIGURE 5-220. ESTIMATED TEMPERATURE RESPONSE DURING APOGEE ENGINE FIRING

An analytical method of treating the problem of particle radiation from a solid rocket engine exhaust plume has been devised and shown to provide reasonable agreement with experimental data where fluxes as high as 30 Btu/sec ft² were measured. Radiation from the exhaust of solid propellant engines is thought to originate mainly from metal oxide particles formed in the combustion process. The analytical approach presupposes that these particles are the sole source of radiation and that the total exhaust radiation intensity may be computed simply by summing the intensities of the individual particles.

Preliminary calculations have been made for the Advanced Syncom vehicle which indicate a maximum radiation flux level of 2 Btu/sec ft² in the region of the sun sensors and axial jets. They are based on the JPL apogee motor with the current loading of aluminum in the propellant. Based on this flux rate and a burning time of 46 seconds, a temperature rise from 80°F to a maximum of 250°F is expected to occur in the sun sensors because of apogee motor firing. Although the results of Syncom 2 apogee motor firing tests at Tullahoma, Tennessee, in June 1963 indicated an apparent temperature

reportedly reached by the sun sensors considerably higher than 250°F, failure did occur in one of the seven sensors. Consequently, further investigation will be made to determine if an immediate heating problem exists in the sun sensors. The expected heating of the reaction system axial jet in the same radiant flux field has not as yet been considered.

- **High Gain Antenna.** The apogee motor insulation analysis has been extended to include the high gain antenna and cruciform portion of the vehicle structure. The intent of the analysis is to determine the temperature range of antenna and cruciform region during normal orbital operation and to determine the need for radiation insulation between the apogee motor and cruciform for protection of the antenna components during apogee motor firing. Network input parameters have been calculated for ultimate computer solution of both the transient and steady state problem.

- **Temperature Sensors.** Coordination meetings have been held to discuss the location of Advanced Syncom temperature sensors. A total of 16 sensors will be included on the vehicle. Tentative general location of the sensors has been established and is shown in Table 5-49. The fuel and oxidizer tank sensors are repeated in each of the two telemetry encoders, providing 11 temperature readings with each encoder.

REACTION CONTROL SYSTEM

Development of the reaction control system for the Advanced Syncom satellite has been subcontracted to the Marquardt Corporation. Development began on 7 February 1963 and has proceeded through a phase I and an interim program covering demonstration of system feasibility and propellant depletion tests with an engineering model. A description of the reaction

TABLE 5-49. TEMPERATURE SENSOR DESCRIPTION

| Sensor Description and Location | Number of Sensors | Telemetry Encoder Number |
|---|-------------------|--------------------------|
| Reaction control system — fuel and oxidizer tanks | 4 | I and II |
| Near traveling-wave tube collector — ribs 3 and 9 | 2 | I and II |
| Solar panels | | |
| Near radial jet 1 | 1 | II |
| Near either end of vehicle | 1 | I |
| Antenna cruciform | 1 | I |
| Sun sensor | 1 | II |
| Aft bulkhead | 1 | I |
| Battery | 1 | I |
| Apogee motor mounting point | 1 | II |
| Radial jet 1 mounting surface | 1 | II |
| Forward shelf near telemetry transmitter | 1 | II |
| Axial jet 1 mounting surface | 1 | I |

control system and a summary of the technical activities conducted during the interim program are given here.

System Description. The storable hypergolic liquid bipropellant reaction control system consists of two completely independent control units, each of which has equal propellant capacity for accomplishment of the spacecraft control functions. A schematic presentation of the proposed control unit (two units per spacecraft) is presented in Figure 5-221.

Each control unit consists of the following components:

- 1) A vernier velocity control thrust chamber
- 2) A spacecraft orientation and spin rate control thrust chamber
- 3) Two spherical fuel/propellant pressurant tanks
- 4) Two spherical oxidizer/pressurant tanks

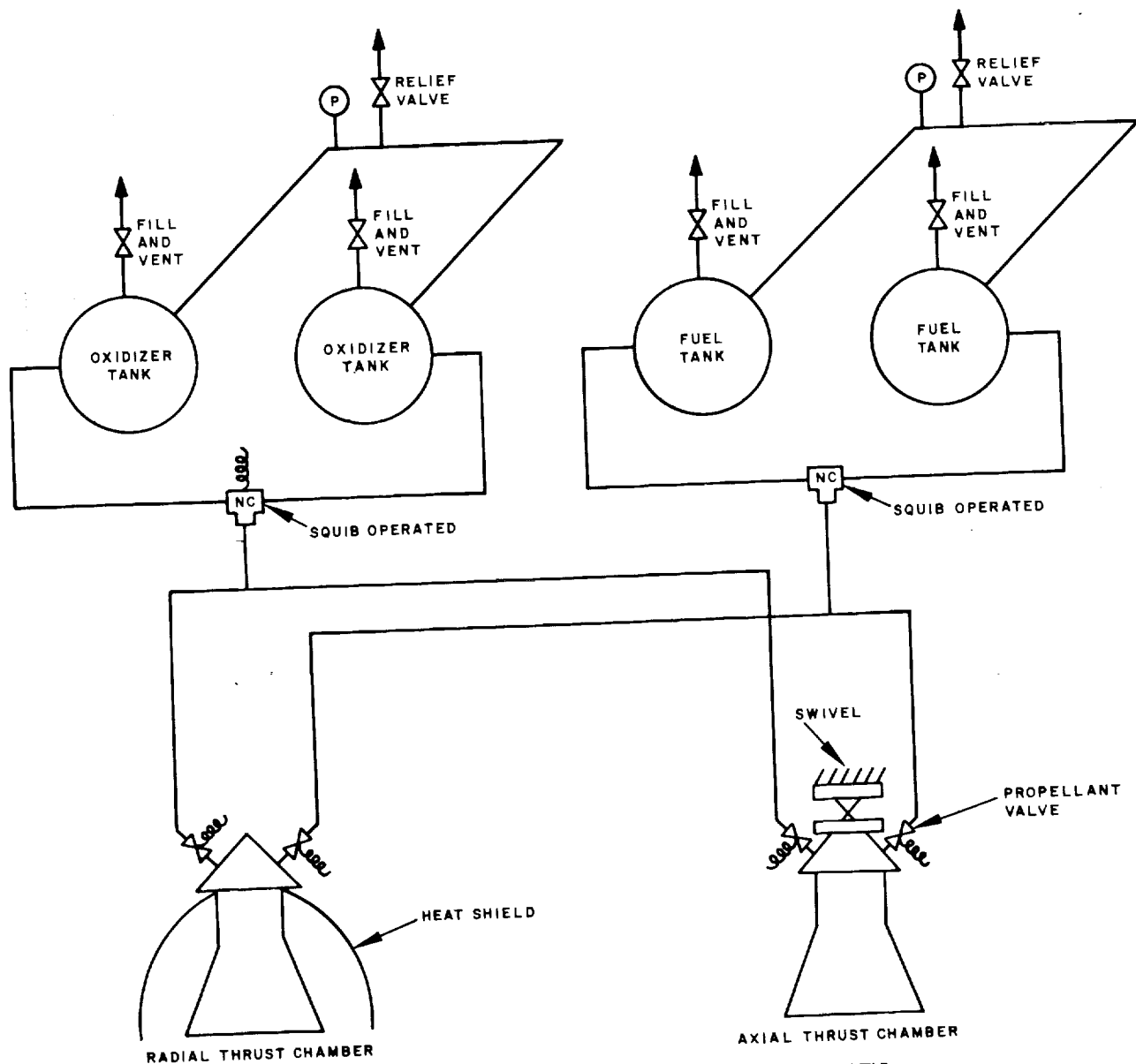


FIGURE 5-221. REACTION CONTROL SYSTEM SCHEMATIC

- 5) Associated propellant supply lines and tank pressurant fill and vent lines
- 6) Fluid filters
- 7) Fill, drain, vent, and isolation valves
- 8) A tank pressurant relief device (burst diaphragm)

- 9) Appropriate instrumentation for in-flight monitoring of propellant system pressures and temperatures.

Each of the thrust chambers referred to here consists of a combustion chamber, expansion nozzle, injector head, and two propellant control solenoid valves with integral fixed flow orifices.

The two thrust chambers provided in each control unit are both functionally and dimensionally identical. These thrust chambers are designed for 5 pounds nominal vacuum thrust at 90 psia combustion chamber pressure with a 40:1 exit nozzle expansion ratio. The combustion chamber ($L^* = 13$ inches) and expansion nozzle are radiation cooled and made from molybdenum alloy, molybdenum-disilicide coated. The steady-state operating temperature of the thrust chamber is 2800° F.

The injector configuration shown schematically in Figure 5-222 is a single doublet with a fixed orifice design and impingement angle to give optimum propellant momentum ratio, stream geometry, and pressure drop. The injector head also provides the mounting surfaces for the thrust

chamber, solenoid valves, heat shield (radial engine only), and assembly mounting to the spacecraft. The injector head is made from stainless steel and incorporates an insulated oxidizer passage designed to eliminate problems associated with heat soak-back to the injector head from the combustion chamber. Integral fixed orifices are incorporated in each propellant control solenoid valve to eliminate variations in required propellant flow rates resulting from variations in propellant supply line length and consequent pressure drop.

The propellant supply system of each of the control units consists of four fluid tanks (two oxidizer, two fuel) of nominally equal volume, associated propellant supply lines, fill, drain, vent and isolation valves, and filters located up-

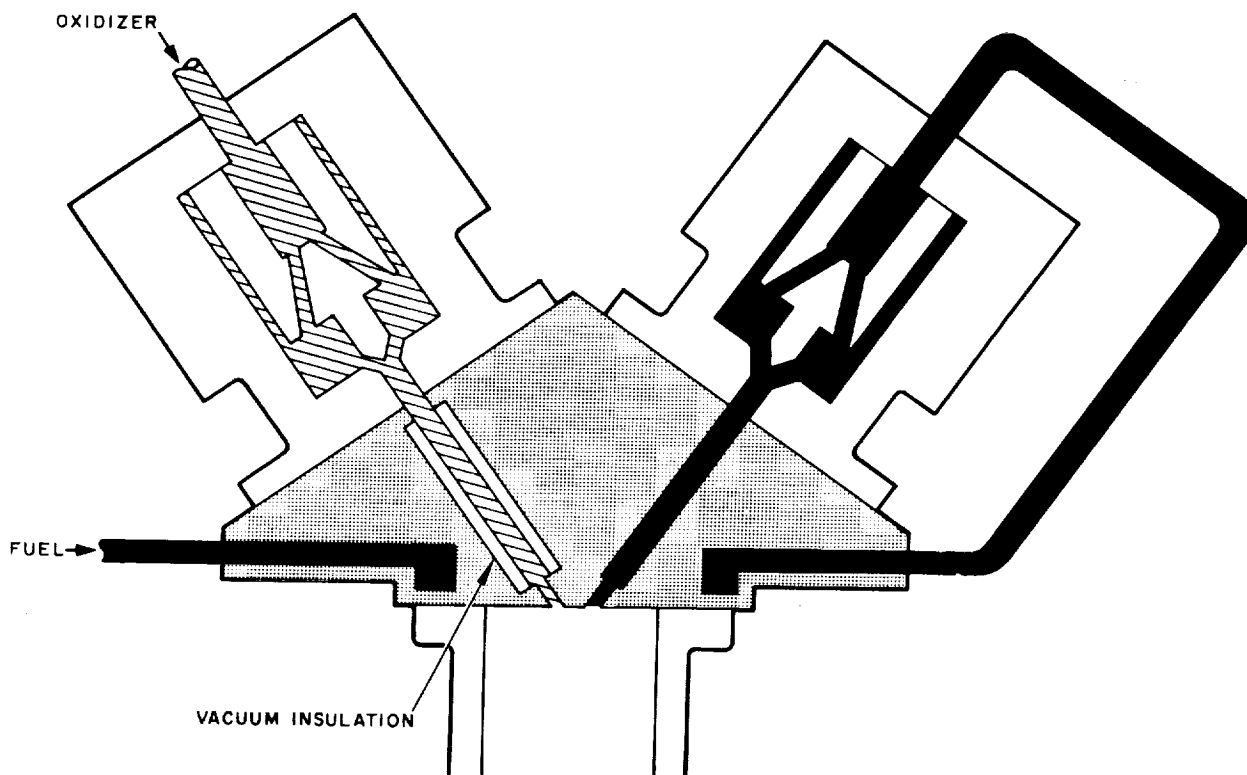


FIGURE 5-222. REACTION CONTROL SYSTEM INJECTOR

stream of each of the propellant control solenoid valves to eliminate the admittance of dirt or contamination to these valves. Each of the tanks is made from 17-7PH steel and is a thin walled vessel of 11½ inches internal diameter. Each tank is trunnion mounted to provide for pressure deflections, tolerance accumulation, and supporting structural deflection. Tank construction is accomplished by using heat-treated forged hemispheres machined to correct thicknesses and then girth welded. All tank fittings are machined integral with the respective hemisphere.

Stainless steel lines are used throughout the system for propellant supply and pressurant charging. Pressurant fill and drain valves are provided at the top of each tank, a single valve being used in each tank to accomplish both the fill and drain functions. All propellant fill and drain valves and pressurant fill and vent valves are capped to minimize leakage of propellants or pressurant. It should be noted that teflon parts used in these valves are adequately protected from space vacuum and space radiation effects by the capped metal body in which they are contained. A three-way valve (multiple inlet, common outlet, with all three ports normally closed) is provided in both the oxidizer and fuel systems to prevent the flow of propellants to the propellant control solenoid valves before system use. This three-way valve, operated by either of two squibs, also serves to eliminate the possibility that propellants will be transferred between pairs of propellant tanks (between two fuel tanks or between two oxidizer tanks) during maneuvers associated with spacecraft orbit injection and orientation.

Each common pair of tanks has a relief valve/burst disc assembly welded to the pressure equalizing line. The purpose of this device is to eliminate any possibility of catastrophic failure of the

propellant tanks caused by over pressurization and attendant destruction of the spacecraft.

The valve assembly, shown in Figure 5-223, consists of a burst disc and relief valve in series with a filter between them. The disc, which forms a gas tight welded seal, is exposed to the system gas pressure on one side and ambient pressure on the other side. The disc is designed to burst within a pressure range well above the nominal charging pressure of the system but still much lower than the tank proof pressure. Thus, an abnormal pressure excursion is required to burst the disc.

Downstream of the disc, the filter serves to trap any fragments resulting from the bursting of the disc and to protect the relief valve from system contaminants. The relief valve does not function until the disc has burst. Its cracking and reseal pressure ranges are between the system pressure level and the disc burst pressure, permitting it to vent the overpressure which caused the disc to burst, but not allowing system pressure to decay below operating levels. A checkpoint is provided for testing the relief valve function without damaging the burst disc.

The instrumentation used for in-flight monitoring of propellant system pressures and temperatures consists of absolute pressure transducers and temperature probes. The absolute pressure transducers are of the diaphragm type and compatible with MON-5 and MMH propellants. Diaphragm-type transducers are chosen over bourdon type transducers for ease of cleaning and passivation, high frequency response, lower hysteresis, and compatibility with telemetering requirements.

A propellant combination comprised of mixed oxides of nitrogen (MON-5) oxidizer and monomethyl hydrazine (MMH) fuel was selected to satisfy the Advanced Syncom spacecraft bipro-

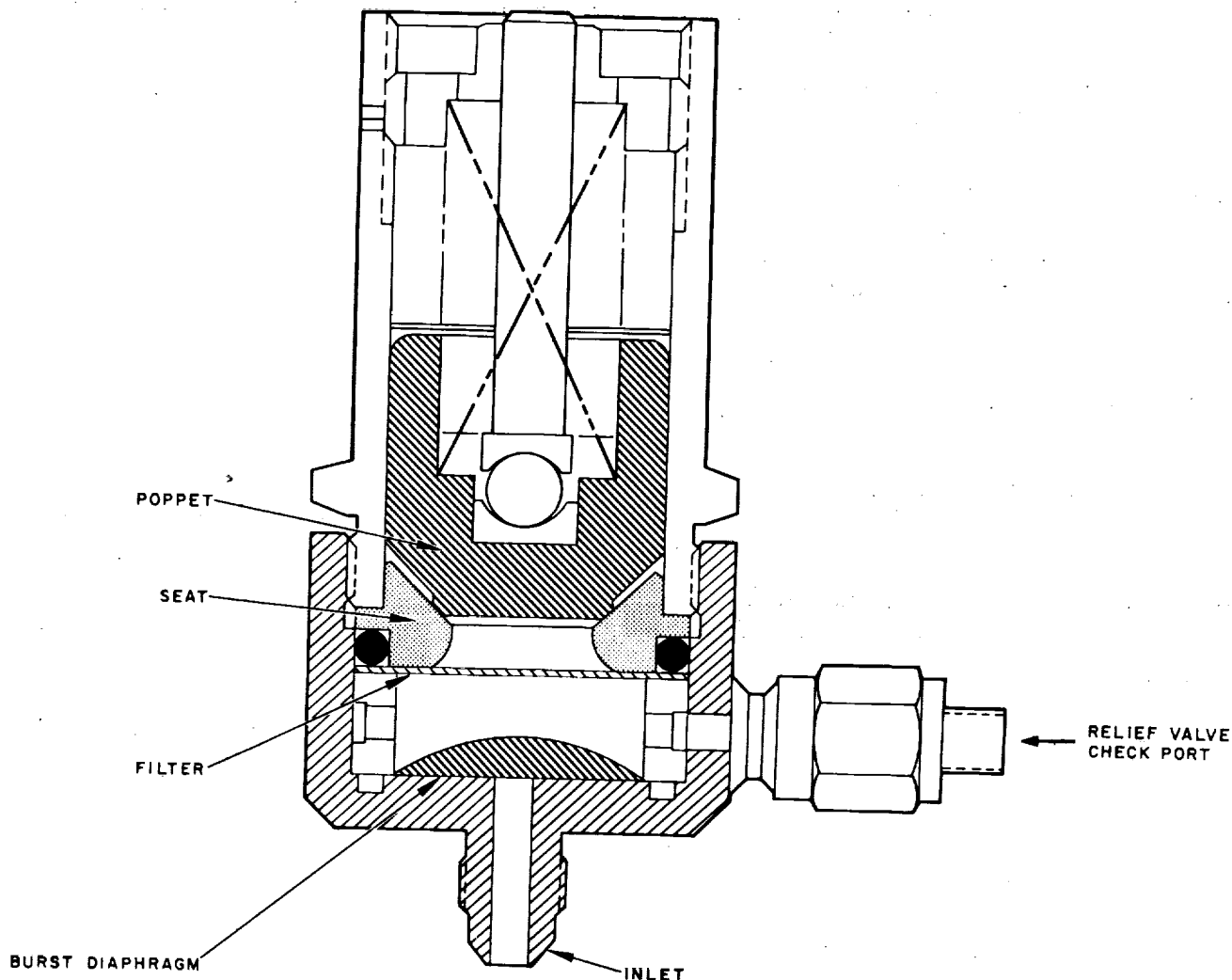


FIGURE 5-223. RELIEF VALVE ASSEMBLY

pellant reaction control system storage, environment, and performance requirements specified in Performance Specification X-254044. The nominal design oxidizer/fuel (O/F) ratio is 1.62.

The choice of 1.62 as the nominal design O/F ratio for the selected propellant combination allows the use of equal volume oxidizer and fuel tanks at equal pressurization levels to minimize O/F variation as tank pressure decays. Although

the maximum theoretical specific impulse (I_{sp}) performance of the selected propellant combination occurs at an O/F ratio of 2.25 (See Figure 5-224, the maximum theoretical characteristic velocity (C^*) occurs at an O/F ratio of 1.75 with essentially negligible variation in theoretical C^* performance down to an O/F ratio of 1.62. Experience with rocket engine configurations similar to that used here has indicated that experimental engine performance more closely

follows theoretical "frozen" flow performance than theoretical "shifting equilibrium" performance. The theoretical specific impulse is shown in Figure 5-225. This can be explained by the fact that the stay time in the engine exit nozzle is so short that essentially no recombination or, at best, limited recombination of the dissociated exhaust species occurs. Hence, maximum experimental I_{sp} performance occurs at approximately that O/F ratio where theoretical C^* performance op-

timizes, i.e. over O/F range of 1.62 to 1.75. for the selected propellant combination. Therefore, the choice of equal volume propellants (at the nominal environmental temperature of 70°F) and an O/F ratio of 1.62 optimizes delivered performance and overall system requirements.

Based on the thrust requirements and tank design considerations, an initial tank pressurant pressure of 300 psi was selected for the Advanced Syncom reaction control system. During reaction control system operation, tank pressurant pressure will decay steadily until a pressure of 149 psia is reached at the point of termination of expulsion of the propellants. It should be noted that the range of tank pressurant pressures referred to above is sufficiently high to suppress the vapor pressure of the MON-5 or N_2O_4 oxidizer and maintain it in the liquid state up to a temperature of 145° F.

The axial engine (Figure 5-226) is mounted on a centrifically operated swivel mechanism. The device controls the spin speed of the vehicle to 100 rpm. The swivel mechanism is spring-biased so that at 100 rpm the axial motor fires parallel to the spin axis and imparts no spin moment.

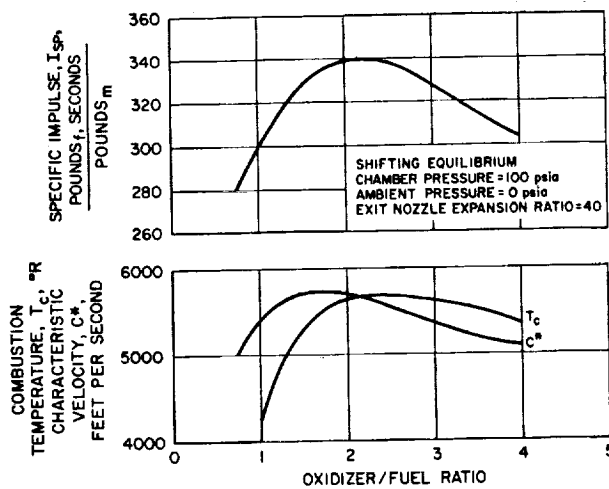


FIGURE 5-224. MAXIMUM THEORETICAL CHARACTERISTIC VELOCITY

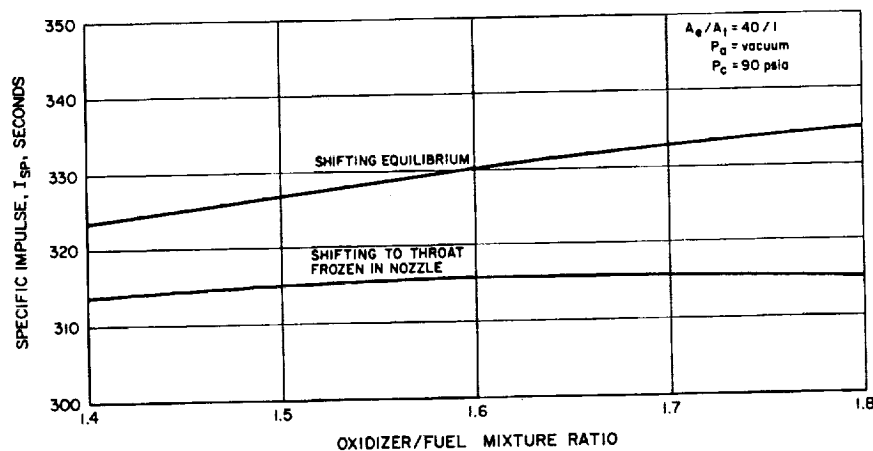


FIGURE 5-225. THEORETICAL SPECIFIC IMPULSE

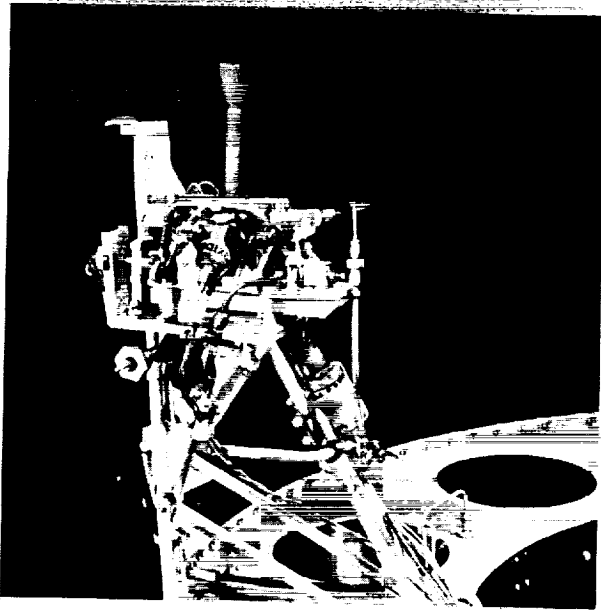


FIGURE 5-226. AXIAL ENGINE

At greater or lesser speeds, a spin moment can be imparted to the vehicle by operating the axial engine. The mechanism pivots on flexures thereby eliminating any friction. Initially the mechanism is locked in the axial position and is released after spin up by a squib-actuated device. The swivel mechanism is shown in Figure 5-227.

Development Status. The development status of the Advanced Syncom bipropellant reaction control system with respect to current spacecraft requirements is summarized in this section. Marquardt Corporation recently completed the advanced development phase of the Advanced Syncom reaction control system development and fabrication program. The final milestone of this phase was a hot firing acceptance test of an engineering model with a reaction control unit installed in a Hughes-supplied spacecraft structure subassembly.

Subsequent changes in the design of the spacecraft structure will require redesign of some reaction control system components. In addition,

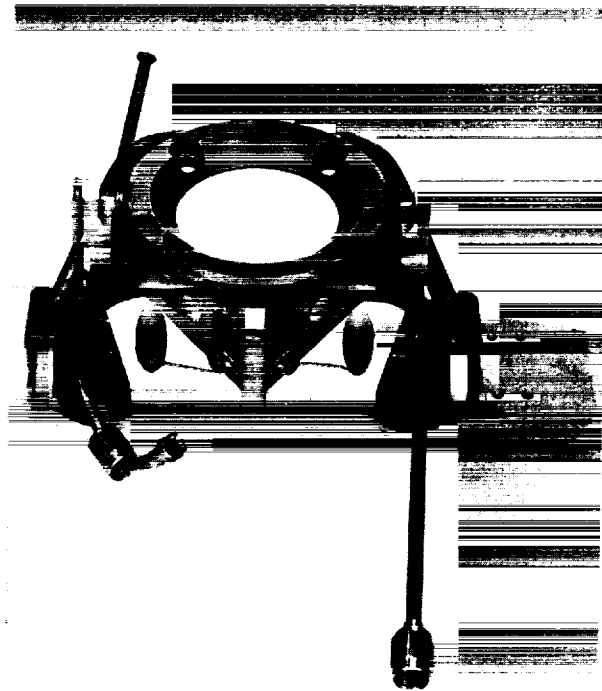


FIGURE 5-227. SWIVEL MECHANISM

certain components will require further refinements to increase the reliability for flight models.

The following discussion describes the current engineering model reaction control system, the requirements for flight models, necessary changes for system components, and additional developmental testing required.

Performance. A tabulation of the current Advanced Syncom engineering model performance and that required by Hughes Specification X-254044 is presented in Table 5-50.

- **Impulse.** The reaction control system design requirement was to provide for a total velocity increment of 2300 fps to a spacecraft initially weighting 575 pounds (including the fully loaded reaction control system), plus 600 lb-sec impulse for spin control. The resultant total impulse was approximately 100 percent greater

5. Spacecraft System Design

TABLE 5-50. REACTION CONTROL SYSTEM PERFORMANCE

| Parameter | Initial Specification | Engineering Model |
|---|---|----------------------------------|
| Total impulse, pound-seconds | 33,300 | 30,600 |
| Thrust, pounds | 5 | 5 |
| Maximum | 3 | 3 |
| Minimum | | |
| Response, milliseconds | | |
| Start | 12 | 15 (approximate) |
| Decay | 12 | 15 (approximate) |
| Mission effective I_{sp} , seconds (continuous and pulsing) | 290 | 239 |
| Duration, steady state, minutes | 14 | 30 |
| Pulse centroid, repeatability, milliseconds | ± 5 (single pulse)
± 1 (50 pulses) | Less than ± 5 (not assessed) |
| Total weight (exclusive of brackets), pounds | 186 | 191 |
| Temperatures, °F | | |
| Ambient | 0 to 130 | 15 to 130 |
| Tanks | 40 to 80 | 40 to 80 |
| Heat shield | 200 | 380 |
| Electrical power | | |
| Watts | 50 | 29.5 |
| Volts | 24 | 24 |
| Reliability | 0.995 | Not determined |

than that necessary to complete the Advanced Syncom mission for 5 years. Complete system redundancy was accomplished by providing two separate propulsion units, each containing the propellant quantities and system components necessary to perform the mission. (The engineering model contains only one unit.)

Spacecraft weight after apogee motor burnout was increased to 765 pounds during the advanced development program. In addition, the required 290 mission specific impulse has not been achieved. The current mission specific impulse of the engineering model is 239 seconds, based on an 80-percent steady-state/20-percent one-sixth pulsed mode duty cycle and a thrust decay from 5 to 3 pounds. The increased spacecraft weight and reduced specific impulse, in effect, reduce the system propellant redundancy by 30 percent for a 5-year mission.

- **Durability.** The axial thrust chambers are required to operate in the continuous, steady-state mode for a 14-minute period and for random 1-minute intervals to accumulate a total life of 80 minutes. Both radial and axial engines are required to operate continuously in the pulse mode for 1½ hours.

The molybdenum disilicide coating of the engineering model engines fails at approximately 3000° F. Once this coating is removed, the base molybdenum alloy material is readily attacked by the oxidizer. Injectors designed to provide the 290 mission specific impulse produced chamber wall temperatures which exceeded the coating operating temperature in less than 5 seconds of steady-state operation. When the injector was re-designed for reduced performance (260 seconds, steady-state specific impulse), the current engineering model axial engine demonstrated two 30-minute periods and one 13-minute period, totalling 73 minutes of continuous steady-state operation.

Pulse mode operation results in a chamber wall temperature that does not exceed the operating regime of the coating; therefore, no redesign of the radial engine was necessary. During recent tests, the radial engine with heat shield demonstrated 80 minutes (clock time) of continuous pulse mode operation plus intermittent pulsed operation exceeding 1½ hour requirement.

- **Response.** A maximum response time of 12 milliseconds from start signal to 95 percent of rated thrust and from shutdown signal to 5 percent of rated thrust is required. During engineering model engine acceptance tests, a starting response time of 21 to 23 milliseconds was demonstrated with the radial engine. The reworked axial engine demonstrated 45 to 50 milliseconds starting response. Shutdown responses of approximately 13 milliseconds for the radial and 35

milliseconds for the axial were recorded. Some of the recorded time can be attributed to instrumentation response. A tradeoff exists between the starting and shutdown response by varying the spring rate of the solenoid valve. In addition, starting responses can be improved by relocating the solenoid valve calibrating orifice. Therefore, radial and axial engine response times are expected to be proven within the 12-millisecond requirement by employing these adjustments and using high response measuring techniques. Both radial and axial engines have demonstrated a centroidal pulse center repeatability of less than ± 5 milliseconds between consecutive pulses.

Component Design

- **Thrust Chamber.** The engineering model feasibility tests were conducted during the advanced development phase using a thrust chamber configuration that delivered 280 seconds impulse (measured) at the 5-pound thrust level. The engines were found incapable of operating in a continuous duty cycle because of the high heating rates associated with the design. To alleviate this condition, a one-third duty cycle (1 second on, 2 seconds off) was chosen for firing the axial engine during initial tests of the engineering model. The radial engine is required to operate only in a one-sixth duty cycle and did not present an overheating problem. The engineering model system feasibility tests were completed with these duty cycles without incident.

During a subsequent test phase at Marquardt, additional system testing was required on the engineering model in which the axial engine would operate in a steady-state mode. The combustion efficiency of the radial engine was reduced by increasing the fuel injection orifice from 0.016 to 0.030 inch. Proper mixture ratio control was maintained by adjusting the upstream orifices.

At the 5-pound thrust level, specific impulse varied from 180 to 260 seconds, with evidence of erratic burning occurring in the combustion chamber. The phenomenon was caused by the low pressure drop on the fuel side, which allowed dynamic coupling between the combustion process and the feed system. The fuel passage on a second injector was drilled to 0.023 inch. This new configuration was tested and produced stable burning and a performance level of 248 seconds at the 5-pound thrust level. This axial engine was chosen to be used on the engineering model for the two depletion tests of the interim program. The radial engine remained unchanged from the phase I effort in which the fuel orifice was 0.016 inch and the oxidizer orifice was 0.019 inch diameter.

A parallel research and development effort at Marquardt has produced a thrust chamber which is suitable for use in flight vehicles. A comparison of the research and development chamber and that used in the engineering model is shown in Figure 5-228. The changes consist of shortening the chamber length (from 3 inches for the engineering model thrust chamber to 1.17 inches for the improved chamber) and adding a short propellant premix section to the research and development chamber.

The engine thrust is 5 pounds with a premix chamber pressure of 106 psia. The propellant supply pressure corresponding to this thrust level is 300 psia. Since the system is a pressure decay type, thrust will decrease in relation to the supply pressure. The final propellant supply pressure is 149 psia, producing a terminal thrust of 3 pounds at a chamber pressure of 69 psia. The thrust decay range of 5 pounds to 3 pounds, resulting from a decrease in tank pressure from 300 psia to 149 psia, results in a maximum possible total engine burning time of approximately 4260 seconds, as shown in Figure 5-229. Thrust

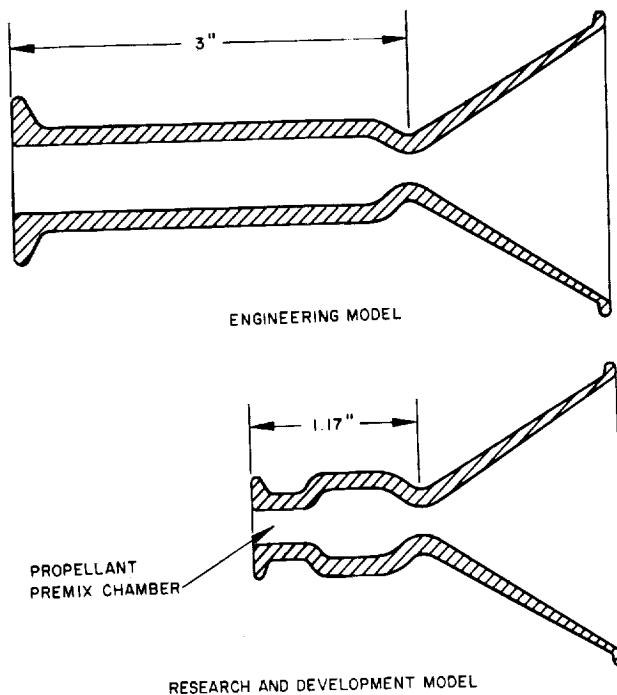


FIGURE 5-228. THRUST CHAMBER COMPARISON

chamber life in excess of 110 minutes has been documented for the Advanced Syncom configuration.

Figure 5-230 presents specific impulse versus thrust level as measured in altitude tests during operation for the research and development engines. The maximum, minimum, and nominal curves represent the test data scatter measured during tests of this engine. The measured specific impulse performance for the proposed engine decreases from 280 seconds (nominal) at the 5-pound steady-state thrust level to 248 seconds (nominal) at the 3-pound steady-state thrust level. In the pulsing mode, the specific impulse varies between 240 seconds at the 5-pound thrust level to 190 seconds at the 3.0-pound thrust level. Using the minimum level of the steady-state data and integrating over the curves shown in Figure 5-230, a minimum mission average I_{sp} of 250 seconds is obtained.

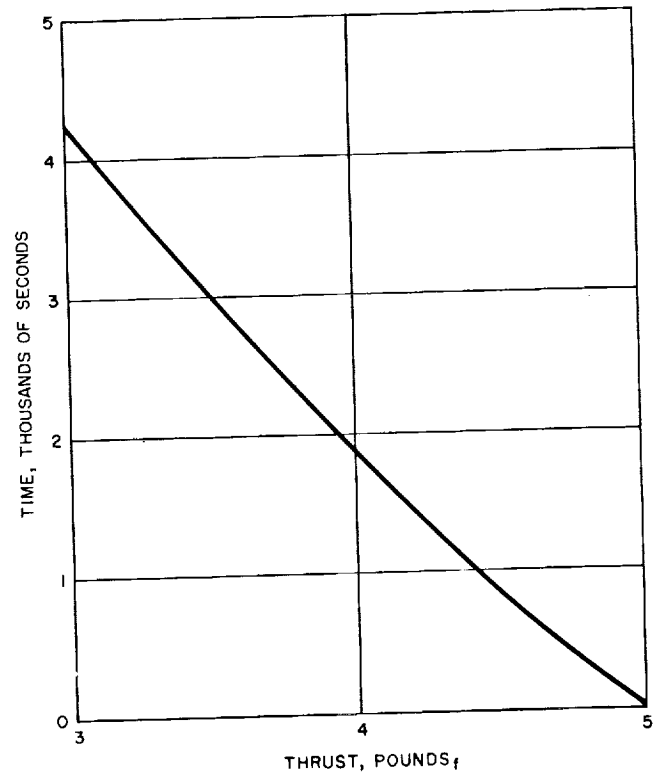


FIGURE 5-229. MAXIMUM POSSIBLE TOTAL ENGINE BURNING TIME

Figure 5-231 presents measured thrust chamber wall temperature of the unshielded engine at the maximum temperature region along the combustion chamber wall. The maximum combustion chamber temperature is 3000° F. These temperatures were determined during steady-state firing operations only. The 0.1-second ON time for the pulse mode is not sufficient to stabilize chamber wall temperature. Therefore, the maximum temperature of 3000° F will not be reached during the Advanced Syncom mission. The first steady-state firing correction begins at a thrust level of 4.6 pounds which corresponds to a chamber wall temperature of 2880° F. At the end of the first steady-state firing correction, the chamber wall temperature has dropped to 2800° F. Operation at these temperatures provides high confidence that the thrust chamber will demon-

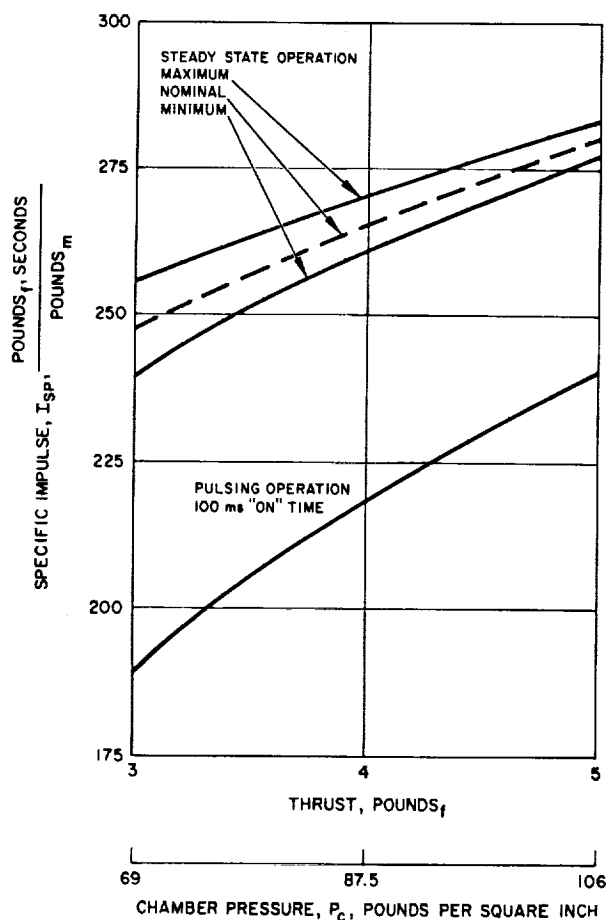


FIGURE 5-230. SPECIFIC IMPULSE VERSUS THRUST LEVEL

strate the required durability for performing the Advanced Syncom mission.

The current engineering model axial and radial engine performance results in a mission average specific impulse of 239 seconds. Higher performance can be obtained with these engines only at the expense of decreased thrust chamber life. The improved thrust chamber, however, provides a minimum mission average I_{sp} of 250 seconds which is equivalent to a propellant gain of approximately 10 pounds. In addition, the improvements will provide a reduction in re-

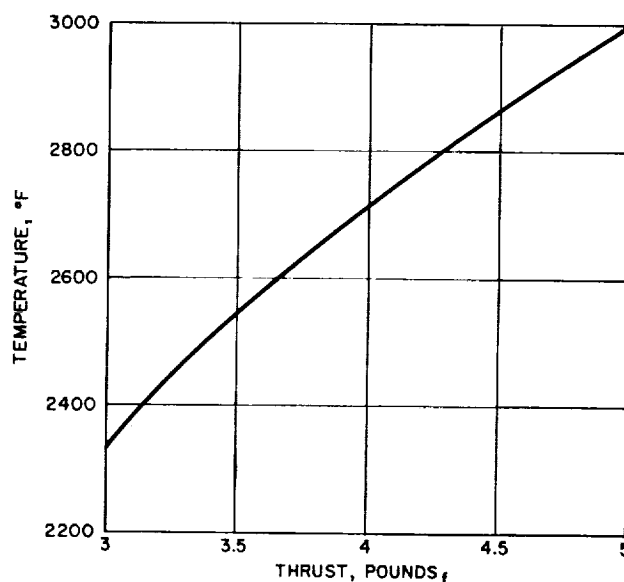


FIGURE 5-231. MEASURED THRUST CHAMBER WALL TEMPERATURE

sponse time and will alleviate spacecraft installation and mounting problems.

- **Solenoid Valves.** Solenoid valves (Figure 5-232) currently used on the engineering model engines require two changes to meet Advanced Syncom flight model specifications. The existing design must be altered to include a filter element, and an all-welded design must be developed. Current engineering model valves were brazed to facilitate assembly. The braze material used is not compatible with the propellant long-term storage requirement.

- **Tanks.** Reconfiguration of the Advanced Syncom structure requires corresponding changes in propellant tank design. The engineering model tanks, currently mounted in a plane perpendicular to the spin axis, are loaded through an aft quick-disconnect fitting on the tank and pressurized through a forward quick-disconnect on the pressure equalizing line. The redesigned structure will require tanks mounted fore and aft, with

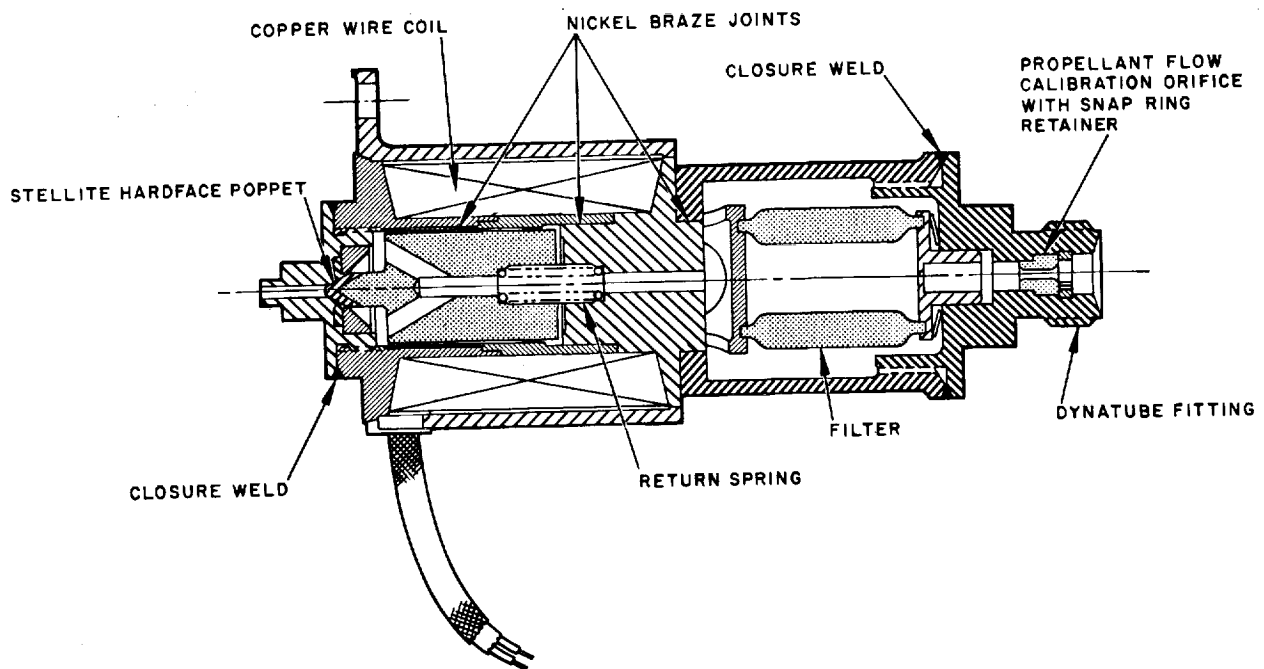


FIGURE 5-232. SOLENOID VALVES

one common loading and pressurizing fitting concentric with the forward tank mount.

The engineering model tanks leaked through the flanges of the 17-7 PH material fittings during testing. The leakage was attributed to porosity resulting from the elongated 17-7 grains being oriented transversely to the flange. The tanks were temporarily fixed by gold plating. Flight hardware tanks with fittings fabricated such that the flange grain structure is oriented longitudinally will be constructed.

- **Heat Shield.** A heat shield is required to limit radial engine envelope temperatures as presented to surrounding structural elements to 200° F. Based on preliminary thermal analysis, a 5-inch-diameter heat shield envelope, including insulation, is necessary to produce the required thermal gradient. The engineering model structure provided only a 3-inch-diameter hole in the solar panels through which the heat shield must

protrude; therefore, most of the insulation in the vicinity of the panels had to be eliminated. The redesigned structure includes a 5½-inch-diameter hole. Although heat shield measurements were made, a rigorous evaluation of the heat shield was not performed during the advanced development phase. Tests must be conducted to verify analytical evaluation.

- **Squib Valves.** The engineering model reaction control system contains a three-way squib-actuated propellant pre valve. The purpose of this valve is to eliminate the possibility of propellant transfer between common tanks during prelaunch and launch phases and to serve as a backup safety device preventing premature or inadvertent firing of the engines.

A disadvantage of the current valve design is that without firing the squib actuator, it is not possible to conduct a complete systems check. In addition once the squib is fired, the valve can-

not be reused. Two alternatives are being considered to assure adequate preflight systems checks: 1) employ reusable squib-actuated valves or latch solenoid valves instead of the squib valves currently used and 2) eliminate the pre-valve entirely. The second approach required confirmation of range safety requirements with respect to the Advanced Syncom and a detailed study of propellant transfer characteristics during launch.

- **Spin Control Mechanism.** The engineering model spin control mechanism, illustrated in Figures 5-226 and 5-227, consists of the axial engine assembly, the rigid bar frame to which it is mounted, a base plate, and a vibration damper. The bar frame is linked to the base plate by flexure pivots and torsion bar springs. This type of spring bias allows for transmittal of propellants through the bars without the tendency of stress loading the lines at the solenoid and spacecraft interfaces. During the launch sequence, the mechanism is locked in the nominal or 100-rpm position (thrust axis parallel to the spin axis) by two squib-actuated pin-pullers. To enhance system reliability, the use of one pin-puller will be investigated during the flight hardware phase.

Currently, a base plate adapter is required at the spacecraft interface. A more optimum base plate-to-spacecraft attachment will be developed.

The reaction control system spin control mechanism employs a friction damper for engineering model demonstrative purposes only. Various alternative techniques for damping the mechanism have been investigated; however, final selection of a method has not been made.

Weights. A summary of the existing reaction control system weight status and projected changes is given in Table 5-51.

TABLE 5-51. REACTION CONTROL SYSTEM WEIGHT

| Component | Quantity/
Unit | Weight/
RCS Unit,
pounds | Predicted
Flight
Weight,
pounds |
|---|-------------------|--------------------------------|--|
| Engine | 2 | 2.45 | 2.45 |
| Heat shield | 1 | 1.02 | 1.07 |
| Squib valve | 2 | 1.40 | * |
| Fill and drain | 4 | 0.96 | 0.96 |
| Fill and vent | 2 | 0.48 | * |
| Relief valve | 2 | 1.38 | 0.76 |
| Pressure transducer | 2 | 0.46 | 0.10 |
| Temperature transducer | 2 | 0.01 | 0.01 |
| Filter | 2 | 1.26 | 1.26 |
| Variable restrictor | 2 | 0.48 | 0.23 |
| Spin control mechanism | 1 | 2.30 | 1.66 |
| Tank | 4 | 12.00 | 13.48 |
| Tubing, fittings | — | 3.25 | 3.25 |
| Dry unit weight | | 27.45 | 25.23 |
| Dry system weight | | | 50.46 |
| Propellant | | | 130.00 |
| O/F error | | | 3.03 |
| Unusable propellant in lines | | | 0.45 |
| Expulsion efficiency (½ percent) | | | 0.64 |
| Propellant loading accuracy (½ percent) | | | 0.64 |
| N ₂ weight | | | 1.36 |
| Total system weight | | | 186.58 |

*To be removed on future units.

Interim Program. Presented here is a synopsis of the final report of the interim test program conducted at Marquardt Corporation on the Advanced Syncom engineering model reaction control system. The contracted items of work were to modify an axial engine to increase its durability and to conduct two propellant exhaustion tests on the engineering model at altitude conditions. The propellant exhaustion tests were to provide mission impulse, mixture ratio, expulsion efficiency, and spacecraft temperatures, as well as operational experience on the system.

The axial engine on the engineering model as delivered in phase I could not meet specification steady-state durability requirements, although at 5 pounds thrust, specific impulse was 280 seconds. To increase durability, combustion efficiency was reduced by increasing the fuel injection orifice from 0.016 inch to 0.030 inch diameter, whereupon the fuel pressure drop decreased from 55 psi to approximately 10 psi.

An orifice was installed upstream of the injector to provide proper mixture ratio control, and the engine was fired at the 5-pound thrust level. Analysis of the performance data gave specific impulses ranging from 180 seconds to 260 seconds with evidence of combustion instability. This phenomena was caused by the low injector pressure drop, which allowed dynamic coupling between the combustion process and the fuel feed system. A second injector fuel passage was drilled to 0.023 inch. This new configuration was tested and produced stable burning and a performance level of 248 seconds at the 5-pound thrust level. As flow and thrust were decreased (and the injector pressure drop decreased), some resonant burning was observed but at a much lower thrust level than on the previous configuration. This engine was used on the engineering model for the two propellant exhaustion tests. The radial engine had an injector configuration that delivers a specific impulse of 280 seconds at the 5-pound thrust level. The fact that this chamber operates only in a pulsed mode permits operation periods in excess of specification durability requirements.

Prior to testing, the engineering model was instrumented such that the following data could be obtained: chamber pressure, fuel and oxidizer tank pressures and temperatures, six structure interface temperatures, swivel mount angular position versus spin speed, and propellant valve current traces. In both tests the model was loaded with 41 pounds of oxidizer and 25 pounds of fuel. The tests were conducted at a cell pressure of less than 0.15 psia, while the system was rotated at a nominal 100 rpm. The tests were run to simulate a worst case Syncom mission in the following sequence:

- 1) 8000 pulses in a one-sixth duty cycle (0.1 second on, 0.5 second off) with the radial engine

- 2) 1200 pulses in the one-sixth duty cycle with the axial engine
- 3) 14 minutes of steady-state firing with the axial engine
- 4) One pulse of 0.1 second duration with the radial engine
- 5) 34 seconds of steady-state firing with the axial engine
- 6) Alternate 4 and 5 above to depletion of the fuel supply.

The first system test was conducted without incident. Pressure versus thrust data, generated from individual rocket acceptance tests, were used to convert the system chamber pressure to an equivalent specific impulse. Upon summing up all the impulse bits, a total mission impulse of 17,112 lb-sec for the first test was obtained. This gives an average mission impulse of 259 seconds for the loaded propellant weight of 66 pounds. At the conclusion of the test, it was found that 3.75 pounds of oxidizer remained and the fuel tanks were empty, giving a mission oxidizer-to-fuel mixture ratio of 1.49 for the complete test. The test procedure required both fuel and oxidizer tank pressures to be set to 300 psia; however, the fuel tank was pressurized to 264 psi and the oxidizer tank to 294 psi. These errors were attributed to a faulty pressure gage in the loading equipment.

Thermocouples were located on the radial engine heat shield, the axial engine swivel mount, the propellant valves, and the thrust mounts. During the 8000 pulses of the radial engine, the outer surface of the heat shield rose to 370° F. This is higher than the Hughes requirement of 200° F; however, this value will be significantly reduced by changes incorporated in the flight design.

During the second propellant exhaustion test, the radial engine chamber pressure began to decrease after approximately 200 pulses and reached zero after 1000 cycles. Postrun inspection of the radial engine showed the teflon valve seat to be severely extruded into the flow passage, accounting for the behavior of the engine. Cold flow of the Teflon seat had been experienced previously. The valve design has been changed to incorporate a thinner Teflon section. This modification to the radial engine will be made before delivery of the engineering model.

The depletion test was completed with the axial engine, using a duty cycle of 1 minute on and 30 seconds off. The engineering model is shown during test in Figure 5-233 and after testing in Figure 5-234. The mission specific impulse for the second test was 249.7 seconds. Oxidizer remaining was 1 pound and all the fuel was expended. The mission average mixture ratio was 1.60.

The initial tank pressures were also low for the second depletion test, being 284 psi and 274 psi, respectively, for the fuel and oxidizer tanks.

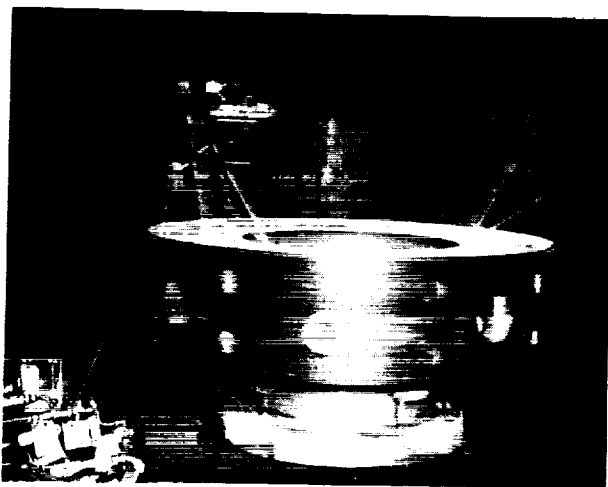


FIGURE 5-233. ENGINEERING MODEL DURING TEST NO. 2

The fact that the tank pressures were more nearly equal in the second test accounts for the lesser amount of residual oxidizer and the greater accuracy of mixture ratio.

The axial engine is mounted on a swivel mechanism, which directs the thrust of the engine to control vehicle spin speed. The swivel mechanism is designed to give no spin moment at the nominal speed of 100 rpm. The null speed of the swivel mechanism on the engineering model was 85 rpm. There is no provision for adjusting the existing design. Since manufacturing tolerances could not be held close enough to meet specification requirements, an adjusting mechanism has been added to the design for future units. Table 5-52 presents a summary of system performance for the two exhaustion tests.

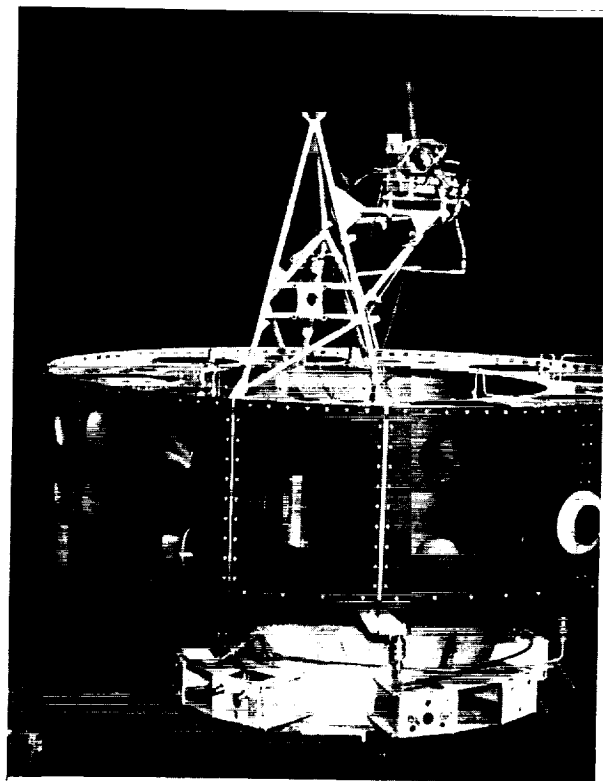


FIGURE 5-234. ENGINEERING MODEL AFTER TEST

TABLE 5-52. SYSTEM PERFORMANCE SUMMARY

| | Test 1
19-20
September
1963 | Test 2
25
September
1963 |
|---|--------------------------------------|-----------------------------------|
| Propellant load, pounds | | |
| Oxidizer | 41.0 | 41.0 |
| Fuel | 25.0 | 25.0 |
| Propellant off load, pounds | | |
| Oxidizer | 3.75 | 1.0 |
| Fuel | 0.0 | 0.0 |
| Mission O/F | 1.49 | 1.60 |
| Total engine burning time, seconds | 4537.6 | 4329.3 |
| Mission total impulse, lb-sec | 17112.89 | 16479.21 |
| Mission average specific impulse, seconds | 259.3 | 249.7 |
| Percentage of total engine burning time | | |
| Axial engine | 82.04 | 96.05 |
| Radial engine | 17.96 | 3.95 |
| Temperatures (maximum), °F | | |
| Heat shield | | 370 |
| Fuel valve | | 255 |
| Radial engine (attach. boss) | | 270 |
| Swivel mount | | 185 |
| Spacecraft (axial engine truss) | | 145 |

APOGEE ENGINE

System Description. The Advanced Syncom apogee engine is to be supplied as GFE/NASA and is being designed and fabricated by JPL. The design evolved as a direct scale-up of the Syncom 2 apogee engine with the following exceptions:

- 1) The nozzle is bolted to the case flange rather than threaded.
- 2) Attachment to the spacecraft will be by means of shear bolts rather than tension bolts.
- 3) The igniter will be a pyrogen rather than a pyrotechnic type.
- 4) A contour nozzle will be utilized rather than a conical nozzle.

