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NAVIGATION/TRAFFIC CONTROL SATELLITE MISSION STUDY

VOLUME III
SYSTEM CONCEPTS

CRAIGIE, CAPRIOGLIO, RENN, DRUCKER, PIERCE, ET AL.

JUNE 1969

Electronics Research Center
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Prepared under Contract No. NAS 12-595 by

TRW
SYSTEMS GROUP

ONE SPACE PARK • REDONDO BEACH, CALIFORNIA 90278

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ACKNOWLEDGMENT

The Navigation/Traffic Control Satellite Mission Study has been an extensive team effort. While numerous technical personnel made contributions to the study, the following TRW Systems people made significant contributions to this volume:

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Section 3	Design and Development/ Preoperational Navigation/ Traffic Control Satellite System	C. W. Renn, F. M. Holmes
Section 4	Expansion to a Worldwide Operational System	A. N. Drucker, T. P. Nosek, H. T. Ekstrand
Section 5	Program Plan and Costs	K. M. Pierce, M. Arm J. H. Craigie
Section 6	Applications	N. A. Jolley, W. F. Storer, D. D. Otten, J. H. Craigie, A. Burbank, M. C. Chapman, J. B. Gardner

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1. INTRODUCTION

This volume of the Navigation/Traffic Control Satellite Mission Study Final Report describes the system which has been synthesized during this study. Section 1 outlines the Navigation/Traffic Control Satellite System concept and the basic design and test philosophy. Section 2 describes the recommended North Atlantic Operational System and contains the detailed description of spacecraft, user hardware, and ground stations. Section 3 describes the Design and Development/ Preoperational System which is needed in order to develop and demonstrate the hardware, software, and operational aspects of the Navigation/Traffic Control Satellite System. Section 4 describes the expansion to more complete North Atlantic/U.S./Europe coverage and thence to worldwide operation. Section 5 delineates the program required to bring about the D and D/Preoperational, North Atlantic Operational, and Worldwide systems, including budgetary and planning cost estimates. Section 6 describes a number of important applications for this system including collision avoidance, solar flare warning, and Continental U.S. air traffic control.

1.1 SYSTEM CONCEPT

The Navigation/Traffic Control Satellite System concept has been an outgrowth of the Ad Hoc Joint Navigation Satellite Committee (Reference 1) and the continuing National Aeronautics and Space Administration program, both at the Office of Space Science and Applications, and at the Electronics Research Center.

1.1.1 Basic Elements of the System

In addition to people, procedures, and the natural environment of the earth itself, there are three basic elements of the Navigation/Traffic Control Satellite System which need to be considered in its conceptual design. The first element is made up of the satellites themselves, which provide basic navigation data and high quality, highly reliable communications relay services. The second basic element of the system — called user equipment or user hardware, is made up of the electronics equipment used by the various system customers. It consists of antennas, receivers, data processors, displays, and the like. The third major element of the system is made up of the various ground stations which provide satellite system support such as satellite tracking and station-keeping, and user mission support such as air traffic control, meteorological advisory services, and relay of company communications.

1.1.2 Functions

The Navigation/Traffic Control Satellite System must be configured such that the following functional services can be provided:

- Communications
- Surveillance
- Navigation
- Collision avoidance.

The communications, surveillance, and navigation functions make up the basic elements of a number of missions, such as air traffic control, search and rescue operations, and recovery of manned and unmanned spacecraft. Other missions, such as scientific or commercial exploration, usually involve two or more of these functions. In the air traffic control mission, which is the one considered in greatest detail in the Mission Study, the relative emphasis and importance of all four basic functions will vary between the North Atlantic case and the Continental United States case; and will in all likelihood, vary significantly. For this reason it is important in its early conceptual phase to examine these potential roles, and the potential capability of this system, and to configure the system such that it is capable of performing any or all suitable roles.

1.1.3 Users

The Ad Hoc Joint Navigation Satellite Committee made it clear that there was a wide range of potential customers for such a system. NASA has, from the outset, intended that the operational system be available and economically beneficial to a broad spectrum of users including:

- Large aircraft
- Large ships
- General aviation
- Small marine craft
- Specialized users such as scientific projects or expeditions.

The reason that the Navigation/Traffic Control Satellite System must cater to a broad spectrum of users is simply one of economics. The system will be more viable and of greater economic benefit if it can be applied to a wide variety of uses by many different subscribers.

1.1.4 Coverage

The Ad Hoc Joint Navigation Satellite Committee recognized that an immediate need exists for a more efficient air traffic control system over the North Atlantic Ocean. For this reason the North Atlantic Ocean Area has received primary emphasis in the study; but, recognizing the obvious benefits of a worldwide system of this type, NASA incorporated into the study requirements the ability of the system to function on a worldwide basis and called for an examination of the impact on the system design of the expansion to worldwide coverage.

1.2 SYSTEM DESCRIPTION

1.2.1 Satellites

1.2.1.1 Satellite Configuration Description

The NTC Satellite shown in Figure 1 will exist in three similar configurations with a high degree of commonality between configurations. In fact, the basic structure and dimensions will be identical. Configuration A will be the Design and Development/ Preoperational spacecraft. It will weigh approximately 650 pounds (dry) and will have as its payload one voice, one data, and one ranging channel. Configuration B will be an operational version of Configuration A and will be employed in synchronous inclined elliptical orbits. It will differ from Configuration A only in the yaw control design. Configuration C will be the standard operational spacecraft and will be employed in synchronous equatorial orbits. It will weigh

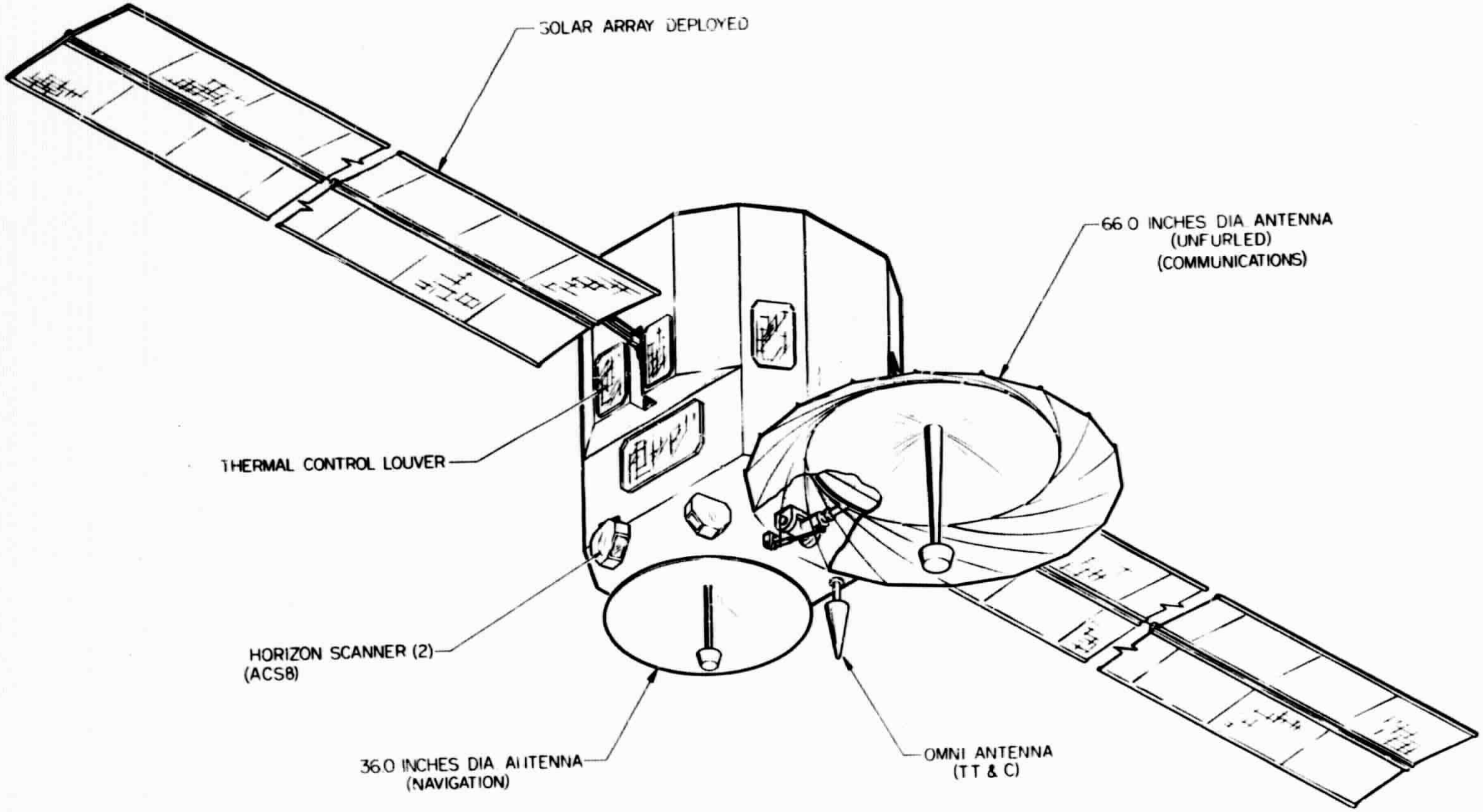


Figure 1. Navigation/Traffic Control Satellite

approximately 950 pounds (dry) and will provide four voice, two data, and one ranging channel. It will be similar to Configuration A, but will have greater communications capacity and therefore requires more power, thermal control, propellant, and the like.

All satellites will be three-axis stabilized spacecraft with an earth-oriented antenna and sun-oriented solar panels. Nitrogen gas jets provide the control capability. The structure will be partially cylindrical and partially rectangular, and will entail honeycomb mounting platforms for the various spacecraft components. Thermal control will consist of both active and passive techniques, e.g., heaters, insulation, louvers, and surface coatings. Propulsion will consist of approximately a 100-pound thrust liquid engine with an I_{sp} of 300 seconds. A gyro reference package is used for injection only. Pitch and roll control torque is provided by gimballing the engine during this phase. Earth sensors are employed for attitude sensing. Configurations A and C will use a pitch momentum wheel for yaw control, and Configuration B will use a yaw sun sensor and a yaw reaction wheel. All spacecraft will employ a high-gain antenna for voice and data communications and an earth coverage antenna for position determination. For electrical power, Configuration A will have an accordian foldout array designed to provide 300 watts of power at the end of three years. Configurations B and C will employ accordian foldout arrays designed to provide adequate power at the end of seven years.

1.2.1.2 Orbital Constellations

As described in Section 4 of this volume, the NTCS Design and Development/Preoperational Program uses two synchronous, equatorial Configuration A satellites with geostationary subsatellite points at 15 degrees and 56 degrees W longitude. The initial North Atlantic operational capability is attained with Configuration C satellites at those two points, plus a third Configuration C synchronous equatorial satellite with its geostationary subsatellite point at 35 degrees W longitude. As the system expands to worldwide operation, coverage in the North Atlantic becomes more complete and when four Configuration C equatorial, and one or two Configuration B inclined satellites are in view from the North Atlantic at all times, the coverage also includes the CONUS and Europe.

1.2.2 User Hardware

The user hardware analysis described in this report is limited to commercial aviation users. From a position determination viewpoint, a fairly wide range of users was examined in Reference 1, and from a communications standpoint, the user hardware for voice and data communications is much the same for all classes of users except that some users would not require as sophisticated input/output and display hardware as other users might require.

The resulting commercial carrier avionics package is an integrated satellite communications/navigation system that provides capabilities for communicating voice and printed messages as well as supplying aircraft position information in digital form to the ground terminal. The operational user hardware is described in Section 2.3. The D and D/Preoperational hardware approach is discussed briefly in Paragraph 3.3. In essence, the recommendation is made that the user hardware for the D and D/Preoperational phase be prototype and/or development flight hardware.

1.2.3 Ground Stations

The Design and Development/Preoperational phase of the NTCS Program will involve the setting up of remote tracking stations at Shannon and Gander, and a temporary Master Control Center at the Federal Aviation Agency's National Aviation Facilities Experimental Center (NAFEC), as well as obtaining operational support at NASA's Rosman and Goddard facilities. For the North Atlantic Ocean Area Operational phase, a remote tracking station at Ascension Island is added to the Shannon and Gander net, and a permanent master control center is set up at J. F. Kennedy Airport in New York. For worldwide coverage, two additional master control stations and three additional remote tracking stations will be required. The operational and D and D/Preoperational ground station approaches are described in Paragraphs 2.4 and 3.4 respectively.

1.3 MISSION ANALYSIS

1.3.1 Major System Requirements

Analyses were performed which developed North Atlantic communications and air traffic control surveillance requirements. The analyses which developed these requirements can be found in Volume II. Recommended air traffic control surveillance requirements for a 60-mile lateral separation in 1975 are:

- Accuracy: 1 nmi (1σ) position uncertainty
- Fix Rate: Subsonic - 1 fix per 80 to 100 sec
  Supersonic - 1 fix per 20 to 24 sec
  Total: 10,000 to 12,000 fixes/hr
- Provides: $\geq$5 observations of heading error blunders of $\geq$15 degrees

Recommended North Atlantic communications requirements are:

- Aircraft: 11 voice (peak load)
  1 emergency voice
  3 data
- Marine: 2 data
  1 emergency data
  Off-peak aircraft voice
- Search and Rescue: Preempt 1 or 2 aircraft voice.

1.3.2 Navigation/Traffic Control Satellite System Capability

The system described in this volume will more than meet the requirements just outlined. Furthermore, the performance margins over and above those requirements provide a great deal of operational utility, flexibility, and growth potential, and have been achieved at a very modest expenditure in terms of satellite system costs and complexity. The voice and data requirements postulated for the North Atlantic Ocean Area (NAOA) can be met with three Navigation/Traffic Control (NTC) Satellites. In the ultimate worldwide configuration, an aircraft in most of the North Atlantic, the Continental United States, or Europe would be able to see four synchronous equatorial, and one synchronous inclined satellite. This constellation would provide him with a fix with one sigma position determination uncertainties of approximately

- Latitude 70 feet
- Longitude 200 feet
- Altitude 140 feet

and a communications capability of seventeen voice and nine 1200 bit/sec data channels. With certain minor modifications to the user hardware, the system could also provide the user with 0.2 to 0.4 ft/sec velocity and rate of climb accuracy. If the inclined satellite is lost, all six elements of position and velocity are still available; but for aircraft in the North Atlantic, the altitude and rate of climb information suffer large geometric dilution of precision (GDOP) and is therefore the least accurate element of the position or velocity. If one of the four synchronous inclined satellites should also be lost, altitude and rate of climb information is no longer available; but, assuming aircraft altitude is known to several hundred feet, very good latitude and longitude information is still obtainable. Finally, if only two synchronous equatorial satellites are available excellent latitude and longitude information is available but the accuracy is time dependent. The user's clock will have been calibrated prior to takeoff, but cannot continually be automatically calibrated in flight as is the case when three or more satellites are visible. Clearly, with respect to position determination capability as well as communications capacity, the Navigation/Traffic Control Satellite System provides an operational capability of unprecedented quality; operational redundancy; and—in the event of a single satellite failure—an adequate operational capability.

1.4 REFERENCES

1. L. Jaffe, et al, "Final Report of the Ad Hoc Joint Navigation Satellite Committee," May, 1966.

2. OPERATIONAL NAVIGATION/TRAFFIC CONTROL SATELLITE SYSTEM

2.1 GENERAL

The Navigation/Traffic Control Satellite Mission Study involved the examination of a number of position determination techniques and communication subsystem approaches, both voice and data; and the selection of the most promising overall system design approach. This section describes that selected approach. The worldwide NTCS System is comprised of a number of multipurpose, communication-plus-navigation satellites in synchronous equatorial, and in synchronous inclined elliptical orbits, providing virtually worldwide coverage, but clearly favoring the Northern Hemisphere. The NTCS System is a high performance system in terms of accuracy and capacity. The traffic control system envisioned is virtually impervious to population growth.

The system designed is not yet optimized, however. Satellite design can clearly be improved with regard to weight and reliability, as well as flexibility of voice versus digital channel assignments. The orbital constellation selected, though very attractive, might well be replaced with the Aerospace Corporation "Y" orbits. Launch and injection strategy and the possible use of dormant spares in orbit have yet to be adequately investigated, but within the limitations imposed by time and funding, a very attractive system has been synthesized and will be described in the remainder of Section 2.

Phase II of the NTCS Program will provide initial coverage in the North Atlantic with three satellites in circular synchronous equatorial orbit as indicated in Figure 2 and as described in Paragraph 4.4.2 of this volume. During Phase III, the expansion to worldwide coverage, North Atlantic coverage will be provided by first four, and then six satellites. This growth is described in Paragraphs 4.4.3, 4.4.4, and 4.5 of this volume.

2.2 OPERATIONAL SPACECRAFT

2.2.1 Summary

The global NTCSS orbital configuration as currently envisioned will consist of eight satellites in synchronous equatorial orbits which provide the bulk of the communications services and also support the position and velocity determination function. Two or four smaller satellites will be placed in synchronous, elliptical, 52.5-degree inclined orbits to provide high latitude communications, altitude and rate of climb information, and increased position and velocity determination accuracy. The satellites in inclined orbits (Configuration B) are operational versions of the D&D satellites (Configuration A) to be described in Section 3.2. The synchronous equatorial operational satellite (Configuration C) will have a communications capacity of four voice and two data channels per spacecraft in addition to the navigation channel. The remainder of Section 2.2 will describe Configuration C, unless otherwise specified.

The Titan IIIB/Agena launch vehicle can be used to place the operational satellites (Configuration C) into synchronous equatorial orbits. A liquid bipropellant subsystem is incorporated into the satellite to accomplish the apogee injection. When launched aboard a Titan IIIB/Agena, approximately 2300 pounds can be placed into a synchronous equatorial transfer orbit. The proposed satellite design wet weight is estimated to be 2011 pounds including adapter and separation system and contingency. A summary of the satellite subsystem weight is given in Table 1.

Table 1. Satellite Weight Summary
(Configuration C)

Item	Weight (lb)
Structure and thermal control	98.2
Power supply	273.9
Electrical integration	80.0
Attitude control	93.7
Telemetry and command	29.3
Position determination and communication	90.2
Antennas	32.9
Propulsion	161.4
Contingency (10%)	86.0
SATELLITE DRY WEIGHT	945.6
Propellant and pressurant	1,015.0
Adapter and separation	50.0
BOOSTER PAYLOAD WEIGHT	2,010.6

A block diagram of the Configuration C satellite is given in Figure 3. The position determination and communications subsystem incudes the navigation signal generation function and the required voice and data communication channel capability. The low level output of the navigation channel portion of the transmitter is fed to either one of two redundant solid state amplifiers producing 50 watt peak (5 watt average) power at 1550 MHz. The output of the power amplifiers is fed to an earth coverage parabolic antenna (boresight gain 20.5 db). The communications portion of the subsystem has the capability to handle four voice (3 kHz) and two data (1200 bps) channels via two separate transmitters. The output of

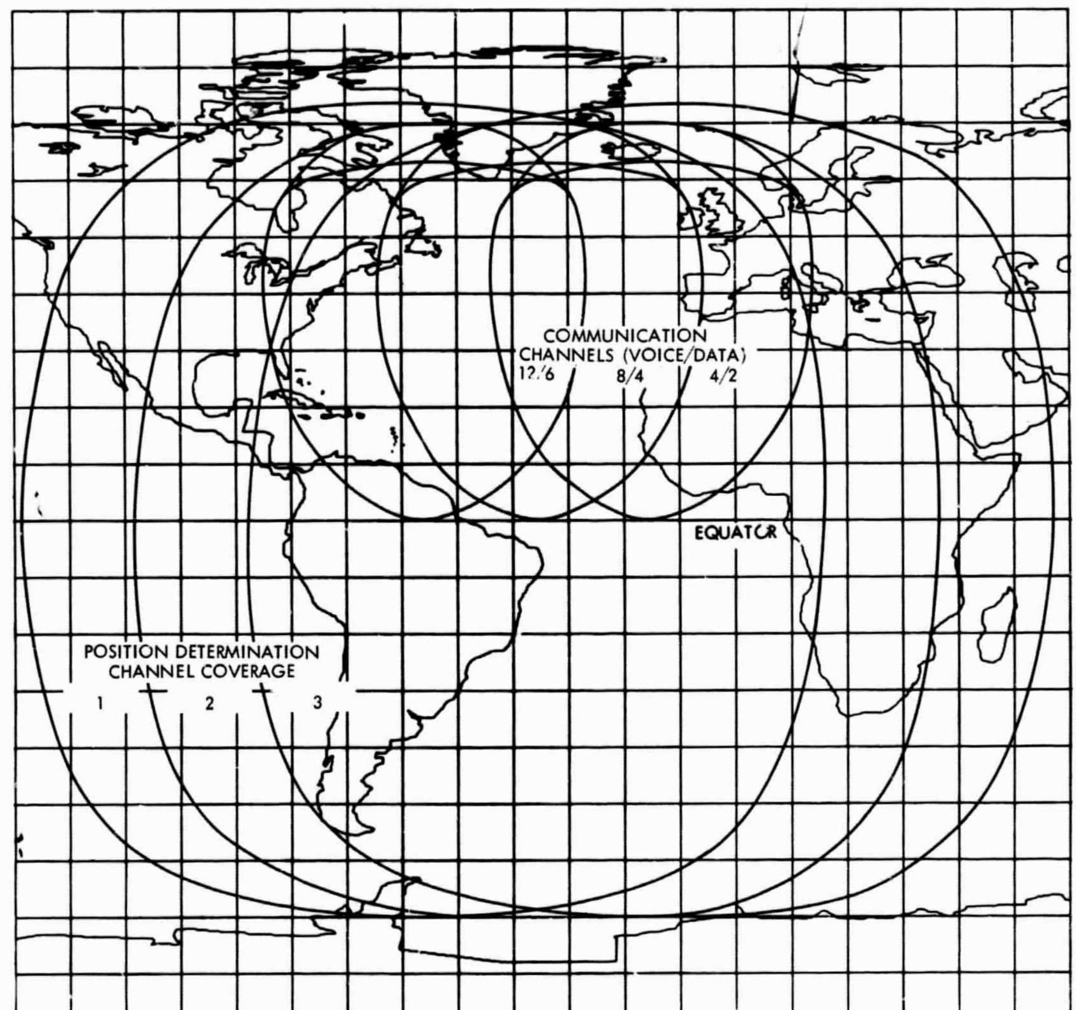
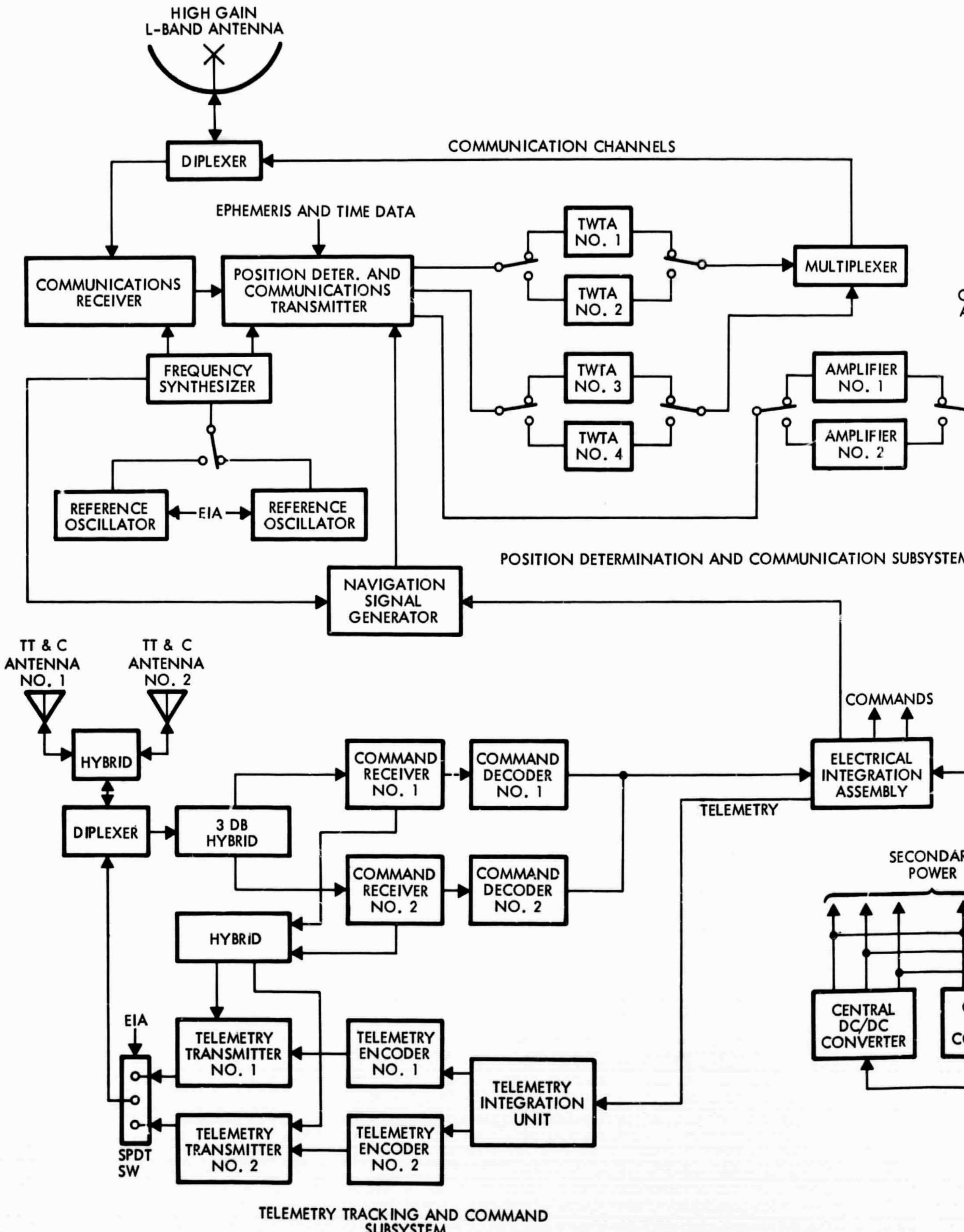


Figure 2. Phase II - Initial Operational Coverage


HIGH GAIN
L-BAND ANTENNA
DIPLEXER
COMMUNICATION CHANNELS
EPHEMERIS AND TIME DATA
COMMUNICATIONS RECEIVER
POSITION DETER. AND COMMUNICATIONS TRANSMITTER
TWTA NO. 1
TWTA NO. 2
TWTA NO. 3
TWTA NO. 4
MULTIPLEXER
AMPLIFIER NO. 1
AMPLIFIER NO. 2
FREQUENCY SYNTHESIZER
REFERENCE OSCILLATOR
EIA
REFERENCE OSCILLATOR
POSITION DETERMINATION AND COMMUNICATION SUBSYSTEM
NAVIGATION SIGNAL GENERATOR
TT & C ANTENNA NO. 1
TT & C ANTENNA NO. 2
HYBRID
DIPLEXER
3 DB HYBRID
COMMAND RECEIVER NO. 1
COMMAND DECODER NO. 1
COMMAND RECEIVER NO. 2
COMMAND DECODER NO. 2
HYBRID
COMMANDS
ELECTRICAL INTEGRATION ASSEMBLY
TELEMETRY
EIA
TELEMETRY TRANSMITTER NO. 1
TELEMETRY ENCODER NO. 1
TELEMETRY INTEGRATION UNIT
TELEMETRY TRANSMITTER NO. 2
TELEMETRY ENCODER NO. 2
SPDT SW
CENTRAL DC/DC CONVERTER
TELEMETRY TRACKING AND COMMAND SUBSYSTEM

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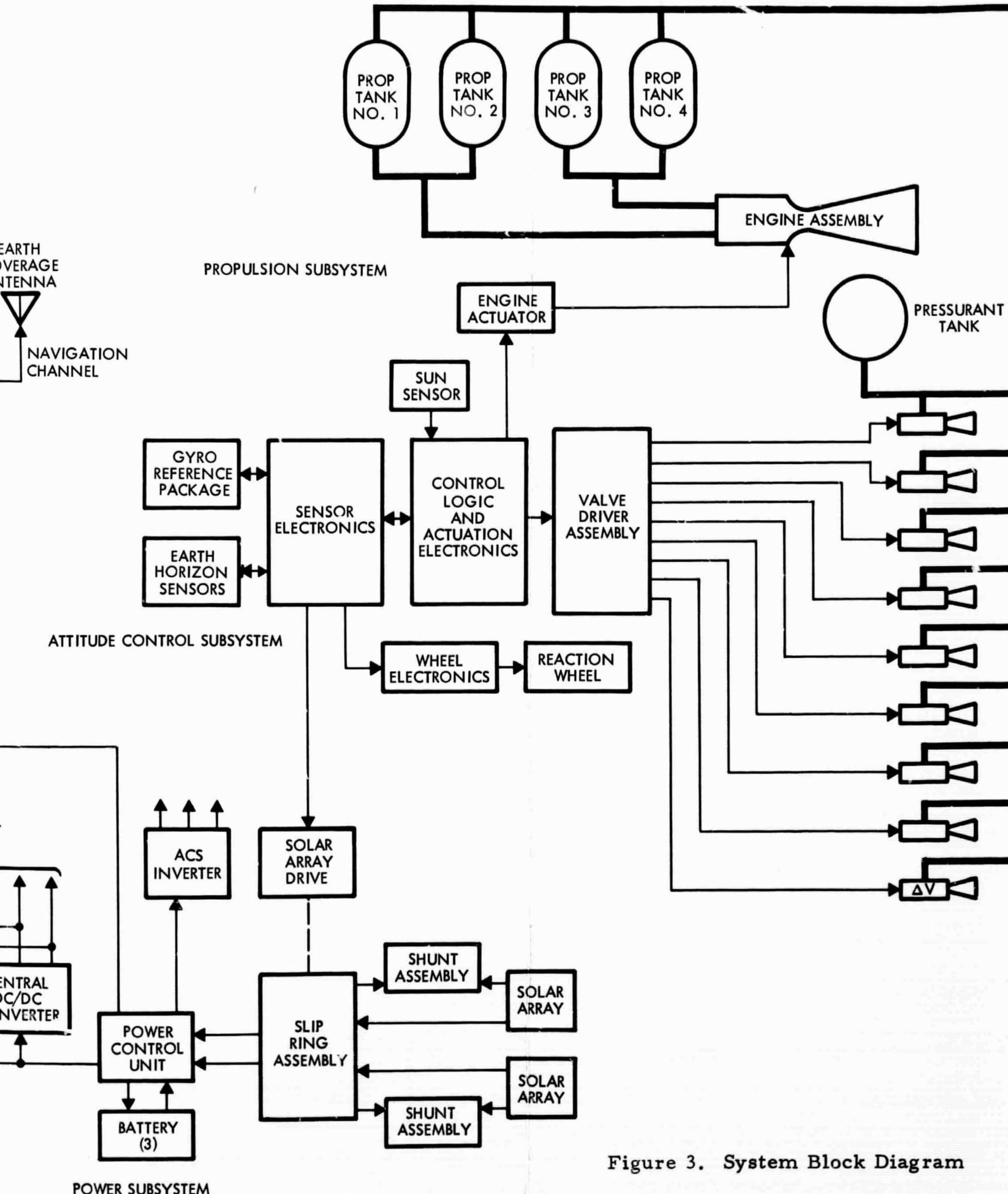

Figure 3. System Block Diagram

B

each transmitter, consisting of two voice and one data channel, is fed to one of two redundant traveling wave tube (TWT) amplifiers. These amplifiers have 100 watt output at L-band (40 watt per voice channel and 20 watt per data channel). The two separate RF carriers from the two traveling wave tubes are diplexed and fed to a high gain (23 db beam edge) L-band antenna.

The telemetry, tracking, and command subsystem (TT&C) is compatible with the USBS and incorporates full redundancy for receivers, decoders, transmitters, and encoders. Full omnidirectional antenna coverage is provided by two radiating elements connected to the TT&C subsystem by a hybrid. One antenna is mounted forward on the spacecraft (conical spiral) and the other on the aft end (Archimedean spiral).

The attitude control subsystem (ACS) makes use of a pitch momentum reaction wheel in combination with cold gas nitrogen thrusters. A gyro reference package is used to provide orientation reference during the transfer orbit to synchronous equatorial altitudes.

The propulsion subsystem uses a hypergolic bipropellant (nitrogen tetroxide and monomethylhydrazine) providing approximately 100 pounds of thrust at a specific impulse of 300 seconds. The nitrogen pressurant used for the propellant tanks is used in the attitude control system.

The satellite power supply is a combination solar cell/rechargeable nickel-cadmium battery system. The solar arrays are composed of N-on-P silicon cells in an accordian foldout configuration; approximately 1400 watts can be supplied by the solar array assembly at beginning of life, although refinements in the analysis late in the study indicated that only 1195 watts will be needed. Three nickel-cadmiun batteries are used to provide full eclipse operation and operation during the transfer orbit (before the solar array is deployed). A summary of the satellite subsystem power requirements is given in Table 2.

2.2.2 Configuration Tradeoff Analyses

As noted above in Section 2.1, the operational NTCSS consists of satellites placed in two different orbit planes. For the geostationary orbits, considerable operational and design experience has been accumulated on such programs as ATS, the INTELSAT series, and TACSAT.

Table 2. Summary of Satellite Power Requirements (Configuration C)

Subsystem	Power (watts)
Attitude control	31.1
Power subsystem*	29.5
Electrical distribution	8.2
Telemetry and command	43.8
Position determination and communications	611.0
SUBTOTAL	723.6
Contingency (5%)	36.2
TOTAL EQUIPMENT REQUIREMENT AT SOLAR ARRAY	759.8

*Does not include power required for battery charging.

Consequently it is logical to consider these satellite designs as candidates for NTCSS. The highly inclined orbits, however, are unique, and they will require a different design than the standard dual spin configurations identified above. The extent of these design differences and the reasons for them will be discussed in further detail later in this section. The fact that the dual spin design is not readily adaptable to operating in the inclined orbits leads to the first tradeoff consideration — should the operational NTCSS consist of two separate satellite/booster designs or is it feasible to utilize a single spacecraft design for both orbits? The potential advantages of a single design in terms of minimum cost, minimum replenishment problems, and in terms of achieving maximum utilization of the R&D design are obvious. In addition, three particular spacecraft subsystem design tradeoffs — attitude control system, antenna, and solar array — received special emphasis and are treated in detail in Volume II, Section 6.

It is clear that any configuration which operates satisfactorily in the inclined orbits will also be acceptable for a single spacecraft design. Of the several spacecraft design approaches, the choice is basically whether a large angular scan antenna is preferred over an oriented solar array. Oriented solar arrays have been developed on such spacecraft as OGO and Nimbus. A large angular scan spacecraft antenna has never been developed or flown. Hence the oriented solar array appears to be the preferred approach. Similarly, the design of an oriented array mounted on a despun platform has never been established, whereas the 3-axis stabilized design has been flown on the spacecraft noted above. Furthermore, as indicated in Volume II, Section 6, for the projected power requirements, the 3-axis stabilized design enjoys a significant weight advantage. Therefore, the 3-axis stabilized design has been selected as the optimum approach for the NTCSS.

2.2.2.1 Configuration Description

As a result of the tradeoff analyses described in Volume II, Section 6, a 3-axis stabilized spacecraft configuration was developed. Configuration analyses on the selection and placement of the spacecraft antennas and selection of solar array design were performed and are given in Volume II, Section 6. The proposed configuration is shown in Figure 4.

The communications antenna is stowed upright and deployed 90 degrees to be operational. The center rigid part of the antenna is of a lightweight aluminum honeycomb. The rim frame for attachment of the aluminum wrap-around ribs adds rigidity to the antenna and anchors the Mylar covering for the unfurlable outer antenna section. Ribs are spaced at 9 degrees. A strap is used to hold the ribs secure during launch and is cut in two places by pyrotechnic strap cutters at deployment. A detail of this unfurlable antenna perimeter is shown in Figure 4. The large antenna is placed on the centerline of the satellite and rotates 90 degrees to deploy along this same centerline. The placement of the antenna shown allows a nearly 180 degree clear view for the horizon scanners that are placed at 90 degrees to each other. The earth coverage antenna is of aluminum honeycomb construction and is attached by three tubular legs to the body of the spacecraft. A detailed weight estimate for both of the antennas is given in Volume II.

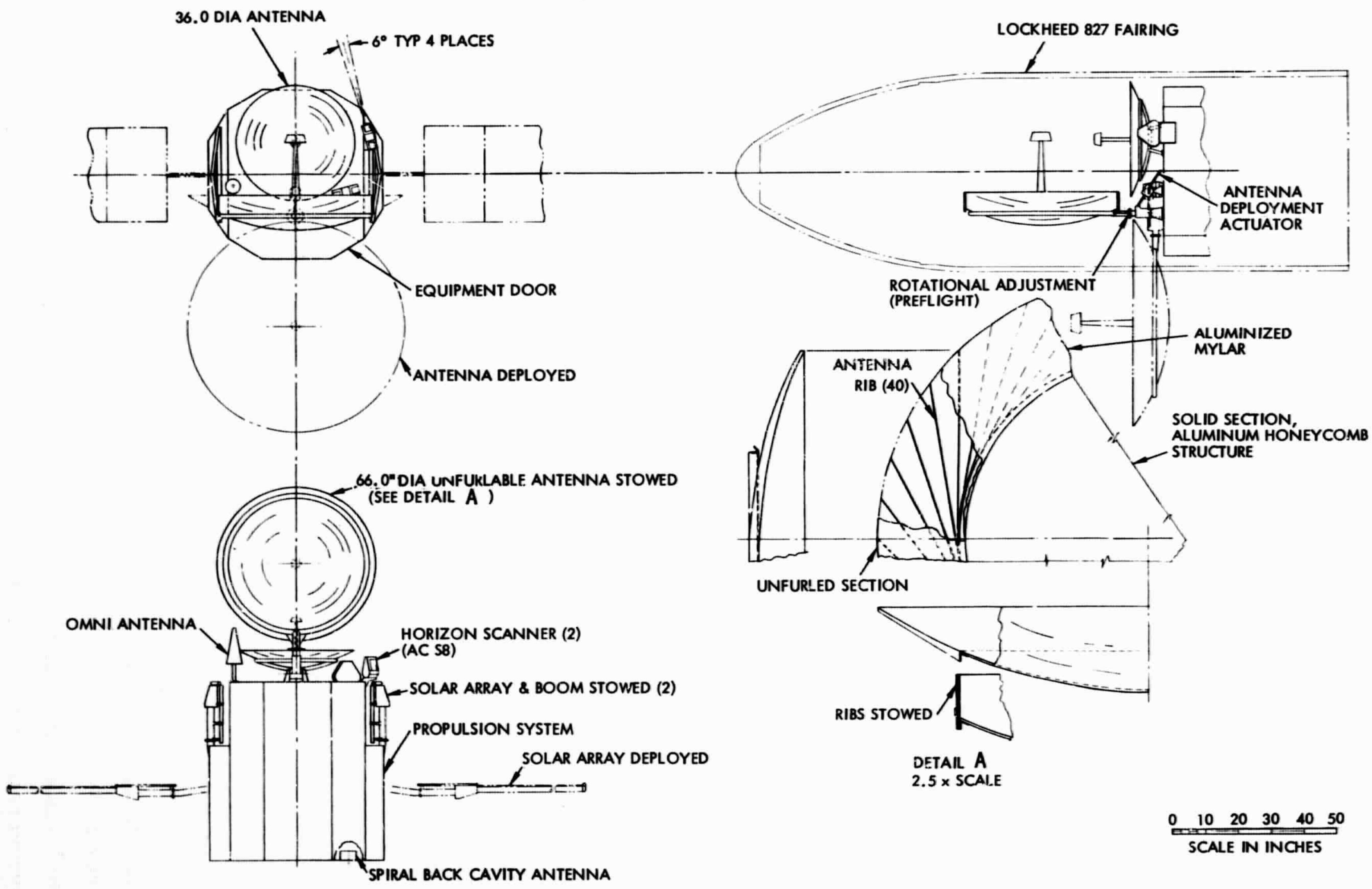

Figure 4. NTC Satellite (Configuration C)

Thermal control of the spacecraft equipment is also an important consideration which is facilitated by the use of the three-axis system. Only the sides of the satellite facing the arrays are without sun exposure. High heat dissipation units must therefore be mounted on these surfaces. The entire external area must be covered by louvers or insulation except for sensors, jets, etc.

The propulsion system is readily enclosed by a 12-sided polygon which is continued above the propulsion section for equipment storage except for an area on each side left for stowage of the solar arrays. The satellite can be enclosed by the Lockheed 827 lightweight fairing as shown in the figure.

Figure 5 furnishes further details of the satellite and an inboard profile of the propulsion subsystem. Some equipment items identified by the code given in Table 3 are placed on a bulkhead above the propulsion subsystem. This bulkhead can be perforated as required to allow heat interchange between the propellant and equipment. Two batteries and some equipment units associated with the propulsion system are mounted below the bulkhead. The bulk of the equipment is mounted above the propulsion subsystem flat against the exterior panels for the best cooling. The panels facing the solar arrays are held in place by two internal aluminum truss assemblies which also support the top panel, the small antenna, horizon scanners, and other external top-mounted equipment. The end panels above the propulsion system are hinged to provide access to the equipment. Units of adjacent equipment are located on the basis of minimum length connecting harness runs. Sufficient room is left between units for the connecting harness.

The solar array mechanical design is shown in Figure 6 and described in detail in Volume II.

2.2.3 Position Determination and Communication Subsystem (PD&CS)

The block diagram of the subsystem is shown in Figure 7. The transponder is used to relay voice and data signals between aircraft and ground stations. In addition, a navigation signal, together with accurate time of the day and internally stored ephemeris and clock update data, is generated and transmitted by the satellite.

Reception and transmission is accomplished simultaneously on a high gain antenna through the use of a diplexer. Transmitted and received signal frequencies are spaced 100 MHz apart so that the diplexer can provide

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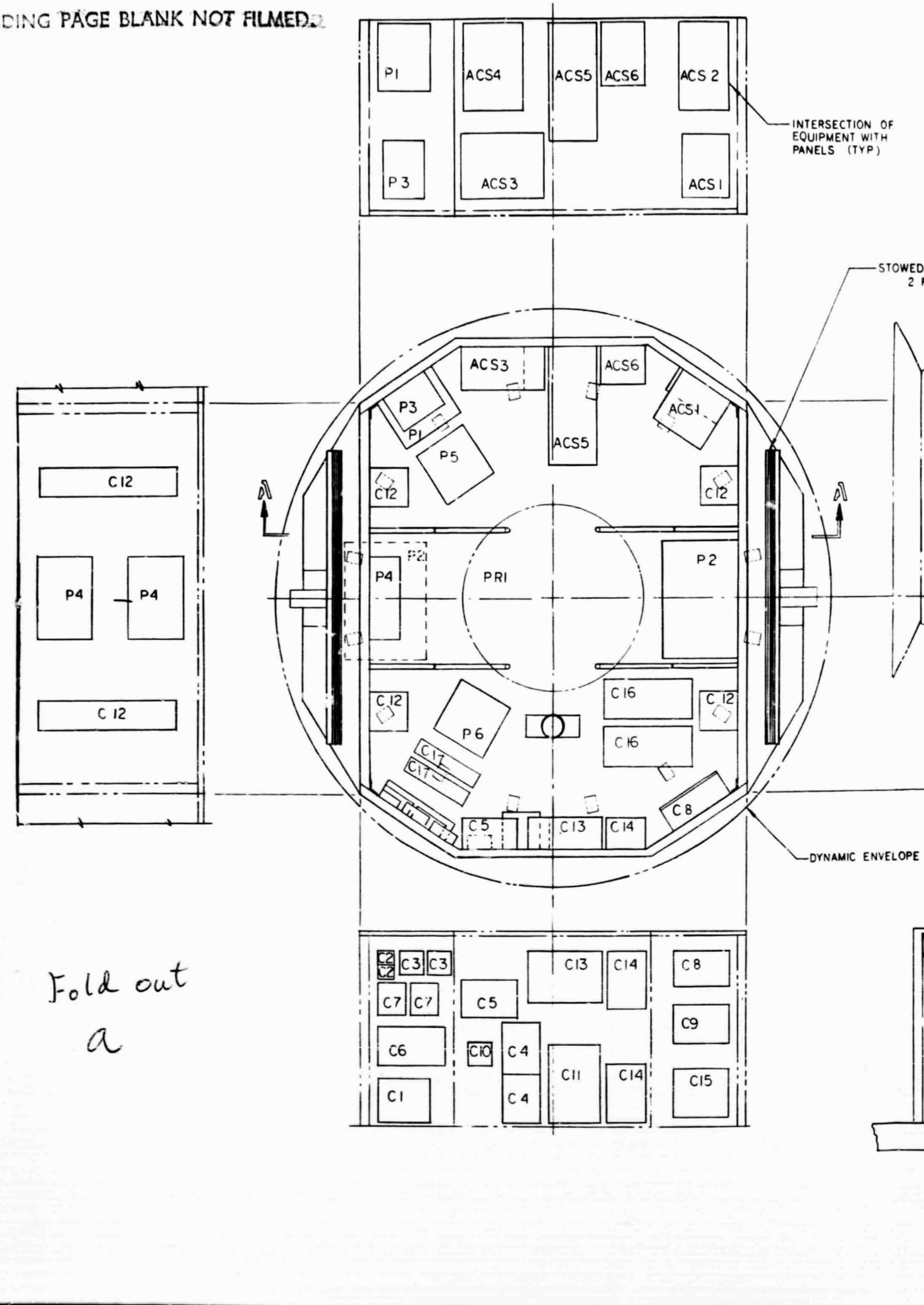

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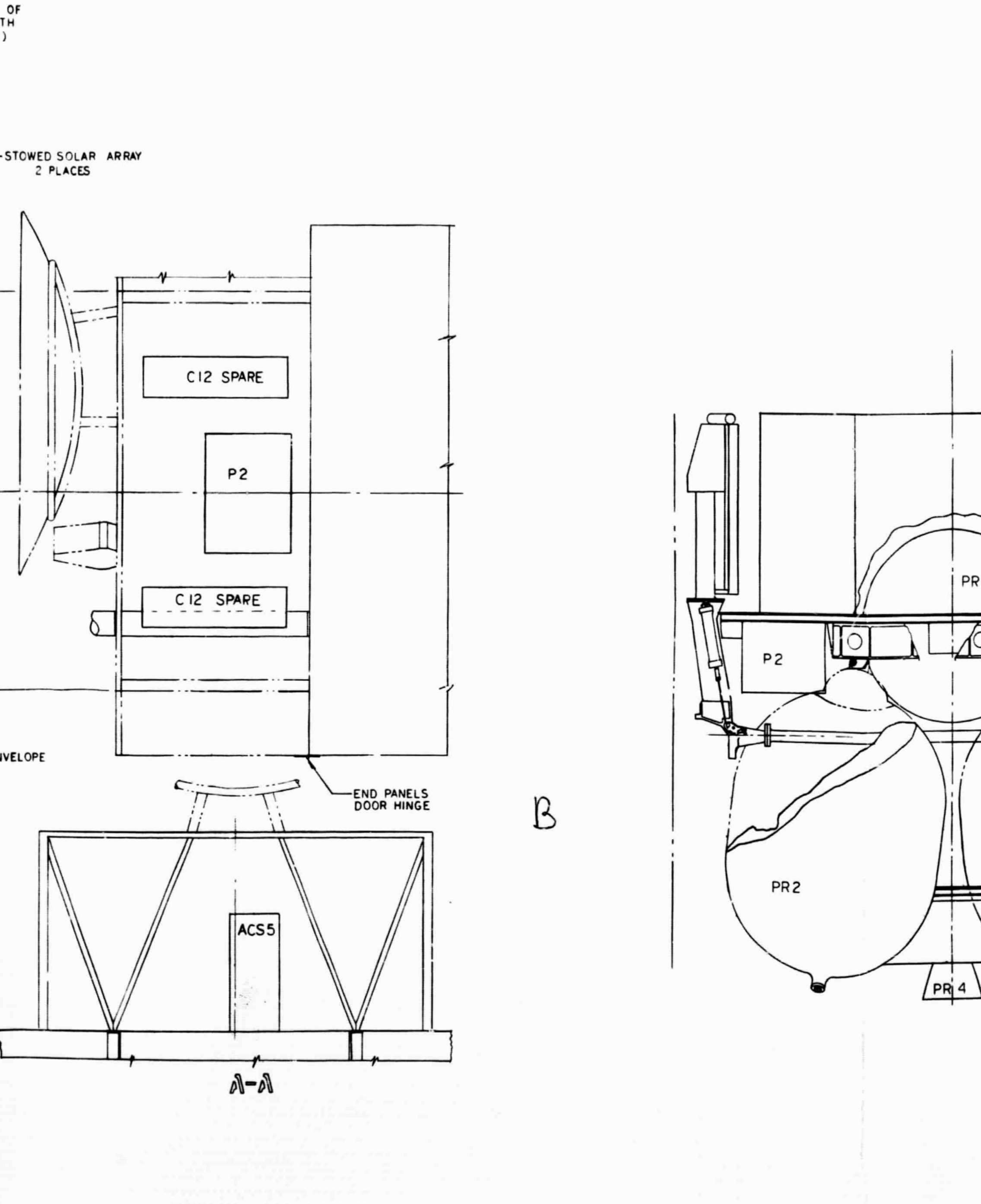
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TH
STOWED SOLAR ARRAY
2 PLACES
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P2
C12 SPARE
NVELOPE
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ACS5
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PR
P2
PR2
PR 4

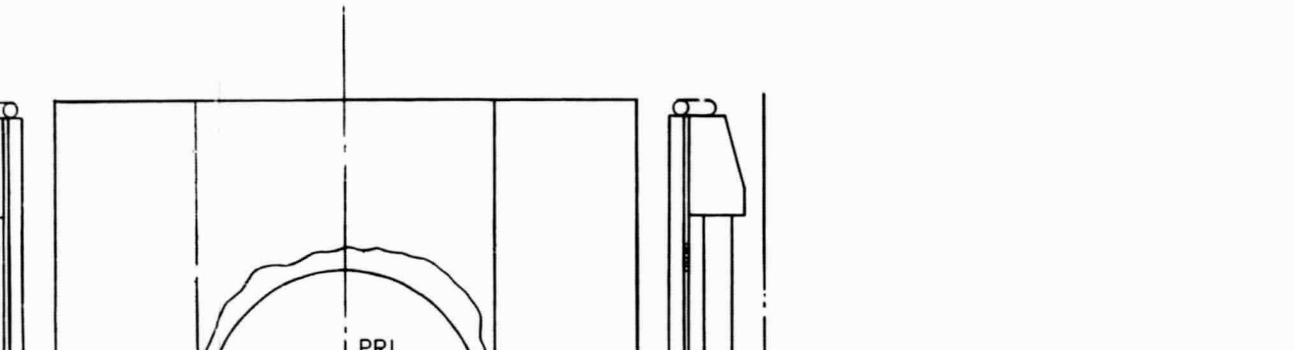

Figure 5. **Inboard Profile NTC Satellite (Configuration C)**

C

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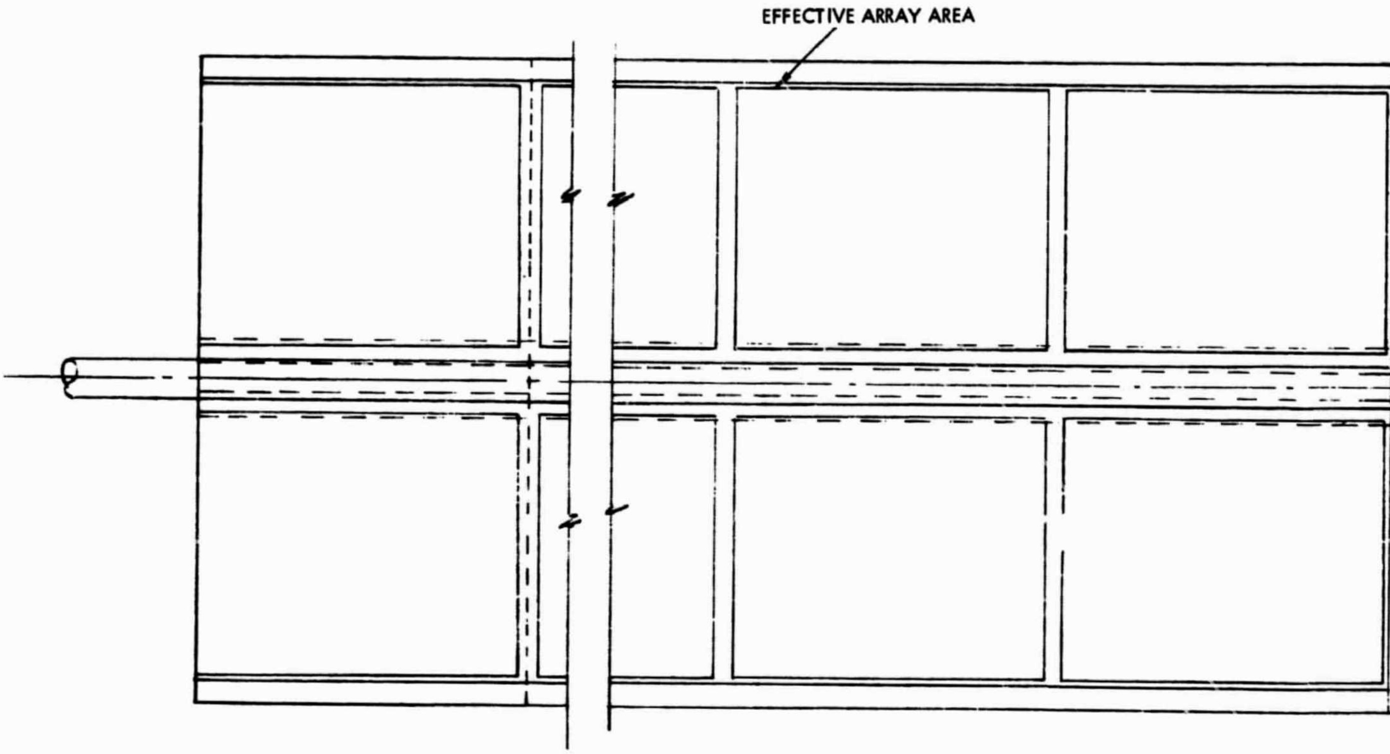

SOLAR ARRAY

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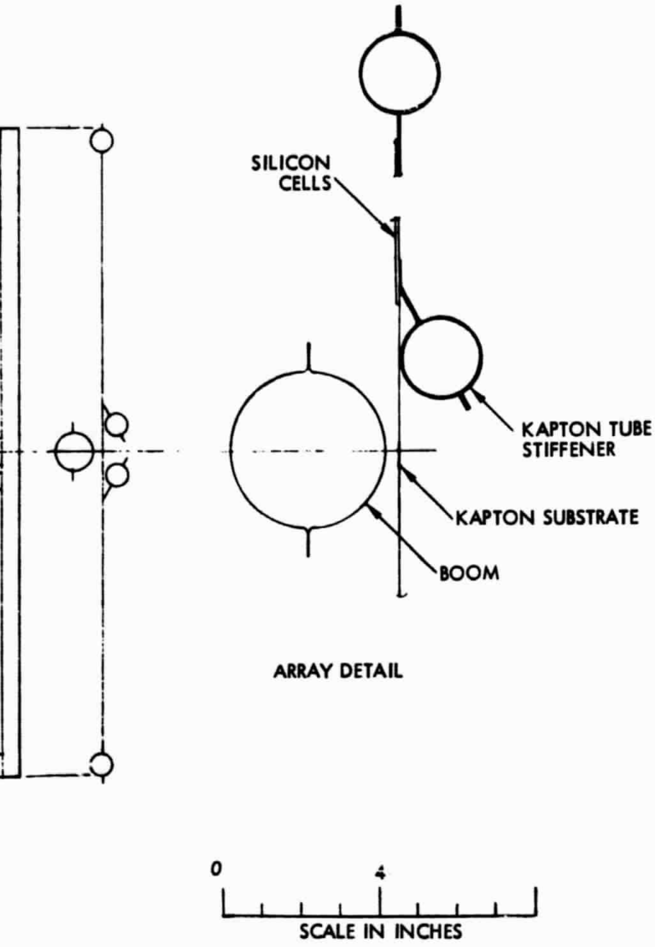

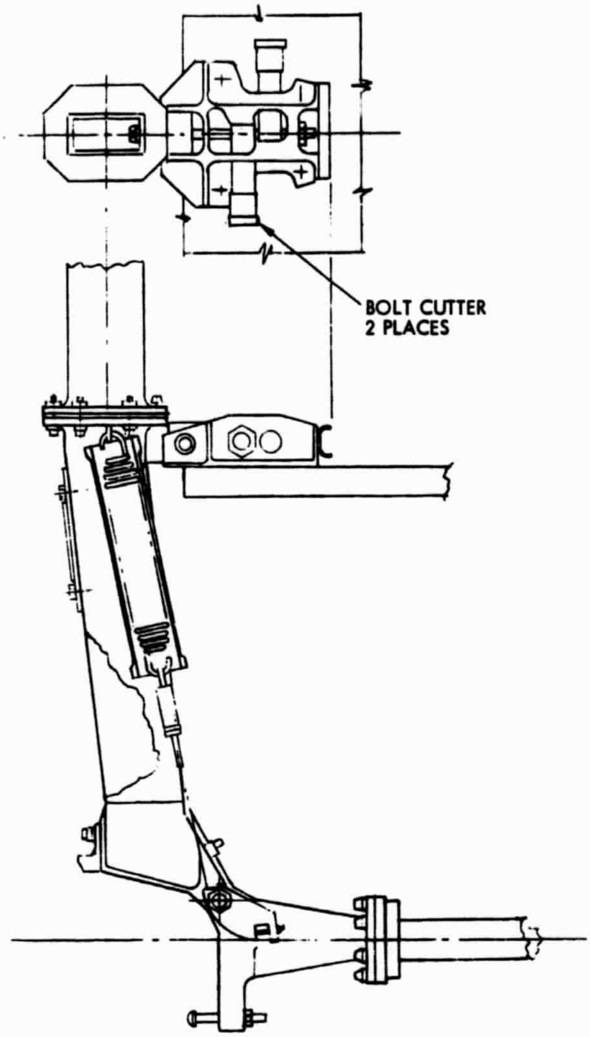

Figure 6. Solar Array Detail, NTC Satellite (Configuration C)

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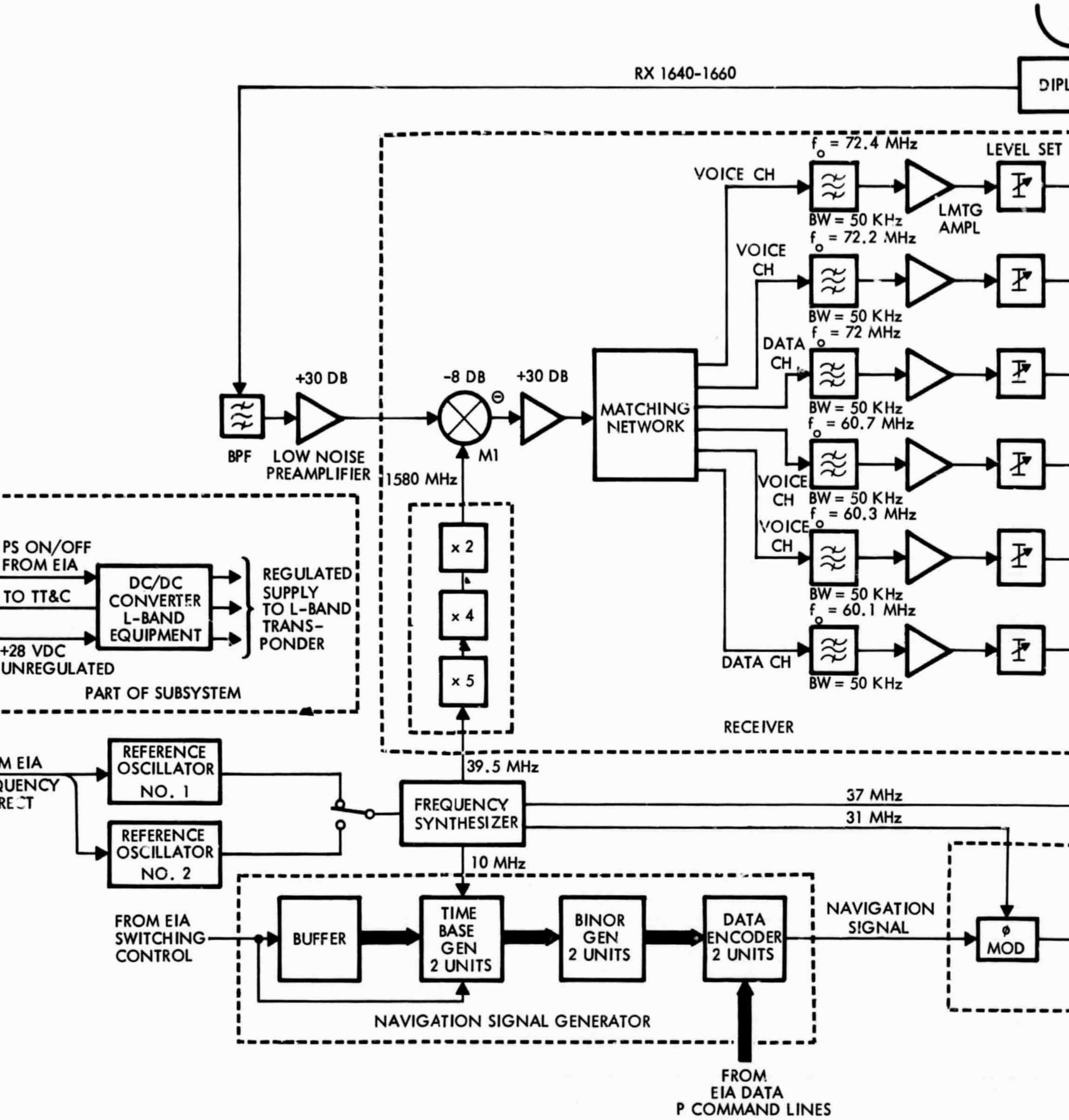

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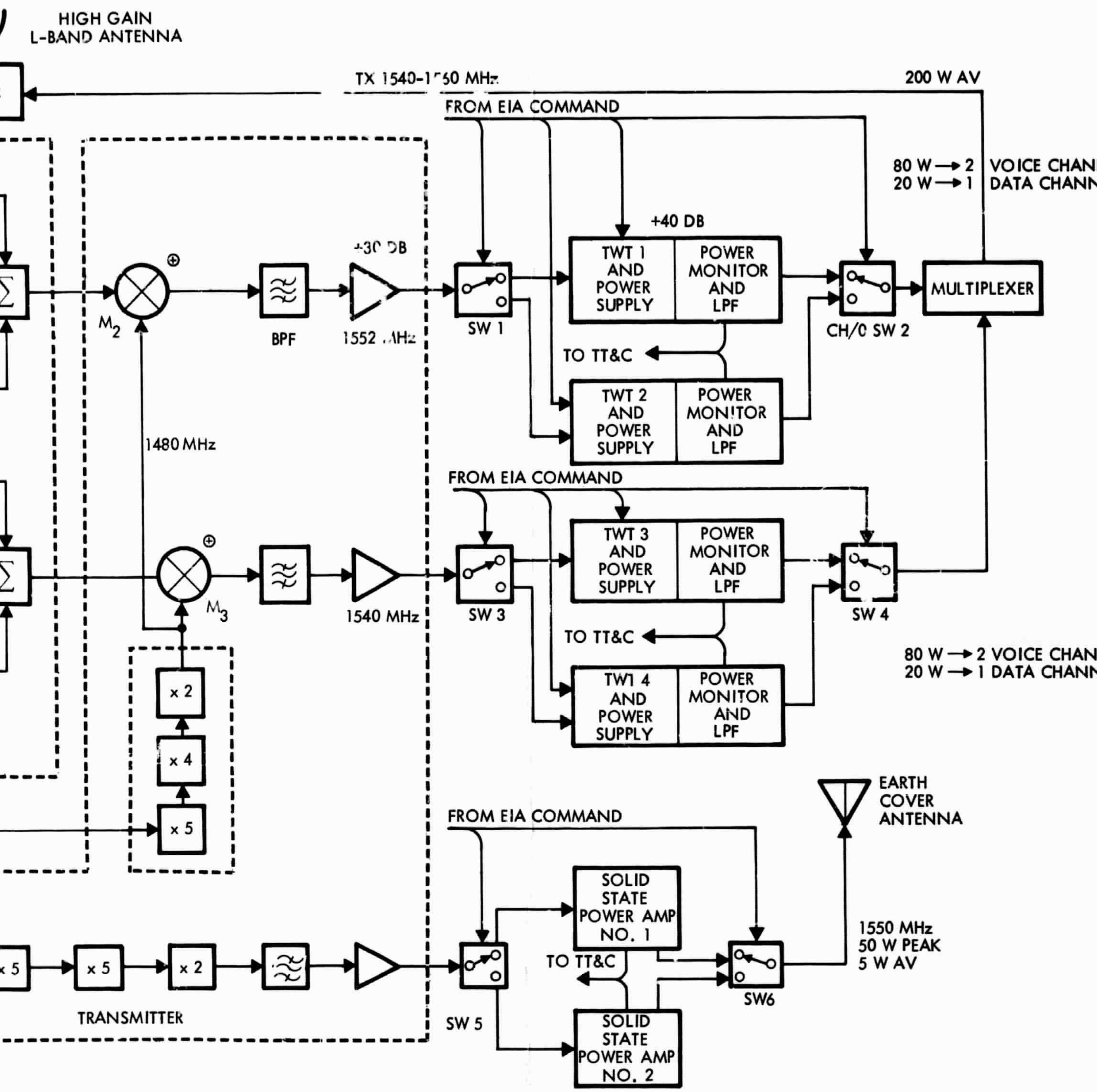

Figure 7. Position Determination and Communication Subsystem Block Diagram

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Table 3. Navigation Traffic Control Satellite Equipment Identification Code

Code	Equipment
C1	Diplexer (S-band)
C2	Hybrid
C3	Command receiver
C4	Command decoder
C5	Telemetry transmitter
C6	Telemetry encoder
C7	Receiver selection
C8	L-band diplexer
C9	Frequency multiplexer (L-band)
C10	Isolator
C11	Transmitter
C12	TWT and power supply
C13	Receiver
C14	Solid state power amplifier
C15	Frequency synthesizer
C16	Reference oscillator
P1	Power control unit
P2	Battery
P3	Central dc/dc converter
P4	Shunt Assembly
P5	Electrical integration unit
P6	Telemetry integration module
ACS1	Gyro reference assembly
ACS2	Sensor electronics
ACS3	Control electronics
ACS4	Actuation electronics
ACS5	Reaction wheel
ACS6	Reaction wheel electronics
ACS7	Sun sensor assembly
ACS8	Horizon sensor assembly
ACS9	Solar array drive
PR1	Pressurance (N) tank
PR2	Oxidizer (N_2O_4)
PR3	Fuel (MM_H)
PR4	Engine assembly

adequate signal isolation. Four voice and two data channels are received on carriers located in the frequency band from 1640 to 1660 MHz. The carriers are received and converted to IF where they are separated, amplified, and, in turn, recombined into two groups, stepped up to L-band (1540 to 1560 MHz) and retransmitted via two TWT's. The outputs from the two TWT's are frequency combined in the frequency selective multiplexer and in turn applied to the diplexer. The input to the TWT's are spaced 12 MHz apart so that the design of the high power multiplexer is eased. The voice channels are transmitted at 40 watts per channel while the data channels are transmitted at 20 watts per channel. The navigation channel is transmitted on another carrier (1550 MHz) using a separate earth coverage antenna at 50 watt peak or 5 watt average output power.

A tentative frequency plan showing the frequencies received and transmitted by the satellites is shown in Figure 8. Transmit/receive frequency parts are spaced 100 MHz apart as shown in the figure. The satellite receive frequencies are in the 1640 to 1660 MHz band while satellite transmit frequencies are in the 1540 to 1560 MHz band. Within these bands four frequency groups have been designated as groups A, B, C, and D. Each group consists of two, 2-MHz wide bands which are spaced 12 MHz apart. Each band contains two voice and one data channel carriers which are spaced apart in steps of 100 kHz. Consequently, a total of 40 channel frequency assignments are available within each frequency group. All channels will not be utilized, however, since the voice and data channels will be spaced within each group in a manner which minimizes system IM problems. The two bands in each group are transmitted separately by the two TWT amplifiers in the transmit portion of the satellite transponder. For the navigation signal transmissions all satellites transmit at a frequency of 1550 MHz. Since the navigation signal bandwidth will be no greater than 2 MHz, the band from 1549 to 1551 MHz is reserved for the navigation signal.

The block diagram of the transponder in Figure 7 assumes a satellite operating in frequency group A. For example, the frequency assignments of the three carriers within each 2-MHz wide band could be located such that a data channel is transmitted by the satellite at a frequency of 1540. 1 MHz, 2 voice channels at frequencies of 1540. 3 and 1540. 7 MHz, another data channel at 1552 MHz, and finally two more voice channels at 1552. 2 and 1552. 4 MHz. Of course, other satellites in the network would operate either in a different frequency group or in frequency group A but with different frequency assignments for the six voice and data channels.

Since the equipment is to be operative for a period of at least five years, redundancy is provided in each case where reliability and life expectancy of individual components would limit the operation of the system. The TWT's with a life expectancy of 30, 000 to 50, 000 hours may establish the satellite

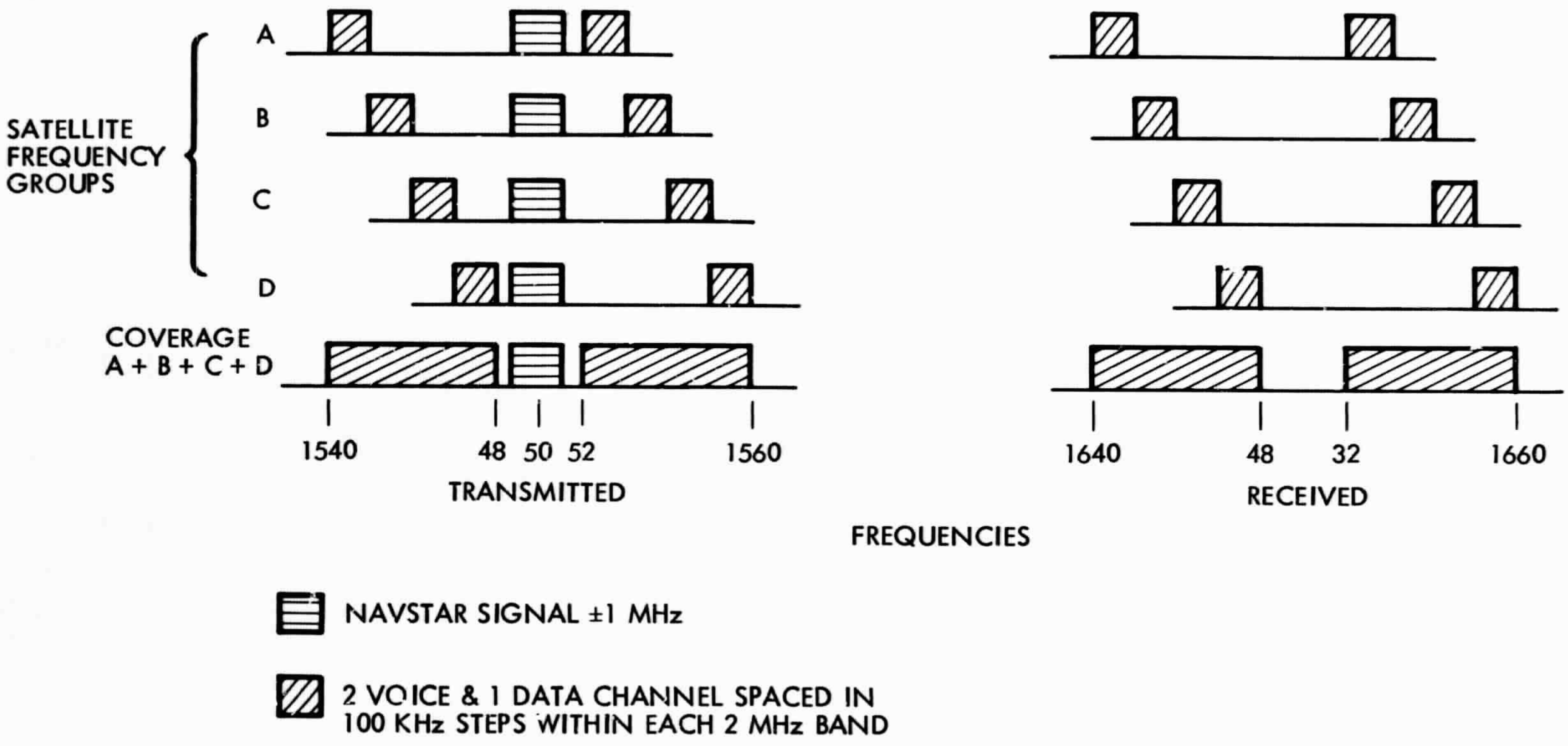


Figure 8. Frequency Plan for Satellite Network

life. One element predominantly determines the life of the TWT — the cathode coating. During continuous operation, a partial cathode coating depletion takes place followed by a decrease in power. To alleviate this problem, redundant TWTs are provided, the redundant unit being left on cold standby. The transfer can be accomplished either automatically by means of the changeover switches or by ground station command via TT&C.

A high stability crystal-controlled oscillator is proposed. Since the unit provides all the timing and reference signals for the spacecraft navigation signal generator and L-band carriers, one-to-one redundancy is required to meet the ultimate reliability requirements. Both units will be running continuously so that both units will be subjected to the same aging effects.

The solid state power amplifier associated with the navigation carrier has a peak power of 50 watts and is provided with a cold standby unit which is switched in by means of two changeover coaxial switches. The changeover is automatic in the case of RF power failure or a ground command via TT&C S-band link.

2.2.3.1 L-Band Diplexer

The L-band diplexer is a frequency selective unit comprised of two bandpass filters to permit simultaneous transmission and reception on the same antenna.

An additional bandpass filter following the diplexer is necessary to further reduce the onboard-generated RF power. This combination provides sufficient attenuation of the transmitter signals so that the receiver is not desensitized by the high power transmitter.

The unit specifications are:

- L-Band transmit frequencies: 1540 to 1560 MHz
- L-Band transmit power: 100 watts per TWT
- Received frequencies: 1640 to 1660 MHz
- Isolation between transmitted and received signals: 100 db minimum
- Passband insertion loss: 0.3 db maximum
- Power handling: 150 watt minimum
- Input/output impedance: 50 ohms
- VSWR: 1.2 maximum
- Connectors: TNC

- Operating temperature: $-30^{\circ}C$ to $+70^{\circ}C$
- Size: 10 x 4 x 3.5 inches maximum
- Weight: 3.0 pounds maximum

2.2.3.2 <u>Low Noise Preamp</u>

The low noise amplifier that follows the diplexer is a wideband 30 db gain L-band preamplifier with a maximum noise figure of 5.5 db. Although the unit is capable of operating in the frequency range 1 to 2 GHz, it has been optimized for the lowest noise figure at the operating frequencies 1640 to 1660 MHz.

2.2.3.3 <u>Receiver</u>

The receiver consists of the following elements:

- Mixer
- Multiplying chain (40X)
- Channel separating crystal bandpass filters
- IF amplifiers
- Level setting and summing networks

The received signal is filtered and amplified before being applied to a balanced low-distortion mixer with approximately 8 db of insertion loss. The local oscillator for the mixer (1580 MHz) is derived from the reference oscillator via the frequency synthesizer and the multiplying chain (40X). The frequency synthesizer produces 39.5 MHz which is multiplied by 5 x 4 x 2, then filtered and amplified to approximately +3 dbm. The 60 to 72 MHz IF signal is further amplified and passed to a matching network. This network drives six channel-selecting crystal bandpass filters, possessing a 60 db minimum rejection to the adjacent channels. Each filter is followed by an IF amplifier with a gain such that each channel will be limited to 0 dbm prior to being fed to the summing networks.

The voice and data channels are divided into two groups with a 12 MHz separation between groups. Each group contains 2 voice and 1 data channel which are spaced in steps of 100 kHz. The voice and data channels have 25 kHz and 5kHz information bandwidths, respectively. However, each transponder channel will be filtered in a 50 kHz wide band to accommodate both the information bandwidth required and uncertainties due to doppler shift and transmitter oscillator stability.

2.2.3.4 Transmitter

The outputs of each of the summing networks are applied to their respective L-band mixers where the two IF groups are translated to L-band. Both mixers are driven from a common local oscillator multiplier (1480 MHz) which derives its input from the reference oscillator and frequency synthesizer. The 37 MHz output from the synthesizer is multiplied (5 x 4 x 2) then filtered and amplified to approximately +3 dbm. The mixers, which are identical to those in the receiver unit, are followed by bandpass filters for selecting the appropriate upper sidebands. The TWT driver amplifiers that follow are solid-state devices exhibiting 30 db of gain and capable of 30 mw of output drive level at L-band.

2.2.3.5 Power Amplifier TWT

The power amplifiers receive their inputs from the solid-state drivers via the coaxial changeover switches. Each power amplifier is capable of handling two voice and one data channel at a total output power level of 100 watt average. Each unit operates in the saturated mode, exhibiting a beam efficiency of 45 percent and overall dc to RF efficiency of 36 percent. Each power supply is capable of being switched on and off by a command signal from the electrical integration assembly (EIA) unit. The output filter rejects the undesired harmonics, while the output power monitor provides a direct indication of the RF output power in the form of a dc signal. A 3 db loss in power or a complete RF power failure will automatically switch in the redundant TWT. This can also be accomplished via a ground command. Space qualified 100 watt units at S-band have been developed by both Watkins-Johnson and Eimac. Both companies claim that a minor scaling-down effort is required (6 months redesign, $70,000) for conversion to L-band operation.

The unit specifications are:

- Operating frequency: 1540 to 1600 MHz
- Power output (saturated): 100 watts minimum
- Gain: 40 db minimum
- Bandwidth: 10 MHz minimum (0.5 db)
- RF to dc efficiency: 36 percent minimum
- Input/output impedance: 50 ohms

- VSWR input/output: 1.13 maximum
- Noise figure: 35 db maximum
- Harmonics output: not less than 60 db down with respect to unmodulated carrier
- Spurious outputs: not less than 80 db down in any Hz band in the frequency range of 1500 to 1600 MHz band
- Operating lifetime: 50,000 hours minimum
- Operating temperature range: -30°C to 80°C (base temperature)
- Vibration (operating): 20 G's
- Shock (nonoperating): 200 G's 1 ±0.5 msec
- Weight: 10 pounds maximum
- Size: 15.0 x 6.0 x 4.0 inches maximum
- Telemetry: (a) Output power maintenance +2 v dc (150 w)
  (b) Helix current +2 v dc
  (c) Temperature +2 v dc
  (d) Beam voltage +2 v dc

2.2.3.6 Changeover Switch

The switches that precede and follow the TWTs are coaxial single-pole double-throw fully-latching types, i.e., they do not require input power to maintain a given state once switched. The units are commanded by the EIA in connection with the TWT power monitors or ground command. The combination of the two switches allows the redundant TWT to be switched in or out.

2.2.3.7 L-Band Multiplexer

The purpose of the unit is to frequency multiplex the output of the two L-band 100 watt power amplifiers. The unit specifications are:

- Two L-band input frequencies: 12 MHz apart in band 1540 to 1560 MHz
- Frequency response: passband flatness ±0.5 db maximum
- Isolation between 1540 and 1560 MHz: not less than 40 db
- Passband insertion loss: 0.5 db

- Power handling: 150 watt minimum
- Input/output impedance: 50 ohm
- VSWR: 1.2 maximum at each port
- Connectors: TNC
- Operating temperature: $-30^{\circ}C$ to $+70^{\circ}C$
- Size: 10 x 4 x 3.5 inches maximum
- Weight: 3.0 pounds maximum

2.2.3.8 Reference Oscillator Assembly

This unit provides an ultra-stable reference signal from which the clock and all the timing pulses for range and range rate measurements are derived. The oscillator operates at 5 MHz and utilizes a third overtone crystal.

Integrated circuitry is used throughout the package in order to provide reliability and miniaturization. To achieve an ultimate stability of 1×10 pp 10^{11} (per day), the crystal is operated at its zero-temperature coefficient. A triple proportional oven is used to achieve a high degree of temperature control.

For frequency adjustment, two quartz piston-type multiturn trimmers are incorporated in the oscillator circuit. One trimmer acts as a course frequency adjustment, and the second functions as a fine frequency adjustment. The total range is $\pm 2 \times$ pp 10^{7} with a resolution of ± 1 pp 10^{11} is achieved. The trimmers can be varied electromechanically. This type of a bidirectional frequency control utilizing mechanical capacitors provides an effective frequency memory in case of a power interruption.

All the voltages supplied to the oscillator and oven circuitry are regulated and filtered to prevent noise pulses or varying voltages from affecting the overall frequency stability.

In order to ascertain that the redundant unit is running at exactly the same frequency as the on-line unit, the frequencies of each are compared, the difference stored, and then transmitted to the ground station.

Frequency correction can then be commanded to unit 2. This is not shown in the block diagram, but could easily be implemented as a part of the oscillator assembly.

The unit specifications are:

- Operating frequency: 5 MHz
- Temperature control: triple proportional oven
- Drift rate: $1\text{-}2 \times 10^{-12}$ per hour; $1\text{-}2 \times 10^{-11}$ per day, maximum
- Aging rate: $1\text{-}2 \times 10^{-12}$ per hour after seventh day of continuous operation
- Power supply: ±12 volt
- Power supply sensitivity: $\pm 1 \times 10^{-11}$/5% change in supply
- Load sensitivity: $\pm 10^{-11}$/20% load
- Power consumption: 2.5 watt, including remote frequency control
- Remote frequency control, bi-directional: fine control resolution $\pm 10^{-11}$; coarse control resolution $\pm 2 \times 10^{-7}$
- Remote frequency control signal: digital pulses -50 to 100 ms
- Signal output level: 1 volt rms into 50 ohms
- Harmonics: not less than 40 db down
- Spurious outputs: not less than 80 db down
- Temperature range: -20 to +60°C
- Temperature sensitivity: $\pm 5 \times 10^{-12}$ -10°C to +55°C
- Weight: 6 pounds, including remote frequency control
- Dimensions: 3.5 x 4.5 x 9.0 inches maximum

2.2.3.9 Frequency Synthesizer and Switch

The frequency synthesizer uses the highly stable 5.0 MHz signal from the reference oscillator to generate 39.5, 37, and 31 MHz signals for the receiver and transmit mixers and the navigation channel phase modulator. The unit also produces a 10 MHz signal for timing and sequencing within the navigation signal generator. The outputs from each of the two reference oscillators are applied to a diode changeover switch, and when both signals are present, oscillator 1 is selected. Lack of signal from oscillator 1 will switch in oscillator 2. The logic signal from TT&C overrides the built-in logic, and oscillator 2 can be commanded from the ground. Two diodes per pole are used to provide at least 80 db suppression for the unused oscillator.

2.2.3.10 Navigation Channel

The elements of the navigation channel are:

- Navigation signal generator
- Phase modulator-transmitter
- Power amplifier

Position Determination Signal Generator. The position determination (navigation) signal generator is comprised of four major elements: the telemetry buffer and memory, the time-base generator, the BINOR code generator and the data output encoder. Figure 9 shows a block diagram of the assembly indicating the basic functional interfaces between the time-base generator where all timing and control signals are derived and the other units. Uplink messages from the TT&C subsystem are received in the input telemetry buffer. The starting address of data to be stored in memory is collected and transferred to the memory telemetry address register in parallel. Subsequent data are stored in the memory in a serial bit-by-bit fashion. Telemetry data storage will occur on a non-interference basis with normal readout of ephemeris data for transmission to the user.

Time Base and Code Generator Unit. The design for the time base and code generator unit is straightforward; a block diagram and some timing diagrams are shown in Figure 10. The time base unit counts down from the oscillator reference frequency in binary fashion to 12.39 seconds per cycle. A second set of counters generates time-of-day information from the 5-kHz flip-flop in the first set of counters. A group of 13 frequencies in the first set (320 kHz to 78.125 Hz) is used to generate the BINOR code for range measurement.

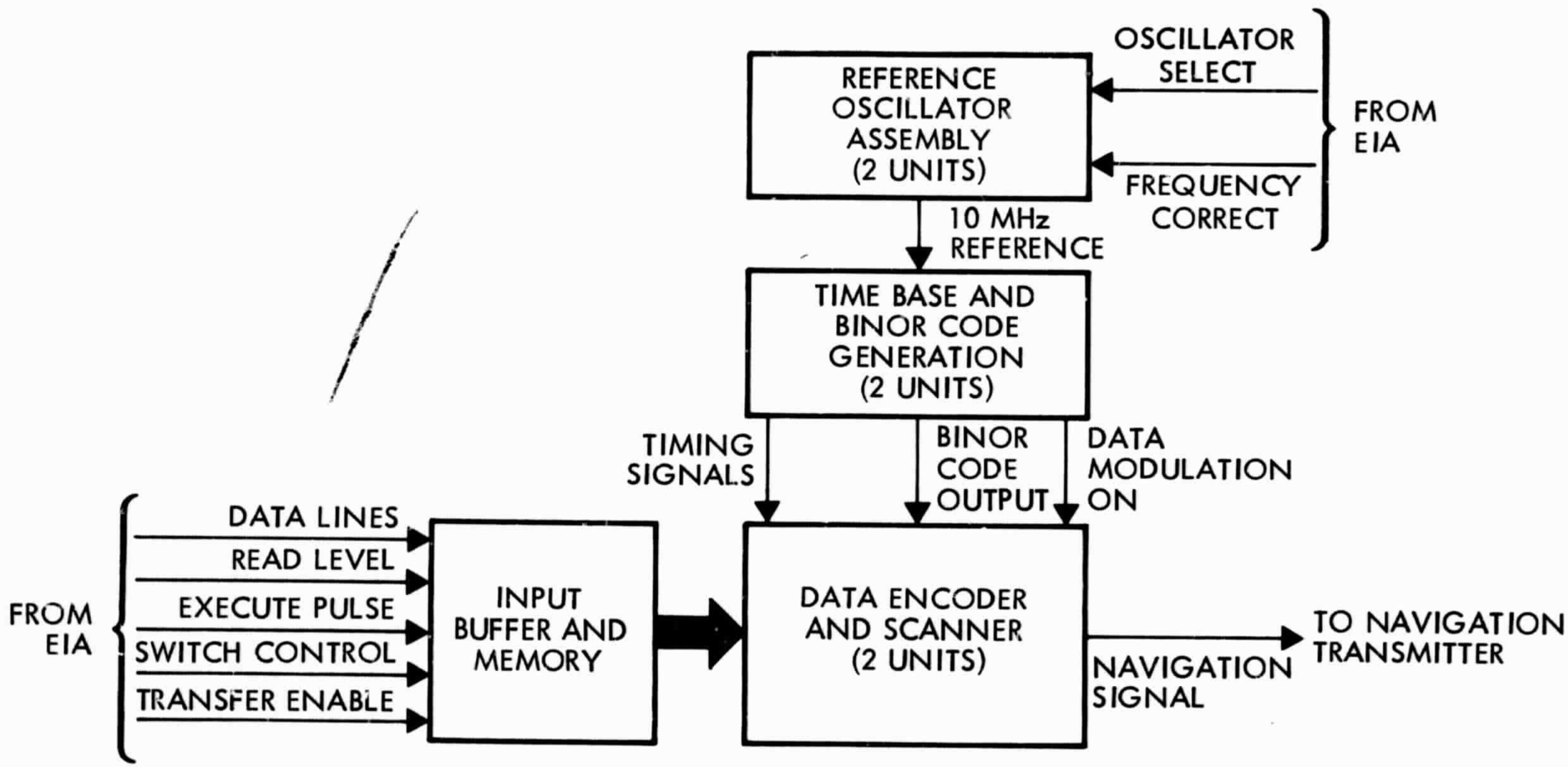


Figure 9. Navigation Signal Generator Overall Block Diagram

The majority logic block of Figure 10 is used to generate the BINOR ranging code. After examining several ways of performing this function, this was chosen because of its simplicity and reliability. This method is based on multilevel logic, using modulo 2 adders. For 13 inputs, as little as 36 gates can define the proper output, most gates having only 3 inputs. A brute-force approach to the same decision making would take several hundred gates.

<u>Data Encoder and Scanner</u>. Figure 11 gives a logic diagram for encoding data for transmission to satellite users. These data are received by the satellite from command stations at regular intervals (ephemeris and time correction information) and stored in the encoder registers. In addition, a more frequent time reference update occurs near the start of data transmission during internal timing signal DS1 (see Figure 12). At this time, the time-of-day information is transferred in parallel into the time reference register where it is held until the next update 12.39 seconds later. The only other information to be generated is fixed for each satellite (ID and frame sync codes).

The information coming out of the scanner has a format of 15 words of 11 bits each. Figure 13 gives the exact arrangement of the data. Each bit is 1/625-seconds long. The first 35 bits of the data transmission allow time for the user preprocessor to achieve bit synchronization. Then, 165 bits of information are sent (the last 6, not counting parity, allow for future expansion), giving a total of 200 bits.

Timing signals DSP and DS1 through DS165 are generated from the last four flip-flops of the ÷2048 counter and the first five flip-flops of the ÷121 counter (625 Hz to 4.88 Hz). The "Data Modulation On" signal limits the scanning to the time that data are transmitted, and inhibits interference of data with the ranging code. A ÷11 counter and time derivatives from it are used to control entry of ephemeris and time-correction data into the correct registers in a serial mode. A set of signals for control of information transfer from the electrical integration assembly (EIA) to the data encoder registers is shown in Figures 9 and 11. Data bits are transferred 8 at a time via an 8-bit input register with parallel input and serial output. A 5-kHz signal from the time base and code generator unit has been selected to control the shift rate; this rate is somewhat

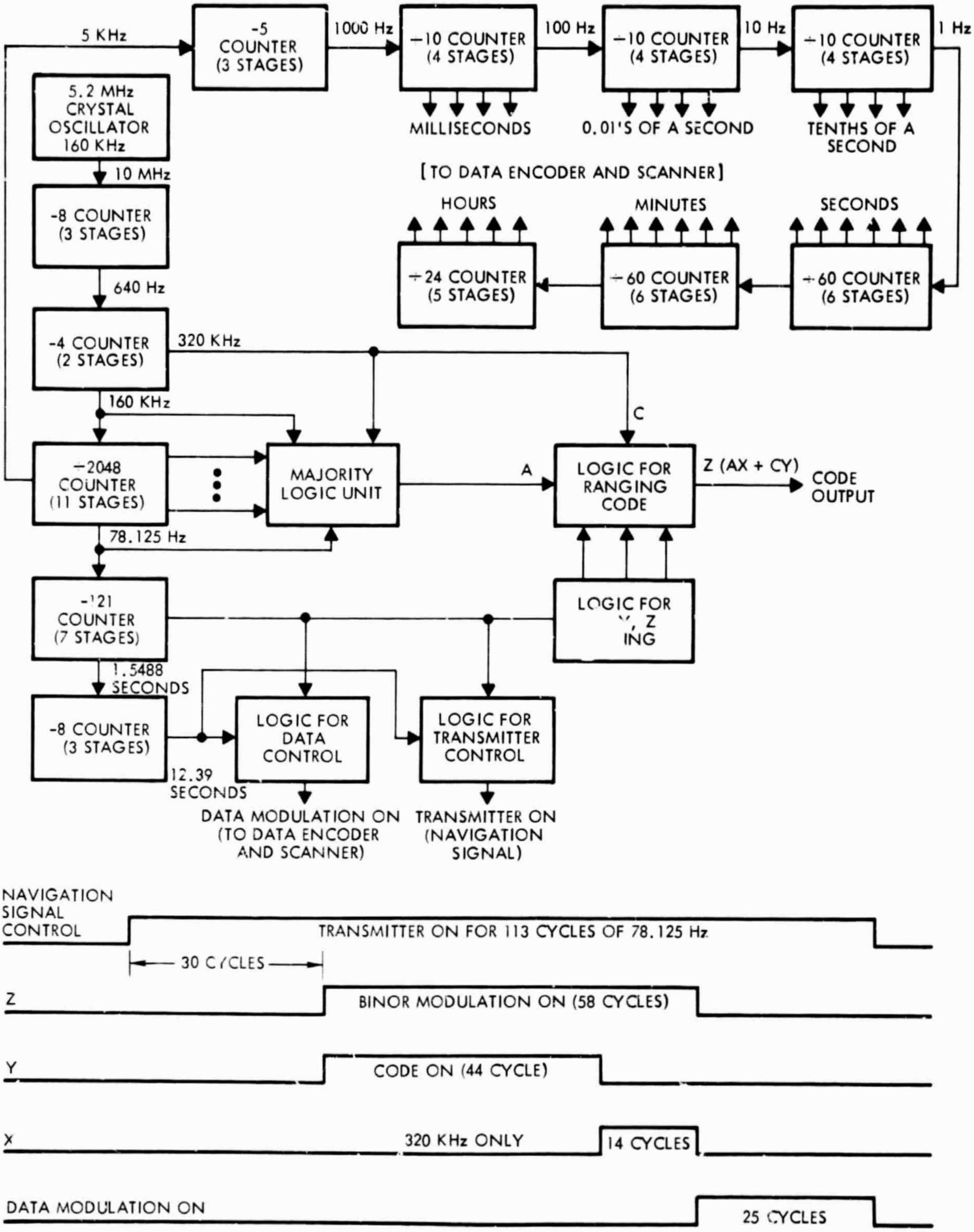


Figure 10. Time Base Unit and Code Generator

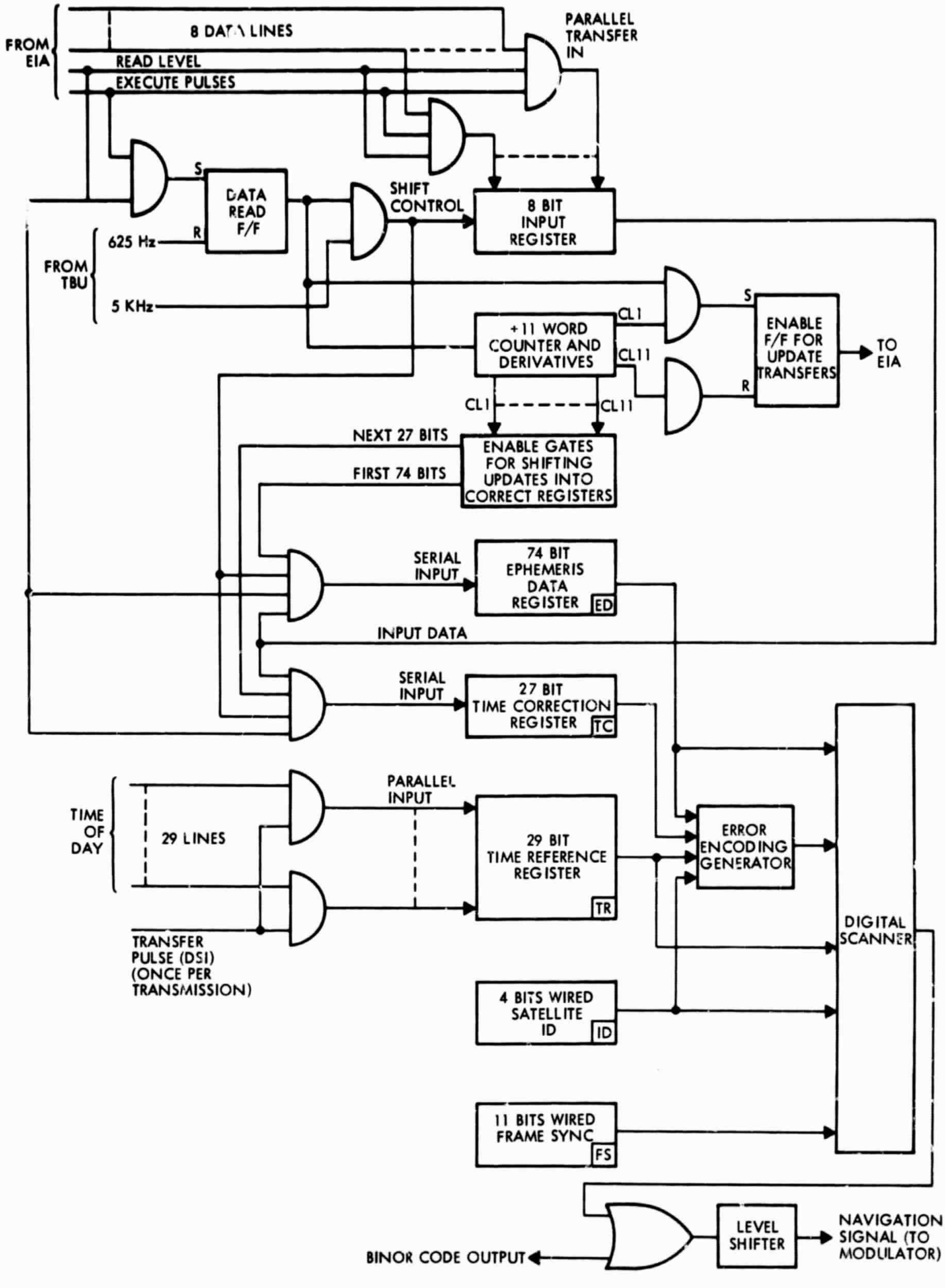


Figure 11. Data Encoder and Scanner

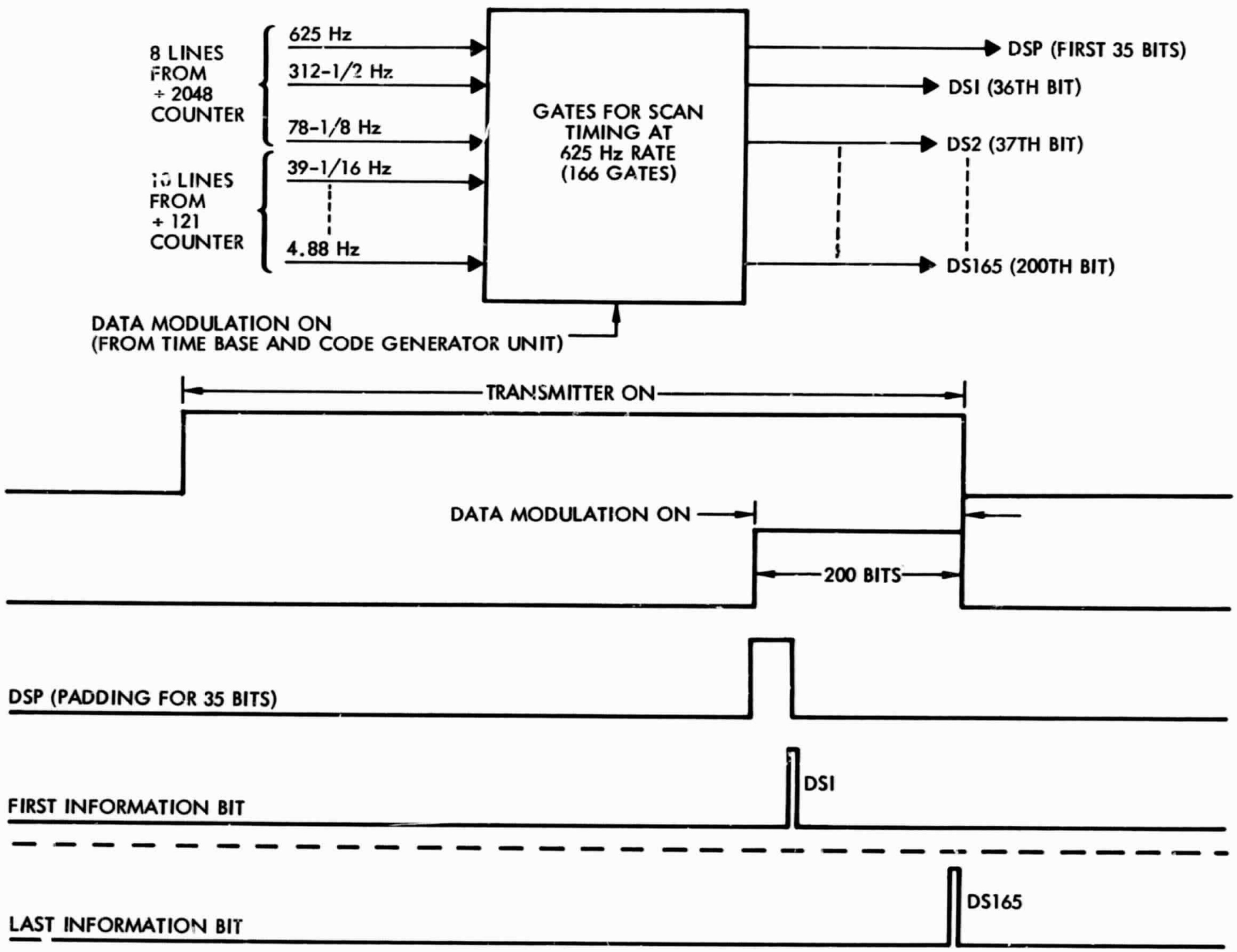


Figure 12. Timing Signals for Error Encoding and Digital Scanning

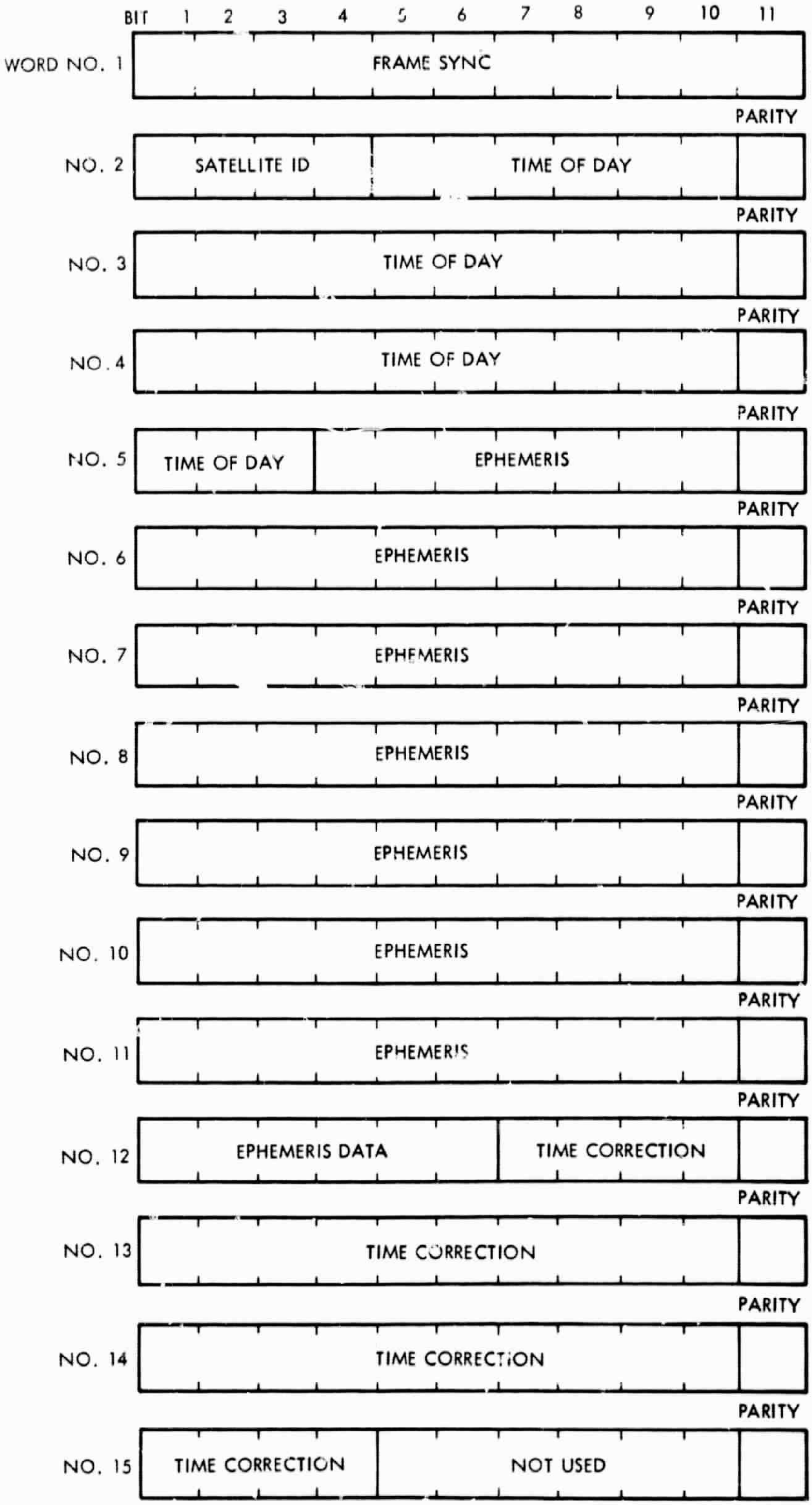


Figure 13. Data Format

arbitrary and can be changed if required. A transfer enable signal is sent to the EIA to indicate when message transmission is complete; this signal is true at the start of transmission (initiated by the EIA) and for 11 words thereafter. Figure 14 gives details of the digital scanner and error encoding generator, which are shown together since most of the input signals are the same. The DSP and DS1 through DS165 signals are gated with the proper information and then fed into a modulo 2 adder along with the output of the 625-Hz flip-flop of the time base and code generator unit. Since the data rate is also at 625 Hz, a split-phase data representation results, which is the desired form for the data. The ranging code is OR gated with the split-phase data and level shifted to meet modulator requirements. The final signal has equal plus and minus levels, which will be adjusted in amplitude to final transmitter modulation index.

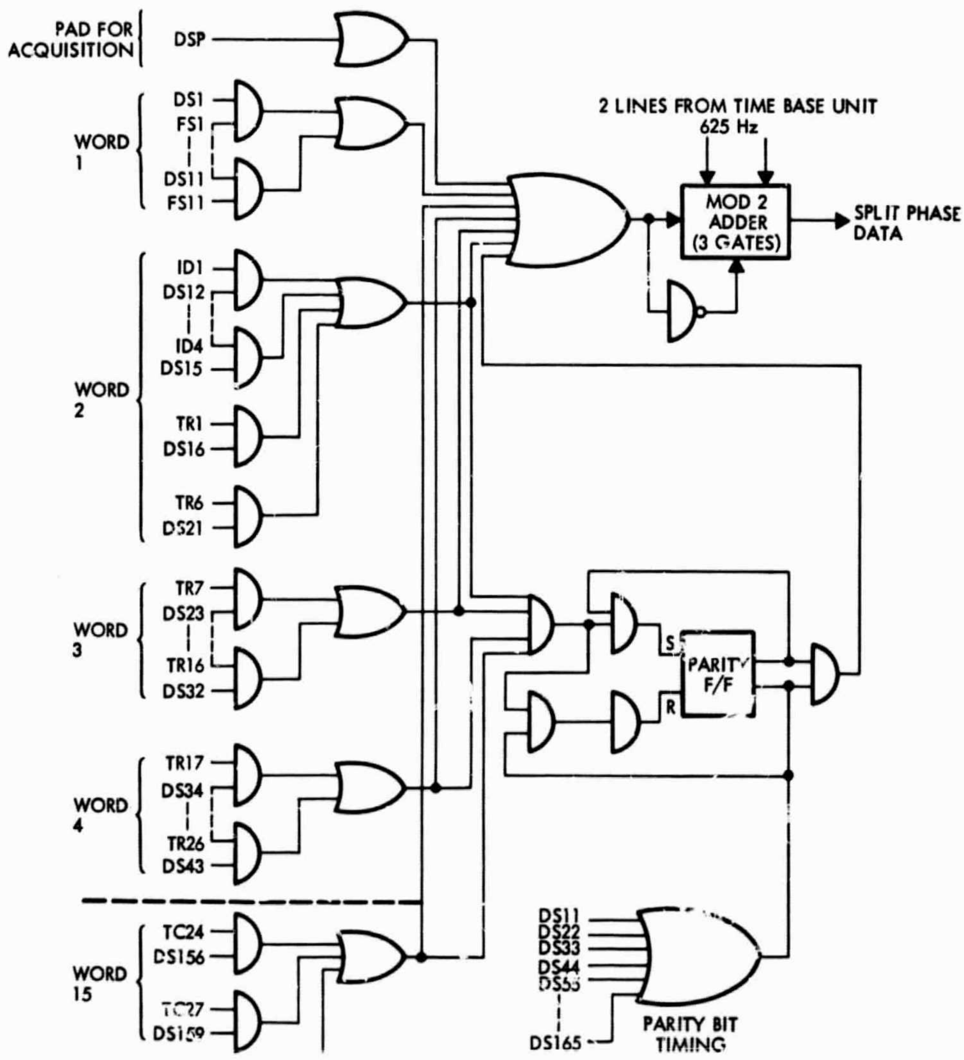


Figure 14. Details of Error Encoding Generator and Digital Scanner

Error Encoding Generator. Even parity is generated with a single flip-flop. The data are fed into this flip-flop as well as to a modulo 2 adder, as described in the preceding subsection. The data are scanned in proper sequence with timing signals shown in Figure 12. The parity flip-flop will, therefore, be in the correct state at the time it is scanned for parity insertion into the data at the input of the modulo 2 adder. Meanwhile, the parity flip-flop is reset so that it can start properly in generating parity for the next word. The reset logic (DS11 through DS165) can be set up (optically) directly from the counter outputs as the eleventh pulse of each 625-Hz count.

Phase Modulator — Transmitter. Digital navigation signal from the data encoder is applied to the biphase modulator which derives its 31 MHz carrier from the frequency reference oscillator via the synthesizer. The phase modulation is impressed upon the carrier at low modulation index, followed by multiply by 5 x 5 x 2, thus multiplying the modulation index to the required value. A bandpass filter and driver amplifier follow the multiplying chain providing a sufficient power drive for the solid-state amplifier at 1550 MHz.

Solid-State Power Amplifier. The design of the 50 watt solid-state power amplifier is based on the facts that the maximum dc to RF efficiency, reliability, and minimum size and weight are required for a space qualified unit. There are two basic design approaches possible. The first design is based upon existing discrete components and techniques.

The power amplification is accomplished at 400 to 600 MHz and then multiplied in a varactor diode to L-band frequency. The second approach was solid-state L-band devices which are presently being developed by the TRW Semiconductor division. This design has a superior dc-to-RF efficiency because it avoids the loss in the varactor multiplier. This design concept is shown in Figure 15. An overall efficiency of 30 percent is expected.

With microstripline within the module, considerable reduction in lead length is achieved, increasing both the operating frequency and power. Fifteen-watt devices at 2 GHz have been demonstrated using these techniques. The individual modules are designed with a 50-ohm input and output impedance so that their performance can be checked prior to integration into the system. The hybrid combiner and divider networks can

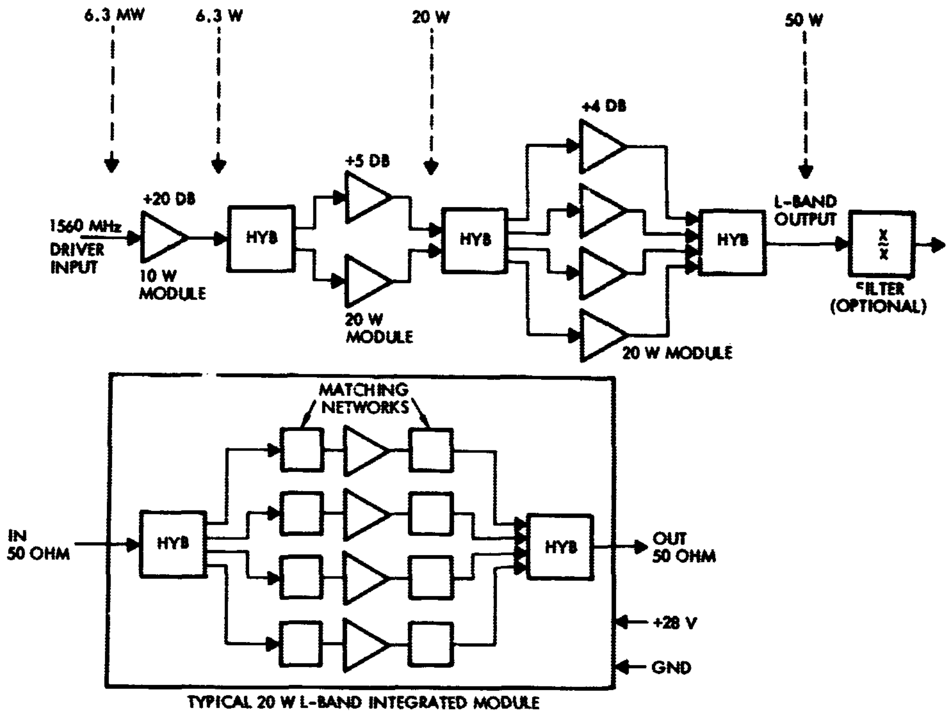


Figure 15. Design Concept for Direct L-Band Amplification Transmitter

also be constructed utilizing the stripline approach. Use of the stripline has the important advantage that later mistuning of the networks is unlikely since distances will remain fixed.

Since the availability of dc power within the spacecraft is always limited, it is important that onboard equipment is designed to provide maximum dc-to F efficiency. Since the information from the navigation signal generator has a duty cycle of approximately 10 percent, the use of a solid-state class C amplifier provides an efficient use of prime power. During the time interval when no information (signal) arrives at the power amplifier, no current is drawn. Had the TWT been used, a steady power consumption corresponding to the peak power would be consumed.

2.2.4 Tracking, Telemetry and Command Subsystem

The Tracking, Telemetry and Command (TT&C) subsystem as shown in Figure 3 is comprised of telemetry transmitters, telemetry encoders, diplexer, two S-band conical antennas combined in a 4.5 db hybrid, command receivers, command decoders, and a 3 db receiver hybrid. All the active units are provided with a one-to-one active redundancy, with the exception of the transmitter unit which is backed by a cold standby. The designs for the TT&C subsystem are based upon components which have already been developed and flight-proven on other spacecraft programs. The telemetry data includes housekeeping information, satellite command status, and high resolution timing information. Commands are transmitted at 50 bps data rate from the ground station to initiate normal satellite functions, propulsion engine firings, and resetting of the satellite oscillator frequency. Satellite ephemeris and oscillator phase correction data is transmitted from the ground station to the satellite for onboard storage.

2.2.4.1 Subsystem Description

Telemetry Encoder. The proposed telemetry encoder is similar to a design developed for the INTELSAT III satellite. Analog telemetry signals are time-multiplexed by a ripple counter and sampling gates at a 1 minute frame rate; 63 channels comprise a complete frame. Critical data may be super-commutated to a rate of once every 15 seconds. Earth sensor pulses can be telemetered to the ground station (for attitude control command backup) by two IRIG subcarriers. Sun sensor signals may be amplitude modulated on the subcarriers. The 63-channel PAM data frequency modulates a third IRIG subcarrier. The outputs of the modulators are summed and routed to the transmitter.

Telemetry Transmitter. The telemetry transmitter accepts the encoder subcarriers and phase modulates them onto the S-band carrier. A minimum of 2 watts output power at 2200 MHz is produced. The all-solid-state S-band transmitter developed for the SGLS program is proposed for use on the NTC satellite. Frequency stability is achieved by a temperature-compensated crystal oscillator which is isolated from succeeding stages by a buffer amplifier. A block diagram of the telemetry transmitter is given in Figure 16.

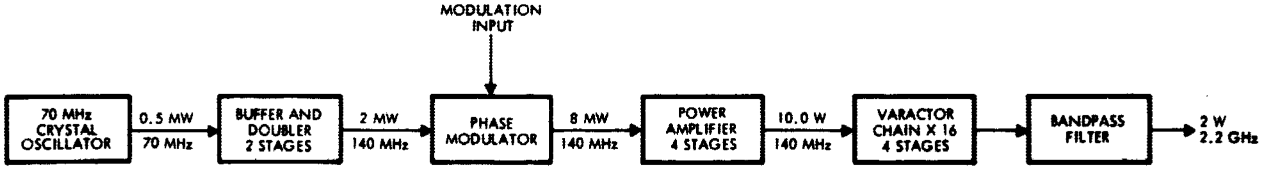


Figure 16. S-Band Telemetry Transmitter Block Diagram

The subcarriers are phase modulated onto a 140 MHz RF carrier by a varactor phase modulator. The carrier and phase deviation are multiplied by sixteen to achieve the final output frequency. Amplification to 10 watts is achieved at 140 MHz (where efficiency is high) using four transistor amplifier stages. The 140 MHz phase modulated carrier is multiplied by sixteen using four push-push varactor doublers; final output power is 2 watts at 2200 MHz. Spurious harmonics are removed by a high-Q cavity bandpass filter.

Diplexer. The diplexer allows the receiver and transmitter to be coupled to a common antenna with maximum isolation between both units and minimum insertion loss between the units and the antenna. The diplexer developed and space-qualified for the SGLS program has a receiver channel insertion loss of 1.6 db, a transmitter channel insertion loss of 0.7 db, and a transmitter channel insertion loss of 0.7 db, and a transmitter-to-receiver isolation of 80 db. Receiver image rejection is a minimum of 85 db and attenuation of the transmitter second harmonic is 80 db.

Command Receiver. A flight-qualified S-band receiver available from the SGLS program is proposed for the NTC satellite. The phase-lock receiver is all solid state and applies dual conversion. Important features of this receiver include coherent mixer injection frequencies, low frequency IF amplifiers, and local oscillator and product frequencies chosen such that high level harmonics of the incoming frequency are not generated. This feature is vitally important since a self-locking mode is possible if a subharmonic of a received frequency is present and of threshold magnitude.

Command Decoder. The command decoder is similar to that used on the Vela spacecraft. It demodulates the FSK subcarriers from the command receiver and processes the 50 bps data stream into a 9 bit parallel readout. Each command or data word appears as binary information on these 9 lines for 80 microseconds at the end of a command sequence.

A 60 microsecond execute signal appears on the tenth line and occurs in the center of the command output interval. A sequential circuit disables the command decoder and turns off the power to most of its circuits whenever no tones are present. The data lines and the execute line are set to the electrical integration assembly (EIA) which provides command processing and distribution.

Link Performance. The telemetry and command link power budgets are given in Tables 4 and 5. A 30 feet diameter ground station antenna was assumed. The telemetry link performance margin is 4.6 db; the command link performance margin is 24.3 db.

Table 4. Power Budget-Telemetry Link, 2200 MHz (PAM/FM/PM)

Parameter	Value
Satellite transmitter power (2 w)	+33.0 dbm
Circuit loss	1.50 db
Average Omni (antenna system gain)	0.0 db
Space loss (22,000 n mi) (2200 MHz)	189.9 db
Polarization loss (maximum)	3.0 db
Ground antenna gain (30-ft parabola)	43.0 db
Ground circuit loss	0.5 db
Net transmission loss	151.9 db
Total received power	-118.9 dbm
Receiver noise spectral density ($T_s = 290^{o}K$)	-174.0 dbm/Hz
Received carrier-to-noise density ratio (C/Φ)	55.1 db
Required C/Φ	50.5 db
Performance margin	4.6 db

Table 5. Power Budget — Command Link, 1800 MHz (PCM/FSK/PM-50 bps)

Parameter	Value
Ground transmitter power (100 w)	50.0 dbm
Circuit loss	2.5 db
Ground antenna gain (30-ft parabola)	43.0 db
Space loss (22000 n mi) (1800 MHz)	189.6 db
Polarization loss (maximum	3.0 db
Satellite antenna gain (omni)	0.0 db
Satellite circuit loss	1.5 db
Net transmission loss	153.6 db
Total received power	-103.6 dbm
Receiver noise spectral density (N. F. = 6.0 db)	-168.0 dbm/Hz
Carrier Performance	
Carrier modulation loss (1.0 rad peak deviation)	2.3 db
Received carrier power	-105.9 dbm
Carrier loop noise BW ($2 B_{LO}$ = 1-kHz)	30.0 db
Threshold SNR in $2 B_{LO}$	6.0 db
Threshold carrier power	-128.0 dbm
Performance margin	+22.1 db
Data Performance	
Data modulation loss (1.0 rad peak deviation)	4.1 db
Received data power	-107.7 dbm
Data noise bandwidth (50 Hz)	17.0 db
Threshold SNR in data bandwidth ($P_e = 10^{-5}$)	19.0 db
Threshold data power	-132.0 dbm
Performance margin	+24.3 db

2.2.5 Antenna Subsystem

Based on a preliminary study of the antenna performance requirements in relation to the navigation/air traffic control mission requirements, the following antenna assembly is proposed. The selection of this antenna configuration was preceded by a tradeoff of various antenna configurations and appropriate techniques to provide variable beam shaping or a number of individually sha d beams. Included in the tradeoff were phased array techniques and combinations of reflector antenna/phased array approaches.

The antenna assembly consists of three separate functional units; the high gain directional antenna, the earth coverage antenna, and the omnicoverage telemetry/command antenna. The antenna designs are described in the following paragraphs.

2.2.5.1 High Gain Directional Antenna

The high gain directional antenna, as indicated in Table 6, is a 5-1/2 foot diameter parabolic reflector fed by a sleeve-dipole turnstile. The reflector consists of a rigid center section 4-feet in diameter, surrounded by a 3/4-foot wide deployable circular extension attached to the perimeter of the 4-foot center section. The deployable section is of a swirl-about design folded around the perimeter of the center section when in stowed position. This approach has been selected because it will provide an operational parabolic reflector with only approximately 3 db of gain reduction in case of failure to deploy. Also, this approach can be implemented at less weight than other approaches studied.

Table 6. High Gain Directional Antenna Parameters

Type	Partially Deployable Parabolic Reflector
Feed configuration	Sleeve-dipole turnstile
Reflector size	5.5 foot diameter
Focal length	0.5 D
Frequency	1540 to 1660 MHz
Net gain (boresight)	26.0 db (1600 MHz)
Beamwidth (3 db)	8 degrees
Antenna losses (total)	3.5 db
Antenna weight (total)	27.2 pounds

The reflector is sized for a net gain of 26 db above isotropic at boresight, assuming 50 percent minimum overall antenna efficiency. The net antenna gain is maximized by selecting an edge taper which minimizes the sum of the spillover and aperture illumination taper losses. A sleeve-dipole turnstile feed is employed to obtain acceptable performance for all parameters of importance such as VSWR, axial ratio off and on axis, and aperture efficiency. A suitable feed configuration is shown in Figure 17.

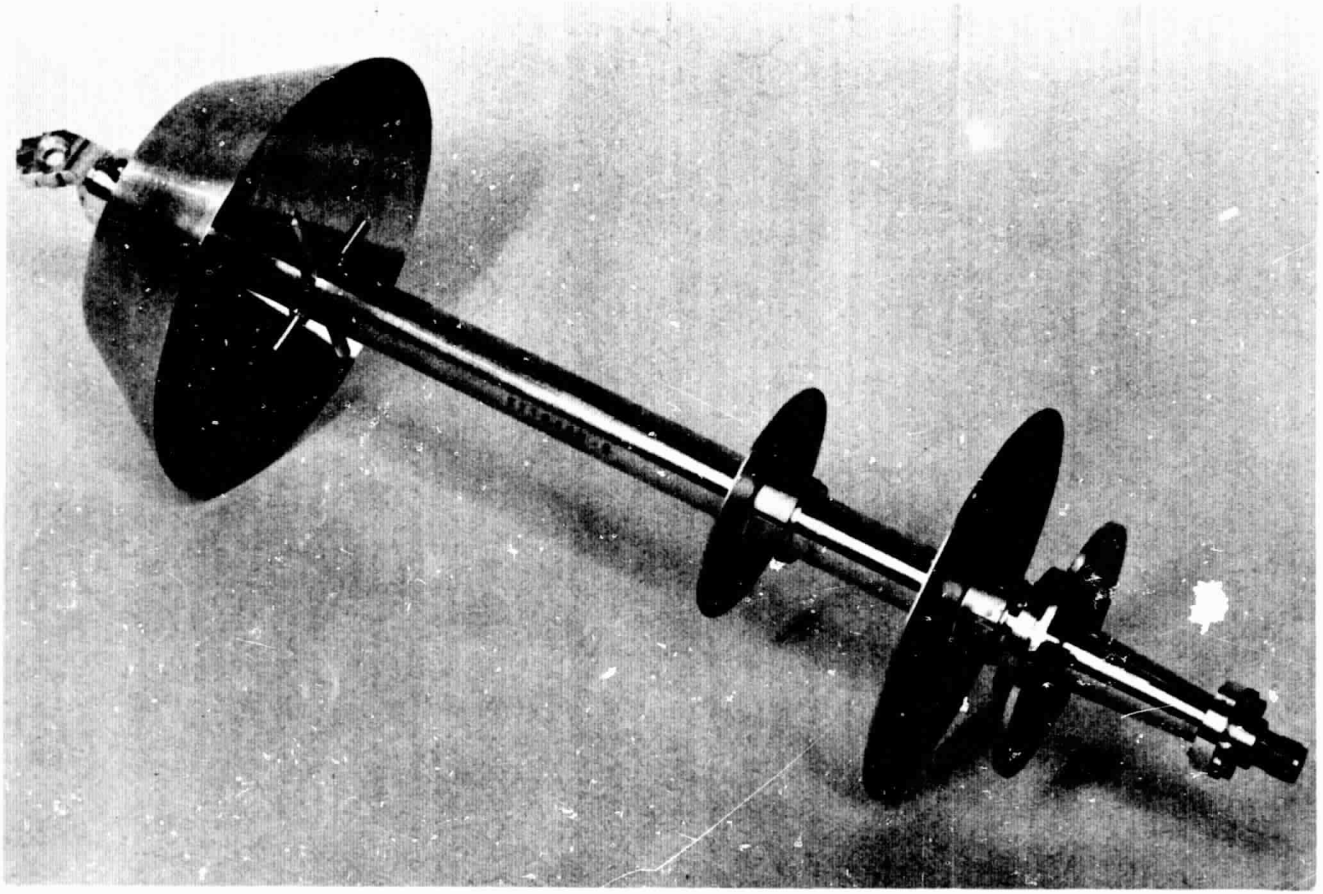


Figure 17. Crossed Dipole Reflector Feed

Reflector losses are listed in Table 7 and total antenna losses are summarized in Table 8. These losses represent a careful evaluation of the reflector efficiency losses, including the additional loss shown under reflector surface errors caused by the irregularities and discontinuities in the deployable reflector section.

Table 7. High Gain Directional Antenna Efficiency Losses

Illumination	1.10
Aperture blockage	0.20
Reflector surface errors	0.65
Cross polarization	0.15
Primary feed phase errors and defocusing	0.25
Aperture mismatch	0.25
Illumination spillover	0.40
	Total 3.00 db

Table 8. High Gain Directional Antenna Losses

Reflector/feed	3.00
Diplexer	0.25
Coaxial cable	0.25
	Total 3.50 db

2.2.5.2 Earth Coverage Antenna

A broad beam for optimum earth coverage is implemented using a reflector/feed configuration of the type shown in Figure 18. The reflector diameter has been selected to provide maximum earth edge gain at ten degrees above horizon. It is derived by calculating the earth edge gain as a function of reflector diameter for a given aperture efficiency and beamwidth factor. Results are shown in Figure 19 using a 53 percent reflector efficiency and a 3-db beamwidth value of $72\lambda/D$, where λ is the free space wavelength and D is the reflector diameter. The edge gain is adjusted for the difference in altitudes between earth edge and boresight to the spacecraft. For reflector diameters other than 36 inches, the earth edge gain decreases because either the aperture gain decreases or the beamwidth narrows and the first pattern null moves into the angular range of interest. The curve shifts along the abscissa with increasing frequency in the direction of decreasing diameter.

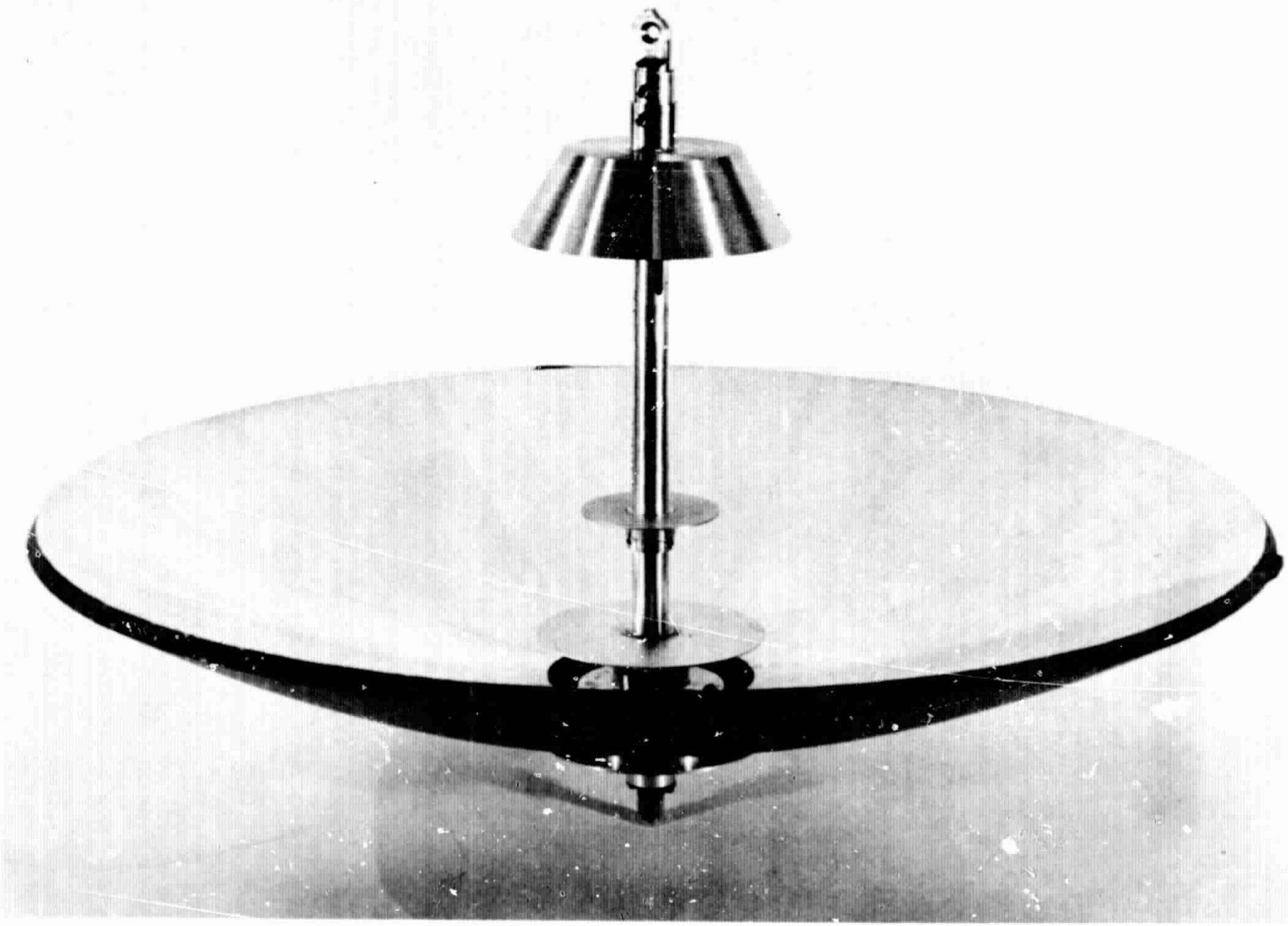


Figure 18. Earth Coverage Reflector/Feed Assembly

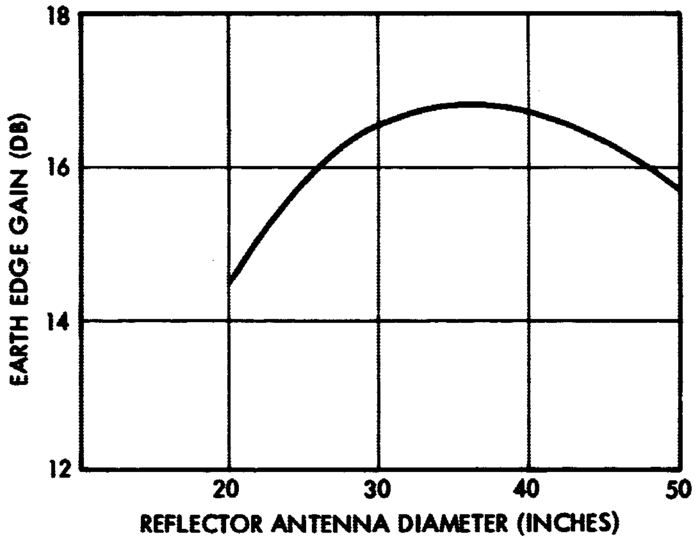


Figure 19. Earth Edge Gain — Antenna Gain at 10 Degrees Above Horizon as a Function of Reflector Diameter Circular Synchronous Orbit

A reflector/feed configuration, similar to the proposed design has been qualified for spacecraft environment. To minimize weight, the 3-foot parabolic reflector is constructed of expanded aluminum honeycomb cone sandwiched between two layers of fiberglass cloth face sheets. The front surface is metalized to provide a continuous conductive and reflective surface.

Circular polarization is obtained by the orthogonal orientation of the crossed dipoles and by feeding the balanced dipoles 90 degrees out of phase with equal power from a common split tubular balun. Unequal length sets the dipole pairs at the proper phase difference.

The cup reflector is located a quarter wavelength behind the crossed dipoles. It reflects the incident energy back into the parabolic reflector in phase with the dipole radiation in that direction. The shape and size of the cup control the primary illumination taper across the reflector to obtain equal E and H plane beamwidths. Two discs are placed on the center tubular support in front of the parabola to compensate for the feed blockage effects and provide low axial ratio characteristics for both off and on axis. The performance/design parameters of the earth coverage antenna are listed in Table 9. Efficiency and antenna losses are provided in Tables 10 and 11, respectively.

Table 9. Earth Coverage Antenna Performance/Design Parameters

Type	Parabolic reflector
Feed configuration	Turnstile-crossed dipoles
Reflector size	3.0 feet diameter
Focal length	0.4 D
Frequency	1550 MHz
Net gain (boresight)	20.5 db
Net earth edge gain (ten degrees above horizon)	16.8 db
Beamwidth (3 db)	14.5 degrees
Antenna losses (total)	2.85 db
Antenna weight (total)	3.7 pounds

Table 10. Earth Coverage Antenna Efficiency Losses

Illumination	1.0
Aperture blockage	0.30
Reflector surface errors	0.25
Cross polarization	0.15
Primary feed phase errors and defocusing	0.25
Aperture mismatch	0.25
Illumination spillover	0.40
Total	2.60 db

Table 11. Earth Coverage Antenna Total Losses

Reflector/feed	2.60
Coaxial cable	0.25
Total	2.85 db

<u>Omnicoverage Antenna Assembly</u>. The omnidirectional radiation coverage is provided by a set of two spiral antennas mounted one on top and one at the bottom of the spacecraft diametrically opposite to each other. The spiral antenna on top is a conical log spiral, the other antenna element is a flat, Archimedean spiral. Both elements inherently provide circular polarization. This omnidirectional antenna configuration has been selected because it provides coverage over 85 percent of the total radiation sphere.

The omnidirectional antenna assembly is shown as a block diagram in Figure 20. The power is fed to each antenna through a 4.5 db power divider to equalize the ERP radiated by each antenna. (The unequal power divider is required because the gain of the two antennas is different.)

The conical log spiral antenna has two complementary arms which cause the conical spiral to have a balanced impedance of approximately 110 ohms. A matching transformer balun is used to match the antenna elements to a 50 ohm unbalanced coaxial feed line. The infinite balance technique of wrapping the antenna elements with the feed cable is employed, in addition to the matching transformer incorporated in the feed cable at the tip of the antenna.

The cavity-backed Archimedean spiral antenna has two complementary arms to provide a balanced impedance of approximately 160 ohms. Feeding the Archimedean spiral using a Robert's balun has been found to be advantageous and convenient.

The radiation pattern of the two antennas is shown in Figure 21. The broader pattern is due to the conical log spiral, while the narrower pattern is of the cavity backed Archimedean spiral antenna. As seen, the interference region is minimum.

The physical configuration of the conical log and Archimedean spiral are shown in Figures 22 and 23. Table 12 lists design parameters of the omnicoverage antenna configuration.

Table 12. Design Parameters Omnicoverage Antenna

Frequency	1800 to 2300 MHz	
Conical log spiral	1.0 pounds	4.7 inches diameter, 11.6 inches high
Archimedean spiral	1.0 pounds	5.0 inches diameter, 3.0 inches high

2.2.5.3 Weight Estimate

Tables 13 and 14 list the detailed weight breakdown for the two directional antenna assemblies.