The apogee engine is diagrammed in Figure 5-235. The engine consists of a 410 chromium steel case, with a rolled and welded cylindrical

section, joined to drawn and machined elliptical dome ends. The case is lined with a nitrile-base rubber material (Gen-Gard V-52). Attachment to the spacecraft thrust tube will be by means of shear bolts through a head-end skirt located at the case circumference. The igniter is a pyrogen type and incorporates redundant 1-watt/1-ampere squibs and ALCLO ignition material. The nozzle is comprised of a graphite "C" carbon throat insert seated in a tape-wrapped composite exit cone structure of modified phenolic impregnated glass and carbon cloth. The propellant is a JPL formulation.

Development Status. All materials utilized in the Syncom 2 design have been carried over to the Advanced Syncom apogee engine design; this includes the case, insulation/liner, nozzle/insert, and igniter materials. The capabilities of new materials have been evaluated through subscale engine tests using Syncom 2 apogee engine hardware. As a result of these tests, based on observed char depths, a modified phenolic was selected as the optimum impregnating material for the composite nozzle structure.

A series of igniter tests were conducted in a special heavy-wall ignition test engine to establish the relative merits of pyrogen and pyrotechnic igniters and determine the optimum charge of igniter material. These tests clearly demonstrated the superiority of the pyrogen-type igniter for the Advanced Syncom application. All flight and heavy-wall component design effort has been completed, including an optimized flight nozzle contour. Subcontractors for the fabrication of flight hardware, including cases and nozzle body blanks, have been selected. Two cases and one nozzle have been delivered. To maintain better control of the physical and ballistic properties of the propellant, a 150-gallon mixer was purchased and is currently being installed at Ed-

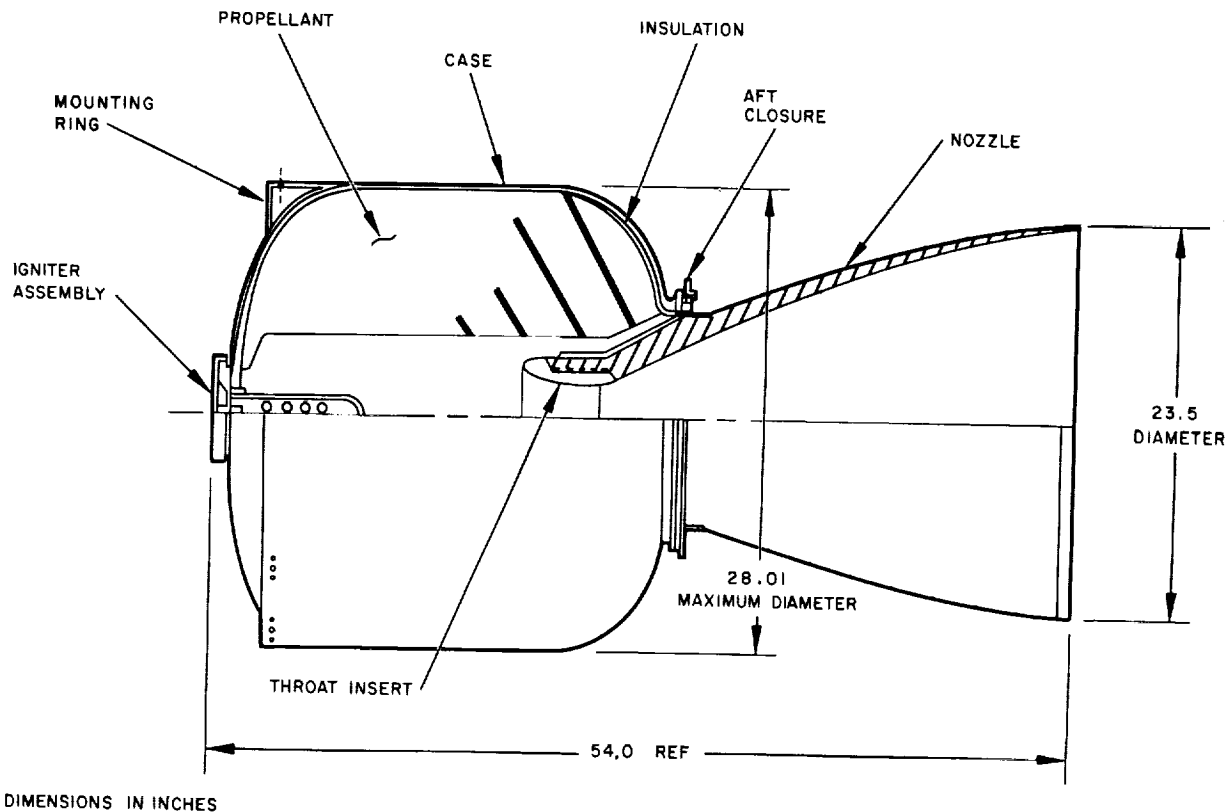


FIGURE 5-235. ADVANCED SYNCOM APOGEE ENGINE

wards Air Force Base. The use of this mixer will permit loading an engine from one batch of propellant. Installation and checkout of the mixer will be completed during 1963.

To date, three heavy-wall engine tests (Figure 5-236) have been conducted under ambient conditions at EAFB. All were cast with multiple propellant batches and contained approximately 655 pounds of propellant — somewhat less than the flight engine design loading. The exterior dimensions of the heavy-wall engine approximate those of the flight engine; the extra wall and insulation thicknesses therefore resulted in a lower available loading volume. All tests were completely successful from a performance standpoint. In the initial test, however, the throat

insert cracked circumferentially just aft of the throat. The nozzles in the other two tests, involving considerably different insert designs, performed satisfactorily.

The ballistic data from the three tests were very nearly identical. Maximum chamber pressure was 240 ± 2 psia and occurred at 29.0 seconds of operation; action time was 40.0 ± 0.5 seconds; peak ignition pressure was 230 ± 2 psia. Thrust data were not reduced because nozzle flow separation rendered it invalid. Nozzle throat erosion did not exceed 1.05 percent in any of the three tests. One additional test is planned in the heavy-wall series, to be conducted in November 1963; the initial flight-weight engine test will be conducted in December 1963. A typical pressure-time trace is shown in Figure 5-237.

5. Spacecraft System Design

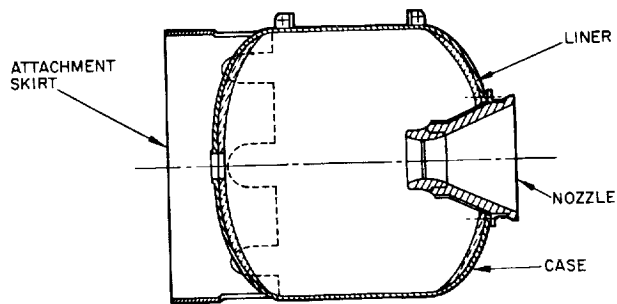


FIGURE 5-236. ADVANCED SYNCOM APOGEE ENGINE
HEAVY WALL MODEL

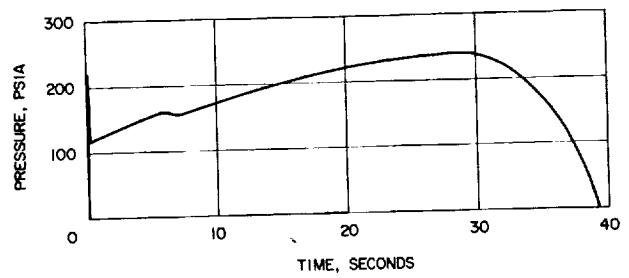


FIGURE 5-237. CHAMBER PRESSURE VERSUS
TIME FOR HEAVY WALL MODEL



6. RELIABILITY AND QUALITY ASSURANCE

FAILURE MODE ANALYSIS

A failure mode analysis has been completed for the Advanced Syncom spacecraft on a system and subsystem level. All inputs and outputs to minor control items were considered in the analysis. The detailed analysis indicated there was only one area in which a specific failure would result in loss of communications. This area included the RF input to the transponder receivers, where a receiving antenna failure could occur, and the RF connection between the transmitter multiplexer and the power splitter.

In the case of the transponder, a failure of any one of the series units would cause a failure of that transponder mode. Although detailed failure mode analysis forms are not included, a detailed analysis was performed which assured that the transponders were "straightforward" and no hidden or subtle effects were found.

Phased Array Control Unit. The phased array control unit is designed with four redundant systems, one in each quadrant. Some interconnections make it possible for some parts of different quadrants to be operated together as a system. This interconnection scheme tends to increase the versatility of the spacecraft, but it also adds new potential failure modes (possible shorts of interconnecting wires). The subject failure mode analysis revealed only one area where all four quadrants were tied together such that a short to ground would affect all of them. This area is the "Timer Select, PACE Lock, Beam Position" input to the telemetry encoder. A short to ground here would prevent telemetering this information from any quad-

rant. This failure mode could be eliminated by running separate interquadrant wires. All other failures (except shorting the unregulated bus to ground) are of an intraquadrant nature, even though they may be caused by shorting interquadrant wires. A typical example is the 2.81-minute output of the central timer. A short to ground of this wire (within a quadrant or between quadrants) applies ground directly to the command 24-volt line through the transistor in the output buffer. This will then cause the command regulator to turn off or burn out. The transistor in the output buffer may also be damaged. The addition of an output series resistor to the output buffer could prevent the overloading of the command regulator in the event of a short. This same condition exists with the output buffer for the 194-cps output from the central timer to the telemetry encoder.

In general it was found that a short or an open would affect only a small portion of the system, and the four times redundant design could tolerate several failures and still function. Complete failure of the phased array control electronics (PACE) is not catastrophic. After such a failure, the antenna would still have a gain of 8 db and illuminate the earth with a pancake pattern so that the system would be able to operate at reduced capability.

As part of the phased array control unit, the jet control electronics (JCE) circuits are four times redundant with a backup command for each function. Thus in addition to on-board control of jet firing, direct ground command jet firing is possible. However, these backup commands are connected from the command decoder

to an OR gate in the output of the JCE. A short or open at or beyond the output of this OR gate would not only result in loss of the quadrant JCE capability but would also prevent backup operation in this quadrant. Separate wiring and isolation for backup would correct this situation.

An open or short to ground at the input or output to the squib driver circuits will not cause squib firing. If the output were shorted to the unregulated bus, the squib would blow. These wires should be routed to prevent this type of short.

Transponder. The multiple access transponder is essentially a series string of active and passive networks. A common master oscillator, with output properly multiplied, provides the receiver local oscillator signal and also the carrier for the retransmission. A common -4-volt supply, produced in the master oscillator amplifier, and a common -24-volt regulated voltage out of the series regulator provides all bias.

A failure mode analysis was made of the entire transponder, with emphasis on catastrophic failures and loss of bias or signal caused by opens or shorts. Most failures can be summed as follows:

- 1) An open or a short anywhere on the -4-volt bus will cause complete failure of a transponder.
- 2) An open or a short anywhere on the -24-volt bus will cause complete failure of the transponder. A short will usually cause the series regulator overload circuits to turn off the regulator and remove most of the load from the spacecraft unregulated bus.
- 3) An open or a short in the received signal anywhere from the output of the receiver

multiplexer to the input of the phase modulator, P/N 475131, will cause a loss of transponded intelligence, but the carrier would still be transmitted. The same result would occur if a short or open occurred in the local oscillator signal between the output of the master oscillator amplifier, P/N 475123, and the input to the mixer, P/N 475100.

- 4) An open or short in the carrier signal, from the output of the master oscillator, P/N 475122, through the modulator, P/N 475131, and into the 3-db hybrid, P/N 475171, of the transmitter, will cause a complete failure of the transponder. The carrier will be lost and cannot be utilized as a beacon.

The frequency translation transponder is similar, and failures may be summed as follows:

- 1) An open or a short anywhere on the -4-volt bus will cause complete failure of a transponder.
- 2) An open or a short anywhere on the -24-volt bus will cause complete failure of the transponder. A short will usually cause the series regulator overload circuits to turn off the regulator and remove most of the load from the spacecraft unregulated bus.
- 3) Any open or short between the receiver multiplexer and the input to the limiter amplifier, P/N 475104, would cause a loss in intelligence transponded, but the beacon and carrier would still be present.
- 4) Any open or short between the master oscillator, P/N 475113, output and the dual filter hybrid would mean a complete loss of the transponder, including beacon.

6. Reliability and Quality Assurance

- 5) Any open or short between the dual filter hybrid and the input mixer, P/N 475100, would provide a loss of transponded intelligence but again the beacon would still be operative.
- 6) Any open or short from the dual filter hybrid to the 3-db hybrid, P/N 475171, in the transmitter would cause a complete transponder failure.

Assuming equal value for each of the four multiple access transponders and four frequency translation transponders, a loss of only one would mean a reduction of $1/8$ in mission capability. This means of redundancy (i.e., multiple transponders) instead of redundant circuitry per transponder appears to be the most judicial choice.

The only telemetry directly attached to the transponders is the received signal strength of the frequency translation transponders. This would indicate a loss only if the transponder failed completely, or if the receiver portion, including the local oscillator string, failed. However, secondary telemetry clues to failure are transmitter power, channel 4, 5, 6, and 7 of frame 5, which would indicate a loss of power if the carrier or beacon were lost, and bus current, channel 1 and 4, frame 1, which could indicate a change of current if the transponder was or was not turning on.

In listing failures, the fact that identical failures may occur in any of the four quadrants has been considered but only one is listed. If multiple failures were to occur, the effect on the mission would be increased proportionately. There are only two areas in which a single failure would cause a complete loss of the communication function, and one area where two failures would cause complete loss.

A short or an open to the signal path from the receiver antenna array with its coupling to the multiplexer or to the connection between the transmitter multiplexer and the power splitter will cause a complete failure. However, the physical construction in these areas is such that any failure would be caused by severe physical damage from booster separation, etc., and is not considered likely.

Two failures, one to each unregulated bus, would cause complete failure. A permanent short to ground of either bus will make one half of the spacecraft inoperative. However, dividing the bus further would remove allowance for solar cell degradation. With the present two buses, each feeding two quadrants, any degradation of cell output will not prevent operation of one quadrant by each. With one bus per quadrant, the same degradation, which is assumed to be uniform over the solar cell areas, would make all quadrants inoperative.

Telemetry and Command. No single open in the telemetry and command subsystem wiring will be catastrophic. In fact, the worst case of a single open wire is the loss of a command regulator which in turn means the loss of one command receiver, one timer, one PACE, and the commutated data from one telemetry encoder. No single short to ground will be catastrophic, the worst case being a short in the unregulated bus which will cause half the spacecraft to be inoperative. The only catastrophic failure found was the apogee motor fire signal from a command decoder output shorted to 24 volts, which would fire the apogee motor.

In general, if an output of a command decoder is opened, the other three decoders can still be used. If an output to a regulator is shorted to ground or the voltage is lowered, the function will be executed, and from then on the other

three decoders must be used for that particular function. A short to ground or lower voltage of a fire signal, i.e. fire axial jet, will not be sufficient to cause the function to occur. A short to a higher or positive voltage would appear as a continuous command.

An open or short on an input to the telemetry encoder will only cause the loss of that input, except in the case of the pressure signals and beam position. If a pressure excitation signal or beam position signal is shorted to ground, there would be no pressure, temperature, or beam position readings from any telemetry.

Detailed Analysis. Table 6-1 is an example of the detailed analysis of failure modes and effects in all mission phases. The likelihood of a particular failure is not indicated. The majority of failures considered were opens and shorts of inputs to a unit. In most cases this is a solder connection. The failure rate of a solder connection is 0.004 by 10^{-6} failure per hour. This rate was

obtained from the Reliability Application and Analysis Guide, Martin Company MI-60-50 (first revision).

The numbers assigned in the "degree" column were based on the following assumptions concerning weight of a four-quadrant spacecraft:

| | Mission | Quadrant | Item |
|---------------------------------|---------|-------------|-------------|
| Phase array control electronics | 0.5 | 0.12 | 0.12 |
| Communications receiver | 1.0 | 0.25 | 0.12 |
| Communications transmitter | 1.0 | 0.25 | 0.12 |
| Telemetry regulator | 1.0 | 0.25 | 0.25 |
| Telemetry encoder | 0.92 | 0.23 | 0.23 |
| Telemetry transmitter | 1.0 | 0.25 | 0.25 |
| Command regulator | 1.0 | 0.25 | 0.25 |
| Command receiver | 1.0 | 0.25 | 0.25 |
| Command decoder | 1.0 | 0.25 | 0.25 |
| Jet control electronics | 1 { 0.5 | 0.25 { 0.12 | 0.25 { 0.12 |
| Backup control | 1 { 0.5 | 0.25 { 0.12 | 0.25 { 0.12 |
| Reaction control system | 1.0 | — | 0.5 |

TABLE 6-1. PACE BOARD

Function: Supplies controlled power to RF phase shifters in phased array antenna system

| Part, Component, Unit, or Subsystem | | Failure Mode Description | Effects Description | | | Telemetered Indication |
|---|--|--------------------------|------------------------------------|--|------------------|---|
| Identification | Description | | On Item | On Mission | Degree, 0 to 1.0 | |
| To auxiliary board, quadrants 1 through 4 | PACE lock, beam position, timer select (typical 3 outputs) | Short | Output grounded. | No telemetering of this information. | 0.04 | F5/C1, C2, C3—no telemetry for timer select, PACE lock, or beam position. |
| | | Open | No telemetry from this quadrant. | Other quadrants ok. | 0.01 | |
| From command board quadrant 1 | Timer select, quadrant 1 (typical 4-quadrant inputs) | Short | This quadrant cannot select timer. | Remaining quadrants can select timer in this quadrant. | 0.12 | F5/C1, C2, C3—timer select command will not have been executed. |
| | | Open | This quadrant cannot select timer. | Remaining quadrants can select timer in this quadrant. | 0.12 | |

TABLE 6-1 (continued)

| Part, Component, Unit, or Subsystem | | Failure Mode Description | Effects Description | | | Telemetered Indication |
|-------------------------------------|---|--------------------------|---|--|------------------|---|
| Identification | Description | | On Item | On Mission | Degree, 0 to 1.0 | |
| To power amplifier | Summing network, output 1 (typical 1 to 16 outputs) | Short | Input to power amplifier grounded. | One-half wave output to phase shifter even if other quadrants used. | 0.06 | No telemetered indication; however, beam would dip twice each revolution. |
| | | Open | No input to power amplifier from this quadrant. | Other quadrants can feed power amplifiers. | 0.03 | |
| From ψ sensor, quadrant 1 | ψ | Short | No ψ input. | This quadrant PACE, JCE, ψ_2 angle counter inoperative. | 0.12 | F5/C1, C2, C3, F3/C3, C4, C5—lose PACE lock. ψ_2 angle erratic. |
| | | Open | No ψ input. | This quadrant PACE, JCE, ψ_2 angle counter inoperative. | 0.12 | |
| To auxiliary board, quadrant 1 | ψ_2 counter clock | Short | This quadrant ψ_2 angle counter inoperative. | Other quadrants ok. | 0.12 | F3/C3, C4, C5— ψ_2 angle register not counting. |
| | | Open | This quadrant ψ_2 angle counter inoperative. | Other quadrants ok. | 0.12 | |
| From auxiliary board, quadrant 1 | Time-of-day correction (typical 4-quadrant inputs) | Short | No time-of-day correction from this quadrant. | Telemetry and command, PACE, timer, command regulator +24-volt output shorted to ground. | 0.25 | No telemetry or command for this quadrant. |
| | | Open | No time-of-day correction from this input. | Other quadrant time-of-day inputs may be used. | 0.12 | F5/C1, C2, C3—beam would not correct for time of day |
| From auxiliary board, quadrant 1 | PACE, +24 volts | Short | Quadrant 1 PACE inoperative. | Other quadrants may be used. | 0.12 | F5/C1, C2, C3—no PACE lock, or beam position. |
| | | Open | Quadrant 1 PACE inoperative. | Other quadrants may be used. | 0.12 | F5/C1, C2, C3—no PACE lock, or beam position. |
| From auxiliary board, quadrant 1 | PACE, -24 volts | Short | Quadrant 1 PACE inoperative. | Other quadrants may be used. | 0.12 | F5/C1, C2, C3—no PACE lock, or beam position. |
| | | Open | Quadrant 1 PACE inoperative. | Other quadrants may be used. | 0.12 | F5/C1, C2, C3—no PACE lock, or beam position. |

TABLE 6-1 (continued)

| Part, Component, Unit, or Subsystem | | Failure Mode Description | Effects Description | | | Telemetered Indication |
|-------------------------------------|--|--------------------------|--|---------------------|------------------|--|
| Identification | Description | | On Item | On Mission | Degree, 0 to 1.0 | |
| To auxiliary board, quadrant 1 | B116 typical also for B117, B118, B119 | Short | Quadrant 1 PACE and JCE inoperative. | Other quadrants ok. | 0.12 | No telemetered indication; phased array beam would dip and drift erratically. |
| | | Open | Quadrant 1 JCE can fire erratically. | Other quadrants ok. | 0.12 | Pulse envelope will indicate erratic or erroneous firing. |
| To auxiliary board, quadrant 1 | PACE lock | Short | Quadrant 1 JCE inoperative. | Other quadrants ok. | 0.12 | F5/C1, C2, C3—PACE lock lost. |
| | | Open | Quadrant 1 JCE can fire erratically. | Other quadrants ok. | 0.12 | Pulse envelope will indicate erratic or erroneous firing. |
| From command board, quadrant 1 | Antenna reference position (typical 4-quadrant inputs) | Short | This quadrant input cannot feed beam positioner. | Other quadrants ok. | 0.12 | F5/C1, C2, C3—No change in beam position when commanded. |
| | | Open | This quadrant input cannot feed positioner. | Other quadrants ok. | 0.12 | F5/C1, C2, C3—No change in beam position when commanded. |
| From command board, quadrant 1 | Eclipse mode ON (typical 4-quadrant inputs) | Short | This quadrant cannot turn eclipse mode on. | Other quadrants ok. | 0.12 | No telemetered indication of eclipse mode ON or OFF. If command fails, system will lose lock at later time (going in or out of eclipse). |
| | | Open | This quadrant cannot turn eclipse mode on. | Other quadrants ok. | 0.12 | |
| | Eclipse mode OFF (typical 4-quadrant inputs) | Short | This quadrant cannot turn eclipse mode off. | Other quadrants ok. | 0.12 | |
| | | Open | This quadrant cannot turn eclipse mode off. | Other quadrants ok. | 0.12 | |

CRITICAL COMPONENT TEST PLANS

The critical component test plans specified for the Advanced Syncom program are presented on the following pages. Each plan details

the number of units required for critical testing and the routing of each unit. These tests are designed to assure a maximum confidence level for each critical item while holding to a minimum the number of units involved.

CRITICAL TEST PLAN FOR SUN SENSORS
ADVANCED SYNCOM

REVISED:

| | | | |
|---|--|-------------------------|-----|
| PREPARED
R. J. Parfitt | HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | NUMBER
475302-601 | |
| APPROVED 10-11-63
<i>[Signature]</i> | | CODE IDENT
NO. 82577 | REV |
| APPROVED | | PAGE 1 OF 6 | |
| APPROVED | | DATE: 10/10/63 | |

SPECIFICATION SHEET

INDEX

| | <u>Page</u> |
|-----------------------------------|-------------|
| 1.0 SCOPE | |
| 1.1 Purpose | 3 |
| 1.2 Description | 3 |
| 2.0 APPLICABLE DOCUMENTS | 3 |
| 3.0 REQUIREMENTS | |
| 3.1 General | 3 |
| 3.2 Acceptance Tests | 4 |
| 3.3 Qualification Test Cycling | 4 |
| 3.4 Accelerated Life Test Program | 5 |
| 3.5 Life Test | 5 |
| 4.0 QUALITY ASSURANCE PROVISIONS | 6 |

CRITICAL TEST PLAN FOR SUN SENSORS
ADVANCED SYNCOM

HUGHES AIRCRAFT CO.
CULVER CITY, CALIF.

SPECIFICATION SHEET

NUMBER

475302-601

CODE IDENT
NO. 82577

REV

PAGE 2 OF 6

DATE 10/10/63

1.0 SCOPE

1.1 Purpose - The purpose of this test plan is to establish requirements and procedures for evaluating and demonstrating the operation of the sun sensors under simulated launch orbit mission profiles of Advanced Syncom.

1.2 Description - Each sun sensor is a device consisting of two photovoltaic cells mounted in a mechanical housing that supports them and masks them except for a finely defined field of view. Mounted on the periphery of the satellite, they produce ψ pulses each time their fields of view pass the sun. These pulses are essential for spin rate control, orientation control, and antenna phase control.

2.0 APPLICABLE DOCUMENTS

| | | |
|-----|------------|--|
| 2.1 | 475302-100 | Hughes Drawing, Sun Sensor Assembly |
| 2.2 | 475302-600 | Test Plan, Solar Sensors |
| 2.3 | 475000-601 | Environmental Requirements for Qualification Tests of Major Control Items and Selected Minor Control Items |
| 2.4 | 475000-602 | Environmental Requirements for Flight Spare Acceptance Tests of Major Control Items and Selected Minor Control Items |
| 2.5 | 475000-607 | Visual-Mechanical Inspection Requirements for Control Units |

3.0 REQUIREMENTS

3.1 General - A minimum of 38 flight solar sensors shall be procured. Of these, 8 shall be used for the test program, and shall be characteristic and identical in configuration to flight production units and shall be inspected, tested, and handled in the same manner as flight production units.

During temperature-vacuum-spin testing each unit shall be energized with a light source of known intensity in an attempt to simulate sun input, and the electrical outputs shall be loaded to match the spacecraft load characteristics.

Signal outputs shall be monitored continually during tests for any intermittencies or other indications of malfunctions.

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|---|-------------------------|-----|
| CRITICAL TEST PLAN FOR SUN SENSORS
ADVANCED SYNCOM | NUMBER
475302-601 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 3 OF 6 | |
| | DATE 10/10/63 | |

SPECIFICATION SHEET

3.2 Acceptance Tests

3.2.1 A representative sample of eight solar sensor assemblies will be selected for the test program. Unit tests in accordance with test plan 475302-600 will initially be performed on the eight assemblies. Each shall be subjected to one cycle of each of the following acceptance level environments:

- 1) Shock and vibration
- 2) Thermal-vacuum and spin
- 3) Boost acceleration
- 4) Acoustical Noise

3.2.2 Parameters of the test cycle environments shall be as specified in the above listed document 475000-602.

3.2.3 A complete functional test shall be performed on each unit following each of the environmental test exposures in accordance with Test Plan 475302-600.

3.2.4 The eight sun sensor assemblies shall be divided into four groups of two each, and shall be tested in the following order relative to environments 1, 2, 3, 4 as listed above.

| | |
|-----------|------------|
| Group I | 1, 2, 3, 4 |
| Group II | 2, 3, 4, 1 |
| Group III | 3, 4, 1, 2 |
| Group V | 4, 1, 2, 3 |

3.2.5 Upon completion of acceptance testing of all eight units, one unit shall be selected from each group to be placed on accelerated life testing, with the second unit from each group to undergo qualification test cycling.

3.3 Qualification Test Cycling

3.3.1 Four sun sensor assemblies shall be subjected to one cycle each of the following qualification level environments.

- 1) Shock and vibration
- 2) Temperature-vacuum-spin
- 3) Boost and spin acceleration
- 4) Acoustical Noise

3.3.2 Parameters of the test cycle environment shall be as specified by 475000-601.

3.3.3 A complete functional test shall be performed following each of the environmental test exposures in accordance with Test Plan 475302-600.

| | | |
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| CRITICAL TEST PLAN FOR SUN SENSORS
ADVANCED SYNCOM

HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | NUMBER
475302-601 | |
| | CODE IDENT
NO. 82577 | REV |
| | PAGE 4 | OF 6 |
| | DATE 10/10/63 | |

SPECIFICATION SHEET

3.3.4 The four sun sensor assemblies shall be tested in the following order relative to environments 1, 2, 3, and 4.

- | | | |
|----|-------------|---------------|
| 1) | First unit | 1, 2, 3, 4, 1 |
| 2) | Second unit | 2, 3, 4, 1, 2 |
| 3) | Third unit | 3, 4, 1, 2, 3 |
| 4) | Fourth unit | 4, 1, 2, 3, 4 |

3.4 Accelerated Life Test Program

3.4.1 Four sun sensor assemblies shall be subjected to concurrent temperature-vacuum-spin tests under the following conditions:

- a) Under a vacuum of at least 10^{-5} Torr, and under calibrated simulated sunlight, and spinning, the temperature shall be cycled between the extreme high and low temperature values as listed in Environmental Requirements for Flight Spare Acceptance Tests of Major Control Items and Selected Minor Control Items, 475000-602. The temperature shall be cycled from high to low and back to high in 1 hour, 40 minutes. The cycle shall be completed 200 times.
- b) The spin rate shall be in accordance with 475000-602, Environmental Requirements for Flight Spare Acceptance Test of Major Control Items and Selected Minor Control Items.

3.4.2 Each sun sensor shall successfully pass the requirements of Test Plan 475302-600 upon completion of the test.

3.5 Life Test

3.5.1 Following successful completion of the acceptance test program, the same four units shall be operated under laboratory ambient conditions for a period of at least 1 year.

3.5.2 Each sensor shall be firmly mounted and its electrical output shall be loaded to match the load offered by the spacecraft: A simulated solar light source shall be mounted so as to excite the cells. The source shall be cycled on and off at a rate of 100 cycles (simulating the spin rate of the spacecraft).

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| <p>CRITICAL TEST PLAN FOR SUN SENSORS
ADVANCED SYNCOM</p> | <p>NUMBER
475302-601</p> | |
| | <p>CODE IDENT
NO. 82577</p> | <p>REV</p> |
| <p>HUGHES AIRCRAFT CO.
CULVER CITY, CALIF.</p> | <p>PAGE 5 OF 6</p> | |
| | <p>DATE 10/10/63</p> | |

SPECIFICATION SHEET

3.5.3 The four simulated light sources should be capable of continuous operation for one year. If a light fails, the time count will terminate, and resume when the light is repaired.

3.5.4 Each sun sensor output will be monitored at the beginning of the test, and then monitored every 24 hours for two weeks, with note made of any change. If no change is noted, monitoring will be made once every week for the rest of the year. If, at any time, a change is noted, the output will be switched to a recorder, and will be continuously monitored as long as it continues to change. If no further change is noted within one week, continuous monitoring may be ended, and the cell output may be checked once every 24 hours. If, after two weeks, no further change is noted, checks may be decreased to once a week.

4.0 QUALITY ASSURANCE PROVISIONS

Same as stated in test plan 475302-600

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|---|-------------------------|-----|
| CRITICAL TEST PLAN FOR SUN SENSORS
ADVANCED SYNCOM | NUMBER
475302-601 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 6 OF 6 | |
| | DATE 10/10/63 | |

SPECIFICATION SHEET

CRITICAL TEST PLAN FOR PHASE SHIFTERS

REVISED:

| | | |
|----------------------------------|--|------------------------|
| PREPARED
<i>R. J. Parfitt</i> | | NUMBER
475152-601 |
| APPROVED <i>11-23</i> | | CODE IDENT NO. 82577 I |
| APPROVED | HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 1 OF 4 |
| APPROVED | | DATE: 10-10-63 |

SPECIFICATION SHEET

INDEX

| | <u>Page</u> |
|----------------------------------|-------------|
| 1.0 SCOPE | |
| 1.1 Purpose | 3 |
| 1.2 Description | 3 |
| 2.0 APPLICABLE DOCUMENTS | 3 |
| 3.0 REQUIREMENTS | |
| 3.1 General | 3 |
| 3.2 Ambient Pressure Test | 3 |
| 3.3 Thermal Vacuum Tests | 4 |
| 3.4 Qualification Vibration | 4 |
| 4.0 QUALITY ASSURANCE PROVISIONS | 5 |

| | | |
|--|-------------------------|-----|
| CRITICAL TEST PLAN FOR PHASE SHIFTERS | NUMBER
475152-601 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 2 OF 4 | |
| | DATE 10-10-63 | |

SPECIFICATION SHEET

1.0 SCOPE

1.1 Purpose: The purpose of this test plan is to establish requirements and procedures for evaluation and demonstrating the operation of the Phase Shifters under simulated launch and orbit mission profiles of Advanced Syncom.

1.2 Description: The phase shifter is a device consisting of an RF input coupler, a tube of ferrite in a circular waveguide inside a four-pole, two phase electromagnetic field coil, and an output coupler. An RF signal applied to the input port will leave the output port shifted in phase by an amount determined by the field coil current. The field coils are driven by signals from the phased-array control electronics (PACE) which provide the correct phasing angles so that the output of all the antenna elements are properly phased to form a beam directed toward the earth.

2.0 APPLICABLE DOCUMENTS

| | | |
|-----|------------|--|
| 2.1 | 475152-100 | Hughes Drawing, Phase Shifter |
| 2.2 | 475152-600 | Test Plan, RF Phase Shifter |
| 2.3 | 475000-601 | Environmental Requirements for Qualification Tests of Major Control Items and Selected Minor Control Items |
| 2.4 | 475000-602 | Environmental Requirements for Flight Spare Acceptance Tests of Major Control Items and Selected Minor Control Items |
| 2.5 | 475000-607 | Visual-Mechanical Inspection Requirements for Control Items |
| 2.6 | 475030-600 | Test Plan, Antenna, Communications |

3.0 REQUIREMENTS

3.1 General: A total of 66 flight phase shifters will be produced. Of these, 6 shall be selected for critical component testing. They shall be characteristic and identical in configuration to flight production units, and shall be inspected, tested and handled in the same manner as flight production units.

Each phase shifter shall be numbered, one through six, and shall retain this identity throughout the critical testing.

3.2 Ambient Pressure Test:

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|--|-------------------------|-----|
| CRITICAL TEST PLAN FOR PHASE SHIFTERS | NUMBER
475152-601 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 3 OF 4 | |
| | DATE 10-10-63 | |

SPECIFICATION SHEET

3.2.1 After the six phase shifters have met the performance requirements of Test Plan 475152-600, they shall be subjected to temperature tests at laboratory ambient pressure. Each phase shifter shall be mounted in a special adapter, as used in Test Plan 475152-600, and the phase shift between the output and the input shall be measured as a function of current through the motor stator winding.

RF losses and power outputs of the two outputs on each phase shifter shall be measured and compared.

3.2.2. All readings shall be made at a stabilized 0° F and again at stabilized 150° F.

3.3 Thermal Vacuum Tests:

3.3.1 Phase shifters numbers 1, 2, 3 and 4 shall then be tested in a thermal vacuum at a pressure of 10^{-5} Torr or less. The phase shifters shall be mounted in a special adapter, as used in Test Plan 475152-600, and shall be stabilized at a temperature of 40° F for a period of 48 hours. The same performance tests as in 3.2.1 above shall be repeated once every 24 hours.

3.3.2 The temperature shall then be increased to 130° F and stabilized, and shall remain in this condition for 48 hours. The performance tests of 3.2.1 above shall be repeated once every 24 hours.

3.4 Qualification Vibration:

3.4.1 After completion of 3.2 and 3.3 above, phase shifters numbers 3, 4, 5 and 6 shall be subjected to qualification vibration environment as defined in 475000-602. Performance tests shall be conducted in accordance with Test Plan 475152-600 before and after the vibration tests.

4.0 QUALITY ASSURANCE PROVISIONS

4.1 Same as stated in Test Plan 475152-600, RF Phase Shifter, and Test Plan 475030-600, Antenna, Communications.

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|--|-------------------------|-----|
| CRITICAL TEST PLAN FOR PHASE SHIFTERS | NUMBER
475152-601 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 4 OF 4 | |
| | DATE 10-10-63 | |
| SPECIFICATION SHEET | | |

CRITICAL TEST PLAN FOR THE
CENTRAL TIMER

REVISED:

| | | | |
|---|--|-------------------------|-----|
| PREPARED
J. W. Babb | HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | NUMBER
475303-601 | |
| APPROVED 10-14-63
<i>J. C. Meyer</i> | | CODE IDENT
NO. 82577 | REV |
| APPROVED | | PAGE 1 OF 4 | |
| APPROVED | | DATE: 10-10-63 | |

SPECIFICATION SHEET

INDEX

| | | |
|-----|---|-------------|
| 1.0 | SCOPE | <u>Page</u> |
| 1.1 | Purpose: | 3 |
| 1.2 | Description: | 3 |
| 2.0 | APPLICABLE DOCUMENTS | 3 |
| 3.0 | REQUIREMENTS | |
| 3.1 | General: | 3 |
| 3.2 | Test Samples: | 3 |
| 3.3 | Temperature Cycling (Non-Operating): | 4 |
| 3.4 | Vibration and Shock: | 4 |
| 3.5 | Acceleration: | 4 |
| 3.6 | Noise: | 4 |
| 3.7 | Start Tests: | 4 |
| 3.8 | Thermal-Vacuum, Temp. Cycling and Spin Tests: | 4 |
| 3.9 | Operating Life Test: | 4 |
| 4.0 | QUALITY ASSURANCE PROVISIONS | 4 |

| | | |
|--|-------------------------|-----|
| CRITICAL TEST PLAN FOR THE CENTRAL TIMER | NUMBER
475303-601 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 2 OF 4 | |
| | DATE 10-10-63 | |

SPECIFICATION SHEET

1.0 SCOPE

1.1 Purpose: It is the purpose of this test plan to evaluate and demonstrate the operation of the central timer, P/N 475303, under simulated Advanced Syncom mission profiles (launch and orbit operation).

1.2 Description: The Advanced Syncom central timer (four per spacecraft) has four separate functions to perform in the spacecraft. First, it provides selectable time-of-day correction signals to the Phased Array Control Electronics at 2.81 minute intervals throughout the life of the vehicle. Second, it provides a 194 cps commutating clock frequency to the telemetry encoder throughout the life of the vehicle. Third, the central timer provides an output to clear the command register at 315 minute intervals throughout the life of the spacecraft. Fourth, it generates the Apogee Motor Ignition Signal 315 minutes after separation from the second stage of the launch vehicle.

2.0 APPLICABLE DOCUMENTS

2.1 The following documents form a part of this test plan:

- | | | |
|-------|------------|--|
| 2.1.1 | 475303-600 | Central Timer in Process Test Specification |
| 2.1.2 | 475000-602 | Env. Requirements for Flt. Spare Accept. Tests of Maj. Cont. Items and Sel. Minor Cont. Items. |
| 2.1.3 | 475000-601 | Env. Reqmts. for Qualification Tests of Major Control Items and Selected Minor Control Items. |

3.0 REQUIREMENTS

3.1 General:

3.1.1 The central timers selected for this test shall be characteristic and identical to the flight production units and shall be assembled, inspected, tested and handled in the same manner as all flight production units.

3.1.2 Performance Tests in accordance with 475303-600 shall be conducted prior to and after each environmental test.

3.2 Test Samples:

3.2.1 A representative sample of five timers shall be selected for this test. Two timers will be segregated after temperature cycling (non operating) for operating life tests; the remaining three timers shall be subjected to the tests listed in Paragraphs 3.4 through 3.9.

| | | |
|--|-------------------------|-----|
| CRITICAL TEST PLAN FOR THE CENTRAL TIMER | NUMBER
475303-601 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 3 OF 4 | |
| | DATE 10-10-63 | |

SPECIFICATION SHEET

3.3 Temperature Cycling (Non-operating): All five timers will be temperature cycled 100 times between the minimum and maximum temperature limits as specified in 475000-601. The purpose is to cause poorly constructed units to fail prematurely so they may be culled.

3.4 Vibration and Shock: Three central timers shall be subjected to vibration and shock levels called out in 475000-601, Environmental Requirements For Qualification Tests of Major Control Items and Selected Minor Control Items. In addition the three units shall be subjected to vibration and shock stresses at $1\frac{1}{2}$ and 2 times the levels called for in 475000-601.

3.5 Acceleration: Spin and boost three units shall be subjected to acceleration levels as specified in 475000-601. In addition these timers shall be subjected to stresses at $1\frac{1}{2}$ and 2 times the levels called out in 475000-601. The units shall be spun at each level initially oriented such that the tuning fork will be subjected to the same spin forces it will experience in the spacecraft.

3.6 Noise:

3.6.1 Acoustical Noise: The three timers shall be subjected to acoustical noise levels specified in 475000-601.

3.6.2 Electrical Noise: The timers shall be operated with a 3 volt peak-to-peak square wave superimposed on the ± 24 volt supply to ensure that electrical noise on the supply lead will not cause the unit to malfunction.

3.7 Start Tests: The timers shall be subjected to start tests to ensure that when power is removed and then is reapplied very slowly (at a rate not to exceed 10 volts/min.), the oscillators will have sufficient loop gain to restart oscillation at room temperature. This test shall be repeated 5 times at the minimum and maximum temperature specified in 475000-602.

3.8 Thermal-Vacuum, Temperature Cycling and Spin Test: The three timers shall be operated in a thermal-vacuum of 1×10^{-5} Torr or less for 30 days. The heat sink temperature shall be cycled at least 50 times between the minimum and maximum temperature limits as specified in 475000-602. The timers shall be subjected to spin acceleration identical to that to which they are subjected in the spacecraft.

3.9 Operating Life Test: The two timers previously segregated (Paragraph 3.2.1) plus the timers which survived all the previous tests will be put on operating life test. This test shall consist of electrically energizing each timer to simulate actual operating conditions and monitoring the electrical parameters. The test duration shall be a minimum of three months. During the operating life test the heat sink temperature shall be cycled from maximum to minimum (as specified in 475601) daily.

4.0 QUALITY ASSURANCE PROVISIONS

Same as stated in 475303-600, Central Timer In Process Test Specification.

| | | |
|--|-------------------------|-----|
| CRITICAL TEST PLAN FOR THE CENTRAL TIMER | NUMBER
475303-601 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 4 OF 4 | |
| | DATE 10-10-63 | |

SPECIFICATION SHEET

CRITICAL TEST PLAN FOR TRAVELING-WAVE TUBE
ADVANCED SYNCOM

REVISED:

| | | | |
|--|--|-------------------------|-----|
| PREPARED
J. C. Meyer | HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | NUMBER
475125-601 | |
| APPROVED
10-14-63
<i>J. C. Meyer</i> | | CODE IDENT
NO. 82577 | REV |
| APPROVED | HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 1 OF 5 | |
| APPROVED | | DATE: 10/10/63 | |

SPECIFICATION SHEET

INDEX

| | <u>Page</u> |
|----------------------------------|-------------|
| 1.0 SCOPE | 3 |
| 1.1 Purpose | 3 |
| 1.2 Description | 3 |
| 2.0 APPLICABLE DOCUMENTS | |
| 2.1 Test Specifications | 3 |
| 3.0 REQUIREMENTS | 3 |
| 3.1 General | 3 |
| 3.2 Power Aging | 3 |
| 3.3 Cycling | 3 |
| 3.4 Life Tests | 4 |
| 4.0 QUALITY ASSURANCE PROVISIONS | 5 |

| | | |
|--|---------------------------------|------------|
| <p>CRITICAL TEST PLAN FOR TWT
ADVANCED SYNCOM</p> | <p>NUMBER
475125-601</p> | |
| | <p>CODE IDENT
NO. 82577</p> | <p>REV</p> |
| <p>HUGHES AIRCRAFT CO.
CULVER CITY, CALIF.</p> | <p>PAGE 2 OF 5</p> | |
| | <p>DATE 10/10/63</p> | |

SPECIFICATION SHEET

1.0 SCOPE

1.1 Purpose - It is the purpose of this test plan to evaluate and demonstrate the operation of traveling-wave tube (TWT) under simulated Advanced Syncom mission profiles (Launch and Orbit operation).

1.2 Description - The TWT amplifies the RF output of the Advanced Syncom spacecraft to 4 watts over the frequency range of 3977 to 4194 mc.

2.0 APPLICABLE DOCUMENTS

2.1 Test Specifications

| | |
|------------|---|
| 475125-600 | Test Plan, TWT |
| 475000-602 | Environmental Requirements for Flight Spare Acceptance Tests of Major Control Items and Selected Minor Control Items. |
| 475000-601 | Environmental Requirements for Qualification Tests of Major Control Items and Selected Minor Control Items. |
| 475000-607 | Visual-Mechanical Inspection Requirements for Control Items. |

3.0 REQUIREMENTS

3.1 General - The TWTs selected for this test plan shall be characteristic and identical to the flight production units and shall be assembled, inspected, tested and handled in the same manner as all flight production units. There will be a total of 56 flight TWTs. Of these 56, 20 will be selected for flight use and 30 will be scheduled for the following test program.

3.2 Power Aging - After the tubes have met the requirements called out in paragraph 4.1.2 Test Plan 475125-600, they will all be subjected to a 2000 hour power aging test at laboratory ambient conditions. Each of the 30 tubes will then be subjected to the expected launch environment which will consist of vibration, shock, acceleration (boost and spin) and temperature. The tubes will be operational during the environmental exposures and performance in accordance with paragraph 4.1.2 of Test Plan 475125-600, will be recorded prior and subsequent to each environmental exposure.

3.3 Cycling - At the conclusion of the environmental exposure each tube will be cycled 200 times. A cycle is defined as 15 minutes off, 15 minutes on. (The filaments to be turned on 4 minutes prior to applying high voltages). The tubes shall meet the electrical performance requirements of paragraph 4.1.2 of Test Plan 475125-600 at the conclusion of the cycling test.

| | | |
|---|-------------------------|-----|
| CRITICAL TEST PLAN FOR TWT
ADVANCED SYNCOM | NUMBER
475125-601 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 3 OF 5 | |
| | DATE 10/10/63 | |

SPECIFICATION SHEET

3.4 Life Tests - The 30 tubes will be sorted into 9 groups and the following tests performed for a minimum of one year. The grouping and tests to be performed on each group follows:

| <u>Group</u> | <u>No. Tubes</u> | <u>Test</u> |
|--------------|------------------|--|
| 1 | 3 | Cycling 1 hr. on, 1/4 hr. off (r min. heater warm-up). |
| 2 | 3 | Cycling 1 hr. on, 1/4 hr. off (no heater warm-up). |
| 3 | 4 | Continuous operation with base plate temperature of 175° F. |
| 4 | 5 | Continuous operation with Flight type power supply (475174-100 units). This group will be subjected to 200 operational cycles 6 hot and cold temperatures (reference Test Plan 475000-602) under vacuum conditions (10 ⁻⁵ Torr or less). At the completion of the thermal vacuum they will continue test under laboratory ambient conditions. |
| 5 | 4 | Continuous operation at laboratory ambient conditions. |
| 6 | 3 | Same as group 5 except collector voltage will be adjusted to obtain 3 times the normal helix current. |
| 7 | 3 | Stored at laboratory ambient conditions. Each tube to be operated once each 6 month for sufficient time to check the performance in accordance with Paragraph 4.1.2 of Test Plan 475125-600. |
| | 2 | Continuous operation with the RF output shorted in such a manner as to result in maximum reflected power. |
| | 3 | Operate continuously at laboratory ambient conditions with the cathode temperature elevated (operate filament at 7 volts). |

3.4.1 Description of expected results from life tests.

Group 1 tests - The attempt here is to see if on-off cycles affect the life of the tube.

| | | |
|---|-------------------------|-----|
| CRITICAL TEST PLAN FOR TWT
ADVANCED SYNCOM | NUMBER
475125-601 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 4 OF 5 | |
| | DATE 10/10/63 | |

1022 CC REV. 1-62

SPECIFICATION SHEET

Group 2 tests - To determine if there is any difference between the life of tubes in Group 1 and 2. If there is not, it would result in a considerable weight, part and complexity savings of the power supply.

Group 3 tests - Determine the effects of high temperature operation.

Group 4 tests - This test simulates the actual operation of the tube with power supply. 200 cycles was chosen as the maximum number of on-off cycles that any given tube would see in orbit.

Group 5 tests - This test simulates the normal operation and will be the reference test from which the other tests can be compared.

Group 6 tests - This test will determine the effects of high helix current on cathode wearout.

Group 7 tests - This test will determine the effects of non-operation (simulating the redundant tube in the spacecraft).

Group 8 tests - This test is to determine the lifetime of the attenuator.

Group 9 tests - This test is to determine the effects of elevated heater temperature on the life of the tube. Hopefully, it will result in data to verify accelerated testing.

3.4.2 In addition to the above tests, 10 tubes fabricated during the interim program have been selected and placed on life test at laboratory ambient conditions.

3.4.3 Frequency of Tube Performance Recording - Cathode activity measurements in accordance with paragraph 4.1.24 of Test Plan 475125-600 will be made at 1000 hours, 3000 hours, 5000 hours and 7000 hours and then in steps of 3000 hours. Complete electrical performance measurements in accordance with paragraph 4.12 Test Plan 475125-600 will be made at 2000 hours, 5000 hours, 7000 hours and then in steps of 3000 hours.

4.0 QUALITY ASSURANCE PROVISIONS

Same as stated in Test Plan 475125-607.

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|--|-------------------------|-----|
| CRITICAL TEST PLAN FOR TWT
ADVANCED SYNCOM | NUMBER
475125-601 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 5 OF 5 | |
| | DATE 10/10/63 | |

SPECIFICATION SHEET

PROPOSED CRITICAL TEST PLAN FOR SQUIB DRIVERS

The squib driver circuits are constructed of standard proven solid state devices using techniques well within the "state-of-the-art" and therefore should not be considered as critical components. However, when the pyrotechnic switches (critical component) were designed out of the Advanced Syncom System, it was suggested that the squib drivers be substituted as a critical component. If, after careful consideration, it is decided that the squib driver is a critical item, the following is an outline of the test program.

REVISED:

| | | | |
|---|--|-------------------------|-----|
| PREPARED
J. W. Babb | | NUMBER
475306-601 | |
| APPROVED 10-14-63
<i>[Signature]</i> | | CODE IDENT
NO. 82577 | REV |
| APPROVED | HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 1 OF 4 | |
| APPROVED | | DATE: 10-10-63 | |

1412-CC REV. 11-61

SPECIFICATION SHEET

INDEX

| | <u>Page</u> |
|---|-------------|
| 1.0 SCOPE | |
| 1.1 Purpose: | 3 |
| 1.2 Description: | |
| 2.0 APPLICABLE DOCUMENTS | 3 |
| 3.0 REQUIREMENTS | |
| 3.1 General: | 3 |
| 3.2 Test Samples: | 3 |
| 3.3 Vibration and Shock: | 4 |
| 3.4 Acceleration (Spin and Boost): | 4 |
| 3.5 Noise: | 4 |
| 3.6 Thermal Vacuum and Temperature Cycling: | 4 |
| 3.7 Extended Storage Life Test: | 4 |
| 4.0 QUALITY CONTROL PROVISIONS | 4 |

| | | |
|--|-------------------------|-----|
| CRITICAL TEST PLAN FOR SQUIB DRIVERS | NUMBER
475306-601 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 2 OF 4 | |
| | DATE 10-10-63 | |

SPECIFICATION SHEET

1.0 SCOPE

1.1 Purpose: It is the purpose of this test plan to evaluate and demonstrate the operation of the squib driver under simulated Advanced Syncom Mission Profiles (launch and orbit operation).

1.2 Description: The squib drivers receive a fire signal input from the central timers or from the command decoder and supply power to the various squibs in the Advanced Syncom spacecraft. These squib driver circuits are constructed of standard proven solid state devices using techniques well within the "state of the art".

2.0 APPLICABLE DOCUMENTS

The following documents form a part of this test plan:

| | |
|------------|--|
| 475306-600 | Squib Driver in Process Test Specification |
| 475000-602 | Environmental Requirements For Flight Spares
Acceptance Test of Major Control Items and
Selected Minor Control Items |
| 475000-601 | Environmental Requirements for Qualification
Tests of Major Control Items and Selected
Minor Control Items. |

3.0 REQUIREMENTS

3.1 General:

3.1.1 The squib drivers selected for this test shall be characteristic and identical to the flight production units and shall be assembled, inspected, tested and handled in the same manner as all flight production units.

3.1.2 Performance tests in accordance with 475306-600 shall be conducted prior to and after each environmental test.

3.2 Test Samples: A representative sample of three squib drivers (each containing 4 separate driver circuits) shall be selected for this test. One squib driver will be segregated for extended storage life test. The two remaining units shall be subjected to the tests listed in Paragraphs 3.3 through 3.7. The drivers shall not fire randomly during any of the test environments.

3.3 Vibration and Shock: The two squib drivers shall be subjected to vibration and shock levels as specified in 475000-601.

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|--|-------------------------|-----|
| CRITICAL TEST PLAN FOR SQUIB DRIVERS | NUMBER
475306-601 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 3 OF 4 | |
| | DATE 10-10-63 | |

SPECIFICATION SHEET

3.4 Acceleration (Spin and Boost): The squib drivers shall be subjected to the acceleration environments specified in 475000-601.

3.5 Noise:

3.5.1 Acoustical Noise: These units shall be subjected to the acoustical noise levels as specified in 475000-601.

3.5.2 Electrical Noise: The squib drivers shall be operated with a 3 volt peak-to-peak square wave superimposed on the ± 24 volt supply to ensure that electrical noise on the supply lead will not cause the unit to malfunction.

3.6 Thermal Vacuum and Temperature Cycling: The two squib driver packages shall be operated in thermal-vacuum of 1×10^{-5} Torr or less for 10 days with the heat sink temperature cycled at least 50 times between the minimum and maximum limits called out in 475000-602.

3.7 Extended Storage Life Test: The unit segregated in Paragraph 3.2 shall be stored at room temperature for a period of one year. Each month it is to have power applied for 24 hours. None of the four circuits shall fire. At the end of 24 hours an input fire signal is to be applied. All four circuits must fire.

4.0 QUALITY CONTROL PROVISIONS

Same as stated in 475306-600.

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|--|-------------------------|-----|
| CRITICAL TEST PLAN FOR SQUIB DRIVERS | NUMBER
475306-601 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 4 OF 4 | |
| | DATE 10-10-63 | |
| SPECIFICATION SHEET | | |

PHASE I TESTS OF BIROPELLANT INJECTOR
SOLENOID VALVES
ADVANCED SYNCOM

REVISED:

| | | | |
|--------------------------------------|--|-------------------------|-----|
| PREPARED
P. J. Sengstock | HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | NUMBER
475000-608 | |
| APPROVED 10-11-63
<i>C. Meyer</i> | | CODE IDENT
NO. 82577 | REV |
| APPROVED | | PAGE 1 OF 5 | |
| APPROVED | | DATE: 10/10/63 | |

1412-CC REV. 11-61

SPECIFICATION SHEET

INDEX

| | <u>Page</u> |
|---------------------------|-------------|
| 1.0 SCOPE | 3 |
| 1.1 Purpose | 3 |
| 1.2 Description | 3 |
| 2.0 APPLICABLE DOCUMENTS | 3 |
| 3.0 REQUIREMENTS | |
| 3.1 General | 3 |
| 3.2 Environmental Testing | 3 |
| 3.3 Endurance Testing | 4 |

| | | |
|---|---------------------------------|------------|
| <p>PHASE I TESTS OF BIPROPELLANT INJECTOR
SOLENOID VALVES
ADVANCED SYNCOM</p> | <p>NUMBER
475000-608</p> | |
| | <p>CODE IDENT
NO. 82577</p> | <p>REV</p> |
| <p>HUGHES AIRCRAFT CO.
CULVER CITY, CALIF.</p> | <p>PAGE 2 OF 5</p> | |
| | <p>DATE 10/10/63</p> | |

SPECIFICATION SHEET

1.0 SCOPE

1.1 Purpose - It is the purpose of this test plan to evaluate and demonstrate the ability of the solenoid valves to withstand anticipated environments and meet the required number of pulse cycles.

1.2 Description - The solenoid valve allows propellant to enter the injector head when actuated.

2.0 APPLICABLE DOCUMENTS

- | | | |
|-----|------------------|--|
| 2.1 | 475000-602 | Environmental Requirements for Flight Spare Acceptance Tests of Major Control Items and Selected Minor Control Items |
| 2.2 | 475000-601 | Environmental Requirements for Qualification Tests of Major Control Items and Selected Minor Control Items |
| 2.3 | MTS 0612, Rev. A | Marquardt Company Test Specification |
| 2.4 | 475000-607 | Visual-Mechanical Inspection Requirements for Control Items |

3.0 REQUIREMENTS

3.1 General

3.1.1 Four valves will be purchased from the Marquardt Company (TMC). These valves shall be acceptance tested by TMC in accordance with MTS 0612 prior to delivery. Leakage shall not exceed 10 cc/hr of helium at 300 psi.

3.2 Environmental Testing - Two of the four valves will be subjected to environmental testing. Advance Syncom solenoid drivers (PN 475307-100) will be used to actuate the valves. Response time shall be measured just prior to each leak test. The test will be conducted as follows:

3.2.1 Vibration Tests - The two valves will be vibrated at the levels listed in 475000-602. Vibration will be applied at 45 degrees to the longitudinal axis with nitrogen at 300 psi connected to the input. During vibration the valve will be checked for leakage. After each vibration test, the valves will be pulsed 500 times (75 ms ON, 525 ms OFF) with water as the propellant. The valves will then be tested for leaks in accordance with paragraph 3.1.2 of MTS 0612.

| | | |
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| PHASE I TESTS OF BIPROPELLANT INJECTOR
SOLENOID VALVES
ADVANCED SYNCOM | NUMBER
475000-608 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 3 OF 5 | |
| | DATE 10/10/63 | |

1982 CC REV. 1-68

SPECIFICATION SHEET

3.2.2 Shock Test - The two valves will be subjected to a sixty-g terminal peak sawtooth shock impulse of 15 ms duration. The test set-up shall be the same as paragraph 3.2.1 and the same performance tests shall be conducted at the beginning and conclusion of this test.

3.2.3 Spin Test - The two valves will be subjected to spin acceleration equivalent to 150 rpm at a 30" radius with acceleration applied at an angle of 45 degrees to the longitudinal axis. The test set-up shall be similar to paragraph 3.2.1. The valves will be pulsed 250 times during spin. Upon completion of this test the valves will be tested for leaks in accordance with paragraph 3.1.2 of MTS 0612.

3.2.4 High-Low Temperature Tests - The two valves shall then be subjected to high-low temperature extremes.

3.2.4.1 General - The upstream pressure on the valves shall be 300 to 325 psig. The downstream pressure during pulsing shall be 20 to 25 psi less than the upstream pressure. The downstream pressure during soak and leak testing shall be 0 psig.

A pulse series shall consist of 2,200 pulses at 75 ms ON and 525 ms OFF.

The propellant shall be water.

The valves shall be leak tested in accordance with paragraph 3.1.2 of MTS 0612.

3.2.4.2 After initial hook-up the valve will be operated for one pulse series at ambient temperature and then tested for leaks. The temperature shall then be changed to 100° F.

3.2.4.3 After a one-hour soak the valves will be tested for leaks, operated one pulse series and tested for leaks. Then total leakage over a 24 hour period shall be recorded. The valves will then be operated one pulse series and tested for leaks.