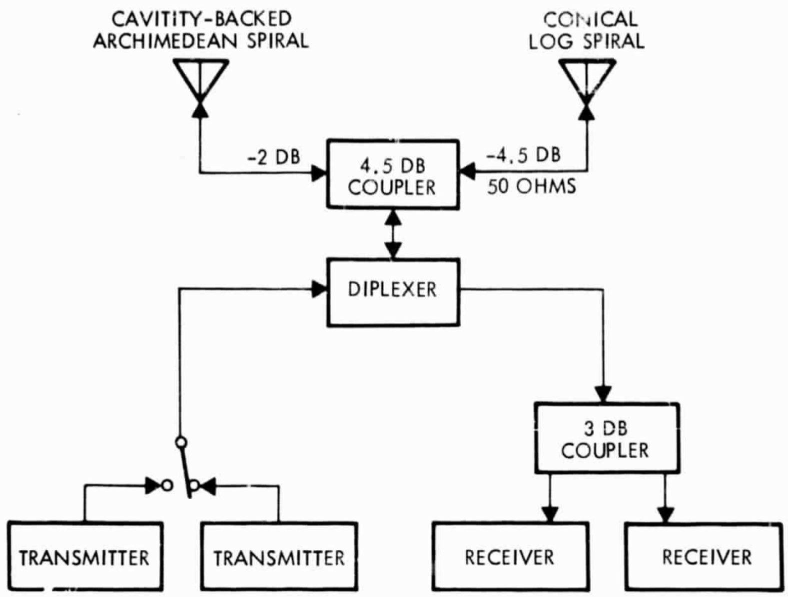


Figure 20. Omni-Coverage Antenna Block Diagram

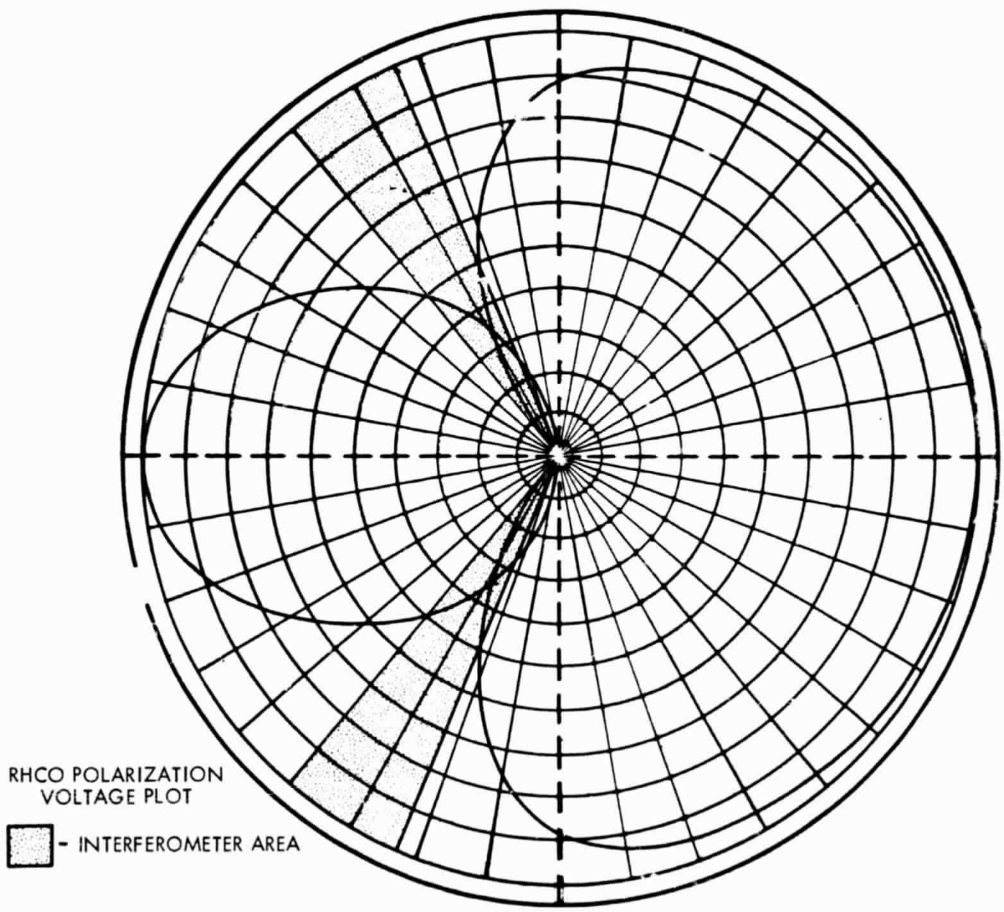


Figure 21. Omni-Antenna Array Ideal Pattern

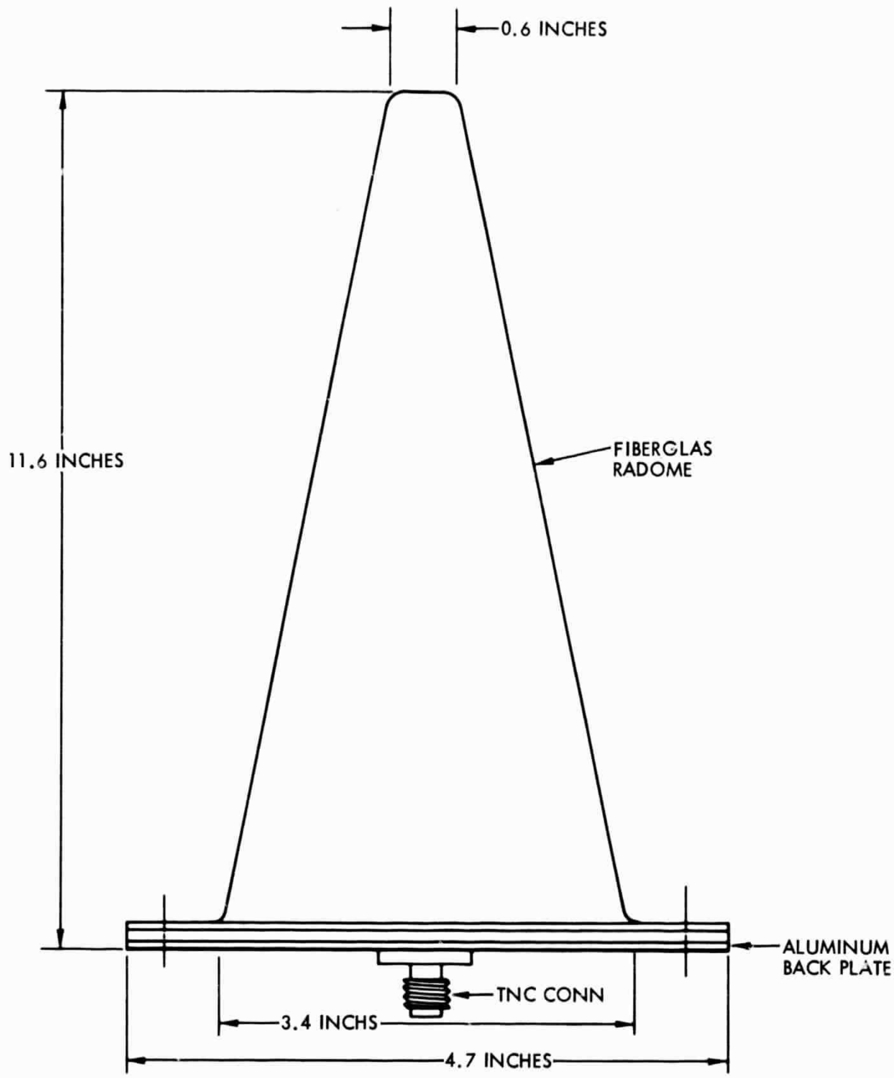


Figure 22. Conical Log Spiral

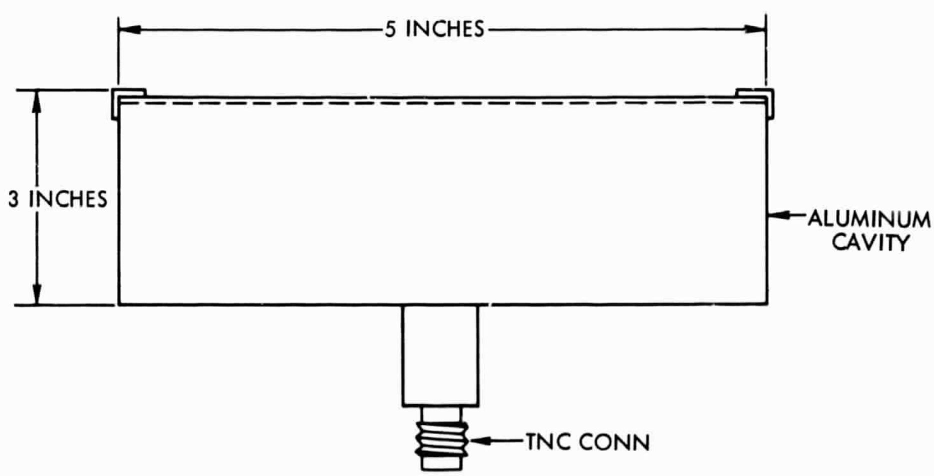


Figure 23. Archimedean Spiral

Table 13. Weight Breakdown of the High Gain/ Narrow Beam Antenna Assembly

Face sheets/honeycomb cone	2.26
Ribs-deployable section	3.07
Mylar covering	1.00*
Frame/ribs	3.25*
Stiffening ring	2.08
Rivets	0.50
Antenna stem	0.52
Antenna stem end fittings	2.00
Strap cutter (Pyrotechnic)	1.50
Collapsible rim strap	0.25
Screw rod/motor-actuator	3.00*
Actuator FTGS	1.00
Antenna adjustment mechanism	1.00
Antenna support FTG	1.00
Antenna feed	1.50
Antenna feed support	0.80
Contingency (ten percent)	2.47
	Total 27.20 pounds

Table 14. Weight Breakdown of the Earth Coverage Antenna Assembly

Reflector-honeycomb	1.27
Reflector support (fixed)	1.00
Reflector feed	0.75
Feed support	0.35
Contingency (ten percent)	0.33
Total	3.70

*Could be reduced slightly by employing a strain energy deployable rim.

2.2.5.4 Alternate Antenna Configurations

The antenna configuration proposed is based on the results of a tradeoff study between phased array and reflector antennas and their applicability to providing two simultaneous beams of different beamwidth (and aperture gain). Such a tradeoff could include many techniques associated with phased arrays and reflector antennas. Some of the more practical approaches are listed in Table 15.

Table 15. Multiple/Controlled Beams

Multiple reflectors	Maximum volume, simple, arbitrary control
Reflector/modified feed	Blockage, spillover, limited in scope
Reflector/modified area	Added complexity-reflector element interaction
Array/phase	Simultaneous beam difficult
Array/amplitude	Controlled IC amplifiers

The first approach uses as many separate reflectors as beams desired. This approach is simple, reliable, and effective. If only a few beams are required, the approach is very practical.

Using one reflector and a dual feed reduces the required volume. However, the number and type of beams are limited. Even using an electronically controlled feed provides only limited beam control, the change in beamwidth achievable amounts to 50 percent.

An alternate approach is aimed at changing the effective radiating aperture of a reflector. In this approach, the reflector is surrounded by controlled antenna elements which, when energized, provide a radiating aperture larger than the reflector. This approach is limited in beam control and number of beams obtainable.

A phased array of many individually controlled radiating elements provides greater control over the array radiation pattern. This control may be implemented by programming the phase and/or the amplitude at the elements of the array.

A wide range of ' eam control is achievable by changing the phase of the elements to provide the desired beamwidth (phase spoiling). Theoretical patterns have been computed for a 1:20 ratio of beam broadening. The net gain decreases in proportion to the beam broadening.

When amplitude control of the array elements is used to change the beam, several schemes are possible. Using several diplexers and separate feed networks, an array can be configured to radiate or receive several different shape beams at slightly different frequencies. In theory, control of the number and type of beams is unlimited.

Phased arrays provide greater flexibility in pattern control than reflector configurations. Implementing integrated circuits in the phased array provides maximum control, since phase and amplitude control can be achieved at each radiating element using the electronic phase shifters and distributed power amplifiers. Figure 24 shows an array of deployable helix radiators which are applicable to spacecraft phased arrays. Figure 25 shows a block diagram for an array providing two separate beams—one broad, the other narrow. Separate feed networks, distributed power amplifiers, and signal diplexing make this approach realizable. Reflector configurations are competitive with phased arrays when only limited beam change or control is required. Ultimately, however, phased arrays are required to achieve maximum flexibility and control. For limited applications, an assembly of two or three different size reflectors will meet the requirement for beams of different beamwidth and net aperture (or earth edge) gain. Therefore, in view of the requirements of this system, two different size reflectors can very efficiently provide two separate beams.

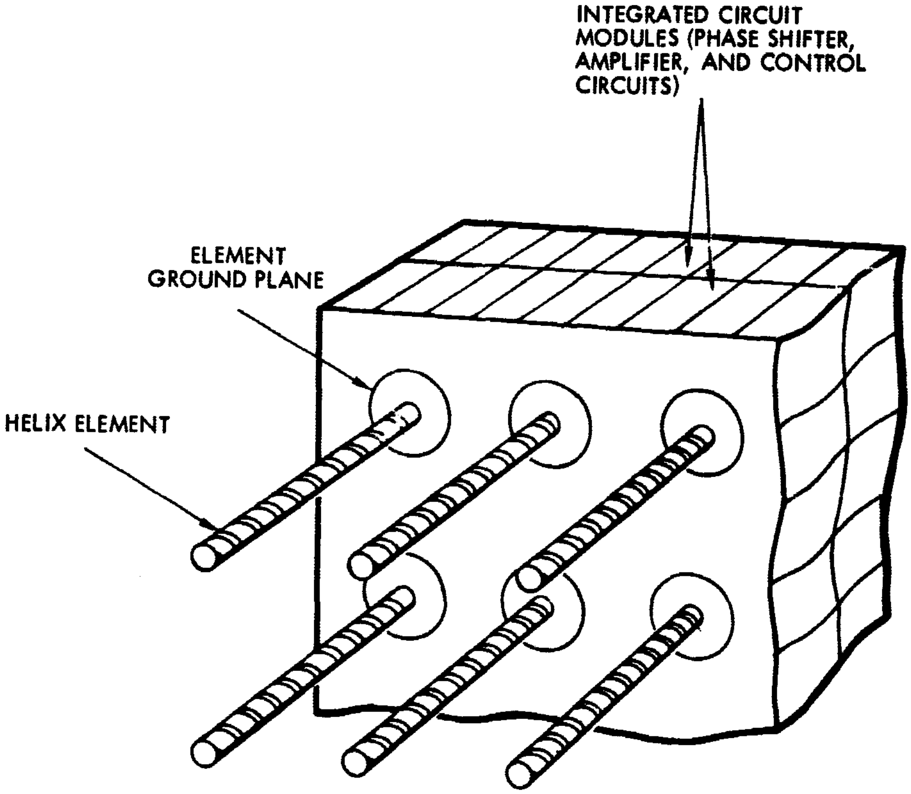


Figure 24. Phased Array of High Gain Deployable Halices Applicable to Beam Zooming by Phase Spoiling or by Amplitude Taper Variation

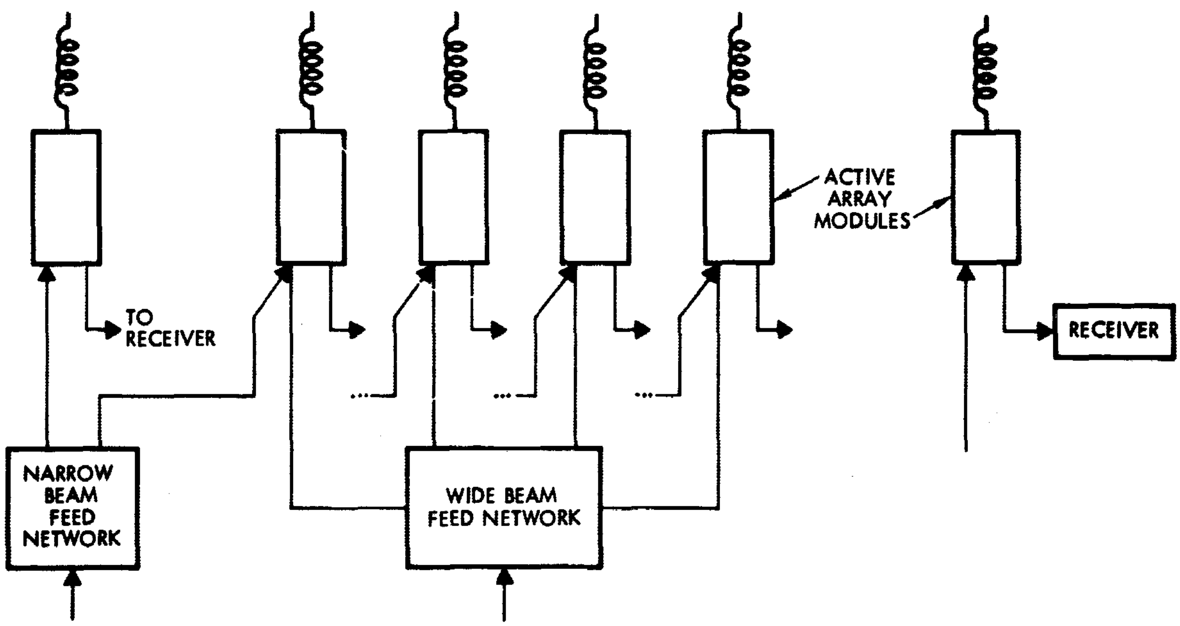


Figure 25. Two-in-One Phased Array Using Active Array Modules and High Gain Array Elements

2.2.6 Attitude Control Subsystem

The performance of the attitude control system is largely established by the communication and navigation antenna pointing requirements. The spacecraft body axes (x_b, y_b, z_b) must be oriented so they coincide with an earth pointing set of reference axes (x_r, y_r, z_r) to the specified accuracies. The z_r axis is directed at the local vertical, the x_r axis is contained in the orbit plane, and the y_r axis is normal to the orbit plane. The performance accuracy is constrained by the mission requirement to minimize the high gain antenna (communication) and navigation antenna earth pointing uncertainties, thus maximizing effective gain and coverage. The desired attitude control accuracy is ±0.5 degree.

The ACS provides the following functions:

- Sensing, signal processing, control logic and actuation for three-axis attitude control of the NTC satellites
- Thrust vector control for the propulsion engine
- Single axis solar array control
- Attitude and vernier velocity control (cold gas thrusters)

Attitude reference is provided by a three-axis gyro reference package during the transfer orbit and a two-axis IR earth horizon sensor during orbital operation. Sun sensors are used for solar array pointing reference. Control torques are provided by nitrogen cold gas jets, propulsion engine thrust vector control and the solar array drive.

2.2.6.1 Subsystem Description

A block diagram of the chosen three-axis control system is shown in Figure 26. The basic gyroscopic stiffness is provided by the pitch reaction wheel; the wheel spin vector is oriented along the spacecraft pitch axis. Two-axis earth horizon sensors provide roll and pitch attitude information. Orientation about the pitch axis is controlled by means of reaction torques from the wheel drive motor. Earth-orbital kinematics result in an interchange of roll and yaw attitude errors in a sinusoidal manner, allowing attitude control for both roll and yaw channels to be accomplished by means of a single set of jets; no yaw sensor is required. The accumulation of pitch momentum due to secular-type disturbances is corrected by jets operating in an on-off mode.

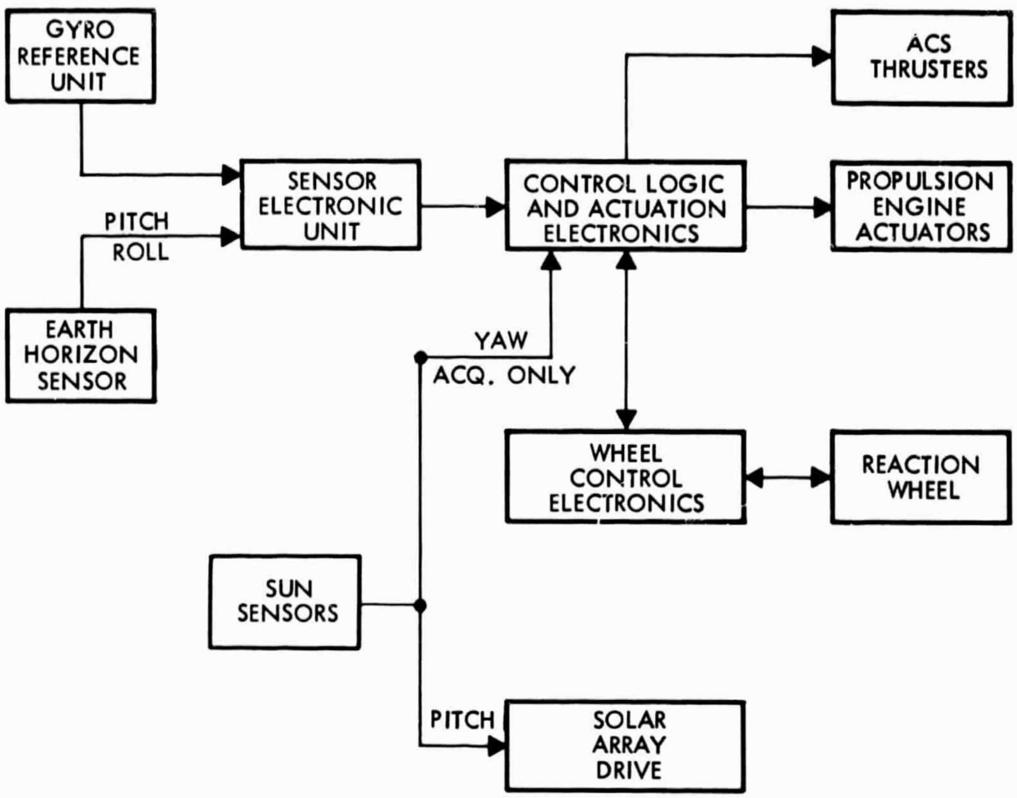


Figure 26. NTC Pitch Momentum Bias ACS Block Diagram

Single axis attitude control of the solar array is provided by the array drive motor. Sun sensor output indicates the angular misalignment between the component of the sun line and plane normal to the array axis and the normal to the array. The sun line component in the plane normal to the array axis can be acquired from any arbitrary orientation at deployment and after eclipse so long as the sun line is within ±35 degrees of the array normal.

The gyro reference assembly provides a three-axis attitude reference for the spacecraft during the coast, powered flight, and earth acquisition modes of operation. Electrical integration of the rates sensed by the gyros provides error signals with respect to inertial reference.

The actuation electronics unit provides output signals to the attitude control thrusters, the solar drive and the engine actuators. The thruster assembly will consist of a pressure regulator, relief valve, pressure transducer, nine solenoid valve and thrust nozzles (eight for attitude control and one for vernier velocity correction), and associated lines and fittings.

2.2.6.2 Control Modes

The attitude control system operates in the following modes:

- Coast
- Powered flight
- Earth acquisition
- Orbit positioning
- Attitude hold or cruise
- Stationkeeping

Coast Mode. This mode is for spacecraft operation immediately upon separation from the injection vehicle. In this mode the spacecraft is in an elliptical orbit with apogee at earth synchronous altitude. The duration of this mode depends on mission parameters and may be as long as 5 hours. Since the primary attitude reference for this mode is provided the gyro reference unit, it is essential to reduce the time duration of this mode. While in this mode, the spacecraft is required to perform rotation maneuvers to orient the apogee engine thrust vector in the proper direction for orbit transfer burn. This maneuver may consist of both pitch and yaw rotation of the spacecraft.

Powered Flight Mode. The powered flight phase is performed by the bipropellant engine to add the velocity required to transfer from the inclined initial elliptical orbit into an equatorial circular orbit at synchronous altitude. Attitude preference for the thrust vector control system is provided by a strapdown inertial unit, the gyro reference unit. The gyro reference unit provides error signals for engine gimbal steering. The primary function of the two-axis engine gimbal control loops is to null the disturbance torques generated by thrust vector displacement from the spacecraft center of mass. The residual angular momentum after engine cutoff is removed by the attitude control system.

Acquisition Mode. Earth acquisition should be performed immediately following completion of the orbit transfer operation. The gyro reference unit may be used to provide the initial acquisition orientation commands, provided acquisition is initiated before the gyro drift becomes excessive. In the event that the gyro reference unit has drifted into saturation, the alternative approach is to perform acquisition using the sun reference. The gyro unit is caged to provide spacecraft angular rate information. Acquisition is performed by the following steps:

- Null body rate using gyro rate reference
- Deploy and lock solar array drive shaft to body; array surface normal to spacecraft body X-axis
- Perform sun acquisition using sun sensor; this will align spacecraft X-axis normal to sun line
- Initiate roll search about the sun line using gyro rate reference
- Upon confirmation of earth lock by the earth horizon sensor, release solar array to control by the solar array drive subsystem
- Establish yaw orientation using sun angle information; reorient yaw axis to align roll axis with orbit plane
- Run up pitch momentum bias reaction wheel.

The acquisition sequence with a flexible solar array may require that the array be deployed after the requisition maneuver has been completed. It is essential that the sun sensor be located so that it can detect angular motion about any normal to the spacecraft X-axis. The following sequence is used for earth acquisition with the array in the stowed position.

- Null body rate using gyro rate reference
- Perform sun acquisition using sun sensor. This will align spacecraft X-axis normal to sun line.
- Initiate earth search by roll (X-axis) motion about the sun line using gyro rate reference.
- Establish earth lock by the earth horizon sensor.
- Establish yaw orientation using sun angle information. Reorient yaw axis to align roll axis with orbit plane.
- Run up pitch momentum bias reaction wheel.
- Deploy solar arrays and communication antenna.

Orbit Positioning. Final orbit positioning is used to move the spacecraft into station. This orbit trim operation is necessary to make up ΔV errors which occur during the apogee engine transfer orbit burn. (This is essentially a ground control operation.) For large ΔV discrepancy it may be desirable to use the bipropellant apogee engine for this mode, for small ΔV deficit this operation is best performed by the stationkeeping propulsion thrusters. The sequence for using the apogee engine for this operation consists of the following.

- Initialize the gyro reference package, cage and uncage the gyros
- Transfer attitude reference to the gyro reference unit operating in a position mode
- Command a 90 degrees pitch maneuver to align the apogee motor thrust vector to the orbit velocity vector
- Set in the proper engine burn time and fire apogee engine
- At completion of the apogee burn, reorient spacecraft to earth reference by means of the commands to the gyro reference unit.

Attitude Hold. In this mode the attitude reference is provided by the earth horizon sensors. This normally requires only pitch and roll attitude reference. Yaw reference is provided by the inertial property of the pitch momentum wheel and the roll/yaw coupling of the orbital kinematics. The attitude control error deadbands for this mode are set at ±0.5 degree. The pitch control is provided by the reaction wheel torque; the roll/yaw control is provided by a mass expulsion jets. The pitch component of the cyclic disturbance torque is stored in the pitch momentum reaction wheel. The secular pitch disturbance torque is also stored by the pitch wheel, however, the accumulated secular torque must be removed by means of the mass expulsion thrusters. The roll/yaw cyclic disturbance is stored as angular displacement from the nominal attitude position. The wheel bias momentum is sized so the maximum cycle disturbance components are contained within the attitude control error deadband. The secular roll/yaw disturbances are removed by means of the mass expulsion thrusters.

Stationkeeping Mode. This is a ground command mode. The linear impulse to provide the necessary stationkeeping ΔV may be supplied by the apogee engine or by special low level thrusters. Three alternatives are possible for accomplishing the stationkeeping operation:

1) The use of the apogee engine for stationkeeping requires reorientation maneuvers to realign the thrust in proper direction. The gyro reference would be used to provide the attitude reorientation reference.

2) A single low level thruster (1.0 lbf thrust level) may be used if the thruster is located with the thrust vector directed through the spacecraft center of mass.

3) The attitude control thruster may be used if a thruster control logic unit is used to give priority to the attitude control functions. The use of the attitude control thrusters to perform the out-of-plane stationkeeping may be impractical: ΔV thrusting would require a long time due to the low thrust level.

2.2.6.3 ACS Subsystem Parameters

Disturbance Torques. The major disturbance torque affecting the operation of the spacecraft at synchronous altitude is solar radiation pressure torque. The major solar pressure torque is assumed to be caused entirely by the solar arrays. The following parameters are assumed in the solar torque calculations:

- Solar array area 70 ft^2 each array
- Solar pressure constant 0.943×10^{-7} lb/ft^2
- Normal solar ray incident angle
- Center of pressure to center of gravity displacement: X-axis 4 inches; Y-axis 2 inches; Z-axis 2 inches.

The solar radiation pressure torque components about each body axis is sinusoidal at orbit frequency. Table 16 summarizes the solar pressure torques.

Table 16. Solar Pressure Torque

Type	t_x	t_y	t_z
Cyclic (peak)	1.1×10^{-6} ft-lb	6.6×10^{-6} ft-lb	1.1×10^{-6} ft-lb
Secular	2.2×10^{-7} ft-lb	1.32×10^{-6} ft-lb	2.2×10^{-7} ft-lb

The secular component of solar pressure is tentatively assumed to be 20 percent of the peak torque for each axis. The magnitude of secular torque directly affects the mission propellant requirement.

During the ΔV thrusting operation misalignment between the thrust vector and spacecraft center of mass would result in attitude disturbance. This disturbance momentum may be reduced by proper alignment of the thrusters; however, center of mass uncertainties would still contribute to to the disturbances. An assumption is made that 1 percent of the translation impulse is coupled into a rotational impulse. The magnitude of the coupled rotational impulse directly affects the mission propellant consumption.

Pitch Momentum Bias Sizing. The pitch momentum bias wheel is sized such that the cyclic torque accumulated each orbit is contained within the attitude control deadband. The total momentum accumulated each half cycle (orbit) is given as:

$$\Delta H_x = \Delta H_z = \text{ave}\left[(12 \text{ hrs})(3600 \text{ sec/hr})(1.1 \times 10^{-6} \text{ ft-lb})\right]$$

$$= 3.03 \times 10^{-2} \text{ ft-lb-sec}$$

The peaks of the ΔH_x and ΔH_z occur 6 hours apart for synchronous altitude operation. The momentum bias necessary to contain the attitude variation within the control system deadband is given as:

$$H = 2 \times 57.3\left[\frac{3.03 \times 10^{-2}}{0.5}\right] = 6 \text{ ft-lb-sec}$$

The factor of two is used because the transient attitude excursion is twice the steady-state attitude error.

Thruster Sizing. The selection of attitude control thrust level has considerable latitude. The important parameter is the control impulse magnitude. The minimum impulse level is chosen to give a full 0.5 degree attitude correction each time the attitude control deadband of ±0.5 degree is exceeded. The minimum impulse required is

$$I = \frac{7}{57.3}(0.5) = 0.06 \text{ ft-lb-sec}$$

The present state-of-the-art thruster control valve has response time on the order of 25 to 30 milliseconds. It is reasonable to expect a control moment arm of 4 feet for a 1100 pound spacecraft. The upper bound in thrust level is given as

$$F = \frac{\ell \Delta t}{I} = \frac{4(.025)}{0.06} = 0.16 \text{ lb}$$

The thruster size could be as large as 0.16 lbf. A smaller thruster level could be used by increasing the thruster on-time by the same proportion. The essential point is that the thrust impulse be maintained at 0.06 ft-lb-sec.

Control Impulse Requirement. The impulse requirement estimates are based on the following assumptions:

- 7 years mission life
- Secular disturbance torque equal to 20 percent of peak cyclic components
- Tipoff rate of 1 deg/sec each axis
- E-W stationkeeping using nitrogen cold gas thrusters
- Residual angular momentum at apogee engine cutoff is 0.1 percent of linear impulse
- Mass properties: I_{xx} = 700 slug-ft^2; I_{yy} = 410 slug-ft^2; I_{zz} = 1000 slug-ft^2; W = 1200 pounds
- N_2 gas specific impulse 70 seconds
- Neglected all orientation maneuver impulse requirement.

Table 17 contains a summary of the required propellant.

Table 17. Summary of the Minimum Propellant Required

Angular Momentum Components	Impulse Required	Propellant Weight
Tipoff momentum	37 ft-lb-sec	0.13 lb
Residual momentum at apogee engine cutoff	372 ft-lb-sec	1.32 lbs
Wheel momentum transfer	40 ft-lb-sec	0.14 lb
E-W stationkeeping	798 lb-sec	11.40 lbs
Accumulated secular momentum	446 ft-lb-sec	1.60 lbs
Total N_2 gas		14.59 lbs

2.2.6.4 System Hardware Description

Most of the attitude control system hardware can be selected from already developed components. A list of candidate subsystem components is given in Table 18.

2.2.7 Power Subsystem

The electrical power subsystem is required to perform the following functions:

- Provide electrical power (from batteries) for spacecraft equipment from launch until the solar array is deployed and illuminated by sunlight. The batteries will also supply the system power during eclipse periods.
- Convert solar energy into electrical energy for system loads and battery charge.
- Provide load and battery power control and regulation.
- Convert electrical power from the main bus voltage to various regulated voltages as required by the spacecraft equipment.

Table 18. Typical Hardware Components

Unit	Unit Required	Weight (lbs)	Power (watt)	Size (inches)
Earth horizon sensor, AOGO type	2	2 (4.0)	12	5 x 8.5 x 3
Sensor electronic assembly	1	5.0	2.5	6 x 4 x 6
Sun sensor a) OGO type	1	2.0	-	6 x 4 x 5
Reaction wheel, OGO yaw unit	1	16.1	53.0 peak starting 6.0	12 dia x 5
Gyro reference unit	1	12.5	20	6 x 6 x 6.5
Solar array drive assembly	1	8.5	4.0	4 x 4 x 16
Array drive electronics	1	3.5	3.0	4 x 4 x 6
Control electronics assembly	2	2 (8.0)	3.5	6 x 12 x 6
Reaction wheel control electronics assembly	1	5.5	2.5	4 x 6 x 6
Valve driver assembly	2	2 (1.2)	0.2	5 x 6 x 1.5
Engine actuator unit	2	2 (1.0)	15 peak 5.0	2 x 3 x 7

The design of the spacecraft to meet the ascent and subsequent orbital life imposes several major constraints on the electric power subsystem. These are:

- Orbit period is 24 hours (synchronous)
- Duration of maximum eclipse is 1.2 hr/orbit
- Solar array consists of two paddles or their equivalent extended from the body of the spacecraft
- Solar paddles shall be flat and may be rotated only about one axis perpendicular to the axis of the spacecraft such that the sun vector will vary a maximum of 23-1/2 degrees off normal with respect to the plane of the solar cells.
- Design life is 7 years. Therefore adequate allowance for solar cell radiation degradation must be made. Also suitable charge control for the batteries must be used to secure long battery life.
- Spacecraft requires power continuously throughout orbit life; it is desirable to have undiminished power during eclipses, but a reduction in communications capacity (with unreduced navigation capability) is acceptable.

The satellite power requirements are shown in Table 19. The normal average power load at the solar array during each orbit is 760 watts. Power requirements during the transfer orbit are 109 watts resulting in an energy requirement of 600 watt-hour during this period. During the worst case eclipse operation (1.2 hours), 920 watt-hours are required. For a reduced eclipse operation of 2 voice, 1 data, and 1 navigation channel, 468 watts are required for a maximum duration of the eclipse or 562 watt-hour of energy. The minimum eclipse operation requirement is to have the navigation channel only in operation, resulting in a requirement of 176 watts (211 watt-hour). During normal load in sunlight about 67 watts are required for battery recharge after a maximum eclipse.

General Description. The electric power subsystem consists of a solar array, a slip ring assembly, three nickel-cadmium batteries, a power control unit, shunt element assemblies, and a central dc-dc voltage converter for communications, attitude control, and other subsystems. The solar array and power control unit provide 28 vdc power to the spacecraft bus during full sunlight portion of the orbital operation. Batteries

Table 19. Power Requirements

Subsystem	Transfer Orbit	Average Power Requirements (watts)		
		Full Eclipse Operation*	Reduced Eclipse Operation**	Minimum Eclipse Operation***
Attitude control	36.6	31.1	31.1	31.1
Power control unit	15.0	29.5	29.5	29.5
Electrical distribution	8.2	8.2	8.2	8.2
Telemetry and command	43.8	43.8	43.8	43.8
Position Det. and Comm.	--	611.0	333.0	55.0
Contingency 5 percent	5.2	36.2	22.3	8.4
Total	108.8	759.8	467.9	176.0

*Normal orbit load (4 voice, 2 data, 1 nav channel)

**2 voice, 1 data, 1 nav channel

***Nav channel only

provide 26 vdc power during eclipses. A functional block diagram of the electrical power subsystem is shown in Figure 27.

2.2.7.1 Solar Array

The solar array previously described must be capable of supplying the system power requirements after 7 years of radiation degradation and when 23.5 degrees off normal to the sun vector. Preliminary calculations indicate that 24 panels of 17 x 30-inch size should give an adequate power margin. The net useful array area covered by solar cells is 16 x 13.4 in two areas for each of the 24 panels. More area can be added by increasing the number of panels and the boom length. The total thickness of each panel is very small and contributes very little to the storage volume. Silicon solar cells 0.008-inch thick is bonded to 0.003-inch thick Kapton substrate. Protective cover glass up to possibly 0.012-inch thick will be bonded to the cells' outer surfaces. The final thickness of cover glass to be chosen will be the result of a tradeoff between weight/volume and cost considerations, with the lower limit dictated by the radiation environment from which the cells must be protected over the desired lifetime.

N-on-P type cells of either 10 or 2 ohm cm bulk resistivity will be used. The choice of the latter will be dependent on the total radiation flux that will be experienced for the period involved. It is expected that 2 x 2 cm silicon cells laid flat and suitably interconnected electrically to give the required voltage and current will be used.

Slip Ring Assembly (SRA). This assembly provides the capability for continuous rotation of the solar array shaft with respect to the spacecraft. The SRA contains the necessary slip rings for transmission of power and signals between the rotating array and the equipment compartment.

Batteries. Batteries are normally connected directly to the solar array and power bus, and in conjunction with the power control unit set the bus voltage between 22 and 32 volts. Three 22-cell, 20 amp-hr, nickel-cadmium batteries are proposed. During normal operation the batteries will supply the spacecraft load during periods of solar eclipses and when system peak loads exceed solar array capability. In the event of one battery malfunction, the failed unit is disconnected. The batteries

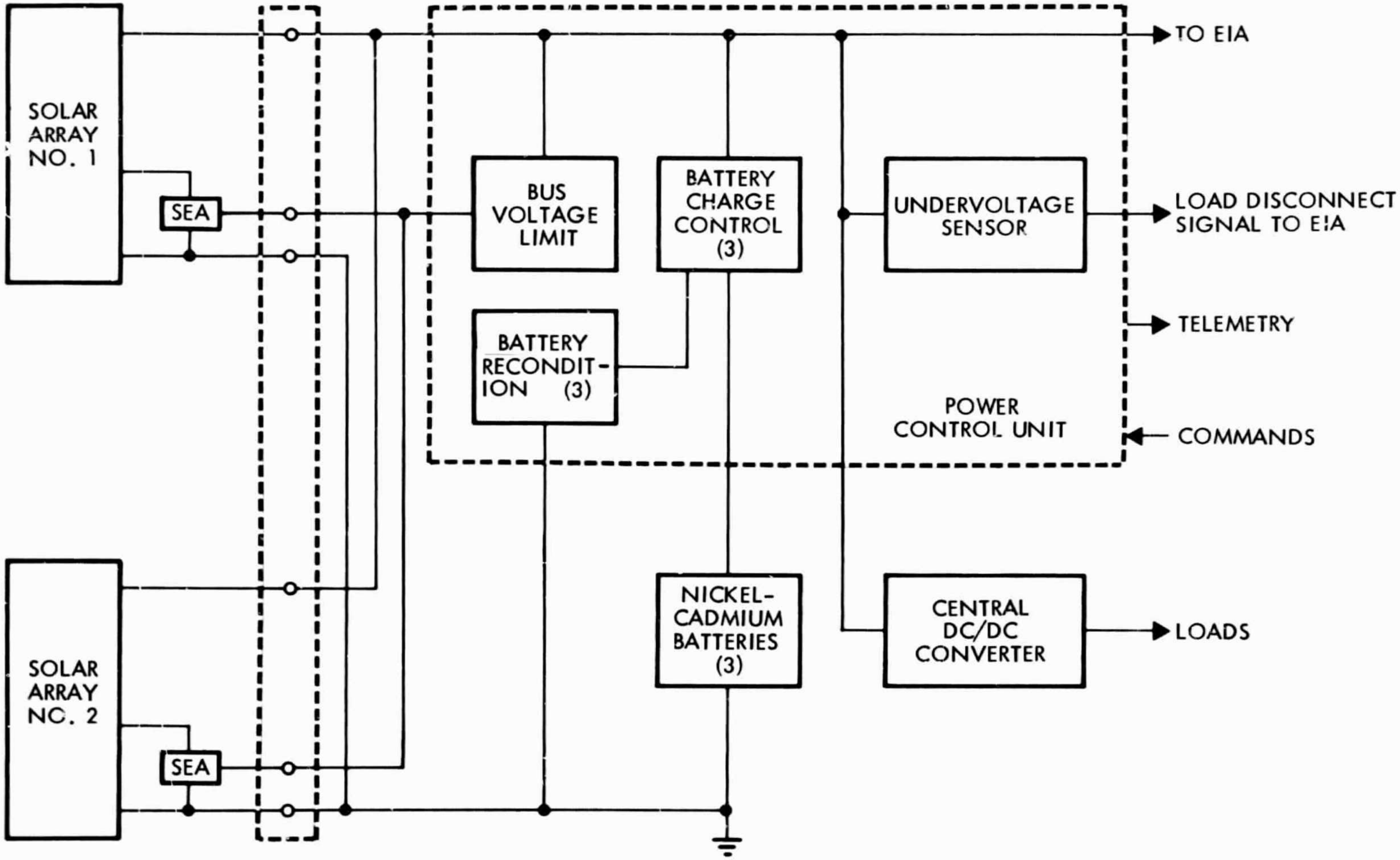


Figure 27. Power Subsystem Block Diagram

will be periodically commanded from the ground to go through a reconditioning cycle of a deep discharge and a suitable recharge.

Power Control Unit (PCU). The PCU provides bus voltage regulation, undervoltage protection, battery charge control, and trickle charge control. Sensing, logic, and switching are included to detect an undervoltage condition. The power control unit provides for battery overcharge protection by both limiting the bus voltage as a function of battery temperature and inserting a trickle charge control resistor into the battery circuit if a high battery temperature is reached. The unit includes provisions for command override of all switching functions and provides power system status telemetry data.

Central dc-dc Equipment Converter. The dc-dc voltage converter supplies the regulated voltages required by communications and other electronic equipment.

2.2.7.2 Solar Array—Battery Interface

In the power system, the battery is normally floated on the load bus. The interface between the battery and the load is not considered a serious problem, since the battery will deliver power at any normal demand rate to the load bus at an acceptable voltage and throughout the entire anticipated temperature range. A problem can arise, however, in the interface between the battery and solar array. The battery when partially discharged is capable of accepting all of the power which the solar array can deliver. However, once the battery is fully charged, it will accept, at most, a limited amount of current without causing the battery cells to be driven to voltages at which gas evolution occurs inside the cells. If the voltage of the load bus is limited to a value below the gassing potential, the battery will accept only that amount of current which it can tolerate with no destructive pressures being generated. If the temperature of the battery were narrowly controlled for any rate of heat dissipation, all that would be needed for charge control would be a simple voltage-limiting device. Since the accuracy of temperature control and the rate of heat removal are limited, additional charge control functions are necessary.

The battery has some overcharge acceptance capability with all overcharge current resulting in the evolution of heat. As the battery begins to dissipate heat (and tends, therefore, to increase in temperature) the battery voltage corresponding to a given constant overcharge current tends to fall. If the voltage impressed across the battery remains the same, the current rises and causes higher heat evolution and currents. This positive feedback of thermal effects can lead to destructive temperatures in a short time if uninterrupted. Consequently, the limiting voltage must be varied in accordance with battery temperature or alternate means must be provided to limit temperature excursions during charging.

Solar Array Cell Characteristics. The silicon solar cells have a nominal solar energy conversion efficiency of approximately 10.5 to 11.0 percent at 28°C and 140 mw/cm^2 air mass 0 (AMO) irradiation.

Battery Type Selection. The following factors must be considered for a battery that is to be used to meet a 7-year life requirement in space.

a) Cycle life

b) Temperature range

c) Depth of discharge

d) Charge control method.

The mission requirement of about 420 deep discharge cycles over a 7-year period plus a 6 to 9-month prelaunch life requirement during fabrication, acceptance, and prelaunch activities rules out silver-zinc and silver-cadmium batteries since their life is limited to 1 and 2 years, respectively, based on present technology.

The nickel-cadmium system has the following life capability:

a) Nickel-cadmium cells tested by TRW have operated more than 6 years, and extensive data exist demonstrating life of more than 3 years with high reliability.

b) The mission profile life tests, conducted on the nickel-cadmium system for the OGO-EGO mission started in early 1962 and completed in 1966, showed no failures. These tests were run at 90°F which exceeds the average expected temperature of this mission.

c) The nickel-cadmium battery proposed is essentially the same battery design which has accumulated nearly 5 million cell-hours of test time at TRW with no design failures and no evidence of design life limitations in cells dating back to 1961.

Battery Sizing. For a maximum eclipse of 1.2 hours, the 760 watt normal load demand represents 35 amp-hr of battery discharge at 26 volts (920 watt-hour). Only 2 cycles/year will reach this discharge level and approximately 580-deep-discharge cycles will occur during the 7-year life. For a maximum allowable depth of discharge for the battery of approximately 65 percent, the rated battery must be approximately 60 amp-hr. Three 20 amp-hr batteries will be used. These batteries will be more than adequate to supply the required 600 watt-hour during the transfer orbit phase. Twenty-two series connected cells will be used in each battery in order to obtain the primary bus voltage range of 22 to 32 volts.

In the event of one battery failure, the remaining batteries can support a normal minimum load including a communications capacity of two voice and one data channel, in addition to the navigation channel (562 watt-hour). If two batteries should fail, the system can still provide the navigation function during eclipse periods (211 watt-hour).

2.2.7.3 Summary of Subsystem Characteristics and Design Margin

The selected solar array-battery system is sized to supply the required continuous 760 watts during each orbit for a period of 7 years under specified orbital conditions. Table 20 shows the performance of the power subsystem as a function of orbit time in years.

2.2.8 Propulsion Subsystem

2.2.8.1 General

It is required that the propulsion subsystem, when incorporated in any of the three configurations, be capable of augmenting the launch vehicle capability such that the required payloads are placed into their respective orbits. Configurations A and B will be launched aboard the Thor/Delta booster; the estimated capability is 720 lbs for synchronous equatorial orbit and 780 for the inclined orbit. Configuration C will be launched into synchronous equatorial orbit aboard a Titan IIIB/Agena. A summary of the required in-orbit capability is given in Table 21.

The orbit transfer from parking orbit to final position, initiated by the Agena D for Configuration C (synchronous equatorial) satellites, must

Table 20. NTC Power Requirements

Requirements	Wattage
System requirements including contingency but excluding battery charge	760 67 (battery charge)
System requirements at bus (all loads)	827
System requirements at solar array (all loads)	15 (diode loss)
	842
Assume electron and proton damage over 7 years = 23 percent (for 0.012 thick cover slides)	
Then necessary solar array capability new = 842/0.77	1093
Must meet above capabilities when solar array is 23.5 degrees off normal	
Therefore solar array capability when normal = 1093/cos 23.5 degrees	1195

Table 21. Launch Vehicle/Propulsion Subsystem In-Orbit Capability

NTC Satellite	Orbit	Launch Vehicle	Capability (lbs)
Configuration A (D and D)	Synchronous Equatorial	Thor-Delta	720
Configuration B (Operational)	Inclined*	Thor-Delta	780
Configuration C (Operational)	Synchronous Equatorial	Titan III-B/Agena	1330

*i = 52.5 degrees, e = 0.35

Apogee = 27,260 n mi

Perigee = 11,340 n mi

be completed by the propulsion subsystem. This function is nominally performed by a single long burn after separation from the Agena followed by a series of shorter burns to converge on the desired trajectory. For the satellites in inclined orbits, it is necessary to perform both perigee and apogee burns after separation from the Thor/Delta launch vehicle. A possible alternative means of satisfying the mission requirements for Configuration C would have been to use a solid rocket motor and a monopropellant vernier system. However, the requirement for multiple burns to place the required satellite into elliptical orbit and the desire to maintain identical propulsion systems for both types of satellites (synchronous equatorial and inclined) results in rejection of this scheme. The differing energy requirements for orbit injection and initial positioning between both types of satellite orbits can be handles by the proposed liquid bipropellant propulsion system.

2.2.8.2 Subsystem Description

A block diagram of the proposed propulsion subsystem is shown in Figure 28. It is a hypergolic bipropellant pressure fed system providing approximately 100 pounds nominal thrust with 380,000 lb-sec minimum total impulse capability. It makes use of the space-storable propellant combination of nitrogen tetroxide (oxidizer) and monomethylhydrazine (fuel). The single rocket engine is gimballed, fixed thrust, and radiation cooled. Nitrogen gas is used as pressurant for the propellant tanks and is also furnished to the cold gas attitude control system. The propulsion subsystem is composed of a pressurant tank, a pressurant control module, four propellant tanks (two oxidizer and two fuel), a propellant control module, a fill and vent module, and an engine module. The operational temperature range is 30 to 90°F; dry weight is approximately 160 pounds, wet weight is approximately 1180 pounds. A weight summary is given in Table 22.

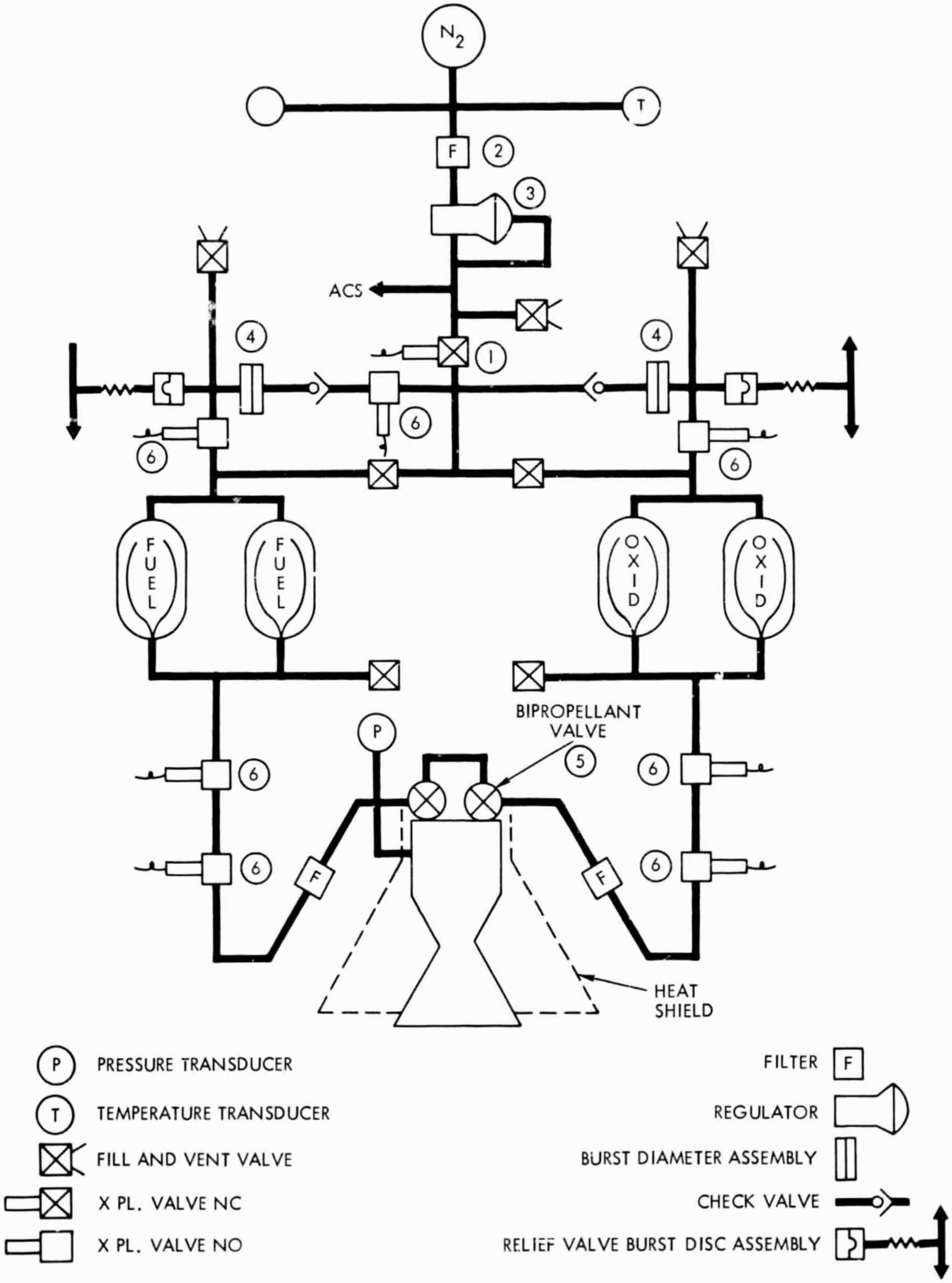


Figure 28. Schematic Diagram of Liquid Bipropellant System

Table 22. Propulsion Subsystem Weight Summary

Pressurant tank	45
Propellant tanks	69
Pressurant control	13
Pressurant valves	5
Fill and vent module	3
Propellant control	7
Propellant valves	5
Engine module	10
Lines and Fittings	4
Dry Weight	161
Propellant	982
Pressurant	33
Propulsion System Total Weight	1176

The operating sequence of the bipropellant system may be described by reference to Figure 28: the normally closed explosive valve (1) is fired releasing nitrogen pressurant which is filtered (2) and regulated (3). Burst discs (4) are ruptured, pressurizing the fuel and oxidizer tanks. From that point on propellant flows are controlled by dual flapper torque motor valves (5), and actuated on command for an unlimited number of firings. After the final burn, normally-open explosive valves (6) are fired to render the system inert.

Engine Module. The engine module is an all columbium, fixed thrust, radiation cooled, liquid bipropellant engine developing approximately 100 pounds of thrust at a combustion chamber pressure of 90 psia.

A radiation cooled heat shield is used to prevent excessive heat transfer to the spacecraft and the engine is single gimballed for thrust vector control (gimbal angle ±5 degrees).

Pressurant Tank. High pressure nitrogen gas (4000 psia) is stored in the pressurant tank; its dual purpose is to expel propellants and to be used with the cold gas attitude and ΔV control system. Pressurant tank volume is approximately 3500 in^3 and its empty weight is approximately 45 pounds maximum.

Pressurant Control Module. The pressurant control module controls the pressure and flow of nitrogen gas from the pressurant tank to the propellant tanks. It consists of redundant pressure regulators, relief valves, high and low pressure telemetry transducers, check valves, manual valves, pyrotechnic start valves, and pyrotechnic valves for pressurant isolation and reactivation operations.

Propellant Tanks. Four identical propellant tanks are used, two for oxidizer and two for fuel. Each consists of a titanium tank shell, bladder, and stand pipe. Three ports are used: propellant, propellant feed, and pressure.

Propellant Control Module. The propellant control module controls the flow of propellants from their tanks to the engine. Two identical but isolated flow paths are provided: one for oxidizer and one for fuel. Each flow path consists of pyrotechnic valves, pressure and temperature telemetry transducers, filter, and feed system orifice.

Fill and Vent Module. The fill and vent module provides the capability of servicing the propulsion subsystem with propellants and pressurant. It consists of poppet-type manual valves mounted to a panel. For flight usage the valves are torqued closed and capable of providing redundant sealing.

2.2.9 Electrical Distribution Subsystem

The electrical distribution subsystem (EDS) is the integrating element for all other electrical subsystems of the NTC satellite. It distributes power, processes commands, conditions telemetry signals, sequences operations, actuates ordnance, and distributes command telemetry and RF signals.

The subsystem is composed of three parts —the electrical integration assembly (EIA), the telemetry interface module (TIM), and the spacecraft harness.

2.2.9.1 Electrical Integration Assembly

The electrical integration assembly performs the following functions:

- Processes and distributes commands
- Conditions telemetry signals
- Controls undervoltage bus
- Controls spacecraft ordnance
- Programs and executes onboard sequences.

The EIA weighs approximately 9 pounds and together with the TIM draws approximately 6.2 watts of dc power.

2.2.9.2 Telemetry Interface Module

The telemetry interface module cross-straps telemetry inputs to the redundant encoders. It weighs approximately 6 pounds.

2.2.9.3 Spacecraft Interconnecting Harness

The spacecraft harness distributes electrical power and signals to all spacecraft subsystems. It includes cabling within the main body of the spacecraft, cabling across antenna support arm and deployment mechanisms, solar array harness, umbilical ordnance, and test connectors and spacecraft/booster interstage cabling. Estimated weight is approximately 10 pounds for the solar array harness and 55 pounds for the remainder of the spacecraft harness.

2.2.10 Thermal Control Subsystem

2.2.10.1 Thermal Requirements

The thermal control subsystem is required to provide acceptable thermal environment and temperature levels for the satellite and its components during all mission phases including ascent, transfer orbit, orbit injection, and orbital operation (including eclipse periods). A listing of component temperature limits is given in Table 23. Inherent in the long mission life is the requirement to provide sufficient flexibility in the design of the thermal subsystem to accommodate variations in subsystem functional performance. Examples of these variations include solar panel degradation, component failures, degradation of thermal coatings, and allowance for alternate operating modes.

2.2.10.2 Subsystem Description

Thermal control of the satellite and subsystem equipment is achieved through use of surface coatings, base plate design, multilayer insulation and bimetal-actuated louver assemblies. Acceptable temperature levels are maintained through the use of multilayer aluminized Mylar or Kapton insulation assemblies, louver assemblies, passive radiating surfaces, component internal power dissipation, thermal coatings, heat sinks, and conductive heat path control.

Louver assemblies are incorporated on the lower portion of the satellite to provide thermal control dynamic response required to compensate for variations in large power dissipation. A radiator panel is placed between each louver assembly and the external environment. Second-surface mirrors are placed on the face of the panels to maintain low radiator temperatures and minimize solar heat input. The louver blade is operated by a bimetallic actuator which senses the local component mounting temperature.

Multilayer aluminized Mylar and Kapton insulation assemblies enclose all of the lower portion of the satellite except that portion required for the louver assemblies. Mylar is used wherever possible because of its low cost, ease of fabrication and light weight. In the areas that are exposed to high engine temperatures, Kapton is required because of its higher allowable temperature limits.

Table 23. NTC Component Temperature Requirements

Subsystem	Temperature Requirements (°F) (minimum — maximum)	
POWER SUBSYSTEM		
Batteries	40	95
Shunt assembly	- 85	300
DC-DC converter	15	130
PCU	15	120
Solar array assembly	-268	129
ATTITUDE CONTROL SUBSYSTEM		
Sun sensor assembly	0	120
Gyro reference assembly	92	120
Engine actuator assembly	0	110
Solar array drive assembly	- 15	125
Horizon sensors	0	120
Sensor electronics	0	120
Control electronics	0	100
Actuator electronics	0	100
Reaction wheel	0	120
PROPULSION SUBSYSTEM		
Pressurant control module	0	120
Propellant tanks	40	100
Nitrogen tank	40	100
Propellant supply module	40	100
POSITION DETERMINATION AND COMMUNICATION SUBSYSTEM		
Transmitter	0	110
TWT and power supply	0	190
Receiver	0	110
Power amplifier (solid state)	0	110
Frequency synthesizer	0	110
Oscillator	50	70
Navigation signal generator	0	110
RF switches	0	110
L-band diplexer	0	110
TELEMETRY AND COMMAND SUBSYSTEM		
Command receiver	0	110
Command decoder	0	110
Telemetry transmitter	0	110
Data encoder	0	110
Diplexer	0	110
Hybrid	0	110
ANTENNA SYSTEM		
High gain antenna	-250	200
Earth coverage antenna	-250	200

The temperature of internally mounted electronics equipment is controlled by passive means, including structure design control to minimize heat leaks, component surface coating and component location restrictions.

The internal and external surface coatings, insulation, louvers, etc., will be chosen to provide the required environment at minimum cost and weight.

The number of independently actuated louvers on each panel of the modules makes them inherently reliable; the failure of a given louver on a panel would degrade performance slightly but would not induce catastrophic failure. The other components of the subsystem are passive and, therefore, highly reliable.

2.2.11 Mass Properties

A preliminary weight estimate is presented in Table 24 for the NTC Configuration C Satellite. The satellite is compatible with the Titan IIIB/Agena launch vehicle, with the propulsion system sized so that sufficient propellant can be loaded to provide an injection velocity at synchronous apogee of 6400 fps to a 2,250 pound satellite (approximately the payload capability of the Titan IIIB/Agena less a 50 pound allowance for the adapter). In addition, an allowance of 200 fps has been made to cover a possible repositioning maneuver, commensurate with a 90 degree longitude shift.

The satellite is estimated to weight 1,961 pounds including a 10 percent hardware contingency. Based upon this estimate, and utilizing the launch vehicle capability to its fullest extent, we have approximately 133 pounds of available weight. As indicated in Paragraph 2.1, much of this weight margin would be used to further increase the reliability of the satellite.

2.2.12 Reliability

A discussion of assumptions, methods, and failure rates and their data sources used in performing a reliability analysis of the navigation/air traffic control satellite is presented in this section. Reliability block diagrams for the system and its subsystems are included. The reliability of each subsystem and of the system was determined for each of the seven years of mission duration. Table 25 presents a summary of the reliability findings.

Table 24. **Preliminary Weight Estimate — Navigation Traffic Control Satellite**

Item	Units	Weight (lb)
STRUCTURE AND THERMAL CONTROL		98.2
Top cover	(1)	6.3
Side panels	(8)	17.1
Bottom panel installation	(1)	19.0
Truss assembly		32.9
N_2 tank supports		1.0
Propellant tank supports		3.0
Engine supports		1.0
Miscellaneous hardware		1.0
Insulation		5.5
Louver assemblies		6.4
Heat sinks		5.0
POWER SUPPLY		273.9
Solar array paddles	(2)	80.4
Batteries	(3)	150.0
Power control unit	(1)	10.0
Central dc/dc converter	(2)	8.0
Inverter	(1)	2.5
Shunt assembly	(2)	17.0
SA deployment motors	(2)	6.0
ELECTRICAL INTEGRATION		80.0
Electrical integration unit	(1)	9.0
Telemetry integration module	(1)	6.0
Solar array harness		10.0
Cabling and connectors		55.0
ATTITUDE CONTROL SYSTEM		93.7
Gyro reference assembly	(1)	12.5
Horizon sensor assembly	(2)	8.0
Sensor electronics	(1)	5.0
Control electronics	(2)	16.0
Actuation electronics	(1)	8.7
Sun sensor assembly	(1)	1.1
Solar array drive and resolver	(1)	8.5
Engine actuator	(2)	2.0
Reaction wheel	(1)	16.1
Reaction wheel control electronics	(1)	5.5
Nozzles, valves, and plumbing		10.3
TELEMETRY AND COMMAND SYSTEM		29.3
Command receiver	(2)	8.4
Command decoder	(2)	3.4
Telemetry transmitter	(2)	7.2
Telemetry encoder	(2)	3.8
Diplexer	(1)	2.0
Hybrid	(2)	1.5
Dc-dc converter	(1)	3.0

Item	Units		Weight (lb)
POSITION DETERMINATION AND COMMUNICATION SYSTEM			90.2
Transmitter	(1)		2.5
TWT and power supply	(4)		40.0
Receiver	(1)		4.0
Power amplifier (solid state)	(2)		9.0
Frequency synthesizer	(1)		2.0
Reverence oscillator	(2)		12.0
Amplifier	(1)		0.5
BPF	(1)		0.8
Navigation signal generator	(2)		12.0
SPDT RF	(6)		2.4
L-band diplexer	(1)		2.5
Frequency multiplexer	(1)		2.5
ANTENNA SYSTEM			32.9
High gain antenna	(1)		27.2
Earth coverage antenna	(1)		3.7
Telemetry and command antenna	(2)		2.0
PROPULSION SYSTEM			161.4
Propellant tank installation	(4)		69.0
Pressurization system			63.0
N_2 tank installation	(1)	45.0	
Control module	(1)	13.0	
Valves	(2)	5.0	
Propellant feed system			19.4
Lines and fittings		4.0	
Fill and vent	(1)	3.0	
Supply module	(1)	7.4	
Supply valve	(2)	5.0	
Engine installation			10.0
CONTINGENCY (10 PERCENT)			86.0
TOTAL DRY SATELLITE WEIGHT:			945.6
PROPELLANT AND PRESSURANT			1,015.0
Residuals			34.0
Propellant		30.0	
Nitrogen		4.0	
Expendables			981.0
Impulse propellant		952.0	
Impulse ACS nitrogen		26.0	
Nitrogen leakage		3.0	
GROSS SATELLITE WEIGHT:			1,960.6
ADAPTER AND SEPARATION			50.0
Adapter			35.0
Separation system			15.0
BOOSTER PAYLOAD WEIGHT:			2,010.6

Table 25. Reliability for Indicated Year*

Subsystem \ Year	1	2	3	4	5	6	7
Position determination and communication	0.9715	0.9185	0.8481	0.7672	0.6815	0.5958	0.5134
Power	0.9956	0.9908	0.9857	0.9793	0.9630	0.9023	0.7401
Propulsion	0.9848	0.9797	0.9745	0.9694	0.9643	0.9593	0.9543
Attitude control	0.9430	0.8825	0.8114	0.7326	0.6501	0.5679	0.4892
Antenna	0.9930	0.9930	0.9930	0.9930	0.9930	0.9930	0.9930
Thermal control	0.9998	0.9996	0.9993	0.9991	0.9989	0.9987	0.9985
Telemetry and command	0.9897	0.9720	0.9485	0.9202	0.8883	0.8538	0.8173
Structure	0.9900	0.9900	0.9900	0.9900	0.9900	0.9900	0.9900
Electrical integration assembly	0.9916	0.9831	0.9745	0.9659	0.9572	0.9484	0.9396
TOTAL SYSTEM	0.8664	0.7388	0.6003	0.4658	0.3435	0.2328	0.1337

*A computer program was used to calculate the results in this table and to arrive at a system MTTF of 45.52 months for a 7-year design life of the satellite. This MTTF incorporates both the effects of failure due to random causes and due to wearout. The program thus calculates the MTTF by the relation

$$\mathrm{MTTF} = \int_0^{\text{seven years}} R(t)\,dt = 45.52 \text{ months}$$

2.2.12.1 Reliability Assumption

The basic exponential reliability mathematical model for black boxes was used in performing the analysis; modifications to account for binomial, standby redundancy, and active redundancy reliability configuration were incorporated. Equations used for different configuration are as follows:

a) No redundancy

$$R = e^{-\lambda t}$$

b) Active parallel single redundancy (equal failure rates)

$$R = 2e^{-\lambda t} - e^{-2\lambda t}$$

c) Single standby redundancy, perfect switching

$$R = e^{-\lambda_a t}\left[1 + \frac{\lambda_a}{\lambda_i}\left(1 - e^{-\lambda_i t}\right)\right]$$

where

λ_a = failure rate of active (operating) equipment

λ_i = failure rate of standby equipment

d) Double standby redundancy, perfect switching

$$R = e^{-\lambda t}\left[R_1 - R_2 - R_3\right]$$

$$R_1 = 1 + \lambda_a/\lambda_i + \lambda_a/2\lambda_i + \lambda_a^2/2\lambda_i^2$$

$$R_2 = e^{-\lambda_i t} (2\lambda_a/\lambda_i + \lambda_a^2/\lambda_i^2)$$

$$R_3 = e^{-2\lambda_i t} (\lambda_a/2\lambda_i + \lambda_a^2/2\lambda_i^2)$$

e) Binomial type redundancy, equal λ's, k out of n equipment required for success

$$R = R_s^n + nR_s^{n-1} (1 - R_s) + \ldots + \frac{n(n-1\ldots(n-k+1)}{k!} R_s^k (1 - R_s)^{n-k}$$

where

R_s = reliability of each equipment

The failure rates used and their sources are given in ea·h subsystem reliability discussion. Standby failure rates of 10 percent of the active failure rates were assumed. A 10 percent duty cycle was imposed on two components (telemetry transmitter and telemetry encoder in series with another pair in standby redundancy) in the telemetry and command subsystem. Each one-shot item (i.e. components used once) was assigned a probability of success for its "one operation" mission. The probabilities of success used for each battery (in the power subsystem) in the analysis were 0.999 (1 year), 0.992 (2 years), 0.964 (3 years), 0.895 (4 years), 0.770 (5 years), 0.590 (6 years), and 0.386 (7 years).

2.2.12.2 Reliability Analysis

Table 26 presents a listing of components, quantities used, component configurations, failure rates, and failure rate information sources for each of the nine subsystems of the spacecraft. The total system reliability block diagram is shown in Figure 29; diagrams for the subsystems follow in Figures 30 through 37. The reliabilities shown in each box reflect a 3-year operating period. Again, the system has not been optimized for reliability. Using the substantial weight margin available, substantially higher reliabilities can be achieved. Appendix D of Vol. II contains a discussion of satellite reliability as it relates to spares requirements.

Table 26. Subsystem Equipment and Failure Rates

Subsystem	Equipment	Quantity Used	Configuration	Failure Rate (No. x 10^{-9}/Hr)
1. Position determination and communication	L-band diplexer	1	Nonredundant	40
	Frequency multiplexer	1	Nonredundant	40
	Isolat[illegible]	1	Nonredundant	40
	Trans[illegible]	3	Two in standby redundancy, 1 nonredundant	8000 (active) 600 (nonredundant)
	TWT and power supply	4	Two sets of standby redundant equipment	10172 (active)
	Receiver	1	Nonredundant	850
	Solid state power amplifier	2	Standby redundancy	5000 (active)
	Frequency synthesizer	2	Standby redundancy	2294 (active)
	Reference oscillator	2	Standby redundancy	8772 (active)
2. Power	Power control unit	1	Internal redundancy	190
	Battery	3	Binomial type redundancy, 1 of 3 required for vast majority of mission	See probabilities discussed in text
	Dc/dc converter	2	Standby redundancy	2000 (active)
	Slip ring assembly	1	Nonredundant	240
	Shunt assembly	1	Nonredundant	20
	Solar arrays	1	Nonredundant	40
3. Propulsion	Nitrogen tank	1	Nonredundant	20
	Pressurant control assembly	1	Nonredundant	500
	Propellant tanks	1	Nonredundant	80
	Propellant control assembly	1	Nonredundant	Probability of success = 0.995
	Engine assembly	1	Nonredundant	Probability of success = 0.995
4. Attitude control	Gyro reference assembly	1	Nonredundant	Probability of success = 0.995
	Horizon sensor assembly	4	Binomial type redundancy; 3 required for success	2119

Table 26. (Continued)

Subsystem	Equipment	Quantity Used	Configuration	Failure Rate (No. x 10^{-9}/Hr)
(Attitude control)	Sensor electronics	1	Nonredundant	1409
↓	Control electronics	3	Double standby redundancy	28, 277
	Actuation electronics	1	Nonredundant	500
	Sun sensor assembly	1	Nonredundant	1,457
	Thruster assembly	1	Nonredundant	200
	Solar array drive	3	Active redundancy 1 nonredundant	800 (active redundancy) 870 (nonredundancy)
	Engine actuator	1	Nonredundant	40
	Reaction wheel	1	Nonredundant	64
	Wheel control electronics	1	Nonredundant	1230
5. Antenna	High gain antenna	1	Nonredundant	Probability of success = 0.995
↓	Earth coverage antenna	1	Nonredundant	Probability of success = 0.995
	TT&C antenna No. 1	1	Nonredundant	≈0
	TT&C antenna No. 2	1	Nonredundant	≈0
6. Thermal	Louvers		Highly redundant	5
↓	Heaters		Highly redundant	20
	Insulation		Highly redundant	≈0
7. Tracking, telemetry and command	Diplexer	1	Nonredundant	40
↓	Hybrid	1	Nonredundant	10
	Command receiver	2	Active redundancy of one each in series	5278 } 7078
	Command decoder	2		1800
	Telemetry transmitter	2	Standby redundancy of one each in series	2854 } 17,247 10% duty cycle
	Telemetry encoder	2		14393
	Transfer switch	1	Nonredundant	250
	Receiver selection logic	1	Nonredundant	400
8. Structure	Structural members	<200	Nonredundant	Probability of success = 0.99
9. Electrical integration	Electrical integration unit	3	Two in active redundancy, 1 nonredundant	1026 (active) 615 (nonredundant)
↓	Telemetry integration modules	1	Nonredundant	190
	Cable and connectors	1	Nonredundant	150

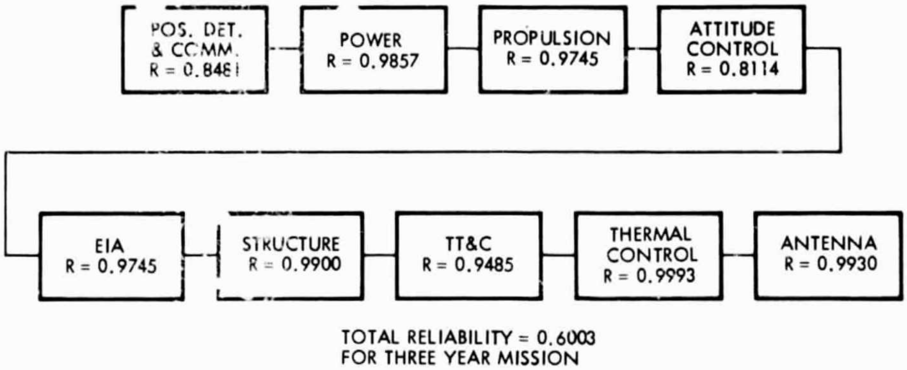


Figure 29. Reliability Block Diagram of Navigation/Air Traffic Control Satellite

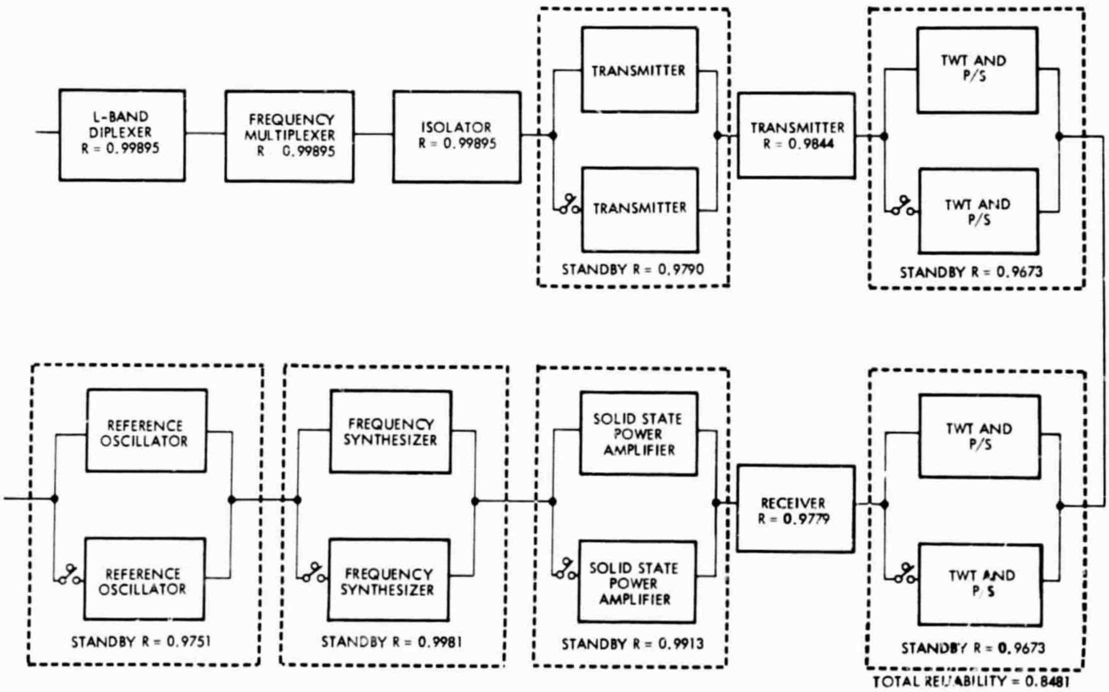


Figure 30. Position Determination and Communication Subsystem Reliability Block Diagram

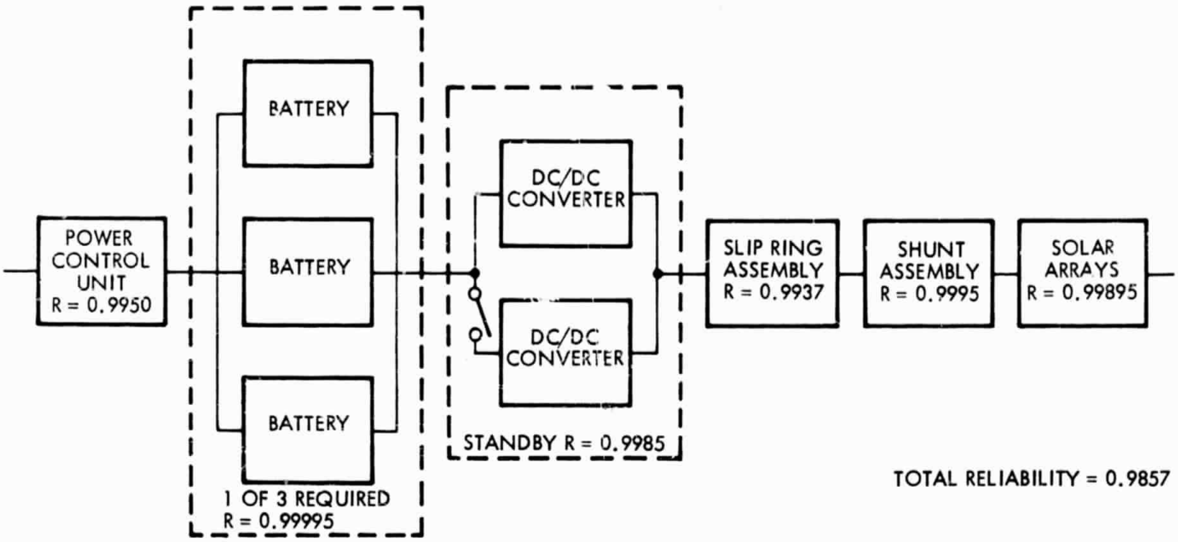


Figure 31. Power Subsystem Reliability Block Diagram

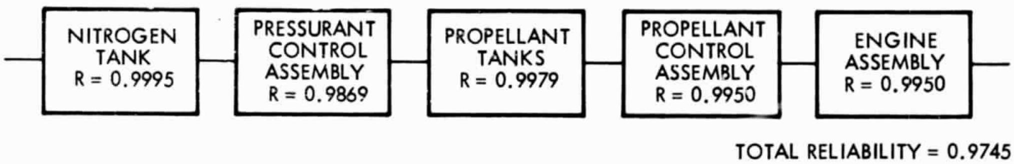


Figure 32. Propulsion Subsystem Reliability Block Diagram

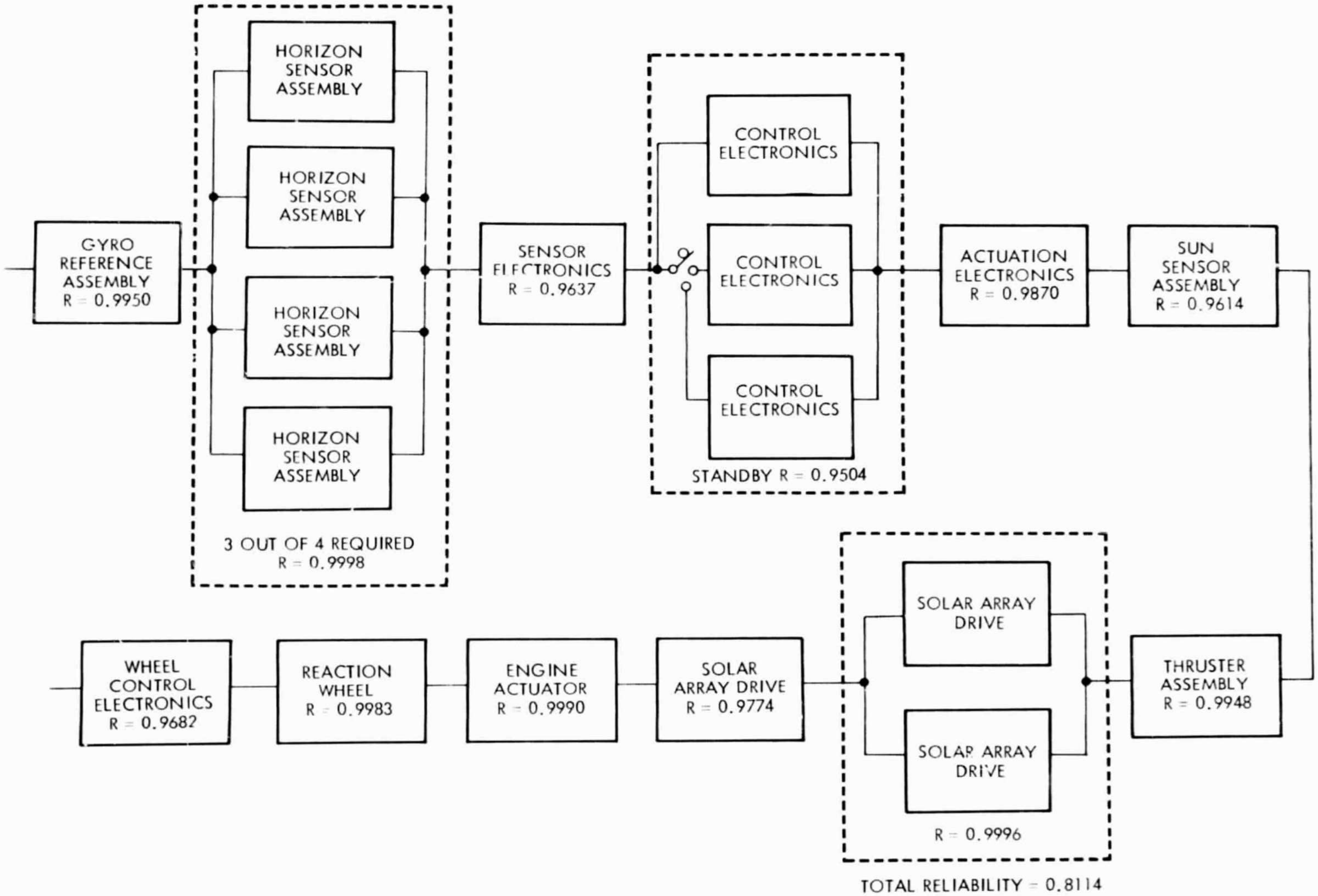


Figure 33. Attitude Control Subsystem Reliability Block Diagram

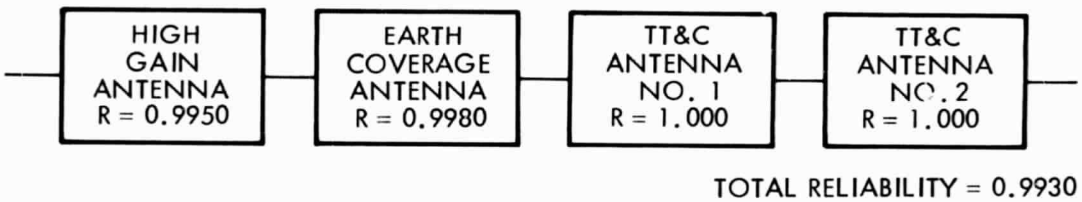


Figure 34. Antenna Subsystem Reliability Block Diagram

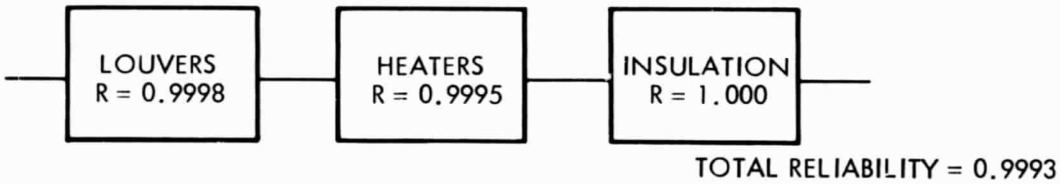


Figure 35. Thermal Subsystem Reliability Block Diagram

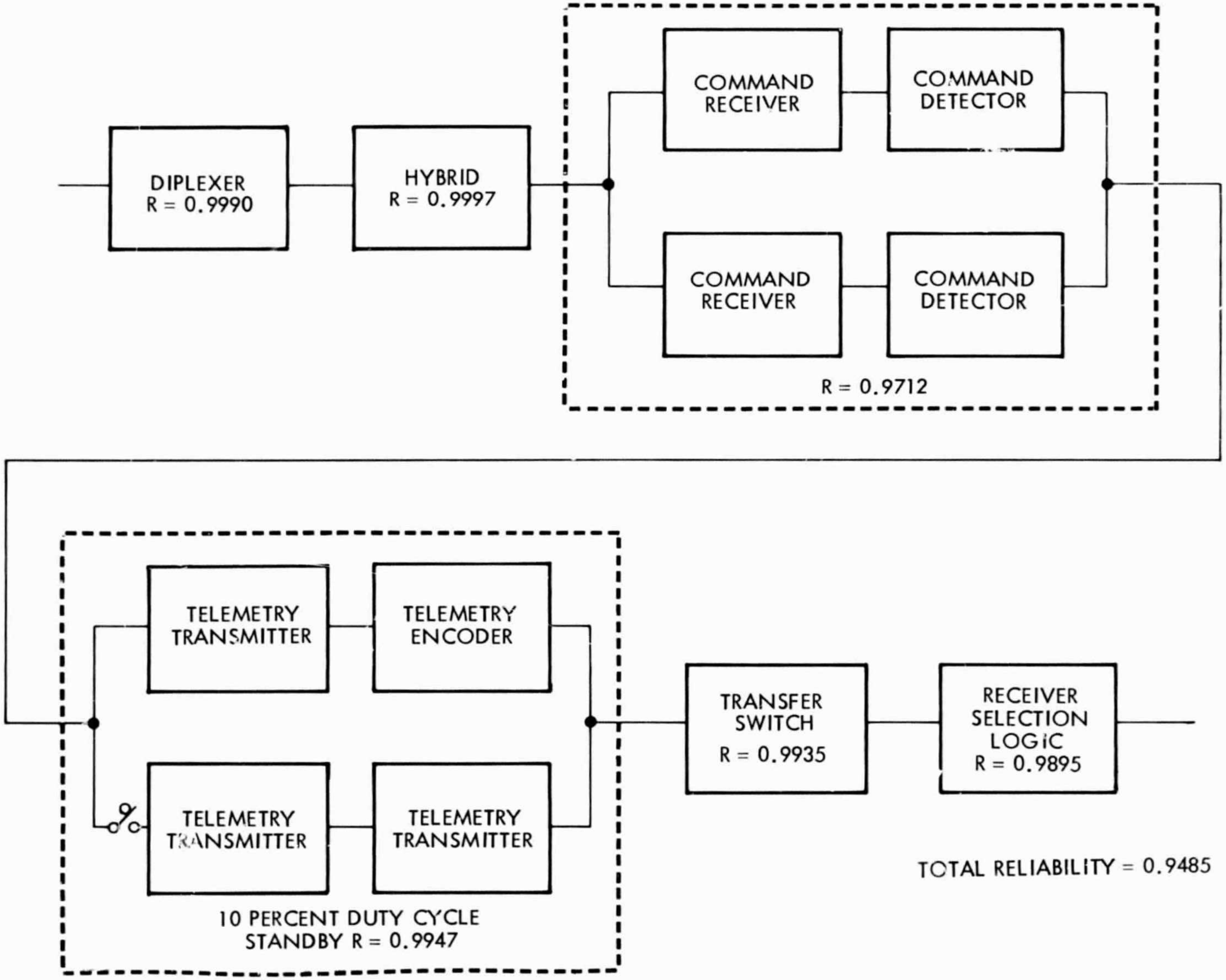


Figure 36. Telemetry and Command Subsystem Reliability Block Diagram

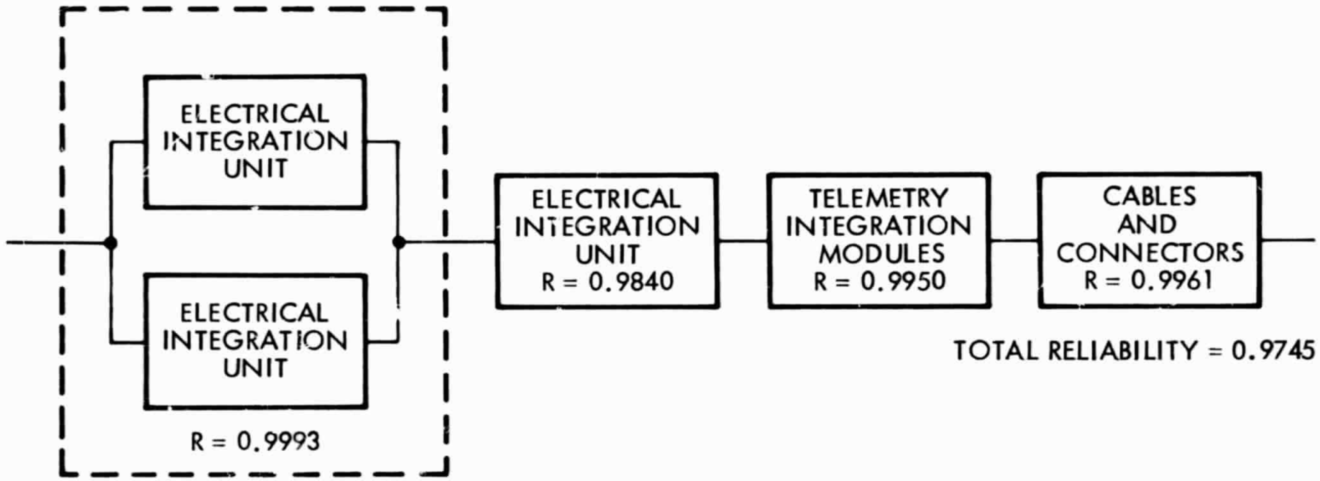


Figure 37. Electrical Integration Subsystem Reliability Block Diagram

2.3 USER HARDWARE

2.3.1 General

The user hardware is part of an integrated satellite communication/navigation system for large aircraft that provides capabilities for communicating voice and printed messages as well as supplying aircraft position information in data form to the ground terminal. Transfer of the aircraft received position information, time of day, and altitude is made as an automatic function (i. e., it does not involve action or operation by any of the aircraft crew). Included within this transfer (called Autorep) is a piggyback special message that is set by the personnel on board the aircraft. This special message includes an emergency alert and a request to allow transmission of digital weather and company communications.

The aircraft has one Autorep transmit/receive (T/R) data channel, two printed message T/R data channels (this is referred to in the text as weather/company message), twelve voice T/R channels (one of which is used for emergency) and one position determination channel which receives only NAVSTAR signals transmitted to the aircraft by the satellites. The NAVSTAR signals can also be processed in the aircraft to give navigation information.

2.3.2 User Satellite Communication/Navigation System

The user system is shown in block diagram form in Figure 38, consisting of three main subsystems:

- The transmitter/receiver (T/R) unit
- The digital unit
- The NAVSTAR unit.

Additional periphery equipment consists of display and control units.

The T/R unit transmits and receives to and from the satellite system via a radio frequency link operating in the L-band spectrum. Modulation and detection of the voice and data communication are provided in this unit along with a translated down (i. e., from L-band) RF signal that is supplied to the NAVSTAR processor.

The digital unit arranges, stores, and releases, in the form of data, both processed NAVSTAR information and weather/company printed messages. The NAVSTAR information fills an Autorep message format in

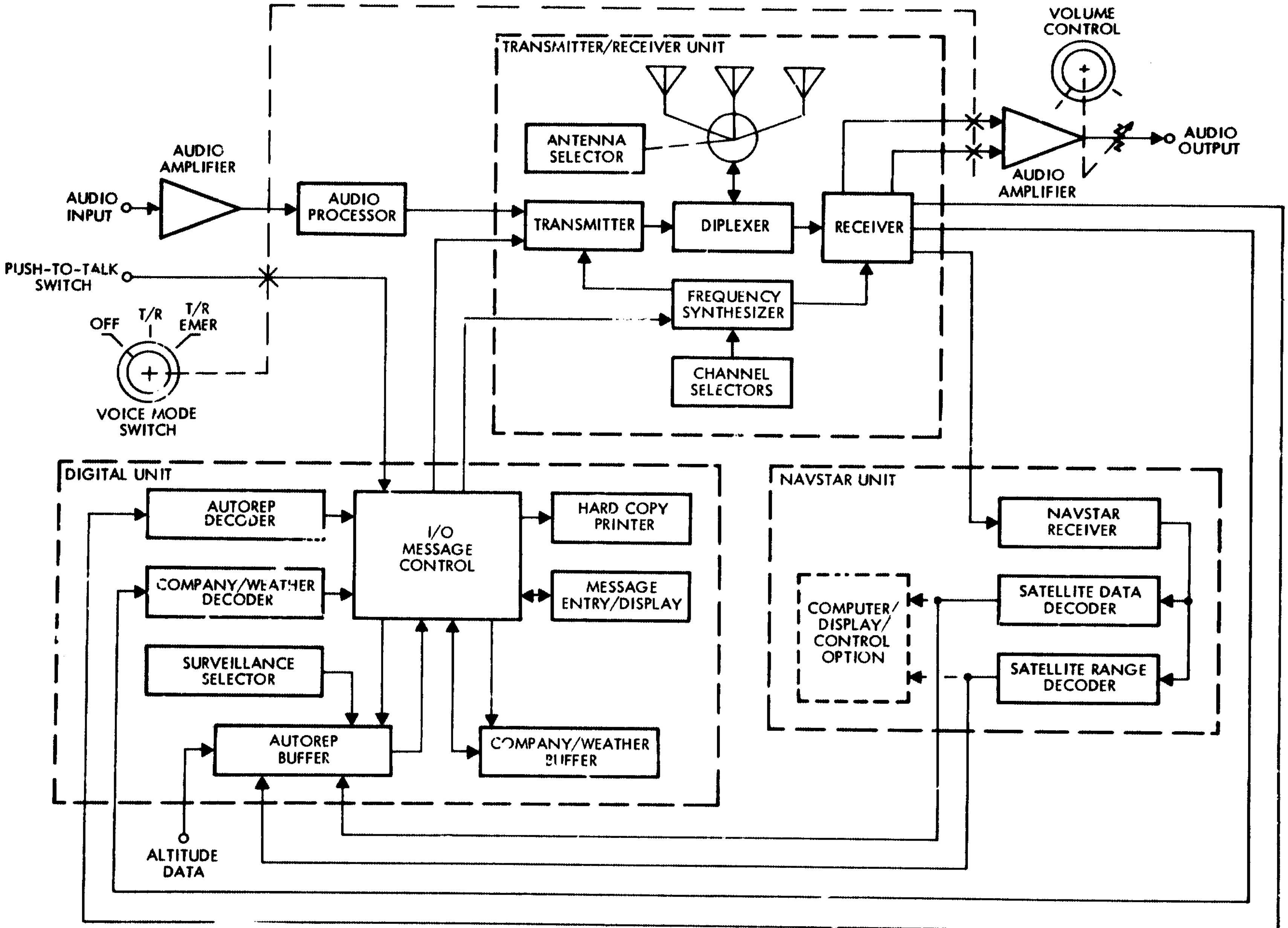


Figure 38. User Hardware Functional Block Diagram

this digital unit, which is then communicated automatically to the ground terminal via the satellite on command from the ground. The weather/company are printed messages that are communicated also via the satellite (as are all communications) to and from the ground terminal. Weather/company messages from the aircraft are first stored in this digital unit and then released on command from the ground as a time interleaved data signal with the autorep message.

The NAVSTAR unit receives the RF signal from the T/R unit, amplifies and detects the NAVSTAR signal, then processes the information received from a group of satellites. This processed signal identifies each satellite and gives the time of day and range information of the aircraft to each satellite. This processed information is relayed to the digital unit for retransmission to the ground terminal.

2.3.3 Transmitter/Receiver Unit

2.3.3.1 T/R Functions

The T/R unit consists of five main functions as shown in Figure 38. These functions are:

- Antenna
- Diplexer
- Receiver system
- Transmitter system
- Frequency synthesizer and channel selectors.

The antenna, an array of 3 slotted dipoles having a gain of 3 to 6 db over a 160 degree cone and flush mounted, receives and transmits information from and to the navigation satellite. To obtain 160 degree coverage it is necessary to have three of these antennas and switch from one to another depending on the satellite location. This switching may be manual or automatic, or both.

The diplexer isolates the energy between the transmitting and the receiving section of the system. Information in the form of both voice and coded data is processed through the transmitter and receiver. A separate discussion will cover the data processing of the information with this analog system. The following is a discussion of the salient features of this particular system.

Receiving Frequencies. The receiving frequency band has been selected to cover 1540 to 1560 MHz. Within this 20 MHz bandwidth there are allocated:

- Eight data channels - four channels for Autorep signal* and four channels for weather/company communications
- Fifteen** voice communication channels
- One special emergency voice channel ("guard channel")
- A 2-MHz bandwidth channel for NAVSTAR navigation information.

System considerations must be given to the channel allocation for these signals. The navigation and Autorep are operated at a high duty cycle, and are considered to operate continuously. Although at first only one channel will be operated for the weather/company communications, it is expected that as traffic becomes greater a second channel will become necessary. The weather/company messages are received less frequently than the autorep signals. Voice messages for one user will operate only about 5 percent of the time and emergency messages, although the receiver must be open all the time, are very much less frequent than normal voice messages.

One of the considerations in any multiple signal environment receiver is the generation of intermodulation (IM) products by the receiver system itself. Selected positioning of signals can reduce the effects of IM and effectively improve a system performance. For this reason special allocations should be made in the placement of the different types of received signals.

Transmitter Frequencies. The transmitter frequencies must also be considered on a system basis. The transmitting frequency band has

*One Autorep channel is used for communications between the ground and the satellite.

**One additional voice channel has been allocated for marine use.

been selected to cover 1640 to 1660 MHz. The system must be capable of transmitting over any one of:

- Eight data channels - four channels for autorep and four channels for weather/company.
- Twelve voice communication channels
- One special emergency voice channel ("guard channel").

In transmission only one of these channels will be operated at any time. In order to limit peak power requirements the voice transmission which averages only 5 percent of the transmitter time for one aircraft is interleaved with the data. The Autorep transmitted data which requires about 0.25 second takes first priority and will actually block out a fraction of a second of aircraft transmitted voice messages if an Autorep report is commmanded during a voice transmission. The weather/company messages have equal priority with voice, in that a voice transmission will not block out an outgoing company message, nor will a company or weather transmission block out a voice transmission. It is felt that no significant operational penalties will result from this arrangement.

Isolation. To reduce the fundamental transmitter frequencies as seen at the receiver input, it is necessary to use a diplexer. Today's diplexers are capable of as much as 50 to 60 db isolation. Special diplexer design can achieve 90 db isolation with 0.8 db insertion loss while giving 60 db or better rejection of the transmitter harmonics. Typically, the size of this diplexer would be about 6 x 6 x 12 inches. With the transmitter power of +25 dbw (316 watts) the transmitter signal seen at the receiver would have a power level of -35 dbm. Greater isolation of the transmitting frequencies, at the receiver front end, can be obtained by adding additional filtering in the receiving path in the diplex or between the diplex and the preamplifier. This filtering would result in increased insertion loss between the antenna and the preamplifier and hence degrade the receiver noise figure by the amount of the insertion loss, which would be approximately 0.5 to 1.5 db.

Other Problems. Since low noise operation is required, the diplexer insertion loss must be minimized. This means that the receiver front end must be able to operate with high cross modulation signals. These signals, higher in frequencies by about 100 MHz, should not cause any overloading of the receiver. It should be pointed out that the isolation obtained from the diplexer is sensitive to the antenna voltage standing wave ratio (VSWR).

Tests performed on an Avantek Model AS-1000 preamplifier showed no degradation of performance in the noise figure or with any cross modulation when a -5 dbm signal was used to simulate the transmitter signal as seen at the receiver. Since the special design diplexer (described above) would give a large amount of isolation it is possible to maintain the transmitter signal level below -35 dbm at the receiver input. Therefore, this diplexer will provide adequate isolation between the transmitter and receiver without degrading receiver performance.

Type of Input Required. The system will require several input signals. These are:

- Data (input form of video) used to modulate the RF signal
- Audio for the voice communication
- Interlace signals from the digital unit
- Selection of operating frequencies.

Type of Outputs. There are five outputs. These are:

- Data in the form of video - one for Autorep and one for weather/company
- An open voice channel receiver for emergency (i.e., the guard channel) in the form of an audio signal
- Received voice as an audio signal
- The NAVSTAR navigation signal in the form of RF that has been translated down in frequency from the received signal.

2.3.3.2 Frequency Synthesizer

The frequency synthesizer generates all the required frequencies for both receiver and transmitter.

Crystal Oscillator. The basic frequency reference for the frequency synthesizer is a crystal oscillator. Today, with temperature ovens using proportional control, stabilities of parts in 10^{-9} can be obtained. With temperature compensated crystal oscillator TCXO circuits, over a temperature range of -10 to +60°C, stability in the order of one part in 10^{-6} are obtained. There should be an improvement in the stability of a TCSO over the next few years to where these devices should give stabilities of parts in 10^{-7} or better. The advantages of TCXO are less cost, smaller size, less weight, and less power consumption.

Divider. To utilize the output from the crystal oscillator, it will be necessary to divide the output frequency by a factor which will be used as the reference frequency in the individual frequency synthesizer units. The output frequency from the divider will be equal to the smallest frequency increment required. That is, if the difference between operating channels is some whole number of megahertz plus 100 kHz, then the output from the divider would be 100 kHz.

Multiplier. The output frequencies from the individual frequency synthesizer are too low for direct application to the transmitter. These signals are mixed with the output of the multiplier to produce the required transmitting frequencies. The crystal oscillator frequency is multiplied higher to obtain two different frequencies for use by the synthesizer mixers. There is one other output from this multiplier, f_{M1}, which is used by the receiver for its first mixer. This multiplied output is used to mix with the incoming RF signal of 1540 to 1560 MHz to provide the first IF.

Frequency Source. Individual frequency sources used to produce the selected operating frequencies for the voice receiver and transmitter, the weather/company receiver and transmitter, etc. These frequency sources consist of a voltage control oscillator (VCO), a digital divider, a phase detector, and a loop filter.

A remote selector can be used to select the setting for the digital divider. Changing the diving rate of this digital divider changes the VCO output frequency. The divided crystal reference frequency establishes the smaller VCO frequency change. Thus the synthesizer can be set to any frequency within its range in multiples of the crystal divider output. Some logic will be required in the remote selector to select the different receiving and transmitting frequency positioning for their respective channel operation.

Dual Mixer. The output of the individual frequency synthesizer used in the voice, weather/company, and Autorep transmitting frequency generating chain and the two outputs from the crystal multiplier circuit are applied to the dual mixer to produce the required transmitting frequencies. The dual mixer consists of two cascaded mixing stages.

The individual synthesizer output is first up-converted by a factor of about three in the first mixer. The output from this mixer is translated up in frequency by again about three in the second mixer. This method of frequency translation is used to facilitate filtering of the undesired signals. Its advantage is lower cost.

A provision to gate the dual mixer output is also provided. The output of the mixer is gated ON only when information is to be transmitted.

2.3.3.3 Receiver

The receiver consists of a series of amplifiers and mixers which process the incoming signals at 1540 to 1560 MHz into selected bands of desired information.

Preamplifier. The incoming signal from the antenna is first put through a diplexer before application to the preamplifier. Not only will there be signals containing the desired information but there will be other signals together with energy from the user's own transmitter appearing at the preamplifier input port.

Intermodulation and cross modulation distortion must be minimized to prevent interference with the desired incoming information. Also, this preamplifier must have a low noise figure to ensure proper reception. Production transistorized preamplifiers of today have noise figures in the order of 3.5 to 5 db. The expected noise figure for preamplifiers in this required frequency range by the year 1972 will be 3 db or better.

First Mixer. The preamplifier is primarily used to increase the input signal level. This increase of signal level is necessary to ensure that no degradation of the system noise figure takes place during the required additional signal processing. The first signal processing is a translation of the input frequency from L-band to the VHF band. The first mixer accepts this L-band signal and mixes it with the multiplied crystal frequency, thereby obtaining a translated frequency in the VHF band.

First IF. This VHF signal is then further amplified in the first IF before further processing of the signal is to take place.

Second Mixer. The second mixer is used to select the particular signal which is then further processed in the individual receivers. There are four of these mixers for the four receivers, plus an output to the NAVSTAR receiver. The selection of a given signal is determined by the mixing frequency generated in the individual frequency synthesizer.

Receivers. All of the four receivers are of the same design. The input signal to the receiver is further increased in power in a band restricted amplifier that permits only the desired signal to continue through the system. When the proper level of the signal is achieved then detection can take place and an audio or video output is produced. A small amount of preamplification of video or audio signals is accomplished in the receiver before outputting the information to the appropriate device for either further amplification or continuing processing as required.

2.3.3.4 Transmitter

The transmitter includes hybrid combiners, modulators, limiters, low-level amplifiers, drivers, and a final power amplifier capable of 300 watts of RF output power. This transmitter is all solid state design. Although no solid state amplifier of this power (i.e., 300 watts) is available today the technology exists today to produce such a transmitter.

When more than one signal is applied to an RF power amplifier then the transmitter capability must be rated in peak power terms. This means that for multiple signals the average power of each signal must be reduced or the transmitter peak power must be increased to ensure that the peak power limitation of the amplifier is not violated. In particular, when two signals are to pass through the amplifier, then the average power of each signal must be 6 db below the peak power capabilities of the amplifier. Thus, if this system must be capable of transmitting both voice and data simultaneously where each signal average power is 300 watts, then the transmitter peak power would have to be rated at 1200 watts. Alternately, two 300-watt transmitters may be used but their radiation must not affect each other. One way that this type of operation can be permitted would be to have two antennas, one for each transmitter.

Hybrid Combiners. Combining the two data channels, these are the Autorep together with weather/company, and then the voice channel is accomplished with a distributed constant hybrid. These hybrids permit the combination of signals of different phase, amplitude, and/or frequency while maintaining a constant impedance at all the hybrid ports, together with at least 20 db of isolation between the input ports.

Modulators. There are two phase modulators used in the transmitter: one for the voice channel, and a second for the data channel. The data for the Autorep and weather/company are interleaved to form one modulating signal. The incoming data transmitting frequencies are switched between two preselected frequencies in the frequency synthesizer so that only one frequency, either for the Autorep or another for the weather/company transmitter, is applied to the data modulator at one time in its proper sequence.

The voice channel only operates during an actual voice transmission. When a voice transmission is to take place the activation of the press-to-talk switch by the operator permits the voice transmitting frequency to be supplied to the modulator from the frequency synthesizer.

Limiter. The limiter circuit can be either a true limiter circuit or a power-sensing feedback circuit. In either case the operation would be the same in that it would ensure that the two channels (i.e., voice and data) are of equal output levels which would produce the maximum transmitter power within the capability of the RF power amplifier.

2.3.3.5 RF Power Amplifier

The voice and data channels are hybrid combined and applied to the input of the RF power amplifier. If this amplifier will have to handle multisignals, the intermodulation (IM) distortion of the amplifier must be better than 25 db. IM is present to a lesser or greater degree in all amplifiers. When more than one frequency is applied to an amplifier then spurious signals are generated, such as

$$NW_1 \pm MW_2$$

where the sum of N + M is called the "order" of the IM distortion generated. The odd order products are spaced from the fundamental signal by a spacing equal to the separation of the original applied signals. Thus, there can be spurious signals, resulting from the combining of the voice and data transmitting frequencies, such that these undesired signals would fall on another voice or data channel. To reduce this effect the RF power amplifier distortion must be as low as possible.

The interleaved signals (i.e., voice and data) are first amplified in low level stages to produce appropriate driving signal energy. These amplifiers can be made very linear, hence they have very low distorted outputs. IM products of 40 db or better for the third order are achieved today.

This signal is further amplified in a first and second driver before application to the final amplifier. When higher powers are involved, such as in the driver and power amplifier, a tradeoff exists between efficiency together with cost, and the produced IM distortion. Any IM generated by these drives will add to the IM of the final power amplifier.

Since the transmitter uses phase modulation and adopting the interleaved operation so that only one signal is transmitted at a time, then it would be possible to relieve the IM specification to about 15 db. This would permit a cost savings and an improvement in the overall transmitter efficiency.

The final power amplifier consists of a large number of combined power modules. The use of many combined subunits enhances the operational reliability of the amplifier due to the fact that a malfunction of one module due to the output shorting or opening only degrades the output power by approximately one-and-one-half times the nominal module power.

Overall efficiency for the RF power amplifier in the order of 35 to 40 percent should be obtained with an IM distortion of at least 25 db. Better devices by 1972 should permit improvement of both the amplifier overall efficiency and IM product. This efficiency could be increased 40 to 50 percent by allowing IM distortion in the order of 15 db.

2.3.4 Digital Unit

2.3.4.1 General

The user digital hardware is configured to handle three types of messages as follows:

- The Autorep message — raw position data transmitted from aircraft to ground on a roll call basis.
- The weather message — from ground to aircraft at ground time convenience, and from aircraft to ground on a ground permission basis.
- The company message — from ground to aircraft at ground time convenience, and from aircraft to ground on a ground permission basis.

The Autorep message serves as a carrier for certain short coded messages which ride piggyback. The Autorep messages are transmitted frequently enough so that this does not present significant delay problems in the use of those messages and the technique results in a higher utilization efficiency in the communication channels.

The Autorep message format and content is illustrated in Figure 39. With the exception of the altimeter data and the piggyback word, all of the data in the Autorep message are extracted automatically from the output of the NAVSTAR preprocessor. An overall block diagram of the digital unit is shown in Figure 40.

2.3.4.2 Autorep Sequence

The Autorep sequence is initiated by the command from the air traffic control ground station. The command message is received, demodulated, and examined for correct address (each airplane has a unique address) in the Autorep address decoder. The main body of the message is a code which is interpreted by the decoder as an order to transmit the Autorep message. The data in the Autorep buffer is sent to

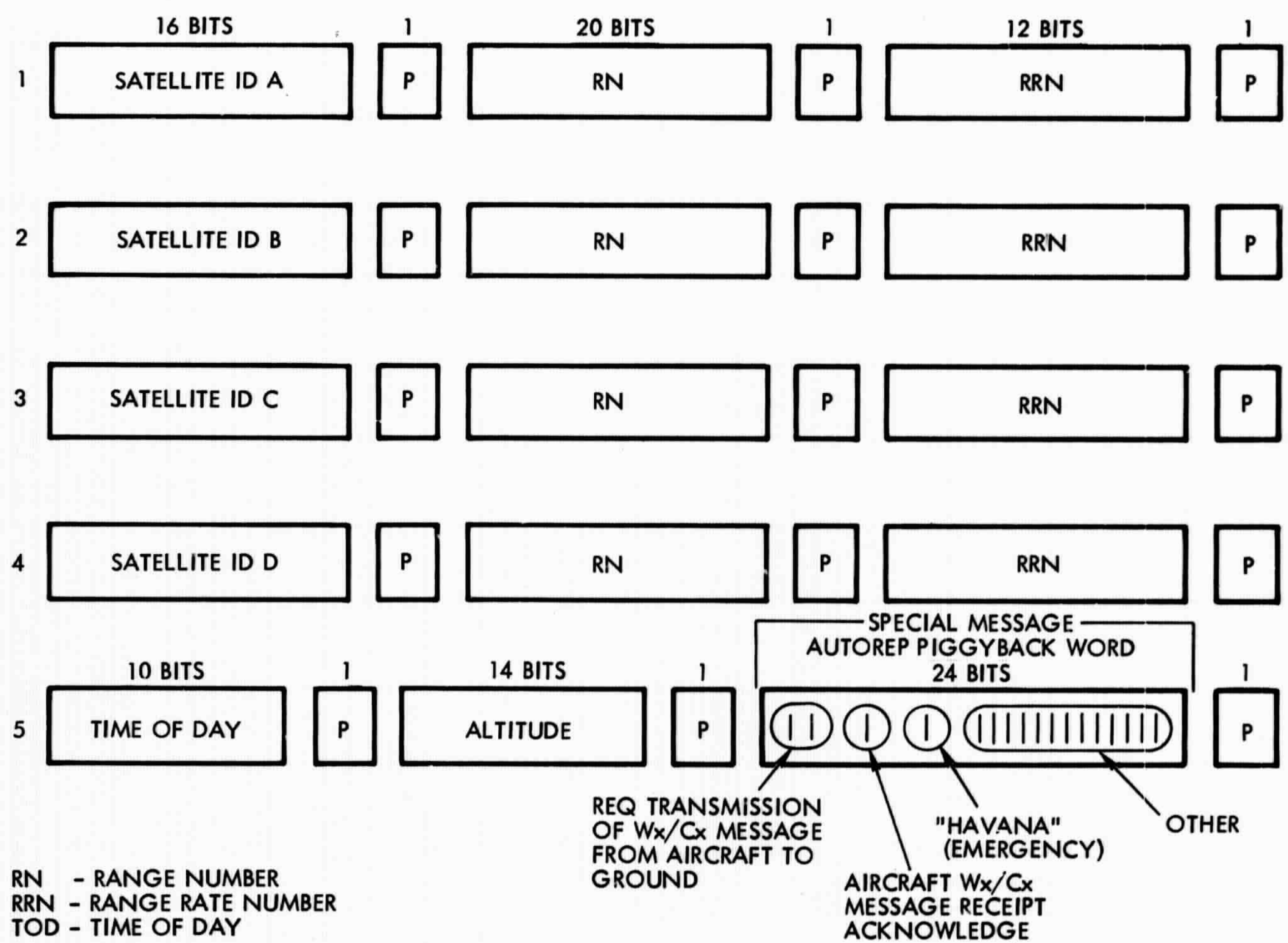


Figure 39. Autorep Message Format

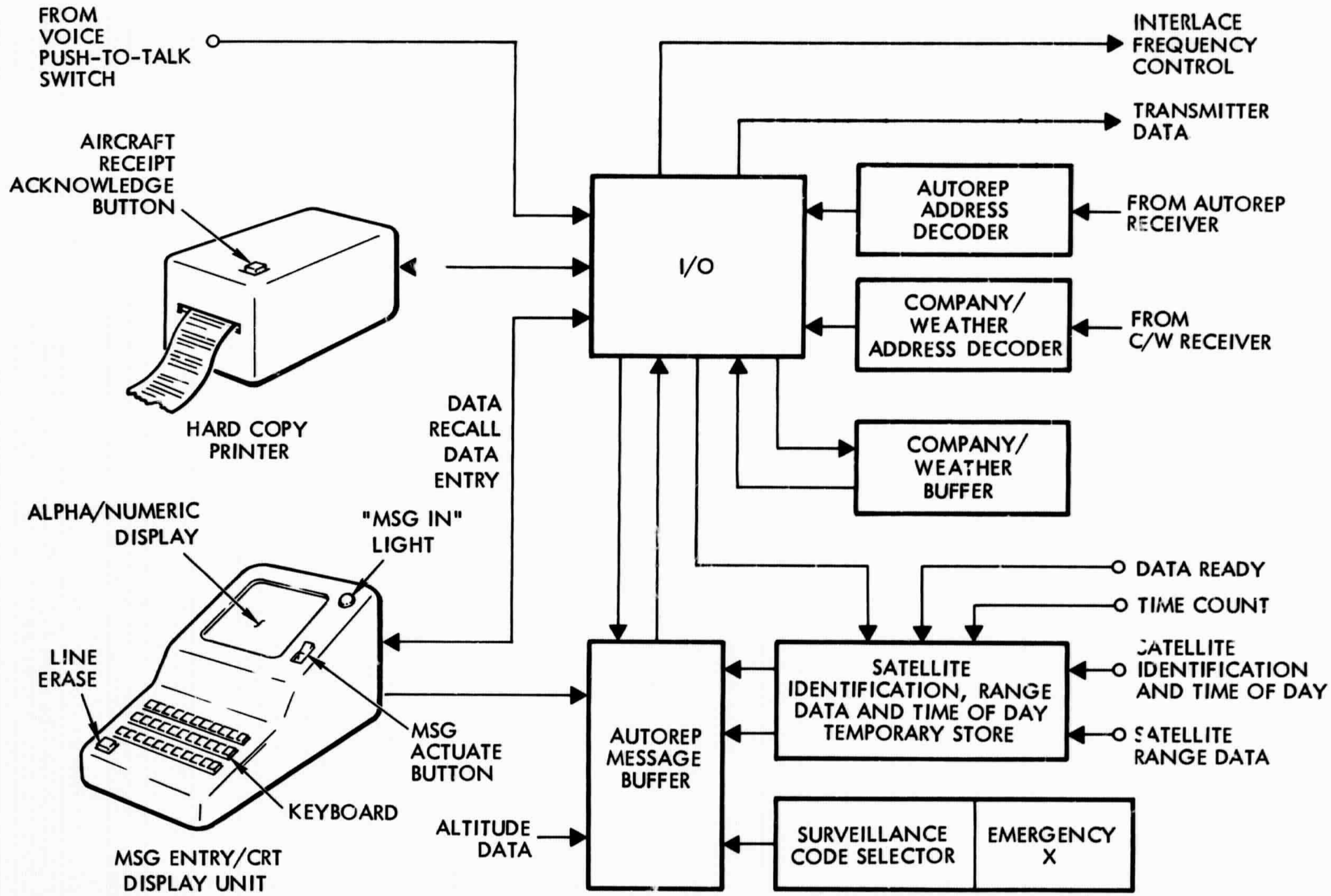


Figure 40. **Digital Unit Block Diagram**

the message formatter and output control which combines the data from the Autorep buffer with the address code of the airplane and sends the message to the ground via the modulator and transmitter. Data from the NAVSTAR processor is selected and stored in the Autorep buffer. The Autorep buffer is updated repetitively but is locked out during message transmission to avoid partial overwriting of data during transmission. The Autorep message itself is, in effect, an acknowledgement of the ground station message which orders transmission of the Autorep message. Therefore, no further acknowledgements are required, and the Autorep buffer terminates its cycle upon completion of the transmission, and updating of data commences again.

2.3.4.3 Company/Weather Message Sequence From Aircraft To Ground

These messages are assembled on the alphanumeric display and keyboard machine. The message is constructed a line at a time, and as the lines are constructed they appear on the CRT display — the operator, pilot, or stewardess does that with the keyboard. As the lines are formed they are stored in the message buffer. Mistakes are corrected by means of the line erase button which permits the last line of the message to be retyped. Having constructed a message which consists of a maximum number of 10 lines, the complete message is stored in the message buffer. It is now possible to take one action which will result in transmitting this message to the ground without any further action required of the operator. This process is initiated with the message actuate button. The "Message In" indicator is illuminated and latched. The "Message Actuate" button causes a tag or code bit to be inserted into the next Autorep message which will be sent to the ground. The ground station will then recognize this special tag as a request for permission to transmit a message. The tag also identifies the message as weather or company. The ground station resolves the message priority problem and calls for transmission of company and weather messages in proper order. When it is permissible to transmit the message, the ground station activates the process by itself without any further requirements on aircraft personnel. However, if a voice transmission is activated prior to the receipt of permission to transmit a company/weather message, then the tagged or coded bit in the Autorep message (that requested C/W transmission) is removed until the voice transmission is completed. The priorities are reversed when a company or weather message is in process of transmission. This will mean a delay of not more than one second in the voice transmission. It should be pointed out that the aircraft-to-ground company/weather messages are restricted in length due to the buffer word size.

The company/weather message sequence of operations is as follows. The ground station will send a message to the aircraft which will be received and demodulated and address decoded as usual. This message will consist of a special code which is a transmit order. The transmit order will be decoded in the decoder unit and sent to the message formatter and output control. This unit will cause the message to be read out of the message buffer and transmitted to the ground. The message is read out of the message buffer through the IO channel to the message formatter and output control where it is combined with the address code which identifies it as coming from this particular airplane. The combined message is then transmitted to the ground. The ground station, having received the message that originated in the airplane, now acknowledges receipt of that message. This is done by transmitting a very short acknowledge message which is demodulated, address checked, and decoded in the aircraft. Receipt of the ground receipt acknowledge now comes back to the message entry CRT display device and extinguishes the "Message In" light which completes the cycle. The pilot knows the message has been received by the ground. This was accomplished by pushing only one button — the message actuate button. It might take as long as several minutes before the light goes out in heavy traffic, but considering the priority of the messages, we feel that is acceptable.

2.3.4.4 <u>Company/Weather Message Sequence from Ground to Aircraft</u>

These messages are received, demodulated, address decoded, and routed into the message buffer. They are stored temporarily in the message buffer and then printed on the hard copy printer. The message which is printed on the hard copy printer is going to have some kind of an end code, like the newspaper "30," and upon receipt of this the operator will initiate an aircraft receipt acknowledge. He does this simply by pushing a button on top of the hard copy printer. This generates a very short message code which is transmitted to the ground piggyback via the autorep message. A bit is placed in the special message section of the Autorep message shown in the lower right hand corner of Figure 39.

Use of the Autorep special message word eliminates several special acknowledge messages going back and forth between the airplane and the ground. This type of operation eliminates message overhead, namely, the start-up and turn-off of conversations. If an aircraft were sent two messages, without an intervening acknowledge, we have an ambiguity, i.e., the people on the ground are not completely sure whether the aircraft received one message or the other, or both. But they know that he received at least one. They might assume at least that his equipment is in order. Tradeoffs might indicate use of more bits if it is desired to actually show how many messages had been acknowledged.

2.3.4.5 Hard Copy

A hard copy of every message that leaves the airplane or comes into the airplane is made for record. The message buffer resolves all of the data rate incompatibilities between the printer, keyboard, and message handling equipment. A study should be made to consider operational feasibility and cost advantages of combining the hard copy printer with the keyboard and eliminating the CRT display.

2.3.4.6 Autorep Message Format

The Autorep message consists of 5 lines of information as shown in Figure 39. The first four lines consist of data associated with the satellite. The data associated with each satellite are:

- Satellite identification: 16 bits
- Range: 20 bits
- Range rate: 12 bits

It should be noted that no provision has been proposed in the present equipment to implement the range rate data, but it will be included in this message analysis since it is a very likely (and very small) user hardware growth item; thus the first four lines of information (from each of the four satellites) consist of satellite identification, range number, and range rate number. Line number 5 contains: the time of day selected from the latest satellite—10 bits; altimeter data—14 bits; and a 24-bit piggyback word.

Each time a new set of data is completed giving the satellite identification, range number, and time of day, the three pieces of information of lines 2, 3, and 4 of the Autorep message are transferred up one line higher (i.e., line 2 is moved to line 1, line 3 to line 2, and line 4 to line 3). The information that was in line 1 is no longer used and the new data from the NAVSTAR processor in temporary storage is moved into line 4 and the 10-bit time of day registrar of line 5.

Each line is shown with 3 parity check bits. Use of the piggyback word warrants further study. It is conceivable that codes should be used which would be the equivalent of the selective identification feature codes on today's radar, so that various aircraft might belong to groups and subgroups, i.e., people flying on instrument flight rules, on top of the clouds, in the clouds, in a climb, in a descent, etc. Changing from one subgroup to another is of operational significance in an air traffic control sense. Codes of this type in a special message or special word would be useful. The codes which we can define now are:

- Request for permission to transmit company/weather message: 2 bits
- Aircraft receipt acknowledge: 1 bit
- Emergency: 1 bit.

2.3.4.7 Surveillance Code Select Unit

This unit selects codes which may be developed in future studies for insertion into the piggyback word. It would be manually operated. In addition to this surveillance code selector a switch would be provided that would place an emergency bit into the autorep piggyback word.

2.3.5 NAVSTAR Unit

2.3.5.1 General

The NAVSTAR system receives an RF signal that has been translated lower in frequency from the L-band spectrum transmitted by several satellites. The satellites broadcast their NAVSTAR information on a time-shared basis so that only one satellite transmission is received at any one time by the aircraft.

The total transmission from each satellite takes about one-and-one-half seconds. This transmission consists of a BINOR code and a data format. The BINOR code consists of a series of 13 coherent square waves which are harmonically related by multiples of two. These signals are added under a given set of rules which result in a 2^{13} bit long code. This PINOR code is followed by the data format of the satellite. This data format, shown in Figure 41, identifies the satellite, gives the time of the transmission (i.e., time of day), the satellite position (i.e., ephemeris data), and a correction of the time of day.

2.3.5.2 NAVSTAR Receiver

The NAVSTAR receiver has a 4 MHz bandwidth and uses a phase-locked loop to generate a reference carrier used in the signal detection circuitry. This NAVSTAR receiver is shown in block diagram form in Figure 42. The receiver has over a 40 db dynamic range with less than ±5 nsec of group delay variation. In order to meet this fundamental requirement, nonlimiting broadband IF circuitry with coherent AGC is used exclusively in the information channel.

A secondary requirement is the incorporation of a loop and quadrature bandwidth switching capability; this is necessitated by a need for rapid carrier acquisition prior to data transmittal and by the uncertainties associated with the receiver's oscillators and by Doppler. The acquisition bandwidth (1650 Hz) allows the receiver to acquire the minimum level carrier (-127 dbm) in 0.38 second with a reliability of 99.9 percent. After carrier acquisition and prior to data transmittal, both the loop and quadrature channels switch to a 50 Hz noise bandwidth. This gives the necessary signal-to-noise ratio (SNR) in both channels to accommodate the 9 db reduction of carrier power during data transmission. It also diminishes the loop sensitivity to the lowest frequency component (78 Hz) of the BINOR code. It is interesting to note that a side benefit of a high-gain coherent AGC in conjunction with an entirely linear loop is the complete stability of the loop bandwidth and damping factor in relation to the lowest BINOR code spectral component.

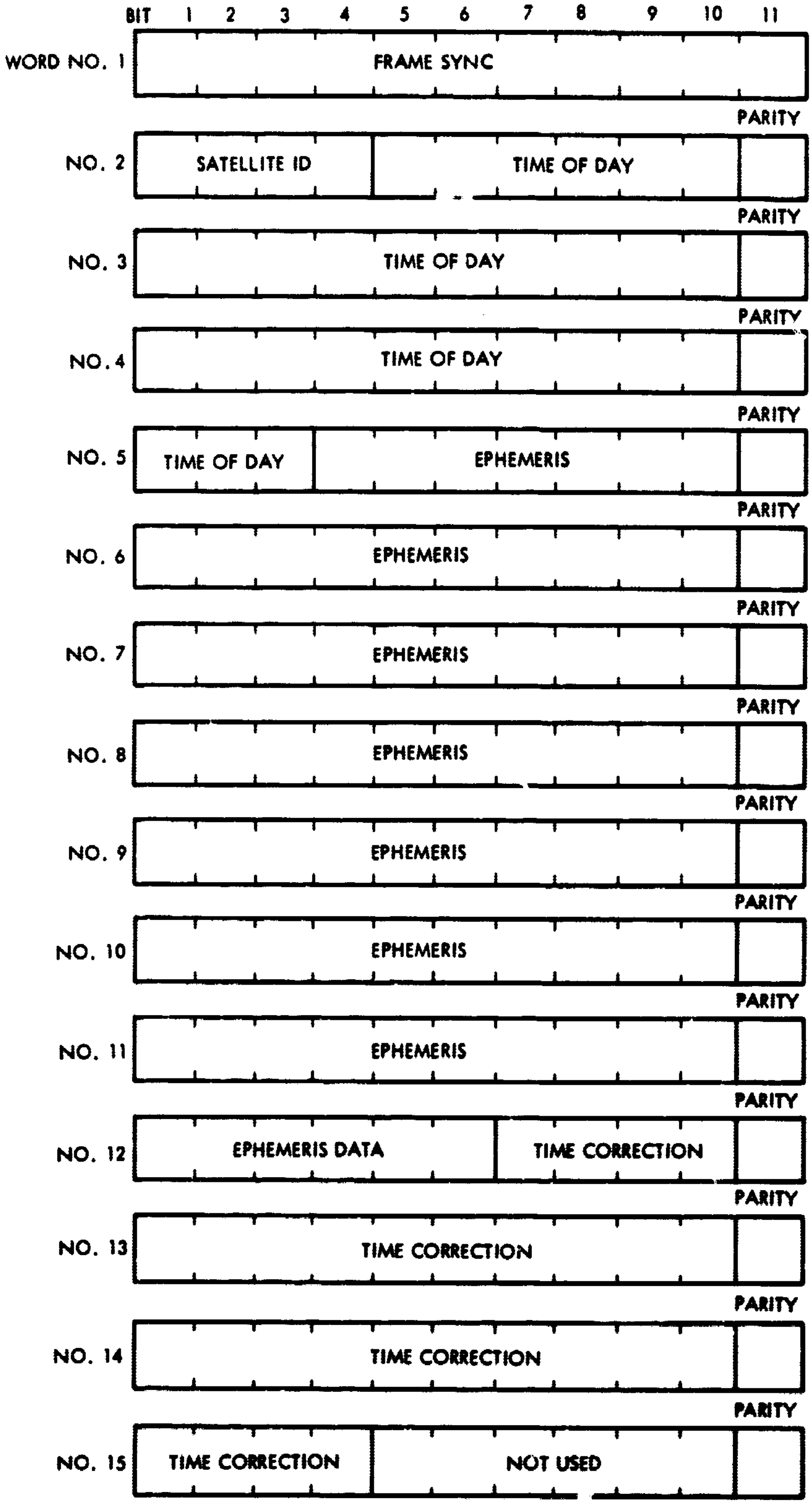


Figure 41. Data Format

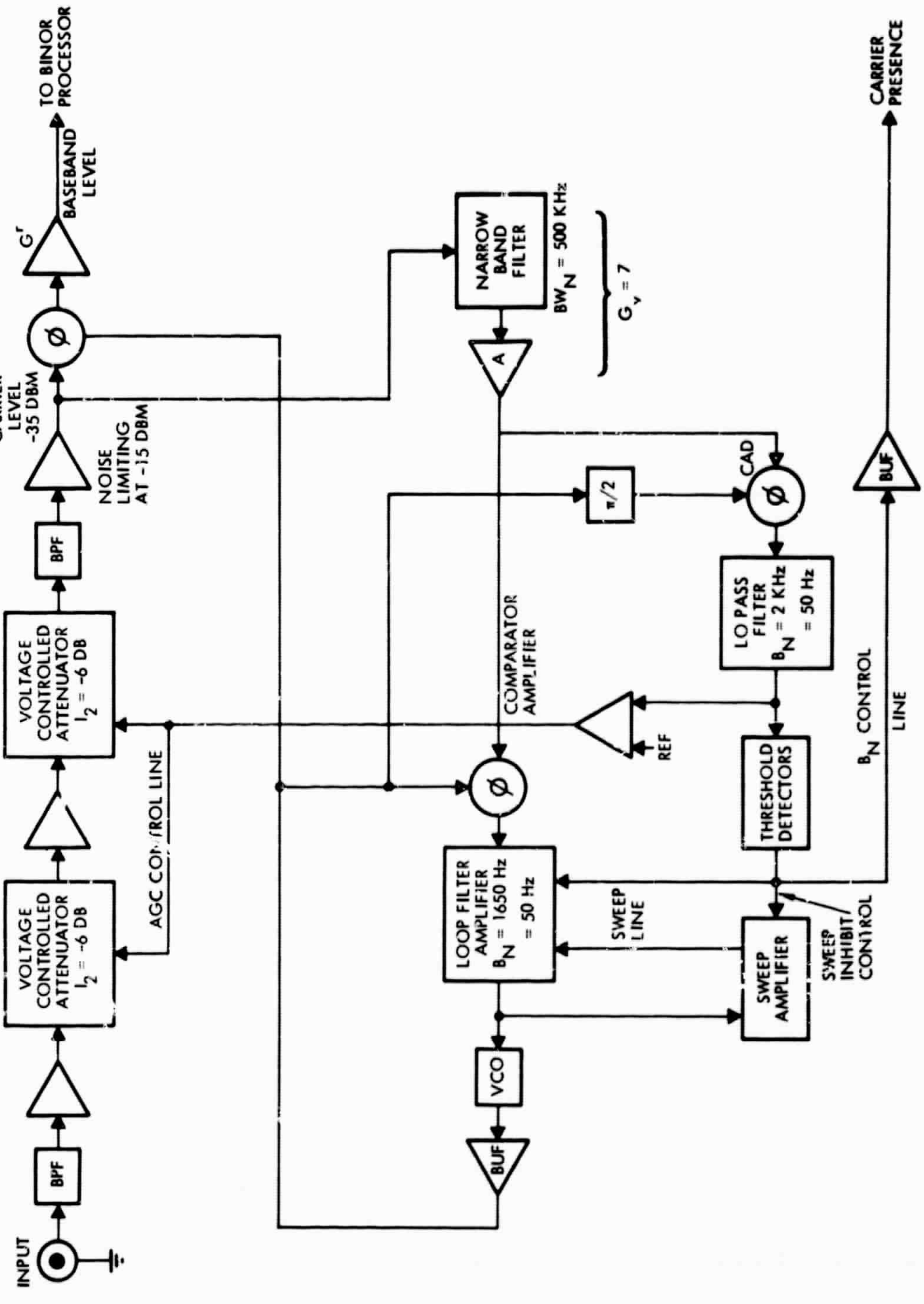


Figure 42. BINOR Receiver Block Diagram

At the completion of the code and the receiver-processor data updating interval, at which time the carrier is removed, the receiver reverts back to the carrier search mode.

Sweep and Loop Tracking Circuitry. After carrier acquisition but prior to the BINOR code transmission interval, the carrier-to-noise ratio (CNR) at the IF output may be as poor as -26 db. Moreover, this ratio will degrade to -35 db during the code transmission interval. In order to improve this CNR and to set the carrier level to a level convenient to the operation of the coherent detectors, the IF output is filtered (B_n = 500 kHz) and then amplified. This drives both detectors at signal levels sufficiently high to avoid thermally-caused offsets while not exposing them to excessive noise levels which also can cause offset difficulties.

During the carrier search mode, the sweep amplifier integrates current into and out of the loop filter amplifier causing the VCXO to sweep a sawtooth search pattern with a search period of 0.36 second and a 0.02 second retrace period.

During the search period, the coherent amplitude detector (CAD) and the comparison amplifier outputs are zero and, correspondingly, the IF gain is maximum. Both the loop and amplitude channel noise bandwidths (1650, 2000 Hz) are consistent with the requirement of 99.9 percent probability of acquisition in 0.39 second for carrier levels of -127 dbm. With carrier acquisition, the threshold detector disengages the sweep amplifier and both channels constrict to 50 Hz noise bandwidths. Moreover, the comparison amplifier compensates the IF gain until the signal level at the IF output is the required -35 dbm. The response times of the IF voltage controlled attenuators are sufficiently rapid that the modulation-caused carrier level drop of 9 db will not reengage the sweep amplifier. The actual carrier level compensation time is approximately 10 msec and is determined by the threshold filter.

IF Amplifiers. The IF amplifier consists of a cascaded series of amplifiers, bandpass filters, and voltage controlled attenuators. The voltage control attenuators permit the receiver to have over 40 db of dynamic range while maintaining the input to the phase detector at a level of -35 dbm. The IF amplifiers make up for the insertion losses of the voltage controlled attenuators and the bandpass filters and to supply the phase detector with the required power level.

2.3.5.3 NAVSTAR Processor

The NAVSTAR processor basically consists of a satellite range decoder and a satellite data decoder. The range decoder acquires and processes the BINOR code from each satellite to give range information. The data decoder operates on the data format from which it abstracts the satellite identification and time of day. Both the range data and the satellite ID and time of day are sent to the Autorep message buffer.

The following is a description of the operation of the two decoders. Much of the subsequent material was taken directly from Volume III of Reference 1, and is presented here to provide a more self-contained report in this volume.

BINOR Code Preprocessor. The BINOR range measurement technique uses a binary code 2^{13} bits long. The acquisition procedure for the code consists of acquiring a clock component with a phase-lock loop followed by 12 correlations in sequence with 12 squarewaves, each at half the frequency of the preceding wave. The code is derived from the highest frequency squarewave of 320 kHz (to give 30-feet rms range error) and the lowest frequency squarewave of 78.125 Hz (to give 2100 n mi range ambiguity). After all of the correlations have been performed, the lowest frequency squarewave will be in-phase with the transmitted code sequence. The desired range can then be secured by measuring the phase delay between the derived in-phase lowest frequency squarewave from the received signal and a reference squarewave of the same frequency generated internally. The preprocessor contains a 20 MHz reference oscillator used in measuring the phase delay. The range count is averaged over eight periods during each satellite transmission interval in order to minimize quantization errors.

The satellite data is extracted by the PCM signal conditioner and bit synchronizer providing NRZ-L data and a sync signal to the data buffer. The data demodulator receives a split-phase code at a frequency equal to the data bit rate of 625 bps. The data are received following the end of the BINOR code ranging signal.

Code Acquisition Network. A block diagram of the acquisition circuitry for the BINOR code is shown in Figure 43. The circuitry consists of a phase-lock loop for acquiring the code clock phase, a two-level loop lock indicator, and 12 correlators. Each correlator, in turn, consists of a multiplier, integrate and dump filter, and sample-and-hold circuit. The block diagram, however, portrays only a conceptual design. In the actual design, the 12 correlators will be replaced by one time-shared correlator, since only one correlation occurs at a time. The total circuitry is simplified by this approach as the added gates and control signals contain much less circuitry than the eleven correlators they replace. To simplify the explanation on the code acquisition, however, the conceptual design configuration as shown is presented here for discussion.