3.2.4.4 The valves will then be cooled to 33° F and paragraph 3.2.4.3 repeated.

3.2.4.5 The valves will then be heated to 200°F and paragraph 3.2.4.3 repeated.

3.3 Endurance Testing - The other two valves will be tested under the following conditions until both valves have been pulsed 10⁶ times.

3.3.1 The test will be run at ambient temperature with 65° F distilled water under 20 to 25 psig pressure.

| | | |
|--|---------------------------------|------------|
| <p>PHASE I TESTS OF BI-PROPELLANT INJECTOR
SOLENOID VALVES
ADVANCED SYNCOM</p> | <p>NUMBER
475000-608</p> | |
| | <p>CODE IDENT
NO. 82577</p> | <p>REV</p> |
| | <p>PAGE 4 OF 5</p> | |
| <p>HUGHES AIRCRAFT CO.
CULVER CITY, CALIF.</p> | <p>DATE 10/10/63</p> | |
| | <p>SPECIFICATION SHEET</p> | |

3.3.2 The valve body temperature will be measured and recorded. The tests will be stopped whenever the valve temperature exceeds 200° F. After an appropriate cooling period, the tests will be continued.

3.3.3 The pulse rate shall be 75 ms ON, 525 ms OFF, except every 1000th pulse shall be 14 minutes ON and 5 minutes OFF.

3.3.4 Leak tests shall be made at the following intervals: 1000, 2000, 4000, 8000, 16,000, 32,000, 64,000, 128,000, 256,000, 512,000, and at completion of test.

| | | |
|---|---------------------------------|------------|
| PHASE I TESTS OF BIPROPELLANT INJECTOR
SOLENOID VALVES
ADVANCED SYNCOM | NUMBER
475000-608 | |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | CODE IDENT
NO. 82577 | REV |
| | PAGE 5 OF 5 | |
| | DATE 10/10/63 | |

1922 CC REV. 1-62

SPECIFICATION SHEET

CRITICAL TEST PLAN FOR BIPROPELLANT
INJECTOR SOLENOID VALVES
ADVANCED SYNCOM

REVISED:

| | | | |
|--------------------------------------|--|-------------------------|-----|
| PREPARED
P. J. Sengstock | HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | NUMBER
475000-609 | |
| APPROVED 10-11-63
<i>J. Meyer</i> | | CODE IDENT
NO. 82577 | REV |
| APPROVED | | PAGE 1 OF 5 | |
| APPROVED | | DATE: 10/10/63 | |

SPECIFICATION SHEET

INDEX

| | <u>Page</u> |
|----------------------------------|-------------|
| 1.0 SCOPE | |
| 1.1 Purpose | 3 |
| 1.2 Description | 3 |
| 2.0 APPLICABLE DOCUMENTS | 3 |
| 3.0 REQUIREMENTS | |
| 3.1 General | 3 |
| 3.2 Test Samples | 3 |
| 3.3 Acceptance Tests | 4 |
| 3.4 Qualification Test Cycling | 4 |
| 3.5 Environmental Life Tests | 5 |
| 3.6 Extended Life Test | 5 |
| 4.0 QUALITY ASSURANCE PROVISIONS | 5 |

| | | |
|--|---------------------------------|------------|
| <p>CRITICAL TEST PLAN FOR BIPROPELLANT
INJECTOR SOLENOID VALVES
ADVANCED SYNCOM</p> <p>HUGHES AIRCRAFT CO.
CULVER CITY, CALIF.</p> | <p>NUMBER
475000-609</p> | |
| | <p>CODE IDENT
NO. 82577</p> | <p>REV</p> |
| | <p>PAGE 2 OF 5</p> | |
| | <p>DATE 10/10/63</p> | |

SPECIFICATION SHEET

1.0 SCOPE

1.1 Purpose - It is the purpose of this test plan to evaluate and demonstrate the operation of the injector solenoid valves under simulated Advance Syncom mission profiles (Launch and Orbit operation).

1.2 Description - The injector solenoid valves (eight per spacecraft), control fuel, and oxidizer flow to the thrust chambers.

2.0 APPLICABLE DOCUMENTS

- | | | |
|-----|------------------|--|
| 2.1 | X254044, Rev. A | Bipropellant Reaction Control Specification |
| 2.2 | MTS 0612, Rev. A | The Marquardt Company Performance Test Specification |
| 2.3 | 475000-602 | Environmental Requirements for Flight Spare Acceptance Tests of Major Control Items Selected Minor Control Items |
| 2.4 | 475000-601 | Environmental Requirements for Qualification Tests of Major Control Items and Selected Minor Control Items |
| 2.5 | 475000-607 | Visual-Mechanical Inspection Requirements for Control Items |

3.0 REQUIREMENTS

3.1 General

3.1.1 The test valves selected shall be characteristic and identical in configuration to all flight production units and shall be assembled, inspected, tested and handled in the same manner as all flight production units.

3.1.2 Advanced Syncom solenoid drivers (475307-100) will be used to energize the solenoid valves.

3.1.3 Performance tests in accordance with MTS 0612 shall be conducted, as necessary to complete Table III (MTS 0612, page 7), prior to and after each environment test.

3.2 Test Samples - A representative sample of 16 valve assemblies will be selected for the test program. Performance tests will be initially performed on all test specimens in accordance with the Performance Test Specification, MTS 0612.

| | | |
|---|-------------------------|-----|
| CRITICAL TEST PLAN FOR BI-PROPELLANT
INJECTOR SOLENOID VALVES
ADVANCED SYNCOM | NUMBER
475000-609 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 3 OF 5 | |
| | DATE 10/10/63 | |

SPECIFICATION SHEET

3.3 Acceptance Tests

3.3.1 The 16 valves shall be divided into four groups of four each and subjected to one cycle of each of the following environments at acceptance levels called out in 475000-602.

- 1) Shock and vibration
- 2) Acoustical noise
- 3) Boost acceleration
- 4) Thermal-vacuum and spin

3.3.2 During thermal-vacuum and spin the valves shall be pulsed 200 times at 75 milliseconds "ON", 525 ms "OFF".

3.3.3 During environments 1, 2 and 3 the valves shall be tested for leakage in accordance with MTS 0612, except the period shall be the length of the test.

3.3.4 The four groups of valve assemblies shall be tested in the following order relative to environments listed above:

| <u>Group</u> | <u>Environment</u> |
|--------------|--------------------|
| 1 | 1, 2, 3, 4 |
| 2 | 1, 2, 4, 3 |
| 3 | 3, 4, 1, 2 |
| 4 | 2, 4, 3, 1 |

3.3.5 Upon completion of acceptance testing, one valve assembly from each group will be placed on an environmental life test; the rest will undergo qualification test cycling.

3.4 Qualification Test Cycling

3.4.1 Twelve valve assemblies, upon completion of acceptance tests, shall be subjected to the environments listed in the acceptance tests according to the plan devised, except that qualification levels as called out in 475000-601 will be used.

3.4.2 Subsequent to group qualification testing, one valve assembly from each group will be placed on an environmental life test. The remaining valve assemblies shall be subjected to additional cycles of each environment of 3.4.1 as follows:

| <u>Group</u> | <u>Environment</u> |
|--------------|--|
| 1 | 1 - (vibration duration shall be multiplied by 3 and 3 times as many shocks shall be applied). |

CRITICAL TEST PLAN FOR BI-PROPELLANT
INJECTOR SOLENOID VALVES
ADVANCED SYNCOM

HUGHES AIRCRAFT CO.
CULVER CITY, CALIF.

SPECIFICATION SHEET

NUMBER

475000-609

CODE IDENT
NO. 82577

REV

PAGE 4 OF 5

DATE 10/10/63

3.4.2 (continued)

| <u>Group</u> | <u>Environment</u> |
|--------------|---|
| 2 | 2 - (duration shall be multiplied by 3) |
| 3 | 3 - (test shall be repeated 3 times) |
| 4 | 4 - (time shall be 168 hours) |

3.5 Environmental Life Tests

3.5.1 One valve from the acceptance tests and one valve from the qualification test shall be subjected to pulsing (see paragraph 3.5.5) at expected low temperatures (see 475000-602) with N_2O_4 as the propellant.

3.5.2 One valve from the acceptance test and one valve from the qualification tests shall be subjected to pulsing (see paragraph 3.5.5) at expected high temperatures (see 415000-602) with MMH as the propellant.

3.5.3 One valve from the acceptance tests and one valve from the qualification tests shall be subjected to pulsing (see paragraph 3.5.5) at expected high temperatures (see 475000-602) with N_2O_4 as the propellant.

3.5.4 One valve from the acceptance tests and one valve from the qualification tests shall be subjected to pulsing (see paragraph 3.5.5) at expected high temperatures (see 475000-602) with MMH as the propellant.

3.5.5 The valves shall be pulsed for 25,000 operations over a 30 day period. The pulsing operation will include at least two periods of 100 milliseconds "ON", 500 milliseconds "OFF" and two periods of 14 minutes "ON".

3.5.6 Performance tests (see paragraph 3.1.3) shall be conducted at 5,000 pulse intervals.

3.6 Extended Life Test

3.6.1 Following the environmental life tests, the valve assemblies from 3.5 will be operated under ambient conditions with the same propellants and pulsed for 10 pulses (100 milliseconds "ON", 500 ms OFF) once a day for at least 1 year.

3.6.2 Complete functional tests in accordance with MTS 0612 shall be conducted once each month.

4.0 QUALITY ASSURANCE PROVISIONS

4.1 See 475000-607

| | | |
|---|---------------------------------|------------|
| <p>CRITICAL TEST PLAN FOR BIPROPELLANT
INJECTOR SOLENOID VALVES
ADVANCED SYNCOM</p> | <p>NUMBER
475000-609</p> | |
| | <p>CODE IDENT
NO. 82577</p> | <p>REV</p> |
| <p>HUGHES AIRCRAFT CO.
CULVER CITY, CALIF.</p> | <p>PAGE 5 OF 5</p> | |
| | <p>DATE 10/10/63</p> | |

SPECIFICATION SHEET

CRITICAL TEST PLAN FOR BATTERIES

REVISED:

| | | | |
|---|--|-------------------------|-----|
| PREPARED
J. C. Meyer | | NUMBER
475000-606 | |
| APPROVED 10-17-63
<i>J. C. Meyer</i> | | CODE IDENT
NO. 82577 | REV |
| APPROVED | HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 1 OF 5 | |
| APPROVED | | DATE: 10-17-63 | |

SPECIFICATION SHEET

1.0 SCOPE

This specification covers the critical testing of the Advanced Syncom batteries (P/N 475254-100 and 475255-100) and battery cells (P/N 250204).

1.1 Purpose: It is the purpose of this test to evaluate and demonstrate the operation of the batteries under simulated Advanced Syncom mission profiles (launch and orbit operation).

1.2 Description: The energy storage system for the Advanced Syncom spacecraft includes four parallel nickel-cadmium batteries consisting of 22 hermetically sealed, series-connected, six-ampere-hour cells. These batteries provide:

1) Storage of electrical energy for operation during the launch phase and during eclipse.

2) Energy capacity for pulse loads such as the apogee motor igniter and the bipropellant reaction control system solenoids.

The batteries are considered critical since they require:

1) A detailed specification and control of the active elements and mechanical properties such as the seal.

2) A comprehensive selection and test program to meet the long lifetime requirement.

A minimum of 88 cells (equivalent of four Advanced Syncom batteries) representative of flight hardware will be procured for the test program.

2.0 APPLICABLE DOCUMENTS

| | |
|------------|--|
| 30630-001 | Procurement Specification, Sealed Nickel-Cadmium Battery Cell, 6.0 Amp-Hr. |
| 475254-600 | Advanced Syncom Battery Test Plan |
| 475000-601 | Environmental Requirements for Qualification Tests of Major Control Items and Selected Minor Control Items |
| 475000-602 | Environmental Requirements for Flight Spare Acceptance Tests of Major Control Items and Selected Minor Control Items |
| 475000-607 | Visual-Mechanical Inspection Requirements for Control Items. |

| | | |
|--|-------------------------|-----|
| Critical Test Plan for Batteries | NUMBER | |
| | 475000-606 | |
| | CODE IDENT
NO. 82577 | REV |
| | PAGE 2 OF 5 | |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | DATE 10-17-63 | |
| | SPECIFICATION SHEET | |

3.0 REQUIREMENTS

3.1 General: The test cells selected shall be characteristic and identical in configuration to all flight production units and shall be assembled, inspected, tested, and handled in the same manner as all flight production units. Procurement Acceptance Tests in accordance with Hughes Aircraft Company Procurement Specification 30630-001 will be conducted on all test cells.

In addition, 20 cells will be subjected to the qualification tests of Hughes Aircraft Company Procurement Specification 30630-001, paragraphs 4.4.7 through 4.4.13.

3.2 Acceptance Tests: The 88 battery cells shall be divided into four groups of 22 cells each and subjected to one cycle of each of the following environments in accordance with Test Plan Number 475000-602:

- Environment 1) Shock and vibration
- Environment 2) Thermal-vacuum and spin
- Environment 3) Boost acceleration
- Environment 4) Acoustical noise

Performance tests in accordance with Test Plan Number 475254-600, paragraph 4.22 shall be conducted prior to, during, and after each environmental exposure.

The four groups of 22 cells shall be tested in the following order relative to environments 1, 2, 3, and 4.

- Group I 1, 2, 3, 4
- Group II 2, 3, 4, 1
- Group III 3, 4, 1, 2
- Group IV 4, 1, 2, 3

Upon completion of acceptance testing, one-half of the units from each group will be placed on an extended life test in accordance with paragraph 3.3 and the second half of each group will undergo qualification test cycling in accordance with paragraph 3.4.

3.3 Extended Life Test: Forty-four battery cells shall be subjected to concurrent temperature, vacuum-spin tests under the following conditions:

- 1) Test to be conducted for a period of 30 days.
- 2) During the 30-day period the temperature shall be cycled between the high and low temperatures referenced in Test Plan Number 475000-602. The

| | | |
|--|-------------------------|------|
| Critical Test Plan for Batteries | NUMBER | |
| | 475000-606 | |
| | CODE IDENT
NO. 82577 | REV |
| | PAGE 3 | OF 5 |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | DATE 10-17-63 | |
| | | |

SPECIFICATION SHEET

temperature cycling shall include at least 200 cycles while simulating the expected temperature-time environment.

3) The spin rate and vacuum environments shall be in accordance with Test Plan Number 475000-602.

4) Actual operational electrical load cycling shall be programmed.

Battery cell performance, i.e., cell voltages, currents, and temperatures shall be monitored and recorded during the above environmental exposure. A more comprehensive performance check in accordance with paragraph 4.1.2 of Test Plan Number 475254-601 shall be made and recorded prior to and after the environmental exposure.

Following the environmental life tests, the same cells shall be operated with simulated spacecraft loading under laboratory ambient conditions for a period of at least one year. Operational load cycling is shown in the following table. Performance tests in accordance with paragraph 4.1.2 of Test Plan Number 475254-600 shall be made at the end of cycles 1, 41, 42, 82, and 83.

| Cycles | Charge
Current,
Amperes | Charge
Time | Discharge
Current,
Amperes | Discharge
Time | Temp.
°F. |
|--------|-------------------------------|----------------|----------------------------------|-------------------|--------------|
| 1 | 0.165 | 70 days | 0 | 0 | 75° |
| 2-11 | 0.165 | 23.4 hr | 1.1 | 0.6 hr | 40° |
| 12-31 | 0.165 | 22.8 hr | 1.1 | 1.2 hr | 10° |
| 32-41 | 0.165 | 23.4 hr | 1.1 | 0.6 hr | 40° |
| 42 | 0.165 | 140 days | 0 | 0 | 75° |
| 43-52 | 0.165 | 23.4 hr | 1.1 | 0.6 hr | 40° |
| 53-72 | 0.165 | 22.8 hr | 1.1 | 1.2 hr | 10° |
| 73-82 | 0.165 | 23.4 hr | 1.1 | 0.6 hr | 40° |
| 83 | 0.165 | 70 days | 0 | 0 | 75° |

Table 1. Operational Load Cycling

3.4 Qualification Test Cycling: Forty-four battery cells selected upon completion of acceptance testing shall be subjected to one cycle of the following qualification level environments in accordance with Test Plan Number 475000-601.

| | | |
|--|-------------------------|-----|
| Critical Test Plan for Batteries | NUMBER
475000-606 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 4 OF 5 | |
| | DATE 10-17-63 | |

SPECIFICATION SHEET

- Environment 1) Shock and vibration
- Environment 2) Thermal vacuum and spin
- Environment 3) Boost acceleration
- Environment 4) Acoustical noise

The battery cells selected for the qualification test cycling shall be divided into four groups and tested in the following environment of sequence:

- Group I 1, 2, 3, 4
- Group II 2, 3, 4, 1
- Group III 3, 4, 1, 2
- Group IV 4, 1, 2, 3

Battery cell performance in accordance with paragraph 4.2.2(b) of Test Plan Number 475254-600 shall be monitored and recorded during the above environmental exposures. A more comprehensive performance check in accordance with paragraph 4.1.2 of Test Plan Number 475254-600 shall be made and recorded prior to and after each environmental exposure.

Subsequent to the above group qualification test cycling, battery cells from each group shall be selected and subjected to one environment in accordance with the level requirements in Test Plan Number 475000-601 as follows:

- Group I Environment 4 (Vibration and Shock)
- Group II Environment 1 (Thermal Vacuum and Spin)
- Group III Environment 2 (Boost Acceleration)
- Group IV Environment 3 (Acoustical Noise)

At the conclusion of each test, performance data in accordance with paragraph 4.1.2 of Test Plan Number 475254-600 shall be recorded. Results of this test will be submitted to the Project Reliability and Test Section for analysis.

4.0 QUALITY ASSURANCE PROVISIONS

Quality Assurance Provisions shall be in accordance with Test Plan Number 475000-607.

| | | |
|--|-------------------------|-----|
| Critical Test Plan for Batteries | NUMBER
475000-606 | |
| | CODE IDENT
NO. 82577 | REV |
| | PAGE 5 OF 5 | |
| | DATE 10-17-63 | |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | | |
| SPECIFICATION SHEET | | |

CRITICAL TEST PLAN FOR SEPARATION SWITCHES
ADVANCED SYNCOM

REVISED:

| | | | |
|---------------------------------|--|-------------------------|-----|
| PREPARED
J. C. Meyer | | NUMBER
475000-601 | |
| APPROVED 12-1-63
J. C. Meyer | | CODE IDENT
NO. 82577 | REV |
| APPROVED | HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 1 OF 4 | |
| APPROVED | | DATE: 10/10/63 | |

SPECIFICATION SHEET

INDEX

| | <u>Page</u> |
|----------------------------------|-------------|
| 1.0 SCOPE | 3 |
| 1.1 Purpose | 3 |
| 1.2 Description | 3 |
| 2.0 APPLICABLE DOCUMENTS | 3 |
| 3.0 REQUIREMENTS | 3 |
| 3.1 General | 3 |
| 3.2 Performance Tests | 4 |
| 3.3 Qualification Tests | 4 |
| 4.0 QUALITY ASSURANCE PROVISIONS | 4 |

| | | |
|---|---------------------------------|------------|
| <p>CRITICAL TEST PLAN FOR SEPARATION SWITCHES
ADVANCED SYNCOM</p> | <p>NUMBER
475000-610</p> | |
| | <p>CODE IDENT
NO. 82577</p> | <p>REV</p> |
| <p>HUGHES AIRCRAFT CO.
CULVER CITY, CALIF.</p> | <p>PAGE 2 OF 4</p> | |
| | <p>DATE 10/10/63</p> | |

SPECIFICATION SHEET

1.0 SCOPE

This specification covers the critical testing of the Advanced Syncom separation switches.

1.1 Purpose - It is the purpose of this test to evaluate and demonstrate the operation of the separation switches under simulated Advanced Syncom mission profiles (launch and orbit operation).

1.2 Description - The separation switches (2 pole, double throw) have two functions in the spacecraft. Four switches, one for each central timer, keeps reset voltage on the timers until separation of the spacecraft from the Agena. Two switches apply ground to the apogee motor squibs from launch to separation.

2.0 APPLICABLE DOCUMENTS

| | |
|------------|--|
| | Procurement Specification, Separation Switches |
| 475000-601 | Environmental Requirements for Qualification Tests of Major Control Items and Selected Minor Control Items |
| 475000-602 | Environmental Requirements for Flight Spare Acceptance Tests of Major Control Items and Selected Minor Control Items |
| 475000-607 | Visual-Mechanical Inspection Requirements for Control Items |

3.0 REQUIREMENTS

3.1 General - This test plan is based on the assumptions that the nature of operation of the switches is "one-shot" and that the environments described are representative of mission conditions. Testing will be accomplished to

- 1) Show the ability of the switch to operate under expected mission conditions.
- 2) Demonstrate a reliability of 0.999 with a reasonable confidence.

The switches used for this test will be from the same manufacturing lot as those to be used in the flight spacecraft.

| | | |
|---|-------------------------|-----|
| CRITICAL TEST PLAN FOR SEPARATION SWITCHES
ADVANCED SYNCOM | NUMBER
475000-610 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 3 OF 4 | |
| | DATE 10/10/63 | |
| SPECIFICATION SHEET | | |

3.2 Performance Tests - A total of twelve switches will be used for this test plan. Prior to beginning any test, each switch will be tested in accordance with the requirements specified in Procurement Specification No

3.3 Qualification Tests - The twelve switches will be subjected to the following environments at the levels and durations as specified in Test Plan No. 475000-601 except where changes are noted below.

Environments

- 1) Vibration (combined sinusoidal, random, and shock)
- 2) Acceleration (boost and spin)
- 3) Acoustical noise
- 4) Thermal-vacuum (12 hours with one cycle of highest and lowest expected temperatures)
- 5) Low-temperature operation
- 6) High-temperature operation

The thermal-vacuum environment is to be applied to all of the twelve switches at one time. Subsequent to the thermal-vacuum test the switches will be divided into groups of two and tested as follows:

| <u>Switches</u> | <u>Environments</u> |
|-----------------|---------------------|
| 2 units | 1, 2, 3, 5, 6 |
| 2 units | 1, 2, 3, 6, 5 |
| 2 units | 2, 3, 1, 5, 6 |
| 2 units | 2, 3, 1, 6, 5 |
| 2 units | 3, 1, 2, 5, 6 |
| 2 units | 3, 1, 2, 6, 5 |

Each switch shall meet the requirements of Procurement Specification No. before and after each environment.

After the application of environments 1, 2, and 3, each switch will be operated for 100 cycles. Each switch will be operated for 1000 cycles under environments 5 and 6.

4.0 QUALITY ASSURANCE PROVISIONS

Quality Assurance Provision shall be in accordance with Test Plan Number 475000-607.

| | |
|---|--------------------------------------|
| <p>CRITICAL TEST PLAN FOR SEPARATION SWITCHES
ADVANCED SYNCOM</p> | <p>NUMBER
475000-610</p> |
| <p>HUGHES AIRCRAFT CO.
CULVER CITY, CALIF.</p> | <p>CODE IDENT NO. 62577 REV</p> |
| <p>SPECIFICATION SHEET</p> | <p>PAGE 4 OF 4
DATE 10/10/63</p> |

CRITICAL TEST PLAN FOR SWITCH
RF TRANSMITTER
ADVANCED SYNCOM

| | | | |
|-----------------------------------|--|-------------------------|-----|
| PREPARED
R. J. Parfitt | HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | NUMBER
475173-601 | |
| APPROVED
<i>R. Meyer 11-63</i> | | CODE IDENT
NO. 82577 | REV |
| APPROVED | | PAGE 1 OF 4 | |
| APPROVED | | DATE: 11/10/63 | |

SPECIFICATION SHEET

INDEX

| | <u>Page</u> |
|--------------------------|-------------|
| 1.0 SCOPE | 3 |
| 1.1 Purpose | 3 |
| 1.2 Description | 3 |
| 2.0 APPLICABLE DOCUMENTS | 3 |
| 3.0 REQUIREMENTS | |
| 3.1 General | 3 |
| 3.2 Acceptance Test | 3 |
| 3.3 Life Test | 4 |

CRITICAL TEST PLAN FOR SWITCH, RF TRANSMITTER
ADVANCED SYNCOM

HUGHES AIRCRAFT CO.
CULVER CITY, CALIF.

SPECIFICATION SHEET

NUMBER

475173-601

CODE IDENT
NO. 82577

REV

PAGE 2 OF 4

DATE 10/10/63

1.0 SCOPE

1.1 Purpose - The purpose of this test plan is to establish requirements and procedures for evaluating and demonstrating the operation of the transmitter RF switch under simulated launch and orbit mission profiles of Advanced Syncom.

1.2 Description - The RF switch is a 3 port circulator where the direction is controlled by the direction of a permanent magnetic field. Switching is accomplished by applying current to an external coil, which changes the direction of the magnetic field. The current is removed after switching is accomplished. The switch provides a means of alternately driving a common antenna system with two different TWT's.

2.0 APPLICABLE DOCUMENTS

- | | | |
|-----|------------|--|
| 2.1 | 475000-607 | Visual-Mechanical Inspection Requirements for Control Units. |
| 2.2 | 475173-600 | Test Plan, RF Switch, Transmitter |
| 2.3 | 475000-601 | Environmental Requirements for Qualification Tests of Major Control Items and Selected Minor Control Items |
| 2.4 | 475000-602 | Environmental Requirements for Flight Spare Acceptance Tests of Major Control Items and Selected Minor Control Items |

3.0 REQUIREMENTS

3.1 General - A minimum of 21 flight 4 gc ferrite switches shall be procured, of which 4 shall be used for the test program. These 4 shall be characteristic and identical in configuration to flight production units, and shall be handled, inspected, and tested in the same manner as the flight production units.

3.2 Acceptance Test - The 4 switches shall be subjected to the following environments at the levels specified in Test Plan No. 475000-602.

- 1) Vibration and shock (combined)
- 2) Acceleration (spin and boost)
- 3) Acoustical noise
- 4) Thermal-vacuum

Each switch will be identified by number and will be subjected to the above environments in the following sequence.

| | | |
|--|-------------------------|-----|
| CRITICAL TEST PLAN FOR SWITCH, RF TRANSMITTER
ADVANCED SYNCOM | NUMBER
475173-601 | |
| | CODE IDENT
NO. B2577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 3 OF 4 | |
| | DATE 10/10/63 | |
| SPECIFICATION SHEET | | |

Switch NumberEnvironmental Sequence

| | |
|---|------------|
| 1 | 1, 2, 3, 4 |
| 2 | 2, 3, 4, 1 |
| 3 | 3, 4, 1, 2 |
| 4 | 4, 1, 2, 3 |

Performance tests in accordance with Test Plan 475173-600 shall be performed on each switch before and after subjection to each environment.

3.3 Life Test - After successful completion of the environmental tests, each switch shall be electrically cycled 1000 times. RF measurements in accordance with Test Plan 475173-600 will be made after each 100 cycles.

4.0 QUALITY ASSURANCE PROVISIONS

Same as directed in Test Plan 475173-600.

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| CRITICAL TEST PLAN FOR SWITCH, RF TRANSMITTER
ADVANCED SYNCOM | NUMBER
475173-601 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 4 OF 4 | |
| | DATE 10/10/63 | |

SPECIFICATION SHEET

CRITICAL TEST PLAN FOR SWITCH
RF RECEIVER
ADVANCED SYNCOM

REVISED:

| | | | |
|--------------------------------------|--|-------------------------|------|
| PREPARED
R. J. Parfitt | HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | NUMBER
475176-601 | |
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<i>J. Meyer</i> 10-11-63 | | CODE IDENT
NO. 82577 | REV. |
| APPROVED | | PAGE 1 OF 4 | |
| APPROVED | | DATE: 10/10/63 | |

SPECIFICATION SHEET

INDEX

| | <u>Page</u> |
|----------------------------------|-------------|
| 1.0 SCOPE | |
| 1.1 Purpose | 3 |
| 1.2 Description | 3 |
| 2.0 APPLICABLE DOCUMENTS | 3 |
| 3.0 REQUIREMENTS | |
| 3.1 General | 3 |
| 3.2 Acceptance Tests | 3 |
| 3.3 Life Test | 4 |
| 4.0 QUALITY ASSURANCE PROVISIONS | 4 |

| | | |
|---|---------------------------------|------------|
| <p>CRITICAL TEST PLAN FOR SWITCH, RF RECEIVER
ADVANCED SYNCOM</p> | <p>NUMBER
475176-601</p> | |
| | <p>CODE IDENT
NO. 82577</p> | <p>REV</p> |
| <p>HUGHES AIRCRAFT CO.
CULVER CITY, CALIF.</p> | <p>PAGE 2 OF 4</p> | |
| | <p>DATE 10/10/63</p> | |

SPECIFICATION SHEET

1.0 SCOPE

1.1 Purpose - The purpose of this test plan is to establish requirements and procedures for evaluating and demonstrating the operation of the receiver RF switch under simulated launch and orbit mission profiles of Advanced Syncom.

1.2 Description - The RF switch is a 3 port, 6 gc circulator where the direction is controlled by the direction of a permanent magnetic field. Switching is accomplished by applying current to an external coil, which changes the direction of the magnetic field. The current is removed after switching is accomplished. The switch provides a means of alternately driving two receivers from a common RF input.

2.0 APPLICABLE DOCUMENTS

- | | | |
|-----|------------|---|
| 2.1 | 475000-607 | Visual Mechanical Inspection Requirements for Control Units. |
| 2.2 | 475176-600 | Test Plan, RF Switch, Receiver |
| 2.3 | 475000-601 | Environmental Requirements for Qualification Tests of Major Control Items and Selected Minor Control Items |
| 2.4 | 475000-602 | Environmental Requirements for Flight Spare Acceptance Tests of Major Control Items and Selected Minor Control Items. |

3.0 REQUIREMENTS

3.1 General - A minimum of 21 flight 6 gc ferrite switches shall be procured, of which 4 shall be used for the test program. These 4 shall be characteristic and identical in configuration to the flight production units, and shall be handled, inspected, and tested in the same manner as the flight production units.

3.2 Acceptance Tests - The 4 switches shall be subjected to the following environments, at the levels specified in Test Plan No. 475000-602.

- 1) Vibration and shock
- 2) Acceleration (spin and boost)
- 3) Acoustical noise
- 4) Thermal vacuum

Each switch will be identified by number, and will be subjected to the above environments in the following sequence.

| | | |
|---|-------------------------|-----|
| CRITICAL TEST PLAN FOR SWITCH, RF RECEIVER
ADVANCED SYNCOM | NUMBER
475176-601 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 3 OF 4 | |
| | DATE 10/10/63 | |

SPECIFICATION SHEET

| <u>Switch Number</u> | <u>Environmental Sequence</u> |
|----------------------|-------------------------------|
| 1 | 1, 2, 3, 4 |
| 2 | 2, 3, 4, 1 |
| 3 | 3, 4, 1, 2 |
| 4 | 4, 1, 2, 3 |

Performance tests in accordance with Test Plan 475176-600 shall be performed on each switch before and after subjection to each environment.

3.3 Life Test - After successful completion of the environmental tests, each switch shall be electrically cycled 1000 times. RF measurements in accordance with Test Plan 475176-600 will be made after each 100 cycles.

4.0 QUALITY ASSURANCE PROVISIONS

Same as stated in test plan 475176-600.

| | | |
|---|-------------------------|-----|
| CRITICAL TEST PLAN FOR SWITCH, RF RECEIVER
ADVANCED SYNCOM | NUMBER
475176-601 | |
| | CODE IDENT
NO. 82577 | REV |
| HUGHES AIRCRAFT CO.
CULVER CITY, CALIF. | PAGE 4 OF 4 | |
| | DATE 10/10/63 | |

SPECIFICATION SHEET

QUALITY ASSURANCE

The Quality Assurance effort during this phase of the Advanced Syncom program has included planning a quality program for the Development and Launch Program, preparation of control documents, evaluation of tentative suppliers, and inspection of hardware. A concerted effort has been devoted to complete revision and upgrading of the El Segundo Quality Assurance Instructions and Culver City Quality Control Instructions, to provide for greater utilization of standard operating procedures and reduce the total number of unique situations.

Quality Control Operations. A representative of the Quality Control Supplier Evaluation group has been actively participating in the survey of all suppliers of electronics parts to be used in Advanced Syncom. Supplier quality control systems were evaluated in terms of requirements established in NPC 200-3. Suppliers that did not meet these requirements were given conditional or unapproved ratings and requests for corrective action. Re-surveys will be conducted when completion of corrective action is indicated by the supplier.

A facility survey of the Marquardt Corporation was performed in June 1963. Based on the findings, Marquardt was given a conditional rating. Marquardt Quality Control indicated that corrective action will be completed by 20 October 1963. A preliminary Quality Control Plan has been completed and a detail plan, which will comply with NPC 200-2, will be submitted for Phase II of the program. Engine and system tests were monitored by Hughes Quality Control and discrepancies noted. Hughes Aircraft Company Statement of Work No. 170426, Revision E., dated 1 August 1963 waived Marquardt quality control coverage for the remainder of the interim

phase of the program. Marquardt Quality Control documentation to reflect acceptance of the engineering model will not be prepared based on this waiver.

In-process inspection has been performed on the Advanced Syncom hardware fabricated and assembled during this phase of the program. This hardware is classified as experimental, requiring only minimum controls and inspection.

Preprocurement Activities. The Quality Assurance procurement efforts during this report period involved participation in the Procurement Committee evaluation and source selection recommendation of suppliers for the Advanced Syncom batteries and solar cells. In both cases RFPs were sent to five suppliers. NASA Quality Publication NPC 200-3 was made applicable to both of these procurements. The Quality evaluation of each supplier was based on recent surveys, proposal inputs, and the Quality Manuals of prospective suppliers.

Quality Assurance Inspection Instructions. The NASA requirements for inspection instructions were analyzed during this period. Documentation has been generated to state the criteria to be utilized in the preparation and issuance of the instructions. Formal documentation will be released shortly in the form of a Quality Assurance Instruction (QAI).

Available hardware drawings have been viewed to determine the need for specific documentation and provide a basis for the formulation of this documentation. At this time material flow charts have not been finalized; consequently, exact locations of in-process inspection operations cannot be defined. Every effort is being expended toward assuring complete correlation of the inspection operations with the total manufacturing effort.

Hughes Aircraft Company

A listing of Quality Assurance Instructions has been included in the Quality Program Plan, given later in this section. These instructions are in part dependent on results of the documentation review and on final delineation of equipment requirements.

To date, more than 150 instructions have been reviewed and revised relating to general quality operations—procurement, fabrication, assembly, compliance/packaging/shipping inspection operations, and calibration. Also included are specific requirements necessary to satisfy program requirements. This task will be completed by 31 October 1963.

In addition to, and in conjunction with, the above effort, a complete review of the documentation system for Advanced Syncom was undertaken, directed toward standardization and minimizing of efforts involved and ensuring compliance with the requirements of NASA Quality Publication NPC 200-2. This review encompasses all aspects of the Hughes documentation system affecting the product.

Aerospace Group Product Assurance Quality Control Instructions (QCIs) describe the manner in which inspections, tests, sampling procedures, use of forms, and other responsibilities are executed. The following is a list of current QCIs that are used as required during the fabrication, assembly, and test phases of contracts requiring deliverable hardware.

| QCI
Number | Subject |
|---------------|---------|
|---------------|---------|

Administrative — General

- | | |
|---|--|
| A | Quality Control Instruction Initiation |
| B | Quality Control Instruction Distribution |

- | QCI
Number | Subject |
|---------------|---|
| C | Inspection and Test — General Requirements and Responsibilities |
| E | Quality Source Liaison Engineers — Project Assignments |
| 1.1-3 | Quality Control Reporting — All Fabrication Areas |
| 1.1-4 | Use of Hughes Form 778, Request for Inspection and Test |
| 1.1-5 | Control of Nonconforming Supplies |
| 1.1-6 | Replacement of Discrepant or Damaged Supplies Found in Assembly |
| 1.1-7 | Control Point, Corrective Action Requests |
| 1.1-8 | Use of Hughes Form 805, Corrective Action Requests |
| 1.1-9 | Quality Capability Surveys of Suppliers |
| 1.1-11 | Control of Government Furnished Facilities Equipment |
| 1.1-12 | Hughes Form 3236, Variance Authorization |
| 1.1-13 | Hughes Form 672A, Substitution Authorization |
| 1.1-14 | Preprocurement Source Selection — Major Subcontract |
| 1.1-15 | Internal Quality Audits |
| 1.1-17 | Use of Hughes Form 359 or 10301CC Inspection and Test Report |
| 1.1-18 | Control and Use of Quality Control (Inspection) Stamps |
| 1.1-19 | Quality Control History and Inspection Status Record |

6. Reliability and Quality Assurance

| QCI
Number | Subject |
|-----------------------------------|--|
| 1.1-20 | Use of Form 3065 Floor Inspection
Travel Copy |
| 1.1-21 | Document Control Center, Inspection,
and Test Section |
| Inspection Instruction | |
| 2.1-2 | Etched Circuit Boards |
| 2.1-3 | Hughes-Furnished Subcontract Mate-
rials |
| 2.1-4 | Clean Room Control and Surveillance |
| 2.1-5 | Inspection of Micro Resistance Weld-
ing |
| 2.1-6 | Periodic Inspection of Stores and
Warehouses |
| 2.1-7 | Workmanship Standards |
| 2.1-8 | Tooling Inspection |
| 2.1-9 | Final Inspection of Missiles and Space
Vehicles |
| 2.1-10 | Hughes Source Inspection — Inspec-
tion and Test Section |
| 2.1-11 | Bearing Inspection |
| 2.1-13 | NASA Soldering High Reliability Con-
nections, MSFC-PROC-158B |
| Receiving and Shipping Inspection | |
| 2.2-1 | Receiving Inspection Sampling Plan |
| 2.2-2 | Receiving Inspection Instruction Sheet |
| 2.2-3 | Raw Stock Receiving |
| 2.2-4 | Vendor Quality Rating System |
| 2.2-6 | Quality Control Reporting — Receiv-
ing Inspection |

| QCI
Number | Subject |
|-------------------------------|---|
| 2.2-7 | Receiving Inspection — General Re-
quirements and Responsibilities |
| 2.2-8 | Packaging and Shipping Inspection |
| 2.2-9 | Source Inspection — Supplier Evalu-
ation Section |
| 2.2-11 | Quality Control Screening of Scrap or
Salvable Material |
| 2.2-12 | High Reliability Receiving Inspection |
| 2.2-13 | Control of Damaged Government prop-
erty (GP) Receiving Inspection |
| 2.2-14 | Receiving Inspection of Government-
Furnished Property (GFP) and
Bailed Property (BP) |
| 2.2-15 | Quality Control Screening Procure-
ment Documents |
| 2.2-16 | Evaluation of Special Process Sup-
pliers |
| 2.2-17 | Initial Review, Nonconforming, or
Discrepant Supplies/Material |
| 2.2-18 | Maintenance of Specification Files,
Receiving Inspection |
| 2.2-19 | Hughes Source Quality Surveillance,
Supplier Evaluation Section |
| Quality Assurance Engineering | |
| 4.1-1 | Quality Operating Plan |
| 4.1-2 | Certification of Micro Resistance
Welding Operators |
| 4.1-3 | Certification of Metal Finishing Proc-
esses |
| 4.1-4 | Qualification/Certification of Heat
Treating |

Hughes Aircraft Company

| QCI
Number | Subject |
|---|---|
| 4.1-5 | Certification/Qualification of Class
"A" Welding Operators |
| 4.1-7 | Test Verification and Performance |
| 4.1-8 | Qualification of Micro Resistance
Welding Machines |
| 4.1-9 | Welding Inspection, Fusion |
| 4.1-10 | Heat Treating Inspection |
| 4.1-11 | Plating Inspection |
| 4.1-12 | Painting Inspection |
| 4.1-13 | Welding, Resistance, Inspector Quali-
fication |
| 4.1-14 | Accuracy of Test Measurement Equip-
ment |
| 4.1-15 | Qualification and Certification of Pene-
trant Inspectors |
| 4.1-16 | Adhesive Bonding Inspection |
| 4.1-17 | Penetrant Inspection Type I and II |
| <p>Project-oriented QCIs are prepared to provide specific guidance for operating personnel. The following preliminary Advanced Syncom QCIs have been prepared for the development and launch phase of the Advanced Syncom program and are included in the supplement.</p> | |
| 3.12-0 | Inspection Check List |
| 3.13-1 | Quality Requirements |
| 3.13-2 | Screening Assist Work Authorizations |
| 3.13-3 | Screening Purchase Orders |
| 3.13-4 | Receiving Inspection Purchased Parts |
| 3.13-5 | Bonded Stores Inspection |

| QCI
Number | Subject |
|---------------|---|
| 3.13-6 | Screening Work Orders |
| 3.13.7 | Control of Parts Pending Release of
Released Prints |
| 3.13-8 | Control of Nonconforming Supplies
(will be prepared when MRB mem-
bership lists are available). |

Inspection Planning Sheets (IPSs) are prepared for specific processes, operations, and critical assemblies. Inspection Planning Sheets have been prepared for the following:

| Hughes Process No. | Subject |
|--------------------|-------------------------------------|
| 4-99 | Nickel Plating, Electrolytic |
| 4-112 | Protective Paint System |
| 4-142 | Vacuum Deposit, Aluminum |
| 9-24 | Removal of Soldering Flux |
| 16-41 | Adhesive Bonding |
| 16-80 | Adhesive Bonding |
| 16-91 | Foam-in-Place |
| No HP
number | Welded Module |
| No HP
number | Welded Module/Board As-
semblies |

Quality Program Plan. The Advanced Syncom Quality Program Plan is described on the following pages in its entirety.

Project Description. The Advanced Syncom program includes development, fabrication, test, and flight operations of spacecraft to be orbited by Atlas/Agena D launch vehicles and operated in conjunction with NASA communication ground stations for the performance of stationary — orbit experiments.

6. Reliability and Quality Assurance

The program includes development of an advanced engineering model, a ground test model, and four flight spacecraft. The engineering model will be used to finalize the design of the various subsystems and will therefore conform as nearly as possible to flight hardware form factors and weights. No power-aging or burned-in components shall be specified for installation in the engineering model.

The ground test (prototype) and flight models will be fabricated and tested to formal drawings and specifications. The prototype and flight models will be identical in all respects except that prototype components will have a 240-hour burn-in and flight components will have an additional 1260 hours of power aging.

The program also requires the fabrication of twelve communication transponders and five telemetry and command simulators for use in integration and prelaunch testing of the ground communications and telemetry and command stations. Level III Quality requirements as defined in Table 6-2 shall be utilized for these items.

The spacecraft systems are divided into major and minor control items as defined in the document listing later in this section. Prior to acceptance for assembly into the spacecraft, tests will be performed to assure unit compliance with established specifications. Systems test equipment will provide basic initial integration check-out and detailed examination of system parameters.

Reference Documents. Except as noted in the document listing, the following documents serve as the basis for the Quality Program Plan described herein:

NPC 200-2 Quality Program Provisions for Space Systems Contractors

TABLE 6-2. TASK BREAKDOWN BY QUALITY CONTROL LEVELS

F = Complete implementation
R = Implementation at project request
NA = Not applicable

| Quality Control Function | Level I | Level II | Level III |
|---|---------|----------|-----------|
| Design review | R | R | R |
| Change control | NA | F | F |
| Inspection and test planning | R | R | F |
| Inspection and test procedures | R | R | F |
| Statistical quality control | R | F | F |
| Training and personnel certifications | F | F | F |
| Quality standards | R | R | F |
| Supplier surveys | R | F | F |
| Purchase order screening | F | F | F |
| Source inspection | R | R | F |
| Receiving inspection | F | F | F |
| Supplier data | F | F | F |
| Supplier rating | NA | F | F |
| Stores and material control | R | F | F |
| In-process inspection and test | R | F | F |
| Process certification | R | F | F |
| Final inspection and test | R | F | F |
| Calibration | F | F | F |
| Preservation, handling, packaging, and shipping | R | R | F |
| Evidence of inspection | R | F | F |
| Material review board | R | F | F |
| Failure report and corrective action | R | F | F |
| Quality assurance audits and reports | F | F | F |
| Control of Government-furnished articles | F | F | F |

NPC 200-3

Inspection System Provisions for Suppliers of Space Materials, Parts, Components, and Services

MSFC-PROC-158B

Procedure for Soldering of Electrical Connections (High Reliability)

Mil-Std 105D Sampling Procedures
and Tables for Inspection by Attributes

Implementation of NASA documents shall be accomplished through the use of Hughes Space Systems Division Policies, Advanced Syncom Project Bulletins, detailed implementing instructions issued by the Quality organizations, and this Quality Program Plan. Those presently in use are given in the document listing and necessary additions and revisions will be made by the use of Advanced Syncom Project Bulletins which will define the modification.

The Hughes Advanced Syncom Project Office is responsible for compliance with all Quality aspects of the contract. Reporting to the Advanced Syncom Program Manager is the Product Effectiveness Department composed of Quality Assurance, Reliability, and Test. This office provides for the preparation and implementation of the quality program and for the coordination of requirements, approvals, reports, and negotiations with the NASA.

Fabrication of Advanced Syncom subsystems and systems, spacecraft system integration and test, and spacecraft environmental testing occur at both the Hughes Aircraft Company Culver City, California and El Segundo, California, facilities. Since both facilities have a completely staffed Quality Control organization, the Advanced Syncom Program shall utilize the services of each group. Coordination of Quality Control activities is accomplished between the divisions by the Project Office and audited by the Space Systems Division Quality Assurance organization. Detailed implementing documents issued by the quality organizations show uniformity of purpose and direction during these periods of transition and operation.

Design Controls

- Design Review. Quality Assurance review of Engineering drawings, specifications, and test plans is the direct responsibility of the Quality Assurance representatives in the Syncom Project. Additional Quality Assurance participation at all regularly scheduled design review meetings will be as directed by this office. All design reviews will be in accordance with SSD Procedure 9-2.7 and EP 7-17.2, Design Review.

- Change Control. Hughes Aircraft Company operates an Engineering Change Control System in accordance with procedures issued and maintained by the Engineering Data Department and published in the Engineering Procedures Manual. The specific portion of the manual directly related to and in effect on Advanced Syncom is EP 7-10.2, Syncom Project—Engineering Change Management. Quality departments shall assure full compliance of hardware with effectiveness as specified in Engineering change documents on all Level II and III hardware, in accordance with Space Systems Division Procedure 9-3.3, Quality Assurance Change Control.

Quality Planning

- Inspection and Test Planning. The servicing Quality Control Engineering sections will prepare instructions describing the inspection operations to be performed for all major control items. Other inspection instructions will be prepared to describe operations performed during in-process, final, packaging, and shipping inspections. Items of a specialized nature or peculiar to Advanced Syncom hardware will be given individual attention by the issuance/modification of Bulletins or Quality Control Instructions.

Fabrication operations shall be detailed in shop travelers prepared by the Planning Section of the Manufacturing organizations. In addition

6. Reliability and Quality Assurance

to the operations, machine utilization, and tooling required, quality requirements such as inspection and test points and process certifications shall be specified. Prior to issuance, Planners shall submit the travelers to the servicing Quality Control Engineering group for concurrence with the quality callouts. Though terminology for travelers may differ within the Hughes divisions, the practice of submitting these documents to Quality Control is standardized. Formal implementation occurs at Levels II and III.

- Levels of Quality Functions. To assure adequate and economical control of product quality, the functions as defined in this section shall be adhered to. Three levels are used to provide quality levels compatible with the phase of the program in terms of hardware development:

Level I Mockups, breadboard, experimental, and ground support equipment. (For the latter, training and certification and Quality Assurance audits and reports are at the request of project.)

Level II Developmental and engineering

Level III Service test, prototype, qualification, production, flight, and spares

Implementation of the Quality Control functions varies in degree between these levels and ranges from basic controls at the Level I phase to full implementation of all Hughes Quality Control policies at the Level III phase. A detailed breakdown of task versus level is shown in Table 6-2.

Level I items are those which are used to:

- 1) Prove the feasibility of a device, circuit, equipment, system, or principle

- 2) Demonstrate the technical soundness of a basic idea.

Level II items are used to:

- 1) Establish technical requirements for environmental or flight vehicles
- 2) Evaluate conformance to performance requirements of the specification
- 3) Demonstrate the reproducibility of the equipment.

Level III items are used for:

- 1) Qualification
- 2) Service test
- 3) Flight

- Inspection/Test Procedures. Inspection procedures shall be prepared by the cognizant Quality Engineering departments. Test procedures are prepared by the Quality Department for those items for which they construct test equipment (as described in work authorizations) or by Engineering (where Engineering constructs the test equipment). These documents shall be available at all applicable inspection/test stations.

- Statistical Quality Control. All electronic components shall be tested 100 percent for critical electrical parameters, as defined by Hughes specifications, either at the supplier or at Hughes. Items which have had 100 percent testing and source inspection at the supplier shall be sampled at Hughes in accordance with Mil-Std 105D, Table I, Level S4, AQL 0.4 for electrical parameters and Level S2, AQL 4.0 for visual/mechanical parameters. Items which have not been burned in or power aged at suppliers shall be 100 percent tested at Hughes for electrical parameters per specification verification and inspected for visual/mechanical parameters in accord with Table I, Level S2, AQL 4.0.

Personnel Controls

- Training and Personnel Certification.

Training of personnel associated with the Advanced Syncom program will be on a continuous basis for the duration of the contract. Training will be accomplished through periodic on-the-job instruction and formal classroom effort on workmanship requirements, NASA soldering techniques, etc. A Hughes facility which has been NASA approved and which utilizes instructors certified to the requirements of MSFC-PROC-158B shall train and certify all personnel utilized on Advanced Syncom. While certification for other special processes has been performed, new personnel assigned to these tasks shall be certified in accordance with Hughes process specifications.

- Standards. Standards of workmanship other than as defined specifically on applicable drawings and in Hughes specifications shall consist of MSFC-PROC-158B, QCI 2.1-7, and portions of the Space Projects Standard Practice Handbook as specified by project bulletins.

Supplier Controls

- Supplier Surveys. Quality Engineering will participate on all supplier survey operations. Surveys will be conducted in accordance with Aerospace Group document, Quality Capability Survey of Suppliers (Q-60-1). The majority of the surveys required for Advanced Syncom have been accomplished under similar space programs currently under way at Hughes (Syncom 2, Surveyor). Surveys are conducted on both pre- and post-award bases. Suppliers are to be resurveyed when conditions arise that may invalidate a previous survey, such as changes in management, products, processes, quality requirements, etc. Surveys are to be conducted for Level II and III hardware procurement.

- Purchase Order Screening. All purchase orders issued by Hughes will be screened by Quality Control Engineers to assure completeness of requirements in accordance with QCI 2.2-15 and QAI 30-6-8. Appropriate Purchase Order Quality attachments shall be included at this review. Screening for completeness is also performed on interdivisional procurement documents.

- Source Inspection. Source inspection at component suppliers and for all major and minor control items shall be as directed by the Syncom Project Office. The requirement for Hughes source inspection will be incorporated in the procurement document by means of the Standard Attachment, Q-1.

- Receiving Inspection. Advanced Syncom High Reliability parts specifications require extensive testing to be done at the suppliers' facilities, including prescreening, serialization, screening, X-ray, burn-in, power aging, and data submittal. Parts entering receiving inspection designated as high reliability items (by purchase order) are to be processed in the following manner:

- 1) Receiving inspectors will remove a sample quantity of pieces (see *Statistical Quality Control* above) from the lot for high reliability test. The remainder of the lot will be placed in a "hold" area within receiving inspection.
- 2) After high reliability testing is complete, the parts will be sent to receiving inspection for return to the original lot.
- 3) Receiving inspection will hold the entire lot until they have received a release notice from the Components and Materials Department.

6. Reliability and Quality Assurance

- 4) Receiving inspection will then deliver the parts to bonded stores with the release notice.

All parts delivered to Advanced Syncom high reliability purchase orders will have receiving inspection as follows:

- 1) Parts will be counted and serial numbers verified.
- 2) All data and samples required to be delivered with the parts must be verified, such as IBM cards, X-rays, and lead material samples.

All off-the-shelf types of mechanical hardware (i.e., fasteners) are to be sample inspected using Mil-Std-105D, Table I, Inspection Level II, and an AQL of 2.5.

All electrical parts not identified by high reliability purchase orders are to be 100 percent tested in receiving inspection.

- **Supplier Data.** Data is to be submitted to Hughes by all suppliers of high reliability parts. This data will be delivered after 0, 240, 750, and 1500 hours of the burn-in and power aging occurring at the suppliers. The majority of the data will be in IBM card format. Receiving inspection will forward all such data received to the Components and Materials Department. All data generated during the high reliability testing shall also be forwarded to Components and Materials. All other data submitted by suppliers as evidence of acceptability — test data, X-rays, certifications, physical/chemical reports, etc. — shall be delivered to the Components Department and made available to authorized NASA and/or Hughes personnel. Eventual filing of these documents will be in receiving inspection. The receiving inspector/tester will review all supplier data for completeness, required entries, and compliance with specification.

- **Supplier Rating.** The supplier rating system as specified by the Hughes Aerospace Group Supplier Control System shall be utilized for Advanced Syncom. When the rating indicates an undesirable trend, the servicing quality organizations will immediately notify the Advanced Syncom Product Effectiveness personnel, cognizant procurement personnel, and Hughes source inspector.

Fabrication and Assembly Controls

- **Stores and Material Control.** Bonded storerooms with limited access will be utilized. Items shall be so stored as to prevent damage and to retain traceability to the purchase document.

Kits for unit assemblies shall be prepared by bonded stores, retaining traceability by means of storeroom requisitions showing the control item and serial number of the unit in which the parts are installed. Quality organizations shall fulfill their responsibilities regarding storeroom operations to comply with Space Systems Division Procedure 9-3.5.3 which includes scheduled audits of the stores areas. In addition, kits shall be periodically audited both after preparation and in the assembly area. Raw stock shall be inspected and controlled in accordance with QCI 2.2-3, QCI 2.1-6, D.I. 38-27.

- **In-Process Inspection and Test.** Quality shall detail points of inspection, types of inspections, and tests required, and assure their incorporation in the shop travelers. The inspectors will utilize controlled drawings, specifications, workmanship standards, and models as necessary to assist in the performance of these inspection call-outs. Particular emphasis is to be given to added requirements as specified in Hughes Specification 475000-607. Implementing inspection instructions, maintained current, are to be supplied to

Hughes Aircraft Company

the inspector by Quality Engineering. History records per QCI 1.1-19 initiated in this area are to be used for data accumulation and analysis to provide an active corrective-action system. In-Process Inspection and Test is to be implemented fully on Levels II and III items, but by project request only on Level I items.

- **Process Certification.** Levels II and III hardware will be fabricated using only fully certified processes. Process certification records are maintained by Quality Engineering. Certified processes include, but are not limited to, welding, soldering, heat treat, and plating. Processing and certification will be accomplished in accordance with Hughes Process Specifications and their Quality Assurance provisions.

- **Final Inspection and Test.** Final inspection and test shall apply on all Level II and III items. In those instances involving Level I hardware deliverable from one Hughes division to another, Quality Assurance will utilize existing final inspection and test implementing documents.

- **Calibration.** The Hughes Calibration Laboratories are responsible for the mandatory periodic maintenance and calibration of all inspection, test, and laboratory equipment and instruments. Since various equipments are used at all phases of development, full implementation of the Hughes Calibration and Measurement Standards System is to be accomplished at all Levels. In addition, by means of Vendor Surveys, Source Inspection, and procurement document screening, Hughes Aircraft Company Quality Engineers will verify that vendors have:

- 1) A satisfactory calibration system
- 2) Adequate calibration procedures
- 3) An effective policing policy

- 4) Equipment that correlates with Hughes Aircraft Company and specification requirements where duplicate testing is involved.

All calibration records are to be made available to the NASA representative on request.

- **Preservation, Handling, Packaging, and Shipping.** Although full implementation of procedures governing these operations will be accomplished at Level III only, it is mandatory that Hughes personnel responsible for handling any hardware exercise good judgment and acceptable methods (reference test specifications). Hardware delivered between Divisions will be packaged to preclude damage and assure safe transit. Final inspection will treat such items as deliverable hardware.

- **Evidence of Inspection.** Hughes Quality Control shall issue, maintain records for, and assure individual traceability for all inspection stamps. Stamps are designed to identify articles that have undergone in-process, partial, or final inspection, and functional test. The article will be stamped unless it is impractical due to physical limitations or will compromise the quality of the part. When impractical, the stamps will be applied to accompanying documents or containers and will be utilized at all levels where inspection or test surveillance is required.

Product Effectiveness

- **Material Review Board.** The Advanced Syncom Material Review Board will be operated in accordance with the structure established in NPC 200-2 for all Level II and Level III items. The implementing document detailing exact functions, membership, and scope will be prepared as directed by the Project Office.

- **Failure Reporting and Corrective Action.** Servicing Quality organizations are assigned

6. Reliability and Quality Assurance

responsibility for ensuring that all quality aspects of the contract are satisfied for correcting discrepant conditions. This is accomplished by the issuance of a corrective action request to the applicable group, function, or supplier. Quality will maintain files and records of the progress and/or completion of the Corrective Action Request written for the Advanced Syncom program. These servicing organizations will assure adequate corrective action has taken place and by regular audit will review completed actions to assure that problems do not recur within the program. Implementation will be for hardware at Levels II and III.

- **Quality Assurance Audit and Reports.** Quality audits will be conducted by independent audit teams to assure conformance to contract requirements, Hughes and Space Systems Division procedures, and this program plan. Current check lists shall be upgraded and used for these audits. In addition to the audits conducted by the independent teams, audits will also be made by the Space Systems Division Product Effectiveness Staff. Results of all audits will be documented and distributed to cognizant management personnel. Reaudits will be conducted in those areas found to be deficient. Reports covering the areas audited, results of the audits, and corrective actions undertaken or completed as a

result of the audits will be submitted to the Project Office and to the NASA in accordance with Appendix B of NPC 200-2. Although the checklists will cover various aspects of the program on a 100 percent basis, it is anticipated that the audits will be conducted on a quarterly basis utilizing approximately 25 percent of the list each time. This will provide a complete coverage annually. Audits will commence the first quarter after contract go-ahead.

Government Property Controls

- **Government Furnished Articles.** Hughes receiving inspection will inspect all Government-furnished articles intended for use on the Advanced Syncom program. This inspection will be accomplished to the extent practical to detect shipping damage, and to determine that items are complete, qualified for end-use, and of proper type, size, or grade. Adequate protection, periodic inspections, and proper storage shall be provided to minimize the possibility of damage and/or deterioration. Any damage, malfunction, or condition that would render the item unsuitable for its intended use on Advanced Syncom shall be reported to the local NASA representative and to Goddard Space Flight Center.