The phase-lock loop (Figure 43) establishes lock with the phase of the BINOR code clock (or highest squarewave frequency). The phase-lock loop threshold circuit indicates phase lock to the clock frequency and its threshold is set for 22.5 percent clock correlation. The lock-indicator signal G_o initiates the timing sequence for the code squarewave correlations. The first correlation is with the $f_{s/2}$ (160 kHz) squarewave. The squarewave and the code are multiplied together and the multiplier output is integrated for two periods of the code (2/78.125 seconds) and sampled. The polarity of the sampled voltage sets a flip-flop to one of its two states. The state of the flip-flop then determines whether the squarewave is or is not inverted, corresponding to whether the squarewave is in or out of phase with the code. The exclusive OR gate logic serves as the squarewave inverter. The T timing pulses, T_1 through T_{12}, are the integration times in sequence for the integrate and dump filter and are two code periods long. The leading edge of a T pulse serves as the sample signal for the preceding correlation. For example, the leading edge of T_2 samples the integration during T_1.

After the first correlation, the correlations continue until the start of T_{13}, at which time the 78.125 Hz squarewave will be in phase with the code. The code phase, as represented by the 78.125 Hz squarewave phase, is now acquired. After reception of the code, which is on for a sufficient time to acquire the code phase, the 320 kHz clock is received

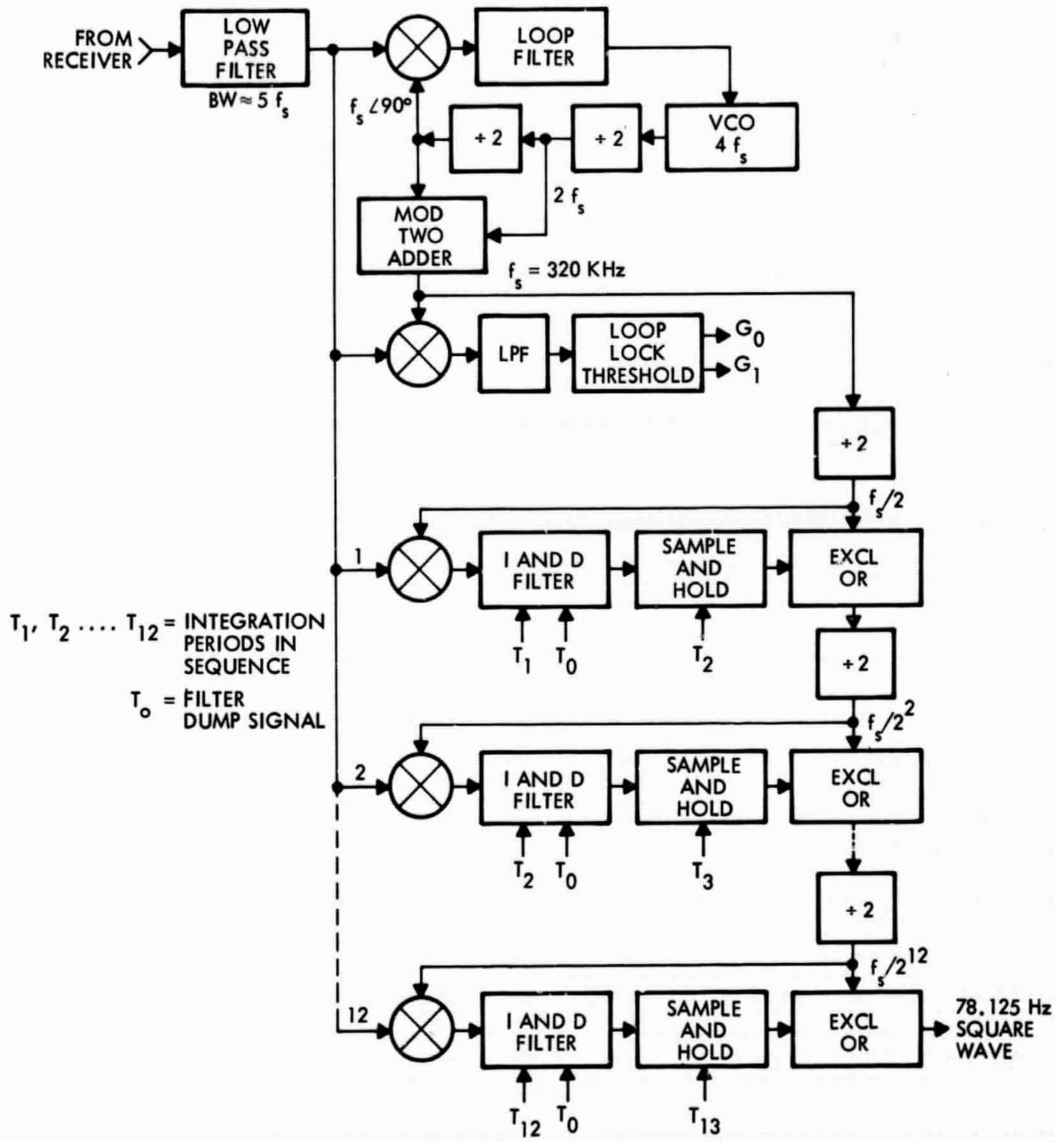


Figure 43. BINOR Code Acquisition – Conceptual Block Diagram

alone without any signal modulation. The clock loop correlation now increases to 100 percent. This event is indicated by loop-lock threshold G_1 and is used to initiate the range measurement. The T pulse T_o occurs eight T times after the T_{13} event and is the dump signal for the filters and the reset signal for the range measurement logic. The event T_o readies the preprocessor for the next range measurement from another satellite.

Range Measurement Unit. The range measurement unit shown in Figure 44 contains the following circuits:

1) 20.48 MHz crystal oscillator — reference
2) Divider — 18 stages
3) Time interval counter — 20 stages
4) Range sample counter — 3 stages
5) Code period or T pulses counter — 6 stages

The lock indicator signal G_o from the code acquisition clock loop sets the code period counter to T_o, initiating the sequence of T pulses representing the correlation time sequence of the code-acquisition

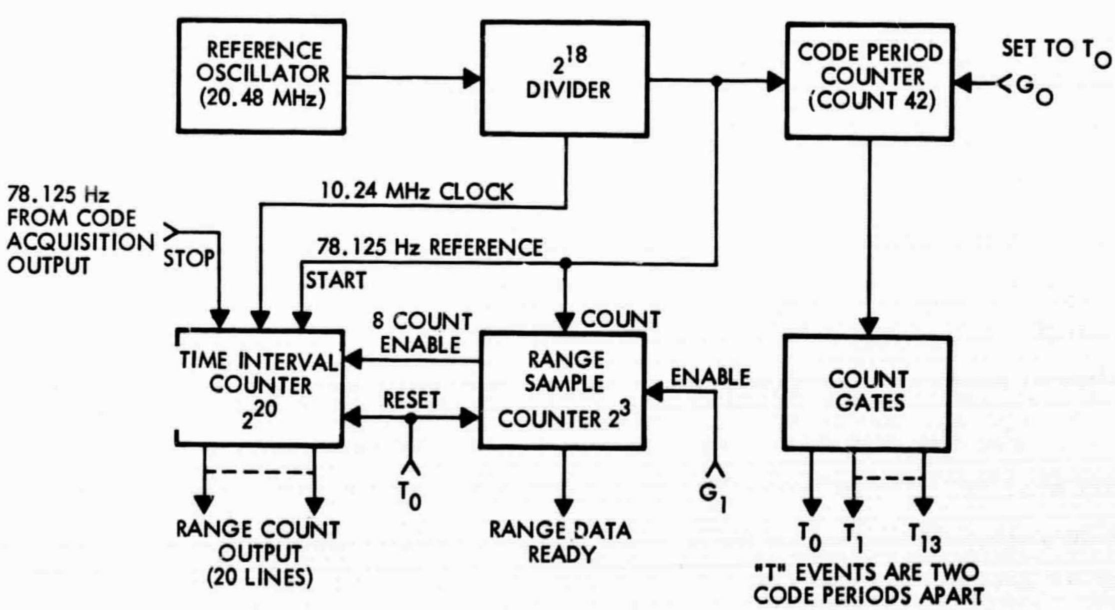


Figure 44. BINOR Code Range Measurement Unit

circuitry. Some time after T_{13}, the lock-indicator signal G_1 occurs and enables the range-sample counter. The range-sample counter counts eight samples of the range measurement by the time-interval counter, disables this counter, and then generates a range-count ready pulse for the computer interface circuitry.

The time-interval counter is driven by a 10.24 MHz symmetrical clock derived from the 20.48 MHz crystal reference oscillator. The interval counter is started and stopped by the leading edges of the reference and code acquired 78.125 Hz signals. The reference squarewave, which is derived from the 20.48 MHz oscillator by a 2^{18} divider, is used also as the clock for generating the T pulses.

One range measurement is represented by the time interval between the start and stop signals of the time-interval counter. Since a 10.24 MHz clock is used to count the timer interval, a maximum timing error of 50 nsec and an rms error of $50/\sqrt{3}$ or 29 nsec results. This would be undesirable for the high-accuracy user; consequently, eight range measurements are made to reduce the rms error by $\sqrt{8}$. The time-interval counter is allowed to accumulate the count over the eight time intervals so that the range-count output must be divided by eight to obtain the correct range count. For one measurement, the capacity of the time-interval counter must be 2^{17} counts of the 10.24 MHz clock. For accumulating eight measurements, the capacity must be increased by eight or to 2^{20} counts. The computer can perform the divide-by-eight function so that the range-count word consists of 20 bits out of the time-interval counter.

Some time after the range count is completed (eight T events after T_{13}), the T_o event occurs and resets the time interval and the range-sample counters to zero so that a new range count can be made on initiation of another G_o signal.

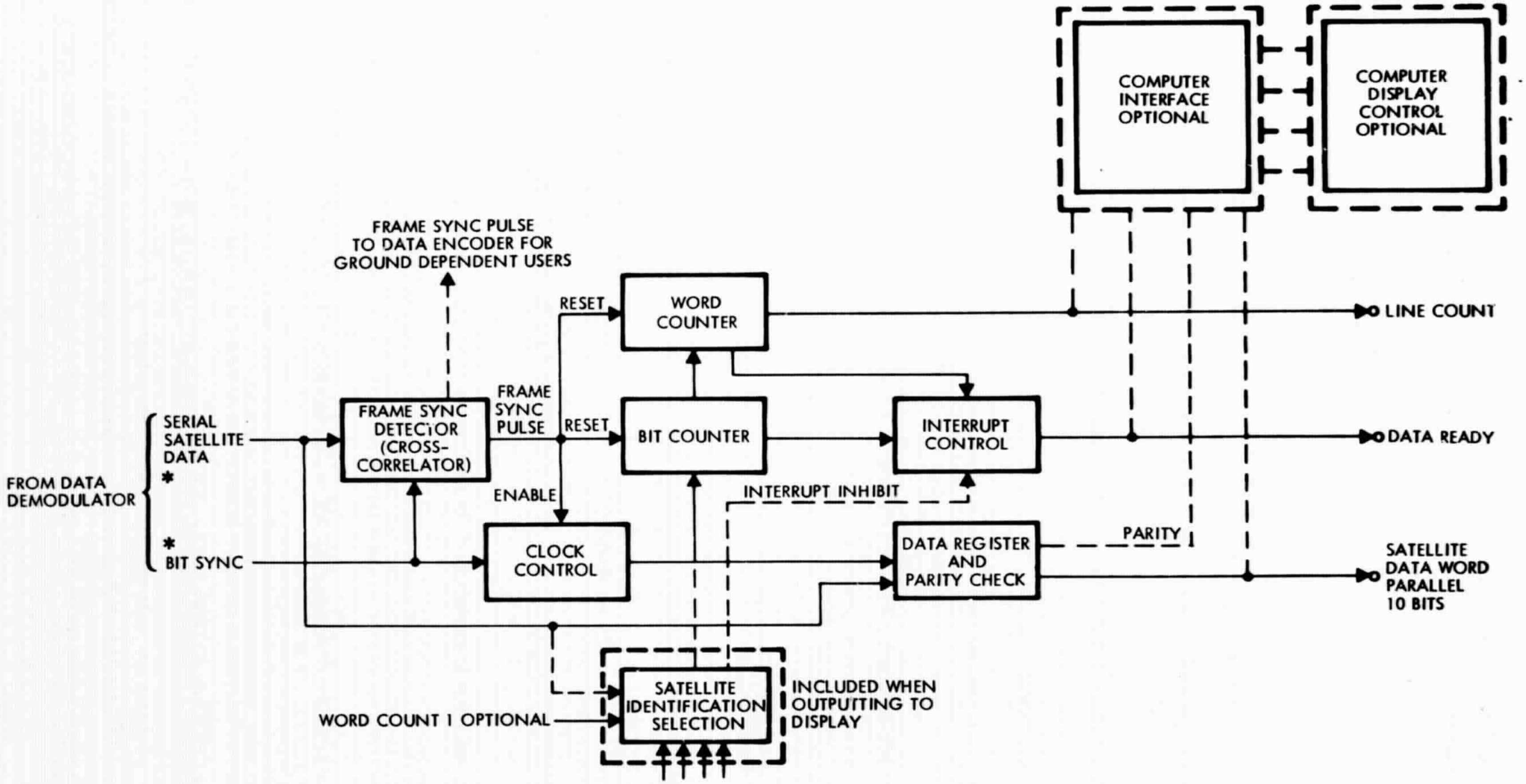
For the low cost, low accuracy user the eight range counts are not necessary. Therefore, the range-sample counter and some logic can be eliminated from the circuitry and the total range-count word would remain at 17 bits.

Data Signal Decoder. The data signal decoder first operates on the signal by conditioning the noisy input and providing bit synchronism and then transfers the data word to a buffer which, in turn, transfers the satellite identification and time of day to the autorep message buffer (Figure 45).

Data Signal Conditioner and Bit Synchronizer. The data signal conditioner and bit synchronizer extract NRZ-L data and a clock signal from a split-phase code input signal under conditions of poor signal-to-noise ratio. A code can be considered as the modulo 2 sum of the NRZ-L data and the clock squarewave. The design of the data signal conditioner and bit synchronizer may be based on the same principles as those employed by TRW on the Pioneer telemetry demodulator. This demodulator design can be simplified while still providing the basic function of signal conditioner in poor signal-to-noise ratio and bit synchronizer. The exact details of this design are not covered at this time but would be developed during the R&D phase of the users equipment.

Data Buffer. A block diagram of the preprocessor data buffer is shown in Figure 43. The data buffer receives inputs from the data demodulator on two lines. One line carries the NRZ data, and the other carries the bit-synchronization signal. At the beginning of each frame, data are continually fed to the frame-sync detector, which contains an 11-bit register and cross-correlation detection logic. Each bit-sync pulse shifts the contents of the register while the incoming data bit is entered. During each bit time, the contents of the register are added, modulo 2, with a hardwired Barker synchronization code. The output is fed to a summing network, and a threshold level test is performed. When correlation is sensed, a frame-sync pulse is generated to reset the bit counter, word counter, and parity check logic in preparation for processing the succeeding data.

Following the establishment of frame synchronization, the data are shifted into the data register while each bit-sync pulse increments the bit counter. Parity is computed by toggling a flip-flop each time a "one" bit is sensed. When the bit counter reaches the eleventh state, the computed parity is compared with the received parity bit. If a positive check occurs,



* BIT RATE IS 625 BPS FOR CW SYSTEMS AND 156.25 BPS FOR PULSE COMPRESSION SYSTEM

Figure 45. Processor Data Buffer Block Diagram

an interrupt signal is fed to the Autorep temporary store as a data ready signal. The contents of the word counter are provided to aid interpretation of the data. Following the interrupt signal, the bit counter is reset, and the word counter is incremented by one. This procedure is repeated for each word of the data frame. If the parity check on any word indicates an error, the data buffer is reset and a parity error signal is fed to the user equipment. Completion of the data frame preprocessing is indicated when the interrupt signal occurs with the word counter in the fifteenth state. The data buffer is cleared and is prepared for sync detection of the data frame from the next satellite in sequence.

The output from the processor data buffer can be made available for those users who take the option to add some form of computer or calculator so that they may obtain position fix onboard the aircraft utilizing this NAVSTAR information. For additional information about the users optional equipment the reader is referred to Volume III of Reference 1.

2.3.6 Alternate User Hardware Configurations

2.3.6.1 Large Aircraft

The system described in the previous paragraphs requires a 4 db antenna to achieve a satisfactory power budget for communications. The navigation power budget indicates that 0 db antenna is adequate. A 0 db antenna may be constructed so it has hemispherical coverage and requires no switching. However, since it is inherently impossible to design a single 4 db hemispherical antenna, the questions arise "How could the communications subsystem be mechanized to use 0 db antennas?" and "Should the system be so mechanized?"

To answer the first question, there are several alternatives to the use of multiple antennas and switching in the user hardware. First is the possibility of increasing the satellite antenna gain and/or transmitter power which reduces the user antenna gain requirement. The obvious disadvantages are either more satellite dc power, reduced numbers of voice and data channels per satellite, reduced area coverage, or a combination of these. The latter approach is probably desirable. A second alternative which can be used to reduce the average power required from the satellite is to configure the downlink between the satellite and the ground station different from the downlink between satellite and user and take advantage of the high antenna gain available at the ground station. The disadvantage with different ground and user downlinks is that all aircraft could not receive messages addressed to the ground station. Thus aircrews could not simply listen to a channel and determine whether or not it was busy. A "busy signal" could be generated at the satellite. This approach clearly merits serious consideration in the preliminary design phase. A third alternative is to reduce the performance margins in the communications link budget. After forthcoming ATS experiments, it should be possible to define exact power requirements for the communications link. These experiments may show that the present link margin in the communications power budget is higher than required. This would be an ideal solution to the problem.

Although all of the above alternatives are worthy of further consideration early in the preliminary design phase, the answer to the second question was in each instance in the negative except for the "Busy Signal" concept, which was not analyzed in detail.

2.3.6.2 Small Aircraft

A small aircraft is defined for this paragraph not on the basis of size - but on the basis of cost. A small aircraft will be defined simply as one which costs substantially less than the business jets and the larger more complex business aircraft. The more expensive aircraft, although normally classified with "general aviation" are almost invariably well equipped and would use a system similar to the system described earlier in this section.

Although small aircraft rarely fly the North Atlantic, if the NTC Satellites are orbited for coverage of North America, many small aircraft can be expected to take advantage of the position determination capability of the system if the user hardware costs can be made competitive. Although the need for communication through a satellite over land areas has not been quantitatively established, it will be assumed herein that small aircraft communications will be limited to direct air-ground-air contacts. With regard to position determination, however, the possibility of using the Autorep surveillance system using direct lire of sight data links between small aircraft and ground stations appears promising. A system with essentially the same capabilities as the baseline system but with a different antenna configuration and a different power budget could be used for transmission of Autorep data between small aircraft and ground stations. The navigation function would be performed in a manner similar to the baseline system, except that price becomes virtually an overriding factor.

Present day small aircraft used in commercial service or as business aircraft are usually equipped with dual navcom radios. A good navcom radio with VOR indicator costs in the vicinity of $2000 installed. Navcom radios are available as low as $800 but they are not suitable for any but the simplest aircraft. If the cost of the navigation radio is arbitrarily evenly divided between communications and navigation, present small aircraft would have invested somewhere in the vicinity of $2000 in navigation equipment. Most aircraft in the $100,000 category also carry Distance Measuring Equipment and ADF resulting in $4000 to $5000 total investment. Initially the cost of navigation satellite equipment will be above $5000, but the possibility exists of producing navigation satellite system user equipment for under this price in the not too distant future.

The implications of a North American navigation satellite system for small aircraft are tremendous if user equipment could be built for a price competitive with VOR/DME navigation receivers. It would be possible to replace the present VOR airway system. It would be useful to compare the cost of erecting and maintaining a North American navigation satellite system to the cost of maintaining the VOR airways network. It is believed that such a study would show the satellite system to be less expensive. It is suggested that the study be made for the time period beginning 1980 when it is anticipated that a working North Atlantic navigation system would be in use, and the technology required to build low cost navigation satellite receivers would be available.

Finally, the use of the position determination data by small aircraft as "minor partners" in a satellite collision avoidance system scheme appears very attractive. All of the above ideas can be mechanized with 0 db aircraft antennas.

2.3.6.3 Large Ships

Large ships of greater than 100 tons displacement may use navigation equipment which is somewhat different than that required for aircraft. The differences arise due to two major factors: first, there is a smaller premium for light weight, compact devices and second, ships move at much lower speed than aircraft and consequently computation may be done at a lower rate.

Some differences exist in the communications as well as navigation equipment. The Communications Load Analysis of Volume II indicates an all-voice only system is not only impractical, but unnecessary for the 13,600 ships of the North Atlantic population. Digital communication with off-peak use of voice channels is a practical solution.

Ship antenna requirements are considerably less difficult to meet than aircraft requirements because there is no necessity to build low profile or flush-mounted devices. Nonetheless, a 0 db gain antenna is recommended for the navigation function and it is likely that the conical log spiral would be the most appropriate. The antennas for ship communications could easily be parabaloidal or horn types. It seems reasonable to consider a trainable antenna for use on ships; that is, an antenna which is pointed in approximately the right direction manually rather than relying on the use of multiple switched antennas as in aircraft. A 4 db antenna, for example, would have a bandwidth of 90 to 100 degrees. Higher gain antennas could be used to further reduce satellite transmission power requirements. For instance, a 10 db (with a beam width of 50-60 degrees) antenna would reduce the transmission power requirements by a factor of 4. The disadvantage of such an approach is that the antennas may have to be stabilized and aimed at a satellite for communication, especially on very rough seas. The approach utilized thus depends somewhat on the satellite constellation. For instance, with equatorial, snychronous satellites the relative direction between the ship and the satellite may be easily calculated at any time and the antenna aimed in the proper direction. If the satellites are in inclined orbits the computation becomes somewhat more difficult, hence the ship communications would probably tend to use the equatorial satellites.

The receiver for use on board large ships would be very similar to that used for aircraft. There seem to be no requirements substantially different from an aircraft. Naturally, packaging for protection from corrosion and other environmental phenomena would be required.

The computer for use on large ships may be quite slow since there is no requirement for rapid navigation computations. However, the memory requirements are similar to those for aircraft. The option exists for marine uses of manual computation of position.

The display for ship use requires latitude, longitude, and system status for the case where a computer is used. For a system relying on manual computations the display would probably read out range data, time, satellite ephemeris, clock correction coefficients and system status; although the last three items might be considered optional for users requiring less accuracy and cheaper sets.

2.3.6.4 Small Marine Users

Most small privately-owned ships would not use a digital computer, but rather the position would be determined by manual calculations. Also, most small ships would not use the communications portion of the system. It is reasonable to expect many small ships would take advantage of the navigation satellite system after the system has been in use for some period of time and the cost of user equipment is suitable.

A conical log spiral antenna is also suitable for small craft. The receiver, in function at least, would be identical to that used in aircraft but packaged for a marine environment. Displays would read out the range information, time and satellite data necessary for manual calculation of position.

2.3.6.5 Marine Scientific Ships

Well equipped scientific ships engaged in various aspects of oceanographic research would undoubtedly take advantage of the navigation satellite system. It is not clear at this time whether they would require, or be eligible to use the communications portion of the system. The one difference between a scientific ship and all other ships may be the inclusion of speed measuring capability. It is possible that ships, especially those engaged in research on ocean currents, would require accurate velocity measurement. In this way it would be possible to get almost instantaneous, precise measurement of ocean currents any place within range of the satellite constellation. The equipment used on the scientific ship would be similar to that used on the large ships mentioned above. For the navigation function a 0 db antenna is again adequate.

2.3.6.6 Portable Configurations

There are several applications for portable navigation satellite receiving equipment. In particular, the portable configurations are useful for mapping, survey, prospecting, geological survey and other scientific and conservation expeditions.

The portable equipment may take one of two forms. In the first case the system would be completely self-contained with antenna, receiver, computer and displays. In this case the user could determine his position, or location, within a short time after setup of the equipment. The second form would be a system with an antenna, receiver, and a recording device. In the second form, the system would only record range differences time and satellite data when the system was used. The actual location would be determined by analysis of the data after return from the field. This second form has the advantage of lower cost since the recording process is simpler and cheaper than the display/computation process, but its other characteristics are similar. In many mapping or survey operations, determining position after returning from the field would not be a serious handicap. With proper design, the recording device used with the portable receiver would produce data in a form readily acceptable to a large computer. Most companies or government facilities which would sponsor field use of the portable navigation satellite receiver would have available some form of computing facility. This will be particularly true in the 1970's when computers of all sizes will be widely available.

The physical characteristics for the two forms of a portable navigation receiver have been estimated and are shown in Table 27.

The antenna is a conical log spiral mounted on a short rod which is held vertically within 15 degrees for proper reception of the satellite signals. It is similar to the conical log spiral mentioned elsewhere in this report with the exception that the antenna pattern extends over an area of 200 degrees rather than 160 , thus eliminating the requirement for accurate vertical orientation.

The portable receiver is essentially identical to the aircraft receiver.

The computer requires approximately 2500 words of memory and the speed is well within the capability of presently available units. The primary concern for the portable computer is cost and power drain. The battery is sufficient, under the worst case temperature conditions, to operate the portable navigation system for at least 144 fixes over a period of not less than 24 hours. A simple digital readout with a battery condition indicator and indicator lights which show that the equipment is operating properly are required for the display.

The recording portable navigation equipment characteristics are also shown in Table 27. Note that there is little reduction in size and weight but very significant reduction of cost. This is due to the elimination of the computer, most of the display and the substitution of a tape recorder.

It was assumed, in this analysis, that the antenna and the receiver were identical to those used in the self-contained system.

The above systems are navigation only and do not contain any facility for communications. It does not seem likely that there is any great

Table 27. Portable Navigation System

	Self-Contained	Recording
Size (ft^3)	0.36	0.06
Weight (lb)	16	15
Cost (dollars)	9550	4460
Battery Life (fixes)	144	160

demand in the civilian community for a satellite communications relay capability in a portable unit. Presently, field expeditions rely on short wave communications and except in the case of emergency, when a transmission might have to be made during unfavorable periods of the day, short wave is adequate. Short wave communications at night are relatively reliable and the equipment is relatively inexpensive.

2.3.6.7 Ground Vehicle Configurations

The same types of users who would have need of a portable navigation receiver system might also have use for a ground vehicle configuration. A system for use in ground vehicles could either be self-contained or recording as above. The cost savings for the recording system would be considerable just as in the case of the portable system.

There are two approaches to constructing the ground vehicle system. One is to construct a system suitable only for use on the ground vehicle. This system would be packaged with great attention to shock and vibration resistance and little attention to the weight of the unit. The second approach to a ground vehicle system is to use the basic portable system described in Paragraph 2.3.6.6 and to provide a power supply so that it may be operated from the vehicle power source. This second approach, while more expensive, is attractive inasmuch as a single receiver system may be used either portably or in the vehicle.

The physical characteristics of a ground vehicle system built exclusively for use in the vehicle is shown in Table 28. Here, as before, both the recording and self-contained types are shown. The system includes a conical log spiral antenna and a binor receiver. The self-contained system includes a computer and displays, while the recording system includes a tape recorder.

Table 28. Ground Vehicle Navigation System

	Self-Contained	Recording
Size (ft^3)	0.3	0.3
Weight (lb)	18	17
Power (watts)	35	32
Cost (dollars)	9100	4200

It is even more unlikely that a requirement exists for communications from a ground vehicle than it does from a portable system. This is the case because the ground vehicle will always be operating nearer to other forms of communication. Also vehicle-borne shortwave receivers are practical, relatively inexpensive devices which perform adequately for almost all occasions.

2.3.6.8 Discussion

It appears that the most likely users of the communications service provided by the NTCS System can afford (in terms of weight, volume and cost) a 4 db antenna. Although large ships could also use, say, 10 db antennas, thus providing an apparent saving in satellite power; this approach has not been recommended since these channels could not then be made available to aircraft at times of low maritime utilization. Therefore, a total system design based on a 4 db communications antenna is compatible with the vast majority of prospective NTCS. Since — as pointed out in the foregoing paragraphs — virtually all of the users of the position determination service can use a 0 db antenna, the system design based on that figure appears attractive.

2.4 OPERATIONAL GROUND STATIONS

2.4.1 General

There are three main classes of ground station functions that are involved in the Navigation/Traffic Control Satellite System operation. They are:

- Launch and injection
- Satellite system support
- User mission support.

The launch and injection portion of the mission will be supported by the usual national range facilities. Their use will not be unique to the NTCS system nor will the NTCS impose any new or strange requirements on the range systems. Therefore, they will not be considered further in the analysis of NTCS ground stations.

The second class of ground stations or ground station data handling functions is involved with providing direct support to the Navigation/Traffic Control Satellite System itself. Satellite system support functions include:

- Satellite tracking
- Reception of satellite telemetry
- Computation of satellite ephemeris and oscillator drift corrections
- Monitoring of satellite subsystem performance and status
- Transmission of latest correction data to satellites for storage on-board
- Generation of satellite commands for stationkeeping.

User mission support, as typified by the combined air traffic control/weather/company operational support functions performed at a station such as the Gander, Newfoundland, Oceanic Control Center, include:

- Surveillance of aircraft position
- Computation and data processing of present and future projected positions of controlled aircraft as well as uncontrolled aircraft in the area of which the Center has knowledge and the associated conflict searches, i.e., the determination of potential airspace violations, and/or mid-air collisions, and the determination of appropriate corrected action.
- The transmission and receipt of voice and data communications to include normal air route traffic control information such as requests for clearance changes, new clearances, traffic advisories, and acknowledgements; weather advisories, both air-to-ground, and ground-to-air; company communications; and messages of an emergency nature.
- Weather observation and forecasting services typical of those aeronautical and maritime meteorological services provided today, augmented to include items such as high altitude radiation warnings for supersonic transports.

2.4.2 Implementation

To compute satellite position and velocities for orbit determination and prediction, the process employed by the NTCS system consists of inverse navigation, i.e., using the navigation signal and the navigation receiving equipment in the tracking stations at known locations. The technique is effectively trilateration.

The basic tracking information consists of range, range rate, and satellite ID received at L-band at the basic BINOR data rate. The data are also used to provide real-time system status check and range and range rate information to the telemetry and command system if required. The telemetry data are derived on board the satellite as a result of monitoring certain spacecraft functions and are transmitted on command from a ground station.

When the NTCS satellites are launched, normal range facilities with the unified S-band system and STADAN net are utilized for ascent trajectory monitoring and positioning in the desired orbit. After final positioning, the dedicated ground system will accept the NTCS satellites for operation and maintenance.

For purposes of identification, the tracking stations, which are normally located apart from the operations control complex are called Remove Tracking Stations (RTS). The Mission Control Complex (MCC) consists of the Tracking, Telemetry, and Command (TT&C) station, a computation facility, communication facilities, and command and control facility. The computation, communication, and command and control functions are located together in the Project Operations Control Center (POCC).

The following summarizes the major ground system elements and their corresponding functions:

Element	Function
Remote Tracking Station(s) (RTS)	Tracking, data transfer
Mission Control Complex (MCC)	
Tracking, Telemetry and Command Station (TT&C)	Tracking, telemetry, command (satellite)
Project Operations Control Center (POCC)	Communications, computation, command and control (ground system), traffic control and mission support, liaison and coordination
Mission Support Module(s) (MSM)	Receive and transmit control and mission support data between satellites and POCC.

One Mission Support Module is added to the POCC for each satellite in active use as control relay point between the POCC and user aircraft and ships. It is clear that the air traffic control and mission support functions of the POCC and the associated MSM would be a part of the appropriate Air Traffic Control Center, e.g., the New York Oceanic Control Center. For analysis of the general problem of aeronautical/nautical traffic control and mission support, however, the functional elements are considered in the manner indicated above.

2.4.3 Requirements

2.4.3.1 Station Locations

For the NTCS to achieve its performance goals in terms of navigation coverage and accuracy, it is necessary that the ground system observe the satellite signals at known locations and compute the satellite position and velocity, as well as the on-board oscillator characteristics. A reasonable geometry is required so that the stations are well separated and not collinear. In the process of locating the stations, logistics support is to be considered. An examination of existing sites shows that Gander, Shannon, and J. F. Kennedy Airport, New York, meet the requirements with the possibility of having to add another site at Ascension. J. F. Kennedy Airport also appears to be an excellent location for the mission control complex.

2.4.3.2 Data Handling

Satellite Ephemeris. Previous studies have shown that the general computation process for the navigation data should normally occur hourly with a redetermination of the satellite oscillator characteristics, although with graceful degradation, this cycle can extend to as much as 8 hours. The RTS need only to take data approximately 15 minutes out of each hour. This factor coupled with data composition reduces the data rate requirement from the RTS to the MCC substantially and will permit the transfer over one of the L-band data links.

It is estimated that the total data processing requirements at the POCC are of the order of 20 to 35 minutes/hour of execution time for a machine of the size and speed of the 7094/II. This includes those functions associated with the navigation part of the system and the general maintenance and upkeep of the spacecraft (telemetry analysis, commands generation, orbit determination and prediction, scheduling, and attitude monitoring, and maneuvering). The entire process is one well suited to time multiplexing; i.e., the TT and C facility and the computation facility can be shared (and scheduled) among several satellites, thereby reducing the overall system cost and complexity.

ATC and Mission Support. The nature of the ATC and mission support data is such that it is unsteady in nature in terms of traffic flow. The type of data, clearances, advisories, emergencies, position reports (Autorep), and weather necessitate that portions of the ground system be dedicated to each communications channel whether it be voice or data. Since the total demand is a function of time, additional channels can be manned for peak periods and shut down during idle or slow times. The system design, however, must be such as to be able to support the peak operation.

2.4.3.3 Availability

The goal of the NTCSS ground system should be to attain essentially 100 percent availability. Furthermore, there should be an extremely low probability that a combination of circumstances can exist to prevent the ground system from performing its function. This can be accomplished through good operating procedures, the use of selective redundancy, and falling back to a degraded but satisfactory operating mode.

Principal candidates for redundancy in ground system are the large antennas and computers where either or both are in-line for operation. Since the NTCSS requires no large antennas, the emphasis is focused on the computers to be employed. A logical design objective is to use multiple processors where the full complement does the entire job at approximately 75 to 80 percent utilization. In the case where the full complement is three systems, for example, one could expect then to accomplish a slightly degraded operation with two out of three, thereby introducing a powerful advantage for continuous (near 100 percent) availability.

The navigation system availability of the NTCSS is greatly enhanced by the general non-real time ground support requirements and the utilization of an 8-hour memory in the spacecraft.

It is expected that the ATC/mission support communication function availability can be similarly improved through the use of commonality across the various equipment in the POCC and MSM to the extent that the computers, displays, and transmitting and receiving equipment can be cross-stropped. Another important consideration in the design would be the capability to perform maintenance during the operational lulls.

2.4.4 Ground System Design

2.4.4.1 Remote Tracking Stations (RTS)

The NTCSS RTS is a small, simple configuration manned by a single operator. It utilizes the following equipment:

- BINOR Receiver Terminal
- Data Processing and Control Equipment
- Data Transmission Terminal.

This equipment is depicted functionally in Figure 46.

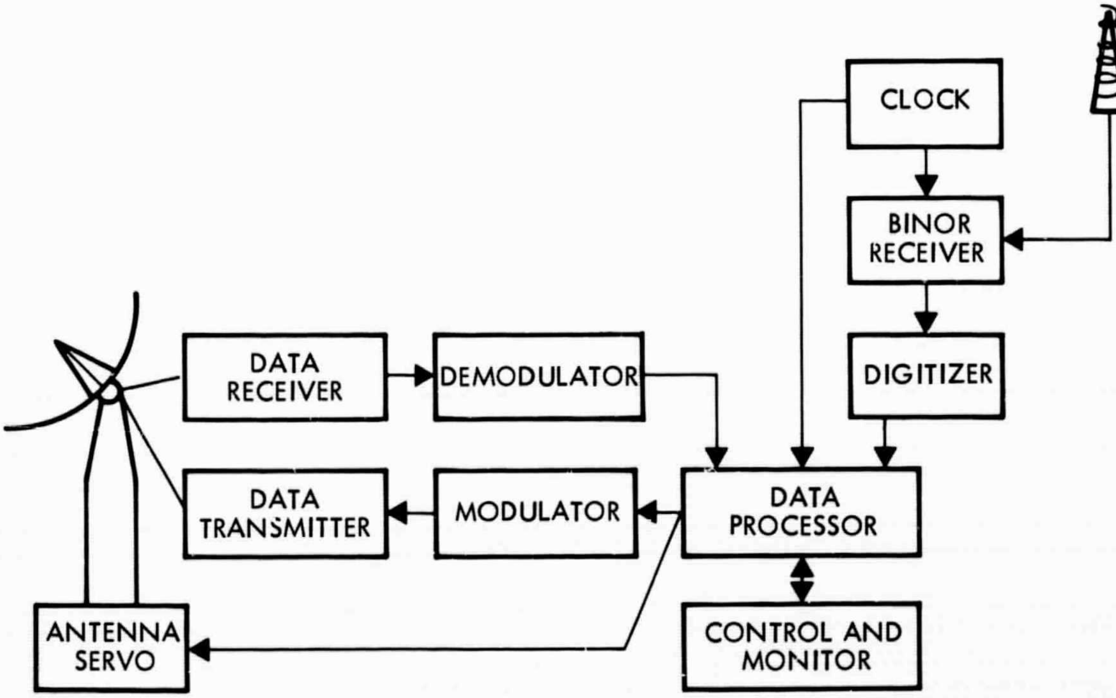


Figure 46. Remote Tracking Station Equipment

BINOR Terminal. The observation of the navigation signals from the satellite is identical to that of a user (aircraft). A BINOR receiver (similar to the user's equipment), a timing reference, and a conical log spiral user antenna is proposed for the RTS. The receiver outputs a measurement of range and range rate for each of the satellites at the TDM rate in use.

Data Processing and Control. The data processing and control unit is shown in Figure 47. The heart of this unit is a small digital computer with a nominal 8000, 16-bit word memory coupled with an atomic timing standard. I/O to the computer is attained through a teleprinter with a paper tape punch and reader.

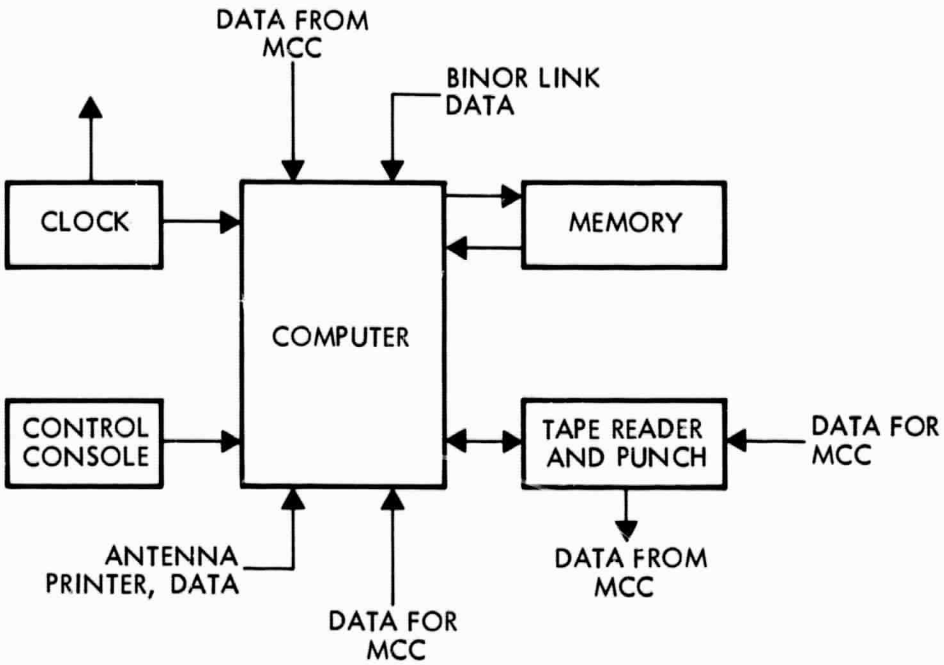


Figure 47. Data Processing and Control Unit

Data Terminal. The data terminal (Figure 48) provides a two-way data link between the RTS and MCC. The tracking data and any administrative data are transmitted over this link on schedule or on demand by the MCC.

A small elevation azimuth parabolic antenna is proposed for the data terminal. Its small size requires no special design features for wind loading. The antenna can be positioned to a specific satellite (by the computer with the ephemeris data received over the BINOR link), or it can be positioned by rotating the control knobs on the pedestal control unit.

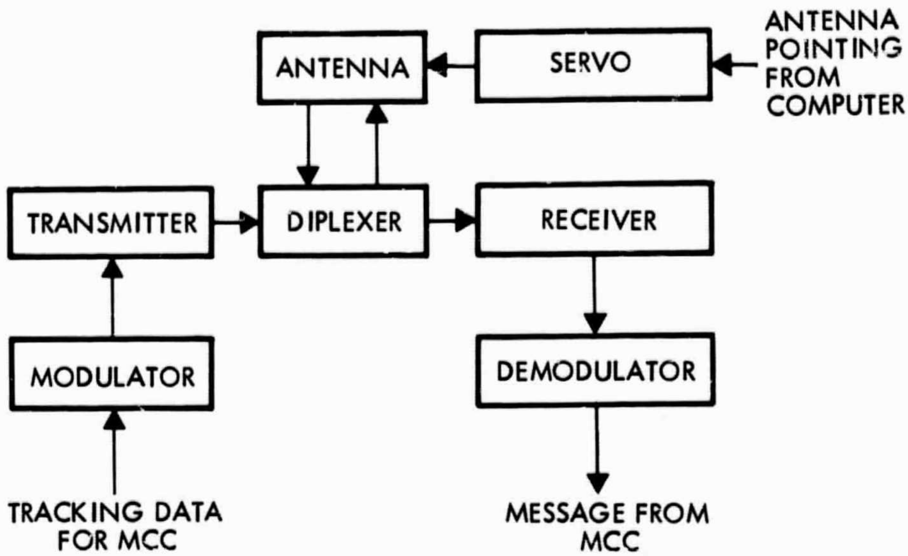


Figure 48. Data Terminal

2.4.4.2 Mission Control Complex (MCC)

The Mission Control Complex incorporates two major subdivisions: the POCC and TT and C station. For ease of operation and coordination, they should probably be housed in the same facility, although POCC/Air Traffic Control Center Coordination and Air Center Facility Space Limitations, for instance, could change this.

The MCC utilizes the following equipment:

POCC

- Terminal communications equipment for interface with the TT and C and the various using and coordinating activities
- Data processors for orbit determination and prediction, telemetry analysis, command generation, NTCSS command and control display, traffic control conflicts resolution, and other supporting analysis
- Display equipment
- Transmitters and receivers for ATC/mission support communications

- Teleprocessing equipment for handling and storage of ATC/mission support data.

TT and C Stations

- BINOR terminal (as in RTS)
- Data transmission terminal (as in RTS)
- Modified unified S-band terminal including antenna, command, and telemetry equipment.

TT and C Station. The TT and C operates under the direction of the POCC. A simplified functional block diagram is shown in Figure 49. The NTCSS TT and C utilizes subsystems of the NASA Unified S-band System (USBS). The spacecraft employs all the normal vehicle components so that the pseudo-random code ranging can be used during the ascent and positioning phases of the spacecraft mission. Once the spacecraft is in proper orbit, however, and its on-board oscillator has been confirmed to be operational, the L-band signal is used for all tracking, thereby negating any requirement for continued use of USBS tracking by the dedicated ground system. The USBS command and telemetry links are used both during launch and by the NTCSS ground system for continuing operations. A block diagram of the NTCSS simplified USBS is shown in Figure 50.

The antenna is a 30-foot az-el drive. The primary operating mode is under the control of the POCC computer. However, it can also be positioned manually using the control knobs on the antenna console in conjunction with a table of az-el values. It can either be operated with a radome for weather protection or in the open with a more passive pedestal.

Project Operations Control Center. The POCC consists of the following subsystem groups: the data processors, displays and Mission Support Modules, and is functionally divided into those elements which provide satellite system support and those which provide ATC/mission support communications. In some cases, elements of each are shared in order to provide redundancy or to allow for more economical operations.

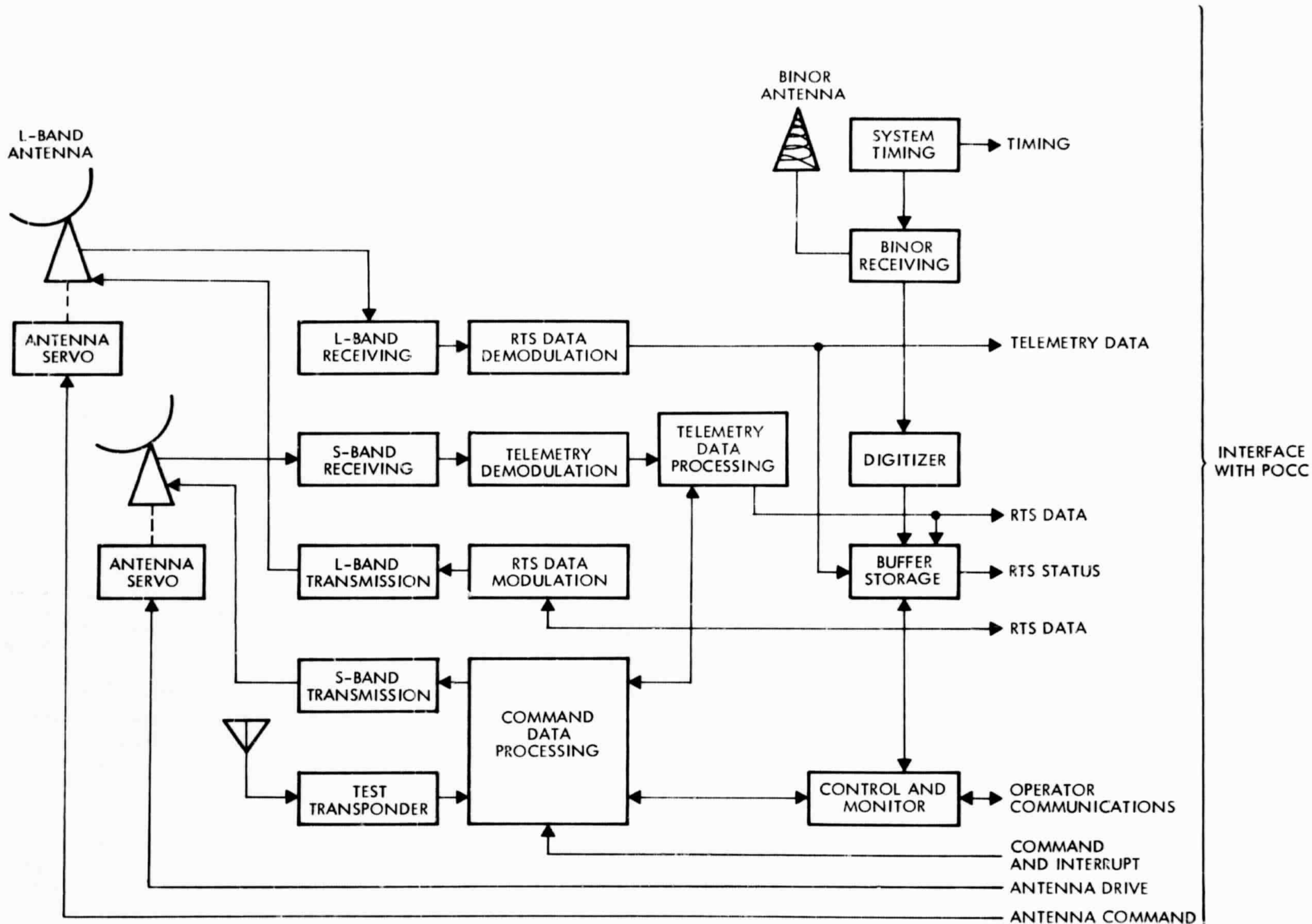


Figure 49. TT and C Functional Block Diagram

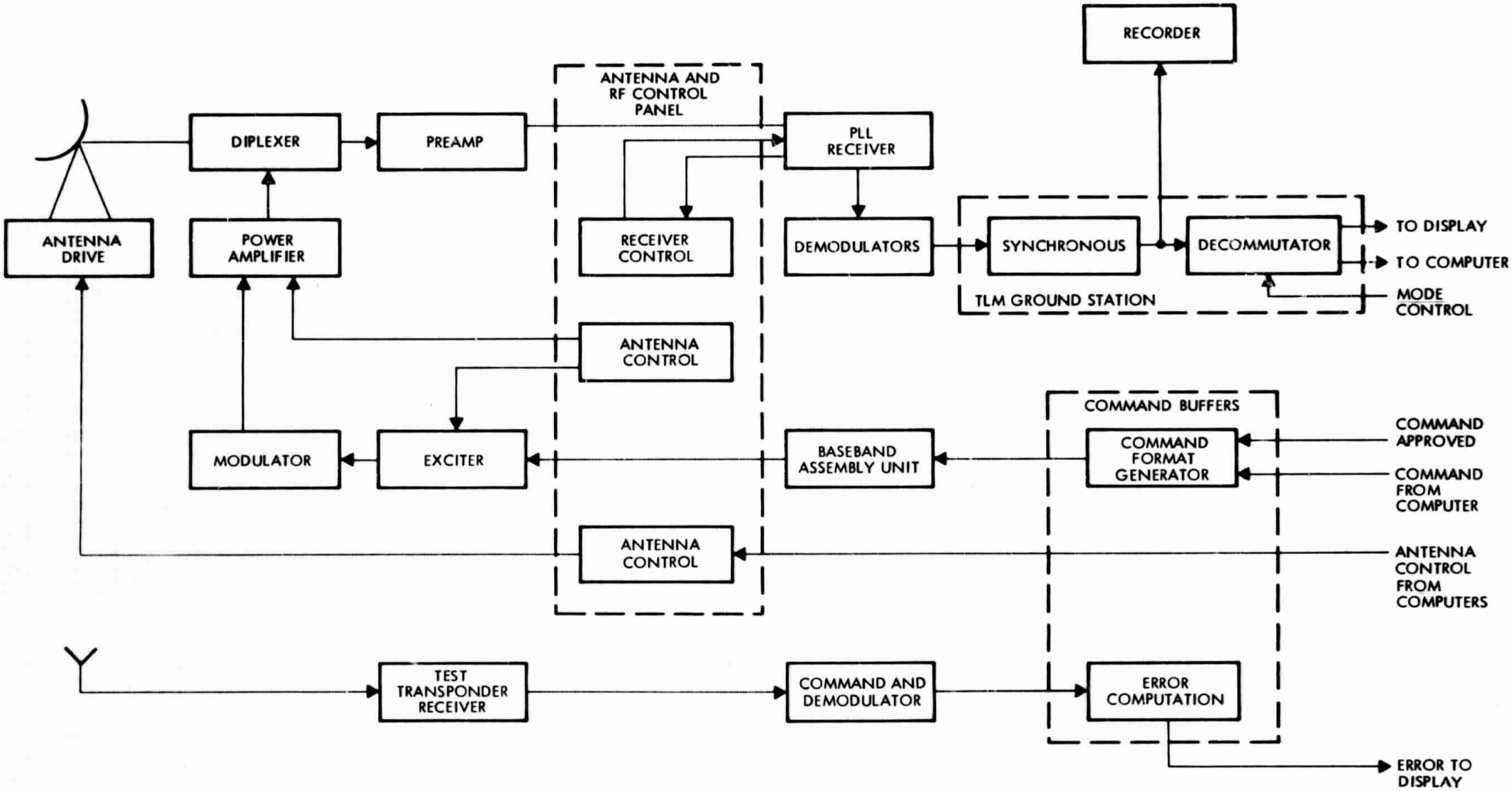


Figure 50. NTCSS USBS Dedicated Station

The Mission Support Module consists of a teleprocessor, transceiver, and antenna equipment to support the communications channels and associated controller display equipment. The intent is to provide for increased capacity and/or growth by adding modules as required in keeping with the traffic requirements.

The POCC block diagram is shown in Figure 51.

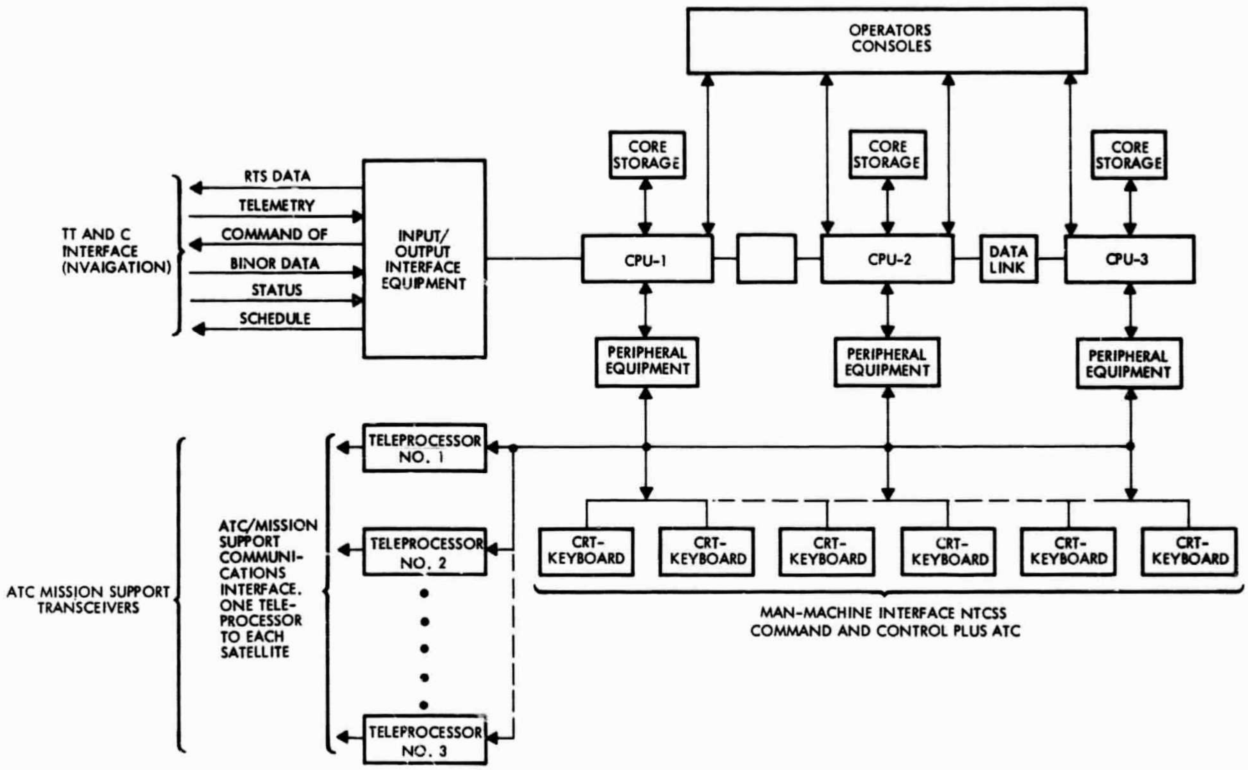


Figure 51. POCC Block Diagram

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3. DESIGN AND DEVELOPMENT/PREOPERATIONAL NAVIGATION/TRAFFIC CONTROL SATELLITE SYSTEM

3.1 GENERAL

This section will describe the system used to develop and demonstrate the traffic control and operational support capability provided by the NTCS System. The Design and Development/Preoperational (Configuration A) satellite is identical in appearance with the Configuration C spacecraft shown in Figure 8 except that fewer solar array panels will be required. The satellite flight test phase will be conducted with two Configuration A satellites in synchronous equatorial orbit, parked over 56 degrees W. longitude and 15 degrees W. longitude, and providing the coverage indicated in Figure 52.

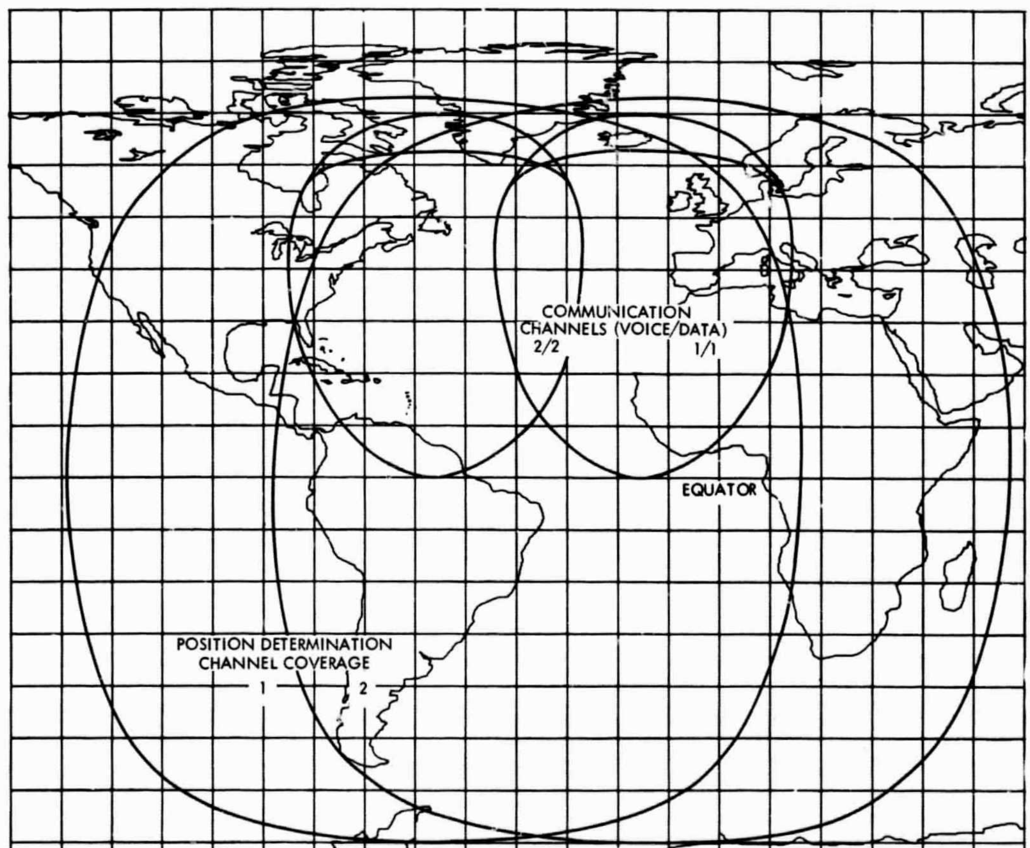


Figure 52. Phase I - D and D Preoperational Coverage

Once the Configuration A Satellites are developed and in service, the international aviation community will have both the operational capability and the management information required to formulate new international agreements concerning separation standards, procedures, and the like. The emphasis here is clearly on development, so that the entire North Atlantic aeronautical and nautical operational support services can rapidly incorporate the new hardware, software, and procedures to take timely advantage of this unprecedented capability.

3.2 D AND D/PRE-OPERATIONAL SPACECRAFT

3.2.1 General

As indicated in Section 1.3, the D and D/Pre-operational spacecraft and the two operational spacecraft configurations are very similar. Thus, the various subsystems are discussed briefly in the following paragraphs.

3.2.2 Position Determination and Communication Subsystem

A block diagram of the Position Determination and Communication Subsystem (PDCS) is given in Figure 53. A common (high gain) antenna plus a diplexer is used to transmit and receive voice and data signals. The satellite transponder design provides a capability to service one voice and one data channel. The BINOR navigation data are transmitted by an independent transmitter and earth-coverage antenna. The subsystem design is similar to the operational configuration, and further details on specific components and a preliminary link power budget is given in Section 2. The total power consumption for this subsystem is estimated to be 330 watts; and subsystem weight will be approximately 56 pounds.

The TWT power amplifier proposed for this program is based on a highly efficient (beam efficiency of 40 to 50 percent) tube such as the Watkins-Johnson WJ-395 with appropriate scaling for operation at L-band. As reported by the manufacturer (Reference 1), it is possible to achieve a power variation of 12 db by appropriate adjustments in helix voltage, beam current, and RF drive with a variation in beam effficiency of a

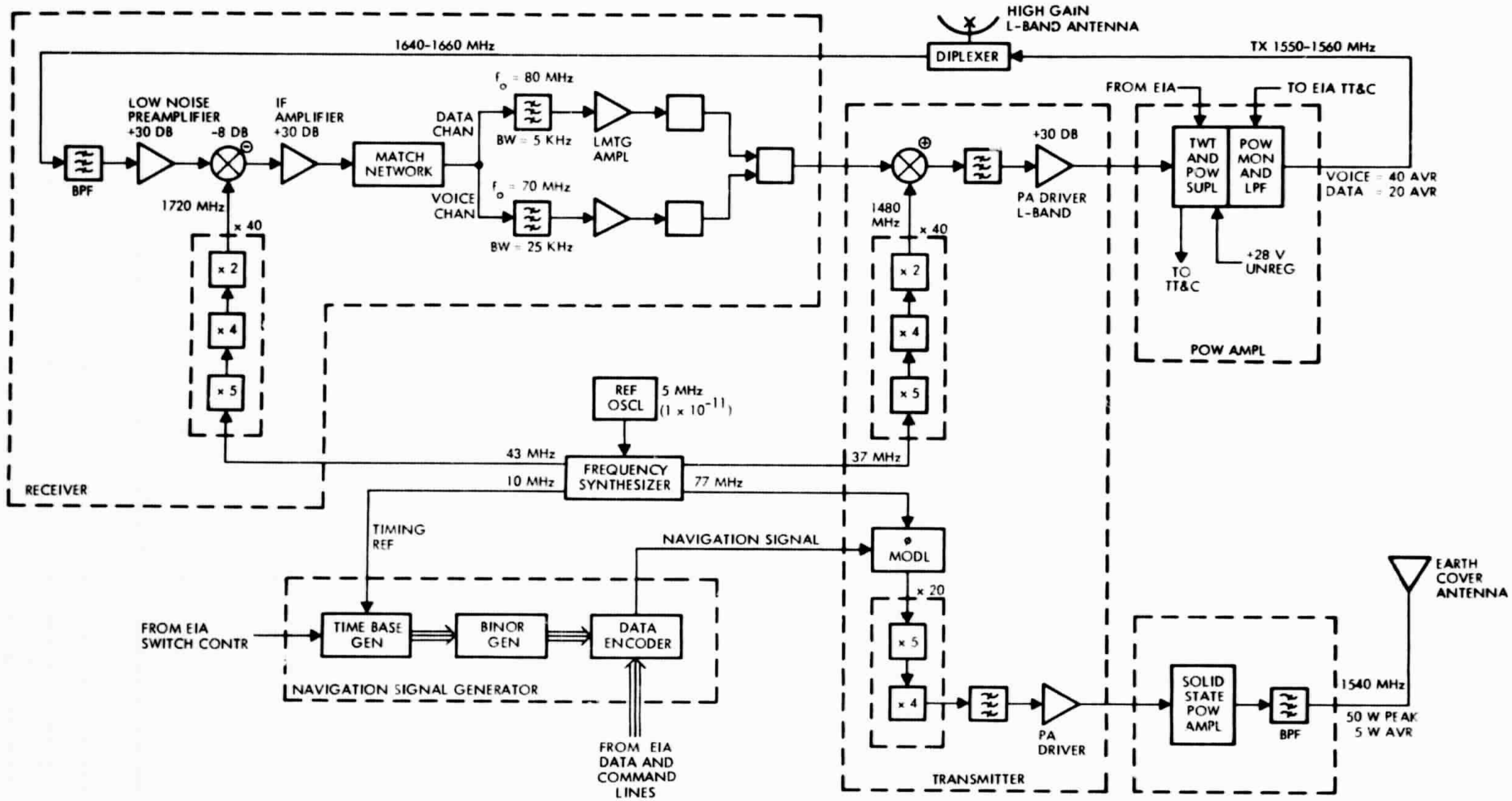


Figure 53. L-Band Position Determination and Communications Transponder

factor of 2. This adjustment feature would be incorporated in the D and D configuration. It would permit a 2- to 3-db increase in the power output of the voice channel, thus providing some additional signal margin for this channel during the pre-operational test program. In fact, it is desirable and planned, for test purposes, to evaluate user reception with variations in satellite RF from 40 to 60 watts per voice channel. This test data will be useful in the design of the user hardware.

3.2.3 Telemetry, Tracking and Command (TT and C) Subsystem

The proposed TT and C subsystem will be identical for the D and D and operational systems and is compatible with the NASA Unified S-Band System (USBS) presently being implemented at selected stations in the STADAN network (see Figure 54). The general features of this subsystem would remain the same even if a VHF operating T and C frequency was assigned to this program. Perhaps the most significant change would be in the design of appropriate TT and C antennas.

3.2.4 Antennas

The NTCS requires antennas to support the following services:

a. An omnidirectional antenna for TT and C.

b. A high gain antenna for the voice/data transponder.

c. An earth-coverage antenna for the BINOR navigation signal.

The proposed antenna consists of a conical log spiral plus an Archimedian spiral for the onmiantenna, plus separate parabolic reflectors for the high gain and earth-coverage antennas. The use of the parabolic reflector for the high gain antenna is a compromise between performance and technical risk. Further study of the possible application of phased array techniques to this problem appear warranted, however. Aside from the performance advantages, such as multiple beams, beam steering, and zooming, which are part of the phased array capability, this antenna could provide 3 to 6 db of effective gain (for a given area of coverage) over a parabola. The attendant reduction in transmitter and solar array power, and the consequent reduction in satellite weight are significant.

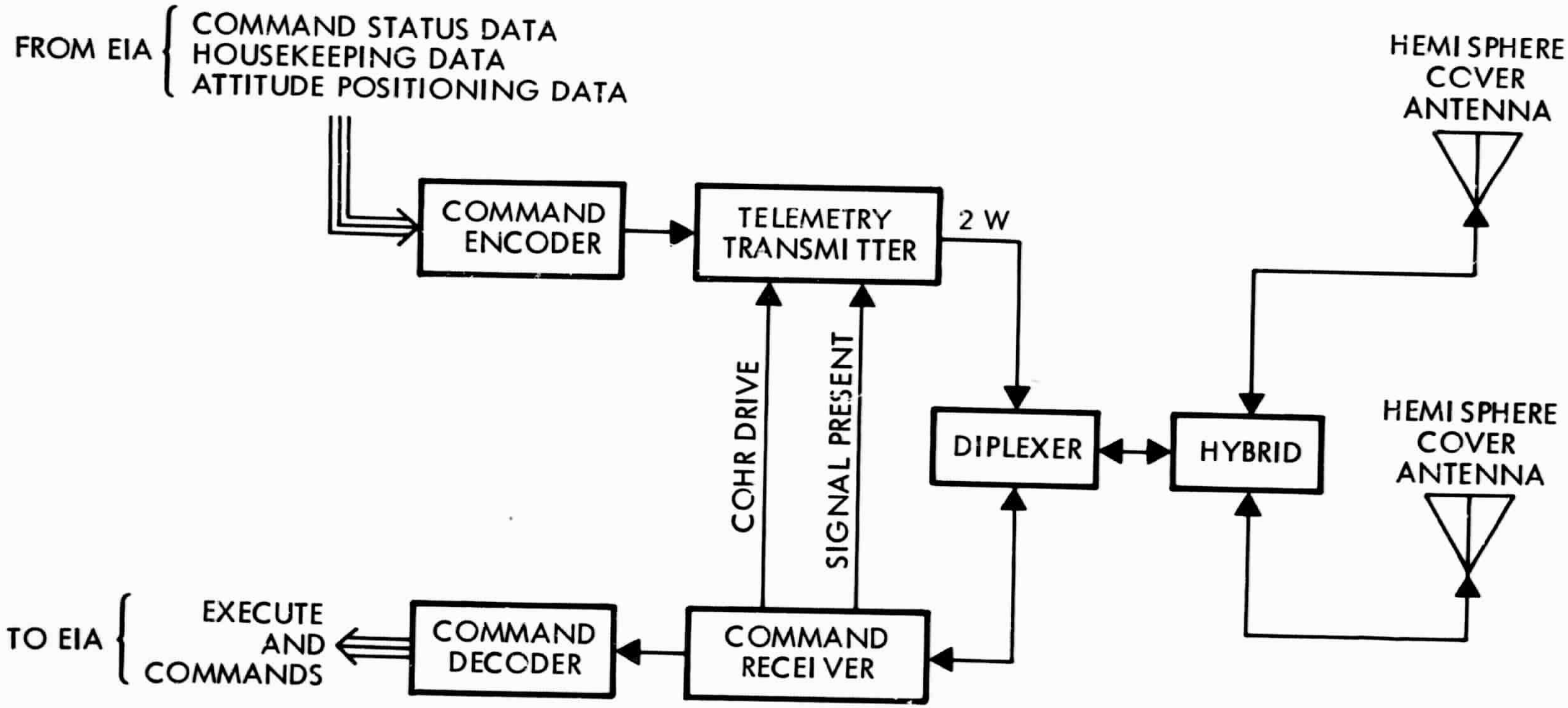


Figure 54. S-Band TT and C Transponder (USBS)

3.2.5 Attitude Control Subsystem

The attitude control subsystem for the D and D spacecraft is almost identical with the system proposed for the operational configuration. The system utilizes a three-axis gyro reference package during transfer orbit and during periods when the propulsion system is operative. At all other times, an ensemble of earth sensors and sun sensors will provide attitude reference for the spacecraft. Attitude control is achieved by a mass expulsion system utilizing dry nitrogen as the working fluid. A reaction wheel with its spin axis oriented along the spacecraft pitch axis is used to increase the attitude stability of the satellite. This reaction wheel provides a means of stabilizing the spacecraft yaw axis. Without this wheel, it would be possible for large yaw errors to develop during eclipses or noon crossings when the sun is no longer available or useful as a reference. These large yaw errors could rapidly deplete the available gas supply and prematurely terminate satellite life.

The sun sensor provides data for attitude control of the spacecraft and also provides signals which drive the solar arrays to their desired orientation. The sun sensor is a null-type sensor, i.e., it has no output when it is oriented properly with the sensor face perpendicular to the sun. Obviously, there is an ambiguous or false null if the sensor is facing away from the sun, and this probability must be considered during the acquisition phase of the flight plan.

The ACS for the operational inclined satellites (Configuration B) is similar to configurations A and C with one major exception. Due to the orbit inclination, it is very inefficient to take the array power loss which is inherent in the configuration A/C design. Rather, it is necessary to include a capability for accomplishing yaw turns so that full array orientation may be achieved at all times during an orbit and at all times of the year. The proposed solution for Configuration B is to replace the pitch wheel with a yaw wheel. While this change increases gas consumption somewhat, the additional gas required over a 5-year period is still less than the weight of an additional wheel. Consequently, it appears desirable to delete the pitch wheel. The yaw wheel will be torqued at some constant rate (variable with the season) to insure full solar array orientation capability.

The reaction wheels required for the smaller satellites will be scaled in size. These wheels and electronics are estimated to weigh 9 pounds and consume 5 watts of power.

All other features of the ACS for the D and D satellite are identical to the Configuration C design.

3.2.6 Electrical Power

The power subsystem for the Configuration A satellite is a scaled-down version of the operational satellite configuration. The basic array size was based on the power requirements given in Table 29. The derivation of the subsystem weight is as follows:

		lb
Array (530 watts at 15 watts/lb)		35
Deployment motors (2 at 3 lb/ea)		6
Battery (1 at 12 A. H.)		23
Power control unit		10
Central converters (2)		8
Miscellaneous (shunts, etc.)		8
	Total	90

This weight estimate is based on the use of the single crystal, accordion-fold array as described in Section 2, and in Volume II, Section 6.

The battery size was selected on the premise that eclipse operation would not be required except for the BINOR channel. The selected battery will provide 100 watts of power for the full eclipse period of 1.2 hours at a 30 percent depth-of-discharge. In addition to providing the reduced eclipse operation, the battery is required for injection, for meeting peak load requirements, and for general power regulation.

3.2.7 Propulsion

The propulsion system for the D and D satellite is completely identical to the operational configuration, except that the quantity of stored propellant will obviously be reduced.

Table 29. Configuration A Power Requirements

		Average Power (watts)
POSITION DET. AND COMMERCIAL SUBSYSTEM		340
BINOR Electronics	30	
Comm. Transponder	280	
BINOR Transmitter	30	
TELEMETRY AND COMMAND		25
ATTITUDE CONTROL		30
Reaction Wheel	5	
Earth Sensor	14	
Array Drive	7	
Control Elect.	4	
POWER		20
ELECTRICAL INTEGRATION		10
MISCELLANEOUS AND CONTINGENCY		20
Subtotal		445
Degradation @ 10%		45
Subtotal		490
Cosine Loss @ 8%		40
Array Power (Beginning-of-Life)		530

3.2.8 Electrical Distribution

The functions of this subsystem are described in Paragraph 2.3.3.7. The D and D component weights were estimated as follows:

		lb
Electrical integrated assembly		9
Telemetry integrated unit		4
Solar array harness		6
Spacecraft cabling		39
	Total	58

3.2.9 Thermal Control

All of the techniques and most of the operational spacecraft hardware are applicable to the D and D pre-operational configuration. All are based on flight-proven hardware, and no problems are anticipated to meet the requirements of this program.

3.2.10 Mass Properties

Table 30 presents the estimated weight breakdown for the D and D satellite. As shown here, this design would fully utilize the projected boost capability of the 9 Castor solid version of the Thor/Delta which is discussed in Section 3.2.12. This should not be construed as a design approach with no margin; there is a 10 percent (60 pound) contingency in the weight estimate; and noted previously, there are techniques available for increasing or optimizing the booster payload capability. For example, a substantial weight increase can readily be obtained for the D and D test configuration by simply varying the orbital parameters of eccentricity and inclination. This could be readily accomplished without compromising any of the principal test objectives. Further, the propulsion tankage volume could be increased—perhaps 10 to 20 percent—to provide some additional payload capability if subsequent design studies established such a requirement.

Table 30. Mass Properties—D and D Configuration

Structure and thermal control		98
Propulsion		161
Attitude control		75
TT&C		21
Electrical integration		58
Power		90
Antennas		33
Position Det. and Commercial		56
	Subtotal	592
	Contingency (10%)	60
	Dry satellite weight	652
	Residuals and pressurant	68
	On orbit weight	720
	Expendables	1400
	Gross satellite weight	2120

3.2.11 Reliability

Based on the results obtained for the operational configuration, it appears reasonable to assume that the system reliability for the D and D satellite will correspond rather closely to Configuration C since many of the major subsystems are identical. For those subsystems which are significantly different (such as power, PD and C, electrical integration), we find that reduced performance requirements in the D and D satellite have been somewhat balanced by a commensurate reduction in redundancy. The net result should be that the reliability of these subsystems will not drastically change from the values shown.

For the D and D test program, the satellite reliability and life are intimately associated with the anticipated flight test schedule. Obviously, the satellite must survive long enough to satisfy all planned test objectives. Operation beyond this period is desirable at least to the extent required to allow the operational spacecraft to be launched. Thus it appears that a satellite design life of three years, and a satellite MTTF of three years were considered to be adequate D and D/Preoperational spacecraft reliability requirements. Operational life beyond this period would probably also be beneficial in that the full position determination capability would be available during the user retrofit and equipping phase when the full communications capacity of the system would not yet be required.

3.2.12 Booster Selection and Performance

Table 31 shows planning estimates of the useful third stage load into a 100 x 19,400 nmi transfer orbit for the Thor/Delta family of boosters assuming an ETR launch. These payload weights are assumed to be based upon a spin-stabilized injection mode during third stage operation. There are certain techniques which have been considered and can be employed to increase payload capability, such as:

a. Use of a lightweight fairing in place of the present fairing.

b. Use of a bi-eiliptic transfer in lieu of a Hohmann transfer.

c. Use of an additional propulsion unit (fourth stage) or use of satellite propulsion system to supplement the booster capability.

d. Optimization of launch azimuth, plane change ratios between the Thor and the payload, use of soft cutoff, etc.

e. Injection into inclined synchronous orbits rather than equatorial synchronous.

f. Injection into elliptical orbits rather than circular orbits.

The first three methods were considered most appropriate and will be discussed in the following paragraphs.

Table 31. Thor Delta Performance

	Configuration	Useful Load 100 x 19,400 n mi Transfer Orbit	*Net Payload in Sync. Equat. Orbit
A	DSV-3L-1B/I. D. /TE-364-3 with 3 Castor I + 3 Castor II	935	490
B	DSV-3L-1B/I. D. /TE-364-4 with 3 Castor I + 3 Castor II	1014	530
C	DSV-3L-1B/I. D. /TE-364-3 with 3 Castor I + 6 Castor II	990	518
D	DSV-3L-1B/I. D. /TE-364-4 with 3 Castor I + 6 Castor II	1098	575

*Based on mass ratio = 1.91

The shroud exchange ratio $\left(\frac{\text{payload weight}}{\text{shroud weight}}\right)$ for the Thor Delta is approximately 0.1. Therefore, use of the A-12 short fairing (265 pounds) or a lightweight magnesium fairing instead of the standard Improved Delta fairing (535 pounds) will permit a useful load increase of 0.1 (535-265), or approximately 27 pounds.

Figure 55 shows the projected launch vehicle performance obtained early in the study (August 1968) for several different Thor Delta configurations. The bi-elliptic transfer can maximize the payload capability of certain booster-upper stage combinations. If the final stage (TE-364) is limiting the payload to apogee, full utilization of the primary boost vehicle capability may not be possible with a Hohmann transfer. Note: The following data were obtained late in the study (April 1969):

- The 6-solid Delta will fly in 1969.
- McDonnell-Douglas is under contract with NASA/Goddard for the design of the 9-solid configuration. Furthermore, they expect a hardware contract very soon and are planning on a December 1970 launch of the 9-solid Delta.
- This 9-solid configuration is now made up of 9-Castor II's rather than 3-Caster II's and 6-Castor II's. As a result, the configuration is now projected to provide a useful load in a 100 x 19,400 nmi transfer orbit of 1235 pounds rather than the previous estimate of 1098 pounds.

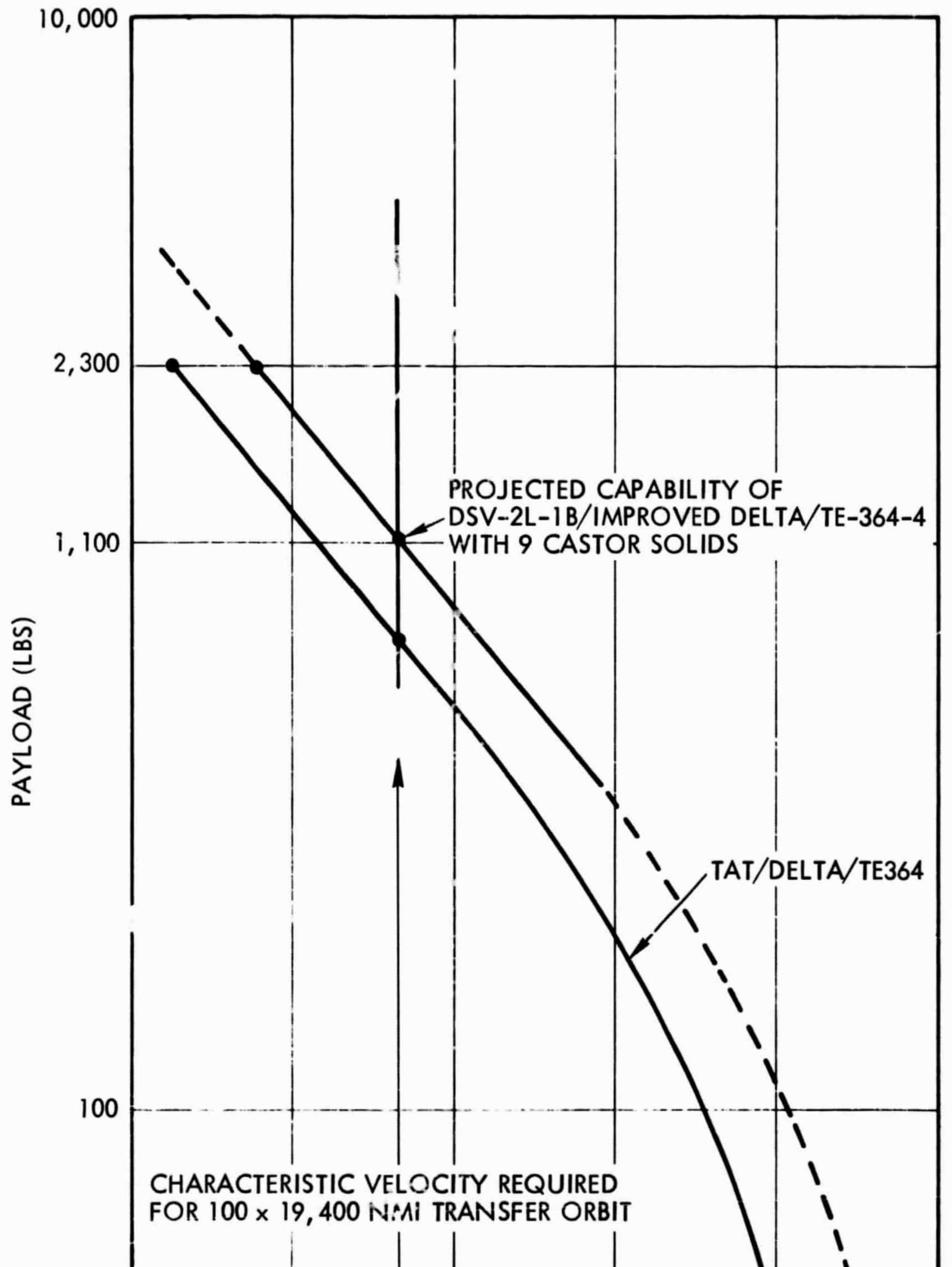


Figure 55. Projected Launch Vehicle Performance

The full capacity of the operational satellite propulsion system can be used in a transfer ellipse burn (see Figure 56) to increase the on-station payload weight. From Figure 55 it is seen that a 2300 pound payload results in a characteristic velocity of 29,200 ft/sec or 4400 ft/sec short of the required value. Hence the satellite propulsion system must provide a ΔV capability of (4400 + 440 + 6000) or 10,840 ft/sec. The mass ratio for this condition is:

$$\ln r = \frac{\Delta V}{gI_{sp}} = \frac{10,840}{9,660} = 1.11 \quad \text{or} \quad r = 3.04$$

which yields a burnout weight of:

$$W_b = \frac{W_i}{r} = \frac{2300}{3.04} = 756 \text{ pounds}$$

This payload weight requires a fuel load of 1544 pounds. For the nominal fuel load of 1400 pounds, the above value would be reduced to 720 pounds. Thus the use of a transfer burn has increased the on-orbit payload weight from 575 pounds to over 700 pounds for the Thor/Delta launch.

For the inclined elliptical orbit, with i = 52.5 degrees and ϵ = 0.35, the characteristic velocity required is 37,300 ft/sec. Figure 55 shows that the Thor/Delta will produce 29,500 ft/sec at a payload weight of 2100 pounds. Hence the propulsion system must make up the deficit of 7600 ft/sec plus gravity loss of 10 percent, plus ETC range safety penalty of 400 ft/sec or 8800 ft/sec. This requires a mass ratio of 2.7 which yields a net payload of 780 pounds injected into orbit. This additional weight capability of 60 pounds over the geostationary satellite could be used to improve the reliability, performance, and/or the satellite life of the inclined operational satellite.

3.2.13 Flight Plan

The D and D pre-operational satellite will be launched on a three-stage Thor/Delta vehicle from ETR. The booster will consist of the long tank Thor (DSV-2L-1B) with nine Castor solids, the improved Delta second stage, and the TE-364-4 third stage. The Thor/Delta combination will place the payload into a nominal 100 n mi parking orbit. Prior to

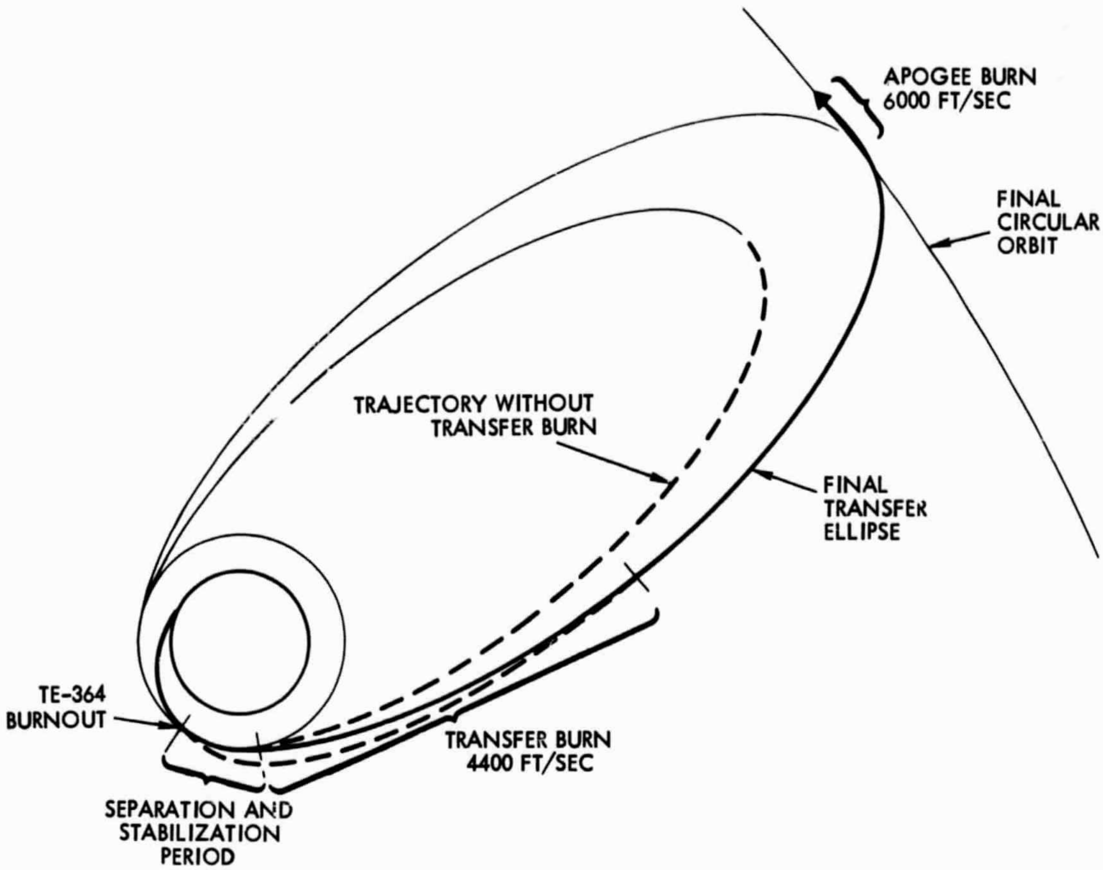


Figure 56. Injection Via Transfer Ellipse Burn and Apogee Burn

separation from the Delta vehicle, the payload will be oriented to a predetermined position and the rockets on the spin table will be fired to spin-stabilize the payload. Following spinup, the payload, consisting of the satellite with the TE-364 motor attached, will separate from the Delta. At the first equatorial crossing, the TE-364 motor will ignite to place the satellite in a nominal Hohmann transfer ellipse with apogee at synchronous altitude. Since the TE-364 cannot provide all of the velocity required for this trajectory, the NTC spacecraft liquid engine will be utilized to provide the velocity deficit. Following the liquid engine burn period, a yo-yo despin mechanism will be actuated to remove spacecraft spin. The on-board gyros in conjunction with the cold gas jets will remove any residual motion. When a fully stabilized condition is achieved, the solar array panels and antennas will be deployed and a stabilized condition re-established. At this point a programed search will be initiated to orient the satellite for the apogee injection firing at the second equatorial crossing. This will consist of sun acquisition by the two-axis sensors located on the solar paddles followed by a series of maneuvers to achieve earth acquisition. When this preliminary orientation is established, a final sequence of commands will place the spacecraft into the correct position for apogee burn. The final burn of the liquid engine will remove the orbit inclination and circularize the orbit. When the final orbit conditions have been achieved, the satellite will again be maneuvered to allow acquisition of the earth, and the final earth-pointing attitude of the satellite will be established. Generally, a drift period will be required prior to the final orientation maneuver to allow the satellite to be stationed at the desired longitude. Typical flight plan milestones are summarized in Table 32.

Table 32. Typical Sequence of Events

Time	Event
T_o	Liftoff; start Stage 1 programmer
(sec)	
90	Start Stage 1 guidance
150	Main engine cutoff; stage separation
180	Jettison fairing
550	Sustainer engine cutoff
1450	Fire spin rockets; stage separation
1500	Stage 3 ignition
1650	Activate yo-yo despin
2000	Deploy solar panels and antennas
(hr)	
$T_o + 1$	Orientation complete; liquid engine ignition
$T_o + 1.5$	First burn complete; start coast phase
$T_o + 5.5$	Second burn liquid engine
$T_o + 6$	Second burn complete
$T_o + 7$	Third burn if necessary; start drift to final position

3.3 D AND D/PREOPERATIONAL USER HARDWARE

It seems highly probable that the Navigation/Traffic Control Satellite System program will be accomplished in a fairly austere fiscal environment. Thus, the development organization must be prepared to run the design and development program on a likewise austere basis. The questions arise, then, "Should the NTCS development program include development of the ultimate user hardware along with the development of the satellite ground systems? Also, should the flight test program be flown with prototype or early production models as opposed to a more austere approach which would call for special ground or aircraft test instrumentation equipment or breadboard models of some user hardware the components?"

There is a real danger in conducting the design and development/preoperational program with engineering model or "lash-up" user hardware. The satellite system design will be essentially firmed up and the user hardware designs will continuously tend to be compromised in order to accommodate systems problems. This biasing of system tradeoffs will not be likely to lead to best overall Navigation/Traffic Control Satellite System solutions. Clearly, the user hardware considerations cannot completely dominate all tradeoffs, but problems should be worked on a system basis. It seems unlikely that this could be done when part of the system is under configuration and cost control (and is the responsibility of the NTCS program manager), and another part is at the breadboard or engineering model stage (and is not the responsibility of the NTCS program manager).

Thus the user hardware must not only be developed to the extent required to allow achievement of the specific technical test objectives, listed in Section 5, but the program should allow development of documentation, such as ARINC characteristics similar to ARINC Characteristic No. 561, "Air Transport Inertial Navigation System," Radio Technical Commission for Aeronautics (RTCA) document, "Universal Air-Ground Digital Communication Systems Standards," or the Air Transport Association's "Airline Policy and Requirements for Airborne

Collision Avoidance"; and to allow the international avionics industry to be in a position to bid competitively for the ultimate user hardware market. Active contributing participation in the flight test program by ARINC, ATA, RTCA, individual airlines, and industrial contractors, based on past experience with inertial navigation systems, collision avoidance systems, and the like, is certainly to be expected. In a very real sense, then, the NTCS program manager may have his cake and eat it, too, i.e., fairly substantial development of user hardware may take place, funded by outside organizations. The reason for this is clear. The entire aeronautical and avionics community recognizes that in a practical sense, if users are to be ready to take advantage of the NTCS System, they must take an active part in the development process so that the airlines and other users will both be equipped and trained when the satellite system becomes available.

TRW therefore recommends that the Design and Development/ Preoperational user hardware consist of the prototype and early production units of the aircraft components described in Section 2.3 and costed in Paragraph 5.3 with three modifications:

- If required, input/output, data processing, and display hardware can be off-the-shelf items early in the flight test program, as long as their functional performance characteristics are similar to the NTCS hardware.
- Clearly, flight test instrumentation for diagnostic and evaluation purposes will be required.
- Additional user hardware for marine and low cost air and marine users should be developed.

3.4 D AND D/PRE-OPERATIONAL GROUND SYSTEM

The ground system which is proposed for the pre-operational NTCSS is to be the conceptual prototype for the operational NTCSS. The design approach is to:

a) Minimize the D and D/pre-operational costs through the use of existing facilities

b) Provide for a meaningful test and demonstration of the NTCSS

c) Establish early participation by the various air traffic control organizations in order to:

- Obtain a realistic system D and D evaluation
- Generate meaningful pre-operational experience
- Ease the phase-in process.

The D and D/Preoperational NTCSS consists of two satellites located in geostationary orbits, a network of ground stations, and a number of specially equipped user vehicles deployed as shown.

The Stadan ground station (shown as Rosman, North Carolina) will support spacecraft operation in terms of monitoring all telemetry data and transmitting all spacecraft commands. NTCSS auxiliary ground stations located at Gander, Shannon, and Rosman will monitor the BINOR ranging signal from each satellite and periodically transmit these data to NAFEC which serves as the pre-operational NTCSS control station. NASA, FAA, and contractor personnel at NAFEC and Goddard Space Flight Center will utilize these data and other pertinent spacecraft status information to track satellite position, monitor oscillator performance, schedule pre-operational tests and evaluate test results. All computations associated with the NTCS will be performed at NAFEC. Table 33 lists the various stations and their functions.

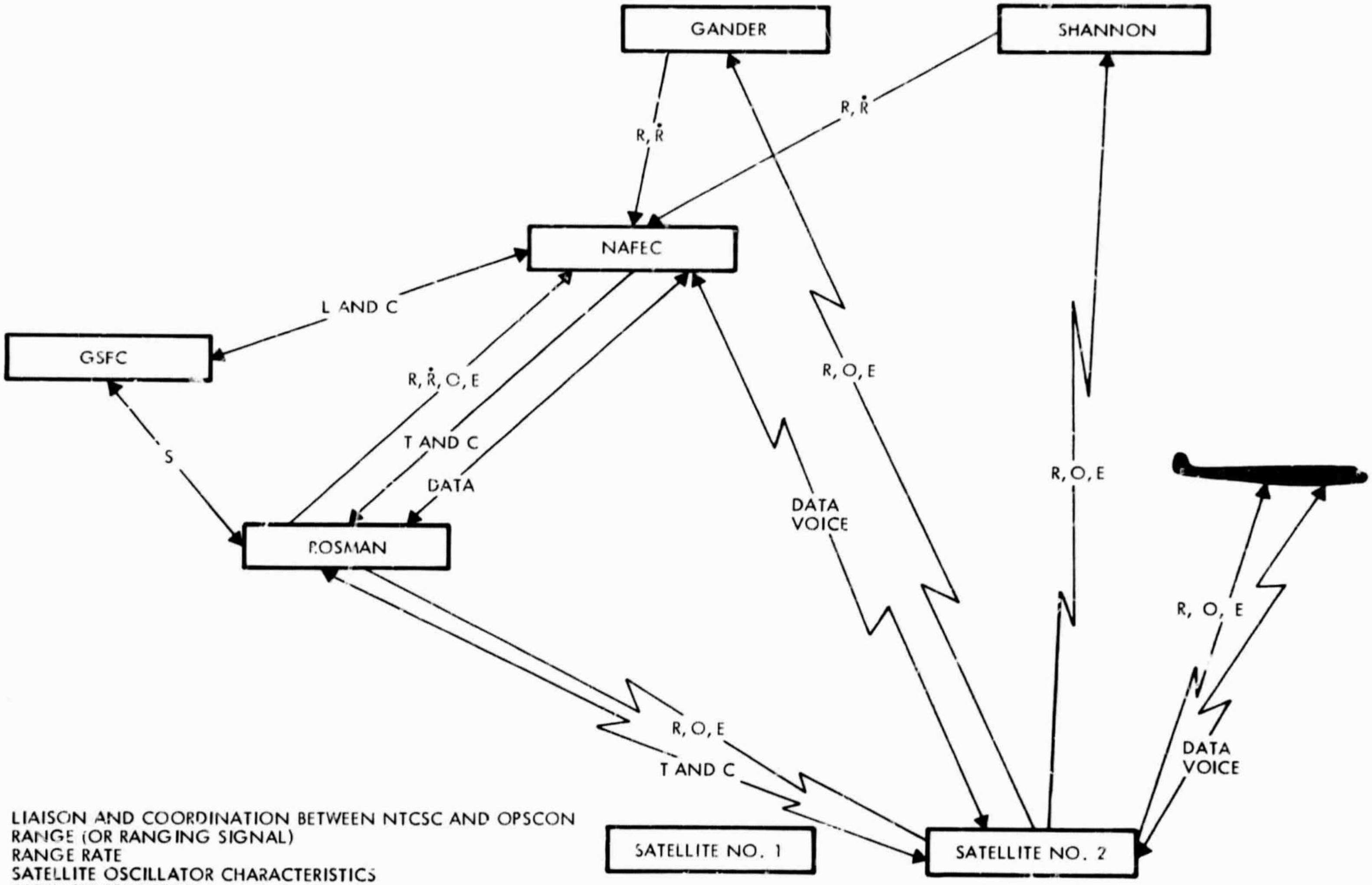
A list of communications links proposed for use in the pre-operational test phase is given in Table 34. Figure 57 depicts the data flow for the preoperational system. With this concept, NAFEC would effectively exercise the communications surveillance and control functions directly, while directing the telemetry and command and navigation functions through Rosman. Gander and Shannon would continue to perform traffic control functions as before, serving only tracking functions for the pre-operational system.

Table 33. Stations and Functions

Station(s)		Functions
Gander, Shannon	a)	Receive BINOR ranging data
	b)	Process
	c)	Transmit to NAFEC
	d)	Continue present traffic control function
Rosman	a)	Receive BINOR ranging data
	b)	Provide TT and C support for the NTCS spacecraft
	c)	Provide two-way communications with NAFEC
NAFEC	a)	System command and control
	b)	Computation center
	c)	Communications center for relay of system information

Table 34. Proposed Communication Links for Use in the Preoperational Test Phase

Link	Description	Frequency	Trans Power	Antenna Gain Transmitter	Antenna Gain Receiver	Remarks
1	Sat to user/ground NAV link	L-Band	50W	23 db		
2	Sat to user/ground Data link	L-Band	30W	23 db		
3	Sat to user/ground voice link	L-Band	60W	23 db		
4	Sat to user/ground telemetry link	S-Band	4W	0		
5	User/ground to sat data link	L-Band		25 db		
6	User/ground to sat voice link	L-Band				
7	User/ground to sat command	S-Band			0	
8	Gander/Shannon to/from NAFEX L-line existing facilities					
9	NAFEC to/from Rosman L-line 2400 bps 3-KC leased line					
10	NAFEC to/from GSFC L-line existing telephone lines					
11	Rosman to/from GSFC L-line existing facilities					



LEGEND

L AND C	LIAISON AND COORDINATION BETWEEN NTCSC AND OPSCON
R	RANGE (OR RANGING SIGNAL)
Ṙ	RANGE RATE
O	SATELLITE OSCILLATOR CHARACTERISTICS
E	SATELLITE EPHEMERIS
S	SCHEDULING DATA
T	TELEMETRY - SPACECRAFT HOUSEKEEPING AND STATUS
C	COMMANDS - SPACECRAFT STATUS CHANGE AND/OR NAVIGATION INFORMATION MEMORY LOAD
DATA	WX, CX, ETC.
VOICE	VOICE CHANNEL

Figure 57. Data Flow for Preoperational System

Many alternatives were available, of course. Among those considered were:

<u>Tracking</u>

Requirements:

Three L-band stations with reasonable geometry (separation). Receiving format and retransmit to computation facility.

Alternatives:

- Use NASA sites and existing communications facilities or send via satellites *
- Use air traffic control sites and selected alternatives, the same as above.

<u>Computation and Control</u>

Requirements:

- Orbit determination and prediction
- Telemetry analysis
- Teleprocessing
- System analysis and support
- Command generation

Alternatives:

- Use GSFC facilities and set up an OPSCON-POCC (Operations Control Center - Proposed Operations Control Center) at GSFC
- Establish POCC at NAFEC (work directly with TT&C and coordinate with GSFC)*

---

*Selected alternatives

Telemetry and Command

Requirements:

- Monitor spacecraft functions
- Command spacecraft as necessary to help appropriately deploy
- Load spacecraft memory periodically with navigation information

Alternatives:

- Establish dedicated facility with minimum TT and C equipment collocated with POCC
- Use STADAN stations and provide project peculiar gear ala OGO, ATS, etc.
- Use USBS equipment and maintain STADAN compatibility

Surveillance and Traffic Control

Requirements:

- Provide aircraft advisory and weather service center
- Voice and data communication terminal for transmission and receipt of information to user (aircraft and ships)

Alternatives:

- Handle through the TT and C facility, i.e., a STADAN station
- Use dedicated facility collocated with the POCC

4. EXPANSION TO A WORLDWIDE OPERATIONAL SYSTEM

4.1 GENERAL

The D and D configuration for the Navigation/Traffic Control Satellite System envisages two satellites in synchronous, circular, equatorial orbits with 56° W longitude and 15° W longitude subsatellite points. Assuming 10-degree minimum elevation angle, these two satellites provide coverage from the U.S. East Coast to London/Paris, across the whole North Atlantic region, including the southern half of Greenland and Iceland. These two satellites permit a limited user navigation capability and primarily serve to provide an early checkout of the ATC satellite communication and navigation concepts. Starting with these two satellites, it is proposed that satellites of different configurations be added through periodic launches, to progressively build up the worldwide NTC Satellite System. Each launch provides an incremental growth of coverage, communication capacity and accuracy in the system without disturbing the operational availability and use of existing system elements. From the two D and D satellites, the system can be built up to eventually include from 10 to 13 satellites, providing a capability for worldwide navigation with 100-foot order of magnitude accuracies.

The philosophy followed in selecting satellite constellations (i.e., the orbital configurations) is to first build up an operational traffic control and mission support service over the North Atlantic, extend this net to cover the continental U.S., then the Pacific, and finally fill in holes in coverage and accuracy to give a worldwide capability. The resulting set of satellite constellations is just one of a virtually unlimited number of ways in which an area coverage system such as this can be time phased into a worldwide system.

Assuming the continued use of barometric altimetry, most classes of users can satisfy their communication, navigation, and collision avoidance needs, while navigating within large areas such as the North Atlantic or CONUS, by using data received from only three synchronous, equatorial satellites. Therefore, the initial operational North Atlantic system will have three synchronous equatorial satellites with subsatellite points at 56°W longitude, 35°W longitude, and 15°W longitude.

For the worldwide system, there will eventually be eight or nine satellites in synchronous equatorial orbits, so placed that at least three satellites will be visible in all major navigation regions of the globe. It should be noted that if these satellites are maintained in precise earth-stationary orbits, significant economies in avionics equipment will be possible for the low cost user, since with earth-stationary satellites, the user will not need to receive satellite update information and compute ephemerides for each satellite in order to determine his X-Y position. However, because of the substantial stationkeeping fuel requirements, precise stationkeeping to provide this feature for low cost users will not be included in the early version of the satellite configurations proposed here.

4.2 INCLINED ORBITS

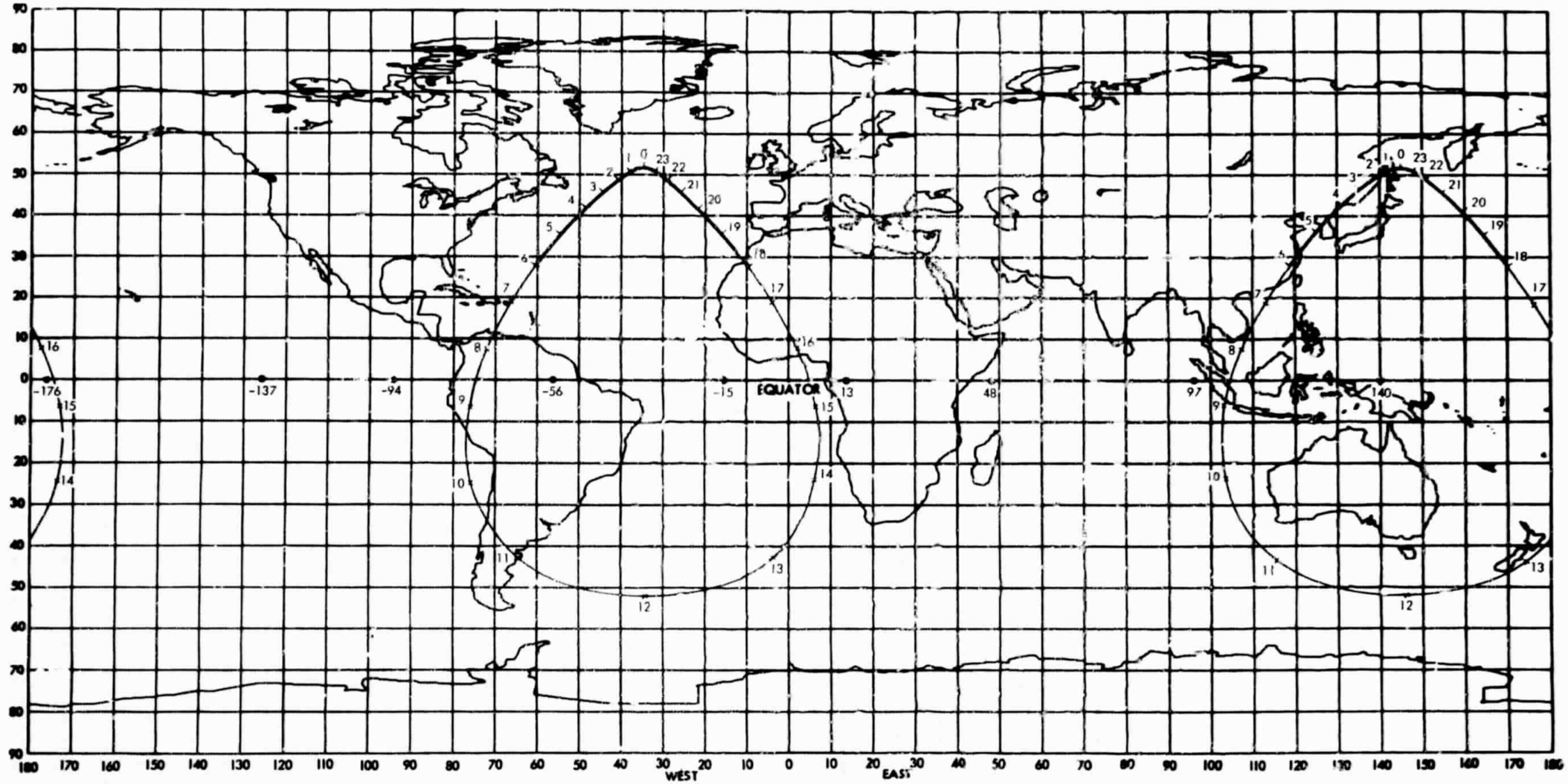
Synchronous equatorial orbits are deficient in several aspects. They cannot be used to derive altitude information in the mid-latitudes; they cannot provide accurate North-South position data in the equatorial regions; and they cannot provide communication and navigational coverage in the polar regions. All of these features, plus considerably improved X-Y position determination accuracy and system redundancy in case of satellite outage, can be obtained by supplementing the equatorial orbit satellites with synchronous satellites in highly inclined orbits. A trade-off study was conducted to determine the optimum inclined orbits that would permit maximum continuous coverage and accuracy for a minimum number of satellites. Included in this study were the so-called Y and X orbit configurations treated in the literature (Reference 1). Since satellite-derived altitude over the North Atlantic, CONUS, and Europe was an important consideration, it was deemed desirable to have at least one satellite at all times over the North Atlantic to give good geometry for altitude determination for the CONUS and Europe, as well as give good near polar communication coverage and good North-South position determination in the equatorial regions.

The inclined synchronous orbit selected has a pear-shaped ground track over the North and South Atlantic with apogee occurring at the apex of the pear, at 52.5°N latitude and 35° W longitude, just halfway across the Atlantic from Nova Scotia to London. The orbit is elliptic with 0.35 eccentricity and 52.5 inclination. In its 24-hour period, a satellite in such an orbit will spend 12 hours in the top quarter of the pear, with its subsatellite point remaining throughout this time above 28°N latitude. If a second satellite with the same ground track is phased 12 hours behind the first one, then the coverage objectives described in the previous paragraph can be achieved with two satellites.

Two satellites with the same ground tracks but 12 hours apart in phasing are actually in different orbit planes, with the two planes rotated 180 degrees relative to each other. Clearly, these require separate launches. If an additional satellite were placed in each of these orbit planes 180 degrees from the original satellite so that there is now a total of four satellites, or two satellites 180 degrees apart per orbit plane, then there would be an additional pear-shaped ground track over the western Pacific with apogee at 52.5° N latitude, 145° E longitude. If desired, these four satellites can be deployed in two launches. With the four inclined synchronous satellites augmenting the nine equatorial synchronous satellites, continuous communication coverage and high accuracy three-axis position determination for all presently anticipated world navigation routes are obtained. The ground traces for all thirteen satellites are shown in Figure 58. The coverage plots for 10 degree minimum elevation angle for the nine synchronous equatorial satellites is given in Figure 59.

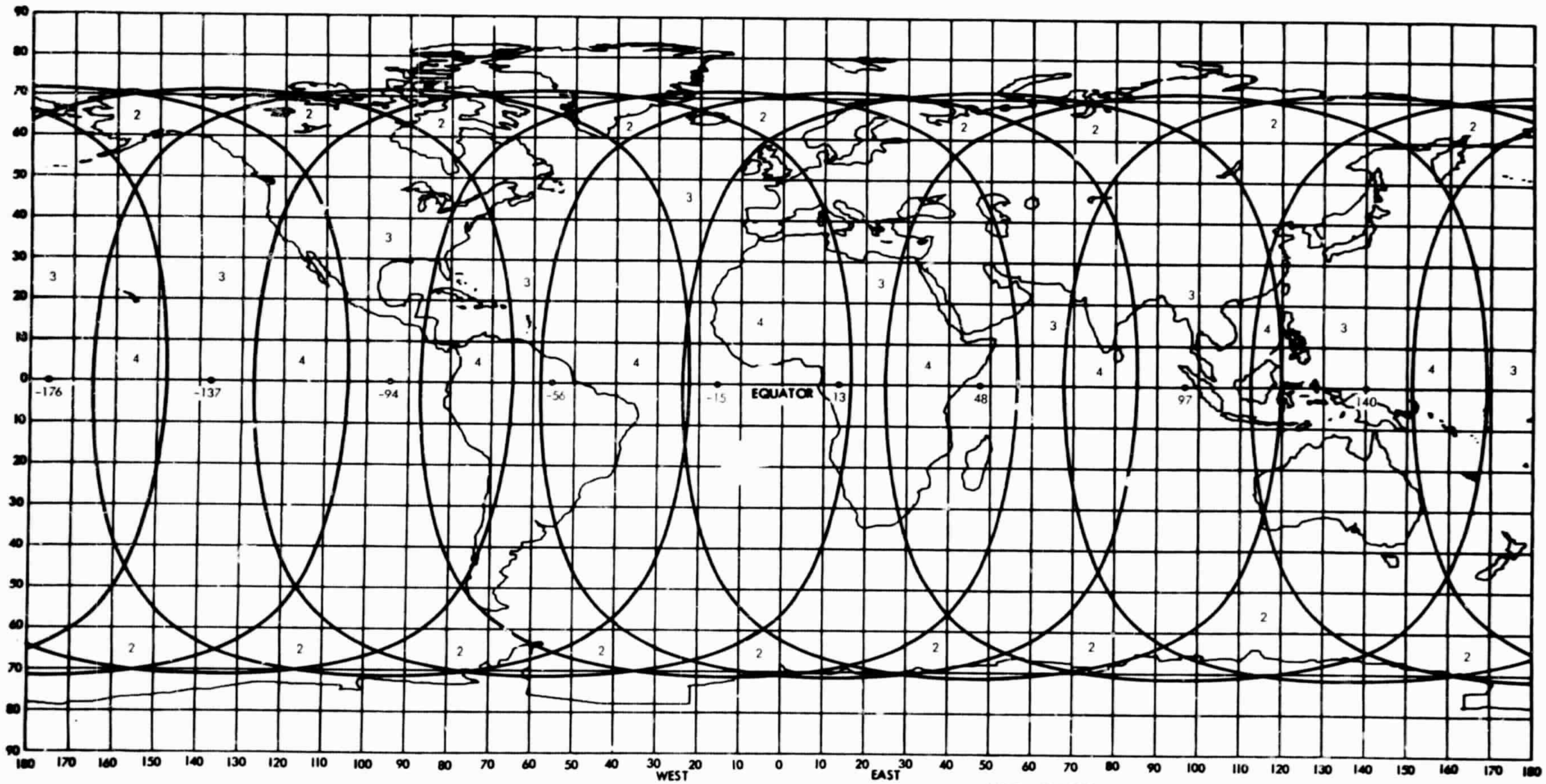
4.3 COVERAGE

The worldwide NTC Satellite System of 13 satellites has considerable redundancy built into it in case of satellite outage. Fror the coverage plots of Figure 59, it is apparent that the positions of the equatorial satellites were chosen so that most of the major world aeronautical terminal areas such as the U.S. East Coast and West Coast, Western Europe, the West and East Coasts of South America, the Eastern Mediterranean, India, Hong Kong, the Hawaiian Islands, etc., have four equatorial satellite coverage, so that in case of one satellite failure, X-Y position determination using only equatorial satellites continues to be possible in the areas normally having only three equatorial satellite coverage, in the event that one is lost, because of the existence of the inclined orbit satellites. For these regions, altitude information from satellite data may not be available, however,



NOTE: NINE CIRCULAR SYNCHRONOUS EQUATORIAL AND FOUR ECCENTRIC SYNCHRONOUS INCLINED). HEAVILY SHADED AREA OF ECCENTRIC ORBIT GROUND TRACES INDICATES REGION ABOVE WHICH THERE WILL ALWAYS BE AT LEAST ONE SATELLITE. NUMBERS ON PLOTS INDICATE HOURS FROM APOGEE. OTHER NUMBERS ON EQUATOR INDICATE LONGITUDE POSITIONS OF EQUATORIAL SATELLITES.

Figure 58. Satellite Ground Traces for Worldwide System



NOTE: THE TOTAL NUMBER OF SATELLITES VISIBLE ARE INDICATED BY THE NUMBERS 8, 6, OR 4 MARKED IN EACH AREA

Figure 59. Ten Degree Elevation Angle Coverage Plots for Nine Circular Synchronous Equatorial Satellites

for certain periods of the day. With regard to a possible failure of an equatorial satellite, it might be noted that in building up the worldwide system some of the early configurations of the satellites will be retired as larger capacity configurations become available. These earlier satellites, which would be dormant, can become spares. Since all equatorial satellites are in the same orbit plane, these spare satellites can be maneuvered over a period of days or weeks depending on how far they have to go and the remaining fuel on board, in order to fill the vacancy created by a failed satellite. For the total start and stop maneuver, it takes approximately 20 feet/second of inplane velocity increment for each degree per day of longitude motion.

In case of failure of an inclined orbit satellite, the phasing of the other inclined satellites might be changed to optimize the position of the remaining satellite to cover the 12 to 15 hours of prime traffic flow in the particular region (Atlantic or Pacific), where the outage occurred. Furthermore, while waiting for a replacement launch, a satellite from the Pacific ground trace could be moved to the Atlantic ground trace, if the more important Atlantic area experienced the outage. It should be remembered that from the standpoint of navigation, a loss of an inclined satellite will generally only affect altitude determination. The North-South position determination capability in the equatorial regions afforded by the inclined satellites will not be seriously compromised by the loss of one satellite, (except for two short equatorial crossing periods), since, with a 52.5° inclination and 0.35 ellipticity, both satellites in a given ground track will at all times be visible in their equatorial coverage areas.

4.4 SATELLITE CONSTELLATIONS

The proposed sequence of launches to develop various satellite constellations up to a 13-satellite configuration worldwide system from an initial 2-satellite D and D coverage of the North Atlantic is detailed in Table 35. In the paragraphs below, each of these satellite constellations is outlined in terms of satellite locations and the consequent coverage and accuracy improvement increments achieved by each new system addition. The accuracy numbers given are the results of an extensive

Constellation Number and Designation	Number of Satellites Launched (Cumulative Number of Satellites Launched in Parentheses)		Major New Coverage Areas (Position Determination)	Number of Axes of Navigational Data Available in New Coverage Area
	Equatorial	Inclined		
1. D and E/Preoperational North Atlantic (Phase I)*	2		North and South Atlantic including U.S. East Coast and W. Europe	1**
2. Initial Operational North Atlantic (Phase II)	1 (3)		Three satellite coverage of North and South Atlantic including U.S. East Coast and W. Europe	2
3. Initial Operational Western World (Phase III)	1 (4)		Four satellite coverage of mid-Atlantic	2
4. Operational Western World (Phase III)	(4)	2	Minimum of four satellite coverage of all of North Atlantic, three satellite coverage of Western U.S., Western Europe, and North Africa	3 (N. Atlantic) 2 (Western U.S., Europe)
5. Operational U.S. and Western World (Phase III)	2 (6)	(2)	Minimum of four satellite coverage of all of U.S.; three satellite coverage of East Pacific including Hawaii and Alaska	3 (CONUS) 2 (Hawaii, Alaska)
6. Initial Operational Worldwide (Phase III)	2 (8)	(2)	Minimum of three satellite coverage of all of Pacific including Hong Kong, Australia, Viet Nam; also Middle East and Eastern Europe	2
7. Operational Worldwide (Phase III)	(8)	2 (4)	Minimum of four satellite coverage for all of Pacific	3
3. Ultimate Operational Worldwide	1 (9)	(4)	Minimum of four satellite coverage for all of Europe and Central Asia	3

*Program phases are defined in Paragraph 5.2.1, this volume.

**Note: 2 axes (X-Y) position determination possible if accurate clock is carried by user and synchronized prior to flight (see discussion).

Table 35. Summary of Launch Sequences and Coverage Areas

accuracy analysis in which typical user measurement uncertainties and satellite ephemeris uncertainties were postulated. Summary plots of the accuracies achievable worldwide in latitude, longitude, and altitude for the 9 equatorial satellite case and for the full 13 satellite case are given in Paragraph 4.4.9.

4.4.1 Constellation No. 1 (D and D/Preoperational, North Atlantic, Phase I)

Two satellites, circular synchronous equatorial:

a) Satellites parked over 56°W and 15°W longitude

b) Coverage provided by both satellites from U.S. East Coast to London/Paris with 10 degrees minimum elevation angle. Southern half of Greenland and Iceland is also covered.

c) With high accuracy ($1:10^{10}$) aircraft stable oscillator (one-way range solution) and zero set at takeoff, provides the following user CEP from New York to Paris:

- End of first hour: ~500 ft
- End of third hour: ~1000 ft
- End of sixth hour: ~1900 ft

(assuming 500-feet barometric altitude and 20-n mi initial user latitude and longitude uncertainties).

d) With low accuracy aircraft stable oscillator (range difference solution), and with latitude known initially to 10 n mi:

- Longitude is determined to ~4000 feet in mid-ocean (35°W longitude), and ~5 n mi at New York and Paris.

e) With latitude initially known to be 1 n mi:

- Longitude is determined to ~500 feet in mid-ocean and ~0.5 n mi at New York and Paris.

4.4.2 Constellation No. 2 (Initial Operational, North Atlantic, Phase II)

Three satellites circular synchronous equatorial:

a) Above plus third satellite parked at 35°W longitude to give satellites at:

- 56°W longitude
- 35°W longitude
- 15°W longitude

b) Total coverage same as Launch Sequence No. 1 (above) except all three satellites cover area indicated.

c) With low accuracy aircraft stable oscillator (range difference solution), provides CEP from New York to Paris of

Assuming 500-feet barometric altitude error ~550 feet

Assuming 200-feet barometric altitude error ~440 feet.

d) High accuracy aircraft stable oscillator and zero set at takeoff provides no significant improvement over above values, as evidenced by CEP from New York to Paris with $1:10^{10}$ stable oscillator (assuming 500-feet barometric altitude error):

- End of first hour: ≈400 feet
- End of third hour: ≈500 feet
- End of sixth hour: ≈550 feet

Accuracy is only mildly time-dependent. A good clock still allows the use of the third satellite in a redundant, one-way ranging sense.

e) Most of above CEP errors comprise latitude errors; one sigma longitude error runs from about ~70 feet in mid-ocean to ~250 feet at New York and Paris.

f) Relative position accuracy CEP between users for collision avoidance:

- Assuming 200-feet barometric altitude disparity between users ≈310 feet
- Assuming 100-feet barometric altitude disparity between users ≈280 feet.

4.4.3 Constellation No. 3 (Initial Operational, Western World, Phase III)

Four satellites circular synchronous equatorial:

a) Above plus fourth satellite parked at 56°W longitude; then first satellite moved from 56°W to 94°W longitude; and third satellite* moved from 35°W to 13°E longitude to give four satellites at:

- 94°W longitude 15°W longitude
- 56°W longitude 13°E longitude

*Note: Two comments are in order here:

- It is not clear that this satellite would be physi·ally "moved." The ability to move the satellite r . ıly is not incorporated into the present design.

- It will probably prove desirable to keep a satellite stationed at this point in order to provide adequate communications capacity as the customer load builds up. In fact, it appears that if this is not done, the 95°W and 13°E satellites will have to be moved toward the Greenwich meridian a few degrees in order to provide this coverage.

b) Coverage provided by three satellites (at improved spacing relative to Sequence No. 2) from U.S. East Coast to London/Paris with 10 degrees minimum elevation angle. Portion of mid-Atlantic has four satellite coverage. Small notch near Greenland, north of US/Europe great circle routes, has two satellite coverage. This can be removed if necessary by moving two outlying satellites inward (but will result in some loss of accuracy). Above geometry also gives two satellite coverage for most of U.S., except Pacific Northwest, and for all of Middle East and Eastern Europe.

c) With low accuracy aircraft stable oscillator (range difference solution), provides CEP from New York to Paris of:

- Assuming 500-feet barometric altitude error ~350 feet
- Assuming 200-feet barometric altitude error ~185 feet

d) High accuracy aircraft stable oscillator and zero set at takeoff provides no significant improvement over above values, as evidenced by CEP from New York to Paris assuming 500-feet barometric altitude error:

- End of first hour: ≈330 feet
- End of third hour: ≈350 feet
- End of sixth hour: ≈350 feet

e) Most of CEP error due to latitude errors; one sigma longitude error running about 50 feet over whole span from New York to Paris.

f) Relative position accuracy CEP between users for collision avoidance:

- Assuming 200 feet barometric altitude disparity between users ≈160 feet

- Assuming 100 feet barometric altitude disparity between users ≈110 feet

4.4.4 Constellation No. 4 (Operational, Western World, Phase III)

Six satellites, four circular synchronous equatorial, two elliptic synchronous inclined:

a) Above (four equatorial) plus fifth satellite in 52.5° inclined, 0.35 eccentricity orbit with apogee over 52.5°N latitude, 35°W longitude (subsatellite point). First launch is timed to provide high accuracy coverage from 1800 to 0700 GMT time (covering high traffic periods). Sixth satellite is launched into orbit plane rotated 180° from fifth satellite giving same ground track as fifth satellite but with 180 degrees phase difference.

b) Total coverage for equatorial satellites same as in previous sequence. Coverage for inclined satellites varies with time, but one satellite will always be over North Atlantic above 28°N latitude. All of U.S. and all of Europe and Middle East will always be covered by at least one inclined, and for at least 8 hours per day will be covered by both inclined satellites.

c) CEP from New York to Paris:

- ~65 feet (essentially independent of barometric altitude error, satellite oscillator stability, and instantaneous inclined orbit configuration).

d) Altitude error from New York to Paris assuming 500-feet barometric altitude error:

- ~80 to ~130 feet depending on time of day.

e) Relative position CEP between users essentially same as above

- ~60 feet.

f) If one equatorial satellite fails, then system should be returned to Configuration 2 geometry with three satellites on equator at 56°W longitude, 35°W longitude, and 15°W longitude, supplemented by two inclined orbits. Under these conditions, CEP degrades to

- ~130 to ~210 feet depending on time of day

Altitude error degrades to

- ~150 to ~330 feet, assuming 500-feet barometric altitude error.

4.4.5 Constellation No. 5 (Operational U.S. and Western World, Phase III)

Eight satellite configuration, six circular synchronous equatorial and two eccentric synchronous inclined:

a) Above coverage plus two additional synchronous equatorial satellites parked over 137°W longitude and 177°E longitude.

b) Above coverage ext :nded to provide three equatorial satellite coverage of all of U.S. west to Hawaii and part of Alaska. Also, at least one inclined satellite will at all times cover the continental U.S. and Alaska.

c) CEP for U.S. and great circle route from Los Angeles/ San Francisco to Hawaii:

- ~50 feet (essentially independent of barometric altitude error or instantaneous inclined orbit configuration)

except for region around Hawaii where inclined orbits over Atlantic are not visible. Here CEP is:

- Assuming 500 feet barometric altitude error ~900 feet
- Assuming 200 feet barometric altitude error ~400 feet

(primarily latitude error; longitude error at Hawaii ≈30 feet).

d) Altitude error for U.S. and great circle route from Los Angeles/ San Francisco to Hawaii:

- ~50 to ~125 feet (independent of barometric altitude error or instantaneous inclined orbit configuration, but varies with position; worst case over Central U.S.)

except for region around Hawaii where there is no satellite derived altitude information.

4.4.6 Constellation No. 6 (Initial Operational Worldwide, Phase III)

Ten satellite configuration, eight circular synchronous equatorial and and two eccentric synchronous inclined:

a) Configuration five plus two additional synchronous equatorial satellites parked over 140°E longitude and 97°E longitude.

b) Above coverage extended west to provide three equatorial satellite coverage of all of Pacific including Japan, Korea, Hong Kong, Philippines, Australia, Viet Nam. Also provides three equatorial satellite coverage of all of Middle East and Eastern Europe.

c) CEP for great circle route Los Angeles/San Francisco to Tokyo:

- Assuming 500 feet barometric altitude error ~490 feet
- Assuming 200 feet barometric altitude error ~240 feet

d) No satellite-derived altitude over Pacific except for region East of Hawaii (see Launch Sequence No. 5).

4.4.7 Constellation No. 7 (Operational Worldwide, Phase III)

Twelve satellite configuration, eight circular synchronous equatorial, and four eccentric synchronous inclined:

a) Above plus two inclined orbits in same two orbit planes as Configuration 4 but phased 180 degrees relative to satellites already in those planes. Provides a ground track with apogee over 52.5°N latitude, 145°E longitude.

b) Provides high accuracy coverage (four or more satellites in good geometry) over most of Northern Hemisphere except for polar region, West Europe, and Central Asia including India. Good coverage also of Southern Hemisphere, but geometry permits good altitude determination for only about half of Southern Hemisphere in a band 0 to 45°S latitude.

c) CEP for Los Angeles/San Francisco to Tokyo great circle route:

- ~50 feet (essentially independent of barometric altitude error or instantaneous inclined orbit configuration)

d) Altitude error for Los Angeles/San Francisco to Tokyo great circle route, assuming 500-feet barometric altitude error,

- ~50 to ~140 feet (depending on longitude and instantaneous inclined orbit configuration; only larger values are slightly affected by barometric altitude error).

4.4.8 Constellation No. 8 (Ultimate Operational Worldwide, Phase III)

Thirteen satellite configuration, nine circular synchronous equatorial plus four eccentric synchronous inclined:

a) Above plus an additional synchronous equatorial satellite at 48°E longitude; satellite at 177°E longitude is then moved to 176°W longitude.

b) Coverage as in previous sequence but including high accuracy coverage (over four satellites at good geometry visible at all times) of Western Europe, North Africa, and Mid-Asia, including India. Small lapses of coverage occur in two equatorial regions around 120°W longitude and 60°E longitude during some periods of the day.

c) CEP for all of Northern Hemisphere 0 to 50°N latitude and most of Southern Hemisphere 0 to 30°S latitude (except for regions indicated above)

- ~30 to ~65 feet (depending on location but essentially independent of barometric altitude error and instantaneous inclined orbit configuration).

d) Altitude error for same area

- ~50 to ~130 feet (same comment as above)

e) CEP for Northern and Southern Hemispheres using data from nine equatorial satellites only, assuming 500-feet barometric altitude error:

- ~1250 feet at ±15° latitude
- ~600 feet at ±30° latitude
- ~400 feet at ±45° latitude
- ~330 feet at ±60° latitude (except for regions where three satellites are not visible)

Assuming 200 feet barometric altitude error

- ~550 feet at ±15° latitude
- ~300 feet at ±30° latitude
- ~250 feet at ±45° latitude
- ~250 feet at ±60° latitude (except for regions where three satellites are not visible)

(most of these errors are latitude errors; longitude errors ≈50 feet).

There is an alternate configuration to Constellation No. 8 for worldwide coverage. This alternate requires only 12 satellites (8 circular synchronous equatorial and 4 eccentric synchronous inclined), but this would require moving all of the equatorial satellites in Constellation 7 to plug up holes in the coverage of Western Europe and South Central Asia. This does not appear to be feasible for an operational navigation satellite system. Conceptually, the 12-satellite system could be achieved, however, if the sequence of satellites launched were intended to first cover the U.S., then the North Atlantic, Europe and Western Asia, Eastern Asia, Central Asia, and finally the Pacific area.

It should also be pointed out that although Constellation No. 8 is called a worldwide system, there is a considerable area of the globe, primarily around Antarctica, the North Pole, and some small equatorial ocean areas in the Pacific and Indian Ocean, which will not continuously be covered by at least three satellites at good geometry. The polar routes from U.S. to Scandinavia will be adequately covered without the additional satellite coverage. Additional satellites, primarily at high inclination, would be required to cover these areas, if, which seems quite unlikely, commercial aviation transits these areas in the foreseeable future and establishes a need for high accuracy navigation/traffic control and operational support services in that region.

4.4.9 Performance Summary

Inspection of Figures 60 through 68 yield some interesting and significant points with regard to Navigation/Traffic Control Satellite System performance.

- Figures 60, 62, and 66 indicate that the latitude error which, for the North Atlantic case, corresponds to cross track error is typically several times the longitude error but both are well below a 1 n mi figure associated with air traffic control surveillance, even for a simplified system, which could permit an order of magnitude accuracy degradation. Figure 66 also indicates the loss of latitude information near the equator for the system without inclined elliptic orbits.

- Figures 61 and 63 indicate that the inclined elliptic orbits provide excellent altitude information. Accuracy could not be degraded by an order of magnitude here, however, and still serve as an accurate altitude surveillance check.

- Figures 61 and 63 also show the excellent coverage in terms of the number of satellites visible for the ultimate operational configuration.

- Figure 62 versus 60 and Figure 63 versus 61 indicate that the accuracies obtainable with this system do not vary significantly as the inclined elliptic orbital satellites trace out their patterns.

- Figures 64 and 65 show the extremely good relative position determination capability that this system offers. This point is very significant if the system is to be used in a collision avoidance or stationkeeping mode.

- If it is determined that altitude information, polar communication coverage, and accurate X-Y navigation near the equator are not requirements, then Figures 66, 67, and 68 show the attractiveness of a fully synchronous equatorial system, i.e., a system without the inclined elliptic orbits.

The assumptions made in this analysis are listed in Table 36.