Documents. The following list of documents shall be utilized in the performance of the Program Plan:

General

| | |
|-----------------|--|
| QCI 1.1-12 | Hughes Form 3236, Variance Authorization |
| QCI 1.1-13 | Hughes Form 672A, Substitution Authorization |
| OCI 1.1-17 | Use of Hughes Form 359, Inspection and Test Report |
| QAI 30-3-1 | Forms, Completion Instructions |
| QAI 30-3-1-778 | Request for Inspection/Test |
| QAI 30-3-1-805 | Corrective Action Request |
| QAI 30-3-1-898 | Shipping Request |
| QAI 30-3-1-1352 | Calibration Failure Report |

Hughes Aircraft Company

| | |
|--------------------------|--|
| QAI 30-3-1-2581 | Equipment Due for Calibration |
| QAI 30-3-1-3449 | Receiving Results Form |
| QAI 30-3-1-3515,
9700 | Purchase Order/Receiving Report |
| QAI 30-3-1-7159 | Inspection and Test Report |
| QAI 30-3-1-7254 | Source Field Report |
| QAI 30-3-1-7295 | Vendor History Tag |
| QAI 30-3-1-7375 | Request for Supplier Corrective Action |
| QAI 30-3-1-7550 | Vendor Permanent Record |
| QAI 30-3-1-7873 | Material Purge Request |
| QAI 30-3-1-7888 | Internal Survey Record |
| QAI 30-3-1-9727 | Internal Routing Instructions |
| QAI 30-3-1-9831 | Quality Control History Record |
| QAI 30-3-1-9831B | Inspection Status Tag |
| QAI 30-3-1-QA-38 | Tool Inspection Report |
| QAI 30-3-1-QA-44 | Quality Assurance Test Verification Report |
| QAI 30-3-1-QA-45 | DI Bonded Stores Inspection Log |
| QAI 30-3-1-QA-46 | SSD Raw Material Log |
| QAI 30-3-1-QA-47 | Procurement Review Log |
| QAI 30-3-1-QA-50 | Space Systems Quality Assurance-Defect Summary |
| QAI 30-3-1-T-2105 | Supplier Rating Record |
| QAI 30-3-2 | Quality Assurance Terminology-Definitions |
| QAI 30-3-3 | Safety Instructions |
| QAI 30-3-4 | Housekeeping |
| QAI 30-3-5 | Quality Assurance Labor Charges |
| QAI 30-3-6 | QAI Preparation, Control and Distribution |

Management

Space Systems Division Manual, Section 9, Product Effectiveness Procedures

| | |
|-----------|--------------------------------|
| QCI 4.1-1 | Quality Control Operating Plan |
| QAP 3.0 | Organization |

Design and Development Control

| | |
|-----------|--|
| EP 4-9 | Engineering Change Management — Development |
| EP 7-17-1 | Advance Syncom Project — Engineering Data |
| EP 9-1 | Design Review |
| EP 7-10-2 | Syncom Project — Engineering Change Management |

6. Reliability and Quality Assurance

Control of Contractor Procured Material

| | |
|-------------|--|
| QCI 1.1-9 | Quality Capability Surveys of Suppliers |
| QCI 1.1-14 | Preprocurement Source Selection — Major Subcontract |
| QCI 2.2-16 | Evaluation of Special Process Suppliers |
| Q-60-1 | Quality Capability Survey of Suppliers |
| QCI 2.2-15 | Quality Control Screening — Procurement Documents |
| QCI 3.13- | Advanced Syncom — Screening Purchase Orders for Subcontract Items |
| QCB A | Air Force Policy Item List |
| QCI 2.1-10 | Hughes Source Inspection — Inspection and Test Section |
| QCI 2.2-9 | Source Inspection — Supplier Evaluation Section |
| QCI 2.2-1 | Receiving Inspection Sampling Plan |
| QCI 2.2-2 | Receiving Inspection Instruction Sheet |
| QCI 2.2-3 | Raw Stock Receiving Inspection |
| QCI 2.2-6 | Quality Control Reporting — Receiving Inspection |
| QCI 2.2-7 | Receiving Inspection — General Requirements and Responsibilities |
| QCI 2.2-12 | High Reliability Receiving Inspection |
| QCI 2.2-4 | Vendor Quality Rating System |
| QAI 30-6-2 | Vendor Supplied Test Data |
| QAI 30-6-3 | Laboratory Analysis of Purchased Raw Materials or Parts, Receiving Inspection and Test |
| QAI 30-6-4 | Supplier Rating and Corrective Action System |
| QAI 30-6-5 | Color Coding of Items Requiring Special Control |
| QAI 30-6-7 | Hughes Source Inspection |
| QAI 30-6-8 | Review of Procurement Documentation |
| QAI 30-6-9 | Receiving Inspection of Space Projects Materials |
| QAI 31-18-1 | Government Source Inspection Policy Lists, Distribution and Use |
| QAI 31-6-2 | Quality Assurance Survey of Suppliers |

Control of Government-Furnished Property (GFP)

| | |
|-------------|---|
| QCI 2.2-14 | Receiving Inspection of GFP and Bailed Property |
| QCI 2.2-13 | Control of Damaged GFP — Receiving Inspection |
| QAI 30-19-1 | Control of Government Property |

Control of Contractor-Fabricated Articles

| | |
|----------|---|
| QCI 3.13 | Advanced Syncom — Screening Manufacturing Requests for Quality Control Requirements |
| QCI 3.13 | Advanced Syncom — Screening Work Orders for Quality Control Requirements |
| QCI 3.13 | Advanced Syncom — Screening Assist Work Authorizations |

Hughes Aircraft Company

| | |
|--------------|---|
| QCI A | Quality Control Instructions Initiation and Preparation |
| QCI 1.1-19 | Quality Control History Record and Inspection Status Tag |
| QCI C | Inspection and Test — General Requirements and Responsibilities |
| QCI 4.1-7 | Test Verification |
| QCI 2.1-9 | Final Inspection Of Missiles and Space Vehicles |
| QCI 3.13 | Advanced Syncom Spacecraft Final Inspection and Shipping Requirements |
| QCI 2.1-5 | Inspection of Micro-Resistance Welding |
| QCI 4.1-3 | Certification of Metal Finishing Process |
| QCI 4.1-4 | Qualification/Certification of Heat Treating |
| QCI 4.1-8 | Qualification of Micro-Resistance Welding Machines |
| QCI 4.1-9 | Welding Inspection, Fusion |
| QCI 4.1-10 | Heat Treating Inspection |
| QCI 4.1-11 | Plating Inspection |
| QCI 4.1-12 | Painting Inspection |
| QAI 30-15-3 | Weight Tolerances |
| QAI 30-16-5 | Corrosion Control |
| QAI 38-9-2-1 | Review of Syncom Planning Documentation |
| QAI 38-9-5 | Test Verification |
| QAI 38-15-1 | Drawing and Change Control |
| QAI 38-16-1 | Inspection of Thermal Paint |
| QAI 38-9-6 | Inspection and Test — Space Systems Equipment |
| QAI 38-9-6-3 | Advanced Syncom Spacecraft Final Inspection Requirements |
| QAI 38-9-6-2 | Advanced Syncom Control Item Inspection Instructions |

Nonconforming Material

| | |
|--|---|
| QCI 1.1-5 | Control of Nonconforming Supplies |
| QCI 2.2-11 | Quality Control Screening of Scrap or Salvageable Material |
| QAI 30-18-2 | Air Force Materials Review Quality Assurance Instructions (QAIs 30-18-2 through 30-18-2-18) |
| Note: Air Force MRB Quality Assurance Instructions are issued as a separate manual | |
| QAI 38-18-2 | Space Systems Division Materials Review Quality Assurance Instructions |
| QAI 38-18-2-2 | Special Project Supplement — Syncom Materials Review |

Inspection, Measuring, and Test Equipment

| | |
|------------|--|
| QCI 1.1-10 | Measurement and Test Equipment Surveys |
| QCI 2.1-8 | Tooling Inspection |
| QCI 4.1-14 | Accuracies of Test Measurement Equipment |
| QAI 30-5-1 | Control of Measurement Accuracy |

6. Reliability and Quality Assurance

| | |
|------------|---|
| QAI 30-5-2 | Control of Measuring and Test Equipment and Calibration Indication Status |
| QAI 30-5-3 | Control of Subcontractor's Measurement Equipment and Tooling |
| QAI 30-5-5 | Calibration Data System Program and IBM Listing |
| QAI 30-5-6 | Certification of Platform Scales for Outline and Mounting Weight Measurements |
| QAI 30-6-6 | Calibration and Control of Thermocouple Wire |

Inspection Stamps

| | |
|-------------|---------------|
| QAI 30-12-1 | Stamp Control |
|-------------|---------------|

Preservation, Packaging, Handling, Storage, and Shipping

| | |
|--------------|---|
| QCI 2.1-6 | Periodic Inspection of Stores |
| QCI 2.2-8 | Packaging and Shipping Inspection |
| QCI 3.13 | Advanced Syncom — Bonded Stores Inspection |
| QCI 3.13 | Advanced Syncom — Spacecraft Final Inspection and Shipping Requirements |
| QAI 38-9-1 | General Bonded Stores Procedures |
| QAI 38-3-1 | Requirements for Environmentally Controlled Areas |
| QAI 38-9-4 | Requirements for Protective Handling of Thermal Control Surfaces |
| QAI 38-10-1 | Acceptance Inspection |
| QAI 38-9-3-1 | Interdivisional Transfer of Syncom Control Items |

Statistical Quality Control

| | |
|------------|---|
| QCI 2.2-1 | Receiving Inspection Sampling Plan |
| QAI 30-8-2 | Classification of Characteristics and Defects |
| QAI 30-8-3 | Sampling Plans |

Training and Certification of Personnel

| | |
|-------------|---|
| QCI 4.1-2 | Certification of Micro Resistance Welding Operators |
| QCI 4.1-5 | Certification/Qualification of Class "A" Welding Operators |
| QCI 4.1-13 | Welding, Resistance, Inspection Qualification |
| QAI 30-16-1 | Verification and Certification of Progresses, Equipment and Personnel |
| QAI 30-16-2 | Assembler and Inspector Certification |

Data Reporting and Corrective Action

| | |
|-------------|---|
| QCI 1.1-7 | Control Point, Corrective Action Requests |
| QCI 1.1-8 | Use of Hughes Form 805, Corrective Action Request |
| QCI 1.1-3 | Quality Control Reporting — All Fabrication Areas |
| QAI 30-18-1 | Corrective Action Procedure |

Audits

| | |
|-------------|-------------------------|
| QCI 1.1-15 | Internal Quality Audits |
| QAI 30-20-1 | Area Surveys |
| QAI 38-20-1 | Area Surveys |

Advanced Syncom Failure Reporting System.
A new form for reporting of trouble and failures has been developed for use within the Aerospace Group (Figure 6-1). The form has been prepared to provide a standard system for use in the Company. Trouble and failure reports (Figure 6-2) must be prepared as soon as an abnormal condition is verified and delivered to the Syncom Project Reliability Office immediately. When a part is replaced, the old part is retained for analysis; it is placed in an envelope and attached to the Trouble and Failure Report. Particularly important is retention of the part serial number, since all parts have been extensively tested before installation and can be identified.

The Trouble and Failure Report is filled out as follows:

Block No.

- 1 Omit.
- 2-9 To be completed by originator.
- 2a Advanced Syncom.
- 2b If applicable, give the spacecraft number: F-1.
- 2c From running time meter, determine length of time unit was in the spacecraft before failure or trouble was noted.
- 2d Number of times unit had been turned on.
- 3 Self-explanatory.
- 4 Self-explanatory.
- 5 Check if a second page is used. If other TFRs have been written relative to the same subject, list the number or numbers.
- 6a,b,c Major control item name, number, and serial number.

- 7a,b,c Minor control item name, number, and serial number.
- 8a,b,c Subassembly name, number, and serial number, such as a solenoid valve.
- 9 Describe in sufficient detail symptoms of the trouble. Do not use such terms as "unit does not work"; be specific.
- 10 Omit.
- 11 To be completed by person locating the failure cause. Give repair instructions here.
- 12 To be completed by person replacing any parts. Quality Control inspection stamp is required after repair.
- 12a Circuit symbol, such as Q3 or R1.

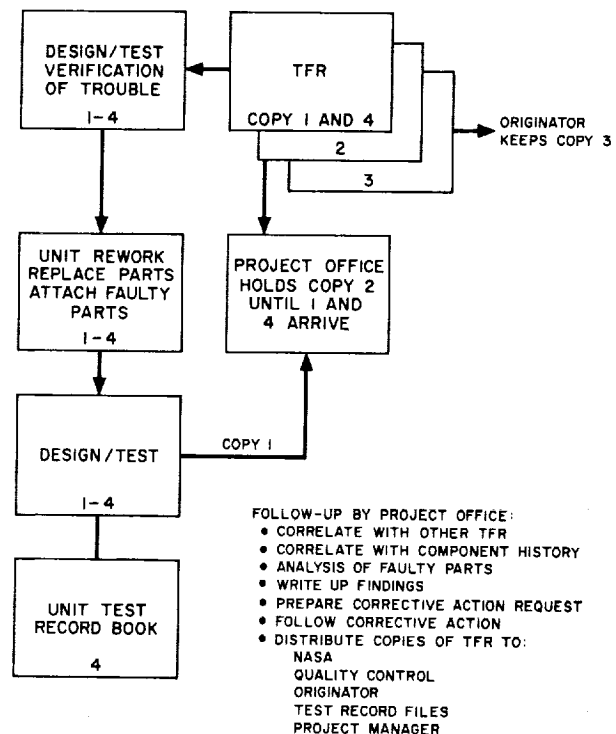


FIGURE 6-1. TROUBLE AND FAILURE REPORT FLOW DIAGRAM



TROUBLE & FAILURE REPORT No.

CONTINUATION SHEET

22 CASE NO.

[illegible]

ADDITIONAL FBI
CONTINUATION
SHEET(S) USED

[illegible]

FIGURE 6-2. TROUBLE AND FAILURE REPORT



TROUBLE & FAILURE REPORT

No. 0000000

| THIS REPORT COVERS A | | ☐ TROUBLE
☐ FAILURE | | 22 CASE NO. | | 1 LOG REP. NO. & LETTER | | | | | | | | | | | | | | |
|--|----------|--|----------|----------------------|---|-------------------------|---|---|---|---|---|---|---|-----|-----|-----|--|--|--|--|
| 10 MAINTENANCE DATA | | 2 SYSTEM NAME | | 3 REPORTING FACILITY | | 4 DATE | | | | | | | | | | | | | | |
| <table border="1"> <tr> <th>(A) UNIT</th> <th>(B) ASSY</th> <th>(C) SUB ASSY</th> </tr> <tr> <td>1</td><td>2</td><td>3</td> </tr> <tr> <td>4</td><td>5</td><td>6</td> </tr> <tr> <td>7</td><td>8</td><td>9</td> </tr> <tr> <td>10</td><td>11</td><td>12</td> </tr> <tr> <td>13</td><td>14</td><td>15</td> </tr> <tr> <td>16</td><td>17</td><td>18</td> </tr> <tr> <td>19</td><td>20</td><td>21</td> </tr> <tr> <td>22</td><td>23</td><td>24</td> </tr> <tr> <td>25</td><td>26</td><td>27</td> </tr> <tr> <td>28</td><td>29</td><td>30</td> </tr> <tr> <td>31</td><td>32</td><td>33</td> </tr> <tr> <td>34</td><td>35</td><td>36</td> </tr> <tr> <td>37</td><td>38</td><td>39</td> </tr> <tr> <td>40</td><td>41</td><td>42</td> </tr> <tr> <td>43</td><td>44</td><td>45</td> </tr> <tr> <td>46</td><td>47</td><td>48</td> </tr> <tr> <td>49</td><td>50</td><td>51</td> </tr> <tr> <td>52</td><td>53</td><td>54</td> </tr> <tr> <td>55</td><td>56</td><td>57</td> </tr> <tr> <td>58</td><td>59</td><td>60</td> </tr> <tr> <td>61</td><td>62</td><td>63</td> </tr> <tr> <td>64</td><td>65</td><td>66</td> </tr> <tr> <td>67</td><td>68</td><td>69</td> </tr> <tr> <td>70</td><td>71</td><td>72</td> </tr> <tr> <td>73</td><td>74</td><td>75</td> </tr> <tr> <td>76</td><td>77</td><td>78</td> </tr> <tr> <td>79</td><td>80</td><td>81</td> </tr> <tr> <td>82</td><td>83</td><td>84</td> </tr> <tr> <td>85</td><td>86</td><td>87</td> </tr> <tr> <td>88</td><td>89</td><td>90</td> </tr> <tr> <td>91</td><td>92</td><td>93</td> </tr> <tr> <td>94</td><td>95</td><td>96</td> </tr> <tr> <td>97</td><td>98</td><td>99</td> </tr> <tr> <td>100</td><td>101</td><td>102</td> </tr> </table> | | (A) UNIT | (B) ASSY | (C) SUB ASSY | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 10 | 11 | 12 | 13 | 14 | 15 | 16 | 17 | 18 | 19 | 20 | 21 | 22 | 23 | 24 | 25 | 26 | 27 | 28 | 29 | 30 | 31 | 32 | 33 | 34 | 35 | 36 | 37 | 38 | 39 | 40 | 41 | 42 | 43 | 44 | 45 | 46 | 47 | 48 | 49 | 50 | 51 | 52 | 53 | 54 | 55 | 56 | 57 | 58 | 59 | 60 | 61 | 62 | 63 | 64 | 65 | 66 | 67 | 68 | 69 | 70 | 71 | 72 | 73 | 74 | 75 | 76 | 77 | 78 | 79 | 80 | 81 | 82 | 83 | 84 | 85 | 86 | 87 | 88 | 89 | 90 | 91 | 92 | 93 | 94 | 95 | 96 | 97 | 98 | 99 | 100 | 101 | 102 | (a) UNIT NO. (b) ASSY NO. (c) SUB ASSY NO.
(d) UNIT NAME (e) ASSY NAME (f) SUB ASSY NAME OR SYMBOL
(g) S/N (h) REF. DESIG. OR SYMBOL (i) S/N (j) REF. DESIG. OR SYMBOL
(k) MFG (l) MFG (m) MFG
(n) (o) (p) | | 5 REFERENCES
TFR CONT. SHEET USED ☐
TFR | |
| (A) UNIT | (B) ASSY | (C) SUB ASSY | | | | | | | | | | | | | | | | | | |
| 1 | 2 | 3 | | | | | | | | | | | | | | | | | | |
| 4 | 5 | 6 | | | | | | | | | | | | | | | | | | |
| 7 | 8 | 9 | | | | | | | | | | | | | | | | | | |
| 10 | 11 | 12 | | | | | | | | | | | | | | | | | | |
| 13 | 14 | 15 | | | | | | | | | | | | | | | | | | |
| 16 | 17 | 18 | | | | | | | | | | | | | | | | | | |
| 19 | 20 | 21 | | | | | | | | | | | | | | | | | | |
| 22 | 23 | 24 | | | | | | | | | | | | | | | | | | |
| 25 | 26 | 27 | | | | | | | | | | | | | | | | | | |
| 28 | 29 | 30 | | | | | | | | | | | | | | | | | | |
| 31 | 32 | 33 | | | | | | | | | | | | | | | | | | |
| 34 | 35 | 36 | | | | | | | | | | | | | | | | | | |
| 37 | 38 | 39 | | | | | | | | | | | | | | | | | | |
| 40 | 41 | 42 | | | | | | | | | | | | | | | | | | |
| 43 | 44 | 45 | | | | | | | | | | | | | | | | | | |
| 46 | 47 | 48 | | | | | | | | | | | | | | | | | | |
| 49 | 50 | 51 | | | | | | | | | | | | | | | | | | |
| 52 | 53 | 54 | | | | | | | | | | | | | | | | | | |
| 55 | 56 | 57 | | | | | | | | | | | | | | | | | | |
| 58 | 59 | 60 | | | | | | | | | | | | | | | | | | |
| 61 | 62 | 63 | | | | | | | | | | | | | | | | | | |
| 64 | 65 | 66 | | | | | | | | | | | | | | | | | | |
| 67 | 68 | 69 | | | | | | | | | | | | | | | | | | |
| 70 | 71 | 72 | | | | | | | | | | | | | | | | | | |
| 73 | 74 | 75 | | | | | | | | | | | | | | | | | | |
| 76 | 77 | 78 | | | | | | | | | | | | | | | | | | |
| 79 | 80 | 81 | | | | | | | | | | | | | | | | | | |
| 82 | 83 | 84 | | | | | | | | | | | | | | | | | | |
| 85 | 86 | 87 | | | | | | | | | | | | | | | | | | |
| 88 | 89 | 90 | | | | | | | | | | | | | | | | | | |
| 91 | 92 | 93 | | | | | | | | | | | | | | | | | | |
| 94 | 95 | 96 | | | | | | | | | | | | | | | | | | |
| 97 | 98 | 99 | | | | | | | | | | | | | | | | | | |
| 100 | 101 | 102 | | | | | | | | | | | | | | | | | | |
| 9 DESCRIPTION OF TROUBLE | | | | | | | | | | | | | | | | | | | | |
| ROUTE TO APPLICABLE TEST AREA FOR VERIFICATION OF TROUBLE/FAILURE ☐ YES ☐ NO SIGNATURE DEPT | | | | | | | | | | | | | | | | | | | | |
| 11 TEST RESULTS FAILURE/TROUBLE VERIFIED ☐ YES ☐ NO OTHER DEFECTS NOTED ☐ YES ☐ NO | | | | | | | | | | | | | | | | | | | | |
| RECOMMENDED DISPOSITION SIGNATURE DEPT | | | | | | | | | | | | | | | | | | | | |
| 12 (A) CIRCUIT SYMBOL (B) PART NO (C) S/N (D) MFG (E) DEFECT OR REASON FOR REPLACEMENT (F) PROJECT APPROVED YES NO (G) RECEIVING LOT NUMBER (H) DO NOT USE CODE PRE FAIL | | | | | | | | | | | | | | | | | | | | |
| 13 CAUSE OF TROUBLE OR RESULTS OF MALFUNCTION ANALYSIS | | | | | | | | | | | | | | | | | | | | |
| 14 ACTION TAKEN—CONCLUSIONS | | | | | | | | | | | | | | | | | | | | |
| 15 CAUSE OF DIFFICULTY | | | | | | | | | | | | | | | | | | | | |
| 16 CORRECTIVE ACTION STATUS DATE | | | | | | | | | | | | | | | | | | | | |
| 17 PROBLEM | | | | | | | | | | | | | | | | | | | | |
| 18 EFFECT ON | | | | | | | | | | | | | | | | | | | | |
| 19 CLASS OF FAILURE | | | | | | | | | | | | | | | | | | | | |
| 20 RELATED CASES | | | | | | | | | | | | | | | | | | | | |

FIGURE 6-2 (CONTINUED). TROUBLE AND FAILURE REPORT

6. Reliability and Quality Assurance

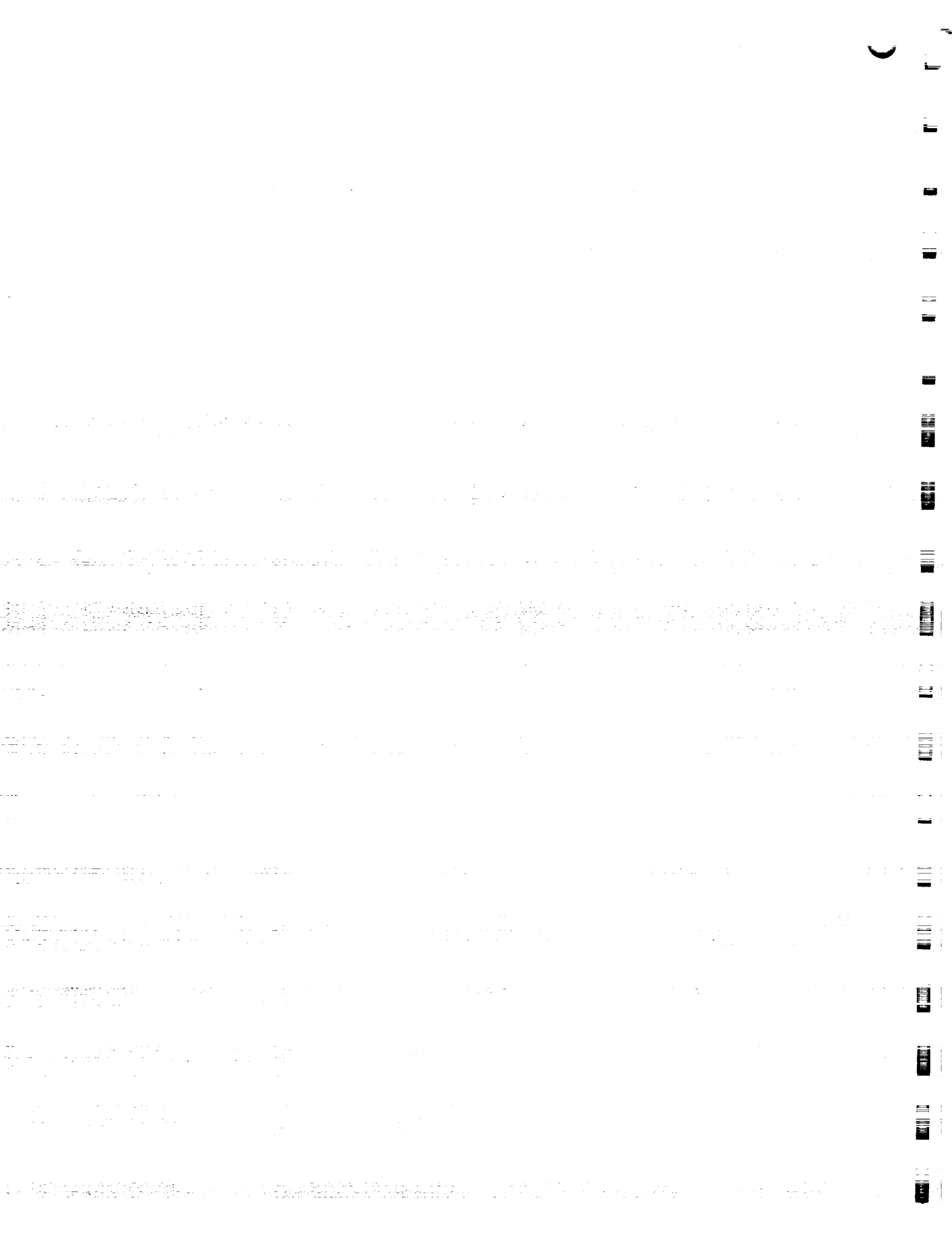
- 12b Hughes 988000-00 part number.
- 12c Serial number of new replacement part.
- 12d Manufacturer of the new part.
- 12e Omit. Will be completed in follow-up action.
- 12f Omit.
- 12g Receiving lot number. Obtained from Bonded Store Requisition.
- 13 To be completed by the cognizant design engineer after corrective action.
- 14 To be completed in follow-up.
- 15-20 To be completed by the Advanced Syncom Project Reliability Office.
- 21,22 Omit.

Distribution of copies of the Trouble and Failure Report is as follows:

- 1) Originator: Sends sheet 2 to the Advanced Syncom Project Reliability Office; inserts sheets 1 and 4 in the unit test record book and sends faulty unit and test record book

to repair area; retains copy number 3. (Copies 1 and 4 must be kept together and are used in Project Reliability Office follow-up action.)

- 2) If the TFR covers an occurrence for which no repair action is required, the originator sends copies 1, 2, and 4 directly to the Project Reliability Office.
- 3) After completion of repair, copies 1 and 4 (with notations as required in block 12) are returned to the unit test record book with the unit for retest.
- 4) The design engineer who receives the repaired unit verifies the corrective action with appropriate tests and completes block 13; he immediately sends copy 1 of the TFR to the Project Reliability Office. Copy 4 is retained in the unit test record book.



7. MATERIALS, PROCESSES, AND COMPONENTS

SPACECRAFT BILL OF MATERIALS

The latest revision (10 October 1963) of the spacecraft advanced bill of materials is given in Table 7-1.

SYSTEM TEST EQUIPMENT

Advance Bill of Materials for Synchronous Controller and Synchronous Controller Auxiliary

Special Components. The synchronous controller and synchronous controller auxiliary contain the following special components:

| | Manufacturer's Number | Quantity |
|--|---|----------|
| Transistors | 2N914 | 109 |
| | 2N1959 | 109 |
| | 2N2510 | 11 |
| | 2N708 | 10 |
| | 2N1132 | 2 |
| | 2N2185 | 4 |
| | 2N2060 | 1 |
| Diodes | 1N3064 | 361 |
| | FD300 | 28 |
| | FZ1 | 7 |
| | 1N967B | 1 |
| Capacitors | (Glass) | 180 |
| Resistors | Carbon composition, 1/4 watt, 5 percent | 756 |
| Miscellaneous (transformers, crystals, tuning forks) | | 74 |

Standard Components. The synchronous controller and synchronous controller auxiliary contain a number of standard Hughes-manufactured logic elements or cards, which were developed for a test console and are in use in several programs. They include the following:

| Card Name | Hughes Number | Quantity per Synchronous Controller | Quantity per Synchronous Controller Auxiliary |
|--------------------------|---------------|-------------------------------------|---|
| Flip-flop | 477854 | 11 | 15 |
| Logic amplifier Modified | | 7 | 10 |
| Inverter | 461303 | 2 | 1 |
| Indicator driver | 477860 | 3 | 20 |
| Clock pulse amplifier | 461305 | 4 | 3 |
| Gated pulse generator | 461359 | 4 | 1 |
| Diode gate | 461302 | 20 | 30 |

The elements listed above contain the following parts:

| | Total Quantity |
|-------------------|----------------|
| Transistors | |
| (Hughes) 928221-1 | 320 |
| (Hughes) 928222-1 | 328 |

Hughes Aircraft Company

Diodes

| | |
|-------------------|------|
| (USN) 1N3064 | 4260 |
| (Hughes) 925296-1 | 105 |
| (Hughes) 925297-1 | 52 |
| (Hughes) 925250-1 | 98 |

Resistors

| | | |
|-----------|------|-----------------------------------|
| RC07GFXXX | 2900 | 1/4 watt
carbon
composition |
| RC42GFXXX | 7 | 2 watt
carbon
composition |

Capacitors

| | | |
|------------|-----|---------------------|
| CY10CXXX | 440 | Glass |
| 910007-221 | 54 | Metallized
film |
| CY15CXXX | 8 | Glass |
| CS13AGXX | 8 | 50 volt
tantalum |

Panel Equipment. Panel equipment is as follows:

| | Quantity |
|--|----------|
| Digital readout, IEE type 120,000 | 24 |
| Rotary switches | |
| One deck nine-position Centralab 2029 | 2 |
| Two deck four-position Centralab 2011 | 1 |
| One deck six-position Centralab 2019 | 1 |
| Backlighted microswitch
(Micro Series 2C27) | 3 |
| Subminiature toggle switches
(Microswitch Series 6AT) | 9 |

| | |
|--|----|
| Helipot IRC HD-150 | 2 |
| Digiswitch (3-digit) (Series 300) | 3 |
| Textronix scopes (Model 555) | 1 |
| Frequency counter (Beckman Model 8370) | 1 |
| Meter (DCO-200 MA)
(Simpson 68F890) | 1 |
| Potentiometer Type A, 100 kilohm
(Clarostat CA1041) | 1 |
| Switch (Microswitch 6AT1) | 2 |
| Switch (Microswitch 6AT5) | 2 |
| Connector DE-9P | 1 |
| Coaxial connector (UG625B B/U) | 3 |
| Amplifier (Nexus) Type DA-1 | 2 |
| Trim potentiometer, 20 kilohm | 6 |
| Oven (Frequency Electronics Inc.
proportional control crystal)
Model FE-20-PC) | 1 |
| Card connectors 922214-1 | 93 |
| Dials (Revodox No. 412) | 2 |
| Plug-in for Textronix scope | |
| Type M | 1 |
| Type D | 1 |
| Current probe | 1 |
| Power supplies | |
| + 24 volt regulated (Transpac TR051) | 2 |
| + 28 volt unregulated
(Transpac F2B15) | 1 |

The advance bill of materials for the communications panel is given in Table 7-2.

TABLE 7-1. SPACECRAFT BILL OF

| Transistors
988 — | | Units per
Spacecraft | 813-1 | 816-1 | 817-2 | 819-1 | 821-2 | 823-1 | 824-2 | 826-1 | 828-1 | 829-1 |
|----------------------|--|-------------------------|-------|-------|-------|-------|-------|-------|-------|-------|-------|-------|
| Unit | | | | | | | | | | | | |
| 100 | Input mixer | 4 | | | | | | | | | | |
| 101 | Regulator XP, multiple
access mode | 2 | | | 4 | | | 2 | | | | |
| 102 | Regulator XP, frequency
translation mode | 2 | | | 4 | | | 2 | | | | |
| 104 | Intermediate amplifier | 2 | | | | | | | | | 8 | |
| 109 | Limiter | 2 | | | | | | | | | | |
| 110 | Preamplifier | 2 | | | | | | | | | 6 | |
| 111 | Postamplifier | 2 | | | | | | | | | | |
| 112 | High level mixer | 2 | | | | | | | | | | |
| 113 | Master oscillator, frequency
translation mode | 2 | | | | 2 | | | | 2 | 12 | |
| 114 | X32 multiplier | 6 | | | | | | | | | | |
| 116 | X3 multiplier | 4 | | | | | | | | | | |
| 117 | X2 multiplier | 4 | | | | | | | | | | |
| 118 | Filter | 4 | | | | | | | | | 4 | |
| 122 | Master oscillator,
multiple access mode | 2 | | | | | | | | | | |
| 123 | Amplifier, master oscillator | 2 | | | | 2 | | | | 2 | 4 | |
| 130 | Preamplifier | 2 | | | | | | | | | 4 | |
| 131 | Phase modulator | 2 | | | | | | | | | 2 | |
| 132 | Doubler amplifier | 2 | | | | 2 | | | | | 6 | |
| 141 | Filter amplifier | 2 | | | | | | | | | | |
| 160 | PACE regulator | 3 | | | 9 | | | | 3 | | | |
| 161 | PACE digital | 3 | | | | | | | 3 | | | |
| 162 | PACE power amplifier | 16 | | | | | | | | | | |
| 163 | PACE regulator | 8 | | | | | | | | | | |
| 172 | Telemetry monitor | 2 | | | | | | | | | | |
| 174 | Traveling-wave tube
power supply | 4 | | 12 | 4 | | | | | | | |
| 201 | Diplexer | 4 | | | | | | | | | 32 | |
| 210 | Command receiver | 4 | | | | | | | | | | |
| 211 | Command decoder | 2 | | | | | | | | | | |
| 212 | Command regulator | 4 | | | 4 | | | | | | | |
| 220 | Telemetry transmitter | 4 | 4 | | | 4 | | | | 4 | 12 | |
| 221 | Telemetry encoder | 2 | | | | | | | | | | |
| 222 | Telemetry regulator | 4 | | | 4 | | 160 | | | | | |
| 252 | Solar panel | 14 | | | | | | | | | | |
| 253 | Solar panel special | 2 | | | | | | | | | | |
| 300 | Wiring harness | 1 | | | | | | | | | | |
| 302 | Sun sensor | 4 | | | | | | | | | | |
| 303 | Central timer | 3 | | | | | | | 51 | | | |
| 306 | Squib driver | 3 | | | 24 | | | | | | | |
| 307 | Solenoid driver | 4 | | | 12 | | | | | | | |
| | Temperature sensor | 11 | | | | | | | | | | |
| 206 | Jet control electronics | 2 | | | | | | | | | | |
| | ψ_2 angle counter | 3 | | | | | | | 3 | | | |
| Total per spacecraft | | | 4 | 12 | 65 | 10 | 160 | 4 | 60 | 8 | 90 | |

[illegible]

7. Materials, Processes, and Components

MATERIALS

| 838-2 | 844-1 | 846-2 | 849-1 | 851-1 | 853-1 | 853-2 | 854-1 | 855-1 | 857-1 | 2N2415 | 861 |
|--------------|--------|-------|-------|----------|-----------|-------|----------|-----------|-------|--------|-----|
| 4
4 | 4
4 | | | 6
6 | 6
8 | | | 6 | | | 8 |
| 12 | | | | | | | | | | | 6 |
| 12
72 | 3 | | 16 | 12
16 | 12
213 | | 3
64 | 9
441 | 3 | | |
| 8 | 16 | | | | | 16 | | | | | |
| 16 | 16 | | | | 12 | | | | | | |
| 4
30
4 | | | | | 4 | | 32 | 226 | | 4 | |
| 8
16 | 8 | | | | 42
12 | | | 86
12 | | | |
| 24
12 | | | 24 | | 12
24 | | 24
16 | 12
48 | | | |
| | | | | | 12
42 | | | 24
111 | | | |
| 226 | 55 | 24 | 16 | 44 | 407 | 16 | 139 | 975 | 3 | 4 | 14 |

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Hughes Aircraft Company

TABLE 7-1. (continued)

| Diodes
988 —
Unit | | Units per
Spacecraft | 701-1 | 703-1 | 704-4 | 704-9 | 704-12 | 704-14 | 705-4 | 705-5 | 708-1 | 709-1 |
|-------------------------|--|-------------------------|-------|-------|-------|-------|--------|--------|-------|-------|-------|-------|
| 100 | Input mixer | 4 | | | | | | | | | | |
| 101 | Regulator XP, multiple
access mode | 2 | | | | | | 2 | | 2 | | |
| 102 | Regulator XP, frequency
translation mode | 2 | | | | 2 | | 2 | | 2 | | |
| 104 | Intermediate amplifier | 2 | | | | | | | | | | |
| 109 | Limiter | 2 | | | | | | | | | | |
| 110 | Preamplifier | 2 | | | 4 | | 2 | | | | | |
| 111 | Postamplifier | 2 | | | | | | | | | | |
| 112 | High level mixer | 2 | | | 2 | | | | | | | |
| 113 | Master oscillator, frequency
translation mode | 2 | | | 2 | | | | | | | |
| 114 | X32 multiplier | 6 | | | | | | | | | | |
| 116 | X3 multiplier | 4 | | | | | | | | | | |
| 117 | X2 multiplier | 4 | | | | | | | | | | |
| 118 | Filter | 4 | | | | | | | | | | |
| 122 | Master oscillator,
multiple access mode | 2 | | | | | | | | | 2 | 2 |
| 123 | Amplifier, master oscillator | 2 | | | | | | | | | | |
| 130 | Preamplifier | 2 | | 2 | | | | | | | | |
| 131 | Phase modulator | 2 | | | | | | | | | | |
| 132 | Doubler amplifier | 2 | | | | | | | | | | |
| 141 | Filter amplifier | 2 | | 2 | | | | | | | | |
| 160 | PACE regulator | 3 | | | | | | | | | | |
| 161 | PACE digital | 3 | | | | 3 | | 3 | 3 | | | |
| 162 | PACE power amplifier | 16 | | | | | | | | | | 20 |
| 163 | PACE regulator | 8 | | | | | | | | | | |
| 172 | Telemetry monitor | 2 | 64 | | | | | | | | | |
| 174 | Traveling-wave tube
power supply | 4 | 8 | | 4 | | 4 | 4 | | | | |
| 201 | Diplexer | 4 | | | | | | | | | | |
| 210 | Command receiver | 4 | | | | | | | | | | |
| 211 | Command decoder | 2 | | 4 | | | | | | | | |
| 212 | Command regulator | 4 | | | | | | | | | | |
| 220 | Telemetry transmitter | 4 | 16 | | 4 | | 4 | | | | | 10 |
| 221 | Telemetry encoder | 2 | | 8 | | | | | | | | |
| 222 | Telemetry regulator | 4 | | | | | | | | | | |
| 252 | Solar panel | 12 | 16 | | 4 | | 4 | | | | | 6 |
| 253 | Solar panel special | 4 | | | | | | | | | | |
| 300 | Wiring harness | 1 | | | | | | | | | | |
| 302 | Sun sensor | 4 | | | | | | | | | | |
| 303 | Central timer | 3 | | | | | | | | | | |
| 306 | Squib driver | 3 | | | | | | | | | | |
| 307 | Solenoid driver | 4 | 8 | | | | | | | | | |
| | Temperature sensor | 11 | | | | | | | | | | |
| 206 | Jet control electronics | 2 | | | | | | | 11 | | | |
| | ψ angle counter | 3 | | | | | | | | | | 1 |
| Total per spacecraft | | 260 | 104 | 24 | 17 | 2 | 19 | 7 | 15 | 2 | 2 | 47 |

TABLE 7-1. (continued)

| Resistors 988601- | | Units per
Spacecraft | 15 | 16 | 17 | 19 | 21 | 22 | 23 | 24 | 25 | 26 | 27 | 28 | 29 | 30 | 31 | 32 | 33 |
|----------------------|--|-------------------------|----|----|------|-----|----|----|----|----|----|----|----|----|----|----|----|----|----|
| Unit | | | | | | | | | | | | | | | | | | | |
| 100 | Input mixer | 4 | | | | | | | | | | | | | | | | | |
| 101 | Regulator XP, multiple
access mode | 2 | | | | | | | | | | | | | | | | | |
| 102 | Regulator XP, frequency
translation mode | 2 | | | | | | | | | | | | | | | | | |
| 103 | | 1 | | | | | | | | | | | | | | | | | |
| 104 | Intermediate amplifier | 2 | | | | 2 | | | | | | 6 | | | | | | | |
| 109 | Limiter | 2 | | | | | | 2 | | | | | | | | | | | |
| 110 | Preamplifier | 2 | | | | 2 | | | | | | | | | | | | | |
| 111 | Postamplifier | 2 | | | | | | | | | | 4 | | | | | | | |
| 112 | High level mixer | 2 | | | | | | | | | | 6 | | | | | | | |
| 113 | Master oscillator, frequency
translation mode | 2 | 2 | | | 2 | | | | | | 4 | | | | | 8 | 6 | |
| 114 | X32 multiplier | 6 | | | | | | | | | | | | | | | | | |
| 116 | X3 multiplier | 4 | | | | | | | | | | | | | | | | | |
| 117 | X2 multiplier | 4 | | | | | | | | | | | | | | | | | |
| 118 | Filter | 4 | | | | | | | | | | | | | | | | | |
| 122 | Master oscillator,
multiple access mode | 2 | | | | | | | | | | | | | | | | 4 | |
| 123 | Amplifier, master oscillator | 2 | 2 | | | 2 | | | | | | | | | | | 2 | | |
| 124 | Attenuator | 6 | | 6 | | | 6 | | | 6 | | 6 | | 6 | | 6 | | | |
| 130 | Preamplifier | 2 | | | | 2 | | | | | | | | | | | | | |
| 131 | Phase modulator | 2 | | | | | | | | | | | | | | | | | |
| 132 | Doubler amplifier | 2 | 6 | | | | | 4 | | | | | | | 4 | | 4 | | |
| 141 | Filter amplifier | 2 | | | | | | | 4 | | 2 | | | | | | | | |
| 152, 53, 54 | | 1 | | | | | | | | | | | | | | | | | |
| 160 | PACE regulator | 3 | | | | | | | | | | | | | | | | | |
| 161 | PACE digital | 3 | | | | | | | | | | | | | | | | | |
| 162 | PACE power amplifier | 16 | | | | | | | | | | | | | | | | | |
| 163 | PACE regulator | 8 | | | | | | | | | | | | | | | | | |
| 172 | Telemetry monitor | 2 | | | | | | | | | | | | | | | | | |
| 174 | Traveling-wave tube
power supply | 4 | 4 | | | | | | | | | | | | | | | | |
| 201 | Diplexer | 4 | | | | | | | | | | | | | | | | 4 | |
| 210 | Command receiver | 4 | | | | 4 | | | 4 | | 8 | | 4 | | | | | | |
| 211 | Command decoder | 2 | | | 922 | | | | | | | | | | | | | | |
| 212 | Command regulator | 4 | | | | | | | | | | | | | | | | | |
| 220 | Telemetry transmitter | 4 | | | | | | | | | | | | | | | | 8 | |
| 221 | Telemetry encoder | 2 | | | 226 | | | | | | | | | | | | | | |
| 222 | Telemetry regulator | 4 | | | | | | | | | | | | | | | | | |
| 252 | Solar panel | 14 | | | | | | | | | | | | | | | | | |
| 253 | Solar panel special | 2 | | | | | | | | | | | | | | | | | |
| 300 | Wiring harness | 1 | | | | | | | | | | | | | | | | | |
| 302 | Sun sensor | 4 | | | | | | | | | | | | | | | | | |
| 303 | Central timer | 3 | | | | | | | | 30 | | | | | | | | 6 | |
| 306 | Squib driver | 3 | | | | 208 | | | | | | | | | | | | | |
| 307 | Solenoid driver | 4 | | | 112 | | | | | | | | | | | | | | |
| | Jet control electronics | 2 | | | 46 | | | | | | | | | | | | | | |
| | ψ_z angle counter | 3 | | | | | | | | | | | | | | | | | |
| Total per spacecraft | | | 14 | 6 | 1518 | 10 | 6 | 6 | 4 | 38 | 8 | 26 | 4 | 6 | 4 | 8 | 12 | 28 | 16 |

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| 92 | 93 | 94 | 95 | 96 | 97 | 98 | 99 | 100 | 101 | 102 | 103 | 104 | 105 | 106 | 107 | 108 | 109 | 110 | 111 | 112 | 113 | 114 | 115 |
|----|----|----|----|--------|----|--------|----|-----|-----|-----|----------|-----|-----|-----|-----|-----|---------------|-----------|--------|-----|-----|--------|-----|
| | | | | | | 2
2 | | | | | | 2 | | | | | | | 2
2 | | | 2
2 | |
| 12 | 12 | 12 | | | | | 2 | | | | | | | | | | | | 6 | | | | |
| | | 2 | | | | | | | | | | | | | | | | | | | | | |
| 2 | | | | | | | | | | | | | | | | | | | | | | | |
| | 51 | 45 | 9 | 9
6 | 84 | 27 | 6 | 6 | 33 | 6 | 42
16 | 39 | 123 | 6 | 33 | 84 | 246 | 213
16 | 51 | 3 | 48 | 3 | 189 |
| | | 8 | | 4 | 4 | | | | | 4 | 16 | | | 4 | | | 8 | 16 | | | | 4 | |
| | | | | 4 | | | | | | | | | 4 | | 4 | | | | 4 | | | 4 | |
| 8 | 4 | | | | | | | | | | | | 4 | | | | | | | | | 4 | |
| 3 | | 6 | | 8 | | | 3 | 3 | 3 | | | 15 | 3 | 3 | | | 8 | 16 | 3 | | 6 | | |
| | | 15 | | 3 | | 6 | | | | | 39 | | 48 | | 6 | | 8
16
96 | 16
81 | 3 | | 30 | 3 | 54 |
| 25 | 67 | 88 | 9 | 34 | 88 | 37 | 19 | 9 | 36 | 10 | 113 | 56 | 178 | 13 | 52 | 84 | 374 | 342 | 71 | 3 | 84 | 22 | 243 |

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| | | | | | | | | | | | | | |
|----|----|----|----|----|----|----|----|----|----|----|----|----|--|
| 65 | 67 | 68 | 69 | 70 | 71 | 72 | 73 | 74 | 75 | 76 | 77 | 78 | |
| 4 | 2 | 2 | | | | 2 | | | 2 | | | | |
| 4 | 2 | 2 | | | | 2 | | | 2 | | | | |
| | | | | 8 | | | | | | | | | |
| | | | | 8 | | | | | | | | | |
| | | | | 4 | | 2 | | | | | | | |
| | | | 2 | 2 | | | | | 2 | | 2 | | |
| | | | | | | | | | | | | | |
| | | | | 4 | | | | | 2 | | | | |
| | 16 | | 4 | 4 | | | | | 4 | | | | |
| 3 | 3 | | | 4 | 6 | | | 2 | 2 | | | | |
| | | | | 2 | 3 | | 3 | 3 | | 3 | 3 | | |
| | | | | 4 | | | 24 | 3 | | 16 | 16 | | |
| | | 4 | | | | | | | | | 4 | | |
| | 4 | | | | 28 | | 8 | 4 | 24 | | 4 | | |
| | | 12 | | | | | 4 | | | | | | |
| 8 | | | 3 | | | | | | | | | | |
| | | | | | | | | 4 | | | | | |
| 19 | 24 | 23 | 10 | 36 | 37 | 6 | 39 | 16 | 38 | 19 | 26 | 5 | |

TABLE 7-1. (continued)

| Resistors 988601 —
Unit | | Units per
Spacecraft | 79 | 80 | 81 | 82 | 83 | 84 | 85 | 86 | 87 | 89 | 90 | 91 |
|----------------------------|--|-------------------------|----|----|----|----|----|----|----|----|----|----|----|-----|
| 100 | Input mixer | 4 | | | | | | | | | | | | |
| 101 | Regulator XP, multiple
access mode | 2 | 2 | 2 | 2 | | | 2 | 2 | | | | | |
| 102 | Regulator XP, frequency
translation mode | 2 | 2 | 2 | 2 | | | 2 | 2 | | | 2 | | |
| 103 | | 1 | | | | | | | | | | | | |
| 104 | Intermediate amplifier | 2 | | | | | | | | | | | | |
| 109 | Limiter | 2 | 2 | | | | | | | | | | | |
| 110 | Preamplifier | 2 | | | | | | | | | | | | |
| 111 | Postamplifier | 2 | | | | | | | | | | | | |
| 112 | High level mixer | 2 | | | | | | | | | | | | |
| 113 | Master oscillator, frequency
translation mode | 2 | | | | 2 | | | | | | | | |
| 114 | X32 multiplier | 6 | | | 12 | 12 | 12 | 12 | 12 | 12 | | | | |
| 116 | X3 multiplier | 4 | | | | | | | | | | | | |
| 117 | X2 multiplier | 4 | | | | | | | | | | | | |
| 118 | Filter | 4 | | | | | | | | | | | | |
| 122 | Master oscillator,
multiple access mode | 2 | | | | | 4 | | | | | | | |
| 123 | Amplifier, master oscillator | 2 | | | | | | | | | 2 | | | |
| 124 | Attenuator | 6 | | | | | | | | | | | | |
| 130 | Preamplifier | 2 | 2 | | | | | | | | 2 | | | |
| 131 | Phase modulator | 2 | 8 | | | | | | | | 4 | | | |
| 132 | Doubler amplifier | 2 | 2 | | | | | | | | | | | |
| 141 | Filter amplifier | 2 | 4 | | | | | | | | | | | |
| 152, 53, 54 | | 1 | | | | | | | | | | | | |
| 160 | PACE regulator | 3 | 3 | 3 | | | | | | | | | | |
| 161 | PACE digital | 3 | 12 | | 27 | 3 | | | 3 | 3 | 24 | 6 | | 69 |
| 162 | PACE power amplifier | 16 | | | | | | | | | | 3 | | |
| 163 | PACE regulator | 8 | | | 8 | | | | | | | | | |
| 172 | Telemetry monitor | 2 | | | | | | | | | | | | |
| 174 | Traveling-wave tube
power supply | 4 | 12 | | | | 4 | | | | 4 | | 8 | |
| 201 | Diplexer | 4 | | | | | | | | | | | | |
| 210 | Command receiver | 4 | | 4 | | | 8 | | | 4 | 32 | 4 | | |
| 211 | Command decoder | 2 | | | | | | | | | | | | |
| 212 | Command regulator | 4 | | | | | | | | | | | | |
| 220 | Telemetry transmitter | 4 | | | | | | | | | 4 | | 4 | |
| 221 | Telemetry encoder | 2 | | | | | | | | | 24 | | | |
| 222 | Telemetry regulator | 4 | 4 | | | | | 4 | 4 | | | | | 4 |
| 252 | Solar panel | | | | | | | | | | | | | |
| 253 | Solar panel special | | | | | | | | | | | | | |
| 300 | Wiring harness | | | | | | | | | | | | | |
| 302 | Sun sensor | | | | | | | | | | | | | |
| 303 | Central timer | 3 | | 6 | 3 | 3 | | | | | | 6 | 3 | |
| 306 | Squib driver | 3 | | 6 | 3 | | | | | | | | | |
| 307 | Solenoid driver | 4 | | | 4 | | | | 8 | | | | | |
| 206 | Jet control electronics | 2 | | | | | | | | | | | | |
| | ψ , angle counter | 3 | | | | | | | | | | | | 36 |
| Total per spacecraft | | | 53 | 23 | 61 | 20 | 28 | 20 | 31 | 19 | 96 | 21 | 15 | 109 |

TABLE 7-1. (continued)

| Resistors 988— | | Units per
Spacecraft | 602-32 | 602-39 | 602-46 | 602-53 | 602-55 | 602-56 | 602-60 | 602-63 | 602-66 | 602-67 | 602-68 | 602-75 | 602-77 | 602-85 | 602-91 |
|---|--|-------------------------|--------|--------|--------|--------|--------|--------|--------|--------|--------|--------|--------|--------|--------|--------|--------|
| Unit | | | | | | | | | | | | | | | | | |
| 100 Input mixer | | 4 | | | | | | | | | | | | | | | |
| 101 Regulator XP, multiple access mode | | 2 | | | | | | | 2 | | | | | | | 2 | |
| 102 Regulator XP, frequency translation mode | | 2 | | | | | | | 2 | | | | | | | 2 | |
| 103 | | 1 | | | | | | | | | | | | | | | |
| 104 Intermediate amplifier | | 2 | | | | | | | | | | | | | | | |
| 109 Limiter | | 2 | 2 | | | 2 | 2 | | | | | | | | | | |
| 110 Preamplifier | | 2 | | | | | | | | | | | | 2 | | | |
| 111 Postamplifier | | 2 | | | | | | | | | | | | | | | |
| 112 High level mixer | | 2 | | | | | | | | | | | | | | | |
| 113 Master oscillator, frequency translation mode | | 2 | | 2 | | | | | | | | | 2 | | 2 | | |
| 114 X32 multiplier | | 6 | | | | | | | | | | | | | | | |
| 116 X3 multiplier | | 4 | | | | | | | | | | | | | | | |
| 117 X2 multiplier | | 4 | | | | | | | | | | | | | | | |
| 118 Filter | | 4 | | | | | | | | | | | | | | | |
| 122 Master oscillator, multiple access mode | | 2 | | | | | | | | | | | | | | | |
| 123 Amplifier, master oscillator | | 2 | | | | | | | | | | | | | | | |
| 124 | | 6 | | | | | | | | | | | 2 | | | | |
| 130 Preamplifier | | 2 | | | | | | | | | | | | | | | |
| 131 Phase modulator | | 2 | | | | 4 | | 4 | | | | | | | | | |
| 132 Doubler amplifier | | 2 | | | | 2 | | 6 | | | | | | | | | |
| 141 Filter amplifier | | 2 | | | | | | | | | | | | | | | |
| 152, 53, 54 | | 1 | | | | | | | | | | | | | | | |
| 160 PACE regulator | | 3 | | | | | | | | | | | | | | | |
| 161 PACE digital | | 3 | | | | | | | | | | | | | | | 3 |
| 162 PACE power amplifier | | 1 | | | | | | | | 3 | 15 | 6 | | | | | |
| 163 PACE regulator | | 8 | | | | | | | | | | | | | | | |
| 172 Telemetry monitor | | 2 | | | | | | | | | | | | | | | |
| 174 Traveling-wave tube power supply | | 4 | | | | | | | | | | | | | | | 4 |
| 201 Diplexer | | 4 | | | | | | | | | | | | | | | |
| 210 Command receiver | | 4 | | | | | | | | | | | | | | | |
| 211 Command decoder | | 2 | | | | | | | | | | | | | | | |
| 212 Command regulator | | 4 | | | | | | | | | | | | | | | |
| 220 Telemetry transmitter | | 4 | 8 | | 8 | | | | | | | | | | | | |
| 221 Telemetry encoder | | 2 | | | | | | | | | | | | | | | |
| 222 Telemetry regulator | | 4 | | | | | | | | | | | | | | | |
| 252 Solar panel | | | | | | | | | | | | | | | | | |
| 253 Solar panel special | | | | | | | | | | | | | | | | | |
| 300 Wiring harness | | | | | | | | | | | | | | | | | |
| 302 Sun sensor | | | | | | | | | | | | | | | | | |
| 303 Central timer | | 3 | | | 18 | | | | | | | | | | | | |
| 306 Squib driver | | 4 | | | | | | | 3 | | | | | | | | |
| 307 Solenoid driver | | 4 | | | 24 | | | | | | | | | | | | |
| Jet control electronics | | 2 | | | | | | | | | | | | | | | |
| ψ_z angle counter | | 3 | | | | | | | | | | | | | | | |
| Total per spacecraft | | | 10 | 2 | 50 | 8 | 2 | 10 | 4 | 6 | 15 | 6 | 4 | 2 | 2 | 4 | 7 |

| 88613- | | 988624- | | | 988625- | | | 988630 | | 988631- | 988611- | | | | | |
|--------|-----|---------|----|--------|---------|----|----|--------|-----|---------|---------|-----|-----|-----|-----|-----|
| 401 | 472 | 17 | 20 | 24 | -9 | 11 | 37 | -49 | -18 | -93 | 157 | 155 | 101 | 307 | 354 | 292 |
| 3 | 3 | | | 16 | | 3 | | 16 | | | | | | | | 2 |
| | | | | 4 | | | 4 | | 4 | | | | | | | |
| | | | | 1
4 | | | | | 4 | | | | | | | |
| | | | | | | | | | | | | 3 | 3 | 3 | 3 | |
| 3 | 3 | | | | | | | | | | | | | | | |
| 46 | 6 | 4 | 16 | 5 | 3 | 4 | 4 | 16 | 4 | 3 | 3 | 3 | 3 | 3 | 2 | 3 |