Table 36. **Assumptions in Error Analysis**

	Absolute Navigation (ft)	Relative Navigation: Aircraft Case (ft)	Relative Navigation: Ship or Land Case (ft)
Satellite position uncertainties:		0	0
σ_{Radial} =	20		
$\sigma_{Intrack}$ =	125		
$\sigma_{Crosstrack}$ =	75		
User measurement uncertainties:			
σ_M =	40	25*	20**
User initial position uncertainties:			
$\sigma_{Altitude}$ =	500	100	5
$\sigma_{x\ position}$ =	1,000,000	1,000,000	1,000,000
$\sigma_{y\ position}$ =	1,000,000	1,000,000	1,000,000

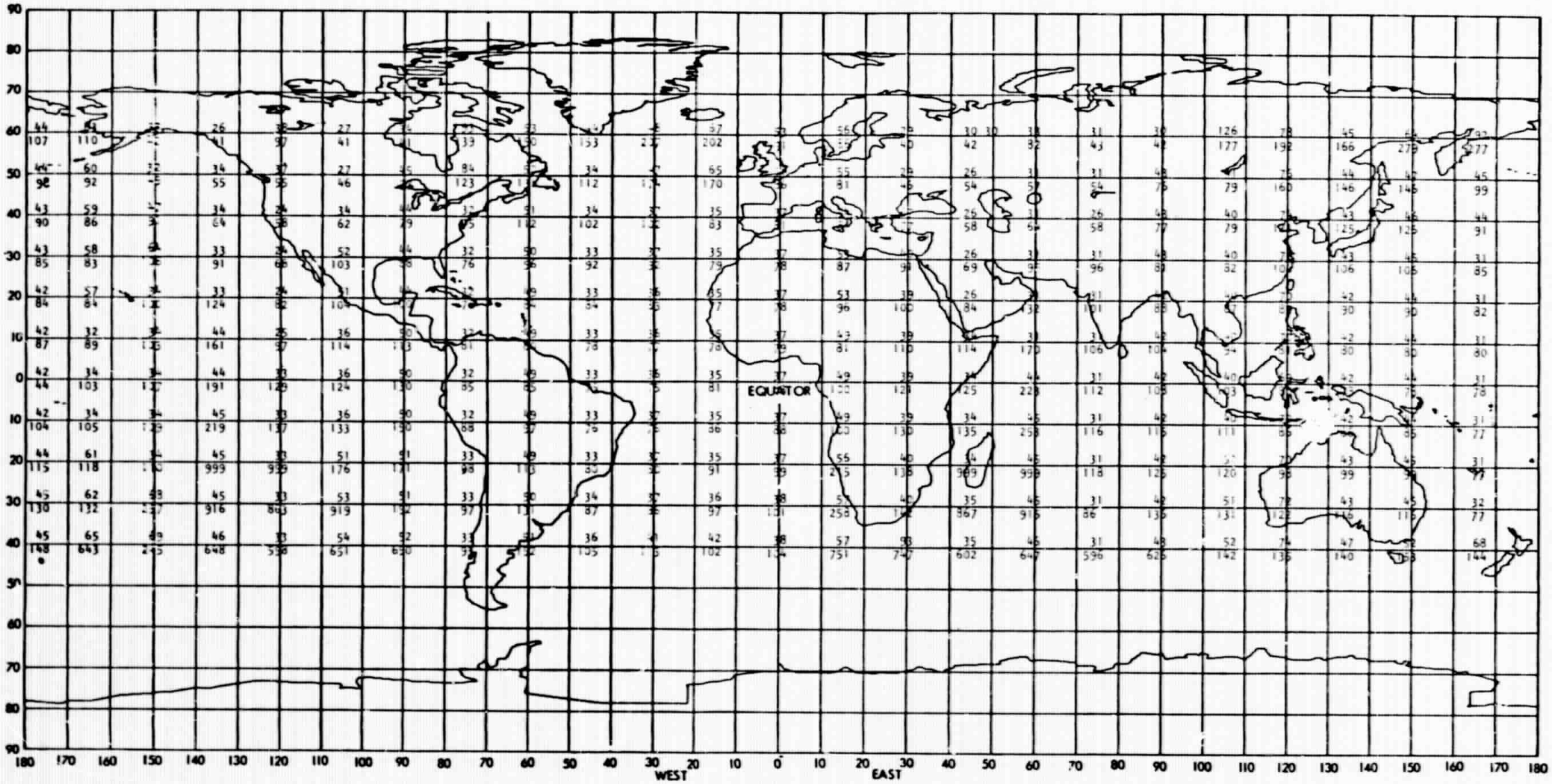


Figure 60. Absolute Navigation Accuracies Given in Terms of One Sigma Longitude Error (Top) and One Sigma Latitude Error (Bottom) for 13-Satellite Operational World-Wide Configuration (with Inclined Satellites at Apogee ±6 Hours)

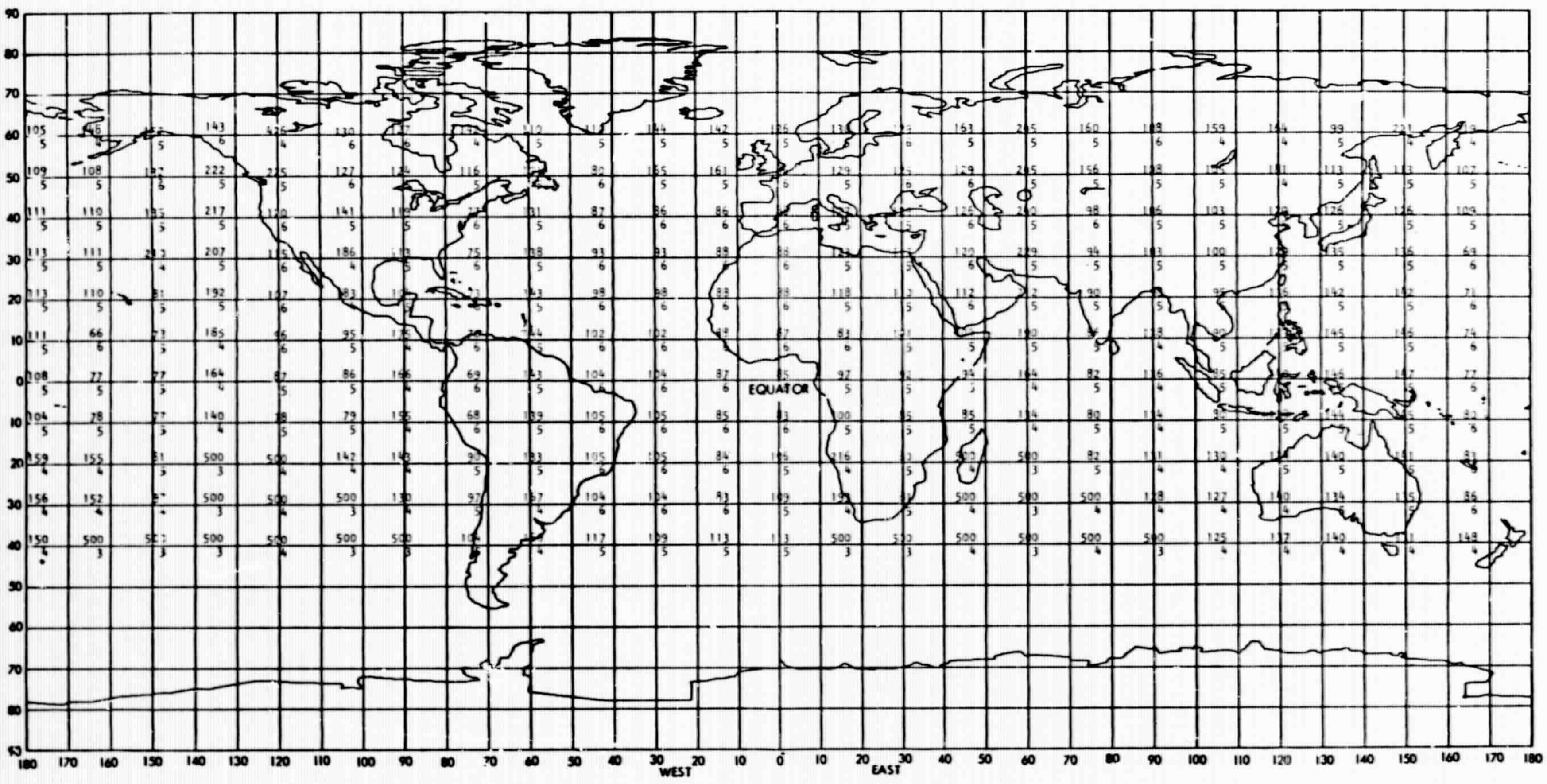


Figure 61. One Sigma Altitude Determination Accuracy (Top) and Number of Satellites Visible (Bottom) for 13-Satellite Operational World-Wide Configuration with Inclined Satellites at Apogee ±6 Hours

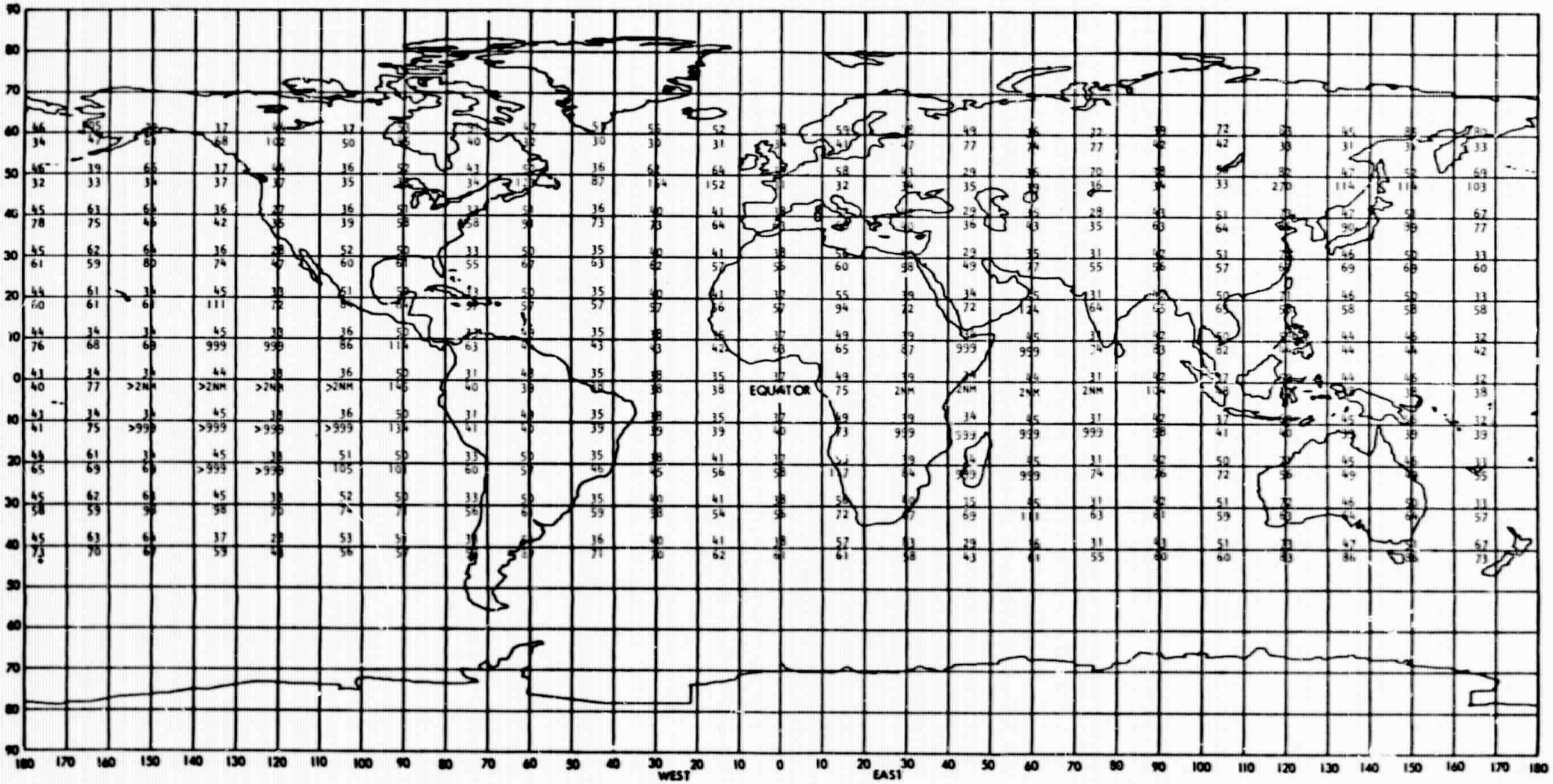


Figure 62. **Absolute Navigation Accuracies Given in Terms of One Sigma Longitude Error (Top) and One Sigma Latitude Error (Bottom) for 13-Satellite Operational World-Wide Configuration (with Inclined Satellites at Apogee or Perigee; i.e., Figure 60 ±6 Hours)**

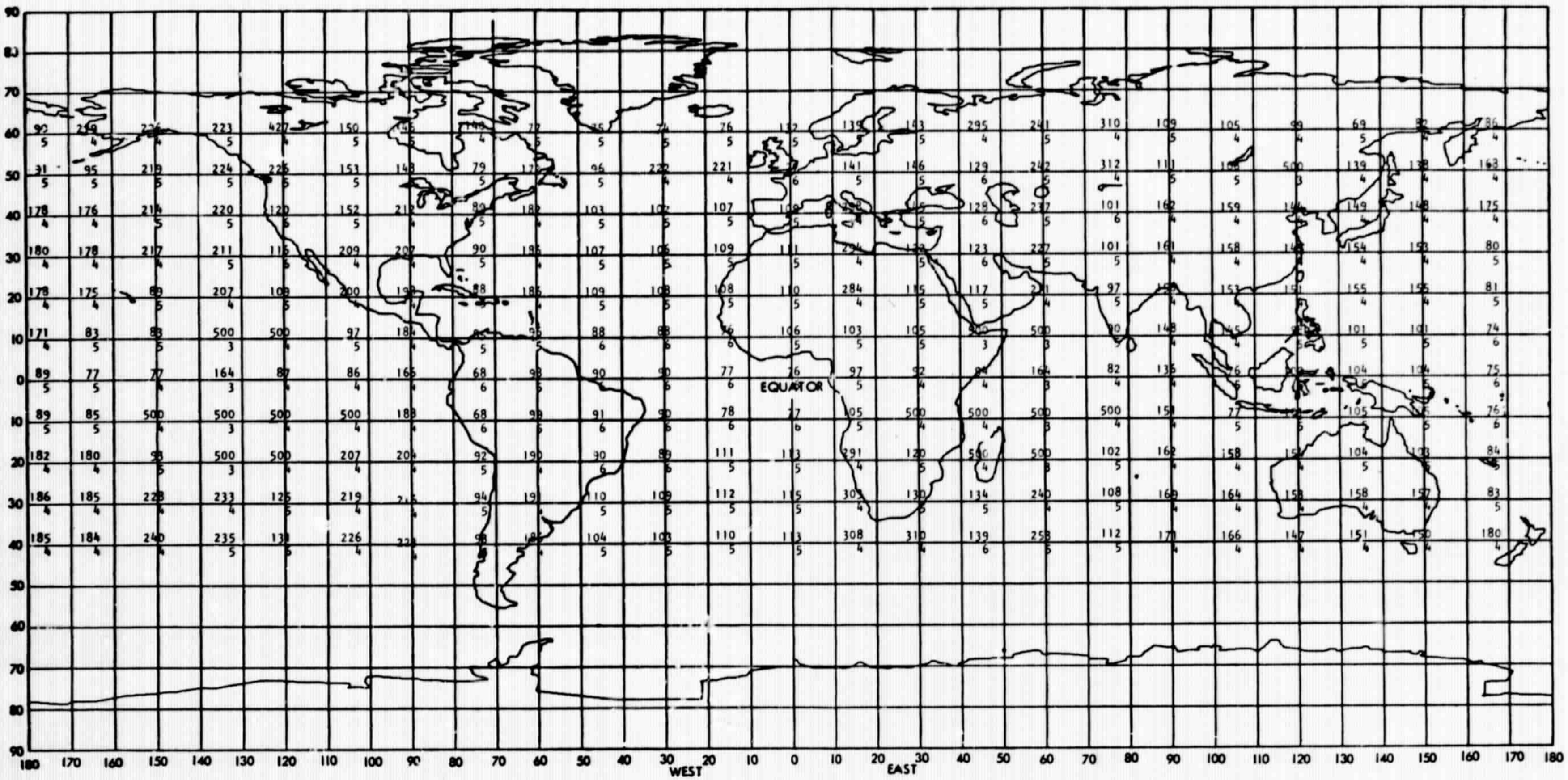


Figure 63. One Sigma Altitude Determination Accuracy (Top) and Number of Satellites Visible (Bottom) for 13-Satellite Operational World-Wide Configuration (with Inclined Satellites at Apogee or Perigee; i. e., Figure 61 ±6 Hours)

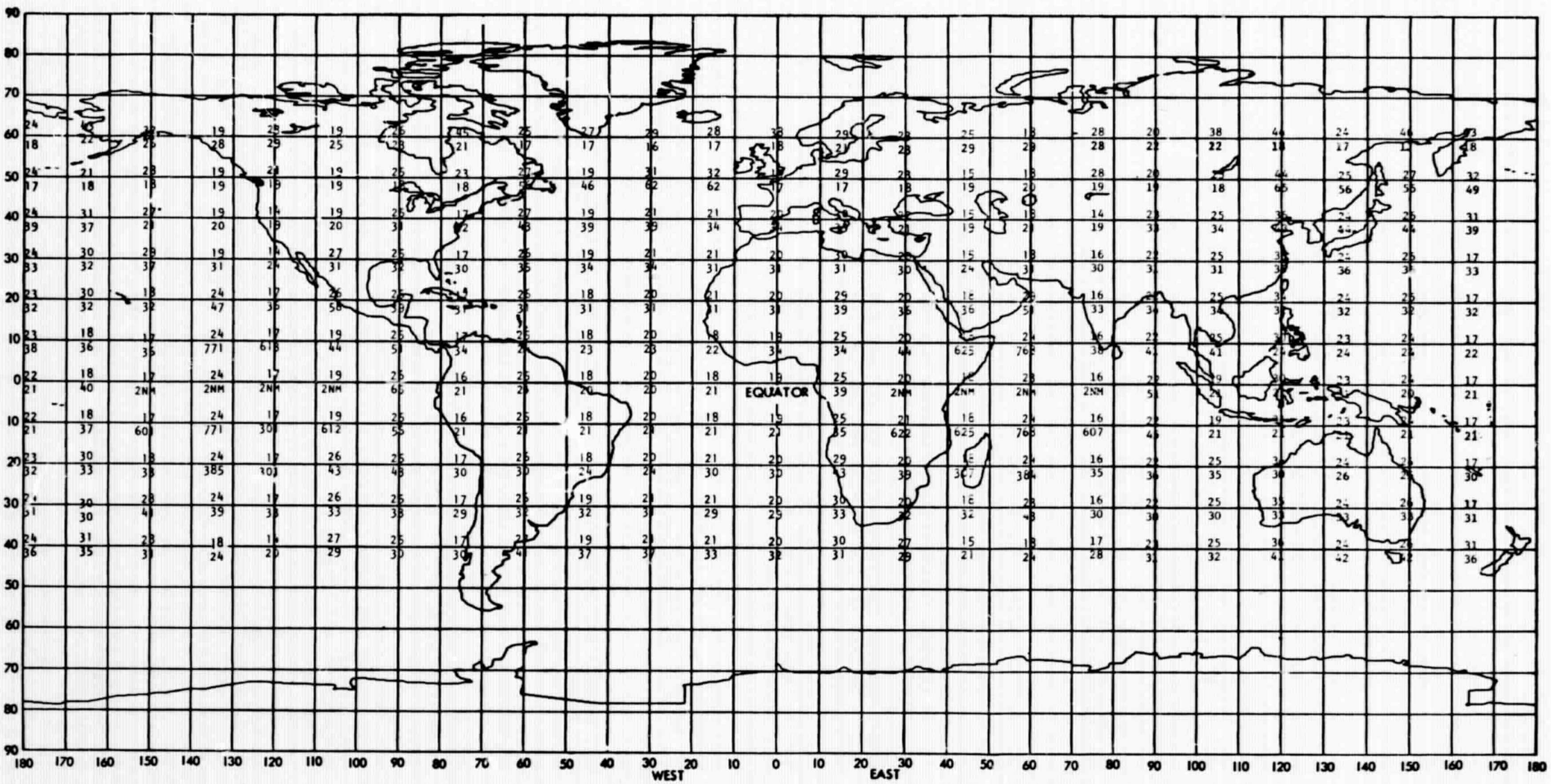


Figure 64. Relative Aircraft Navigation Accuracies Given in Terms of One Sigma Longitude Error (Top) and One Sigma Latitude Error (Bottom) for 13-Satellite Operational World-Wide Configuration (with Inclined Satellites at Apogee or Perigee; i.e., Figure 60 ±6 Hours)

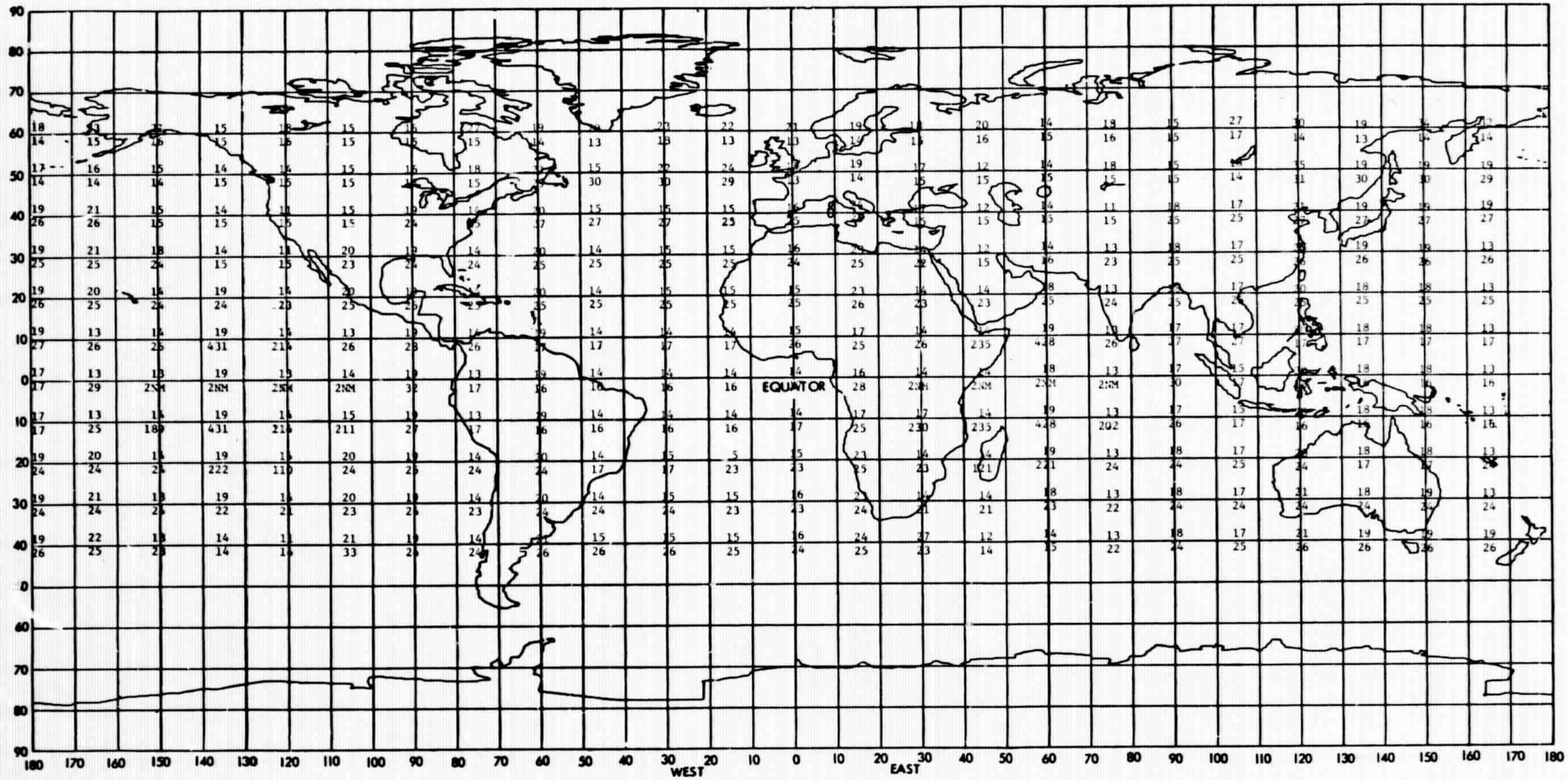


Figure 65. Relative Ship or Land Navigation Accuracies Given in Terms of One Sigma Longitude Error (Top) and One Sigma Latitude Error (Bottom) for 13-Satellite Operational World-Wide Configuration (with Inclined Satellites at Apogee or Perigee; i. e. , Figure 60 ±6 Hours)

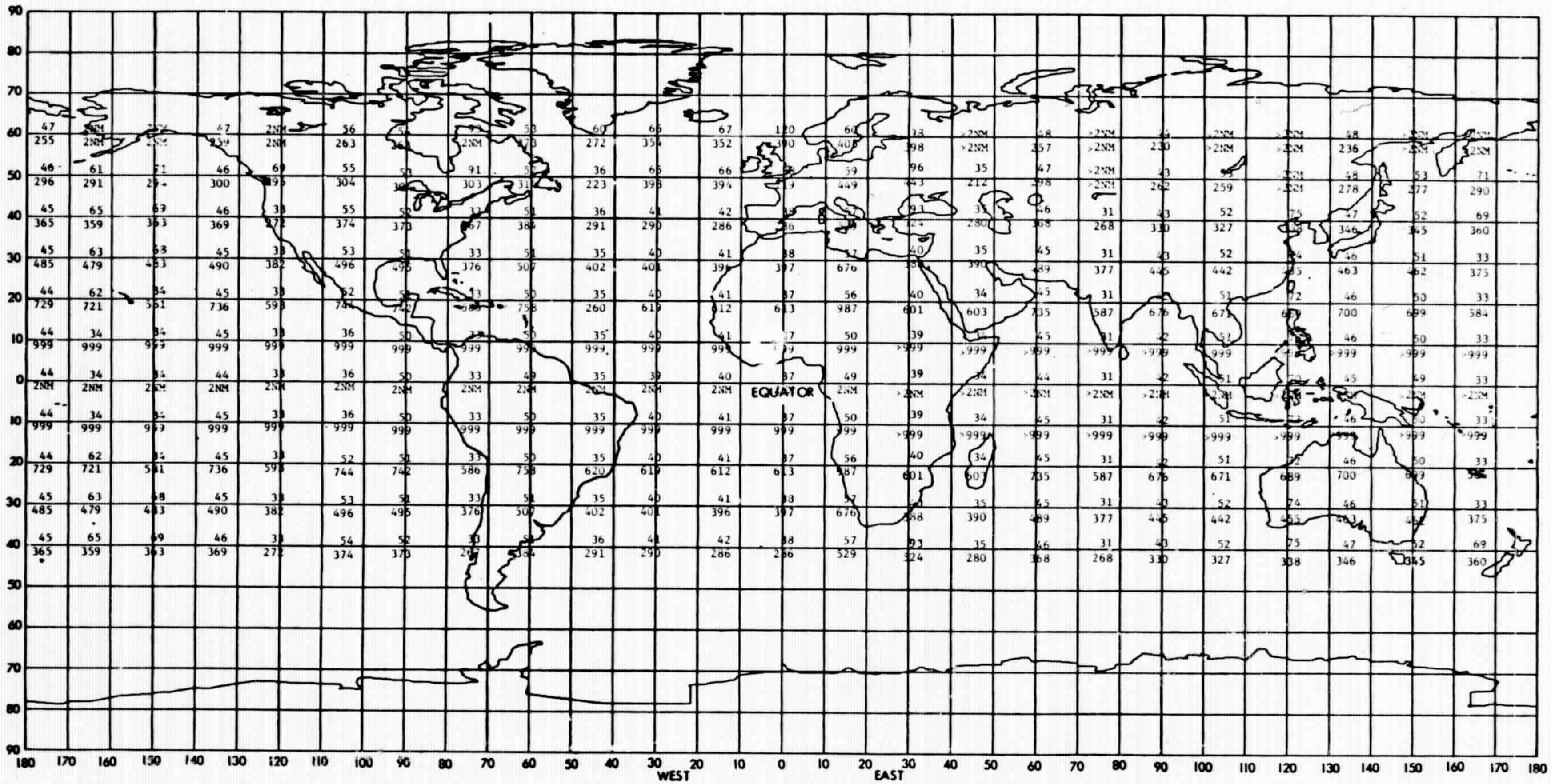


Figure 66. Absolute Navigation Accuracies Given in Terms of One Sigma Longitude Error (Top) and One Sigma Latitude Error (Bottom) for 9-Equatorial Satellite World-Wide Configuration

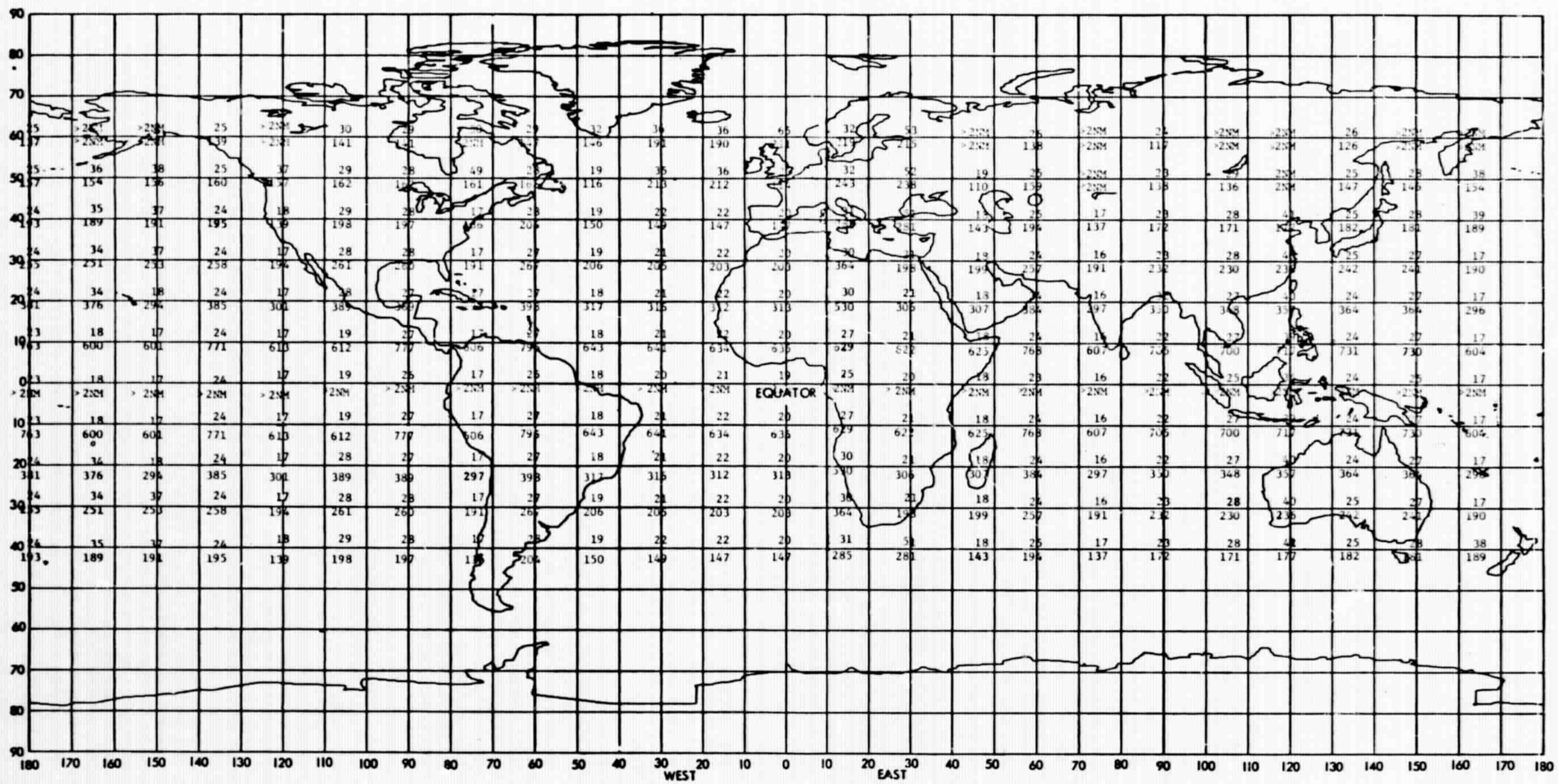


Figure 67. Relative Aircraft Navigation Accuracies Given in Terms of One Sigma Longitude Error (Top) and One Sigma Latitude Error (Bottom) for 9-Equatorial Satellite World-Wide Configuration

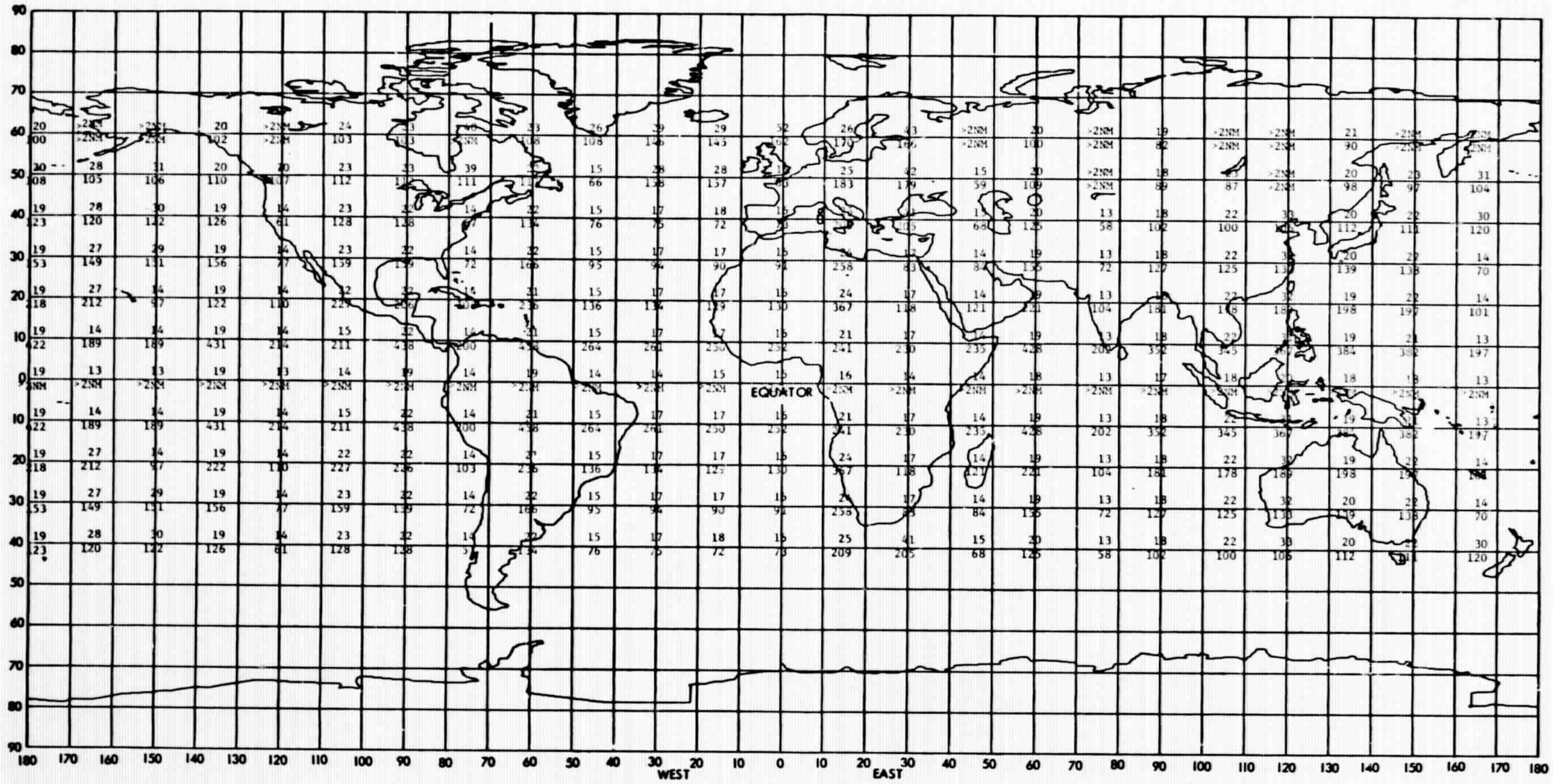


Figure 68. Relative Ship or Land Navigation Accuracies Given in Terms of One Sigma Longitude Error (Top) and One Sigma Latitude Error (Bottom) for 9-Equatorial Satellite World-Wide Configuration

4.5 PSEUDO-STATIONARY CONSTELLATION COMMUNICATION COVERAGE

The communications coverage obtainable with the ultimate worldwide system (Constellation No. 8) is shown in 69. For convenience of notation in Figure 69, the synchronous equatorial satellites to the West of the Greenwich meridian are given even numbers and to the East are given odd numbers. The inclined pair covering the Atlantic are numbered 10 and 11, whereas the inclined pair covering the Far East are numbered 12 and 13. Inspection of the figure shows that the circular 8-degree beam described in Paragraph 2.2.5.1 will give excellent Northern Hemisphere coverage throughout the world. It is not clear, however, that full Northern Hemispherical coverage would make sense for a program sponsored and funded by the United States. Indian Ocean maritime route coverage would clearly be of greater economic benefit to the Western World than USSR/China coverage. Accordingly, coverage indicated by the curves labeled Alt. 3, Alt. 5, Alt. 7 are obtained simply by moving the centerline or aim point of the communications antenna on the No. 3, No. 5, and No. 7 satellites to approximately 2 degrees South, straight down, and approximately 2 degrees North of the subsatellite point, respectively.

The curves labeled 10a, and 11a, show the communications coverage provided by satellites 10 and 11 at the 6- and 18-hour points, whereas the curves 10b and 11b show the coverage 6 hours later. As pointed out in Volume II, Paragraph 4.8.3, the coverage provided by the inclined satellite at apogee is excellent, but for the system as presently configured, the satellite does not provide this coverage for as long as is really desirable. Therefore, the preliminary design phase could well see the inclined satellite inclination and/or eccentricity increasing. (Note: it is also conceivable that the need for this satellite coverage from both a communications and position determination standpoint might not warrant its development and implementation.) Satellites 12 and 13 are shown only at the zero and 12-hour points on the orbit to simplify the chart. The Southern Hemisphere coverage provided by these satellites is only incidental. When the synchronous inclined satellites are near apogee, their coverage area is quite small and they are moving fairly rapidly at this point, the necessary consequence of any eccentric inclined orbit.

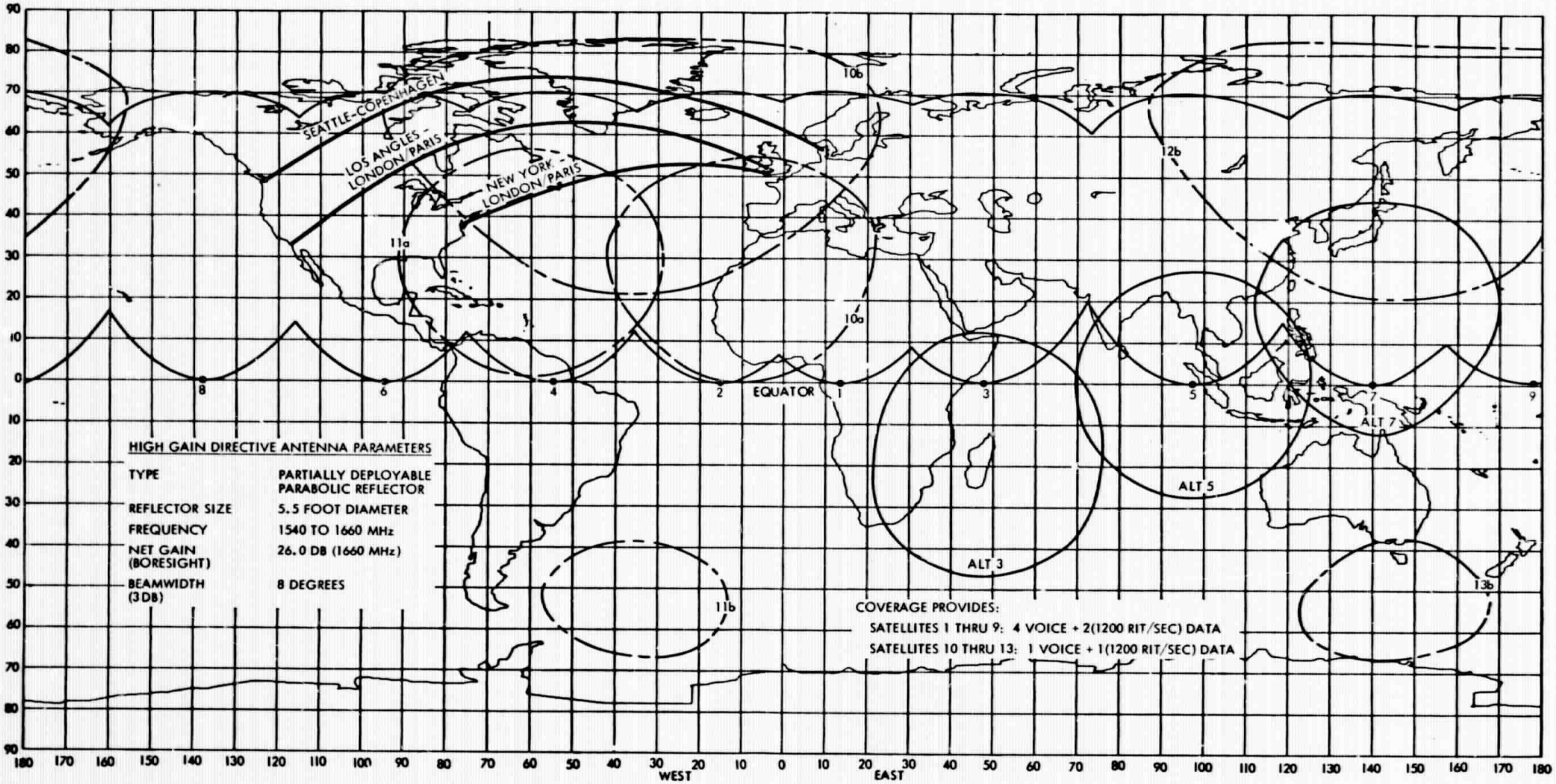


Figure 69. Global NTCS Communications Coverage

4.6 DISCUSSION

The foregoing set of orbital constellations provides an orderly development of the worldwide Navigation/Traffic Control Satellite System, but it should be pointed out that this represents only one of a substantial number of attractive approaches. Another system design concept which was considered has one stationary satellite and three others in elliptical, synchronous inclined orbits, a combination of ellipticity (approximately 0.3) and an inclination (approximately 30 degrees) that results in a ground trace for each of the three satellites that are, roughly, circles about the stationary satellite subsatellite point. The coverage of this "Y-orbit" network is indicated in Figure 4-13. For further discussion of these orbits, see Reference 1. Still other orbital constellations look feasible and attractive for this mission. TRW still considers the synchronous orbits of Reference 2 to have merit. Furthermore, for worldwide coverage, orbits other than synchronous could be used. Putting a satellite up at a lower altitude requires less boost velocity, and the resulting orbits permit centralization of tracking stations and reduce stationkeeping requirements. Coverage, however, is not continuous until the entire network is orbited, and all users must be able to work with the system of satellites that are all in significant relative motion with respect to the user craft.

At the time of final selection of the orbital constellation and satellite design approaches, a number of questions that are beyond the scope of this feasibility study must be resolved. For example, this section has referred throughout to the movement of satellites from one geostationary point to another. If it were operationally desirable to make this movement in a relatively short period of time, e.g., 1 or 2 days, and if the longitude movement were significant, e.g., 30 to 180 degrees longitude, then the satellite design discussed in Sections 2 and 3 would have to be modified. Specifically, once the solar array is deployed, it does not, as presently configured, have the mechanical strength to withstand the accelerations associated with firing the injection motor. If it were deemed satisfactory to move the satellite using longer firing of the

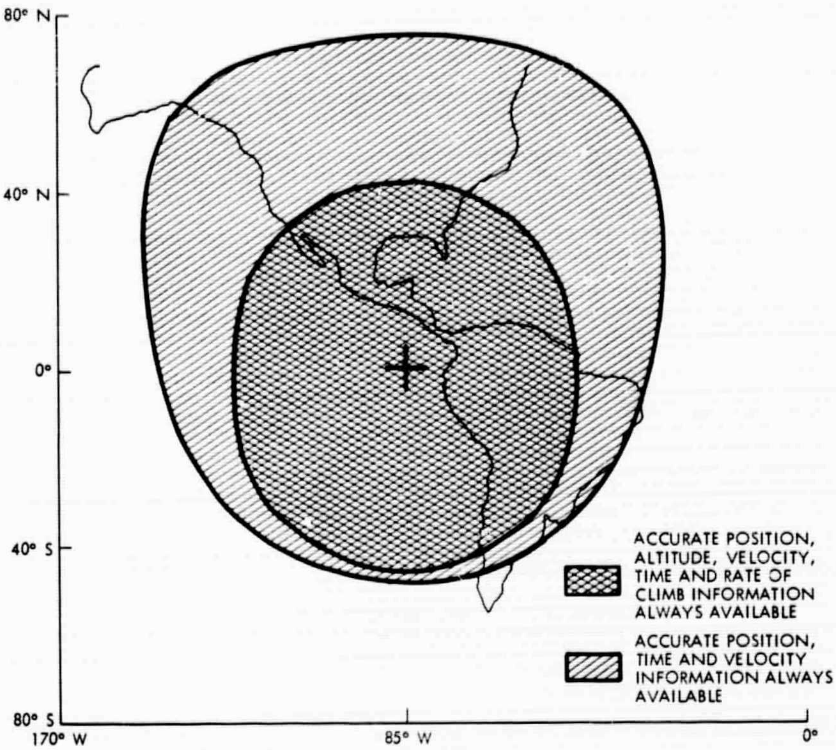


Figure 70. Coverage of Four Satellite "Y" Constellation

stationkeeping rockets, and if the appropriate amount of stationkeeping fuel were designed into the system, then the solar array design as presently conceived is satisfactory. Clearly, in the preliminary design phase, a tradeoff between operational flexibility and system reliability and cost is in order. In summary, when final worldwide coverage is provided, the number of satellites will be between 10 and 18, and the approach recommended here appears to be representative and very attractive.

4.7 REFERENCES

1. R. L. Dutcher, "Parametric Summary - Influences of Synchronous Inclined Elliptical Orbits on Performance of a Satellite Navigation System," Aerospace Corporation Report TOR-0158(3525-14)-1, December 1967.

2. "Study of a Navigation and Traffic Control Technique Employing Satellites," NASA Contract NAS 12-539, TRW Document 08710-6012-R000, December 1967.

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5. PROGRAM PLAN AND COSTS

5.1 INTRODUCTION

This section will present recommendations for a design, development and demonstration program which is required to bring about the operational Navigation/Traffic Control Satellite (NTCS) System capability, and will include the estimated expenditures required. As in the remainder of the study, emphasis is placed on commercial jet aircraft as the primary users of this system, and detailed estimating was limited to this class of users.

This section will outline the recommended program, describe the methodology of cost estimating and present cost estimates for the major design, development, and demonstration program; will emphasize subsystem test programs, and will discuss briefly some recommended NTCS-related systems and technology efforts.

It should be pointed out that there are no real "deficiencies in technology" as was considered likely at the outset of the study. The status of technology today is such that the design, development and demonstration program could be initiated immediately.

5.2 THE NAVIGATION/TRAFFIC CONTROL SATELLITE SYSTEM PROGRAM

5.2.1 Program Phases

The various program phases are defined below and the major milestones are shown in Figure 71.

5.2.1.1 Phase I: Design and Development/Preoperational Program

Phase I consists of the design, development, and demonstration of the basic satellite configuration, the ground control stations, and aircraft user equipment. Major Phase I accomplishments will include the launching of two satellites in July and September of 1972, setting up a temporary Master Control Center at the Federal Aviation Agency's National Aviation Facilities Experimental Center (NAFEC), and two Remote Tracking Stations at Shannon and Gander; development of six sets of user hardware; and the flight test program itself, beginning with the launching of the first satellite, and running for fifteen months to September 1973.

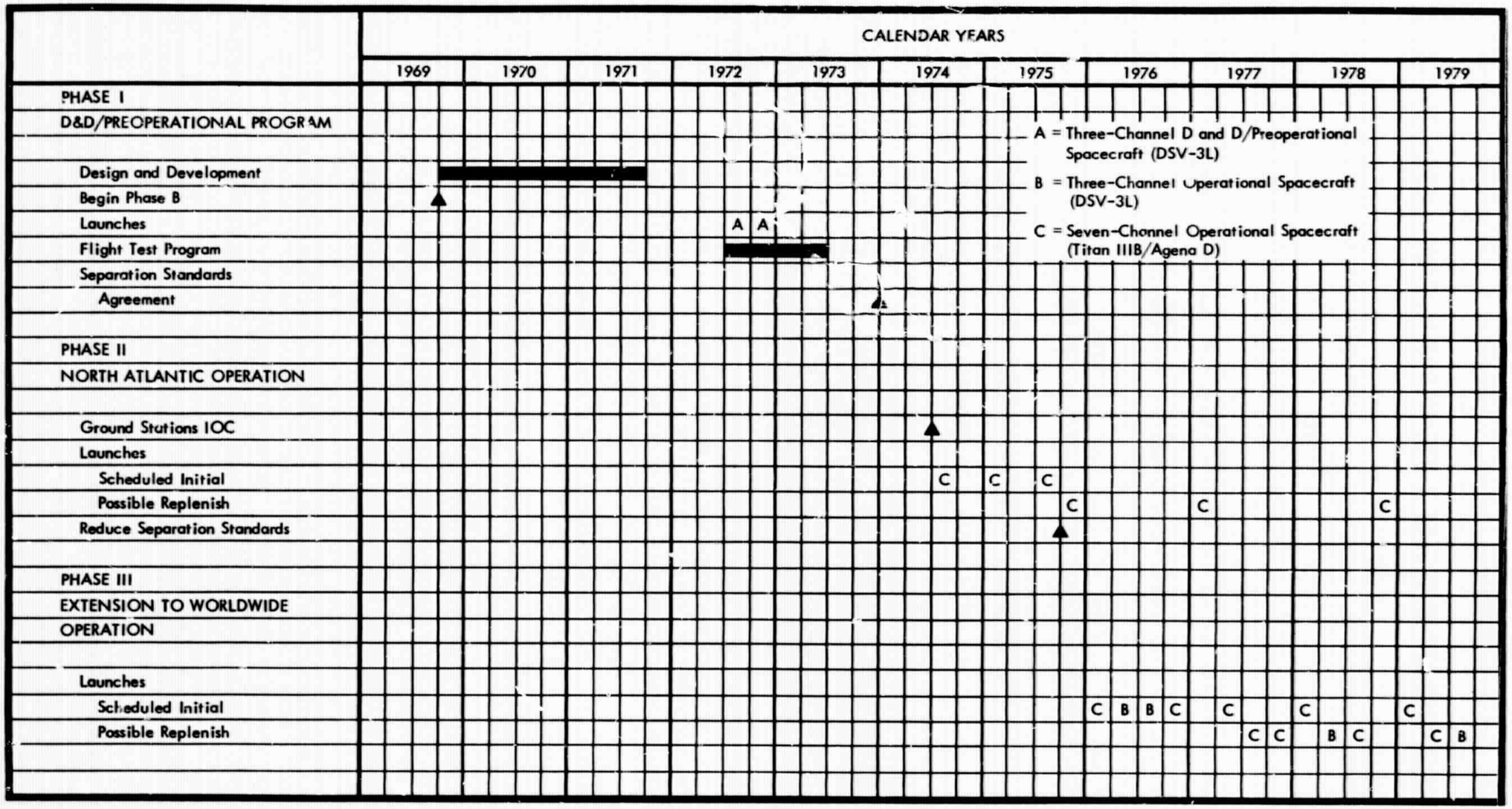

Figure 71. Major Milestones

5.2.1.2 Phase II: North Atlantic Operation

Major accomplishments in Phase II will include qualification of the final satellite configuration. Stationing of three satellites, establishing a dedicated Master Control Station at J.F. Kennedy Airport, and adding an additional Remote Tracking Station on Ascension Island.

5.2.1.3 Phase III: Extension to Worldwide Operation

Phase III involves the stationing of seven additional satellites, and the establishment of two additional dedicated master control stations and three remote tracking stations.

5.2.1.4 Phase Zero: Subsystem Test and Related Systems and Technology Programs

As indicated earlier, the status of satellite and avionics technology is such that no basic research or exploratory development is required in order to advance the state of the art in some area critical to Navigation/ Traffic Control Satellite development.

5.2.2 Program Philosophy

5.2.2.1 Design and Development/Preoperational Program

The D and D program, using the two synchronous equatorial Configuration "A" satellites, was designed with the following major objectives in mind:

1. Development of the Navigation/Traffic Control Satellite System hardware, software, and procedures, in an integrated program wherein all development or operational problems can feed back into a total NTCS System analysis. For instance, the best solution to what appears to be a user problem may be achieved by modifications to ground stations, satellites, software, procedures, or some combination thereof.

2. Demonstration of the system in terms of performance, reliability, and cost.

3. Following the demonstration of system performance, the generation of agreements on reduced separation standards, radio frequency allocations, and all appropriate international regulations should be initiated.

4. Generation of preoperational experience on the part of those organizations and personnel that will make use of the operational system. The training of key personnel, refinement of techniques and procedures, and working out of operational interfaces will bring about an overall system capability which will surpass in many aspects the Continental U.S. traffic control system. This capability on the part of both air crews and traffic controllers, in turn, instill a confidence in the system by the people who are going to use it which is mandatory for an operational system such as this one.

Specific technical objectives of the flight test program include:

1. Position determination. Measurements of user position using the two-satellite configuration to provide (a) an assessment of time-dependent latitude/longitude position determination accuracy for a moving user and (b) an assessment of the ability to determine an accurate range difference line of position for all users.

2. Voice communications. Measurement of carrier-to-noise density ratio and intelligibility as well as basic flux densities as a function of satellite radiated power (40 and 60 watts) and as a function of latitude and longitude of the user, and ionospheric anomalies. Such measurements will be made using diagnostic test messages and test tones.

3. Data communications. Similar measurements will be made over data channels except that bit error rate would be the criterion rather than intelligibility. Again, diagnostic test messages and test tones would be used.

Of special interest will be quantitative measures of performance near the design limits of the system, e.g., at Northern latitudes and minimum satellite elevation angles. Specific development objectives include:

1. Hardware. Design, development, fabrication, and testing of the Configurations A and B satellites, the ground station hardware, and prototype user hardware. At a minimum the user hardware should be developed to the extent required to allow achievement of the specific technical objectives listed above.

2. Software. Several different types of software must be developed including:

   - Navigation equations for various classes of users

   - Data processing software for automatic position/data reporting (Autorep)

- Data processing software for air traffic control purposes, e.g., aircraft position prediction and conflict searches

- Executive routines for mission support operations, including weather/Company business, etc.

3. Procedures, regulations, and International Agreements. As the hardware and software for the Navigation/Traffic Control Satellite System are developed, so must be the standard operating procedures for the various users, the regulations by which air and marine traffic will be governed, and the international agreement and national statutory steps which must be taken to implement these regulations. These management and administrative functions must also be accompanied by a significant documentation program including widespread dissemination of regulations, standard operating procedures, operations and maintenance manuals.

5.2.2.2 Early Operational Program

As indicated earlier, Phase II will involve the additional development required to qualify the Configuration C spacecraft for flight. When the third such satellite is launched and certified as operational, the reduced separation standards agreed to earlier can be placed into effect. As the preoperational satellites and prototype user hardware are augmented and replaced with operational satellites and production user hardware, a great deal of experience will be gained by aviation, maritime, and ground personnel. As the capabilities of the system in terms of hardware and user personnel grow, so will the confidence of user organizations in the system. Since the D and D/Preoperational program will have accomplished much of this, the elements at this step will be a smooth transition with regard to program management from the "D and D" government organization to the "operational" government agency. Another key point, again in the management realm, is that of the lead time required to formulate and obtain agreement on national and international radio regulations, traffic control regulations, and standard operating procedures.

The capability which the Navigation/Traffic Control Satellite System provides will be unparalleled in the history of air traffic control. This system can provide a continuous and complete source of voice and data communications, and a surveillance data which provides a complete time history of all controlled marine and air traffic. These data can be used not only in an operational sense, i.e., to establish proof of airspace violations when they occur, possibly for the assessment of penalties associated with infractions of regulations, but more important, to provide a credible source of accurate navigation/track-keeping data of user craft in a way which can provide precisely the type of information required to reduce separation standards in the North Atlantic Ocean Area. As indicated in Section 2, Volume II, there have been problems associated with gathering data of this type (e.g., Operation Accordion in 1965 and the 4th Meeting of the NAT Systems Planning Group held in Paris on 17-28 June 1968). In the past there has been a great deal of difficulty in correlating and weighting the different sources of data, i.e., from the navigator's log, from aircraft with and without doppler navigation, and from radar stations such as Gander. Once the Navigation/Traffic Control Satellite System has been flight tested, proven, and calibrated, and assuming that it will be periodically spot-checked, the time history of the motions of various ships and aircraft can be used in the following ways:

<u>Operations</u>

- Provide real-time surveillance for traffic control
- Document airspace violations

<u>Operational Development</u>

- Analyze the navigation/track-keeping kinematics
- Analyze the voice and data communications associated with these craft
- Together with other sources of information, such as pilot and traffic controller questionnaires, allow a complete analysis of the traffic control and operational support requirements and capabilities.

These analyses will allow virtually continuous, near-optimization of the use of the data links. For example, if it appears that certain types of aircraft are less well behaved in the track-keeping sense, they can be required to give an Autorep report with greater frequency than other more well behaved craft. Thus it can be seen that — <u>without any adjustment of regulations or modification of hardware</u> — the Autorep load can be essentially adjusted to the total system needs.

It seems clear that such programs (scaled down, and perhaps intermittent) should be performed for the operational life of the Navigation/ Traffic Control Satellite System. In this way the system can be made to almost continually conform to the requirements of a dynamic user environment.

5.3 PROGRAM COSTS

5.3.1 System Elements

Satellites. As described in Sections 1, 2, and 3 of this volume, the NTC Satellite will exist in three similar configurations with a high degree of commonality between configurations. Configuration A will be the Phase I, D and D/Preoperational spacecraft. It will weigh approximately 720 pounds and will have as its payload one voice, one data, and one ranging channel.

Configuration B will be an operational version of Configuration A and will be employed in synchronous inclined elliptical orbits. It will differ from Configuration A only in the yaw control design.

Configuration C will be another operational version, weighing approximately 950 pounds and providing four voice, two data, and one ranging channel. It, too, is similar to Configuration A, but has more power to handle the greater communications capacity.

Ground Stations. The ground system comprises hardware items already developed for other applications. For this reason minimal development cost is anticipated. The hardware has been priced based on vendor catalogs and estimates of the responsible ground system engineers.

The master control complexes and the remote tracking stations will be located at existing facilities. Since the dedicated master control complexes are of sufficient magnitude, costs have been provided for a 3000 square foot operations building for each site. These costs include site preparation and utility installation. All other facilities for both the master control complexes and remote tracking stations were assumed to exist and be available for NTCS uses.

The costs provided for ground control stations do not include estimates for master air traffic control centers.

User Equipment. The estimated costs of the NTCS user equipment were developed for the black box configurations provided by the electrical designers and packaging engineers. This configuration is suitable for commercial aircaft of the SST and 747 classes. No attempts were made in this study to provide cost estimates for other classes of users such as ships or small aircraft. These cost estimates are for providing the user equipment f.o.b. factory and do not include aircraft installation, maintenance, or manuals.

5.3.2 Methodology

The basic parametric data used in developing the Navigation/Traffic Control Satellite System cost models were taken from the TRW Systems data bank and are based on TRW and other contractor experience on related satellite, ground station, and airborne hardware programs. Responsible study engineers selected subsystem equipment based on technical objectives and NTCS program requirements. The estimates contained in these cost models are expressed in 1969 dollars and have been burdened through G and A, but do not include contractor fee.

Since the Navigation/Traffic Control Satellite Mission Study has been basically a feasibility study, the cost figures herein represent that level of detail and accuracy. The effort represented herein is consistent with the design effort itself. It is felt that the preliminary design phase, which would optimize the system, would be the major source of costing changes as opposed to uncertainties in the costing of the existing design which are considered to be in the neighborhood of 20 percent.

Development effort was assumed to include breadboard and test, engineering model fabrication and test, and qualification model fabrication and test; cost estimates were included to cover this effort.

The Navigation/Traffic Control Satellite configuration which is described herein is not an optimized design. In particular, with the weight margins presently available, the satellite MTTF can and should be increased to something like 50 to 75 months. Study cost and time limitations prevented such an optimization. Since the design described herein has a 45-month MTTF, it requires a high replenishment rate. Clearly, the preliminary design phase would include such a design refinement, with a corresponding reliability increase. This reliability tradeoff is discussed in Appendix D, Volume II. The satellite costs and the launch replenishment rate chosen and costed are a reasonable approximation of what the final optimized satellite design would require, i.e., the cost estimates already include approximately a $1.25 M cost increase per satellite to achieve a 60 month MTTF, the projected replenishment rate.

The booster costs for the DSV-3L-TE-364 and the Titan IIIB/Agena D boosters were provided by NASA Headquarters. These figures represent the launch vehicle hardware costs and also the cost of launch and range support.

Assumptions used for the user equipment costing effort were:

1) Development costs include all engineering effort and disciplines required to design and develop the user equipment as well as costs for building breadboard, two engineering models, and one prototype model of all hardware.

2) LSI and analog/RF hybrids technology devices were assumed to be standard production items in the scheduled time spans, based on projected state-of-the-art advances.

3) Commercial level specifications and workmanship standards were utilized as a basis for cost estimates.

4) No AGE costs are included for system demonstration testing. It was assumed that factory test equipment will support this activity during the preoperational phase.

5) <u>User Equipment Production Schedule:</u>

Phase I: D and D/Preoperational: 6 flight systems, 10 total, manufactured in 1972.

Phase II: North Atlantic deployment (arbitrarily estimated as 1000 systems). Production starts in 1971 and runs for five years.

Phase III: Worldwide deployment (arbitrarily estimated as 10,000 systems). Production starts in 1978 and runs for ten years.

The user hardware described in Section 2.3 is, for the most part, unique to the Navigation/Traffic Control Satellite System, whereas the satellite design took advantage of a great deal of related or similar design work. For this reason the user equipment component configurations are described here, from a construction as opposed to a functional point of view, in order to provide greater visibility into the costing exercise.

As mentioned previously, the satellite costing effort was significantly enhanced by a large amount of available and applicable information in the data bank. The NTCS System user hardware is more unique to this system, however. In order to provide additional visibility into the user hardware costing effort, descriptions of the individual user equipment components (from a construction, as opposed to a functional viewpoint) are included below:

- Composite Electronics Unit

  This unit contains three functional sections: digital, analog, and power supply, consisting of a total of 28 multilayer printed circuit boards. The boards are mounted into connectors which are held in a master board. Interconnect between connectors is provided via a wire wrap matrix.

  Digital Function. Nine modularized boards are utilized for digital functions — three for keyboard/message file, one for the company/weather message decoder, one for the Autorep buffer, three for the bit sync, and one for the message output. There are approximately six large-scale integrated circuits (LSI's) assembled to each board.

  Analog Function. Fourteen boards are utilized in seven functions: four for the receivers, five for the synthesizers, one for the three mixers, one for the temperature compensation and crystals, one for the divider/multiplier, one for the preamplifier, one for the mixer divider. These boards average 12 hybrids and 60 discrete components per board.

Power Supply Function. The power supply is mounted in the rear of the enclosure and requires four boards. There will be 315 wires terminating in the six output connectors. In order to dissipate the heat generated on the boards, the enclosure provides cooling air across all modules by use of aircraft forced air supply. Venting is required.

- Control Panel

The control panel is a 6" x 6" x 6" sheet metal enclosure. It contains a voice mode, channel selector, antenna selector, volume control and surveillance switches on its front panel. A speaker and/or jacks are provided for the audio portion. Inside the enclosure are three terminal boards which handle the audio amplifier (input and output) and processes. These boards average 60 discrete components per board.

- Message Entry/Display Unit

The message entry (keyboard)/display unit is designed to interface with the analog/digital electronics package and the printer. It displays information from ground data stations and also permits transmission of information via 20 rectangular switch/indicators to the ground station. This requires 84 wires. It contains a purchased cathode ray display tube and 20 dual type illuminated rectangular switches.

- Hard Copy Printer

A hard copy of every message that leaves the airplane or comes into the airplane is made for record. This is considered to be a purchased item similar to the teletype Model 33RO printer modified for aircraft use.

- 300 Watt Transmitter

Thirty-two power and six driver and modulator modules utilizing transistors (176), resistors (308), and capacitors (220) chips. Two modules are mounted on an aluminum (4" x 4") plate with a substrate of heat conductive beryllium sheet bonded to each plate. Spacers are inserted between modules with 170 interconnects anticipated between modules and 20 external wires for power and signals. Due to the power dissipation of the unit, forced air is required. RF and power inputs and outputs are provided via three connectors. The unit will have a base plate and sheet metal cover 10 x 10 x 7 inches.

Note: The reasons for recommending the use of a solid state transmitter, rather than a less expensive TWT unit, are as follows:

- The reliability of the TWT unit will be lower, and any failure will be a total failure; whereas interchangeable solid state modules will provide a fail-soft capability.

- The TWT unit requires a more complicated power supply, offsetting its weight advantage.

- As the state of the art in solid state devices continues to advance, the costs will become comparative.

- The exceptional long life of this solid state transmitter (it might be comparable to the aircraft lifetime.) is very desirable in minimizing aircraft down time - critical for SST's.

- Diplexer

  The diplexer will be a modified aircraft unit for L-band use. This is similar to the Wavecon diplexer.

- Power Distribution Box and Interconnect Cables

  The power distribution box contains three power relays, 10 RFI protection components, and 20 discrete components mounted on three terminal boards. Eight connectors are provided for all signal and power input/output and it is estimated that 80 interconnections would be required externally.

  The interconnect cables will consist of high grade - low loss antenna cables, and six shrink fit bundles for system interconnect. Two 20-foot cables containing 340 wires each were estimated to connect the display to the digital electronics unit. One 20-foot cable was estimated to connect the digital electronics unit to the transmitter (20 wires). One 5-foot cable (10 wires) was estimated to connect the transmitter to the power distribution box. One 20-foot cable was estimated to connect the digital electronics unit to the hard copy printer.

- Antennas

  The antennas will be an annular slot/dipole or turnstile configuration. They will include the necessary electronics to make them electronically steerable for switching patterns.

- Receivers

  The receivers are made up of identically configured modules that are keyed to the receiver functions. Each function, e.g., the BINOR processor, is designed and built into a single module. Die castings will be utilized for the metal frames of the modules or slices. The four functions, microwave unit, BINOR receiver, BINOR processor, and computer interface are mounted on single, printed circuit cords that are inserted into the metal slice. The mother board will be a stamping with self-aligning connectors. The electronic components will consist of LSI, hybrid circuits, and discrete components.

- Navigation Control and Display Unit (Optional)

  A unit similar to the AC Electronics control and display unit for the Magic 311 Computer was used for this estimate. It weighs 5-1/2 pounds and its volume is 140 cubic inches.

- Navigation Computer (Optional)

  The entire unit is of a modular construction. Each module is made up of four submodules or "sticks" which contain four to six micro-electronic modular assemblies (MEMA's), modular flat packs and a small number of discrete components.

  The memory section consists of layers of wired cores. The layers are wired together and make up the computer memory capacity.

- Systems Engineering/Program Management

  Systems engineering and program management cost estimates have been provided for the D and D/Preoperational phase only. It was assumed that the worldwide user equipment would be individually procured by the users as standard items.

5.3.3 Cost Estimates

Tables 37 through 41 present preliminary cost estimates for the various elements of the Navigation/Traffic Control Satellite System.

5.4 SUBSYSTEM TEST PROGRAM RECOMMENDATIONS

5.4.1 Introduction

TRW has been funded under separate contract (see References 1 and 5) to delineate and study detailed test programs for the navigation portion of the system. These studies will not be repeated here; they will simply be described grossly and test objectives and costs will be called out. One important type of test which was not considered involves the use of the ATS-E. (It was not considered because the decision to add an L-band transponder had not been made.) ATS-E is being modified for an L-band transponder and thus will provide an important capability for additional testing. Study of such tests is currently proceeding under TRW funding.

Limited considerations for testing the voice and data systems are given in Reference 2. No cost information was generated by TRW.

In addition to the above position determination subsystem work, there are a number of other related systems and technology programs which would enhance the overall Navigation/Traffic Control Satellite System program.

Table 37. Navigation/Traffic Control Satellite Program Summary Costs ($000)

	PHASE I D&D/Preoperational		PHASE II North Atlantic		PHASE III Worldwide
	Development	Implementation	Development	Implementation	Implementation
Satellite Program	$38,000	$15,300	$14,350	$56,350	$107,650
	55,800		70,700		107,650
Launch Vehicles		9,600		62,200	110,600
Ground Stations	1,640			10,450	23,730
NTCS Program Costs	67,040		143,350		241,980
User Equipment	15,100	1,200			
	16,300				
TOTAL	83,340		143,350		241,980

Table 38. Satellite Summary ($000)

	PHASE I D&D Preoperational				PHASE II North Atlantic				PHASE III Worldwide		
	Develop.	Implementation			Develop.	Implementation			Implementation		
	Total $	Unit $	Qty	Total $	Total $	Unit $	Qty	Total $	Unit $	Qty	Total $
Satellites A + B											
Spacecraft	30,000	7,000							7,000		
Program Management	4,500	900							900		
Subtotal	34,500	7,900	2	15,800					7,900	4	31,600
Ground Support Equipment	3,500	2,000	1	2,000							
TOTAL	38,000			17,800							
		$ 55,800									
Satellite C											
Spacecraft					11,550	7,500			7,500		
Program Management					1,800	950			950		
Subtotal					13,350	8,450	6	55,850	8,450	9	76,050
Ground Support Equipment					1,000	500	1	500			
TOTAL					14,350			56,350			
PHASE TOTALS		$ 55,800				$ 70,700					$107,650

Table 39. Launch Vehicles and Services Costs ($000)

	Phase I Preoperational		Phase II North Atlantic		Phase III Worldwide	
	Quantity	Total $	Quantity	Total $	Quantity	Total $
DSV-3L/TE-364 (4.7 M each)	2	9,400			4	18,800
1st of Kind Mission Integration	1	200				
Titan IIIB/Agena-D (10.2 M each)			6	61,200	9	91,800
1st of Kind Mission Integration			1	1,000		
Phase Totals		9,600		62,200		110,600
Program Total	182,400					

Table 40. Ground Station Costs ($000)

	D and D		North Atlantic		Worldwide		Total	
	Qty	Total $	Qty	Total $	Qty	Total	Qty	Dollars
Master Control Complex (MCC)	1*		1*		2		3	
Dedicated Complex (at existing facility)				700		1,400		2,100
Project Operations Control Center (POCC)		1,100		7,600		17,400		26,100
TT and C Station		240		2,000		4,480		6,720
Subtotal		1,340		10,300		23,280		34,920
Remote Tracking Station (RTS)	2		1		3		6	
BINOR Terminal		40		20		60		120
Data Processing and Control Equipment		60		30		90		180
Data Transmission Terminal		200		100		300		600
Subtotal		300		150		450		900
TOTAL	3*	1,640	2*	10,450	5	23,730	9	35,820

*D and D estimate for Master Control Complex (MCC) takes advantage of NAFEC facilities; North Atlantic coverage is estimated to include moving NTCS-peculiar equipment to Kennedy Airport, plus the estimates for additional MCC equipment and facility building to establish independent capability as described in Section 2.4.

Table 41. User Equipment Costs ($000)

	Phase I D&D/Preoperational			Phase II North Atlantic	Phase III Worldwide
	Develop	6 Units Implementation	Average Unit	Average Unit Cost[1]	Average Unit Cost[2]
Composite Electronics Unit	3,100	312	52.0	30.0	28.0
Control Panel	50	3	0.5	0.3	0.3
Message Entry/Display Unit	375	33	5.5	3.0	2.3
Hard Copy Printer	35	12	2.0	1.5	1.4
300 W Transmitter	1,500	300	50.0	25.0	23.0
Antenna	140	18	3.0	2.0	1.0
Diplexer	115	15	2.5	0.2	0.2
NAVSTAR Receiver Unit	2,100	63	10.5	4.0	3.3
Cables and Distribution Box	375	105	17.5	11.0	8.5
Subtotal				$77.0	$68.0
Navigation Computer (Optional)	475	180	30.0	12.0	7.0
Navigation Control and Display Unit (Optional)	985	51	8.5	6.0	4.5
System Engineering/Program Management	5,850	108	18.0		
Total	15,100	1,200	200.0	95.0	79.5

1 - Based on 1000 unit production
2 - Based on 10,000 unit production

The test plan described in Reference 1 develops a series of tests for evaluating the navigation system. These tests are aimed at establishing, at a relatively low cost, a high degree of confidence in the ultimate performance of navigation satellite systems operating in the frequency range from 1540 to 1660 MHz.

The specific tests proposed in this plan follow a logical sequence from the laboratory, to the field, to spaceborne testing using a small piggyback-launched satellite. (The ATS-E will, however, provide a test superior to that using the piggyback satellite. Test goals will be similar.) In addition, a demonstration test of the NAVSTAR position location technique using the Initial Defense Communications Satellite Project (IDCSP) satellite network is discussed. The ultimate goal of the test plan is the development of sufficient data and confidence to permit the deployment of a full-scale prototype satellite system.

The test sequence for which cost information is presented includes:

- Laboratory Tests, using a precision oscillator, BINOR code generator, and low power L-band transmitter, connected through an attenuator to a BINOR time division multiplex (TDM) receiver and range acquisition unit. Another test uses an antenna test range and model antennas and aircraft to measure user antenna pattern characteristics.

- Field Tests, using a receiving aircraft and/or helicopter to simulate a user and such signal sources as aircraft, balloons, or ground transmitters simulating the satellite(s).

5.4.2 Test Objectives

Since potential Navigation/Traffic Control Satellite System users range from low speed ships to SST's the complexity of user equipment will vary significantly, depending on the user's dynamics and need for precision navigation. Because of the diversity of user speeds, altitudes, and precision requirements, the system test plan needs to provide data on all of the basic operating characteristics of the system, from which the results to be achieved by a particular user can be determined.

The objective of the test phase is to formulate, demonstrate, and understand the system model. A model of the ranging measurement error sources for the NAVSTAR system has been presented in Reference 1. This model is an updated version of the error model given in Volume II of Reference 1.

As indicated in Table 42, each test is designed to verify certain portions of the range error model. In addition, Table 42 shows the other objectives of each specific test which are connected with the operating performance of key elements of the user and satellite electronics and with demonstration of position determination.

5.4.3 Position Determination Subsystem Laboratory and Field Test Cost Estimates

The following eight pages are budgetary and planning cost estimates for position determination, subsystem, laboratory, and field tests, and are taken directly from Reference 1. The total period of time from start of work to test completion, including data analysis and reporting, is estimated to be about 18 months. The total cost is estimated to be approximately $1.3 Million with a jet transport-type aircraft used in the field tests. An additional \$0.5 Million will be required to complete the helicopter field tests (including the VSTOL ILS test).

Table 43 summarizes estimated laboratory and field test program costs assuming that only the jet transport is used in the field tests. Table 44 provides incremental costs of adding a user helicopter to the field test program. The latter costs include VSTOL ILS tests as well as NAVSTAR system tests in the instrumented helicopter configuration.