988613 — 69

2

Hughes Aircraft Company

TABLE 7-1. (continued)

| Resistors | | Units per
Spacecraft | 988610- | | | | | | | | 988623- | | | | | | | | 9 |
|----------------------|--|-------------------------|---------|-----|-----|-----|-----|-----|-----|-----|---------|-----|-----|----|----|-----|-----|-----|----|
| | | | 197 | 240 | 289 | 301 | 326 | 327 | 359 | 365 | 215 | 275 | 371 | 39 | 68 | 259 | 260 | 319 | |
| 100 | Input mixer | 4 | | | | | | | | | | | | | | | | | |
| 101 | Regulator XP, multiple
access mode | 2 | | 2 | 2 | 4 | 2 | | | | | | | | | | | | |
| 102 | Regulator XP, frequency
translation mode | 2 | | | 2 | | | 2 | | | | | | | | | | | |
| 104 | Intermediate amplifier | 2 | | | | | | | | | | | | | | | | | |
| 109 | Limiter | 2 | | | | | | | | | | | | | | | | | |
| 110 | Preamplifier | 2 | | | | | | | | | | | | | | | | | |
| 111 | Postamplifier | 2 | | | | | | | | | | | | | | | | | |
| 112 | High level mixer | 2 | | | | | | | | | | | | | | | | | |
| 113 | Master oscillator, frequency
translation mode | 2 | | | | | | | | | | | | | | | | | |
| 114 | X32 multiplier | 6 | | | | | | | | | | | | | | | | | |
| 116 | X3 multiplier | 4 | | | | | | | | | | | | | | | | | |
| 117 | X2 multiplier | 4 | | | | | | | | | | | | | | | | | |
| 118 | Filter | 4 | | | | | | | | | | | | | | | | | |
| 122 | Master oscillator,
multiple access mode | 2 | 2 | | | | | | | | | | | | | | | | |
| 123 | Amplifier, master oscillator | 2 | | | | | | | | | | | | | | | | | |
| 130 | Preamplifier | 2 | | | | | | | | | | | | | | | | | |
| 131 | Phase modulator | 2 | | | | | | | | | | | | | | | | | |
| 132 | Doubler amplifier | 2 | | | | | | | | | | | | | | | | | |
| 141 | Filter amplifier | 2 | | | | | | | | | | | | | | | | | |
| 160 | PACE regulator | 3 | | | 3 | | | 3 | | | | | | | | | | | |
| 161 | PACE digital* | 3 | | | 30 | | | | | 60 | | | 3 | | | | | | 6 |
| 162 | PACE power amplifier | 16 | | | | | | | 16 | 16 | | | | | | 128 | 128 | 256 | |
| 163 | PACE regulator | 8 | | | | | | | | | | | | | | | | | |
| 172 | Telemetry monitor | 2 | | | | | | | | | | | | | | | | | |
| 174 | Traveling-wave tube
power supply | 4 | | | 4 | | | | | | | | | | | | | | |
| 201 | Diplexer | 4 | | | | | | | | | | | | | | | | | |
| 210 | Command receiver | 4 | | | | | | | | | | | | | | | | | |
| 211 | Command decoder | 2 | | | | | | | | | | | | | | | | | |
| 212 | Command regulator | 4 | | | 4 | | | 4 | | | | | | | | | | | |
| 220 | Telemetry transmitter | 4 | | | | | | | | | | | | | | | | | |
| 221 | Telemetry encoder | 2 | | | | | | | | | | | | | | | | | |
| 222 | Telemetry regulator | 4 | | | 4 | | | 4 | | | | | | | | | | | |
| 252 | Solar panel | 14 | | | | | | | | | | | | | | | | | |
| 253 | Solar panel special | 2 | | | | | | | | | | | | | | | | | |
| 300 | Wiring harness | 1 | | | | | | | | | | | | | | | | | |
| 302 | Sun sensor | 4 | | | | | | | | | | | | | | | | | |
| 303 | Central timer | 3 | | | | | | | | | | | | | | | | | |
| 306 | Squib driver | 3 | | | | | | | | | | | | | | | | | |
| 307 | Solenoid driver | 4 | | | | | | | | | | | | | | | | | |
| | Temperature sensor | 11 | | | | | | | | | | | | | | | | | |
| | Jet control electronics | 2 | | | | | | | | | | | | | | | | | |
| | ψ angle counter | 3 | | | | | | | | | 3 | 3 | 3 | 12 | 24 | | | | 6 |
| Total per spacecraft | | | 2 | 2 | 49 | 4 | 2 | 13 | 16 | 76 | 3 | 3 | 6 | 12 | 24 | 128 | 128 | 256 | 12 |

*In addition to the parts listed the following parts in various values are purchased in the quantities shown for selection purposes: 988610-

988502-

| 37 | 39 | 42 | 43 | 46 | 47 | 48 | 50 | 51 | 57 | 58 | 59 | 60 | 61 | 64 | 65 | 66 | 67 | 83 | 139 | 155 | 31 | 75 | 72 | 53 | 23 |
|----|----|----|----|----|----|----|----|----|----|----|----|----|-----|----|----|----|----|----|-----|-----|----|----|----|----|----|
| 4 | 2 | 2 | 2 | 2 | | 4 | | | | 2 | | | 267 | 3 | 16 | 3 | 6 | 3 | 6 | 3 | 4 | | | 3 | |
| | | | | | | 6 | | 3 | | | | | | | 8 | | 16 | | | | | | | | |
| | | | | | | | 24 | | | | | 4 | | | | | 4 | | | | | | | | |
| | | | | | | | | | | | 8 | | | | | | 8 | | | | | | | | |
| | | | | | | | | | | | 3 | | | | | | 4 | | | | | | | | |
| | | | | | | | | | | | | | 10 | | | | | | | | | | | | 3 |
| 4 | 26 | 4 | 22 | 2 | 3 | 10 | 27 | 3 | 2 | 2 | 11 | 4 | 277 | 3 | 24 | 3 | 89 | 3 | 6 | 3 | 4 | | | 3 | 3 |

2

7. Materials, Processes, and Components

TABLE 7-1. (continued)

| Capacitors | | 988500- | | | | | | | | | | | | | | | | |
|--|-------------------------|---------|----|-----|----|----|----|----|----|----|-----|-----|-----|-----|-----|-----|-----|-----|
| Unit | Units per
Spacecraft | 16 | 17 | 48 | 52 | 55 | 56 | 59 | 63 | 72 | 136 | 137 | 141 | 151 | 154 | 166 | 185 | 200 |
| 100 Input mixer | 4 | | | | | | | | | | | | | | | | | |
| 101 Regulator XP, multiple
access mode | 2 | | | | | 4 | | | | | | | | | | | | |
| 102 Regulator XP, frequency
translation mode | 2 | | | | | 4 | | | | | | | | | | | | |
| 104 Intermediate amplifier | 2 | | | | | | | | | | | | | | | | | |
| 109 Limiter | 2 | | | | | | | | | | | | | | | | | |
| 110 Preamplifier | 2 | | | | | | | | | | | | | | | | | |
| 111 Postamplifier | 2 | | | | | | | | | | | | | | | | | |
| 112 High level mixer | 2 | | | | | | | | | | | | | | | | | |
| 113 Master oscillator, frequency
translation mode | 2 | | | | | | | | 2 | 2 | | | | | | | | |
| 114 X32 multiplier | 6 | | | 6 | 6 | | | | | | | | | | | | | |
| 116 X3 multiplier | 4 | | | | | | | | | | | | | | | | | |
| 117 X2 multiplier | 4 | | | | | | | | | | | | | | | | | |
| 118 Filter | 4 | | | | | | | | | | | | | | | | | |
| 122 Master oscillator,
multiple access mode | 2 | | | | 4 | | | | | | | | | | | | | |
| 123 Amplifier, master oscillator | 2 | | | | | | | | | 2 | | | | | | | | |
| 130 Preamplifier | 2 | | | | | | | | | 2 | | | | | | | | |
| 131 Phase modulator | 2 | | | | | | | | | 2 | | | | | | | | |
| 132 Doubler amplifier | 2 | | | | | | | | | | | | | | | | | |
| 141 Filter amplifier | 2 | | | | | | | | | | | | | | | | | |
| 160 PACE regulator | 3 | | | | 3 | 12 | | | | | | | | | | | | |
| 161 PACE digital | 3 | | | | | | | | | | 15 | 3 | 9 | 6 | 3 | 9 | | 9 |
| 162 PACE power amplifier | 16 | | | | | | | | | | | | | | | | | |
| 163 PACE regulator | 8 | | | | | | | | | | | | | | | | | |
| 172 Telemetry monitor | 2 | | | | | | | | | | | | | | | | | |
| 174 Traveling-wave tube
power supply | 4 | 4 | | | 4 | 4 | 8 | | | | | | | | | | 4 | |
| 201 Diplexer | 4 | | | | | | | | | | | | | | | | | |
| 210 Command receiver | 4 | | | | | 4 | | | | 4 | | | | | | | | |
| 211 Command decoder | 2 | | | | | | | | | | | | | | | | | |
| 212 Command regulator | 4 | | | | | 8 | | | | | | | | | | | | |
| 220 Telemetry transmitter | 4 | | 4 | | | | | | | 4 | | | | | | | | |
| 221 Telemetry encoder | 2 | | | | | | | | | | | | | | | | | |
| 222 Telemetry regulator | 4 | | | 16 | 4 | 4 | | | | | | | | | | | | |
| 252 Solar panel | 14 | | | | | | | | | | | | | | | | | |
| 253 Solar panel special | 2 | | | | | | | | | | | | | | | | | |
| 300 Wiring harness | 1 | | | | | | | | | | | | | | | | | |
| 302 Sun sensor | 4 | | | | | | | | | | | | | | | | | |
| 303 Central timer | 3 | | | 33 | 6 | 3 | | 3 | 3 | | | | | | | | | |
| 306 Squib driver | 3 | | | 72 | | | | | | | | | | | | | | |
| 307 Solenoid driver | 4 | | | | | | | | | | | | | | | | | |
| Temperature sensor | 11 | | | | | | | | | | | | | | | | | |
| Jet control electronics | 2 | | | | | | | | | | | | | | | | | |
| ψ , angle counter | 3 | | | | | | | | | | | | | | 3 | 6 | | |
| Total per spacecraft | | 4 | 4 | 127 | 27 | 43 | 8 | 3 | 5 | 14 | 15 | 3 | 9 | 6 | 6 | 15 | 4 | 9 |

Hughes Aircraft Company

TABLE 7-1. (continued)

| Capacitors

Unit | | Units per
Spacecraft | 988503- | | | | | | | | | | | | | | |
|------------------------|--|-------------------------|---------|---|----|----|----|----|----|----|----|----|----|----|----|----|----|
| | | | 4 | 7 | 10 | 12 | 14 | 17 | 19 | 23 | 25 | 26 | 27 | 28 | 32 | 35 | 36 |
| 100 | Input mixer | 4 | | | | | | | | | | | | | | | |
| 101 | Regulator XP, multiple
access mode | 2 | | | | | | | | | | | | | | | |
| 102 | Regulator XP, frequency
translation mode | 2 | | | | | | | | | | | | | | | |
| 104 | Intermediate amplifier | 2 | | | | | | | | | | | 2 | | | | |
| 109 | Limiter | 2 | | | | | | | | | | | | | | | |
| 110 | Preamplifier | 2 | | | | | | | | | | | | | | | |
| 111 | Postamplifier | 2 | | | | | | | | | | | | | | | |
| 112 | High level mixer | 2 | | | | | | | | | | | | | | | |
| 113 | Master oscillator, frequency
translation mode | 2 | 2 | | 2 | | | | 4 | | | | | | | | |
| 114 | X32 multiplier | 6 | | | | | | | | | | | | | | | |
| 116 | X3 multiplier | 4 | | | | | | | | | | | | | | | |
| 117 | X2 multiplier | 4 | | | | | | | | | | | | | | | |
| 118 | Filter | 4 | | | | | | | | | | | | | | | |
| 122 | Master oscillator,
multiple access mode | 2 | | | | | | | | | | | | | | | |
| 123 | Amplifier, master oscillator | 2 | | | | | | 2 | 2 | | | | | | | | |
| 130 | Preamplifier | 2 | | | | | | | | | | | | | | | |
| 131 | Phase modulator | 2 | | | | | | | | | | | | | | | |
| 132 | Doubler amplifier | 2 | | | | | | | | | | | | | | | |
| 141 | Filter amplifier | 2 | | | | | | | | | | | | | | | |
| 160 | PACE regulator | 3 | | | | | | | | | | | | | | | |
| 161 | PACE digital | 3 | | | | | | | | | | | | | | | |
| 162 | PACE power amplifier | 16 | | | | | | | | | | | | | | | |
| 163 | PACE regulator | 8 | | | | | | | | | | | | | | | |
| 172 | Telemetry monitor | 2 | | | | | | | | | | | | | | | |
| 174 | Traveling-wave tube
power supply | 4 | | | | | | | | | | | | | | | |
| 201 | Diplexer | 4 | | 8 | | | | | | | | | | | | | |
| 210 | Command receiver | 4 | | | | 8 | | | | | | | | | | | |
| 211 | Command decoder | 2 | | | | | | | | | | | | | | | |
| 212 | Command regulator | 4 | | | | | | | | | | | | | | | |
| 220 | Telemetry transmitter | 4 | | | | 4 | 4 | | | | | | | | | | |
| 221 | Telemetry encoder | 2 | | | | | | | | | | | | | | | |
| 222 | Telemetry regulator | 4 | | | | | | | | | | | | | | | |
| 252 | Solar panel | 14 | | | | | | | | | | | | | | | |
| 253 | Solar panel special | 2 | | | | | | | | | | | | | | | |
| 300 | Wiring harness | 1 | | | | | | | | | | | | | | | |
| 302 | Sun sensor | 4 | | | | | | | | | | | | | | | |
| 303 | Central timer | 3 | | | | | | | | | | | | | | | |
| 306 | Squib driver | 3 | | | | | | | | | | | | | | | |
| 307 | Solenoid driver | 4 | | | | | | | | | | | | | | | |
| | Temperature sensor | 11 | | | | | | | | | | | | | | | |
| 206 | Jet control electronics | 2 | | | | | | | | | | | | | | | |
| | ψ_z angle counter | 3 | | | | | | | | | | | | | | | |
| Total per spacecraft | | | 2 | 8 | 2 | 12 | 4 | 2 | 36 | 14 | 6 | 22 | 16 | 6 | 6 | 8 | |

| | | | | | | | | | 988501- | | | | | | | | | | | | | | |
|---|-----|-----|-----|-----|----|----|-----|-----|---------|---|---|----|----|-----|-----|-----|-----|-----|-----|----|-----|-----|-----|
| 3 | 205 | 300 | 382 | 378 | 85 | 67 | 348 | 498 | 1 | 7 | 9 | 42 | 44 | 125 | 127 | 161 | 181 | 259 | 273 | 38 | 271 | 193 | 195 |
| | 2 | | 12 | 12 | | | | | | | | | | | | | | | | | | | |
| | 3 | | | | 4 | | | | 9 | | | | | | | 6 | 6 | | | | | 6 | 3 |
| | | | | | | | | 4 | 8 | 4 | 8 | 4 | | | | | | 16 | | 8 | 8 | | |
| | | | | | | | | | 12
4 | | | | | | | | | | | | | | |
| | | | | | | | 4 | | | | | | | | | | | | | | | | |
| | | | | | | 3 | | | | | | | | | | | | | | | | | |
| 2 | 3 | 12 | 12 | 4 | | 3 | 4 | 4 | 33 | 4 | 8 | 4 | | | 6 | 6 | 16 | | 8 | 8 | 6 | 3 | |

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TABLE 7-1. (co

| | | Unit | T50-2 |
|--------------|------------------------------|------|-------|
| 100 | Input mixer | | |
| 101 | Regulator XPR
access mo | | |
| 102 | Regulator XPR
translation | | |
| 104 | Intermediate | | |
| 109 | Limiter | | 2 |
| 110 | Preamplifier | | 2 |
| 111 | Postamplifier | | 2 |
| 112 | High level m | | 2 |
| 113 | Master oscil
translation | | |
| 114 | X32 multipli | | |
| 116 | X3 multiplie | | |
| 117 | X2 multiplie | | |
| 118 | Filter | | |
| 122 | Master oscil
multiple a | | |
| 123 | Amplifier, m | | |
| 130 | Preamplifier | | |
| 131 | Phase modul | | |
| 132 | Doubler amp | | |
| 141 | Filter amplif | | |
| 160 | PACE regula | | |
| 161 | PACE digital | | |
| 162 | PACE power | | |
| 163 | PACE regula | | |
| 172 | Telemetry m | | |
| 174 | Traveling-wa
power sup | | |
| 201 | Diplexer | | |
| 210 | Command re | | |
| 211 | Command de | | |
| 212 | Command re | | |
| 220 | Telemetry tr | | |
| 221 | Telemetry e | | |
| 222 | Telemetry re | | |
| 252 | Solar panel | | |
| 253 | Solar panel | | |
| 300 | Wiring harn | | |
| 302 | Sun sensor | | |
| 303 | Central time | | |
| 306 | Squib driver | | |
| 307 | Solenoid dri | | |
| | Temperature | | |
| | Jet control | | |
| | ψ angle co | | |
| Total per sp | | | 6 |

| Connectors
Omni Spectra | | | | | | | Cannon | | | | | Crystals
Bliley | | | | Terminals | | | | | | | | | |
|----------------------------|-------|-----|-------------|------|------|---------|---------------------------------|---------|----|----|----|--------------------|----------|----------|----------|-----------|----------|----------|----------|----------|----------|----------|----------|----------|-----------|
| 321-1 | 571-1 | 204 | 1731 | 1773 | 1775 | 1803 | 1781 | 988201- | | | | | SP 31667 | SP 31669 | SP 31660 | SP 31668 | 978305-3 | 978012-1 | 978323-1 | 988134-1 | 988147-2 | 988151-2 | 978012-4 | 978207-2 | 978105-55 |
| | | | | | | | | -5 | -1 | -2 | -3 | -10 | | | | | | | | | | | | | |
| 8 | 4 | 4 | 4
8
8 | 2 | | 8 | 4
4
4
2
4 | | | | | 2 | 1 | 2 | 1 | 1340 | 10
10 | 4 | | 4 | 12 | 2 | | 4 | 2 |
| | | | | 12 | | 12 | 2
6
4
8
4
4
6 | 24 | | | 4 | 6 | | | | | 8 | | | | | | 28 | | |
| | | | | | | 12
4 | 4
8 | | 4 | | | | | | | | 16 | | | | | | | | |
| | | | | | | 12 | | | | | | | | | | | | | | | | | | | |
| 3 | 4 | 4 | 20 | 14 | | 52 | 60 | 24 | 4 | 4 | 4 | 15 | 2 | 1 | 2 | 1 | 1340 | 44 | 4 | 4 | 12 | 2 | 28 | 4 | 2 |

2

Hughes Aircraft Company

TABLE 7-1. (continued)

| Miscellaneous

Unit | | Spacecraft
Units per | RF Chokes
Delevan | | | | | | | | | | | | | Cores
Micrometals | | | | | | |
|---------------------------|-------------|-------------------------|----------------------|---|---|---|----|----|----|----|----|----|----|----|----|----------------------|-------|-------|-------|-------|-------|----|
| | | | 988670- | | | | | | | | | | | | | T25-4 | T25-6 | T31-4 | T37-6 | T50-3 | T37-4 | |
| | | | 1 | 2 | 3 | 5 | 10 | 12 | 13 | 16 | 18 | 20 | 21 | 22 | 23 | | | | | | | |
| | | | | | | | | | | | | | | | | | | | | | | |
| 100 | Input mixer | 4 | | | | | | | | | | | | | | | | | | | | </ |

7. Materials, Processes, and Components

| Units per
Spacecraft | 988504- | | | | 988505- | 988520- | | 988526- | 988541 | | | 988521 | 988525 |
|-------------------------|---------|----|----|-----|---------|---------|----|---------|--------|---|-----|--------|--------|
| | 5 | 12 | 17 | 28 | -351 | -1 | -2 | -1 | 6 | 8 | 175 | | -14 |
| 4 | | | | | | | | | | | | | |
| 2 | | | 2 | 4 | | | | | | | | | |
| 2 | | | 2 | 2 | | | | | | | | | |
| 2 | | | | 2 | | | | | | | | | |
| 2 | | | | 24 | | | | 20 | | | | 6 | |
| 2 | | | | 2 | | | | 22 | | | | 6 | |
| 2 | | | | 2 | | | | 20 | | | | 6 | |
| 2 | | | | | | | | 16 | | | | 6 | |
| 2 | | | 40 | | | | | | | | | | |
| 6 | | | | | | 14 | 4 | 32 | 2 | 4 | 2 | | |
| 4 | | | | | | 78 | | | | | | | |
| 4 | | | | | | | | | | | | | |
| 4 | | | | | | | | | | | | | |
| 2 | | | | 16 | | | | | | | | | |
| 2 | | | | 18 | | 6 | | 10 | | | | | |
| 2 | 2 | 16 | | | | 6 | | | | 2 | | | 2 |
| 2 | | | | 26 | | 6 | | 16 | | | | | |
| 2 | | | | 22 | | 6 | | 6 | | | | | |
| 2 | 20 | | | | | 6 | 4 | 20 | | | | | |
| 3 | | | 3 | | | 6 | | 22 | | | | | |
| 3 | | | | 3 | | 8 | 4 | 16 | | | | | |
| 16 | | | | | 6 | | | | | | | | |
| 8 | | | | | | | | | | | | | |
| 2 | | | | | | | | | | | | | |
| 4 | | | 4 | 4 | | | | | | | | | |
| 4 | | | | | | | | | | | | | |
| 4 | | | | 52 | | 12 | 16 | 4 | | | | | |
| 2 | | | | | | 44 | | 100 | | | | | |
| 4 | | | 4 | 4 | | | | | | | | | |
| 4 | 52 | | | | | | | | | | | | |
| 2 | | | | | | 48 | 12 | 36 | | | | | |
| 4 | | | 4 | 4 | | | | | | | | | |
| 14 | | | | | | | | | | | | | |
| 2 | | | | | | | | | | | | | |
| 1 | | | | | | | | | | | | | |
| 4 | | | | | | | | | | | | | |
| 3 | | | | | | | | | | | | | |
| 3 | | | | | | | | | | | | | |
| 4 | | | | | | | | | | | | | |
| 11 | | | | | | | | | | | | | |
| 2 | | | | | | | | | | | | | |
| 3 | | | | | | | | | | | | | |
| | 2 | 88 | 19 | 225 | 6 | 234 | 40 | 340 | 2 | 6 | 2 | 18 | 2 |

2

TABLE 7-2. COMMUNICATIONS PANEL ADVANCE BILL OF MATERIALS

| Part | Value | Note | Quantity | Part | Value | Note | Quantity |
|----------------------------|----------------|-------------------------------------|----------|-----------------------------------|------------------------|-------------------------------------|----------|
| HIGH PASS FILTER — X479670 | | | | Resistor | 4.7 kilohms | ¼ watt carbon composition | 1 |
| Capacitor | 21 pf | | 2 | Resistor | 9.1 kilohms | ¼ watt carbon composition | 1 |
| Capacitor | 71 pf | | 2 | Resistor | 16 ohms | ¼ watt carbon composition | 2 |
| Capacitor | 20 pf | | 2 | Resistor | 220 ohms | ¼ watt carbon composition | 2 |
| Capacitor | 9 pf | | 2 | Resistor | 5.6 kilohms | ¼ watt carbon composition | 1 |
| Capacitor | 46 pf | | 2 | Resistor | 270 ohms | ¼ watt carbon composition | 1 |
| Capacitor | 5.5 pf | | 2 | Resistor | 15 kilohms | ¼ watt carbon composition | 1 |
| Capacitor | 16 pf | | 2 | Resistor | 750 ohms | ¼ watt carbon composition | 2 |
| Capacitor | 41 pf | | 2 | Transformer | | Hughes special 2N1141 | 4 |
| Inductor | | Special | 8 | Transistor | | | 6 |
| Resistor | 27 ohms | ½ watt carbon composition | 2 | LOW VOLTAGE POWER SUPPLY — 479673 | | | |
| Resistor | 36 ohms | ½ watt carbon composition | 1 | Capacitor | 2000 μ f | 50 volts | 3 |
| Connector | | BNC | 2 | Capacitor | 0.01 μ f | 150 volts | 4 |
| Q PHASE DETECTOR — X479671 | | | | Capacitor | 15 μ f | 50 volts | 1 |
| Capacitor | 36 pf | | 2 | Diode | | 1N540 | 8 |
| Capacitor | 8.2 pf | | 1 | Diode, zener | | 1N758A | 4 |
| Capacitor | 6.8 pf | | 1 | Diode | | 1N485 | 2 |
| Diode | | HD 5000 | 4 | Diode | | 1N965B | 2 |
| Connector | | Multiple-pin | 1 | Connector | | | 1 |
| Connector | | BNC | 2 | Transmitter | | 2N1649 | 2 |
| Resistor | 510 ohms | ¼ watt 5 percent carbon composition | 2 | Transmitter | | 2N722 | 4 |
| Resistor | 10 kilohms | ¼ watt 5 percent carbon composition | 2 | Transmitter | | 2N174 | 1 |
| Resistor | 100 kilohms | ¼ watt 5 percent carbon composition | 2 | Resistor | 2 kilohms | 1 watt 5 percent carbon composition | 2 |
| Transformer | | Hughes special | 3 | Resistor | 2.2 kilohms | ½ watt 5 percent carbon composition | 2 |
| DRIVER AMPLIFIER — X479672 | | | | Resistor | 1 kilohm | ½ watt 5 percent carbon composition | 2 |
| Capacitor | 0.005 μ fd | Sprague THD-50 | 24 | Resistor | 2.5 kilohms | ½ watt variable | 4 |
| Capacitor | 47 pf | Arco-Elmenco DM10F-470K | 1 | Resistor | 10 kilohms | ¼ watt 5 percent carbon composition | 6 |
| Connector | UG625 B/U | | 3 | Resistor | 5.6 kilohms | ½ watt 5 percent carbon composition | 2 |
| Connector | | Multiple-pin | 1 | Resistor | 20 kilohms | ½ watt 5 percent carbon composition | 2 |
| Diode | | Hughes 1N1871E | 1 | Resistor | 1.5 kilohms | ½ watt 5 percent carbon composition | 2 |
| Inductor | 10 μ h | Delevan 1025-44 | 5 | Resistor | 3 kilohms | ½ watt 5 percent carbon composition | 2 |
| Inductor | 5.6 μ h | Delevan 1025-38 | 1 | Transformer | 115/26 volts, 1 ampere | F-40-X power | 2 |
| Inductor | 0.435 μ h | Hughes special | 1 | | | | |
| Meter | | | 1 | | | | |
| Resistor | 56 ohms | ¼ watt carbon composition | 2 | | | | |
| Resistor | 3.3 kilohms | ¼ watt carbon composition | 2 | | | | |
| Resistor | 1.2 kilohms | ¼ watt carbon composition | 1 | | | | |
| Resistor | 430 ohms | ¼ watt carbon composition | 1 | | | | |

TABLE 7-2 (continued)

| Part | Value | Note | Quantity |
|---|-----------------------------|---|----------|
| 250 VOLT POWER SUPPLY — X479674 | | | |
| Capacitor | 8 μ f,
450 volts, dc | | 1 |
| Choke | 14 henrys | Thermador
#7A1414 | 1 |
| Resistor | 250 kilohms | 10 watt W. W. | 1 |
| Transformer | | Thermador
#5A6146 | 1 |
| Vacuum tube | | 5U4 | 1 |
| Panel lamp
assembly | | GE47 | 1 |
| I PHASE DETECTOR AND ISO. AMPLIFIER — X479675 | | | |
| Capacitor | Select | | 9 |
| Capacitor | 22 μ f | | 2 |
| Capacitor | 510 pf | | 1 |
| Capacitor | 3.3 μ f,
35 volts dc | Tantalum | 2 |
| Capacitor | 0.01 μ f | | 1 |
| Diode | | | 4 |
| Connector | | Multiple-pin | 1 |
| Connector | | Coaxial | 5 |
| Inductor | 0.012 μ H | | 1 |
| Inductor | Select | | 1 |
| Transistor | | 2N1506 | 2 |
| Transistor | | 2N1141 | 1 |
| Resistor | Select | ½ watt 5
percent carbon
composition | 10 |
| Resistor | 100 ohms | ½ watt 5
percent carbon
composition | 1 |
| Resistor | 2 kilohms | ½ watt 5
percent carbon
composition | 2 |
| Resistor | 1 kilohm | ½ watt 5
percent carbon
composition | 1 |
| Resistor | 7.5 kilohms | ½ watt 5
percent carbon
composition | 1 |
| Resistor | 680 kilohms | ½ watt 5
percent carbon
composition | 1 |
| Resistor | 510 kilohms | ½ watt 5
percent carbon
composition | 1 |
| Resistor | 3.3 kilohms | ½ watt 5
percent carbon
composition | 1 |
| Resistor | 11 ohms | ½ watt 5
percent carbon
composition | 1 |
| Resistor | 5.1 kilohms | ½ watt 5
percent carbon
composition | 1 |

| Part | Value | Note | Quantity |
|---|--------------|---|----------|
| Resistor | 160 kilohms | ½ watt 5
percent carbon
composition | 1 |
| Transformer | Select | | 3 |
| 64 MC LIMITER AND DISC MODULE — X479676 | | | |
| Capacitor | .005 μ f | Sprague THD-50 | 33 |
| Capacitor | 1-10 pf | Johanson
JMC-2950 | 4 |
| Capacitor | 3 pf | Arco-Elmenco
DM10F-030K | 1 |
| Capacitor | 15 pf | Arco-Elmenco
DM10F-150K | 1 |
| Capacitor | 4.7 μ f | Sprague
150D474X9035A2 | 1 |
| Capacitor | 5 pf | Arco-Elmenco
DM10F-050 | 1 |
| Capacitor | 33 pf | Arco-Elmenco
DM10F-330K | 2 |
| Capacitor | 47 pf | Arco-Elmenco
DM10F-470K | 1 |
| Connector | UG-625 B/U | | 3 |
| Connector | | Multiple-pin | 1 |
| Diodes | | 1N1871E | 5 |
| Diodes | | 1N750A | 1 |
| Inductor | 10 μ h | Delevan 1025-44 | 2 |
| Inductor | 5.6 μ h | Delevan 1025-38 | 5 |
| Inductor | 0.5 μ h | 11T Hughes
special | 1 |
| Inductor | 0.25 μ h | 8T Hughes
special | 2 |
| Inductor | | 12T | 1 |
| Meter | | | 2 |
| Resistor | 36 ohms | ¼ watt carbon
composition | 1 |
| Resistor | 430 ohms | ¼ watt carbon
composition | 2 |
| Resistor | 4.7 kilohms | ¼ watt carbon
composition | 5 |
| Resistor | 56 ohms | ¼ watt carbon
composition | 3 |
| Resistor | 47 ohms | ¼ watt carbon
composition | 5 |
| Resistor | 16 ohms | ¼ watt carbon
composition | 2 |
| Resistor | 220 ohms | ¼ watt carbon
composition | 2 |
| Resistor | 3.3 kilohms | ¼ watt carbon
composition | 3 |
| Resistor | 30 kilohms | ¼ watt carbon
composition | 1 |
| Resistor | 200 ohms | ¼ watt carbon
composition | 4 |
| Resistor | 6.2 kilohms | ¼ watt carbon
composition | 1 |
| Resistor | 1.2 kilohms | ¼ watt carbon
composition | 1 |

TABLE 7-2. (continued)

| Part | Value | Note | Quantity |
|--|----------------|---------------------------|----------|
| Resistor | 1.0 kilohm | ¼ watt carbon composition | 1 |
| Resistor | 470 ohms | ¼ watt carbon composition | 1 |
| Resistor | 5 kilohms | ¼ watt carbon composition | 2 |
| Resistor | 51 ohms | ¼ watt carbon composition | 1 |
| Resistor | 100 ohms | ¼ watt carbon composition | 9 |
| Resistor | 560 ohms | ¼ watt carbon composition | 3 |
| Resistor | 15 kilohms | ¼ watt carbon composition | 2 |
| Resistor | 150 ohms | ¼ watt carbon composition | 2 |
| Resistor | 2.7 kilohms | ¼ watt carbon composition | 1 |
| Resistor | 18 kilohms | ¼ watt carbon composition | 2 |
| Resistor | 3 kilohms | ¼ watt carbon composition | 2 |
| Resistor | 10 kilohms | ¼ watt carbon composition | 1 |
| Resistor | 2.4 kilohms | ¼ watt carbon composition | 1 |
| Resistor | 5.1 kilohms | ¼ watt carbon composition | 1 |
| Resistor | 7.5 kilohms | ¼ watt carbon composition | 1 |
| Transformer | | Hughes special | 4 |
| Transistor | | 2N1141 | 4 |
| Transistor | | 2N709 | 2 |
| Transistor | | 2N1195 | 3 |
| CRYSTAL VOLTAGE-CONTROLLED OSCILLATOR — 479677 | | | |
| Capacitor | 1000 pf | Feedthrough | 19 |
| | 0.0047 μ f | | 24 |
| | 0.8-8.0 pf | Variable | 7 |
| | 100 pf | | 1 |
| | 0.01 μ f | | 1 |
| | 56 pf | | 3 |
| | 22 μ f | 15 volts | 2 |
| | 0.005 μ f | | 1 |
| Varicap | | MA 4325F | 1 |
| Crystal oscillator | | | 1 |
| Zener diode | 3.9 volts | 1N748A | 1 |
| Connector | | BNC | 4 |
| Choke | 6.8 μ h | | 1 |
| Choke | 15 μ h | | 9 |
| Choke | 5.6 μ h | | 14 |
| Transistor | | 2N1405 | 5 |
| | | 2N1141 | 2 |
| | | 2N707A | 1 |
| | | 2N1506 | 1 |

| Part | Value | Note | Quantity |
|---|---------------|-------------------------------------|----------|
| Resistor | 1 kilohm | Carbon composition | 6 |
| Resistor | 150 ohms | Carbon composition | 1 |
| Resistor | 470 ohms | Carbon composition | 2 |
| Resistor | 510 ohms | Carbon composition | 1 |
| Resistor | 51 ohms | Carbon composition | 1 |
| Resistor | 2 kilohms | Carbon composition | 3 |
| Resistor | 10 kilohms | Carbon composition | 4 |
| Resistor | 47 ohms | Carbon composition | 2 |
| Resistor | 750 ohms | Carbon composition | 1 |
| Resistor | 3.9 kilohms | Carbon composition | 1 |
| Resistor | 1.6 kilohms | Carbon composition | 1 |
| Resistor | 16 ohms | Carbon composition | 2 |
| Resistor | 100 ohms | Carbon composition | 1 |
| Resistor | 220 ohms | Carbon composition | 1 |
| Resistor | 200 ohms | Carbon composition | 1 |
| Resistor | 2 megohms | Carbon composition | 1 |
| Transformer | | Special | 8 |
| BALANCED MODULATOR AND PREAMPLIFIER — X479678 | | | |
| Capacitor | 0.001 μ f | | 14 |
| Capacitor | 120 pf | | 1 |
| Capacitor | Select | | 4 |
| Capacitor | 0.05 μ f | | 1 |
| Capacitor | 160 μ f | | 1 |
| Capacitor | 100 μ f | | 1 |
| Capacitor | 7-45 μ f | Variable | 2 |
| Capacitor | 0.02 μ f | Feedthrough | 3 |
| Capacitor | Select | Variable | 1 |
| Connector | | Coaxial | 3 |
| Connector | | Multiple-pin | 1 |
| Inductor | Select | | 4 |
| Resistor | 75 ohms | ½ watt 5 percent carbon composition | 2 |
| Resistor | 2.7 kilohms | ½ watt 5 percent carbon composition | 4 |
| Resistor | 68 kilohms | ½ watt 5 percent carbon composition | 2 |

TABLE 7-2. (continued)

| Part | Value | Note | Quantity |
|---|----------------------|-------------------------------------|----------|
| Resistor | 1.2 kilohms | ½ watt 5 percent carbon composition | 1 |
| Resistor | 100 kilohms | ½ watt 5 percent carbon composition | 3 |
| Resistor | 68 ohms | ½ watt 5 percent carbon composition | 1 |
| Resistor | 47 kilohms | ½ watt 5 percent carbon composition | 2 |
| Resistor | 12 kilohms | ½ watt 5 percent carbon composition | 1 |
| Resistor | 1 kilohm | ½ watt 5 percent carbon composition | 2 |
| Resistor | 10 ohms | ½ watt 5 percent carbon composition | 2 |
| Resistor | 12 ohms | ½ watt 5 percent carbon composition | 1 |
| Resistor | 510 ohms | ½ watt 5 percent carbon composition | 1 |
| Resistor | 100 ohms | ½ watt 5 percent carbon composition | 1 |
| Resistor | 15 kilohms | 1 watt | 1 |
| Resistor | 51 ohms | 2 watts | 1 |
| Resistor | Variable 2.5 kilohms | | 1 |
| Resistor | Variable 5 kilohms | ½ watt | 1 |
| Transformer | | Select | 4 |
| Tube | | 7360 | 1 |
| Tube | | 6688 | 2 |
| Tube | | 6CL6 | 1 |
| FREQUENCY TRANSLATION TRANSMITTER — X479679 | | | |
| Capacitor | 100 pf | Arco-Elmenco DM10F-101K | 2 |
| Capacitor | 47 pf | Arco-Elmenco DM10F-470K | 1 |
| Capacitor | 51 pf | Arco-Elmenco DM10F-510K | 2 |
| Capacitor | 0.005 μf | Sprague THD50 | 11 |
| Capacitor | 1-10 pf | Johanson JMC-2950 | 3 |
| Capacitor | | Pacific Semiconductor PC-124 | 3 |
| Capacitor | 10 μf, 35 volts | Sprague 150D106X0050R2 | 2 |
| Feedthrough capacitor | | | 2 |
| Variable capacitor | 0.8-8 pf | | 1 |

| Part | Value | Note | Quantity |
|---|-------------|------------------------------------|----------|
| Connector, coaxial | UG-625 B/U | | 2 |
| Connector | M4S-LRN | Winchester four-pin | 1 |
| Inductor | 5.6 μh | Delevan 1025-38 | 4 |
| Inductor | 10 μh | Delevan 1025-44 | 3 |
| Resistor | 270 ohms | ¼ watt carbon composition | 1 |
| Resistor | 300 ohms | ¼ watt carbon composition | 2 |
| Resistor | 430 ohms | ¼ watt carbon composition | 1 |
| Resistor | 4.7 kilohms | ¼ watt carbon composition | 1 |
| Resistor | 9.1 kilohms | ¼ watt carbon composition | 1 |
| Resistor | 820 ohms | ¼ watt carbon composition | 1 |
| Resistor | 2.2 kilohms | ¼ watt carbon composition | 2 |
| Resistor | 2.5 kilohms | ¼ watt carbon composition | 2 |
| Resistor | 27 ohms | | 1 |
| Variable resistor | 100 kilohms | | 2 |
| Rotary switch | PA-6010 | | 1 |
| Transformer | | Hughes special | 3 |
| Piezo electric crystal | 64.7094 mc | F700, International Crystal | 1 |
| Transistor | | 2N1141 | 3 |
| Diodes | | | 2 |
| Isolator | | Hughes 475115 | 2 |
| Bandpass filter | | Rantec FS217 | 1 |
| Traveling-wave tube | | Huggins 210E | 1 |
| X3 frequency multiplier | | Hughes 475116 | 1 |
| Isolator | | MELABS XH 421 | 1 |
| Bandpass filter | | Hughes 475118 | 1 |
| Attenuator | | ARRA 5-6684-50 | 1 |
| Coaxial connectors | | | 27 |
| MISCELLANEOUS SYSTEM AND PANEL PARTS — 479680 | | | |
| Variable attenuator | | ARRA 5684-50 | 1 |
| Variable attenuator | | ARRA 5-6684-50 | 2 |
| Filter | | RANTEC FCXXX | 3 |
| Meter | | Triplett 327PL | 5 |
| Power supply | | +15, +24 volts dc Hughes | 1 |
| | | —15, —24 volts dc Hughes | 1 |
| | | +250 volts dc, 6.3 volts ac Hughes | 1 |

TABLE 7-2 (continued)

| Part | Value | Note | Quantity |
|--|-------------|---------------------------|----------|
| Traveling-wave tube amplifier and power supply | | Huggins 210F | 2 |
| Traveling-wave tube amplifier and power supply | | Huggins 210E | 1 |
| Traveling-wave tube amplifier and power supply | | Huggins 309F | 1 |
| Resistor | 9.1 kilohms | ½ watt carbon composition | 1 |
| | 1 kilohm | ½ watt variable | 1 |
| | 1.8 kilohm | ½ watt carbon composition | 1 |
| Power switch | | MS 35058-21 | |
| Bandpass filter | | Hughes 475118 (mod) | 2 |
| High pass filter | | Hughes special | 1 |
| X2 multiplier | | Hughes 475126 | 2 |

| Part | Value | Note | Quantity |
|---|-------|---------------------|----------|
| Isolator | | Hughes 475115 | 1 |
| X32 multiplier | | Hughes 475114 (mod) | 1 |
| X3 multiplier | | Hughes 475116 (mod) | 1 |
| Mixer | | Hughes 475100 (mod) | 1 |
| Frequency divider ½ | | Hughes 475117 (mod) | 1 |
| Frequency divider ½ | | Hughes 475114 (mod) | 1 |
| Q phase detector | | Hughes special | 1 |
| Phase detector and isolators | | Hughes special | 1 |
| Crystal voltage-controlled oscillator and doubler amplifier | | Hughes special | 1 |
| Balanced modulator | | Hughes special | 1 |

PREFERRED PARTS, MATERIALS, AND PROCESSES

Parts. The approved parts list remains as published in the August 1963 Monthly Progress Report. Of the 70 types presently listed in the Advance Bill of Materials, 48 are fully qualified to the requirements of their specifications, while some others will have only certain added specification requirements to be verified.

The basic elements of parts selection are as follows:

- 1) Use of only space-qualified parts controlled by adequate documentation.
- 2) Addenda to existing Syncom and Surveyor part specifications to provide power aging, additional screening, etc., to enhance reliability.
- 3) Thorough vendor surveys to evaluate manufacturing controls.
- 4) Standardization efforts to minimize the number of types of devices.

This program has led to the following results:

- 1) Every potential supplier of parts has been surveyed by part specialists and project, materiel, and quality control personnel.
- 2) Detailed technical discussions and evaluations of Syncom addenda have been conducted with suppliers to determine their willingness and ability to comply with proposed requirements and compatibility with their in-house manufacturing methods. The attempt has been to avoid unnecessarily increasing costs by requiring actions on the part of suppliers which are redundant to their normal procedures.
- 3) Part specialists, working with designers, have reduced the total number of different semiconductor devices from 76 to 53. Though reliability gains cannot be accurately assessed at this time, reduction in device types will certainly increase reliability control.

Most of the parts procured to the specifications will undergo a period of "burn-in" by the vendor, to eliminate parts that might fail early in use, and a subsequent power-aging period, to determine the consistency of performance, drift, etc. From 1.7 to 3 times the number of parts required for flight spacecraft will be tested in this manner, usually by the vendors, and data will be supplied on each part, to be confirmed by acceptance testing on receipt. The parts with the most consistent performance will be selected for flight use, the balance being used for ground equipment, simulators, etc.

Parts Specifications. A new general specification covering reliability requirements for the space environment has been prepared for procurement of Advanced Syncom semiconductors. The specification is a standard format and will be accompanied by several tables establishing exact requirements for a given part.

It is expected that existing 988XXX basic specifications will gradually be replaced with the new format. In the interim, procurement of three-fourths of the Advanced Syncom requirements will be made with the basic specifications, modified by an addendum that will result in the same total effect as the standard specification. The addendum consists of extracts from the standard specifications; its preparation is complete, and it has been incorporated in 38 specifications to date.

Several parts specifications prepared for Advanced Syncom were included in the September Monthly Report (pages 6-3 through 6-70). These documents are selective examples of the entire specification program.

OSM and Bendix RF Connectors. The Bendix and Omni Spectra miniature RF con-

nectors are presently being evaluated. Problems with connectors previously available were:

- 1) Galling of mating threads — aluminum components showed excessive wear under normal number of matings and unmatings.
- 2) Coarse mating thread — connectors became unmated during vibration.
- 3) Grounding shields—shields could not be grounded securely to the shells.
- 4) Noncaptivated contacts — center contacts were being either pushed in or pulled out on panel or bulkhead jacks.
- 5) Ease of assembly—connectors had many components and were relatively difficult to assemble.

Bendix and Omni Spectra connectors eliminate these problems by the following methods:

- 1) Use of stainless steel for shell material decreases thread wear and galling.
- 2) Finer mating threads discourage uncoupling due to vibration.
- 3) Soldering to metal ferrule provides reliable grounding of shields.
- 4) All bulkhead and panel jacks have captivated contacts.
- 5) Fewer components minimize assembly problems.

The Bendix and Omni Spectra series have these characteristics:

Impedance 50 ohms

Cable RG-55B/U or teflon RF 142/U for OSM type
RG-178/U for OSSM type

| | |
|--|--|
| Voltage standing wave ratio | 1.1:1 over a range of 2 to 10 gc |
| Material | Shell, stainless steel; dielectric, teflon |
| Finish | Gold plate per MIL-G-45204 |
| Size | 1/4 size and weight of BNC type |
| Environmentally resistant construction | |

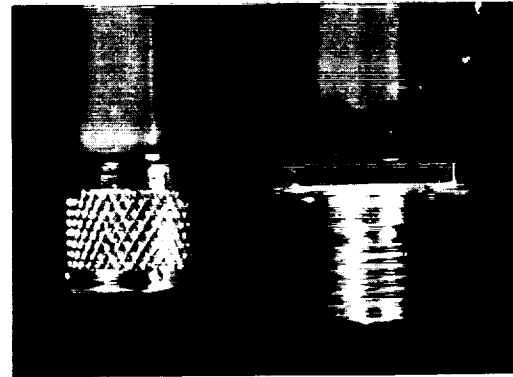
The Bendix and Omni Spectra connectors can be mated interchangeably. Detailed information on the mating faces has been submitted by both vendors and mateability thus verified. All materials and finishes will be space-approved.

Figures 7-1 and 7-2 show the connectors, both apart and mated. Figure 7-2 shows the means by which the contact in the female jack is captivated to prevent motion when the male plug is inserted. Preliminary specifications are being drafted, with both Bendix and Omni-Spectra listed as sources.

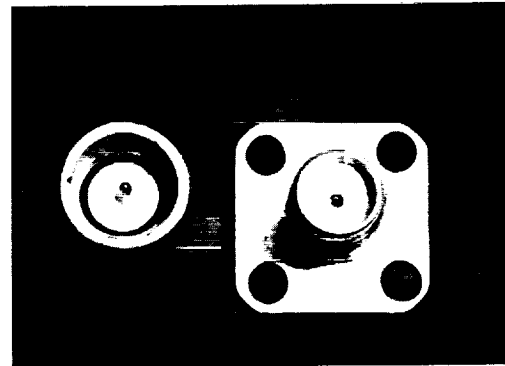
Magnetic Parts. Design of magnetic parts and fabrication of breadboard samples has proceeded as rapidly as circuit design criteria permitted. The breadboard electrical designs for such parts have been completed. Final design for production and corresponding procurement/production specifications will be completed during subsequent phases of the program.

Newly Developed Parts. A small adjustable air capacitor of 0.1 to 1.0 pf range was not initially available commercially; therefore, one was designed and samples constructed for test. A commercial vendor has since proposed a design and samples have been evaluated. It will be covered by specification X988521.

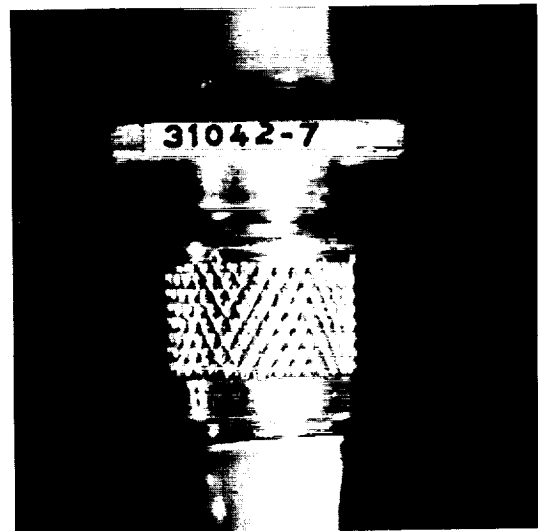
A coaxial switch had been developed for use in the Syncom 2 program but became available too late for such use. It is being evaluated for



a) Side view



b) End View



c) Mated

FIGURE 7-1. BENDIX CONNECTORS

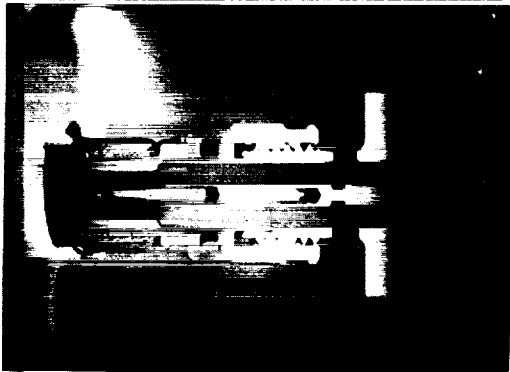


FIGURE 7-2. SECTIONAL VIEW OF MATED OSM CONNECTORS

possible use in the Advanced Syncom program, since it is smaller and lighter than the ferrite switches presently contemplated. It is a hermetically sealed, latching type relay. Samples will be procured equipped with OSM type connectors for further evaluation in subsequent program phases.

Materials. The following materials have been evaluated and approved for Advanced Syncom and are to be added to the list in Appendix B, page B-49, of the April 1963 Monthly Progress Report:

- 1) Epoxy syntactic foam covered by HP 16-108, "Heat Resistant Syntactic Foam, Application of." This material, developed by Hughes, is designed as a low-density (32 to 35 lb/ft³) filler in load-bearing applications where shear, compressive, and bearing stresses are encountered. Minimum room temperature properties are as follows:

| | |
|--|------------------|
| Tensile shear
(aluminum to
aluminum) | 1750 psi minimum |
| Compression | 6500 psi minimum |

- 2) Modified epoxy adhesives covered by HP 16-111 "Adhesive Bonding, Reinforced,

using Modified Epoxy Resins." This is a Hughes-developed material, specifically for bonding solar cells to a variety of substrate materials over wide thermal ranges, facilitating flat-mounting of cells, and improving solar array reliability.

- 3) Modified, transparent epoxy material designed to provide potted cable to connector joints of superior flexibility and capable of being readily repaired. The material cures at room temperature and is covered by HP 16-110 "Potting with Flexible Epoxy Materials."
- 4) Improved hook-up wire covered by HMS 2-1293, "Wire, Electrical, Insulated, Copper Alloy, Modified Polyimide Coated, 250 volts rms (For 200°C Service)"—teflon dielectric. It is available in unshielded (Type I) or shielded and jacketed Type II), Class I (silver coated,) or Class II (nickel coated). Class II is preferred for Syncom.

Processes. The following processes have been developed and approved for Advanced Syncom and are to be added to the Preferred Processes list in Appendix B, page B-77, of the April 1963 Monthly Progress Report.

Items 1, 2, and 3, under *Materials*, involved the development of a process to properly employ the materials. These processes are now preferred, within their scope, for use on Syncom. In addition:

- 1) HP 14-30, "Process for Lead to Lead Electronic Assembly," has been developed and released for comments.
- 2) Soldering per Mil-S-6872, except for etched circuitry, has been replaced by the NASA-MSFC-158B.

7. Materials, Processes, and Components

- 3) HP 16-107, "Laminates, Structural, Epoxy-Glass Fabric, Fabrication of," has been developed to provide a reliable process for reinforced plastics fabrication, at the same time utilizing resin systems approved for the space environment.
- 4) HP 11-45, "Fusion Welding, Electron Beam," has been prepared and released for comments.

RADIATION ENVIRONMENT STUDIES

A test program was started to determine the ionization effects of semiconductors used in the Advanced Syncom spacecraft. Six types of semiconductors, representative of those used in the Advanced Syncom spacecraft, have been selected for the tests. Four are devices suspected of having an unstable surface:

- 988821 (2N2185) Si micro-alloy
- 988825 (2N1041) Ge device
- 988828 (2N1405) Ge device
- 988829 (2N1141) Ge device

Two are devices that presumably have a stable surface:

- 988718 (1N3062) Si device
- 988802 (2N871) planar Si device

With the exception of 2N1041, 12 of each of these devices will be irradiated. Only six will be used for 2N1041.

To obtain data in a short time, one of each of the devices was irradiated at a rate 1.5×10^6 rads/hr from a cobalt source. After a cumulative dose of 2.4×10^5 rads, the 2N871 reverse current increased by a factor of 5 (from 60 to 300 nanoamperes), and the 2N1405 increased from 170 to 400 nanoamperes. The remaining four

devices had increases of approximately 20 percent. This increased leakage, however, will not cause any difficulties in operation of the spacecraft electronics since the circuits were sufficiently derated.

The general program summarized in Table 7-3 will be followed except that the dose rates will be changed to obtain as much information as possible during the remainder of the contractual period. It is anticipated that for each type of device, a series of leakage measurements will lead to a set of plots probably similar to those shown in Figure 7-3, from which the effects of low-dose-rate radiation on semiconductor surface phenomena can be determined. Figures 7-4 and 7-5 show the cobalt 60 radiation source in which the semiconductors are being irradiated.

Studies of Effects of Particle Radiation on Parts. Two studies were made of radiation effects on parts. One of the studies, "A Survey of the

TABLE 7-3. RADIATION PROGRAM SUMMARY

| Major Group | 1.0 rads/hr | | | 10 rads/hr | | |
|---|--|---|---|---|---|---|
| Dose Rate | | | | | | |
| Subgroup | a | b | c | a | b | c |
| Number of Samples | 2 | 2 | 2 | 2 | 2 | 2 |
| Leakage-current measurement at accumulated dose of (rads) | 0, 10, 20, 50, 100, 200, 500, 10^3 , 2×10^3 | | | 0, 10, 100, 200, 500, 10^3 , 2×10^3 , 5×10^3 , 10^4 , 2×10^4 | | |
| Photographs of reverse characteristics at an accumulated dose of (rads) | 0, 10, 100, 10^3 | | | 0, 10, 100, 10^3 , 10^4 | | |
| Bias voltage (V_{ca} or V_a) | Subgroup a | | | Subgroup b | | |
| 2N1041 | 40 | | | 80 | | |
| 2N1141 | 13 | | | 26 | | |
| 2N1405 | 11 | | | 22 | | |
| 2N2185 | 7 | | | 14 | | |
| 1N3062 | 100 | | | 200 | | |
| 2N871 | 22.5 | | | 90 | | |

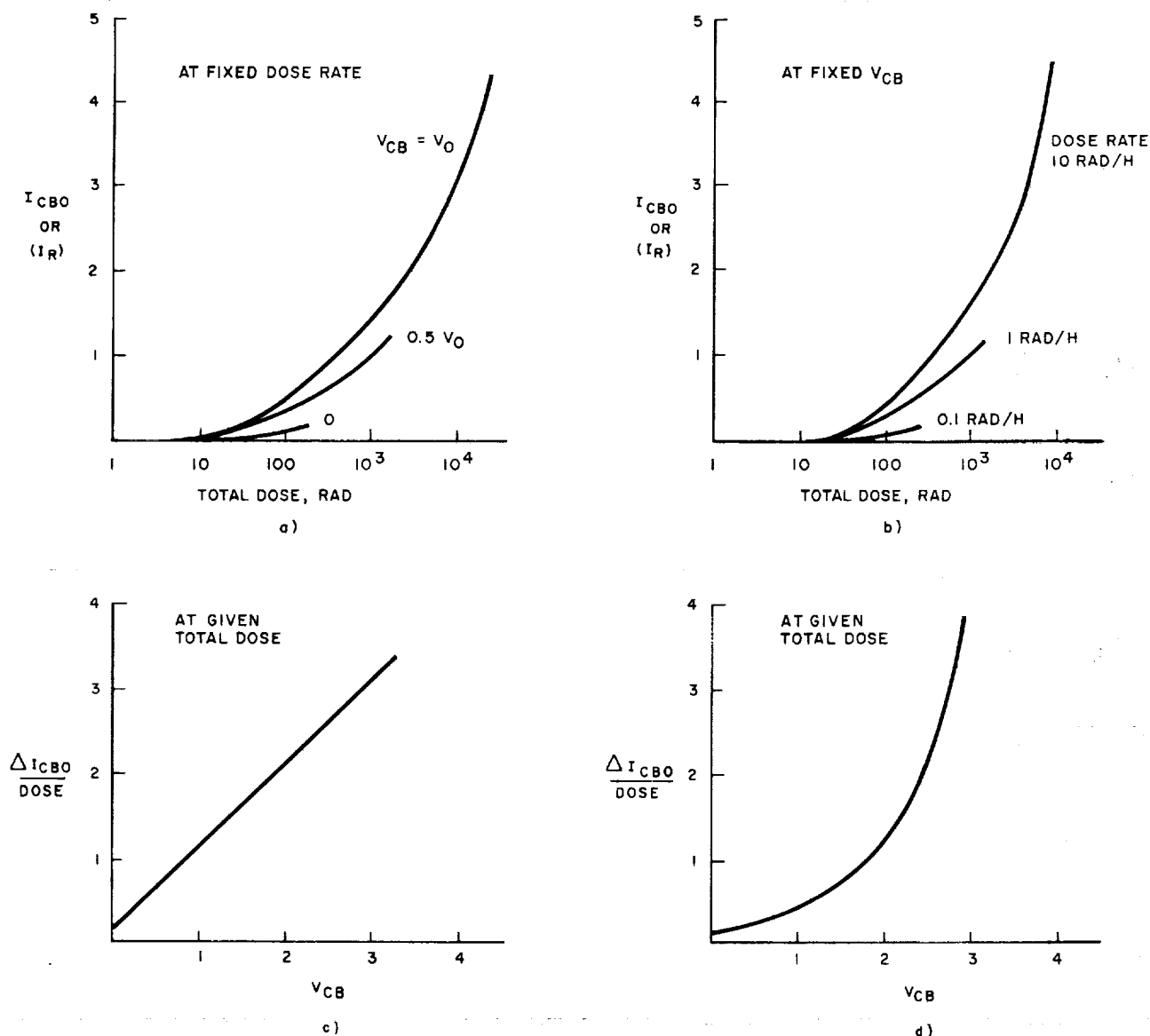


FIGURE 7-3. RADIATION-INDUCED LEAKAGE CURRENTS

Effects of Electron and Proton Radiation on Silicon Solar Cells," presents a unique method, based on defect density, for predicting the lifetime of silicon solar cells or semiconductor devices in a combined field of electrons and protons having a given energy spectral distribution. Silicon solar cell lifetime in Van Allen, solar

flare, and cosmic-ray radiation has been roughly estimated using this method. These computations are based on the best estimates available of the various energy spectra and energy-dependent damage constant for diffusion lengths obtained experimentally. Considerable uncertainty exists in these life estimates, however, because of un-

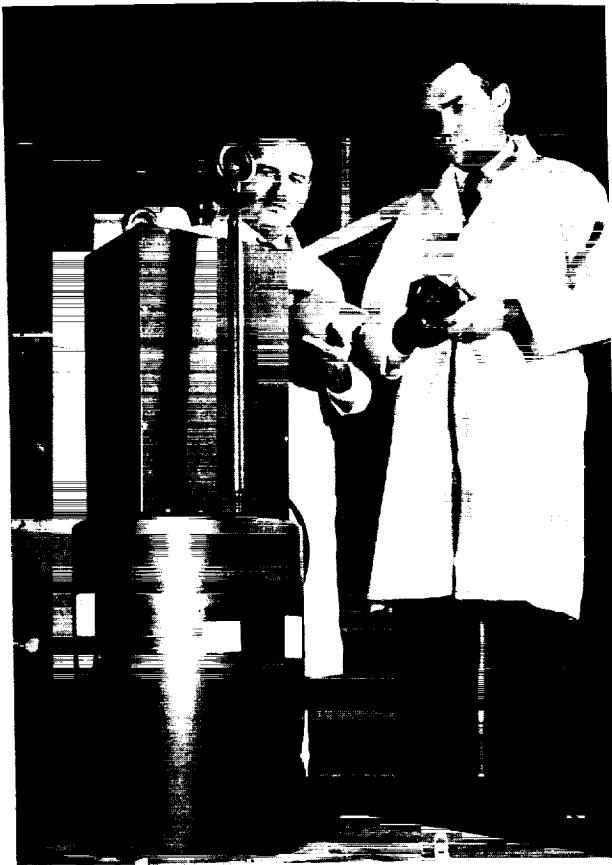


FIGURE 7-4. COBALT 60 SOURCE USED IN IRRADIATING SEMICONDUCTORS

certainties in the environmental conditions, including flux, energy spectrum, and temporal fluctuations at the Advanced Syncom altitude.

The other study, "The Ionization Effects on Transistors Due to Particles Trapped in the Geomagnetic Field at Syncom Altitude," discusses the effects of radiation on transistors and the effects of different types of shielding. The study is included on the following pages.

Ionization Effects Study. Surface effects of ionizing radiation on transistors became a familiar problem when the phenomenon was observed in the transistors used in Telstar. Experimental results indicate that the total radiation

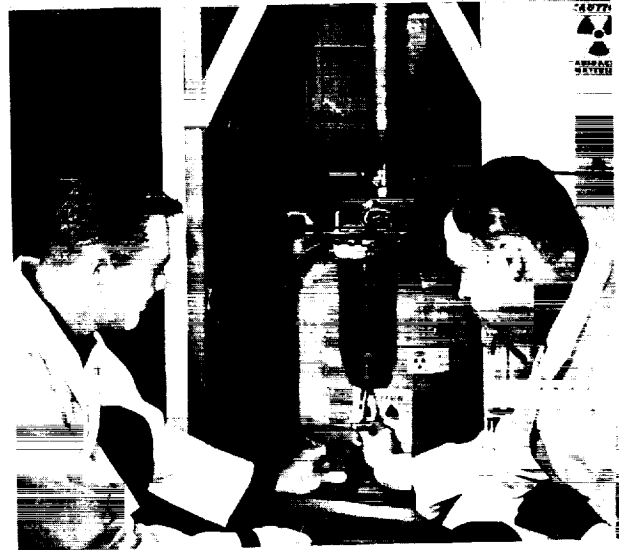


FIGURE 7-5. MOUNTING SEMICONDUCTORS IN COBALT 60 SOURCE

dose seems to be a more significant factor in producing the effect than the dose rate (Reference 7-1). A rough estimate has been made to see whether an accumulated radiation dose over a 5-year period at Syncom altitude would cause a significant surface effect on transistors. Because the size, energy spectrum, temporal fluctuation, and recurrence frequency of solar-flare proton events are unpredictable, only ionizing radiation particles (electrons and protons) trapped in the geomagnetic field at Syncom altitude are considered. Permanent surface damage caused by these particles is not considered.

- **Technical Discussion.** The surface effects in question result from the ionization of gases within a transistor encapsulation and the interaction between the ionized gas and the semiconductor surface.* When an electrical field is not

*Radiation particles penetrating the transistor encapsulation can directly cause permanent surface damage, resulting in an increase in surface recombination velocity and in a reduction of current gain. However, the effects due to this mechanism (disordering of the lattice structure by atomic displacement) are not considered here.

present, electron-ion pairs created by ionization essentially recombine, and the surface phenomenon would hardly be detectable. When a field is present (as in a biased transistor), electrons and ions will be collected at proper electrodes (the semiconductor surfaces) and will form an inversion layer. Sufficient accumulation of electrons and ions at the surface, which causes an appreciable change in the surface state, leads to an ionic leakage (an increase in I_{CBO}). As a result, the current gain will decrease. In this sense, the effect is directly associated with ionization. Consequently, when the number of electron-ion pairs produced by gamma-ray radiation is correlated with the number of pairs produced by the charged particles in question, the data regarding the surface effects on transistors caused by gamma-ray radiation* is immediately applicable to the present problem.

The great difficulty in determining the space radiation environment is due to the insufficient amount of data dealing with the flux, energy spectrum, and temporal fluctuations at a given location. However, based on the assumptions made in Reference 7-2, electron and proton differential energy spectra at Syncom altitude have been plotted in Figures 7-6 and 7-7.

If a transistor is encapsulated within an iron case 10 mils thick, the cutoff energy becomes approximately 425 kev for electrons (Reference 7-3) and nearly 10 mev for protons (Reference 7-4). Protons within the Van Allen radiation belt at Syncom altitude are completely shielded from the interior of the transistor by the iron transistor case and will not contribute to the ionization of gases within the encapsulation. Electron differ-

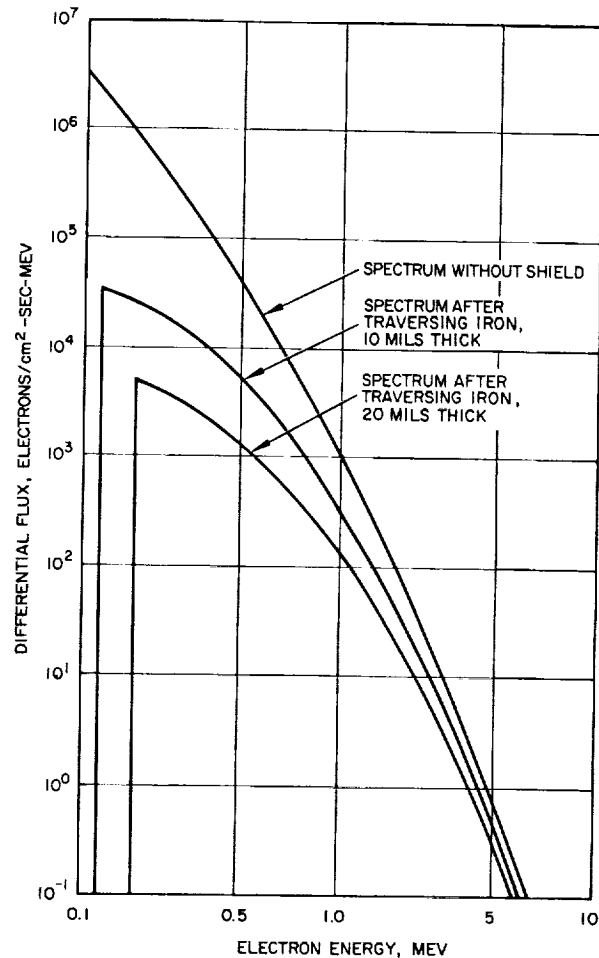


FIGURE 7-6. ELECTRON DIFFERENTIAL ENERGY SPECTRUM (AT SYNCOM ALTITUDE) BEFORE AND AFTER TRAVERSING IRON SHEET

ential energy spectra, after the electrons traverse an iron sheet 10 mils thick and another 20 mils thick, are plotted in Figure 7-6, based on the energy-loss data of electrons in iron as plotted in Figure 7-8 (Reference 7-3), assuming a sharp cutoff. The energy loss is approximately 300 kev for 10 mils and 550 kev for 20 mils of iron at the respective cutoff energy.

According to Figure 7-6, the number of electrons penetrating an iron sheet 10 mils thick is approximately 6×10^3 electrons/cm²-sec. If the encapsulated gas is nitrogen, the energy lost by

*The displacement cross section of gamma rays is very small in comparison with that of protons or electrons, although it depends entirely on the energy of bombarding particles or photons.

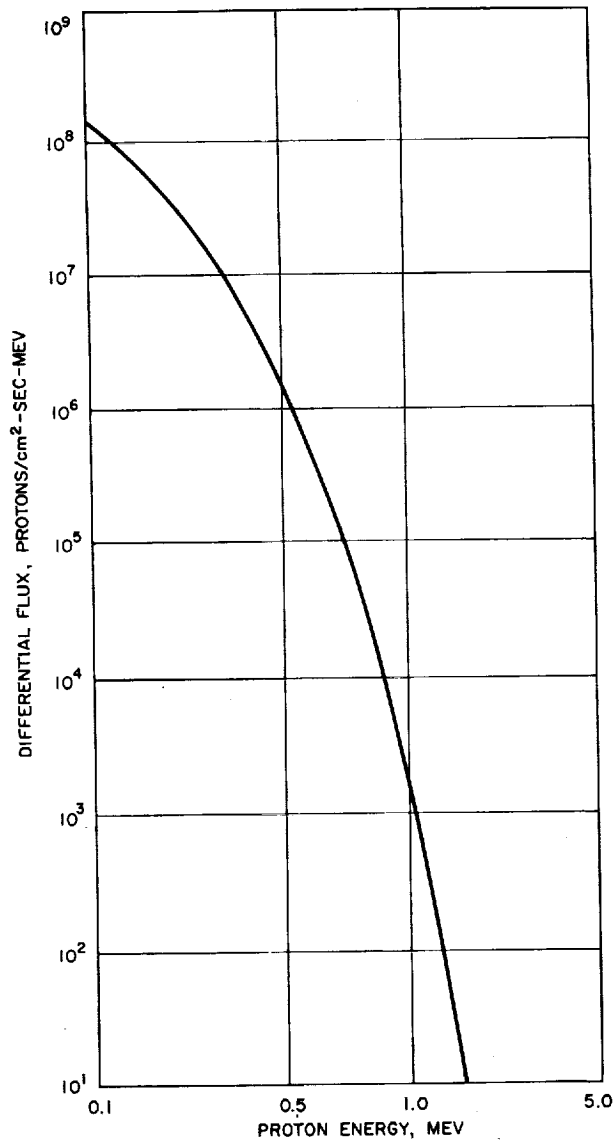


FIGURE 7-7. PROTON DIFFERENTIAL ENERGY SPECTRUM AT SYNCOM ALTITUDE

these electrons traversing a unit path length can be calculated with the aid of the electron energy loss curve in nitrogen gas shown in Figure 7-8. The electrons impart energy to nitrogen gas at the rate of about 18 mev/sec in the first 1 cm of travel. If the energy required to create a pair of nitrogen ions is assumed to be 32 ev, about 5.7×10^5 ion pairs/cc-sec will be produced. This ionizing power is equivalent to a gamma-ray

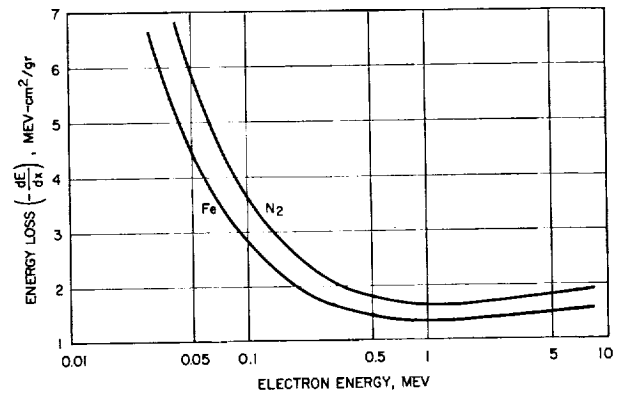


FIGURE 7-8. ELECTRON ENERGY LOSS IN IRON AND NITROGEN

dose of 2.3×10^{-4} rad/sec at the average (gamma-ray) energy of 1.25 mev.* The dose rate thus obtained is approximately 0.83 rads/hr, and the accumulated dose after 5 years is about 3.65×10^4 rads.

Since the radiation dose of electrons is correlated with that of gamma rays (at the average energy of 1.25 mev) through the number of ion pairs created by ionization, the experimental data obtained from the use of a cobalt-60 gamma source (Reference 7-1) are applicable to the present problem. Because the dose rate that the Advanced Syncom will experience is much smaller than the observed experimental dose rate (by a factor of 10^6), no simple conclusion can be drawn from the data. For example, at a high radiation dose rate (of the order of 10^5 rads/hr), the increase in I_{CBO} is proportional to the accumulated dose (see Figure 7-9); however, the effect of a low dose rate on I_{CBO} is completely different in nature from that of a high dose rate, as shown in Figure 7-10 (Reference 7-1).

When a device is removed from the radiation field, its characteristics will partially recover.

*The true absorption coefficient of air at this energy is 2.67×10^2 cm²/gr.

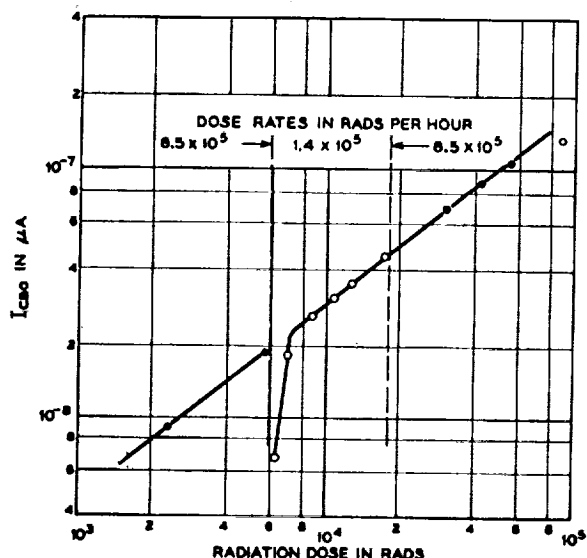


FIGURE 7-9. DOSE RATE EFFECTS ON I_{CBO} AT HIGH DOSE RATES

The recovery time depends on the bias voltage.* If the device is biased, recovery time will be longer (days to weeks) than if no bias is applied (hours to days). At an extremely low radiation dose rate, the recovery processes would take place while the generated ion pairs were simultaneously contributing to the change in the surface state, resulting in a relatively small increase in I_{CBO} . This may constitute a plausible explanation for the phenomena observed in Figure 7-10.

Since the dose rate encountered at Syncom altitude is smaller than the experimental data shown in Figure 7-10 by about a factor of 10, the I_{CBO} of a transistor could increase to 10^{-9} ampere. Thus, if the same device and bias conditions were used in the Advanced Syncom system, the effects would be negligibly small. If the ac-

*This phenomenon is not necessarily restricted to transistors. Semiconductors in general behave in a similar manner when ionization, rather than atomic displacement, predominantly contributes to the phenomenon.

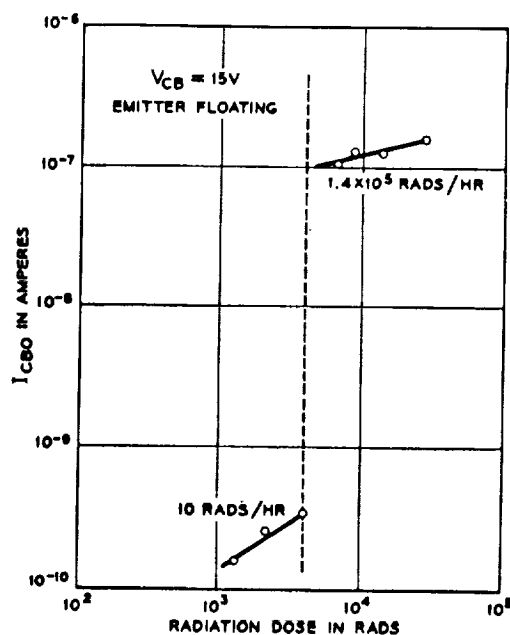


FIGURE 7-10. DOSE RATE EFFECTS IN I_{CBO} AT HIGH AND LOW DOSE RATES

cumulated dose (rather than the dose rate) is regarded as a predominant factor for the degradation, the I_{CBO} could possibly increase to 10^{-5} ampere at a large bias voltage ($V_{CB} = 30$ volts in Figure 7-11). However, such an increase is unlikely for this device since a lower dose rate initiates a degradation at a higher integrated dose, as shown in Figure 7-12.

As seen in Figure 7-11, the degradation of I_{CBO} greatly depends on collector bias voltage, since the ions created within the transistor encapsulation are more efficiently collected at the surfaces under the influence of a higher field. Therefore, a practical method of reducing the surface effects on transistors is to reduce the collector bias.

It should be emphasized that the experimental data in Reference 7-1, on which the above discussions are based, are restricted to a limited number of device types and show a large variation

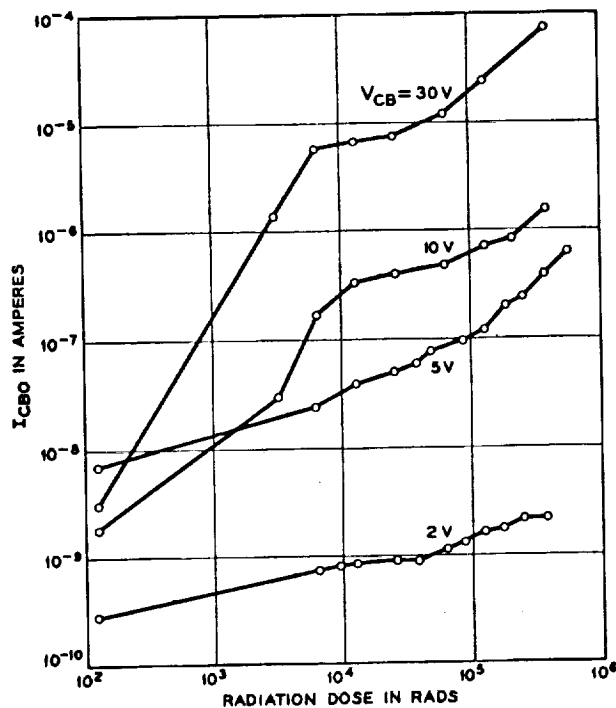


FIGURE 7-11. DEPENDENCE OF DEGRADATION OF I_{CBO} ON COLLECTOR BIAS

in the amount of change of I_{CBO} . The effects of ionizing radiation on I_{CBO} also depend upon the device type, configuration, fabrication and processing techniques and primarily on the stability of the pre-irradiated surface state. Experimental evidence also indicates that a high-gain type of transistor experiences a lesser degradation.

- **Conclusions.** If the transistor case is fabricated from iron (10 mils thick), only electrons, but not protons, trapped in the geomagnetic field at the Syncom altitude would contribute to the ionization of encapsulated gases. Although the accumulated dose may become as high as 3.7×10^4 rads during the 5 years of life, the dose rate would be small (less than 1 rad/hr) and a shielding by the vehicle skin could reduce the dose rate by a factor of 5 or more.

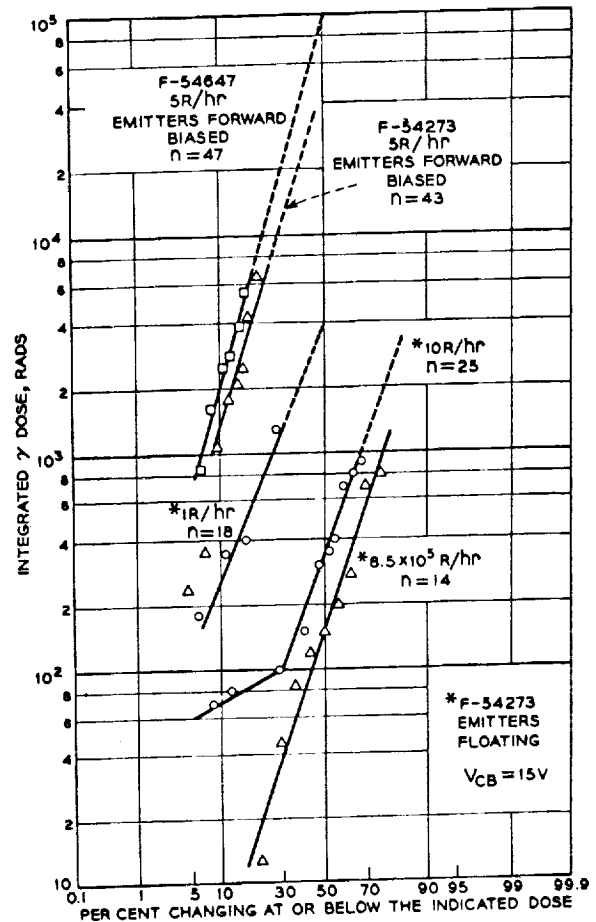


FIGURE 7-12. DISTRIBUTION OF RADIATION DOSE AT INITIATION OF DEGRADATION

Since only meager experimental data are available, an estimate of the effects of ionizing radiation on all the transistor types selected for Advanced Syncom cannot be made with great confidence. Without any consideration of "self" shielding by the vehicle skin, the accumulated dose could possibly induce an appreciable change in I_{CBO} for some types of devices. However, the degradation rate could be made tolerable by selecting proper device types (high-gain devices), choosing an appropriate circuit design (reduction of collector bias), or adopting the proper aging or screening technique. When the shielding effect by the vehicle skin and the low

dose rate are considered, the degradation after 5 years would not be sufficient to cause a significant circuit malfunction. The change in I_{CBO} because of a temperature increase would probably be comparable to the effect caused by ionization.

In the foregoing discussions, the effects of solar flare proton events have been completely ignored. However, since the accumulated dose due to electrons in the Van Allen radiation belt is near or beyond the "initiation dose" for the degradation of I_{CBO} in some devices, a large solar flare event would probably result in an appreciable increase in I_{CBO} before the end of required Syncom life.

- Recommendations. Since the integrated dose calculated is in the critical region in which an appreciable degradation of I_{CBO} might be caused, the following precautions are recommended:

- 1) The reduction of collector bias and case potential
- 2) The selection of high-gain devices
- 3) The exercise of a proper aging or screening technique to eliminate devices with an unstable surface state

To further assure dependable performance, the following additional techniques can be employed:

- 4) The use of evacuated devices
- 5) The use of a thin shield

Simple engineering experiments are highly recommended to determine the difference in the effects of high and low dose rates.

- References

- 7-1. D. S. Peck, R. R. Blair, W. L. Brown, and F. M. Smits, "Surface Effects of

Radiation on Transistors," *The Bell System Technical Journal*, 42, 1963, p. 95.

- 7-2. H. Y. Tada, "A Survey of the Effects of Electron and Proton Radiation on Silicon Solar Cells," TM 755, Hughes Aircraft Company, 1963.

- 7-3. A. T. Nelms, "Energy Loss and Range of Electrons and Positrons," NBS Circular 577, United States Department of Commerce, Washington, D.C., 1956.

- 7-4. J. H. Atkinson and B. H. Willis, "High Energy Particle Data," *University of California Radiation Laboratory Review* 2426, 2, 1957.

MATERIAL AND PROCESS STUDIES

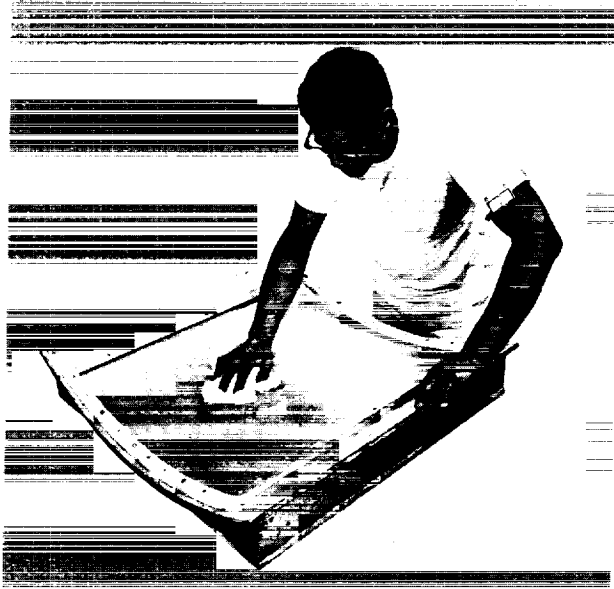
Material Developments. To optimize, within the time limitations imposed, long-term reliability of materials and processes for Advanced Syncom, material development efforts were undertaken. The tasks include solar cell materials and fabrication evaluation, connector development, encapsulation materials development, and thermal materials properties tests.

Solar Array Substrate Development. The objectives of solar array substrate development are 1) composite construction of optimum strength-to-weight ratio, 2) improved panel reproducibility and uniformity, and 3) elimination of as many post-fabrication steps as possible. The substrate incorporates 0.500-inch-thick aluminum honeycomb core, 1/8-inch cell, 0.0007-inch foil. Faces are each three-ply No. 112-Volan A (each 3 mil) glass fabric, impregnated with an epoxy resin system of moderate thixotropy to assure uniform filleting at the core-facing interface. To hold weight to a minimum, the convex (outer) face is only two-ply at the time of substrate fabrication. As discussed later, the third ply is applied during the cell attachment process.

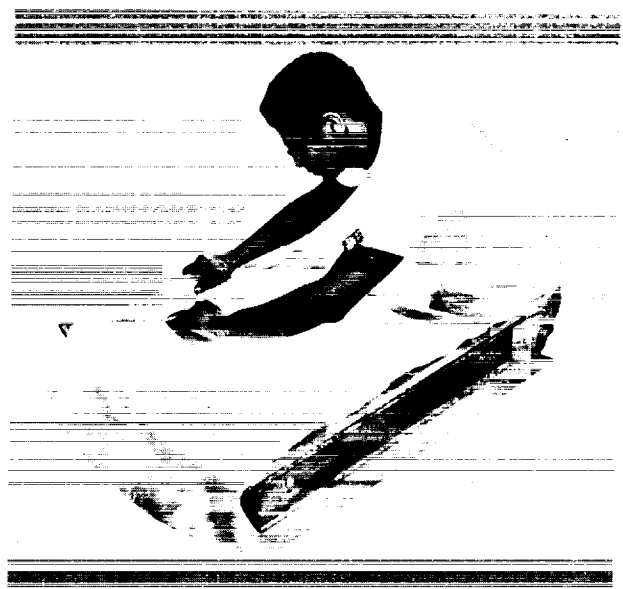
7. Materials, Processes, and Components

The substrate is fabricated in a female aluminum layup tool (Figure 7-13). The 0.5-inch-high nails on the mold surface represent the substrate periphery and permit net fabrication

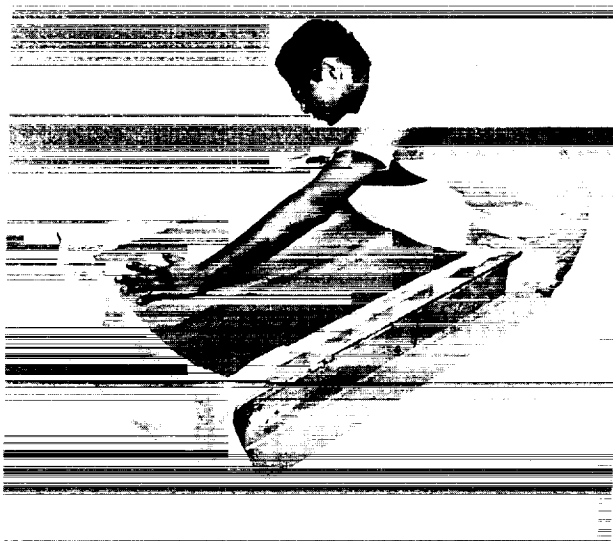
(no trimming required). Edge closure is accomplished by cutting the bottom ply of fabric (ply against mold surface) approximately 1 inch oversize in all directions. This ply is then



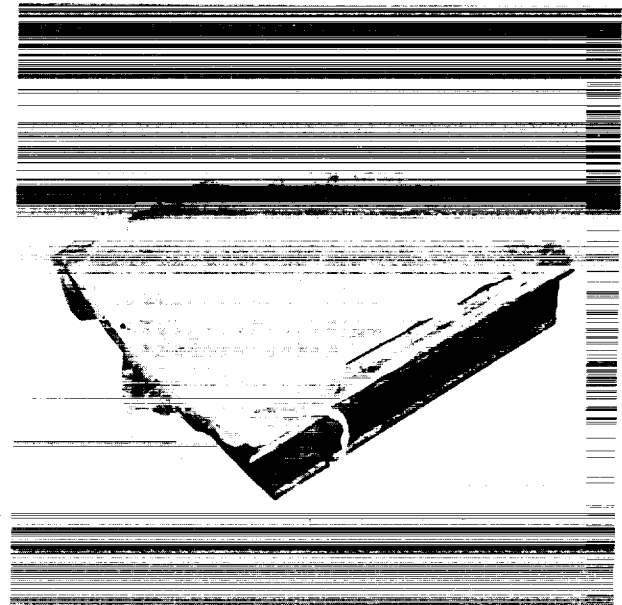
a) Aluminum Cleaning Tool



b) Inserting Glass Cloth into Tool



c) Inserting Honeycomb into Glass-Cloth-Lined Tool



d) Preparation for Vacuum Oven Cure

FIGURE 7-13. SOLAR ARRAY SUBSTRATE FABRICATION

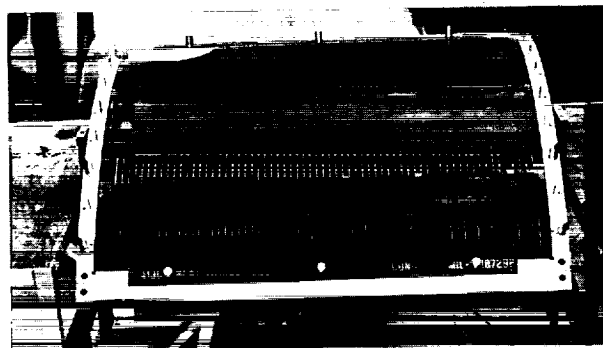
lapped over the top ply of the inner (concave) face, and the completed substrate is heat-cured under vacuum pressure.

The solar cell strings (each string consists of three cells in parallel by 62 cells long) are attached to the substrate in the following manner. One ply of 3-mil glass fabric slightly larger than the cell string is impregnated with the epoxy adhesive and positioned on the prepared substrate. The cell string is then put in place and positive pressure exerted by means of a clamped, rubber-faced holddown pad.

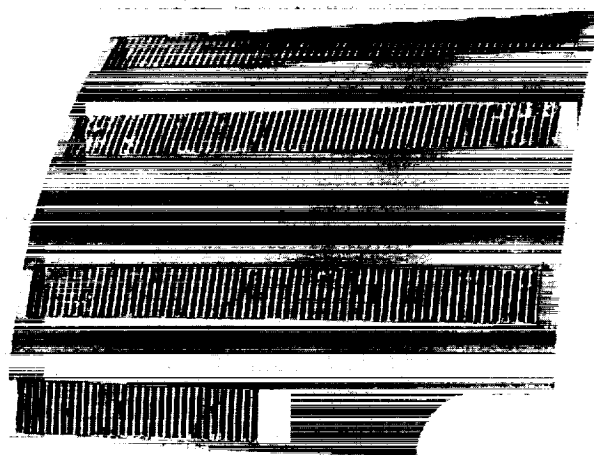
Solar array reliability is significantly improved by these processes. The unique edge-closure permits "net" fabrication which eliminates trimming and the need for the edge-fill putty previously used; the result is an increase in substrate uniformity and reliability. The cell bonding system provides a reinforced bond line, the optimum condition to resist thermal and other environmental stresses. Adhesive weight is also virtually eliminated since the reinforced bond line becomes the integral third ply of the outer face. This technique provides uniform bond line thickness, which is important in assuring uniform stress distribution. The finished panel is shown in Figure 7-14. Because of the high translucency of the resin system, photographic standards of acceptable and unacceptable core cell filleting have been established to make substrate inspection simple and positive.

High Strength Epoxy Syntactic Foam development. High strength epoxy syntactic foam was developed to provide a space-compatible lightweight potting material where relatively high shear and bearing loads must be carried, such as solar panel mounting inserts, etc. Its properties are as follows:

| | |
|------------------------|------------------------------|
| Density | 32 to 35 lbs/ft ³ |
| Tensile shear strength | 1750 psi (minimum) |



a) Solar Cells Being Mounted on Substrate



b) Closeup of Partially Completed Solar Panel

FIGURE 7-14. ATTACHMENT OF SOLAR CELLS TO SUBSTRATE

Compressive strength 6500 psi (minimum)

The resin-hardener system is compatible with laminating systems used by Hughes permitting its use in wet layup construction. A specification covering this material has been released.

Solar Cell to Substrate Adhesive Development. A screening program was initiated to select an adhesive capable of withstanding the environmental and thermal stresses anticipated, whose stress response would have no adverse effect on either substrate or cell, such as cracking, bond line failure, degrading of cell electrical charac-

teristics, or condensation of polymer fractions. Modified epoxies were considered first as a result of the following considerations:

- 1) Their outstanding adhesion is well known.
- 2) A room temperature curing mechanism would be highly desirable from a processing standpoint.
- 3) A well formulated, highly crosslinked epoxy system represents probably the most stable organic material for space applications available today for temperatures up to 300°F.
- 4) Modification, or making the system more flexible, was felt necessary to provide an acceptably low second-order transition temperature. It was also felt desirable that the polymer, at this transition point, not assume a crystalline structure. Such a structure, on thermal cycling, can induce a decrease in polymer integrity with an attendant loss of adhesion.

Screening tests were made of nine formulations, each used to attach a cell to a small piece of substrate. These samples were given 25 thermal shock cycles from 260° to -320°F. These extremes, well beyond the thermal environment anticipated for Syncom, were selected because two of the three test parameters were not representative of the actual mission. The minimum practical test pressure, 10^{-7} Torr, is significantly higher than the 10^{-12} Torr expected in deep space; the available test time of 2 months is equivalent to only about 5 percent of the minimum orbital life. The third test parameter, temperature, remains the only factor capable of being intensified.

By increasing the thermal range, it was felt that greater confidence could be attached to the test results. Specimens were visually and microscopically examined for cell cracking, bond-line

cracking, and loss of adhesion. Surviving specimens were then subjected to vacuum testing at 10^{-7} Torr and 260° F for determination of outgassing characteristics. Examination of the cool glass walls of the test tube for evidence of condensate, and weight loss measurements were the test criteria employed. One system utilizing a modified epoxy exhibited no weight loss or evidence of outgassing; two were marginal (light fog film produced on walls). Two samples of the surviving system were then subjected to 500 thermal cycles from 250°F to -225°F at a rate of thermal change of 20°F per minute. (The anticipated rate of thermal change for Syncom's entering and leaving the earth's penumbra is 10°F per minute; however, because of the restrictions of available calendar time, it was decided that twice as many cycles at 20° F per minute would provide more meaningful data.) At each temperature extreme, the samples were given a 10-minute stabilizing soak. Although the test was designed for a cold extreme of -320° F (liquid nitrogen), the -225° F limit was imposed by the radiative properties of the test equipment. A specification covering this adhesive system has been prepared.

A secondary aspect of this task was consideration of the process to be employed for actual production application of the cells to the substrates. The cells are preprocessed in strings three cells wide by 62 cells long and are designed for flat mounting. The bonding procedure selected is as follows. The substrate convex or outer face will consist of only two plies of glass fabric, not three, at the time of substrate fabrication. This third ply then becomes the cell adhesive "carrier" and is applied concurrently with the cells. This process offers several advantages:

- 1) Weight of cell to substrate adhesive is in effect eliminated since the adhesive serves

also as the impregnant for the third ply of the outer face.

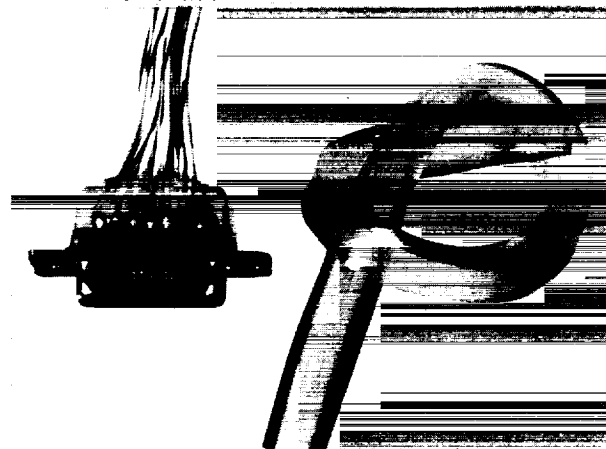
- 2) The glass fabric "carrier" serves not only as a structural element of the composite panel but also provides uniformity of bond-line thickness, important in assuring uniform stress distribution. The carrier also permits close control of adhesive content, precluding excess squeeze-out of adhesive between cells and onto cell surfaces which necessitates tedious cleanup. The glass fabric also provides a reinforced bondline, minimizing the effects of thermal and other environmental stresses on cells or substrate.

Flexible Potting Compound for Cable to Connector Joints. Eighteen material systems for low voltage cable to connector potting were evaluated in an effort to provide a flexible, easy-to-use insulating material that could be repaired with a minimum of effort. Three of these materials were found suitable for the Syncom environment; two were later rejected since they required heat curing and thus did not lend themselves to field repairing or simple processing.

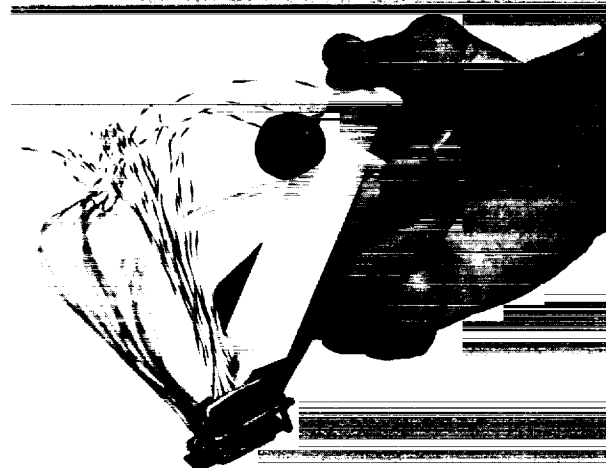
The material selected is a Hughes Materials Technology Department formulation utilizing a blend of two epoxy resins, one of which imparts flexibility to the polymer, and an amine cross-linking agent which provides room temperature curing capability.

The outstanding characteristics of the material (Figure 7-15) are as follows:

- Tough, flexible compound with good dielectric strength and superior handling resistance when compared to conventional elastomers or rigid epoxy materials.
- Easily mixed and applied by standard techniques currently being used. With the use of simple molds and a disposable syringe, individ-



a) Connector Potted with Material (Left) and Closeup of Material (Right)



b) Method of Applying Material

FIGURE 7-15. FLEXIBLE CONNECTOR POTTING MATERIAL

ual connectors can be potted on the bench or attached to a wiring harness within the spacecraft.

- Cure mechanism permits removal of potting shells from connectors after 12 hours at room temperature. Cure is complete within 36 hours.
- Repairability is enhanced because of transparency of the material permitting cutting directly to a questionable connection. Since the

7. Materials, Processes, and Components

material is very tough, cutting for repairs is performed with a knife-tipped soldering iron. When the repair is completed, the compound in a more viscous form can be smoothed into the cavity and permitted to cure. A specification covering this material has been prepared.

Phased Array Control Electronics Encapsulation. To achieve a lightweight, repairable system for encapsulation of a number of welded modules mounted on a conventional circuit board, a method has been developed incorporating advantages evolved from work performed on Syncom 2 and Surveyor. Essentially, this system utilizes conventional etched epoxy circuit boards to which welded electronic modules are to be soldered. Prior to mounting the modules, a thin epoxy laminate is bonded to the top of the etched board. This thin laminate has punched holes exposing the etched circuit islands, yet preventing electrical shorting between the conductors of the board and those in the modules. The laminate is bonded in place with a breakaway adhesive that facilitates removal of single modules for repair. Encapsulation is performed by placing properly proportioned amounts of polyether type foam-in-place compound into a polished mold, then placing the board and welded module assembly over the cavity and allowing the foam to cover the circuitry. Once the foam is hardened, the mold is removed. The foam material selected will cure at room temperature and is completely compatible with the Syncom environment. An example is shown in Figure 7-16.

Repairs are made by cutting through the foam on predetermined marked paths with a brass knife blade. Once the solder is removed from the board-to-lead joint, the entire board is heated to 200°F for 5 minutes and the individually isolated module is removed. Separation occurs at the interface of the etched board and the separator

laminate bonded with the breakaway adhesive. After a new module has been soldered to replace the old one, the cut-out section is refoamed as in the initial process. To protect the foamed unit while it is being handled prior to assembly, a plastic cover is used (Figure 7-17).

Thermal Control Surface Thermophysical Properties. Tests were made to determine the thermophysical (solar absorptance, infrared emittance) properties of various surfaces for passive thermal control. The data obtained is somewhat limited but still useful for the purpose of spacecraft thermal analysis. All specimens were measured for solar acceptance (α) and infrared emittance (ϵ) before and after ultraviolet-vacuum

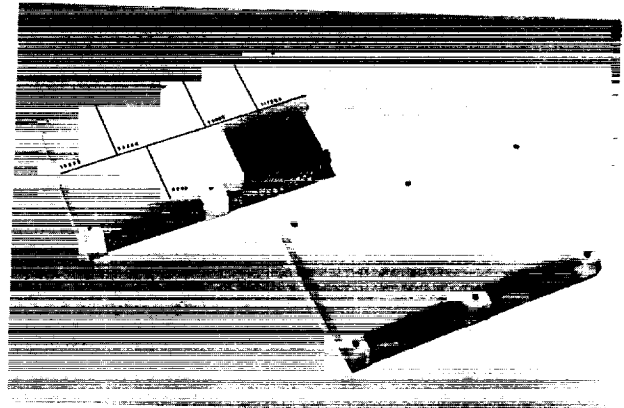


FIGURE 7-16. FOAM ENCAPSULATION

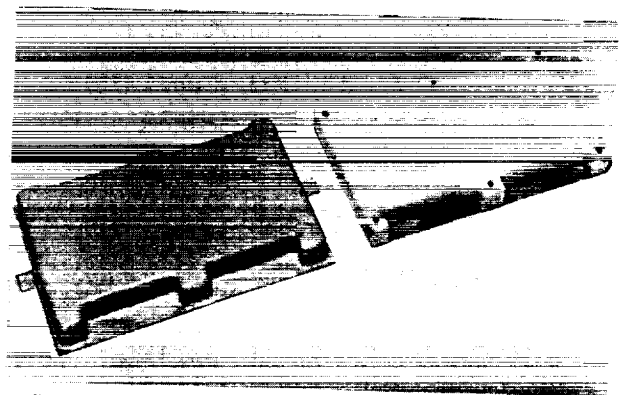


FIGURE 7-17. PLASTIC PROTECTIVE COVER FOR FOAMED MODULE

exposure at five times solar intensity. Although exposure time was short (approximately 250 solar equivalent hours), it was considered adequate to indicate any trend and rate of thermophysical degradation.

Vapor deposited aluminum surfaces 2500 Å thick have been studied. Three specimens were measured indicating initial values of $\alpha = .08$ and $\epsilon = 0.2$. One specimen for angular solar absorptance (α_s) was measured at incident angles of energy of 20 and 70 degrees for the full α_s spectrum. The results are shown in Table 7-4.

Organic White Paint. An evaluation has been made to determine binder-pigment combinations judged to be the most promising candidates for thermal-control paint. Previous work at Hughes and elsewhere in the industry has been on commercially available binder-pigment combinations not originally formulated for use in the space environment. Later polymer research at Hughes and elsewhere holds promise for high-purity solvent-washed binders which, when compounded with equally high-purity pigments, will provide organic coatings of highly acceptable long-time ultraviolet stability, particularly with

TABLE 7-4. REFLECTANCE OF VAPOR DEPOSITED ALUMINUM

| Wavelength,
microns | Specimen
Reflectance | Reference Blackbody
(Magnesium Oxide)
Reflectance | Specimen
Absolute
Reflectance |
|--|-------------------------|---|-------------------------------------|
| 20 DEGREE ANGLE OF SOLAR INCIDENCE | | | |
| 0.30 | 0.920 | 0.980 | 0.9016 |
| 0.41 | 0.955 | 0.985 | 0.9407 |
| 0.50 | 0.910 | 0.985 | 0.8964 |
| 0.60 | 0.875 | 0.980 | 0.8575 |
| 0.70 | 0.870 | 0.980 | 0.8526 |
| 0.81 | 0.850 | 0.970 | 0.8245 |
| 1.00 | 0.905 | 0.980 | 0.8869 |
| 1.54 | 0.977 | 0.975 | 0.9526 |
| 2.05 | 1.000 | 0.945 | 0.9450 |
| 3.39 | -0.950 | 0.920 | 0.9684 |
| | 44.530 | 48.850 | 43.6156 |
| Solar absorptance (α) = 0.128 | | | |
| 70 DEGREE ANGLE OF SOLAR INCIDENCE | | | |
| 0.373 | 0.870 | 0.960 | 0.8352 |
| 0.457 | 0.900 | 0.970 | 0.8730 |
| 0.522 | 0.910 | 0.970 | 0.8827 |
| 0.594 | 0.900 | 0.975 | 0.8775 |
| 0.694 | 0.900 | 0.980 | 0.8820 |
| 0.775 | 0.900 | 0.970 | 0.8730 |
| 0.908 | 0.920 | 0.980 | 0.9106 |
| 1.095 | 0.980 | 0.965 | 0.9457 |
| 1.387 | 0.990 | 0.955 | 0.9455 |
| 2.166 | -0.970 | 0.940 | 0.9691 |
| | 9.240 | 9.665 | 8.9852 |
| Solar absorptance = 0.101 | | | |

respect to solar absorptance—the most seriously affected property.

The best of the combinations to date is a titanium dioxide filled silicone-alkyd with an initial α of 0.22. After 250 solar equivalent hours of ultraviolet exposure, this value increased to $\alpha = 0.28$, compared to the best value reported in the literature or obtained at Hughes on other binder-pigment combinations of $\alpha \approx 0.45$ for the same exposure time. This result is sufficiently promising to warrant additional work.

Solar Cells. Cells of the type to be used on Advanced Syncom (30-mil Corning 7940 fused quartz bonded with Spectrolab ES-10 adhesive) are currently in test. Data will include following:

- 1) α and ϵ before and after 250 solar equivalent hours of ultraviolet irradiation
- 2) Angular α before and after irradiation

Since the solar array represents most of the external thermal control surface, these values are important for thermal analysis.

Traveling-Wave Tube Potting Compound. Tests have been made to determine the long-term thermal stability under vacuum conditions of the alumina-filled polyurethane compound used for potting the collector portion of the traveling-wave tube. This material was formulated for and used on the Syncom 2 program. Its properties include maximum thermal conductivity, compatibility to space environment at tube operating temperature, and sufficient flexibility to compensate for the differentials in expansion and contraction of the various materials within the tube. Thermal degradation of the compound would probably induce a decrease in thermal conductivity, causing tube operating temperature to rise, and a gradual loss of the polymer fractions contributing flexibility, which could in turn cause damaging stresses on components within the tube.

Thermal-vacuum aging tests were run for 6 months on six potted collector sections. The potted sections were held at a constant temperature of 204° F, with the collector bases held at 172° F. Polymer degradation was encountered in at least two of the sections, although no adverse effect was observed on the collector proper.

A parallel program was initiated to develop a compound having higher temperature capability and improved thermal stability with no sacrifice in flexibility. An alumina-filled silicone material met all requirements except for unacceptable outgassing characteristics. Although outgassing of condensable material may or may not pose a thermal control problem, in this case it would certainly result in ultimate severe polymer degradation because of the loss of these low-molecular-weight fractions. Therefore, consideration was given to the possibility of hermetically sealing the collector. This is currently in process together with thermal-vacuum testing to determine the efficacy of such a process. Tests will be completed and data made available by the end of the reporting period.

Improved Electrical Hookup Wire. An improved polyimide-coated FEP teflon-insulated hookup wire has been evaluated and placed under specification. It is available with either silver-plated or nickel-plated conductors. This wire has thinner insulation and improved processing for application of the polyimide coating.

Battery Leak-Sealing and Dielectric Coating. A process has been developed for precluding electrolyte leakage from battery terminals and and fill-tube pinchoff, and at the same time applying an electrical insulation coating to the entire battery case. A polyamide-epoxy resin system in solvent solution is employed for the dip-coating operation. This material is acceptable in the Syn-

com environment and will cure under ambient conditions.

Specifications. A specification covering the electron-beam welding process is being prepared and will be released by the end of the reporting period. A process specification covering point-to-point wiring and assembly procedures has been prepared and will be released before the end of the reporting period.

MATERIAL AND PROCESS STUDIES

3- to 5-Year Life Study. A study was undertaken to determine methods of accelerated testing which might permit prediction of 3- to 5-year behavior of materials with much shorter tests. The study, conducted in May and June, involved

accelerated testing of three types of thermal-control paint in exposure to different intensities of ultraviolet light in vacuum. Two paints — Skyspar and antimony trioxide in an inorganic, silicate-base binder — were known to degrade badly, by different mechanisms. The third — a ceramic pigment in a silicate binder — developed for use on the Surveyor program, was stable.

One of the less stable paints, with an organic base, could be used on the inside of the spacecraft where it would not be exposed to sunlight, while the stable white could be used on the outside. There would be a handling advantage, however, if an organic-base paint, more rugged than the inorganic-base paints, could also be used on the outside. Unless a stable organic combination could be found, however, the advantage would

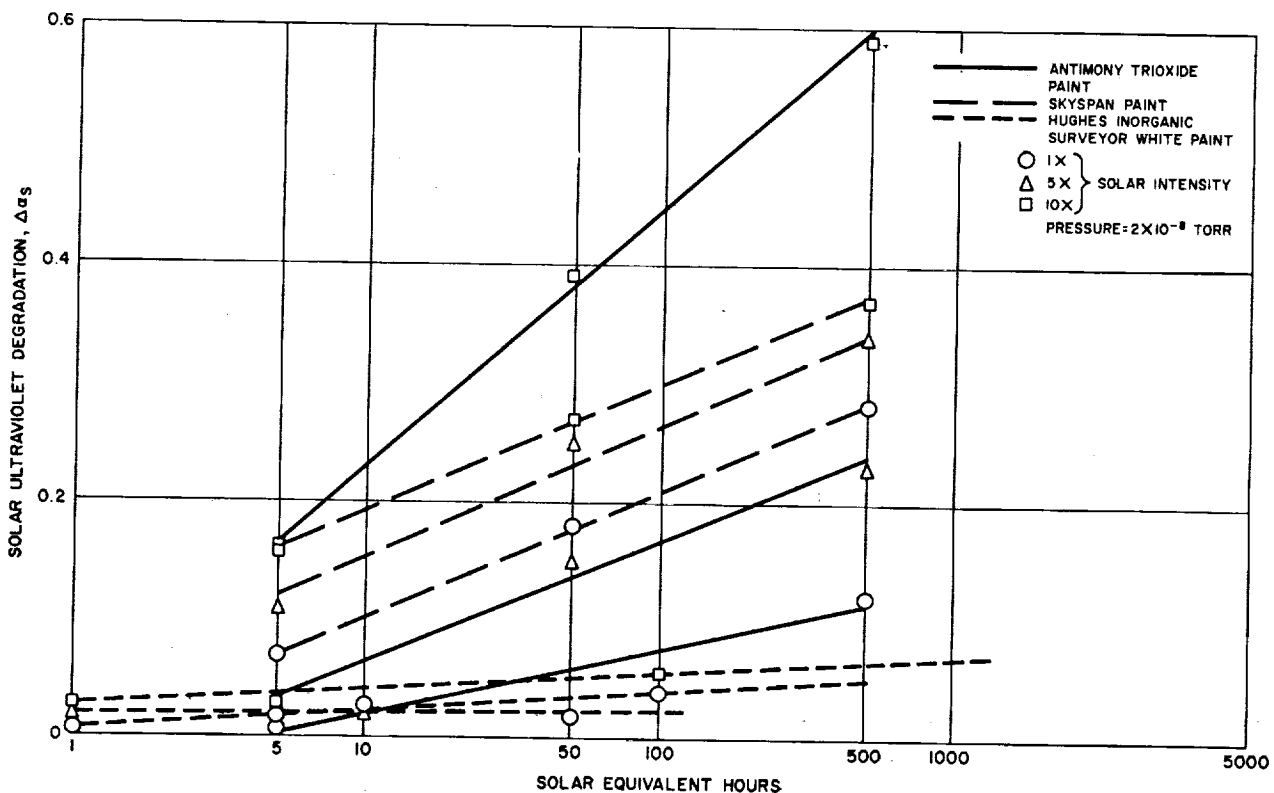


FIGURE 7-18. DEGRADATION OF THERMAL CONTROL PAINTS

7. Materials, Processes, and Components

be negated by the large change in solar absorptance when the vehicle is in orbit.

Twenty-seven tests were completed on these paints. Data from the tests are shown in Figure 7-18. The differences in rates and magnitudes of degradation are evident. In the case of the

Skyspar paint (A423, color SA9185), the degradation is believed to be primarily a function of the organic binder. In the antimony trioxide paint, the binder was an inorganic silicate, somewhat similar to that of the Hughes inorganic binder paint, and the degradation may be a function of the pigment.

8. GROUND SUPPORT EQUIPMENT

INTRODUCTION

This section covers work on system support equipment during the contract period. The work included 1) development of systems test plans, 2) development of test equipment to implement the system test plans, 3) development of major and minor control item test equipment, and 4) development of equipment for use at the ground control sites.

The system test plans written under this contract are documented in Volume 5, System Test Plans. Development of the test equipment is discussed in detail in the supplement to the September Monthly Report.

Discussed here are the two pieces of equipment developed for use at the ground site, the synchronous controller and the synchronous controller auxiliary and the major and minor control item test equipments.

SYNCHRONOUS CONTROLLER

The synchronous controller (Figure 8-1) provides the following major functions and equipment:

- 1) Digital frequency loops which are locked to the two telemetered sun sensor pulses (ψ , ψ_2) and circuitry to separate the two pulses.
- 2) Measurement of the spin period (ψ - ψ) and the orientation angle (ψ - ψ_2) utilizing the frequency locked loops of item 1.

- 3) Control of the transmission of execute pulses to the satellite at selectable spin phase angles for backup operation of the jet control electronics.
- 4) Control of the transmission of 1 to 511 execute pulses to adjust the phase of the antenna beam.
- 5) Circuitry to provide "pseudo ψ " pulses for eclipse operation of the phased array antenna.
- 6) Duration control of the execute tone for operation of the primary mode of the jet control system.
- 7) Controls and displays for operation of this equipment.

The synchronous controller is packaged in two units, one containing the execute control and the execute pulse phase selector subsystem and the other, the ψ and ψ_2 digital locked loop. The execute control provides controls for selecting the desired execute control mode and controlling the execute timing. The execute pulse selector subsystem contains the controls and circuitry to allow creation of an execute pulse at the desired angle relative to ψ , lasting for the proper duration. Controls are also available for adjusting the period and phase of pseudo ψ pulses used in eclipse mode operations. The second unit contains all of the controls for operating the ψ and ψ_2 frequency locked loops. An oscilloscope is available to aid the operator in adjusting the period of the frequency lock loops and phasing the execute pulses for backup operation of the spacecraft jets.



Execute Control Panel. A preliminary layout of the execute control panel is shown in Figure 8-2. Basically, the execute control unit provides the selection of one of the seven available execute modes. Controls are also provided for initiating and controlling the execute duration. The function of the various selector switches, pushbutton switches, and displays are described below.

Execute Mode Selector. The execute mode selector switch permits the selection of the following execute modes:

- 1) Execute and clear spacecraft decoder
- 2) Execute and hold (no spacecraft decoder clear)
- 3) Continuous execute
- 4) Normal control system mode for primary control of the spacecraft control system
- 5) Real time mode for backup of the normal control system mode
- 6) Antenna position mode to program a selectable number of execute pulses for repositioning the spacecraft phased array beam
- 7) Eclipse mode to generate pseudo ψ pulses for operation of the phased array beam during eclipse

- **Execute/Clear.** A single execute pulse is generated upon depressing the "execute" switch. At the termination of the execute pulse, a sufficient number of zero tones are transmitted for clearing the spacecraft command register.

- **Execute/Hold.** A single execute pulse is generated upon depressing the "execute" switch. At the termination of the execute pulse, the transmission of zero tones is inhibited, permitting the command to remain in the spacecraft command register.

- **Continuous Execute.** The execute duration is controlled by depressing and holding the "execute" switch. The execute signal is terminated upon releasing the "execute" switch. No spacecraft decoder clear signal is generated upon releasing the switch.

- **Normal Control System.** The execute signal is provided for control of the spacecraft primary jet control system. The "spin period/pulses" switch is not effective in this mode since only a pulsed execute signal is generated. The execute pulse duration is equal to the spin period and is generated synchronously with reception of a telemetered ψ -pulse ($\psi=0$), and counted.

The number of execute pulses is preselected by the setting of the decimal thumb wheel digit-switch. The binary-coded-decimal "pulses to go" readout displays the selected number of execute pulses.

Two methods of initiating execute are available and are selectable by means of the alternate action "manual/start on GMT" switch. Selection of "manual" and depressing the "execute" switch permits this mode to be initiated manually. Selection of "start on GMT" permits the execute to be initiated at a specified Greenwich Meridian time. In this case, the starting time is selected by means of the minutes and seconds selector switches. The "execute" switch must be depressed and held until the execute has been initiated at the selected time.

- **Real Time Jet Control Electronics Backup Mode.** The execute signals are provided for real time backup of the spacecraft jet control electronics. The "spin periods/pulses" switch permits the option of selecting a continuous execute signal for a specified number of spacecraft spin periods or a pulsed execute at a selected angle for a given number of spacecraft spin periods.

8. Ground Support Equipment

If the pulsed execute mode is selected, the execute pulses are generated at a selectable phase angle with respect to the occurrence of the ψ sun sensor pulse in the spacecraft. The desired phase angle, plus a correction factor which takes into account the propagation time to the spacecraft, is selectable in 1 degree increments by means of the ψ phase selector switches. The method of specifying the desired number of spin periods or

pulses and initiating the execute either manually or at a selected GMT is similar to that of the normal control system.

- Position Antenna. This mode permits the transmission of 1 to 511 execute pulses for adjusting the phase of the antenna beam. Each pulse causes approximately 0.703 degree of beam rotation.

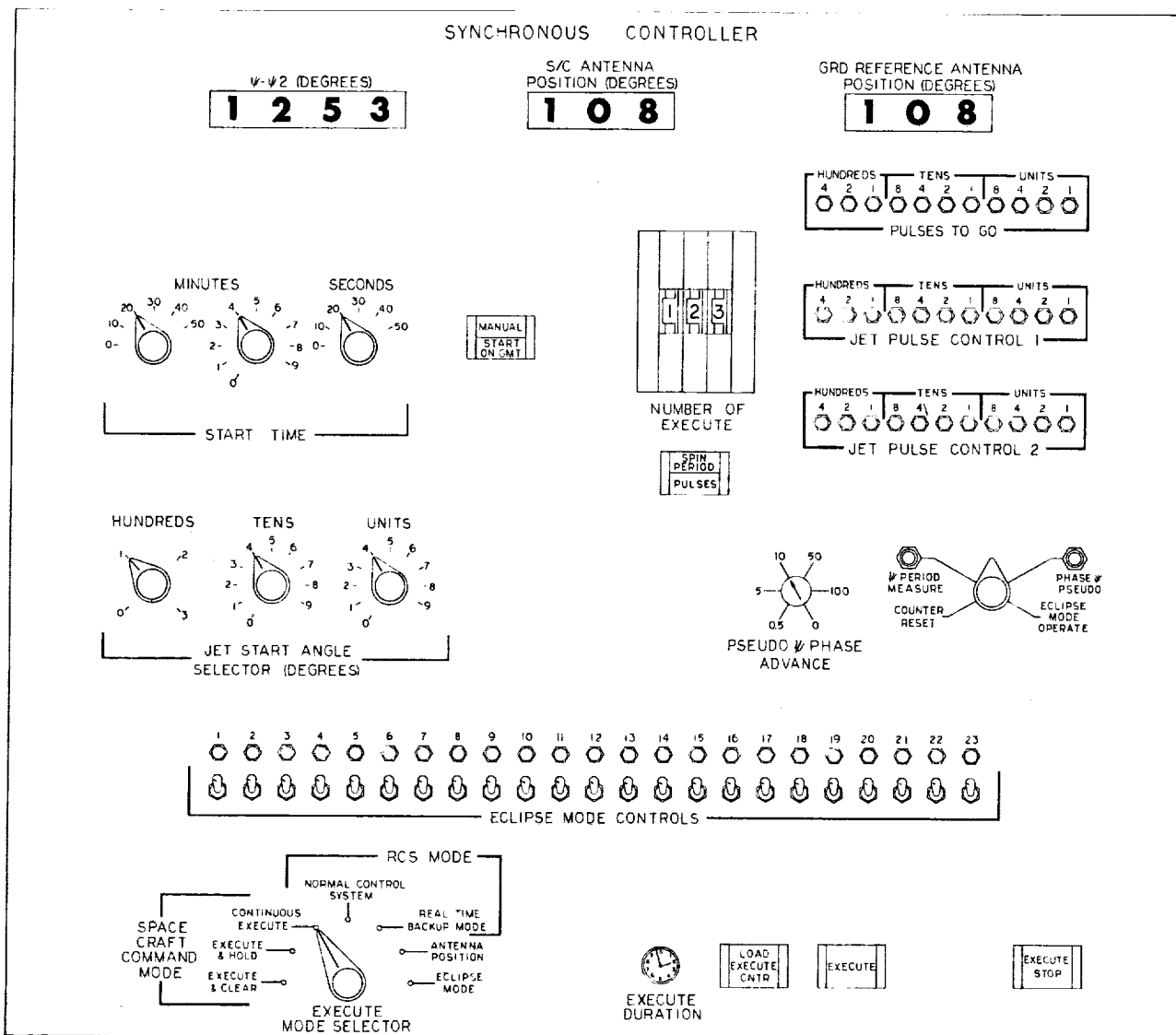


FIGURE 8-2. SYNCHRONOUS CONTROLLER EXECUTE CONTROL UNIT

The desired number of antenna position increments is selectable by means of the decimal digiswitch. The method of initiating the execute is similar to that of the normal control system and real time jet control electronics backup mode. The execute duration is automatically generated in this mode, consisting of a 128-cycle pulse train with a 50-percent duty cycle.

- **Eclipse Mode.** Pseudo ψ pulses are transmitted for eclipse operation of the spacecraft phased array antenna. Prior to initiating eclipse mode operation, pseudo ψ pulses are transmitted to permit phasing. When the transponded execute pulse (pseudo ψ) period and phase are identical to that of the telemetered ψ pulse, the operator can initiate eclipse mode operation.

Depressing the "execute" switch permits the command, "eclipse mode operation on" to be transmitted following termination of the first pseudo ψ pulse. The next pseudo ψ pulse executes the command in the spacecraft command register permitting eclipse mode operation to commence.

Execute Duration. The "execute duration" is a calibrated potentiometer for controlling the duration of the execute pulse. Control is effective when the following execute modes have been selected and initiated:

- 1) Execute/clear
- 2) Execute/hold

The execute duration is variable from 20 to 200 milliseconds with a detent for a 50-millisecond duration.

Load Execute Counter. Depressing this momentary action switch permits the number selected on the digiswitch to be loaded into the BCD execute counter. This switch is effective only for the normal control system, real time backup, and antenna position modes.

Spin Period/Pulses. This alternate action switch is only effective when the real time jet control electronics backup mode has been selected. Selection of spin periods permits a continuous execute signal to be generated for a specified number of spacecraft revolutions. Selection of pulses permits an execute pulse to be generated at a given phase angle with respect to the occurrence of the ψ pulse in the spacecraft. The desired number of execute pulses or continuous spin periods is preselected by the setting of the decimal thumbwheel digiswitch.

Decimal Thumbwheel Digiswitch. This switch permits the selection of 1 to 511 execute pulses when either the normal control system, real time jet control electronics backup, or antenna position mode has been selected. The option is also available, when in the real time jet control electronics backup mode, for selecting a continuous execute for 1 to 511 spacecraft spin periods.

Manual/Start on GMT. This switch permits any execute to be initiated manually or at a specified Greenwich Meridian time when either the normal control system, real time, jet control electronics backup, or antenna position mode has been selected. This mode operates in conjunction with an external GMT clock.

Execute Stop. This alternate action switch is effective only when either the normal control system, real time backup, or antenna position has been selected. Depressing the "execute stop" switch permits countdown of the execute counter to be inhibited. The BCD "pulses to go" readout displays the remaining count in the execute counter. If either the normal control system or real time backup mode has been selected, jet pulse counters 1 and 2 will display the number of transponded executes from the spacecraft jet system. Depressing the "execute stop" switch a

8. Ground Support Equipment

second time permits countdown of execute pulses to resume.

Execute. Depending on the execute mode, this switch initiates various types of executes. The types of execute signal for the associated execute modes are as follows:

- 1) Single pulse, 20 to 200 milliseconds duration, for either execute/clear or execute/hold execute modes.
- 2) Continuous execute signal when in the continuous execute mode and the execute switch is held depressed.
- 3) Continuous execute signal for a specified number of spacecraft revolutions when in the real time jet control electronics backup mode and the "spin periods/pulses" switch is on "spin periods."
- 4) Multiple pulsed modes (normal control system, real time jet electronics backup mode, and position antenna).

Binary Coded Decimal Displays.

- **Pulses To Go.** The pulses to go readout displays the number of execute pulses/spin periods remaining to be executed in the following modes:

- 1) Normal control system and antenna position modes (pulsed modes)
- 2) Real time jet control electronics backup mode (spin periods/pulses)

- **Jet Pulse Control 1 and 2.** The jet pulse control readouts display the number of transponded executes from the spacecraft jet system when either the normal control system or real time jet control electronics backup mode has been selected.

- **ψ - ψ^2 .** This readout displays the ψ - ψ^2 angle in degrees as decoded from the telemetry.

- **Spacecraft Antenna Position.** This readout displays the telemetered value of the spacecraft antenna beam position in degrees. The display is updated during each commutation cycle (approximately 2.5 seconds) of the telemetry encoder.

- **Ground Reference Antenna Position.** This readout will display the ground computed antenna position. The display will be manually updated when antenna position increments are transmitted to the spacecraft upon executing "accept antenna position" command. A correction factor is automatically added to the existing count approximately every 2.81 minutes.

Phase Selector Panel. The ψ phase selector panel controls and displays provide the following major functions:

- 1) Selection of a trigger pulse at 1-degree increments with respect to the telemetered pulse
- 2) Eclipse mode oscillator controls for providing pseudo ψ pulses for eclipse operation of the spacecraft phased-array beam

The ψ phase control can be used to generate an execute pulse in 1-degree increments relative to ψ for operation of control system in the backup mode. The desired ψ pulse is selected on the set of three switches, indicated as "Psi Phase" in Figure 8-2.

ψ Tracker Panel Description. Figure 8-3 is a preliminary drawing of the ψ tracker front panel. Figure 8-1 (top left) is a diagram of the unit. The tracker panel controls and displays provide the following major functions:

- 1) Control for initial setup and operation of the ψ and ψ_2 frequency lock loops

- 2) Displays for indicating the locked/unlocked conditions of these frequency lock loops

Frequency Lock Loop Controls and Displays. There is an identical set of controls and displays associated with each of the frequency lock loops. Since the function of the controls and displays is the same for each frequency lock loop, only one set is described.

- **Input Amplifier Balance.** The input amplifier balance potentiometer permits adjustment of the dc output level from the input amplifier.

- **Voltage-Controlled Oscillator.** The voltage-controlled oscillator helipot provides the control for locking the frequency lock loop to the incoming frequency rate of the ψ or ψ_2 pulses. Varying the helipot permits control of the rate and direction of precession of the input pulse (modified ψ or ψ_2) relative to the gate pulses. The loop is locked when the input pulse occurs within the gating interval and remains there.

- **Frequency Range Switch.** The frequency range toggle switch permits the selection of course frequency ranges, high or low, from the voltage-controlled oscillator.

- **Frequency Lock Loop Displays.** Each frequency lock loop employs two types of displays for indicating status of the frequency lock loops. The indicator displays the status of the reference frequency lock loop as "locked" or "unlocked." The meters associated with each of the frequency lock loops provide a visual indication of the stability of the locked loop. A stable locked loop is indicated by pulsations close to full scale deflection, a loop that is "pulling" or near breaking lock is indicated by deflections on the lower portion of the scale.

- **Frequency Lock Loop Controls.** The potentiometers and toggle switches for preliminary adjustment of the frequency lock loops are located under a panel flap. Assuming the preliminary adjustment procedure has been completed, the only precaution during operation is to leave the "gate" switch ON and the integrator switches at OPERATE.

Operation of Eclipse Mode Circuitry. The eclipse mode circuits in the synchronous controller are illustrated in Figure 8-4. A four-step sequence, controlled by a four-position three-pole selector switch (S1) prepares the system for eclipse mode operation:

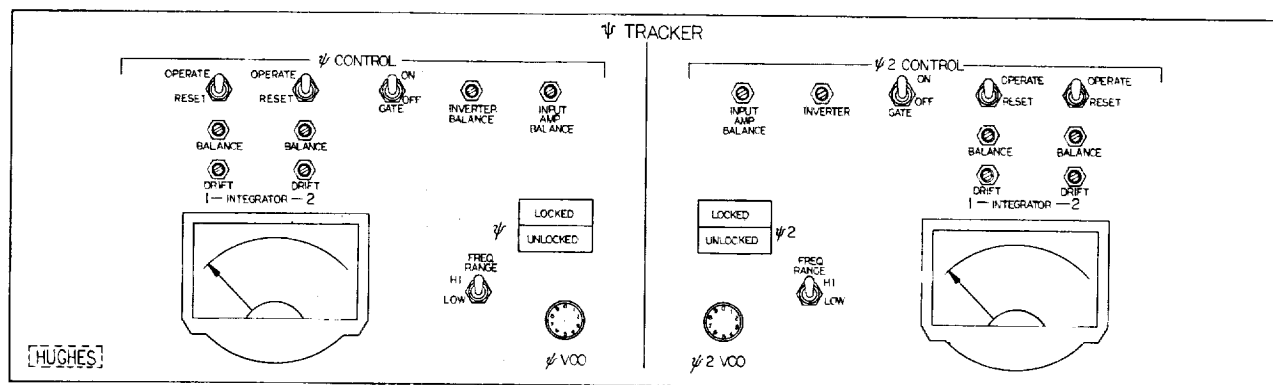


FIGURE 8-3. ψ TRACKER FRONT PANEL

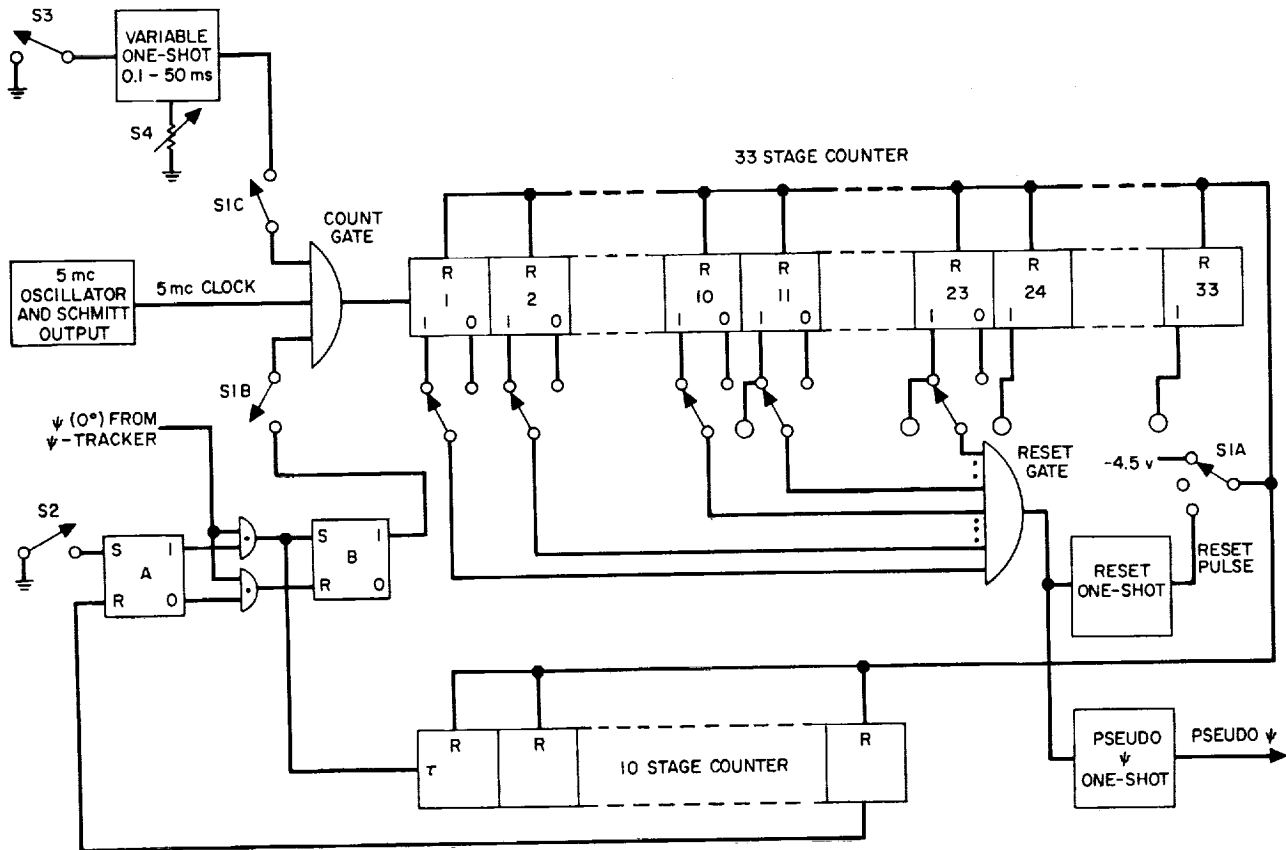


FIGURE 8-4. ECLIPSE MODE CIRCUITS

- 1) Reset:
 - S1A - -4.5 volts
 - S1B - closed
 - S1C - open

-4.5 volts applied through S1A resets all counter flip-flops.
- 2) ψ period measurement:
 - S1A - open
 - S1B - closed
 - S1C - open

Pushbutton S2 is actuated and sets flip-flop-A. The next $\psi(0)$ pulse from the ψ tracker sets flip-flop-B, while it and succeeding $\psi(0)$ pulses are counted in the 10-stage counter. With flip-flop-B set, the count gate passes the 5-mc clock, which is counted in the 33-stage counter.

Following the 1024th $\psi(0)$ pulse, an output of the 10-stage counter resets flip-flop-A, and the next (1025th) $\psi(0)$ pulse resets flip-flop-B. The count gate has then passed the 5-mc clock for 1024 ψ periods, and the 33-stage counter holds a count of the number of 5-mc clocks in this interval. Its 23 most significant stages (which are equipped with binary readout lamps) therefore hold a count of the number of 5-mc clocks in one ψ period. Using the binary toggle switches provided, the same count is selected from the 33-stage counter's 23 least significant stages to drive the reset gate, so the reset one-shot will be triggered each time that count is reached.

- 3) Pseudo ψ pulse phasing: S1A - reset pulse
S1B - open
S1C - closed

In its stable state, the variable one-shot enables the count gate to pass the 5-mc clock, which in turn drives the 33-stage counter. As a result of the previous step, the reset one-shot is triggered each time the count reaches the number of 5-mc clocks in one ψ period, and since the reset pulse it generates resets the counter before the next clock, the counter cycle time is exactly one ψ period. The reset gate which triggers the reset one-shot also triggers the pseudo ψ one-shot, so that a pseudo ψ pulse is also generated each ψ period.

The pseudo ψ pulse causes an execute tone to be transponded via the telemetry link; this transponded execute tone is displayed on a 'scope together with the ψ output of the telemetry receiver. Phasing the pseudo ψ pulse consists of making the displayed pulse trains time-coincident, which means in effect that a pseudo ψ pulse arrives at the spacecraft just as a sun sensor pulse is generated. Phasing is accomplished by repeated actuations of pushbutton S3 which trigger the variable one-shot. The resulting pulses, of 0.1 to 50 milliseconds duration depending on the setting of the variable timing resistance (S4), disable the count gate so that no clocks are passed in that interval. Succeeding pseudo ψ pulses are thereby delayed by the sum of these pulse lengths. In practice, when the transponded pseudo ψ and received ψ pulse trains are initially far out of phase, a few actuations of S3 with S4 set to 50 milliseconds will bring them

close; then the setting of S4 will be gradually reduced as further actuations of S3 are used to "zero in." There should be no problem in achieving coincidence within ± 0.1 millisecond, which at the greatest spin rate equals ± 0.09 degree.

- 4) Eclipse mode operate: S1A - reset pulse
S1B - open
S1C - open

Upon completion of the foregoing steps, the eclipse mode circuitry will be producing correctly timed and phased pseudo ψ pulses, and the eclipse mode operation ON command may be transmitted and executed. The spacecraft will then accept the pseudo ψ pulse train as its space reference for phased-array beam positioning. This fourth position of S1 is provided so that the ψ period measuring circuits and pseudo ψ phasing circuits are disconnected during the eclipse, and inadvertent actuation of S2 or S3 will cause no harm.

Synchronous Controller Circuits. Several circuits will be required to implement the synchronous controller functions that are not available on VATE circuit cards. Most occur in the digital frequency lock loops and in the eclipse mode portion of the unit. Some of the eclipse mode circuits have been designed and breadboarded and are presently being tested, including:

- 1) 5 mc crystal controlled oscillator
- 2) Counter flip-flop capable of 5 mc operation with a fan-in and fan-out of 23
- 3) 75 nanosecond one-shot with sufficient drive to base-reset 23 counter stages

These circuits are used in the counter loop that synthesizes a pseudo ψ pulse train for eclipse mode operation.

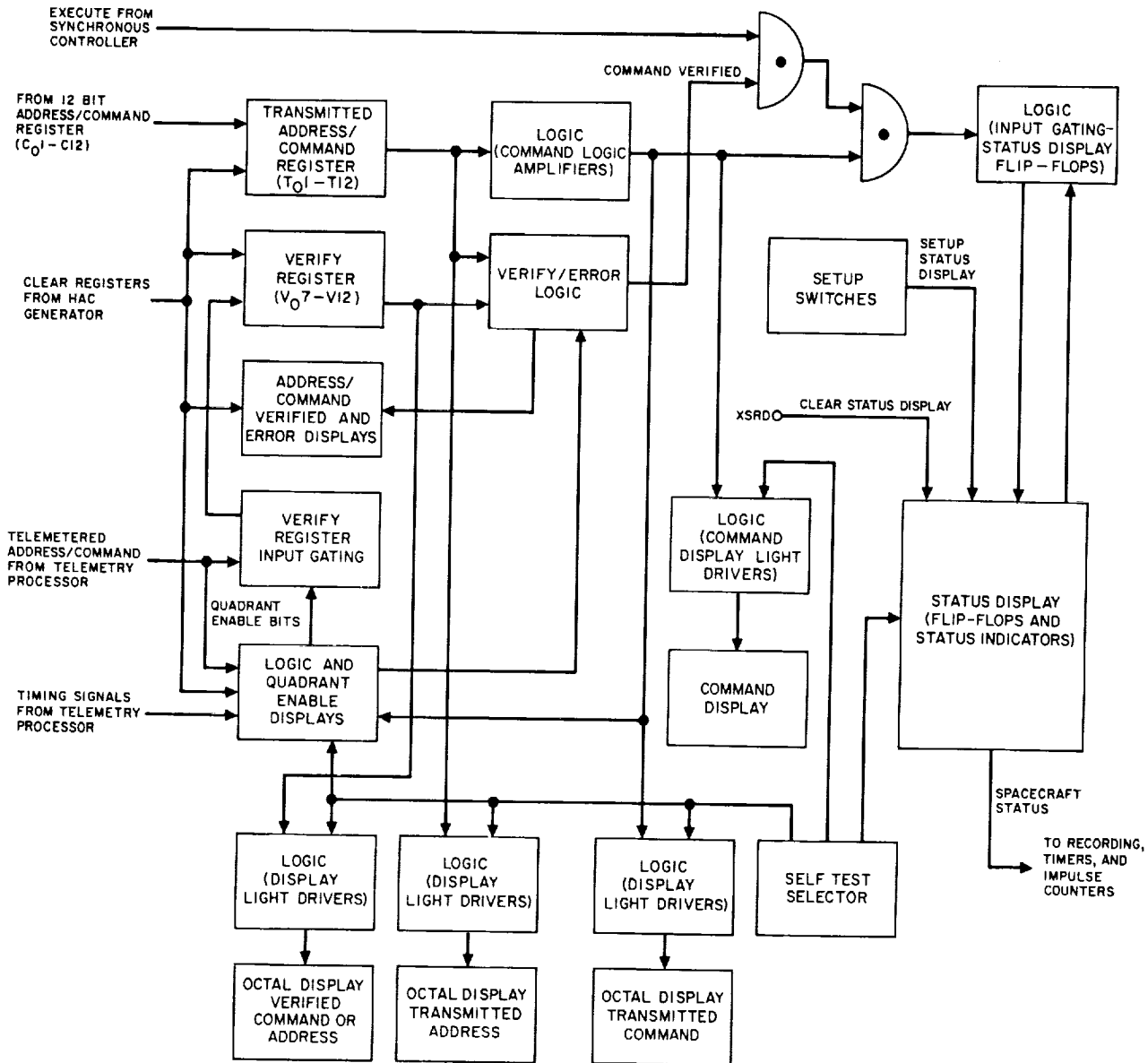


FIGURE 8-5. SYNCHRONOUS CONTROLLER AUXILIARY

SYNCHRONOUS CONTROLLER AUXILIARY

The synchronous controller auxiliary (Figure 8-5) is used in conjunction with an OGO command generator to operate the command subsystem of an Advanced Syncom spacecraft. Figure 8-6 illustrates the front panel of this unit. The function of the synchronous controller auxiliary is to display:

- 1) Command loaded into the command generator (title)
- 2) Transmitted address and command (modified octal)
- 3) Verified address or command (modified octal)
- 4) Address verification

- 5) Command verification
- 6) Quadrant enabled
- 7) Spacecraft status

Displays

- **Selected Command.** This display shows the title of the command loaded into the command generator. The titles will normally appear white against a black background, except in the case of the irreversible commands when the titles will appear red against a black background.

- **Transmitted Address and Command.** This display shows the modified octal representation of the address and command entered into the command generator.

- **Verified Address or Command.** This display will show the contents of the addressed quadrant's command register in modified octal during normal operation. It also has the capability of displaying the contents of any one of the other three spacecraft command registers when manually selected with the corresponding quadrant enabled switch.

- **Address Verification.** This display will show "verified" or "error" when the addressed spacecraft quadrant corresponds or does not correspond to the telemetered enabled quadrant. This display will function only between transmission of the command and transmission of the execute.

- **Command Verification.** This display will show if the telemetered contents of the spacecraft command register correspond to the command transmitted to the spacecraft. If they correspond, "verified" will light; if not, "error" will light. This display will function only between transmission of the command and transmission of the execute tone.

- **Quadrant Enabled.** This display will show which spacecraft quadrant(s) is (are) enabled. It consists of four backlighted micro-switches and will perform a secondary function as described below under *Controls*.

- **Spacecraft Status.** This display shows the current status of the following subsystems on board the spacecraft:

- 1) Telemetry on (quadrants 1, 2, 3, 4)
- 2) Traveling-wave tube 1 filament on (quadrants 1, 2, 3, 4)
- 3) Traveling-wave tube 2 filament on (quadrants 1, 2, 3, 4)
- 4) Traveling-wave tube high voltage on (quadrants 1, 2, 3, 4)
- 5) Multiple access mode on (quadrants 1, 2, 3, 4)
- 6) Frequency translation mode on (quadrants 1, 2, 3, 4)
- 7) PACE on (quadrants 1, 2, 3, 4)
- 8) Phase shifter driver on

Controls

- **Quadrant Enabled.** These backlighted switches provide a secondary function, by allowing the operator to display the contents of any spacecraft command register on the "verified address/command" display.

- **Clear Status.** This switch allows the status display to be completely cleared.

- **Status Preset.** These switches, one rotary and four pushbuttons, are used to set up initial status display conditions. The status item is first selected on the rotary switch, then the desired quadrants are lit by pressing the proper quadrant buttons.

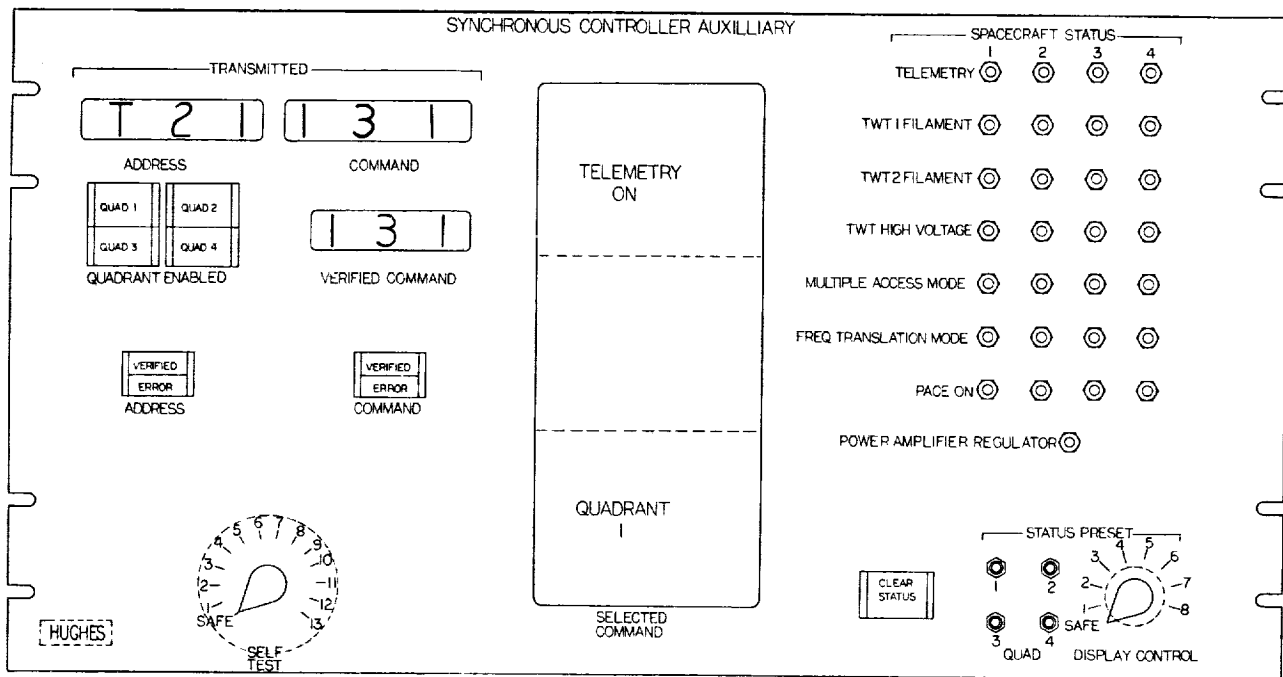


FIGURE 8-6. SYNCHRONOUS CONTROLLER AUXILIARY FRONT PANEL

• **Self-Test.** This switch will allow a complete test of the projection displays and all nonprojection-type indicators. As the switch is rotated through the first 12 positions, the projection displays are sequentially lit. In the last position, power is applied to all nonprojection indicators.

TRANSPONDER RF TEST SET

The breadboard transponder RF test set is in the final stages of assembly. A prereleased set of schematics and an advanced bill of materials have been prepared.

The breadboard configuration of this unit consists of three panels, each 7 inches high, plus four Huggins traveling-wave tubes, 51¼ inches high. A total of 42 inches of 19-inch rack space is required for each transponder test set. Four sets will be required to test one complete four-quadrant spacecraft. Maximum utilization

of existing commercial equipment and previously designed subassemblies resulted in a modular construction that is almost identical to the block diagram (Figure 8-7).

Receivers. Two basic drawings are required to combine all the schematic drawings in the test set:

- 1) Interconnections for receivers and multiple access transmitters (Figure 8-8)
- 2) Schematic — frequency translation transmitter (Figure 8-9)

These drawings show the interconnections between the modules and list the drawing, or part numbers, for each module. A common receiver is used for both modes and is shown on the drawing with the multiple access transmitter.

The receiver input signal is connected to J1 and passes through a series of variable attenuators, filters, traveling-wave tube amplifiers, iso-

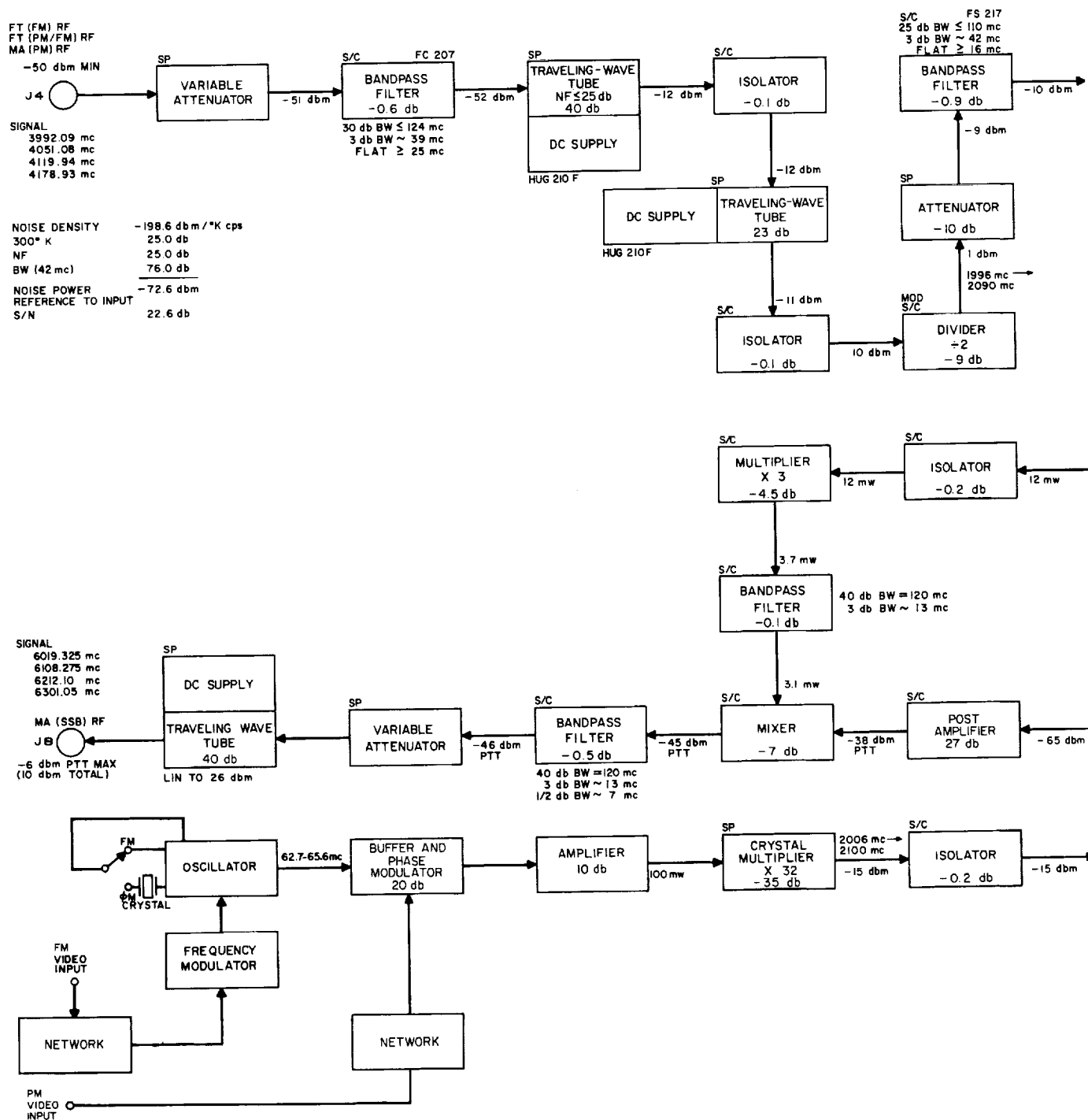
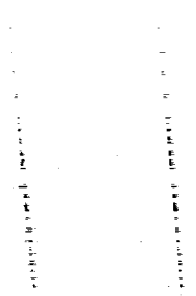
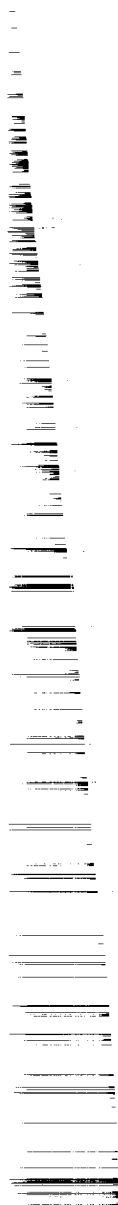
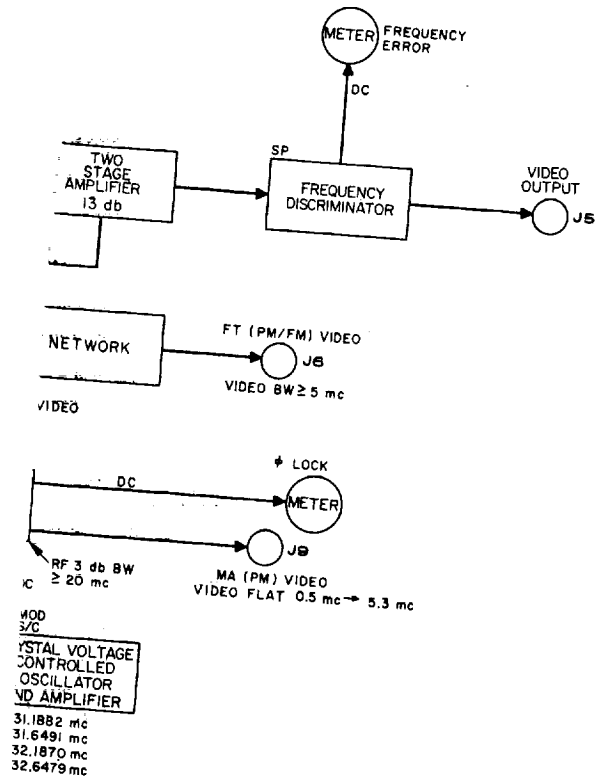


FIGURE 8-7. TRANSPONDER RF TEST SET BLOCK DIAGRAM



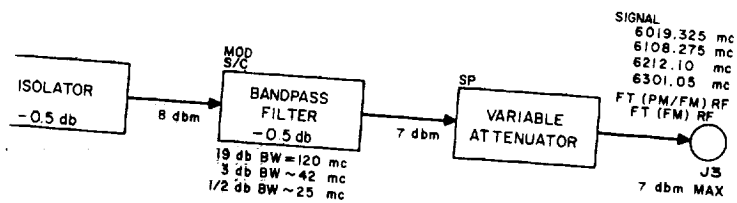




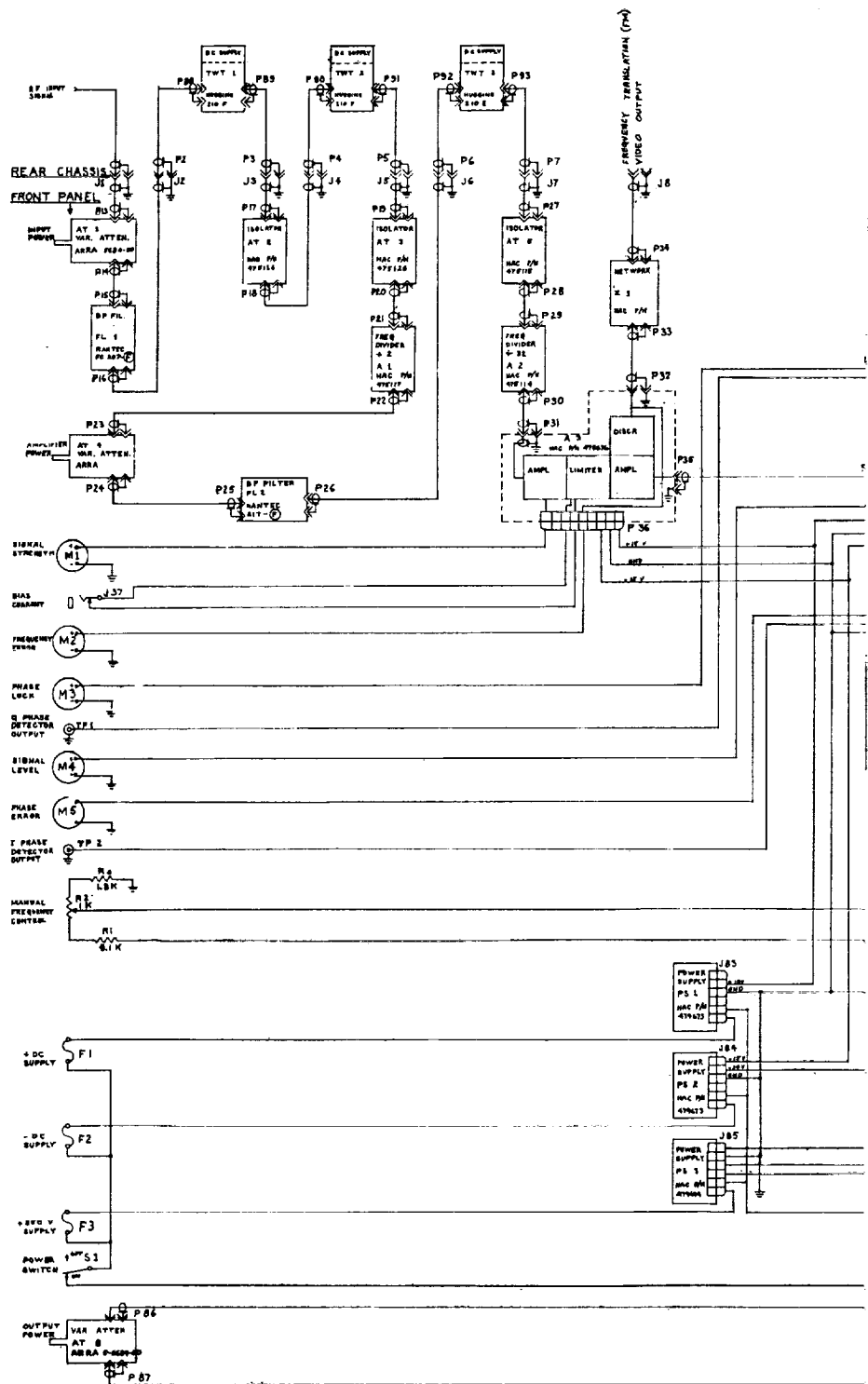
MA (SSB) VIDEO

J7

VIDEO FLAT 0.5 mc $\rightarrow$ 5.3 mc



3



NOTE:

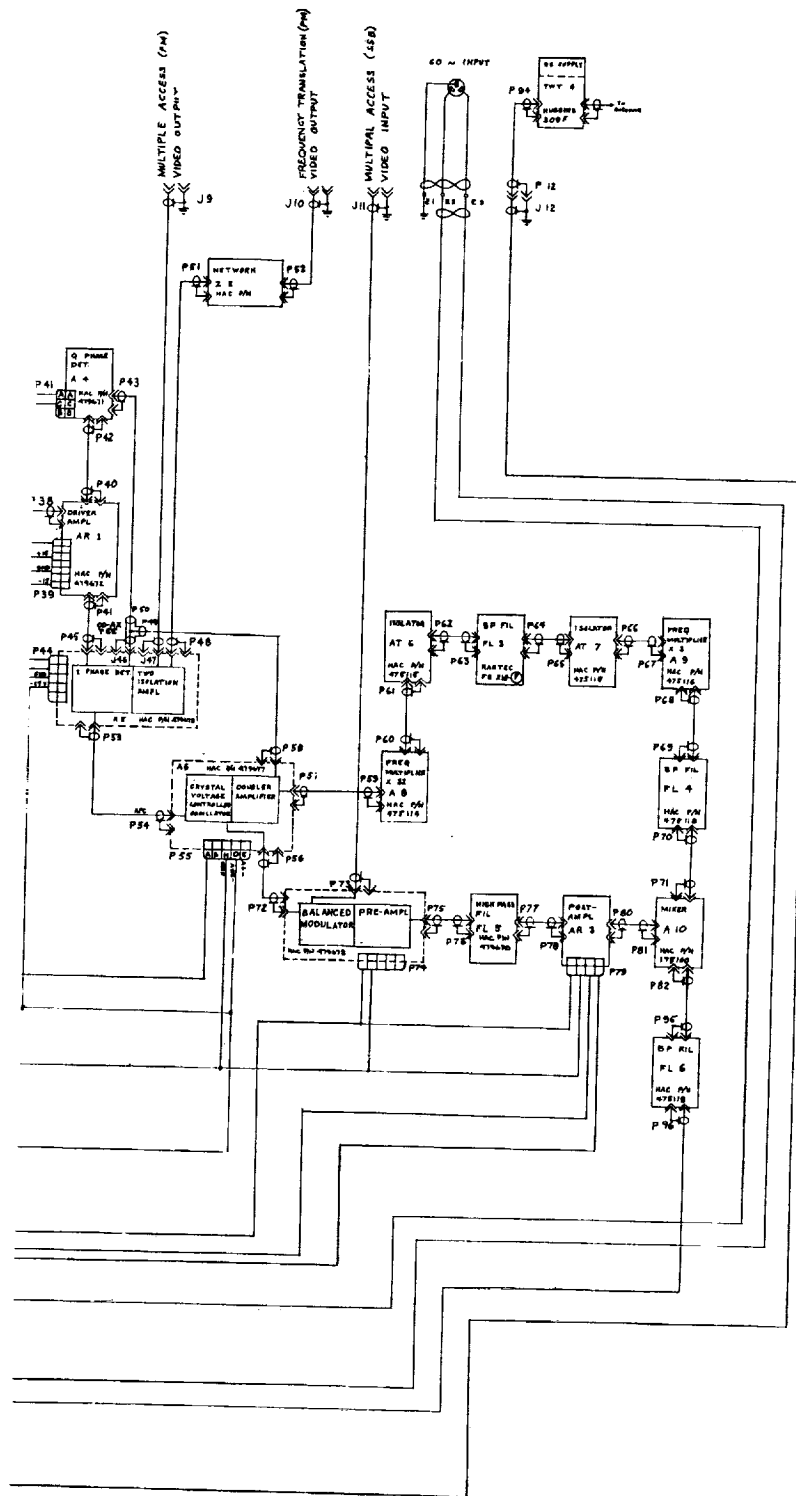
FIGURE 8-8. TRANSPONDER TEST RECEIVERS AND MULTIPLE ACCESS

1. The first part of the document is a list of names and titles, including "The Hon. Mr. Justice" and "The Hon. Mr. Justice".

2.

1. The first part of the document is a list of names and titles, including "The Hon. Mr. Justice" and "The Hon. Mr. Justice".

2.



① - DASH NUMBER INDICATES DIFFERENT FREQUENCIES

TRANSMITTER

2

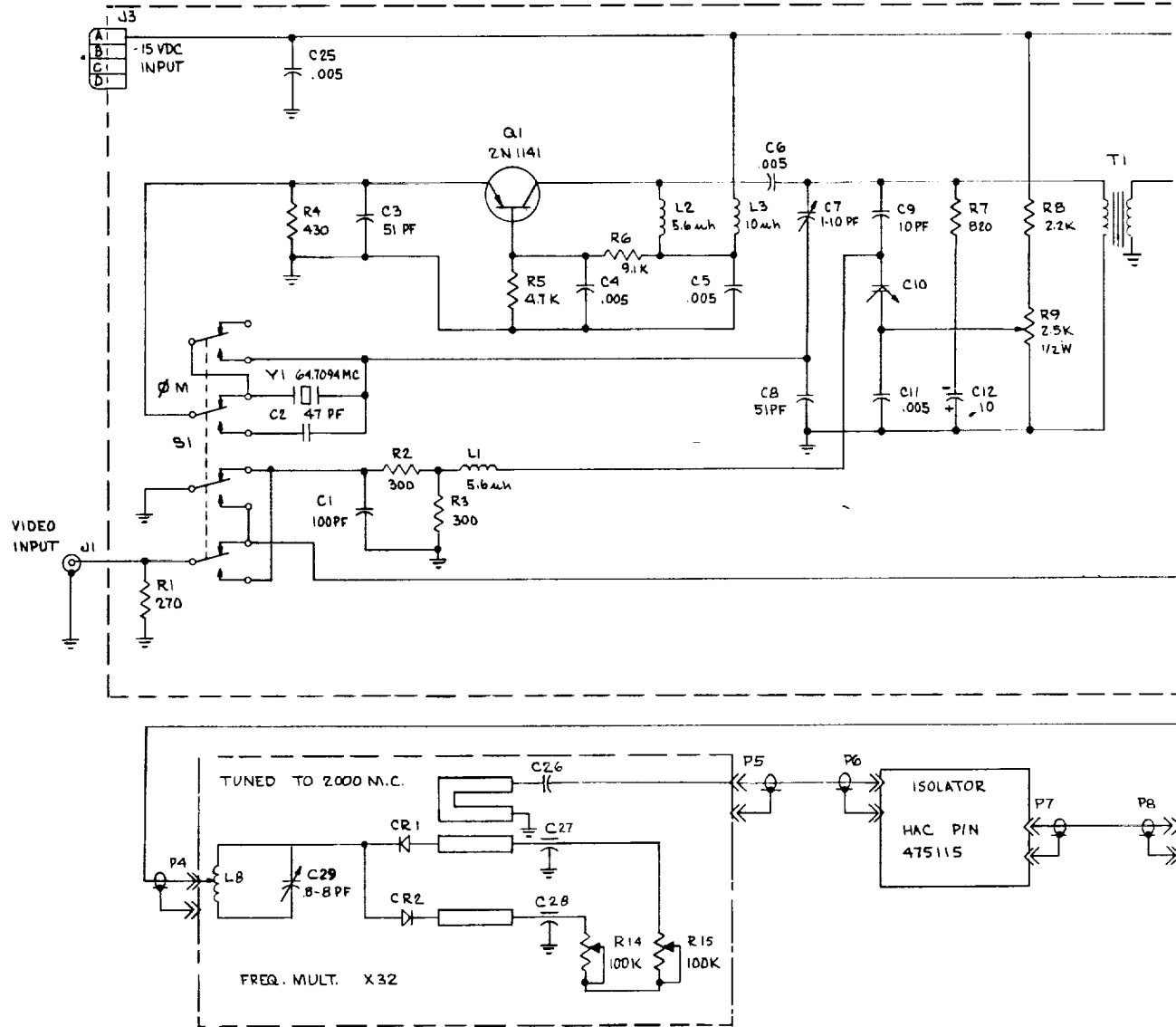


FIGURE 8-9. TRANSPONDER TEST SET FREQUENCY TRANSLATION TRANSMITTER



lators, and frequency dividers until it arrives at the 64-mc limiter and discriminator. During this process, three significant events occur:

- 1) The signal is amplified. The incoming signal may be as low as -50 dbm. The variable attenuators (AT1 and AT4), operating in conjunction with the three pre-adjusted and calibrated traveling-wave tube amplifiers (TWT 1, 2, and 3) supply a 4-milliwatt signal to the 64-mc limiter and discriminator module.
- 2) The frequency range utilized by the spacecraft under test is selected by the pre-tuned bandpass filters (FL 1 and FL 2), which reject all other frequencies. These filters are also used in the spacecraft.
- 3) The incoming signal frequency is reduced from 4 kmc to 64 mc by dividing the frequency first by 2, in A1, and then by 32, in A2. The exact frequency is obtained by dividing the exact incoming carrier frequency by 64. The 64-mc limiter and discriminator are adjusted to the specific frequency for the transponder under test.

The two dividers, A1 and A2, are the same as the multipliers used in the spacecraft transponders. A multiplier may be used as a divider by reversing the input and output connections, as discussed in the August 1963 Advanced Syncom Monthly Progress Report, Section 9, *New Technology*.

The RF signal, thus selected, amplified, and frequency divided down to 64 mc, is connected into the 64-mc limiter and discriminator module. This module performs four functions.

The first function is fulfilled by a two-stage amplifier whose outputs are used to drive the front panel signal strength meter (M-1) and to

drive the limiter. The limiter action is obtained from four diodes (CR2, CR3, CR4, and CR5). The "limited" signal is then amplified again by a two-stage amplifier and split into two parts. One part is passed through an emitter follower, Q8, to the phase detector amplifier. The second signal drives the discriminator. The discriminator output passes through an emitter follower to the video output jack, J32. After passing through an impedance matching network, Z1, this signal is the frequency translation video output (for frequency-modulated signals) at J8.

The driver amplifier input is passed through an emitter follower and two stages of amplification, and the output is again split into two parts. Both parts are passed through emitter followers and are connected to the two phase detectors at P42 and P45. As a secondary function, a sample of the two emitter followers signal is taken to drive the front signal level meter.

The output of the Q phase detector is displayed on panel meter M-3 and indicates the phase error between the received signal and the reference signal derived from the crystal voltage-controlled oscillator and double amplifier. The same circuitry is utilized in the I phase detector and isolation amplifier to produce an AFC voltage which controls the crystal voltage-controlled oscillator frequency.

The instantaneous deviation in the 64-mc signal (the phase modulation of the signal), is detected by the same circuit. The ac component of the detected output is the video signal. The video is split into two parts and amplified by the isolation amplifier. One of the isolation amplifier's output is connected to the phase-modulated multiple access video output jack, J9. The other output — the phase modulation frequency translation video — is connected through a compensation network to J10.

8. Ground Support Equipment

A vital part of the phase detector is the frequency servo loop formed by parts of the I phase detector and the crystal voltage-controlled amplifier. The AFC voltage developed in the I phase detector for controlling the frequency of the crystal voltage-controlled oscillator is connected into the oscillator at J54 and, by utilizing a vari-cap (C11) tunes the output circuit of the oscillator to the proper frequency. The frequency deviation of the oscillator is limited to approximately ± 1 kc. At this frequency (32 mc), 1 kc corresponds to a control of approximately 128 kc drift in the received signal carrier frequency. A manual frequency control is available on the front panel which allows the oscillator to be varied over a greater frequency range.

Tests have been performed on the phase detector to determine its operating characteristics. It was experimentally determined that a signal with a modulation index of $14\frac{1}{2}$ degrees (approximately the modulation index per test tone) will produce an output voltage of 0.35 volt (peak-to-peak) $\pm 1\frac{1}{2}$ db throughout the base range of 500 kc to 6 mc (at J9) in the multiple access mode. Using the modulation index expected in the frequency translation mode, the video output (at J10) was measured at 4.8 volts (peak-to-peak) $\pm 1\frac{1}{2}$ db throughout the range from 100 cycles to 6 mc and was down only 3 db at 60 cycles. (It is a requirement to receive the 60-cycle sync signal for TV reception.)

Two outputs are obtained from the 32-mc oscillator. The first is amplified and connected to the balanced modulator in the multiple access transmitter. The second is doubled in frequency, split into two parts, and amplified. One of the two outputs is connected (through J57) to the frequency multiplier chain for the multiple access transmitter and the other through J58 to the I phase detector as the reference signal.

When the receiver is operating normally and is phase-locked to the incoming RF signal, the front panel meters will read as follows:

| | Approximate Reading, microamperes |
|-----------------------|-----------------------------------|
| Signal strength meter | 200 |
| Frequency error meter | 0 |
| Phase error meter | 0 |
| Phase lock meter | 25 |

Multiple Access Transmitter. The multiple access transmitter is phase-locked to the received carrier. The input to the frequency multiplier, X32 (at J59), is multiplied by 32 and then by 3 to produce a carrier frequency of 6400 mc. Operation of this series of multipliers, bandpass filters, and isolators is similar to that of the receiver RF signal chain, but the frequency is multiplied up instead of divided down. The result is a 6400-mc carrier frequency input to the mixer at J71.

The 32-mc carrier input to the balanced modulator and preamplifier, at J72, is modulated by the video input at J73. The output from the modulator is a double sideband suppressed carrier signal which is amplified and connected to the high pass filter at J76.

The high pass filter consists of two stages of tuned filters, separated by a tee pad. Both stages are identical and are tuned to eliminate the lower sideband frequency. The tee pad is used for impedance matching. The output from the filter is a single-sideband signal.

The single-sideband signal is amplified in the postamplifier. The input is connected at a point following C14, and the balance of the amplifier is identical. The output from the postamplifier is connected to the mixer at J81.

The mixer and bandpass filter are spacecraft parts. The mixer output is filtered to remove spurious frequencies. The output of the filter is a single sideband on a carrier of the proper frequency for transmission to the spacecraft. The signal is then passed through a variable attenuator (front panel control) to a traveling-wave tube for final amplification and transmission to the spacecraft.

Frequency Translation Transmitter. The frequency translation transmitter receives the video input at J1. A front panel switch is used to select either phase or frequency modulation. In the frequency modulation position, the frequency of the oscillator is controlled by the feedback from the RLC network, including a vari-cap, C10. To modulate the carrier, the video input is applied across C10 and C11 to ground. This video signal causes a change in the capacitive reactance of C10, and therefore a change in the oscillator frequency. The result is an FM carrier output from Q1.

In the phase modulation mode, the same principle is utilized, but in a slightly different manner. The oscillator is crystal-controlled in this mode. The video is applied to the junction of two vari-caps, C17 and C18, in the phase modulator section of the module, resulting in phase modulation of the carrier frequency.

The modulated carrier is then amplified and connected to the frequency multiplier, X32. The balance of the transmitter is a series consisting of isolators, filters, attenuator, frequency multiplier, and traveling-wave tube amplifier which operates in the same manner as the other transmitter series. The output power level is controlled by a variable attenuator.

Power Supply. Two of the power supplies are regulated 24-volt supplies, one for +24 volts

dc and the other for -24 volts dc. An additional regulator reduces the voltage to 15 volts. The third power supply is a +250 volt dc unregulated supply that also supplies 6.3 volts ac for vacuum tube filaments.

MAJOR AND MINOR CONTROL ITEM TEST POSITIONS

This discussion covers the various test positions required to implement the major and minor control item "in-process" test specification.

Major Control Items

Communication Transponder. The transponder test sets described earlier in this section will be used to test the complete transponder. The commercial equipment listed in Table 8-1 will be used in conjunction with the transponder test set.

Communication Transmitter. The communication transmitter test position consists of a single bay of test equipment and will be used to perform tests in accordance with the test plan for the transmitter. The equipment that will be included in this position is shown in Table 8-2.

Minor Control Items

Telemetry Transmitter. The telemetry transmitter test position will consist of two bays of test equipment and will be used to perform tests on the 475220 telemetry transmitter in accordance with in-process test specification 475220-600. The equipment included in the test position is listed in Table 8-3.

Command Receiver Test Position. The command receiver test position will consist of three bays of test equipment and will be used to perform tests on the 475200 command receiver in accordance with in-process test specification 475210-600. The equipment to be included is listed in Table 8-4.

8. Ground Support Equipment

Regulator Test Position. The regulator test position will consist of one bay of test equipment. It will be used to perform tests on the following minor control items in accordance with the referenced in-process test specifications:

| | Test
Specification |
|---|-----------------------|
| 475101 Regulator, transponder, multiple access mode | 475101-600 |
| 475102 Regulator, transponder, frequency translation mode | 475102-600 |
| 475160 Regulator, phased array (PACE) | 475160-600 |
| 475163 Regulator, phased array (power amplifier) | 475163-600 |
| 475174 Power supply, traveling-wave tube | 475174-600 |
| 475212 Regulator, command | 475212-600 |
| 475222 Regulator, telemetry | 475222-600 |

The equipment to be included is listed in Table 8-5.

Master Oscillator Test Position. The master oscillator test position consists of two bays of test equipment and will be used to perform tests on the following minor control items in accordance with the referenced in-process test specifications:

| | Test
Specification |
|---|-----------------------|
| 475113 Master oscillator, 16 mc, frequency translation mode | 475113-600 |
| 475122 Master oscillator, 32 mc, multiple access mode | 475122-600 |
| 475123 Amplifier—master oscillator, 32 mc, multiple access mode | 475123-600 |

TABLE 8-1. TRANSPONDER COMMERCIAL TEST EQUIPMENT

| Quantity | Description |
|--|--|
| 1 | Noise generator — AIL07049, 07050, 07111 |
| 1 | Attenuator — Kay Electric 432C |
| 1 | Signal generator — HP 618B |
| 1 | Spectrum analyzer — SPA 4A |
| 1 | Five-bay test equipment cabinet |
| 1 | Traveling-wave tube and power supply — Huggins 304 |
| 1 | Traveling-wave tube and power supply — Huggins 210B |
| 1 | Traveling-wave tube and power supply — Huggins 309B |
| 2 | Traveling-wave tube and power supply — Huggins 210A |
| 1 | Frequency meter — HP G-532A |
| 1 | Scope — Tektronix 555 |
| 1 | Scope, plug-in — Tektronix H |
| 1 | Scope, plug-in — Tektronix L |
| 4 | Signal sampler — Microlab HY10N, HY10B, HZ10N, HZ10B |
| 1 | Test oscillator — HP 650AR |
| 1 | AC voltmeter — Ballantine 314-52 |
| 1 | High frequency mixer — HP 934A |
| 1 | Counter — HP 5243L (rack mount) |
| 1 | Frequency converter — HP 5251A |
| 1 | Frequency converter — HP 5253A |
| 1 | Variable attenuator — ARRA 5684-50 |
| 2 | Variable attenuator — ARRA 5-6684-50 |
| 1 | Noise figure meter — HP 340B |
| 1 | Noise receiver — Marconi TF 1225A |
| 1 | Noise generator — Marconi TF 1226B |
| 1 | Band stop filter unit — Marconi TM 5774 |
| 1 | Transfer oscillator — HP 540B |
| 1 | Synchronizer — DY15796 |
| 1 | Regulated power supply — ± 24 volts |
| Class C | |
| Miscellaneous rack connectors | |
| Running time meters and circuit breakers | |
| Rack wiring and miscellaneous cables | |

The equipment to be included in the test position is listed in Table 8-6.

Hughes Aircraft Company

TABLE 8-2. COMMUNICATION TRANSMITTER TEST EQUIPMENT

| Quantity | Description |
|----------|---|
| 1 | Power supply (traveling-wave tube) — Bertel Product |
| 1 | Spectrum analyzer — Panoramic SPA-4A |
| 1 | Calorimetric wattmeter — HP 834 |
| 2 | Power supplies — H Lab 726AR |
| 1 | Single bay test equipment cabinet |
| | Class C |
| | Miscellaneous test rack connectors |
| | Running time meters and circuit breakers |
| | Rack wiring and miscellaneous cables |

TABLE 8-3. TELEMETRY TRANSMITTER TEST EQUIPMENT

| Quantity | Description |
|----------|--|
| 1 | Signal generator — HP 608DR |
| 1 | Scope — Tektronix RM 503 |
| 1 | Scope preamplifier — Tektronix 53/54C |
| 1 | Scope preamplifier — Tektronix type L |
| 1 | Scope preamplifier — Tektronix type K |
| 1 | Counter — HP 5245L |
| 1 | Frequency converter — HP 5253B |
| 1 | Attenuator — Kay Lab No. 30-0 |
| 1 | Noise figure meter — HP H-51-342AR |
| 1 | VHF noise source — HP 343A |
| 1 | Audio oscillator — HP 200 CDR |
| 1 | Three bay test equipment rack |
| 2 | Power supply — H Lab 865B |
| 1 | Balanced modulator — GR 1000-P7 |
| 1 | VTVM — HP 412AR |
| 1 | Power supply — H Lab 711AR |
| 1 | Power meter — HP 431 |
| 1 | Sweep generator — Telonic HD-5 |
| 1 | Detector mount — Telonic |
| 1 | Spectrum analyzer — Panoramic SPA-4A |
| 1 | Coaxial attenuator — Microlab ADION, 10 db |
| 1 | Coaxial attenuator — GR 874G20, 20 db |
| 1 | Standing wave detector — PRD219 |
| 1 | Standing wave indicator — HP 415 |
| | Class C |
| 1 | Hughes-fabricated telemetry transmitter |
| | Miscellaneous test rack connectors |
| | Running time meters and circuit breakers |
| | Rack wiring and miscellaneous cables |

TABLE 8-4. COMMAND RECEIVER TEST EQUIPMENT

| Quantity | Description |
|----------|--|
| 1 | Counter — HP 5245L |
| 1 | Frequency converter — HP 5253A |
| 1 | Spectrum analyzer — Panoramic SPA-4A |
| 1 | Power meter — HP 431B |
| 1 | Thermistor mount — HP 478A |
| 2 | Attenuator — Microlab |
| 1 | VTVM — HP 410CR |
| 1 | Digital voltmeter — HP 3440 with 3442 |
| 1 | Power supply — H Lab 726AR |
| 1 | Oscillator — HP 205AGR |
| 1 | Receiver — Clarke A167E |
| 1 | Two bay test position cabinet |
| 1 | Oscilloscope — HP 130CR |
| | Class C |
| | Miscellaneous test rack connectors |
| | Running time meters and circuit breakers |
| | Rack wiring and miscellaneous cables |

TABLE 8-5. REGULATOR TEST EQUIPMENT

| Quantity | Description |
|----------|--|
| 1 | Power supply — Kepco Labs No. SC-36-2M |
| 2 | Power resistor decade box — Clarostat No. 240C |
| 1 | Decade box — Gen. Radio No. 1432T |
| 1 | Voltmeter — Triplett No. 327PL |
| 1 | Voltmeter — Weston No. 931 |
| 1 | Reference, standard dc — Polyrange |
| | Sensitive resistor No. 3-CEW-7A |
| 1 | Differential VTVM — Fluke No. 801B |
| 1 | Precision voltage divider — Fluke No. 80A-2 |
| 1 | True rms voltmeter — Fluke No. 910A |
| 1 | Oscilloscope — Tektronix RM45 |
| 1 | Scope preamplifier — Tektronix, type G |
| 1 | Milliammeter — Triplett No. 327PL |
| 1 | Microammeter — Triplett No. 327PL |
| 1 | Ammeter — Weston No. 931 |
| 1 | Test equipment cabinet |
| | Class C |
| | Miscellaneous test rack connectors |
| | Running time meter and circuit breakers |
| | Rack wiring and miscellaneous cables |

8. Ground Support Equipment

TABLE 8-6. MASTER OSCILLATOR TEST EQUIPMENT

| Quantity | Description |
|--|--------------------------------------|
| 1 | Spectrum analyzer — Panoramic SPA 4A |
| 1 | Counter — HP 524CR |
| 1 | Counter, plug-in — HP 525A |
| 1 | Power meter — HP 431B |
| 1 | Thermistor mount — HP 478A |
| 1 | VTVM — HP 410C |
| 1 | Two bay test equipment cabinet |
| 1 | VOM — Triplett |
| 2 | Power supplies — H Lab No. 6242 |
| Class C | |
| Miscellaneous test rack connectors | |
| Running time meters and circuit breakers | |
| Rack wiring and miscellaneous cables | |

X2, X3 Multiplier Test Position. The X2, X3 multiplier test position will consist of four bays of test equipment, and will be used to perform tests on the following minor control items in accordance with the referenced in-process test specifications:

| | Test Specification |
|----------------------|--------------------|
| 475116 X3 multiplier | 475116-600 |
| 475117 X2 multiplier | 475117-600 |

The equipment to be included in the test position is listed in Table 8-7.

X32 Multiplier Test Position. The X32 multiplier test position will consist of one bay of test equipment and will be used to perform tests on the following minor control item in accordance with the referenced in-process test specification:

| | Test Specification |
|---|--------------------|
| 475114 X32 multiplier (frequency translation and multiple access modes) | 475114-600 |

This position will also be used to perform tests on the following groups of units:

| | Test Specification |
|--|--------------------|
| Frequency translation local oscillator group | 475025-601 |
| Multiple access local oscillator group | 475025-602 |
| Multiple access transmitter group | 475025-603 |

The equipment to be included in the test position is listed in Table 8-8.

TABLE 8-7. X2, X3 MULTIPLIER TEST EQUIPMENT

| Quantity | Description |
|---|--|
| 1 | Spectrum analyzer — Panoramics SPA 4A (rack mount) |
| 1 | Microwave oscillator — Gen. Radio No. 1360A (rack mount) |
| 1 | Frequency meter — H.P. 536A |
| 1 | Line attenuator — ARRA 2-3414-30 |
| 1 | Slotted line — HP 805A |
| 1 | Coax termination — Bendix No. 21-8011-63 |
| 1 | Power meter — HP431B |
| 2 | Power supplies — H-LAB No. 6204 |
| 1 | Directional coupler — NARDA 3004 |
| 1 | One bay test equipment cabinet |
| 1 | Thermistor mount — HP478A |
| 1 | Isolator — 2 kmc and 3706 kmc |
| Class C | |
| Miscellaneous test rack connectors | |
| Running time meter and circuit breakers | |
| Rack wiring and miscellaneous cables | |

TABLE 8-8. X32 MULTIPLIER TEST EQUIPMENT

| Quantity | Description |
|--|-------------------------------------|
| 1 | One bay test equipment cabinet |
| 1 | Spectrum analyzer Polarad SA84W |
| 1 | Power meter — HP431B |
| 1 | Thermistor mount — HP487A |
| 1 | Power supply — H LAB 6204 |
| 1 | Attenuator — Microlab AC-20N |
| 1 | Directional coupler — Narda 3002-10 |
| Class C | |
| Hughes-fabricated bias supply | |
| Miscellaneous test rack connectors | |
| Running time meter and circuit breaker | |
| Rack wiring and miscellaneous cables | |

Hughes Aircraft Company

Intermediate Frequency Test Position. The intermediate frequency test position consists of two bays of test equipment and will be used to perform tests on the following minor control items in accordance with the referenced in-process test specifications:

| | Test
Specification |
|---|-----------------------|
| 475104 Intermediate amplifier,
64 mc | 475104-600 |
| 475109 Limiter, 64 mc | 475109-600 |
| 475110 Preamplifier, 64 mc | 475110-600 |
| 475111 Postamplifier, 64 mc | 475111-600 |
| 475130 Preamplifier, 32 mc | 475130-600 |
| 475131 Modulator, phase
32 mc | 475131-600 |
| 475132 Amplifier, double,
64 mc | 475132-600 |
| 475141 Filter, amplifier, 32 mc | 475141-600 |

The equipment to be included in the test position is listed in Table 8-9.

TABLE 8-9. INTERMEDIATE FREQUENCY TEST POSITION

| Quantity | Description |
|--|---------------------------------------|
| 1 | Spectrum analyzer — Panoramic SPA 4-A |
| 1 | Signal generator — HP608CR |
| 1 | Power meter — HP431B |
| 1 | Thermistor mount — HP478A |
| 1 | RF millivoltmeter — HP411AR |
| 1 | VTVM — HP410CR |
| 2 | Power supplies — H LAB 865B |
| 2 | Kay attenuators — No. 30-0 |
| 1 | Noise figure meter — HP342A |
| 1 | Noise source — HP343 |
| 1 | Two bay test equipment cabinet |
| 1 | Scope — Tektronix RM503 |
| CLASS C | |
| Miscellaneous test rack connectors | |
| Running time meters and circuit breakers | |
| Rack wiring and miscellaneous cables | |

Microwave Test Position A. Microwave test position A will consist of three bays of test equipment, and will be used to perform tests on the following minor control items in accordance with the referenced in-process test specifications:

| | Test
Specification |
|---|-----------------------|
| 475100 Mixer, input | 475100-600 |
| 475103 Receiver, directional
coupler, 20 db | 475103-600 |
| 475112 Mixer, high level | 475112-600 |
| 475118 Filter, 6 kmc, frequency
translation and multiple
access modes | 475118-600 |
| 475172 Monitor, telemetry | 475172-600 |
| Rantec bandpass filters, 4 kmc | FC207-600 |
| Rantec multiplexer, receiver | FA106-600 |
| Rantec multiplexer, transmitter | FA107-600 |

The equipment to be included in the test position is listed in Table 8-10.

Microwave Test Position B. Microwave test position B will consist of three bays of test equipment, and will be used to perform tests on the following minor control items in accordance with the referenced in-process test specifications:

| | Test
Specification |
|---|-----------------------|
| Rantec dual filter hybrid, 2 kmc | FS222-600 |
| Rantec bandpass filters, 2 kmc 16
mc bandwidth | FS217-600 |
| Rantec bandpass filters, 2 kmc
6 mc bandwidth | FS218-600 |

The equipment to be included in the test position is listed in Table 8-11.

8. Ground Support Equipment

TABLE 8-10. MICROWAVE TEST POSITION A
TEST EQUIPMENT

| Quantity | Description |
|--|---|
| 1 | Three bay test equipment cabinet |
| 1 | Sweep generator (modified 3.7 to 6.7 GC) — Alfred 633AR |
| 1 | Spectrum analyzer — Panoramic SPA-10 |
| 1 | Noise figure meter — HP H-51-342AR |
| 1 | Noise source (waveguide) — HP No. J-347A |
| 1 | Power meter — HP 431B |
| 1 | Standing wave indicator — HP 415CR |
| 1 | Slotted line, BRM type (3-piece assembly) — MD No. 1000 LX |
| 1 | Precision frequency meter — HP J-532A |
| 1 | Precision frequency meter — HP G-532A |
| 1 | Precision attenuator — HP J-382A |
| 1 | Precision attenuator — HP G-382A |
| 1 | VHF noise source — HP No. 343A |
| 4 | BRM coax to waveguide adapter — MD No. 13,000,000 |
| 4 | BRM coax to waveguide adapter — MD No. 13,000,003 |
| 2 | Ferrite circulator — MD No. 16,100,010 (4.1 gc) |
| 2 | Ferrite circulator — MD No. 16,100,010 (6.2 gc) |
| 2 | Standard termination — MD No. 11,010,010 |
| 6 | Standard termination (miniature) — C.S. No. 20025P |
| 6 | Standard termination (miniature) — C.S. No. 20025J |
| 2 | BRM coax phase shifter (4-6.3 gc) — MD No. 2200C |
| 4 | Type N to J band waveguide adapter — HP J281A |
| 4 | Type N to G band waveguide adapter — HP G281A |
| 1 | Thermistor mount — HP 478A |
| 1 | Crystal detector mount — HP 420A |
| 8 | Type OSM coax to type N adapter (J) — O.S. No. 21020 |
| 8 | Type OSM coax to type N adapter (P) — O.S. No. 21040 |
| 3 | Crystal detector mount — O.S. No. 20080 (4-11 gc) |
| 2 | Pair, matched pair crystal detector mount — O.S. No. 20080M |
| 2 | Coax to type N circulator (4.05 gc) — Melab No. HC405 |
| 2 | Coax to type N circulator (4.25 gc) — Melab No. HC625 |
| 1 | Precision attenuator set — Weinschel No. AS-2 |
| 1 | Variable coax attenuator (3.9 to 10 gc) — PRD No. CX1110 |
| 1 | Scope — Tektronix RM503 (P-7) |
| 1 | X-Y recorder — Moseley 135AR |
| 2 | Power supplies — H LAB 865B |
| 1 | Traveling-wave table amplifier (3.7 to 6.7 gc) — Alfred No. 5-542 |
| 1 | Oscillator (50-250 mc) — G.R. 1215B |
| 1 | Power supply — G.R. 1201B |
| CLASS C | |
| Miscellaneous test position connectors | |
| Running time meters and circuit breakers | |
| Rack wiring and miscellaneous cables | |
| Modular power supplies | |
| Current meters and test panels | |

TABLE 8-11. MICROWAVE TEST POSITION B
TEST EQUIPMENT

| Quantity | Description |
|--|--|
| 1 | Three bay test equipment cabinet |
| 2 | BRM type coax coupler, 3 db — MD No. 1200C |
| 2 | BRM type coax coupler, 10 db — MD No. 1200C |
| 2 | BRM type coax coupler, 20 db — MD No. 1200C |
| 2 | BRM type coax coupler, 30 db — MD No. 1200C |
| 2 | BRM type coax coupler, 6 db — MD No. 1200C |
| 2 | BRM coax miniature isolator (4.1 gc) — MD No. 17,000,001 |
| 2 | BRM coax miniature isolator (6.2 gc) — MD No. 17,000,001 |
| 1 | Sweep generator (1.7 to 3.7 gc) — Alfred 632A |
| 1 | Variable coax attenuator (0-60 db) — Alfred No. E103 |
| 1 | Traveling-wave tube amplifier (1.7 to 3.7 gc) — Alfred 502A |
| 1 | Receiver (30 mc IF with precision attenuation) — LEL to LR 1410 |
| 1 | Mixer preamplifier assembly (1.7 to 2.4 gc) — LEL No. SAC-3-30-08-50 |
| 2 | Broadband coax circulator (1.7 to 2.4 gc) — E and M No. LS 110 LYC |
| 1 | Frequency meter (1-4 gc) — HP 536A |
| 2 | Power supplies — H LAB 865B |
| 1 | Directional coupler, high directivity (1.9-4.0 gc) — HP 777D |
| 1 | Scope, dc — Tektronix RM 503 |
| 1 | BRM coax slotted line — M.D. No. 1000-001 |
| 1 | Tunable detector mount — FXR No. B200A |
| 1 | RF power meter — HP 431B |
| 1 | Coax thermistor mount — HP 478A |
| 1 | X-Y recorder — Moseley No. 135 CMR with remote electronic pen control and rear input terminals |
| 1 | Matched pair crystal detector mounts — OSM 20080 |
| 1 | Spectrum analyzer — Panoramic SPA-4 |
| 1 | Ratio meter — HP 416B |
| 1 | Attenuator set — Weinschel AS-2 |
| CLASS C | |
| Miscellaneous test rack connectors | |
| Running time meters and circuit breakers | |
| Rack wiring and miscellaneous cables | |

DIGITAL SUBSYSTEM MAJOR AND MINOR CONTROL ITEM TEST EQUIPMENT

Module and Unit Test Equipment. During the interim program, two testers were designed and fabricated: the PACE module tester and unit tester. The module tester (Figure 8-10) has been used to test about half of the welded modules

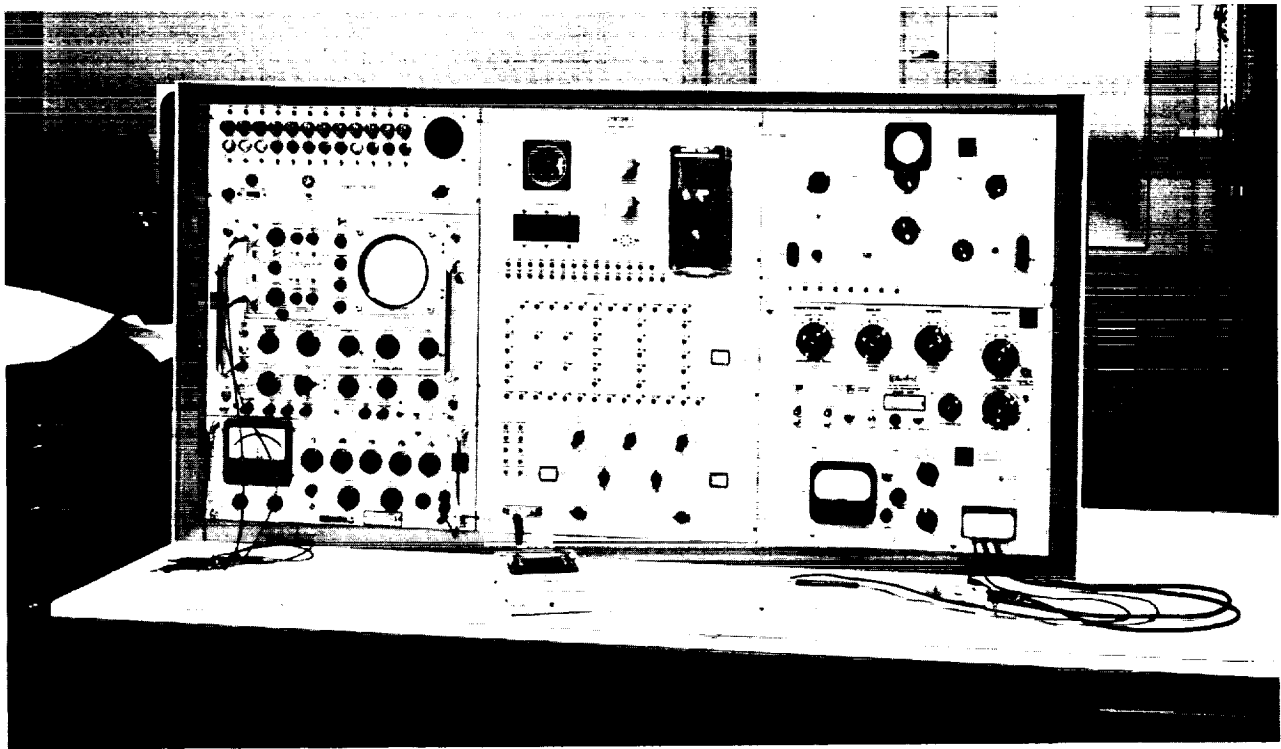


FIGURE 8-10. PACE WELDED MODULE TESTER

being built for the basic PACE. The function board/unit tester (Figures 8-11 and 8-12) has been fabricated and checked out as far as possible until the function boards are fabricated and ready for test.

The testers are comprised of the following equipments:

Module Tester

- Oscilloscope, Tektronix Type RM35A
- Oscilloscope, Tektronix Type CA plug-in
- Oscilloscope, Tektronix Type D plug-in
- Differential voltmeter, Fluke, Model 825A/AB
- Vacuum tube voltmeter, Hewlett-Packard, Model Y94-410BR
- Pulse generator, Rutherford, Model B7F
- Low frequency function generator, Hewlett-Packard, Model 202A
- Power supply, Harrison Labs, Model 6343A (4 each)

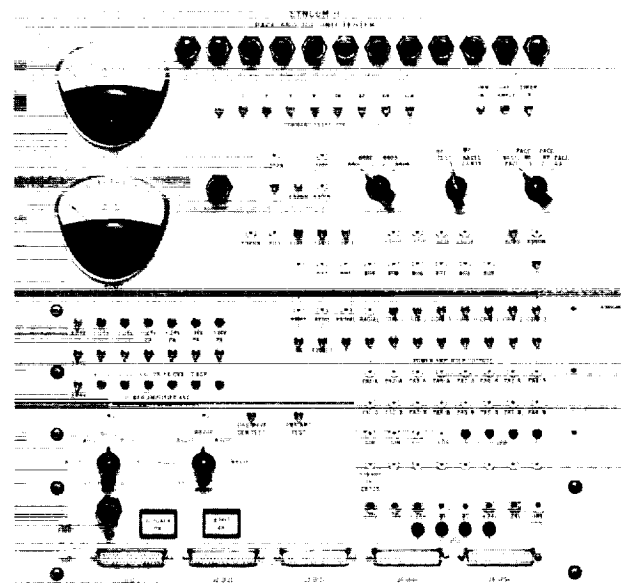


FIGURE 8-11. SPECIAL ELECTRONICS UNIT FOR PACE FUNCTION BOARD/UNIT TESTER

8. Ground Support Equipment

Power supply, Harrison Labs, Model 6346A (6 each)

Power supply, Harrison Labs, Model 6347A (2 each)

Special electronics unit, Hughes-fabricated
Function Board/Unit Tester

Oscilloscope, Tektronix, Type RM45A

Oscilloscope, Tektronix, Type M plug-in

Oscilloscope, Tektronix, Type D plug-in

Electronic counter, Hewlett-Packard, Model 5233L

Digital voltmeter, Beckman, Model 4011R

Low frequency function generator, Hewlett-Packard, Model 202A

Power supply, Transpac, Model TR24R (2 each)

ψ - ψ_2 simulator, Hughes

Special electronics unit, Hughes

In addition to the testers that have already been built, the following testers will be required for a production phase. A preliminary listing of commercial test equipment requirements is included.

PACE module tester (similar to one already built)

Central timer module tester

Telemetry encoder module tester

Command decoder module tester

Synchronous controller card tester

Telemetry processor card tester

Command generator card tester

Solenoid driver unit tester

Oscilloscope

Low frequency function generator

Power supply (36 volts at 2 amperes)

Squib driver unit tester

Oscilloscope

Differential

Power supply (60 volts at 10 amperes)

Central timer function board tester

Oscilloscope

Differential voltmeter

Audio frequency signal generator

Power supply (40 volts at 500 milliamperes)
2 each

Power supply (36 volts at 1 ampere)

General purpose electronic counter

Telemetry subsystem function board/unit tester

Oscilloscope

Electronic counter

VTVM

Differential voltmeter

Power supply (24 volts at 500 milliamperes)

Power supply (36 volts at 1.5 amperes)

Command subsystem function board/unit tester

VHF signal generator

50-ohm step attenuator, 1-db steps

50-ohm step attenuator, 10-db steps

Power Supply (40 volts at 2 amperes)

General purpose electronic counter (0 to 50 mc)

Frequency connector (500 mc)

Power meter

Thermistor mount

Digital voltmeter

Connector, ac-to-dc

Oscilloscope

Synchronous controller unit tester

Oscilloscope

General purpose electronic counter

Power supply (50 volts at 15 amperes)
2 each

Telemetry processor

Oscilloscope

VTVM

Printer

Strip chart recorder with preamplifiers

Random noise generator

Digital voltmeter

Command generator unit tester

Oscilloscope

Electronic counter

Data generator

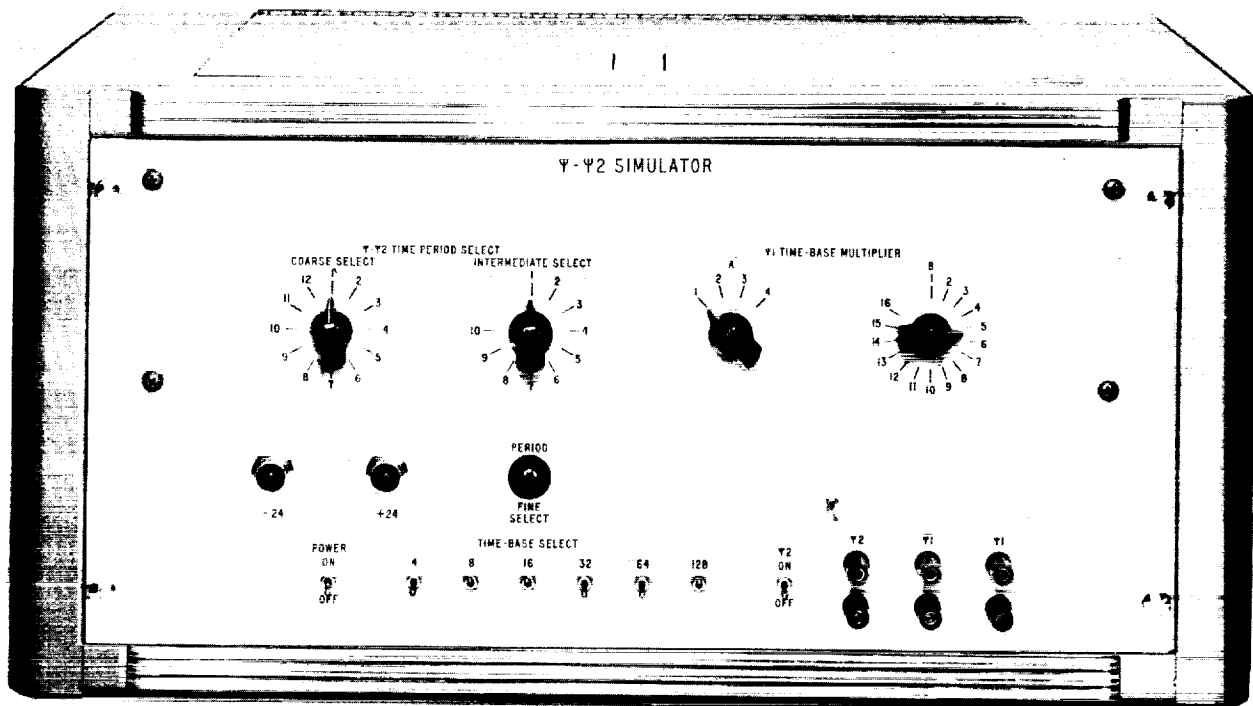


FIGURE 8-12. Ψ - Ψ_2 SIMULATOR

Four ovens will be necessary for temperature cycling during in-process testing. The ovens will be used with several of the testers and so are not listed with any particular testers.

Auto-Tester. In view of past experience with welded module testing, it is considered necessary to use an automatic tester for the Advanced Syncom flight fabrication program. The tester selected is manufactured by Hughes Aircraft Company and has been used on both the Polaris and Minuteman programs. It is card programmed and is capable of making both static and dynamic tests of a module.

The tester is built around two commercial units: on EAI 5001 digital voltmeter for all dc measurements and a DYMEC 5844C measuring system for waveform analysis. The waveform analysis includes rise time, fall time, delay time, pulse width, and pulse amplitude measurements.

The card programmer controls application of all power and loading, application of static or dynamic inputs, and measurement and interpretation of all desired waveform parameters of the module under test.

The tester is equipped with four programmable power supplies to furnish both circuit power and bias voltages to the module. A pulse generator furnishes the required dynamic inputs. Load cards are used with each module to furnish the proper load conditions. Resistances can be inserted in series with each of the inputs to simulate the impedance levels the module would encounter in actual use.

All measurements are converted to digital quantities which are then compared to values punched on the cards. The result of the comparison is conveyed to the operator by the lighting of one of seven lamps. These lamps indicate

8. Ground Support Equipment

whether the value measured was within acceptable limits or whether it was high or low. In addition, a printout is made for each measurement so that a permanent record can be kept of each module's performance.

The tester is capable of making as many as 80 tests a second, so that test time is greatly reduced. Tests already conducted on modules for the Polaris program indicate that modules can be tested in about one-eighth the time taken on a manual tester. The fast testing rate also makes it feasible

to test more parameters for each module, so that more marginal conditions can be detected and corrected. This more severe testing will increase the reliability of the flight versions.

A small amount of redesign in the present tester is being accomplished to make it more usable for Syncom circuits. This redesign includes extending the low frequency measurement capability and implementing a more flexible input/output section.

9. SPACECRAFT HANDLING EQUIPMENT

During the reported program, design effort was directed toward development of four major items of ground handling equipment. No actual hardware was required or fabricated during this period. The following summary describes the equipment and the related design effort.

WEIGHT, CG, AND MOMENT OF INERTIA FIXTURE

A combination fixture which permits the measurement of mass properties was developed. The fixture consists of two parts: 1) a roll fixture (Figure 9-1a) which attaches to the separation

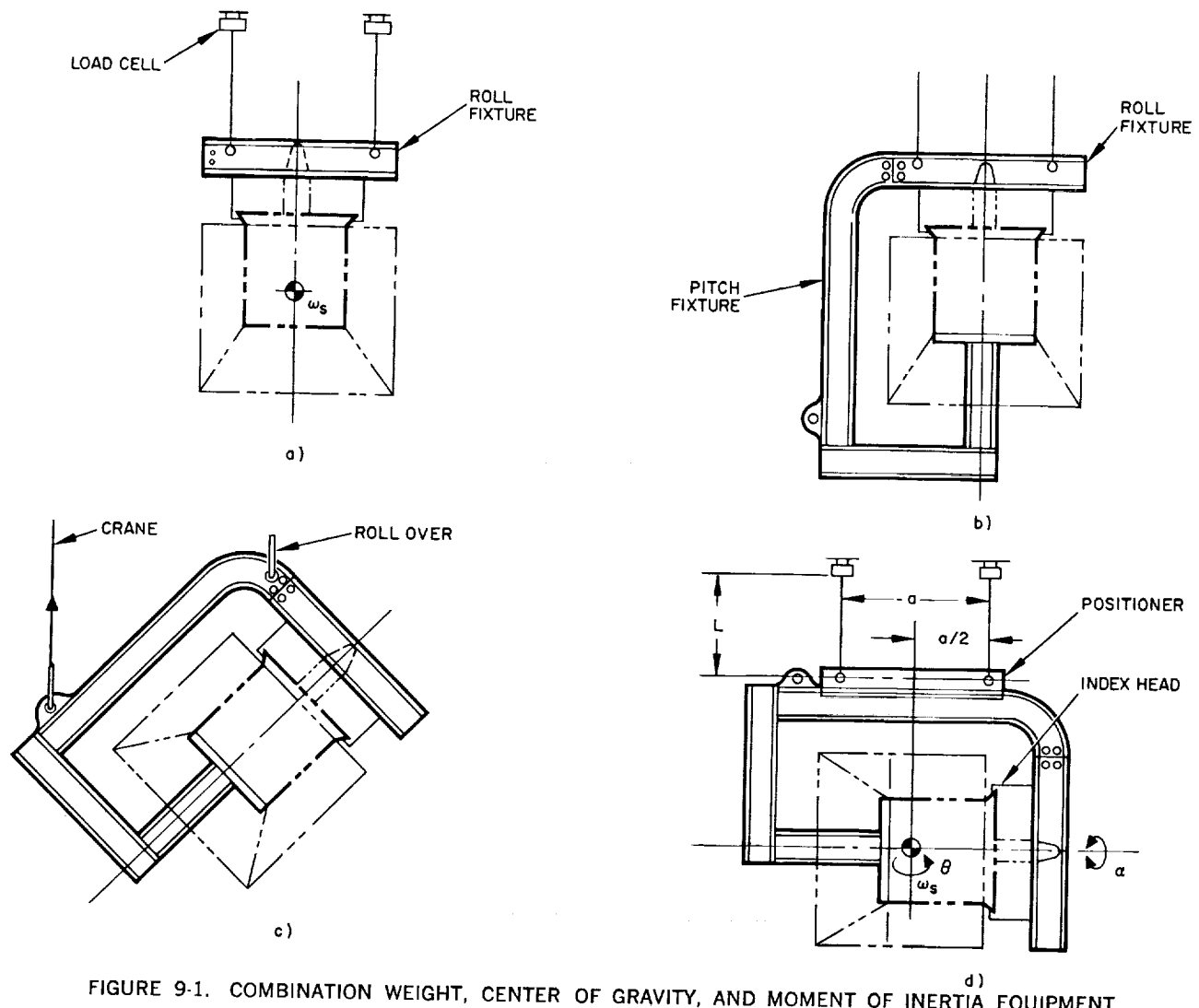


FIGURE 9-1. COMBINATION WEIGHT, CENTER OF GRAVITY, AND MOMENT OF INERTIA EQUIPMENT

plane and 2) an additional structure which attaches to the motor mounting area (Figure 9-1b) and is connected to the roll fixture, thus becoming a fixture permitting measurement of moment of inertia about the pitch and yaw axis. The frame has provisions for attachment of cables from which the spacecraft is suspended for determination of moment of inertia (Figure 9-1d). Two hoists are used to rotate the spacecraft from roll to pitch position (Figure 9-1c). The fixture incorporates a roller-bearing-supported index device which is used to rotate the spacecraft to predetermined positions 45 degrees apart as required. Weight and cg location is determined by suspending the fixture-spacecraft combination from load cells (Figure 9-2). The fixture is made of aluminum, predominantly of welded construction.

SYSTEMS TEST AND SPIN FIXTURE

A special fixture was designed to permit systems tests on a spinning spacecraft. As originally required, the fixture was designed as a mobile piece of equipment and incorporated a specially designed trailer. The fixture was permanently mounted on the trailer and surrounded by a shroud which protected personnel from the spinning spacecraft.

The mobility requirement was eliminated subsequently, and a redesign of the supporting structure was necessary. A new enclosure, to replace the shroud that was incorporated on the trailer, was also required. All these tasks were completed, and the fixture is now designed to be used in a permanent location.

The fixture (Figure 9-3) consists of a pylon assembly supported on a tubular pedestal. The pylon assembly is composed of a housing which supports a drive shaft. A mounting ring is attached to the shaft to permit installation of the

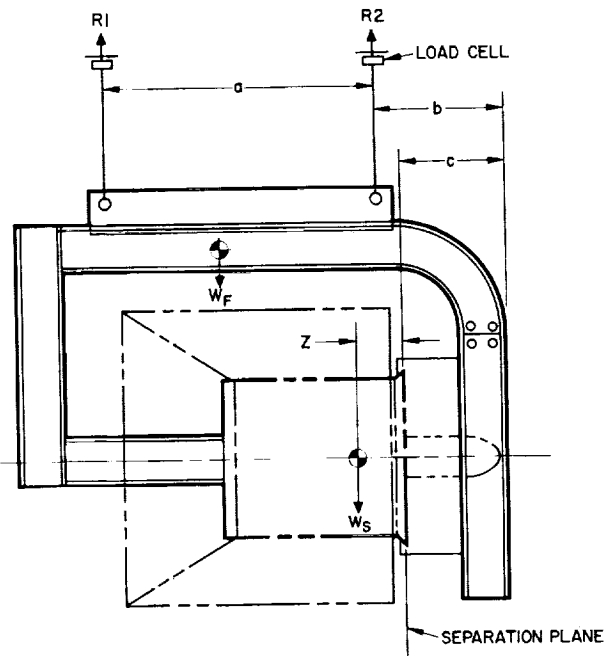


FIGURE 9-2. SPACECRAFT LONGITUDINAL CENTER OF GRAVITY MEASUREMENT

spacecraft on the fixture. A series of slip rings, both low and high frequency, as well as a coaxial rotary joint are incorporated in the pylon assembly for test purposes.

To ensure that variations in rotational speed do not exceed very narrow limits specified as design objectives, a U.S. Motors Varidyne system is used. The system consists of a power unit which converts standard 60 cycle line power to other frequencies and supplies the converted and controlled frequency to a slave motor. The slave motor rotates the drive shaft of the spin test fixture through a V-belt drive. This system permits a succession of stepless speed changes within the desired range and assures maximum speed stability, not affected by line voltage changes, heating, or changes of electrical resistance. A pulse generator and data link is included in the system to provide accurate readout of rpm.

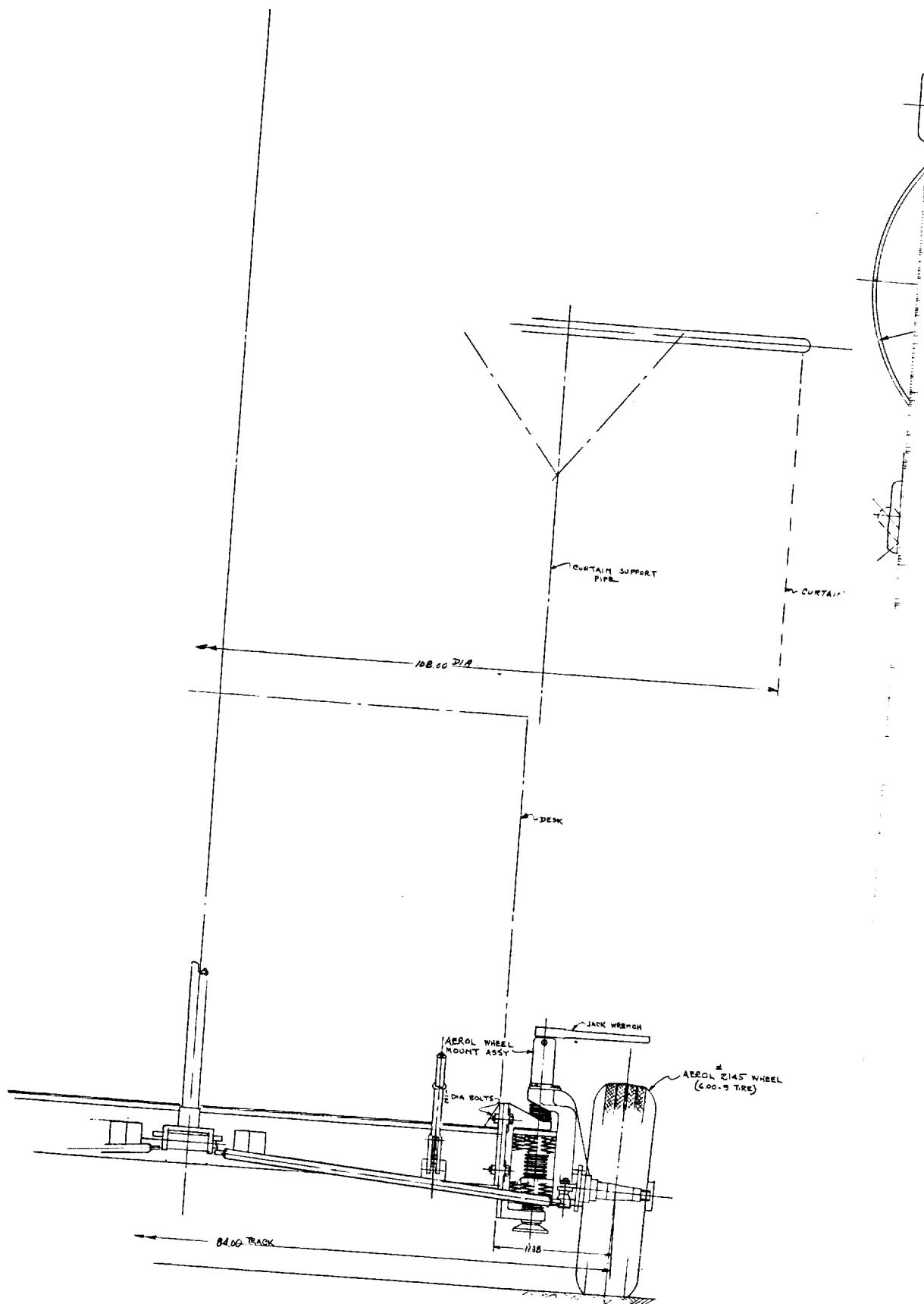
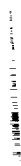
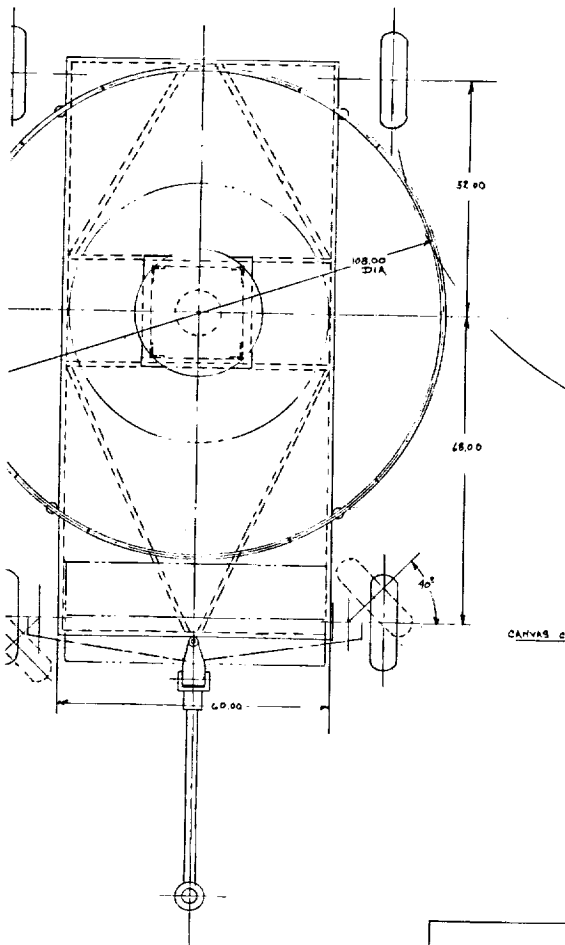


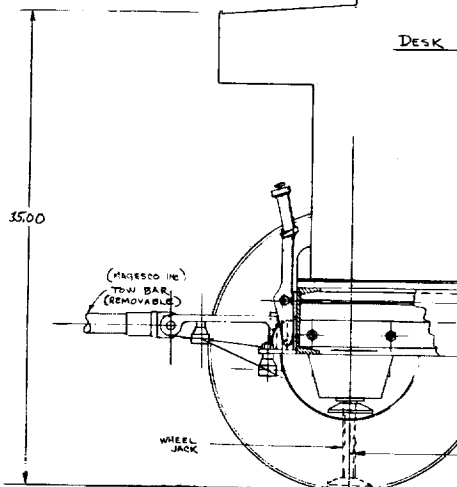
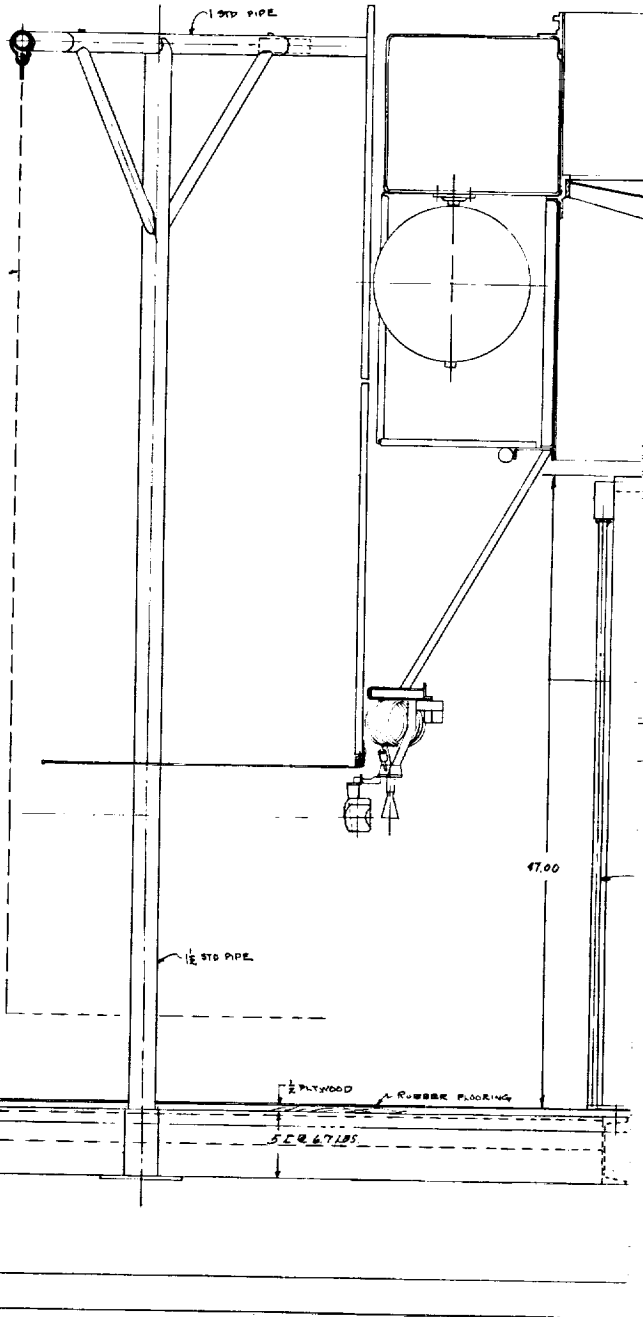
FIGURE 9-3. ADVANCED SYNCOM SPIN FIXTURE





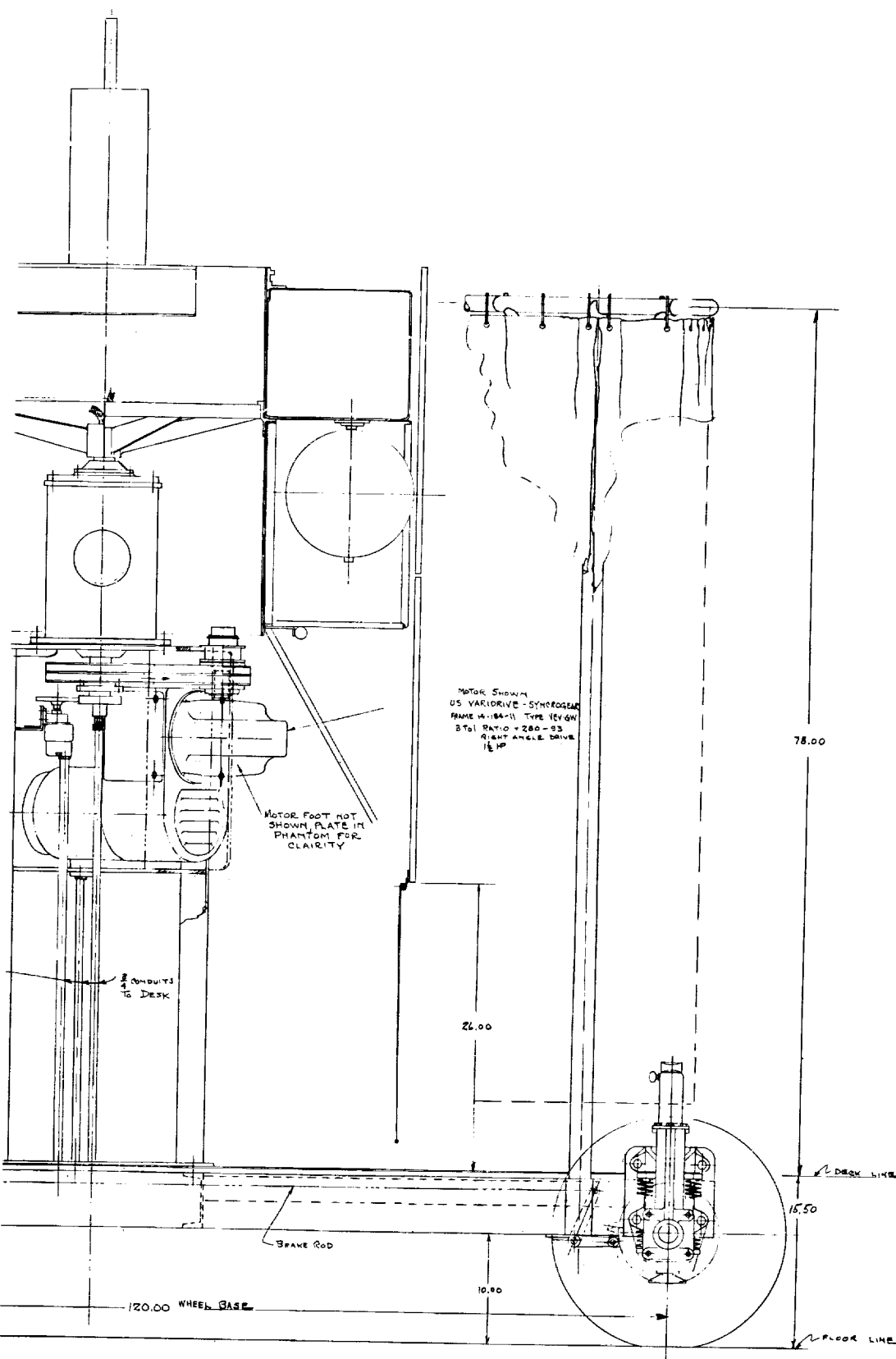
SCHEMATIC
PLAN VIEW TRUCK
SCALE 1/4"

CANVAS CURTAIN



2





2

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MOBILE ASSEMBLY STAND

A fixture to facilitate the assembly and installation of components within the spaceframe as well as performance of various checkout operations was designed (Figure 9-4). A rigid steel tubular frame supports a cantilever arm to which the spacecraft is attached. The arm is pivoted on large bearings and may be rotated by a hand-operated worm gear drive to place the spaceframe in the desired working position. The axis of rotation is located in the vicinity of the combined arm and spaceframe center of gravity, thus requiring minimum effort on the part of the operator. The stand is equipped with casters for mobility within the work area as well as transportation within the plant. Brakes secure the stand in any desired location.

BALANCING MACHINE

A special machine for dynamic balancing of the spacecraft was designed. Two versions were developed. The first (Figure 9-5), designed for spacecraft mounting by the motor mount (i.e., with the antenna pointing up) consists of a tu-

bular A-frame supporting a cradle. The cradle is attached to the A-frame by spring steel flexures placed so as to pick up vibration in two different planes. The lower end of the cradle contains a turntable on which the spacecraft may be mounted. An electric motor drives the turntable through a V-belt. A damper between the cradle and the frame prevents excessive vibration build-up. Electromagnetic dashpots pick up the unbalance and readout meters indicate both the amount and the location of the unbalance. A railing attached to the frame is provided for safety of personnel.

Since this machine did not permit balancing the spacecraft with the apogee motor attached, a second version was designed. In this version, the spacecraft is attached to the turntable by the separation plane (antennas pointing down). This balancer has the following significant differences from the first version:

- 1) Mounting the spacecraft by the separation plane is easier.

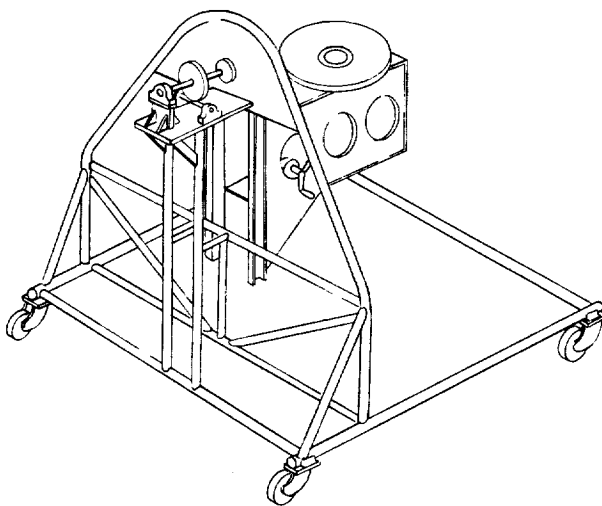


FIGURE 9-4. MOBILE ASSEMBLY STAND

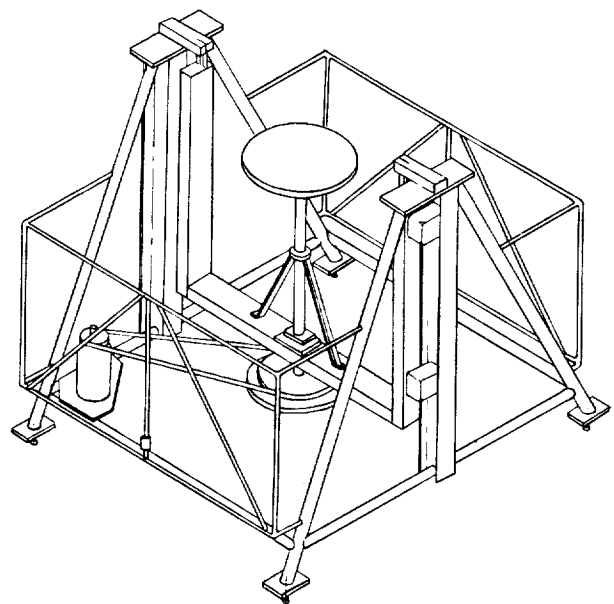


FIGURE 9-5. BALANCING MACHINE

Hughes Aircraft Company

- 2) Flat springs from which the cradle is suspended may be varied in thickness, thus tailoring the frequency of the cradle in a manner permitting more accurate displacement reading.
- 3) Rotation is provided by a U.S. Varidyne variable frequency unit of the same type as that used on the spin test machine. This unit provides a very stable rotational speed and, being mounted outside the balancer, permits remote speed control (up to 200 feet away).

- 4) An amplifier-filter eliminates all "noise" voltages except those generated by unbalance of the rotating spacecraft, providing better accuracy.

- 5) Leveling screws are provided in the frame.

While this balancer was in design, a commercial balancer was proposed and a decision made to purchase the commercial unit. A preliminary statement of work and procurement specification were prepared.

10. NEW TECHNOLOGY

NEW ITEMS

Solar Array Fabrication Technique. A fabrication technique for the solar array has been developed which is believed to have several unique features. The combination of materials and techniques used yields the highest known solar array strength-to-weight ratio and improves the panel reproducibility and uniformity. Of particular significance is the method of fabricating the solar cell strings by use of a machine developed specifically for this purpose and the cell-to-substrate bonding method.

Details concerning the array fabrication are contained in Section 7 of this report under *Solar Array Substrate Development* and *Solar Cell to Substrate Adhesive Development*.

Repairable Foam. An improved method of potting modules in a repairable foam has been developed. By this method, module repairs may be made without damaging the modules or the mounting board.

The mounting board is covered with a thin laminate bonded to the board with a modified epoxy polyamide breakaway adhesive. The polyurethane foam is marked with lines placed so that cutting through the foam on the lines will not damage the underlying structure or circuitry. Thus, when repair is needed, the defective module is unsoldered from the board and the foam is cut through the appropriate lines with a dull brass knife to prevent cutting the etched board. The board is then heated to 200°F to weaken the breakaway adhesive, and the cut foam block

is removed from the warm board. The laminate stripped from the board is replaced with the breakaway adhesive and the repaired or replaced module is resoldered to the board. The board is then refoamed to replace the foam cut away during the module removal. To protect the foamed unit during handling prior to assembly, a plastic cover is used. (See Section 7, Figures 7-17 and 7-18.)

PREVIOUSLY REPORTED ITEMS

All items reported during the contract period are summarized below. Innovations were reported separately from items believed to constitute inventions, in a special section of the Monthly Progress Report; inventions were reported directly by the Hughes Patent Office. The dates shown refer to the report in which the particular item appeared or, in the case of inventions, to the reporting date of the item.

Innovations

Spacecraft Launch Rate and Availability: Fixed Time Replacement Policy (Summary Report, 31 March 1963) — an analytical technique to determine satellite availability and replacement rate.

X3 Multiplier (Summary Report, 31 March 1963) — a varactor multiply-by-three device.

Output RF Power Switch (Summary Report, 31 March 1963) — a ferrite switch, essentially a three-port circulator with provisions for electronically reversing the permanent magnetic field polarity, which operates as a single-pole double-throw switch.

Hughes Aircraft Company

Horizon Scanners (May 1963) — a method for determining the angle between the spin axis and the equatorial plane by the use of earth sensors.

Whip Antenna Orientation (June 1963) — an orientation of whip antennas to produce an optimum radiation pattern.

Solar Cell Support Panels Attachment (July 1963) — a mounting arrangement for the solar panels that prevents deformation of the spacecraft from transmitting loads into the solar panels.

Divide-by-32 Circuit (August 1963) — an efficient and compact varactor divider consisting of five divide-by-two stages operated in series, each stage composed of a unique parallel arrangement of varactor diodes. The divider was designed to operate with an input frequency in the 2000-mc range.

Inventions

Resistance Soldering Technique for Solar Cell Assemblies PD 5334 (1 March 1963) — a technique for the automatic assembly of solar cells into large matrix series-parallel combinations for installation on substrate panels.

Triaxial Antenna PD 5513 (1 July 1963) — an antenna element utilizes complex feed slots to improve the pattern and impedance match of a horizontally polarized antenna.

Capacitor Error Trimmer PD 5548 (15 July 1963) — an improved trimmer capacitor.

Varactor High Level Mixer PD 5623 (1 October 1963) — an efficient modulation device that generates carrier suppressed double sidebands by mixing two frequencies by the variable capacitance effects provided by varactor diodes.

SUBCONTRACTS

During the contract period the following subcontracts have contained the Reporting of New Technology and the Property Rights in Inventions clauses:

Subcontractor: The Marquardt Corporation
16555 Saticoy Street
Van Nuys, California

Statement of Work: Phase I — To design, develop, fabricate, test, and deliver an engineering model of the bipropellant reaction control unit.

Interim Program — to conduct tests to determine the adequacy of the engineering model of the bipropellant reaction control unit and to modify the model to permit steady-state operation.

Completion Date: Phase I — 15 July 1963
Interim Program — 18 October 1963

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