It should be noted that both tables show a small difference in cost when a balloon is used instead of an aircraft for a high altitude signal source. The estimated difference is \$20,000 in favor of a balloon source for the jet transport user tests, and \$4,000 in favor of a jet aircraft source for the helicopter tests. The slight advantage to a jet aircraft source in the latter case accrues because the aircraft need not be tied down for installation and checkout of the signal source instrumentation; installation and checkout are included in the costs for the jet transport user field tests. The balloon costs assume a total of five flights each for the jet and helicopter tests. If problems occur in system performance, however, five flights may not be enough. Figures 72 and 73 provide a feasible schedule for the laboratory and field tests.

Table 42. Objectives of Each Test

Test	Error Model	Other
Laboratory	• Receiver thermal noise • Receiver drift and bias • Quantization • Satellite oscillator	• Receiver acquisition times • User antenna coverage • User antenna multipath reception
Field	• Multipath • User oscillator	• Operation of user equipment in aircraft and/or helicopter • Airborne RFI environment • Refinement of satellite ERP requirements • Limited test of position determination
ERS or ATS-E	• Tropospheric retardation • Ionospheric refraction • Multipath • Overall model*	• Further refinement of satellite ERP requirements
IDCSP		• Tracking station and user software • Demonstration of NAVSTAR position determination technique

*The overall error will be measured at a ground station so that it will not contain multipath errors or any errors that may be peculiar to an aircraft environment.

Table 43. Summary of NAVSTAR Test Program Costs (Fixed Wing Aircraft, no Satellite-Based Testing)

	Cost ($000)						
	Integrated System Lab Tests	Precision Oscillator Lab Tests	Antenna Pattern Lab Tests	RFI and User Oscillator Field Tests	*Aircraft to Aircraft Field Tests	Ground to Aircraft Field Tests	Total
1. Test Planning and Project Coordination	51.5	12.5	15.0	30.0	75.0	75.0	259.0
2. Procure Test Hardware							
a. Oscillator	8.0	8.0					16.0
b. Transmitter	34.7					74.8	109.5
c. Receiver	57.4						57.4
d. BINOR Code Generator	6.0					17.8	23.8
e. BINOR Code Processor	15.2						15.2
f. Antennas			6.3	27.7	12.9	7.3	54.2
g. Antenna Switching Unit					8.9		8.9
h. RFI Measurement Instrumentation				12.5			12.5
i. Test User Oscillators				1.0			1.0
j. Multipath Measurement Instrumentation					15.4		15.4
k. Model Aircraft and Range			9.6				9.6
l. Generator Instrumentation	28.6	7.0		28.1		80.2	143.9
3. Leased Test Equipment							
a. Jet Transport				40.0	40.0	40.0	120.0
b. Signal Source Aircraft (Signal Source Balloon)*					36.0 (16.0)*		36.0 (16.0)*
c. General Instrumentation	20.5	0.6				3.7	24.8
d. Helicopter (Ground Test Support)						8.8	8.8
4. Equipment Installation, Integration, and Check Out							
a. Laboratory	5.8		2.2				8.0
b. Jet Transport Antennas				5.1	11.8		16.9
c. Jet Transport Receivers and Instrumentation				18.0	3.7	2.0	23.7
d. Signal Source Antennas					6.2		6.2
e. Signal Source Transmitters and Instrumentation					6.1		6.1
f. Ground Antennas						2.6	2.6
g. Ground Transmitters and Instrumentation						23.6	23.6
5. Software Development						79.5	79.5
6. Test Operations	35.7	5.5	5.8	7.0	14.0	51.0	119.0
7. Data Reduction, Analysis, and Reporting	19.3	8.0	8.1	9.0	24.6	33.5	102.5
Total	282.7	41.6	47.0	178.4	254.6 (234.6)*	499.8	1304.1 (1284.1)*

*The cost of using a balloon for the signal source (instead of an aircraft) is given in parenthesis.

Table 44. Incremental Cost to NAVSTAR Test Program (Helicopter, no Satellite-Based Testing)

	Cost ($000)				
	Antenna Pattern Lab Tests	*Aircraft to Helicopter Field Tests	Ground to Helicopter Field Tests	VSTOL ILS Field Tests	Total
1. Test Planning and Project Coordination	10.0	30.0	30.0	60.0	130.0
2. Procure Test Hardware					
a. Oscillator					
b. Transmitter					
c. Receiver					
d. BINOR Code Generator					
e. BINOR Code Processor					
f. Antennas	2.1	8.6			10.7
g. Antenna Switching Unit					
h. RFI Measurement Instrumentation					
i. Test Oscillators					
j. Multipath Measurement Instrumentation					
k. Model Aircraft and Range	9.6				9.6
l. General Instrumentation					
3. Leased Test Equipment					
a. Helicopter		16.0	16.0	16.0	48.0
b. Signal Source Aircraft		12.0			12.0
(Signal Source Balloon)*		(16.0)*			(16.0)*
c. General Instrumentation			3.7		3.7
d. Helicopter (Ground Test Support)			8.8		8.8
4. Equipment Installation, Integration, and Check Out					
a. Laboratory	2.2				2.2
b. Helicopter Antennas		9.1			9.1
c. Helicopter Receivers and Instrumentation		21.7	2.0		23.7
d. Signal Source Antennas					
e. Signal Source Transmitters and Instrumentation					
f. Ground Antennas					
g. Ground Transmitters and Instrumentation				11.8	11.8
5. Software Development					
6. Test Operations	5.8	14.0	51.0	39.3	110.1
7. Data Reduction, Analysis, and Reporting	8.1	24.6	33.5	9.9	76.1
Total	37.8	136.0 (140.0)*	145.0	137.0	455.8 (459.8)*

*The cost of using a balloon for the signal source (instead of an airplane) is given in parenthesis.

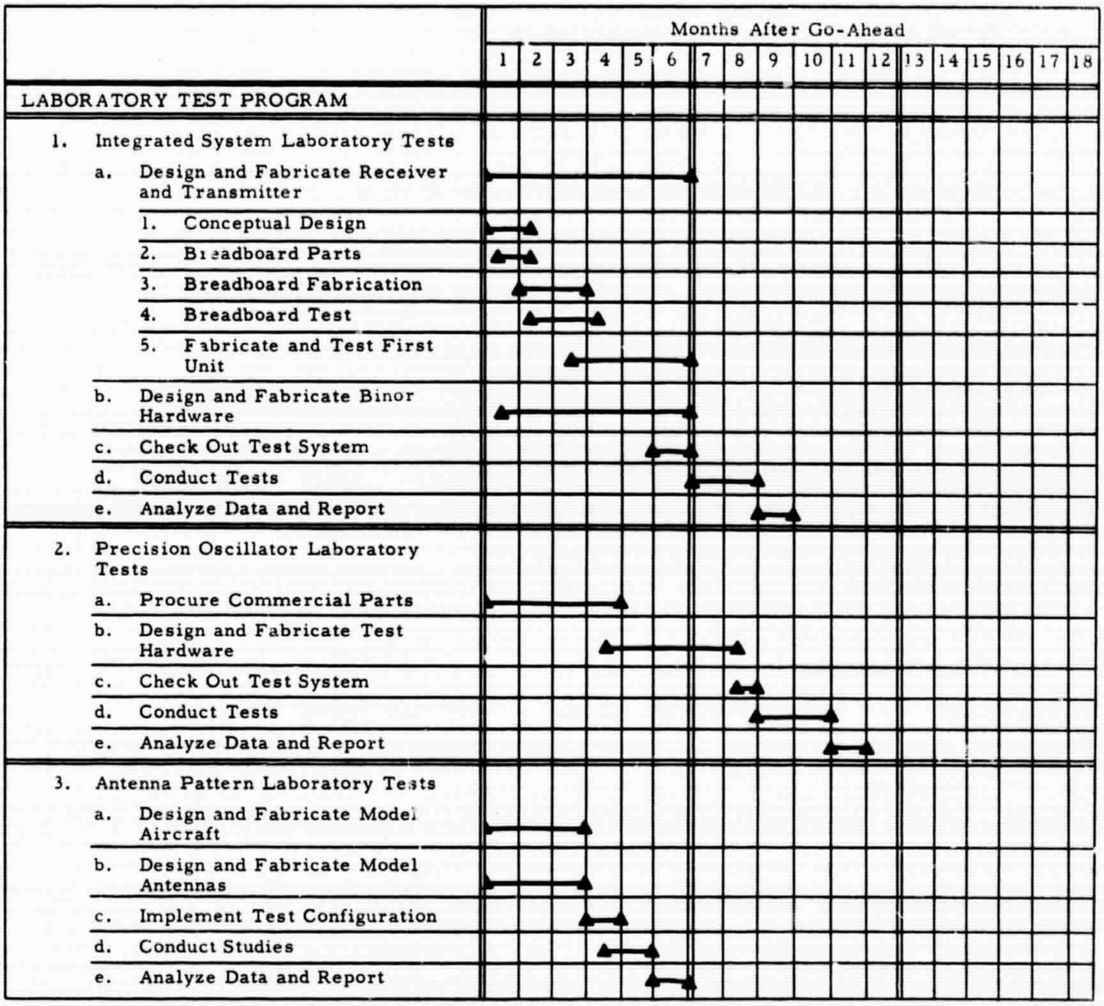

Figure 72. NAVSTAR Laboratory Test Program Schedule

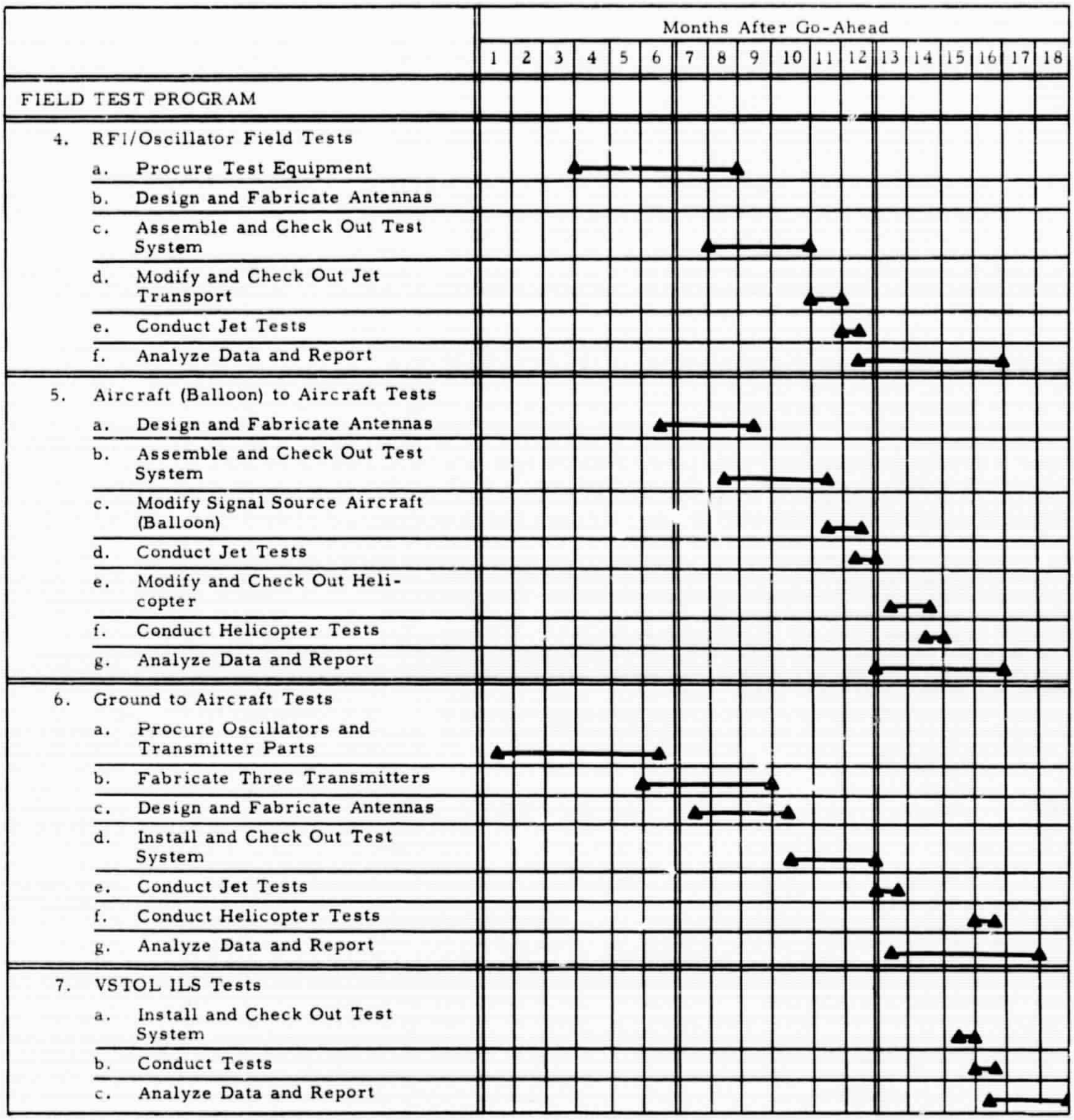

Figure 73. NAVSTAR Field Test Program Schedule

5.4.3.1 Assumptions

The costs estimates in Tables 43 and 44 are based on the following assumptions:

1) The cost of an engineer for one year is 60,000 dollars (including clerical support, overhead, and contractor's fee).

2) The cost of a technician, fabricator, or draftsman for one year is 18,000 dollars (including clerical support, overhead, and contractor's fee).

3) The test program will be conducted over an 18 month period.

4) Only one BINOR code processor and receiver system is required for all tests. The same system will be used for the laboratory tests first, then for the jet transport user field tests, then for the helicopter user field tests.

5) A total of four oscillators, four BINOR code generators, and four transmitting systems are required One set of equipment will be installed in the signal source aircraft or balloon while the remaining three systems will be used on the ground for the position location field tests.

6) The test program will be conducted in the chronological order shown in Figure 73. Equipment associated costs (Tables 43 and 44) are included under the test phase for which the equipment is first required. Succeeding tests requiring the same equipment do not include those costs again.

7) A less expensive oscillator than the one indicated can be used for the integrated system laboratory tests. Table 43, however, reflects a precision oscillator since it will be required for subsequent laboratory and field tests.

8) Receiver and transmitter costs are based on the assumption that no hardware can be "borrowed", but that there are some designs which can be modified to save design time.

9) The transmitter design will use commercial components for the stable oscillator and 4-watt power amplifier. The phase modulator and multiplier will be laboratory developed for the system.

10) The receiver will be completely laboratory developed.

11) Costs have been included to insure that the transmitter and receiver are flightworthy for airborne environments.

12) Test equipment costs are based on monthly leasing rates. For some general purpose test equipment, which may be used throughout the program, it may be cheaper to purchase rather than lease.

13) Receiver engineering tests will include: (a) time delay variation vs temperature and input signal, (b) dynamic range, (c) noise figure vs temperature, (d) phase lock loop performance in the presence of doppler and doppler rate, and (e) carrier acquisition times.

14) No major subcontracts are required.

15) Capital equipment and instrumentation costs include material handling.

16) Fabrication costs include model shop labor as well as part costs.

17) Product Assurance participation is limited to workmanship inspection in the model shop.

18) Formal documentation of fabricated equipment is not included in the estimates.

19) Fabricated equipment will not be accepted and delivered. It will, of course, be structurally compatible with the aircraft environment and its appearance will be consistent with purchased equipment.

20) The airborne computer will process real-time data and output a printed record of computed position at a rate consistent with computer capability.

21) The airborne computer will be used to process postflight and other test data.

22) The BINOR equipment to be fabricated will be similar in concept to that already available at TRW. No increase in operating speed or performance capability is covered.

23) Basic power sources available to all test locations (i.e., 110 volt, 60 Hz, with a power capacity of at least 2 KVA).

24) The scale model aircraft and range for the antenna laboratory tests assumes a one-seventh scale model of the Boeing 707 type jet transport. The model can be constructed or leased, whichever is least expensive.

25) Four antennas are required on the jet transport user aircraft - one on the top, two on the sides, and one on the bottom. Design and installation is estimated under the RFI field tests. The increment for fabrication and installation of three additional antennas is estimated under the aircraft to aircraft tests.

26) Two antennas are required on the helicopter user aircraft - one over the rotor blades and one on the bottom of the aircraft.

27) The jet transport user aircraft (Jetstar class) will be required for approximately 12 weeks at an average estimated rental cost of 10,000 dollars per week.

28) The signal source aircraft (Lear Jet type) will be required for 6 weeks at an average cost of 6,000 per week.

29) The helicopter user aircraft will be required for 16 weeks at an average cost of 4,000 per week. The helicopter tests will be conducted after the jet tests to minimize costs.

30) All software for the airborne computer is estimated under the ground to aircraft position location tests. Some of this software, however, will be used for reduction of post-flight data from other field and, possibly, laboratory tests.

31) Specification development and design or qualification tests are not included because they are not considered necessary.

32) Design or qualification test costs are not included and are not considered necessary.

A possible test configuration which has not been estimated is that of using only the helicopter for the user aircraft field tests. This configuration is not recommended because the jet transport is believed to be the most representative of the user aircraft. If it is desired for other reasons, however, to conduct the field tests using only a helicopter for the user aircraft, the costs would be comparable to those of the jet aircraft only (Table 43).

5.4.4 Related Systems and Technology Programs

Other Navigation/Traffic Control Satellite System related programs are itemized in Table 45.

Table 45. Other NTCS-Related Systems and Technology Programs

SATELLITE

- L-Band Phased Array Antenna
- L-Band Transmitter
- Long Life, High I_{sp} Stationkeeping
- Solar Array/Primary Power Design

USERS

- Aircraft Antenna Design, Installation and Test
- Voice Processing Studies and Test
- Communications Hardware Development and Test

SYSTEM STUDIES

- Continental U.S. Air Traffic Control
- Search and Rescue
- Collision Avoidance
- ATC Interface
- Data Handling
- Position Determination/Communications Subsystem Aircraft Integration
- Marine Applications

These programs represent efforts that can usefully precede the NTCS preliminary design if that phase is delayed. Some of the above efforts (e.g., ATC interface analyses) would be incorporated into a full-scale Navigation/Traffic Control Satellite System design and development program; whereas other (e.g., solar array/primary power design) would simply be dropped and the system design would proceed with the present state of the art, which in all cases is more than adequate.

The application of phased arrays to satellite antennas would greatly influence the system design. The ability to point narrow beams at aircraft requesting service, while at the same time effectively providing full earth coverage, is clearly an attractive approach and should be investigated further than time and funding limitations of this Mission Study would allow. Reference 4 includes a detailed proposal of what can – and should – be done in this area. Similarly, the development of an efficient (40 percent) high power (100-200 watts), solid state L-band transponder is especially desirable in view of the requirement for high reliability, low weight, power and volume spacecraft subsystems. Long life, high specific impulse, and stationkeeping could prove very beneficial, in the event that further design studies showed that it would be economically advantageous to a large number of users to simplify their software and hardware, by taking advantage of the fact that each satellite was, in fact, located exactly at its nominal position. Improvement of satellite primary power-to-weight ratios always proves beneficial to system design and the Navigation/Traffic Control Satellite is no exception. As indicated in Section 6, Volume II, there are attractive solar array techniques which merit further refinement and offer significant promise.

In the user hardware area, in addition to the position determination subsystem work discussed in the previous paragraph, there are several communications oriented toward joint position determination/communications hardware considerations that merit individual attention. Aircraft receiver noise certainly needs to be investigated, especially in the supersonic case via an aircraft antenna design, installation, and test program. Increasing the modulation efficiency of voice signals also deserves attention which could take the form of more efficient voice processing, bandwidth, complex techniques, or even the development of an adequate low-cost vocoder.

The limitations in scope of the Navigation/Traffic Control Satellite Mission Study and the high degree of focusing on the North Atlantic Ocean Area and commercial airline users leaves a number of other system study areas which merit active attention. Of those listed in Table 45, TRW feels that the Continental United States air traffic control is the most important.

5.5 REFERENCES

1. "Test Plan for NAVSTAR Navigation System: Volume I, Test Plan Description," TRW Report No. 08710-6023-R0-00, Contract NAS 12-539, November 1968.

2. "Considerations for an L-Band Satellite Relay Voice Link and Test Plan," TRW Report No. 08710-6025-R0-00, Contract NAS 12-539, November 1968.

3. "Study of a Navigation and Traffic Control Technique Employing Satellites," Volumes I through IV, TRW Report No. 08710-6012-R000, Contract NAS 12-539, December 1967.

4. "High Efficiency Satellite Phased Array Experiments," TRW Systems Proposal No. 12921.000, 30 November 1968.

5. "Applications Technology Satellite-F L-Band Air Traffic Control Communications-Navigation Experiment," TRW Report No. 08710-6015-R000, Contract NAS 12-539, June 1968.

6. APPLICATIONS

6.1 GENERAL

In addition to the air traffic control and other major missions performed by the Navigation/Traffic Control Satellite System there are a number of special applications which have received enough attention on this study to be reported on herein. They include:

- Continental U. S. air traffic control
- Collision avoidance
- Altimetry
- Fishing and oceanographic vessels
- Solar flare sensing and warning
- Space navigation.

In addition to these applications, a number of others have been considered (e.g., References 1 through 6), including:

- Aircraft status and maintenance messages
- Entertainment of passengers
- Public use (e.g., telephone calls, telegraph messages, etc.)
- Point-to-point communications
- Pictorial data, such as cloud cameras
- Manned meteorological satellites
- High altitude winds
- Radiobiological data.

During the course of this study a certain amount of attention was given to these items, especially the first, but the study scope did not allow sufficient effort to be expended here to warrant their inclusion in the final report. The first set of applications will be discussed in the following paragraphs.

For example, TRW conceived an approach for temporary storage and emergency transmission of aircraft status and/or maintenance information. This concept involves sensing of aircraft subsystem data, and the storage of this information in either analog or digital format. Older data is continually replaced by the most recent data in a limited data storage device such as magnetic tape, or digital buffer. This data would be transmitted only when an aircraft is in an emergency condition or if it appeared that such a condition was imminent. This technique would allow emergency transmission and post-flight analysis on, for example, an aircraft which was downed and forever lost in the North Atlantic Ocean; and at the same time would not necessitate continuous wideband transmission of data via satellites from all aircraft. Although the application of this technique as envisioned here involves transmission via satellite link, that type of transmission is not fundamental to the concept. Direct air-to-ground communications of the same type would be applicable for such a scheme. This scheme is reported in the New Technology Section of Volume I. No judgment is made or intended herein as to the patentability of the above items. It was reported because it was an idea which was conceived under contract to NASA and, to the best knowledge of the innovator, represented original thinking.

6.2 APPLICATION OF THE NAVIGATION/TRAFFIC CONTROL SATELLITE SYSTEM TO CONTINENTAL U.S. AIR TRAFFIC CONTROL

6.2.1 Present Operation

The control of air traffic within the Continental United States is intended to provide safe and expeditious movement of aircraft. Aircraft navigation is air-derived, primarily from ground-based facilities, and includes the VOR/DME or TACAN for enroute and terminal area navigation and the ILS localizer, glide slope and marker beacons for final approach. Ground-based surveillance exists primarily for the maintenance of safe separations and relies on position determination from primary and beacon radar as well as pilot reports. Positive control exists on a continuous mandatory basis above certain altitudes along airways and in terminal zones, and also becomes mandatory under prescribed weather minimums within other portions of the controlled air space. Control is exercised by assigning point-to-point clearance along designated airways and may require holding at fixes, delaying maneuvers and speed control as necessary to achieve the separations which are prescribed by regulations and, for specific flight situations, are considered safe by each controller.

6.2.1.1 Control Facilities

The control facilities include Air Route Traffic Control Centers (ARTCC) for enroute control, which use long distance remoting of radar and air-ground communications inputs; the IFR rooms for the terminal area approach and departure control, which employ displays from the short range surveillance radars and possibly precision approach radars; and the airport traffic control towers which provide landing, takeoff and taxi control, and may include radar displays of the approach and taxi areas to augment visual observation of traffic.

The ARTCC's have control over varying volumes of airspace depending upon the complexity of airways structures and loading. The individual work loads are adjusted to manageable proportions by the designation of control sectors sized to the peak demand. The sector controller must coordinate with the controllers of all the contiguous sectors and the IFR rooms or towers contained within his sector both in planning for and the actual transfer of control of arriving and departing aircraft.

This control system is now dependent upon manually prepared and updated records including flight plans, flight progress strips and radar track identity markers. These provide the reference data for both the planning and the real time control of the flight situation. This record operation involves the use of telephone, radio and teletype reports, and requires many handwritten entries, manual estimate time of arrival (ETA) computations[1] and physical transfer of data strips for each aircraft transiting the system. The real-time radar control operation, which involves the manual correlation of identity markers with blips on the radar displays, is especially cumbersome.

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[1] This is the expected time that each aircraft will arrive at its next check point. These "compulsory reporting points" are essentially the nodes of the system, wherein most actual control of traffic takes place.

6. 2. 1. 2 Navigation System

This control system is referenced to the airways the pilot can follow using the ground-based radio navigational aids. These airways follow nominally direct routes between the major terminals, but the navigational aids are, of necessity, not ideally sited because of geographical accessibility and radio propagation effects. The VOR and TACAN bearing signals provide angle references having overall accuracies varying as much as 5 degrees (or ± 9 miles cross track error at 100 miles from the station). If the DME feature is not used, homing guidance is available and precise aircraft position is known only when the aircraft is crossing a ground station or by simultaneous observation of the radial of another station. Continuous position is available when the more complex distance measuring feature is also employed. Under these conditions offset tracks can be flown using VOR/DME or TACAN, especially if an off-course computer and indicator are used; however, the control system is not now adapted to general use of off-airway navigation.

The instrument landing system employs a VHF localizer, a UHF guide slope, and 75 MHz fan markers, and has inherent problems in propagating signals for phase angle measurement in an inconstant environment. Major efforts have resulted in improvement in some of the variables that once plagued the system, but factors such as construction of new buildings or even aircraft waiting at the end of the runway for takeoff clearance may modify the radiated pattern sufficiently to preclude really low approaches.

6. 2. 1. 3 Position Determination for ATC

The air route and terminal area surveillance radars which provide the primary position reference for the controller's decisions have coverage errors similar in magnitude to those of the VOR/DME. However, since these facilities are usually not collocated with the navigational aids, there is no correlation of these angular error patterns. Principal dependence is placed on beacon radar which now provides signal enhancement and limited identity. Eventually the air traffic control radar beacon system will include discrete identity and altitude when the 12-bit response code is employed. However, the air traffic control radar beacon system has the disadvantages of beam spreading, over-interrogation, garbling and dropout. These factors can be critical in a heavy traffic area.

6.2.2 National Airspace System Concept

The National Airspace System (NAS — see References 7, 8, 9) is being implemented to automate many of the ATC clerical functions which involve record preparation, updating, ETA calculations, and radar track identification. However, the NAS does not modify the traditional control methods or the sources of planning and position information. The growing problem in defining control sectors which have both a reasonable geographical size and manageable amount of traffic will become more acute as traffic congestion increases hear major metropolitan areas.

6.2.3 Future Objectives and Problems

Since the NAS will provide a necessary function and also represents a major capital investment that cannot be abandoned soon, there is a challenge to design a new method of air traffic control that will permit the controller-operated NAS to manage the increasing demand for service without compromising the safe, expeditious flow of air traffic.

A stated objective of the DOT/FAA is to shift responsibility for separation-maintenance/collision-avoidance to the pilot and reduce the controller task to flow planning and intervention for exceptions (Reference 8). This objective can be met only by the availability of a new position reference system that permits the pilot to observe in real time the separation and closing rate of his aircraft with respect to others in a threat volume.

A constraint on the expeditious flow of traffic is the landing rate which is affected both by runway capacity and also by the precision and reliability of the approach and landing system (see Reference 9). Many continuing efforts have not produced a cost-effective new solution to the ILS problem and refinements to the present system are near the reasonable technical limits.

Therefore, the continental air traffic control system is constrained in its growth by dependence upon radionavigational aids and radar surveillance facilities that have reached their practical performance limits. Inherent angular accuracy errors and radio propagation and siting deficiencies are major factors that limit the control techniques and system capacity. In order to cope with system demands without ever-increasing delays in major traffic convergence areas, a new approach to continuous precision position determination is essential.

6.2.4 Satellite System Contribution

The potential contributions of satellites to the problem of U. S. air traffic control is enormous. Detailed study of this is, however, far beyond the scope of this contract. Since this has not been done, this section can only allude to possible uses in a qualitative fashion.

The Navigation/Traffic Control Satellite System can overcome the fundamental problem of coverage by ground-based radionavigation and radar facilities, eliminating line-of-sight limitations and the related difficulty in optimum accessible siting. The satellite-based system provides continental coverage from a small number of satellites. Radio signals received by each aircraft from this common reference can either be processed onboard for navigation and/or repeated automatically to the ground-based national airspace system computers which derive positional and rate information for surveillance and traffic control purposes. This data has sufficient accuracy for onboard collision avoidance. Furthermore, highly precise three-dimension position and rate data for approach and landing can be provided by beacons similar to those in the satellites, but located on the ground in the terminal area. Airborne processing of the signals would be unchanged.

6.2.4.1 Possible Changes in Existing Control System

The major impact of the satellite referenced position would be the obsoleting of the radar beacon network which now provides the aircraft position reference to the controllers of the NAS. A principal advantage is the fact that the satellite radiation is passive and lacks the saturation problems related to excessive interrogation of the radar beacon system. Autorep data link requirements are modest — minimizing saturation problems, and making their solution a simple matter of small increases in data link capability. Furthermore, satellites provide a new level of accuracy and additional information. Finally, Autorep puts the data in the NAS system in readily usable form for automation. This can permit reduced separation both longitudinally and laterally. When the position and rate information is available for broadcast from each aircraft the objective of air-derived separation control can relieve the ATC System of a major element of the work load, and permit controllers to concentrate on planning and special situation control.

6.2.4.2 Possible Changes in Navigation

Improvement in the navigation system is required primarily by faster, higher-flying aircraft which are unduly constrained by an airway structure referenced to ground-based facilities. Although the VOR and TACAN stations are sited near the direct paths between principal terminals, actual locations are selected with consideration given to accessibility and radio propagation. The result is often "dog legs" in the airways which are laid out from station to station. Furthermore, although environmental and traffic conditions make routes other than the principal airways advantageous, they are now difficult to define and control. Although VOR/DME or TACAN can provide a reference for offset track flying when a suitable computer and display are used, the error pattern does not correlate with the radar errors asobserved by the ground controller and this uncertainty necessitates very conservative airspace assignments. Conversely, the satellite system provides a common but semi-independent reference system for navigation, collision avoidance, and controller monitoring. Also, the range/range, rather than range/bearing system does not have errors that increase with the distance from the station. This uniform area coverage permits total errors of less than 100 feet as well as accurate rate derivation. Relatively random flight paths could be managed for high density traffics. Controllers would have the inputs needed to predict the situation at convergence points and avoid incipient collisions.

6.2.4.3 Terminal Area Improvements

The terminal area traffic situation is becoming increasingly acute and the existing navigational aid coverage does not facilitate the management of the precise separation intervals needed for maximum landing flow. The dependence on radar-referenced vectoring by controllers places a great burden on the controller and requires considerable confidence in individual abilities. The area coverage precision to be provided by both the satellite reference system and the terminal area ground beacons will permit a new approach to the control of those aircraft having suitable onboard equipment. The pilot will then be able to follow a uniquely determined flight path and make this transition from the approach control pattern to the landing system by simply making a navigation receiver channel change so as to receive signals from the ground-based transmitters.

6.2.5 Summary

The approach to air traffic control system improvement has usually considered only a deliberate evolution of the present operation. Area coverage position accuracies of 100 feet or less and velocities of less than one foot per second can be available using satellites. However, the potential exists for a manageable area control system for equipped aircraft. This system could become largely self-managing except at the final convergence for the final approach. Semi-equipped aircraft would be constrained to the lower altitude route structures and that portion of the system would benefit from the accurate automatic position reporting that could permit reduced enroute intervals and higher terminal area flow rates.

Specific ATC improvements from satellite system include:

- Single universal coverage position determination system replacing air traffic control radar beacon system for NAS processing and display
- Onboard separation interval determination and collision avoidance
- Area or multiple track instead of single airway navigation having uniform high accuracy and correlated to ground position indication
- Terminal area unique track determination for maximum landing rate
- Compatible precise landing system using ground-based transmitters.

6.3 THE USE OF SATELLITE TECHNIQUES FOR A COLLISION AVOIDANCE SYSTEM

6.3.1 Background

"Warning devices to indicate the presence of aircraft operating close to each other should be lightweight, inexpensive, simple to operate, and offer complete protection." The preceding quotation was made over 25 years ago by a former CAA official. Little effort was devoted to the problem of collision avoidance until after the Grand Canyon collision of two commercial airliners in 1956. This tragic accident brought forth a spate of proposals to provide systems and devices which would prevent such occurrences.

As the use of aircraft increases, the changes of collision between aircraft increase even faster. The situation is particularly acute near major air terminals in Los Angeles, Chicago, and New York when mixed classes of traffic arrive at peak hours. The danger of military or private aircraft penetrating commercial aircraft lanes, especially during climb, and descent, also will be an increasing problem.

The airline industry now appears to be on the verge of accepting a Collision Avoidance System (CAS), based on cooperative time-frequency division (TFD) techniques, which holds promise for eliminating major airliner disasters caused by collision. However, no system of collision avoidance can be considered effective for the entire aircraft-user community unless each aircraft in that community is equipped with a compatible collision avoidance system. To provide protection to all aircraft users, warning systems must be considered for use by all classes of aircraft. The cost of currently proposed TFD equipment is expected to be about $50,000 for each installation, a figure which appears incompatible with the cost of an inexpensive private aircraft.

The high cost of the TFD collision avoidance system lies in the complex electronics and the very accurate clock required. The user clock must be accurate to within 0.5 μsec with respect to a standard which is achieved by providing frequent resynchronization from a ground station. User clock uncertainties greater than 2 μsec effectively remove him from participation by placing the user in a "standby mode." A full set of data is transmitted between cooperative users in a uniquely assigned time slot. A minimum message user (such as a private aircraft) is required to employ the same accurate clock system employed in a commercial airliner but a lesser electronics complement.

Some of the proposals made in 1956 involved various forms of radar ranging, infrared detection, bright flashing lights, etc., all of which proved unfeasible at the time. While some of these proposed techniques simply were not feasible technically, others were limited by available technology at the time. Since concern often is directed towards the small aircraft user, consideration should be given to a less expensive system.

The development of satellite technology has occurred during the time that Collision Avoidance Systems have been analyzed. As yet, however, the use of this technology in providing a Collision Avoidance System essentially has been neglected. TRW has considered several techniques for using such satellite systems to provide the CAS function, including:

- Cooperative systems based on the use of a single satellite dedicated for CAS.
- Cooperative systems based on the use of a constellation of satellites capable of providing multiple functions including CAS.

The capabilities of several such satellite-based techniques for providing the Collision Avoidance System function are worthy of further exploration.

The present Air Transport Association position concerning commercial airliner CAS is a significant modification of a previous "do-it-alone" philosophy, which demanded that an airliner be as complete an entity as possible, with minimal reliance on ground systems. The position taken by the airlines on CAS is that it must be totally independent of the basic Air Traffic Control (ATC) system, even though potential conflicts may arise between CAS-directed maneuvers and ATC instructions.

6.3.2 Potential Role of Satellites

The use of satellites for providing a CAS service offers the following advantages over the currently proposed method:

1) Lower user hardware costs

2) Lower total system costs

3) Decreased reliance on barometric altimeters

4) Increased safety potential

5) Improved air-space utilization

6) Broad area coverage including over ocean operation

7) Opportunity for participation by many aircraft classes

The Federal Aviation Administration and the American airline industry, through the Air Transport Association, have taken the position that the best solution to the mid-air collision problem is the existence and operation of a failsafe ground-based air traffic control system. If, for any reason, the ATC cannot fulfill its proper function, an independent collision avoidance system must be provided to effect safe separation between aircraft. With this philosophy, of course, the CAS must be independent of the ATC. This philosophy is a sound one and is discussed in detail in Para. 2.5.2, Vol. II. The satellite-based systems provide the potential for integration with the Air Traffic Control system as well as aircraft navigation systems, in the event that the ATC philosophy changes from independent position determination for surveillance and positive control, to an ICNI and CAS/stationkeeping philosophy (see Para. 2.5.4, Vol. II).

In any event, to provide a reliable Collision Avoidance System requires that critical elements concerning its design and operation be specified carefully. In particular, the CAS so defined must operate reliably in a very congested area, such as that encountered in the Chicago, Los Angeles, or New York City operating areas. The following discussion concerns parameters requiring careful definition if a successful collision avoidance system is to result.

6.3.3 Transmission Interference

Any cooperative CAS requires that the user aircraft radiate some form of signal which can be received by other similarly equipped aircraft. The signals so radiated generally are at a sufficiently high power level to provide an adequate signal-to-noise ratio at a receiving aircraft that, under under certain operating conditions, the signal could be received by aircraft at distances far beyond those at which the transmitting aircraft could be an immediate collision threat. The proposed ATA solution to this problem has been to assign a specific time-of-transmission to each user aircraft, replete with position and other data. Each successive transmission occurs sequentially at increasing operating frequency through a maximum of four such steps before repeating frequency. Thus, for line of sight, adjacent slot interference is reduced, but at the expense of considerable electronics complexity, cost, and RF spectrum. The requirement for reliability is a prime consideration for a collision avoidance system. Protection from interference must be provided but preferably using a simple approach. One such method which appears to offer the necessary degree of protection is the "gate limited" approach. Each user establishes his own "threat volume" and sets a time gate on his receiver-preprocessor to reject transmissions received from outside this threat volume. Such a technique is attractive for a satellite-derived CAS, since, as will be described below, it is easily achievable with minimum user hardware complexity.

In the proposed study program, link power budgets will be computed. The budgets will be used to evaluate the potential interference posed by unwanted signals and a criterion for false alarms caused by such interference will be established.

6.3.4 Time Synchronization

The collision avoidance system requires all participating users to operate from some standard reference time base. Since the transmissions from any user could cause interference to all users within line of sight if the user transmissions were not synchronized, synchronization is a basic CAS requirement. Synchronization requirements are much more acute if accurate one-way ranging between users is required as with the ATA CAS. Methods usually considered to provide time synchronization require three basic elements:

1) A preassigned time slot for each user
2) An accurate, stable user clock
3) A means for maintaining clock synchronization among all users.

To provide the capacity required for a congested area, the collision avoidance system must be capable of accommodating up to 2000 users at any time. To provide reporting on a timely basis and with sufficient frequency (considering that closure rates could approach 6000 ft/sec for two supersonic aircraft), the time allotted to each aircraft transmission must be short. Typical times considered vary from 1000 to 2000 μsec, providing user reports (based on 2000 users in an area) every two to four seconds. Thus, precise time synchronization must be achieved by all user aircraft to avoid "co-slot" interference. Some degree of protection also is provided by guard bands between adjacent slots, but these cannot be more than a few per cent of a slot time without decreasing the reporting frequency to an unacceptably low level.

This proposed program will determine the requirements for time synchronization between users. Methods of achieving the required synchronization will be specified, particularly those based upon the use of a satellite-derived clock signal. Required tolerances and reporting frequencies, as a function of threat volumes, also will be investigated.

6.3.5 User Computing Requirements

Every collision avoidance system user will be required to process the information received from all other participating aircraft within his threat volume. This data must be processed at a sufficient speed to determine a potential collision threat and to provide an indication (if required) of an avoidance maneuver. Thus, every user aircraft will require computational capability sufficiently large to process data from a number of other aircraft.

The requirements for computational speed and memory size are based on the co-slot times (for a preassigned slot system), the equivalent number of data bits in each message, and the computing algorithms used (see Section 4 .1.4). A principal cost element in the CAS computer is the memory system; thus, complicated signal structures requiring large memory capacities should be avoided. The computing algorithms also must be designed carefully to avoid excess memory requirements. Practical operational limits also must be established, such as the gate-limited method mentioned earlier, such that information relating to an aircraft which is not a real threat can be rejected quickly instead of being completely processed and stored.

The proposed program will consider the design of suitable computing algorithms which are rapid and require minimum memory for each CAS system discussed. Computer memory sizing will be augmented by consideration of a slow, high capacity bulk memory, such as provided by tape, if required. Operating speeds, in terms of elementary operations per unit time, will be defined, leading to the selection of candidate existing aircraft computers. Development items will be defined but will be eliminated wherever possible.

6.3.6 Threat Determination/Accuracy

The only function of a collision avoidance system is to evaluate the potential threat of collision between two similarly equipped aircraft and to direct the aircraft to perform appropriate avoidance maneuvers. To provide the threat determination, each CAS-equipped user radiates a signal so structured that any other user receiving this signal can determine whether or not the transmitting aircraft is on a collision course. The most widely used, basic parameter is the so-called "Tau" criterion, defined as

$$\tau(\text{tau}) = \frac{\text{measured range between aircraft}}{\text{range rate between aircraft}}$$

Thus, τ becomes the direct measure of time-to-collision for nonmaneuvering aircraft. With the system under current consideration a false alarm rate substantially greater than zero is possible (for certain noncollision situations) which is to be preferred over the possibility of not detecting a true collision situation. In addition, it is possible that aircraft in a stack could change positions so that a warning of collision would be indicated to at least two aircraft. The subsequent maneuvers may produce other collision warnings to adjacent aircraft, thus sending a ripple through the entire aircraft stack. Aircraft within line of sight of each other will receive information and will be processing such information constantly, unless simple criteria are established to prevent needless computations and consequent large CAS computational requirements. Some form of gate-limiting will alleviate this last problem. A CAS system based upon the use of multiple satellites will not have these problems.

6.3.7 Collision Avoidance Maneuvers

Since a warning from such an independent CAS must involve action outside of ground control, it is important to minimize the system false alarm rate. The system must be designed to provide adequate warning time to allow for relatively benign maneuvers, i.e., maneuvers involving minimum distrubance of the aircraft flight path and minimum likelihood of generating chain reactions involving other aircraft. In the past, an effective means of avoiding collisions, other than through Air Traffic Control, has been the passive control incorporated in the Civil Air Regulations (CAR) prescribing altitude separation between aircraft flying in opposite directions.

A straightforward way to approach the problem of maneuver design is to think of the CAS as a direct extension of the pilot. Thus, the system should emulate the natural response of the pilot to make small corrections to the aircraft flight path whenever an incipient close approach to another aircraft occurs. The pilot determines this first by observing that a close aircraft is coming closer (the subjective measure of closeness being a function of the relative velocity of the two aircraft), and second, by observing that the line-of-sight to the other aircraft is changing slowly, or not at all, relative to his own flight path. With enough warning time, the pilot will change heading slightly to provide adequate separation between the two aircraft at the point of closest approach (assuming that the other aircraft maintains course). To ensure that complementary maneuvers on the part of two aircraft do not exacerbate the situation, Civil Air Regulations provide that two aircraft on anti-parallel courses both change course to the right. For other approach geometries, the CAR defines the right-of-way conditions; for purposes of collision avoidance, these could be supplemented with rules that direct each aircraft to turn in a direction to increase the minimum distance at the point of closest approach. If the pilot is changing altitude when he first becomes aware of a potential close approach to another aircraft, he will prefer to change heading and maintain altitude rate. Changing altitude or altitude rate will be a secondary maneuver to be used only if, for some overriding reason, heading changes are prevented.

An ideal CAS would prescribe collision avoidance maneuvers similar to those which the pilot would make ordinarily. As soon as the system detects a potential threat, it would indicate a maneuver to the pilot to increase this separation. If the CAS range is adequate to provide 30 seconds of warning time before the point of closest approach, the specified maneuver can be a relatively modest heading change.

As an example, consider the following incipient collision condition. Such a condition exists when the relative velocity (the vector difference between the two aircraft velocity vectors) is parallel to the instantaneous line-of-sight (the range vector) between the two aircraft and both aircraft are either in unaccelerated flight or exhibit no components of acceleration normal to the line-of-sight. For unaccelerated flight, the time-to-collision (τ) is defined as the instantaneous range between the two aircraft divided by the relative velocity. To avoid the collision, the relative velocity vector between the two aircraft must develop a component in the direction normal to the line-of-sight. The minimum distance at the point of closest approach is a direct function of this normal component of the relative velocity. To provide a minimum acceptable separation of 1/2 nmi, it is necessary for the relative velocity vector to have a 60 knot (1 nmi/min) or greater component in the direction normal to the line-of-sight for 30 seconds. When the CAS indicates an impedning collision in 30 seconds, the pilot must maneuver the aircraft to provide a 60 knot component of velocity normal to his line-of-sight to the other aircraft. If an airliner traveling at 500 knots, with a line-of-sight 45 degrees from his flight path, is on a collision course with another aircraft and has 30 seconds of warning time, the pilot must change heading 10 degrees to assure a miss at 1/2 nmi. If the line-of-sight is head-on, then the pilot must change heading only 7 degrees to achieve the same results. In the worst case, if the threat aircraft is 90 degrees abeam and the CAS indicates 30 seconds to collision the pilot must change heading by 28 degrees, or, more simply, change speed by 60 knots without changing heading, to miss by 1/2 nmi.

6.3.8 Summary of Satellite CAS-Based Features

Since broad-area coverage can be provided easily by a system of NTC Satellites, the design of a system for collision avoidance based on the use of satellite-transmitted signals appears very attractive. Since the number of dedicated satellites required is small, the space segment cost of the CAS is minimized, and the per user cost becomes very low. System compatibility is assured immediately since all aircraft in the areas of coverage would use the same equipment. Single-satellite CAS features are summarized in Table 46. Multiple-satellite CAS features are summarized in Table 47.

Table 46. Single Satellite CAS Features

- Eliminates need for accurate user clock, multiple ground stations
- No satellite ephemeris required
- Time gating, altitude decoding, and signal doppler could be used to reject unwanted signals
- Requires high rate of data transfer between users

Table 47. Multiple Satellite CAS Features (Simultaneously Visible Constellation)

- Uniquely specified threat aircraft position and velocity in 3-dimension space with highest accuracy
- Complete CA service at low cost using signals from a navigation satellite system
- Order of magnitude less bandwidth required for information transfer between users (existing aircraft VHF communications)
- Negligible false alarm
- Independent of ground-based ATC

6.4 POTENTIAL USE OF A NAVIGATIONAL/TRAFFIC CONTROL SATELLITE SYSTEM BY FISHING AND OCEANOGRAPHIC VESSELS

This section briefly summarizes the potential ways in which fishing or oceanographic vessels might utilize the Navigation/Traffic Control Satellite System.

6.4.1 Central Fishing Boat Navigation

Navigation to and from the fishing grounds is now accomplished with the aid of LORAN and other radio direction finding systems, celestial navigation, radar, and dead reckoning. LORAN has a limited range, and is not installed along many nations' coastlines. For instance, LORAN does not exist south of the Mexican border, and a very substantial part of our fleet operated below the border. A severe handicap associated with celestial navigation is weather — fog or rain can easily prevent a fix from being taken. Fishermen claim (Ref. 12) a normal celestial navigational accuracy of around three nautical miles (one mile at best) providing they can see the stars; and state that generally this is adequate for the purpose of getting to and from the fishing gr ids. In a study for NASA, Campbell (Ref. 13) pointed out that most navigation requirements do not need great precision, "in fact a navigational error of one nautical mile with 0.95 probability available in any part of the world and under all weather conditions would be adequate." Most of the present navigational problems associated with tuna boat daily fishing operations (such as finding or returning to a good area, hazard avoidance, longline fishing gear avoidance) can be handled satisfactorily with existing navigational gear; however, "improvements are very welcome." (Ref. 14).

In the opinion of those contacted, (Refs. 12, 13) additional navigational aids, such as a satellite, would be very useful and valuable particularly in the many areas where LORAN is not available and when weather obscures the stars.

Navigational fixes are taken approximately four or five times per day. The number of possible users is subject to speculation; however, 2000 is a rough estimate of U.S. fishing boats along the entire West Coast. Assuming they each want a fix five times per day, this is a rate of 10,000 requests for a fix per day. As a minimum, if it is assumed that the

130 U.S. high seas tuna boats each want a fix five times per day, then the fix rate is 650 per day. (Ref. 14).

A projected future need for more exact navigational data is in relation to future fisheries forecasting systems and aerial surveillance systems as discussed in Sections 6.4.3 and 6.4.4.

6.4.2 General Fishing Boat Communications

In general, the high seas fleet can now communicate to the U.S. by voice (two-way radiotelephone) from most of their fishing grounds. However, communications from West Africa are most difficult. Apparently the major problem is not distance, but very heavy message traffic. Major two-way voice radiotelephone stations are WOM in Miami, Florida, and KMI in Oakland, California. Each boat transmits several messages per day to the U.S., each lasting several minutes. The boats use a single sideband transmitter. Due to the heavy message traffic, the assistance of ham operators is used often.

Message traffic by voice, assuming two messages per day, each lasting two minutes for each of the 130 high-seas U.S. tuna boats, amounts to 520 minutes per day. These messages are generally not urgent and could be placed during an off-peak communication hour. For example four channels allocated to this application for four hours a day, operating at a 55 percent utilization rate would handle this load.

As an example of how another nation might use the satellite communication channels, the Japanese longline fleet (500 vessels) now communicates each day with the home islands at a rate of 1400 messages per day (400 contain catch statistics and 1000 contain administrative, social, business and weather messages. This message traffic rate completely utilizes the allotted frequencies and time. A high precentage of the message traffic is navigation oriented in the sense that it discusses where the fleet has been, where it is going or should go next and why, based on catch statistics, when boats will be returning home (how many fish have been caught), and where good areas are in order to vector in more boats. The transmission is now by code (telegraph key operated by radiomen) and each of the 1400 messages average around two minutes.

The 400 catch statistics messages each consist of very basic data amounting to only around 180 bits. For 400 messages, this is equivalent to 72,000 bits per day, which at a transmission rate of 1200 bits/second, would utilize only 1 minute per day of a data channel.

The Japanese fisheries forecasting system could probably put to effective use about three to ten times the currently transmitted data from catching vessels. If transmitted automatically at 1200 bits/second, this would require only three to ten minutes of satellite data channel time for the entire fleet. Current constraints appear to be speed limitations of transmission and reception equipment and personnel. Although it would be clearly advantageous to have higher data transmission rates, tradeoff studies to determine the optimum combination of equipment and personnel have not yet been conducted.

6.4.3 Future Fisheries Forecasting Systems

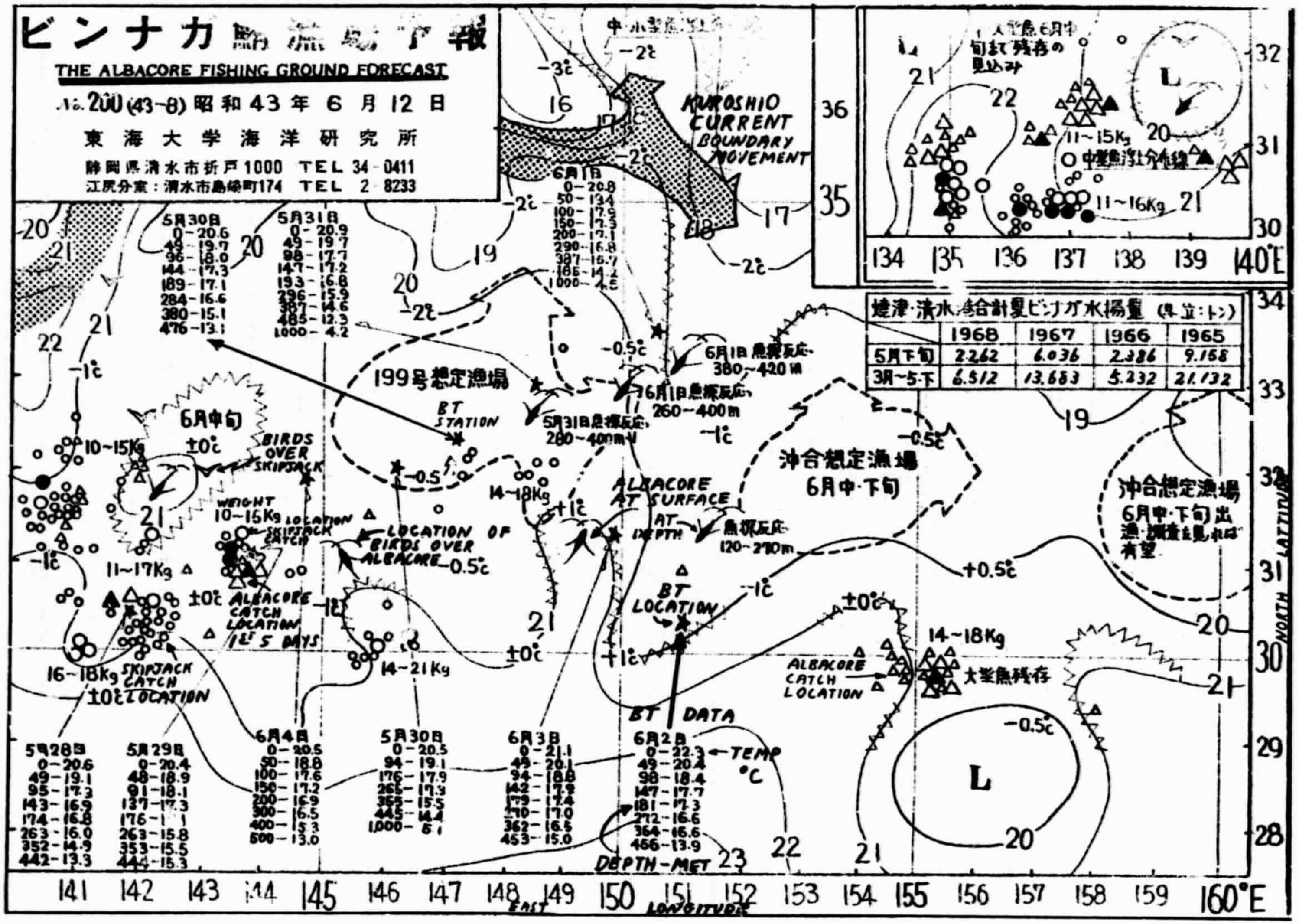
In the future, fisheries forecasting systems will probably be comprised of many integrated platforms and sensors to gather data, correlate it, and utilize the results to better direct and manage fishing effort. System elements might include satellites, aircraft, advanced communications and navigation equipment, computers, oceanographic research ships, exploratory fishing vessels, and catching vessels. It would be a definite advantage in integrating such a system if they all used a common navigational grid and had access to the same communications network.

It will be necessary for research ships (which might be gathering data, for instance, on such factors as chlorophyll concentration, larvae taxonomy and volume, micronekton volume, water column thermal structure, salinity and dissolved oxygen) to know their precise location if the information they are gathering is to be correlated with related data gathered simultaneously. For instance, recent measurements by TRW personnel several miles off the coast of California (Ref. 15) have indicated that a boat and an aircraft attempting to simultaneously record data related to the ocean color of a specific ocean station must obtain data from an area within a tolerance of roughly 100 yards. This tolerance may loosen up considerably (to the order of a mile or so) in open ocean areas quire far offshore, because it is likely the parameters being recorded will not vary as rapidly near shore.

An example of a recent Japanese forecast for their Albacore tuna fishing grounds location is included as Figure 74. This figure is included to show the wide distribution and relative density of the worldwide Japanese fishing fleets. It is seen that the location of specific observables (birds, fish schools, etc.) is given within the range of roughly five n mi and the location of current boundaries and movements is given within about 50 miles. A monthly summary of catch statistics for the Japanese highseas fleet (April 1968, 439 longline vessels, primary catch is tuna, showing the location, intensity, and success of their fishing operations is included as Figure 75. A navigation/communications satellite would be of definite use to such an extensive system. In fact, it was mentioned (during a recent visit to Japan by TRW personnel regarding Japanese fisheries oceanography services (Ref. 16) that Japan may be in the market for a satellite navigation and/or communications system in the next few years.

6.4.4 Coordination of Fishing Effort

If a large area fisheries survey aircraft overflight system — now just conceptual — is actually implemented, a considerable coordinating effort will be required in the area of integrated navigation and communications. A map of the exact cruise track, annotated with the observed fish schools, isotherms, slicks, etc. will require precise navigational aids if it is to be utilized by fishermen. As the aircraft continues on its cruise track, the transmission of the exact location of observed fish schools to the fishing fleet may allow a nearly real-time utilization of the data. The results of some very preliminary assumptions and calculations (velocity = 150 knots, daily sweep area per aircraft = 5000 sq mi, total U.S. West Coast sweep area 50,000 sq n mi, number of aircraft = 10, nav. fix = 6/hr) indicate that a navigational fix rate of 60 per hour for the system seems reasonable for the assumptions given. It would be advantageous if the location of the fish schools could be relayed by the aircraft to the catching vessels within a navigational tolerance of approximately ±one nautical mile. Less than this tolerance is not really necessary because a school can usually be sighted by eye from the ship within a mile, and sometimes from a distance of up to 7 or 8 miles if many birds are feeding on the fish school, and if there is good visibility.



焼津·清水港合計ビンナガ水揚量 (単位:トン)	1968	1967	1966	1965
5月下旬	2.262	6.036	2.386	9.158
3月~5下	6.512	13.683	5.232	21.132

Figure 74. The Albacore Fishing Ground Forecast

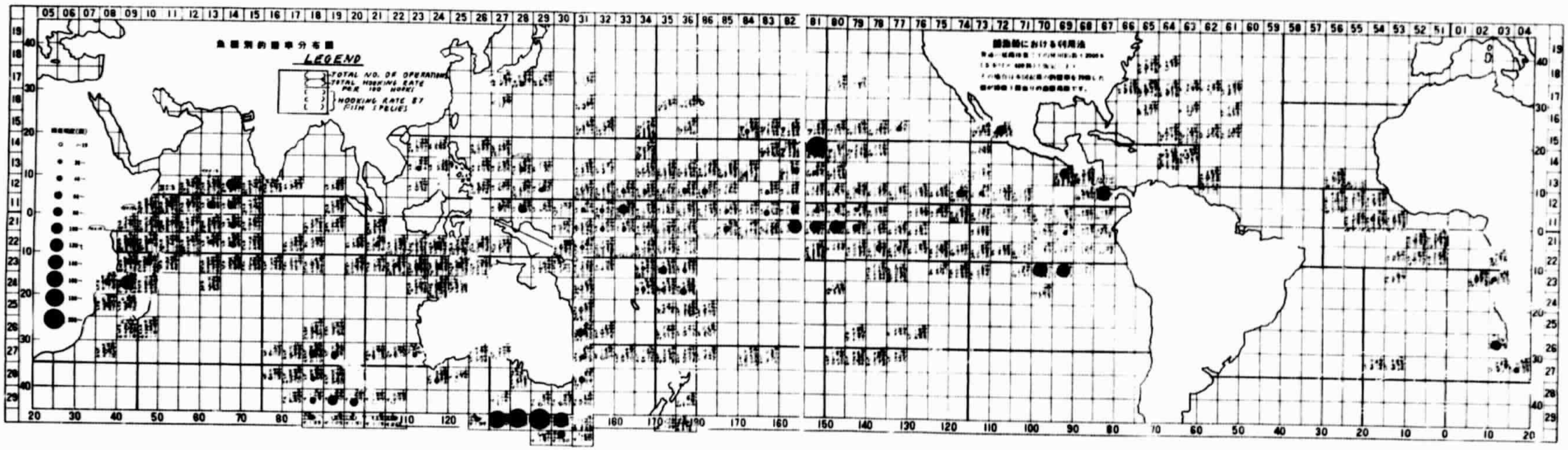


Figure 75. Tuna Catch Statistics From the Japanese High Seas Fleet (Longline) for April 1968

Data transmission requirements (for real time utilization of the aerial survey by fishing boats) would be highly variable — depending largely on the number of schools detected. For the same assumptions given above, a system of ten aircraft each reporting on the average (every ten minutes on a data channel) latitude and longitude of schools sighted along with basic data (school size, species and other observables such as location of slicks, current boundaries, birds, would require time to transmit a total of around 172,000 bits per eight hour overflight day. If a transmission rate of 1200 bits/second can be assumed, this is equivalent to a total channel time of about 2.4 minutes per day. Peak transmission rates may be considerably higher than this. For instance, in recent flights over the Gulf of Mexico the Bureau of Commercial Fisheries reported 150 fish schools sighted along one 40-mile stretch, and 200 more schools sighted in a 100-mile stretch (Ref. 17). If the basic data corresponding to the location of each school was transmitted as each school was sighted, this would require a maximum transmission of approximately 54,000 bits per aircraft per hour.

Multi-platform coordination by future fishing systems may benefit from the accurate fix provided by a NAVSAT. If several platforms are used (search aircraft, capture vessel, helicopter, etc.) by future fleets in the search and capture mode, then an exact fix on a school sighted, for instance, by a helicopter, would be useful to the catching boat.

6.4.5 Trawl Fishing Guidance

In some instances a trawler attempts to return to exactly the same location of a previous large catch, and a precise fix (± 100 ft) on this location would be useful. This need not always be an absolute navigation requirement. Since the boat has usually been at the location of interest within the last hour the return might be handled relative to its previous track.

6.4.6 Regulation of Fishing Boundaries

The NTC Satellite System could be used in a surveillance role to verify the locationof fishing boats, and therby establish if any boundary violation had or had not occurred. (This would require active cooperation of all units since a navigation or cummunications satellite does not have a surveillance capability.) The communications channels might relay the position of all the U.S. ships to the Bureau of Commercial Fisheries, or to international agencies such as the Inter-American Tropical Tuna Commission. In addition to verifying or regulating boundaries, these agencies would benefit from a synoptic picture of the daily fishing effort.

Another use for a NAVSAT would be to establish and verify the location of boats in a regidly enforced grid pattern such as is used by the Japanese North Pacific salmon fleet. The entire North Pacific is divided into grids (about 120 sq mi) and each of the eleven fishing groups (which consist of one Mother Ship and its 28 catching vessels) is allowed to occupy only one grid at a time. Also, the spacing between catching vessels must be at least 8 km. A navigational/traffic control satellite would be quite effective in this case for establishing and verifying each boat's position in relation to the required grid location and the 8 km spacing.

6.4.7 Fixed-Point Oceanographic Data

Moored buoys eliminate spacial uncertainty but the cost of mooring and maintaining them is high. Also, instrumented buoys have a limited data acquisition capability; for instance, they cannot record biological information. A NAVSAT would enable a shop to return to exactly the same ocean station for repeated oceanographic data acquisition. Satellites and on-board computers are now used to get an accurate fix for the correlation of geomagnetic anomaly with topographical data. Neither moored nor drift buoys can chart the progress of shifting oceanographic phenomena such as current boundaries, upwellings, plankton blooms, etc. Ships can track these at great expense by remaining in the area. For aircraft to monitor the progress of such phenomena on a periodic basis good navigational fixes would be required.

The advent of navigational satellites and very accurate navigational fixes will probably bring about the revision of many topographical maps.

6.4.8 Drogue or Free-Drift Buoys

A system of buoys free to drift with the currents to record and transmit data related to fisheries has been hypothesized. These buoys might be launched at specified time intervals in the Kuroshio current near Japan, and would move north with the Kuroshio, and the Subarctic, or North Pacific currents, eventually traveling south again along the U.S. West Coast in the California current, and possible returning in the North Equatorial current. Such buoys would gather and store data and then transmit it when interrogated by a satellite. A combination navigational and communications satellite would be ideal because it would get an accurate fix on the buoy and relay the data as well. Since this sytem is now purely conceptual, information on the number of buoys and data rates is unavailable.

6.4.9 Offshore Oil and Dredging

The offshore oil industry is on the threshold of tremendous growth. Prospecting activity is worldwide with most of the earth's 10.8 million square miles of continental shelves still unexplored. Estimates have been made that only 7% of the continental shelves which have good possibilities for oil discovery have been surveyed. It seems highly probable that a very accurate navigational aid such as a NAVSAT will be useful to offshore oil survey and drilling operations in determining the specific location of bottom and sub-bottom phenomena, and for returning to the exact site of previous investigations. A NAVSAT will be particularly useful in areas beyond the normal range of LORAN, or where LORAN is not available.

One possible application of a NAVSAT might be the exact relocation of previous drilling or experimental sites. Such sites can be marked with a very short range underwater sound source (pinger) to enable it to be located once a boat is in the immediate area (several hundred feet) and to prevent competitors from locating the area. A NAVSAT might be used to navigate within the range of the pinger. Also, a passive navigational system might be desired if the boat location is not to be revealed.

6.4.10 Search and Rescue Operations

Two important factors must be considered in search and rescue operations: (1) the lives of those involved in the mishap, and the recovery of valuable or "sensitive" hardware; and (2) the relative scope and cost of the search operation. A device onboard ships that in an emergency, would continuously transmit a distress signal and the ships locations, as precisely determined by a navigational satellite would greatly enhance the probability of a successful, low-cost rescue.

The emphasis is often shifted away from the latter factor — cost — when human lives are at stake; however, the cost can be surprisingly large. The search for a single aircraft downed in the Pacific in 1964 cost well over $1 million. Estimates have been made which indicate that over 27,000 search and rescue hours have been flown by the Air Force, Coast Guard, and Navy during a 12-month period. It is difficult to estimate the exact costs since these missions were flown by many different aircraft, however the total annual cost is well up in the millions — on the order of 10 million, possible 100 million — depending on what is included in the accounting.

The loss of preperty and lives is substantial also. In 1964, there were 2,308 casualties at sea involving 3,178 vessels. Of these, 390 vessels were totally lost, 191 people died, and 133 were injured. (Ref. 17) These losses were caused by collisions, explosions, fire, material failure, grounding, weather, navigation error, and personnel error, and totalled $68,355,000.

In addition to saving lives, it may in fact, be quite cost-effective (in terms of reduced property loss and search cost) to have an automatic location and distress signal transmitter on-board that obtains its exact position from a NAVSAT, and perhaps transmits its distress signals through the NAVSAT communications channels. Or the on-board equipment could be simple a device that requests a NAVSAT to relay the information necessary to determine the boat's position to the Coast Guard or some other rescue agency.

The ability to pinpoint the exact location of a sinking vessel or a downed aircraft has very significant implications to the military, of course. If an on-board device could call up an automatic navigational fix from a satellite which was then automatically relayed by the satellite to a rescue operations center it would increase the chance of rescue, and decrease the probability that enemy forces would get there first. Another advantage of a precise fix is that a downed pilot, for instance, can remain passive (that is, once downed, he may elect not to use a "beeper" or radio transmitter) unitl he is rescued — thereby avoiding enemy detection. Also, in some instances, in order not to attract attention when conducting clandestine operations, it may be extremely desirable not to have to launch a large scale area sweep search operation. An exact navigational fix on the rescue location would eliminate the need for any extensive search operation. These above problems were discussed by senior Navy officers and TRW personnel on board the carrier Kearsarge during operations in the Gulf of Tonkin in 1966.

6.4.11 Underwater Storage

Many fleets which operate worldwide but have limited access to shore based facilities would like to emplace undersea stations for fuel and other supplies. These might be located on sea-mounts far from any shoreline and would require precise navigation (several hundred feet) in order to consummate reasonable fast locating and transfer of material.

6.5 THE POTENTIAL APPLICATION OF THE NAVIGATION/ TRAFFIC CONTROL SATELLITE SYSTEM TO SOLAR FLARE MONITORING AND WARNING

6.5.1 General

The effects of solar flare eruptions on manned aerospace vehicles, such as very high altitude aircraft and orbital vehicles, fall in two general categories: (1) radiation damage to materials both biological and inert and (2) perturbations of the near earth space environment, including radio blackout from changes in the upper atmosphere. For these reasons, it is desirable to predict when these eruptions might occur and to monitor their occurrence in order to determine the extent of the effects on aerospace operations. Particularly, with the advent of high altitude commercial airline operations, the public will demand that potential hazards to SST passengers from solar radiations be virtually nonexistent. Necessarily, the prediction system must be highly reliable, and the avoidance measures must be unperturbing to the passengers and not prohibitively expensive for the air carrier. The problem of solar flare effects and warning, including a description of the phenomenon, follows.

6.5.2 Description of Solar Flare Phenomena

The effects of solar flares have long been noticed on the earth in the form of geomagnetic storms, polar cap absorptions, sudden ionospheric disturbances, and other geophysical effects. It was known that these effects had some correlation with sunspot activity, but details of the processes were not known. Recently, high energy solar flare particles have been observed (Reference 18). These particles were determined to be protons with kinetic energies up to 30 Bev. With the advent of satellite, balloon, and rocket probes, detailed and rigorous investigation of solar flare phenomena is being undertaken.

The description of all solar flare processes would be exceedingly complex. A very general discussion will be presented on effects such as radio noise emission and optical and X-ray emission. The majority of this discussion will briefly treat solar cosmic ray production, propagation, and composition.

The development of solar flare phenomena as seen from the earth is shown in Figure 76. At time 1 hour, the flare is observed usually to

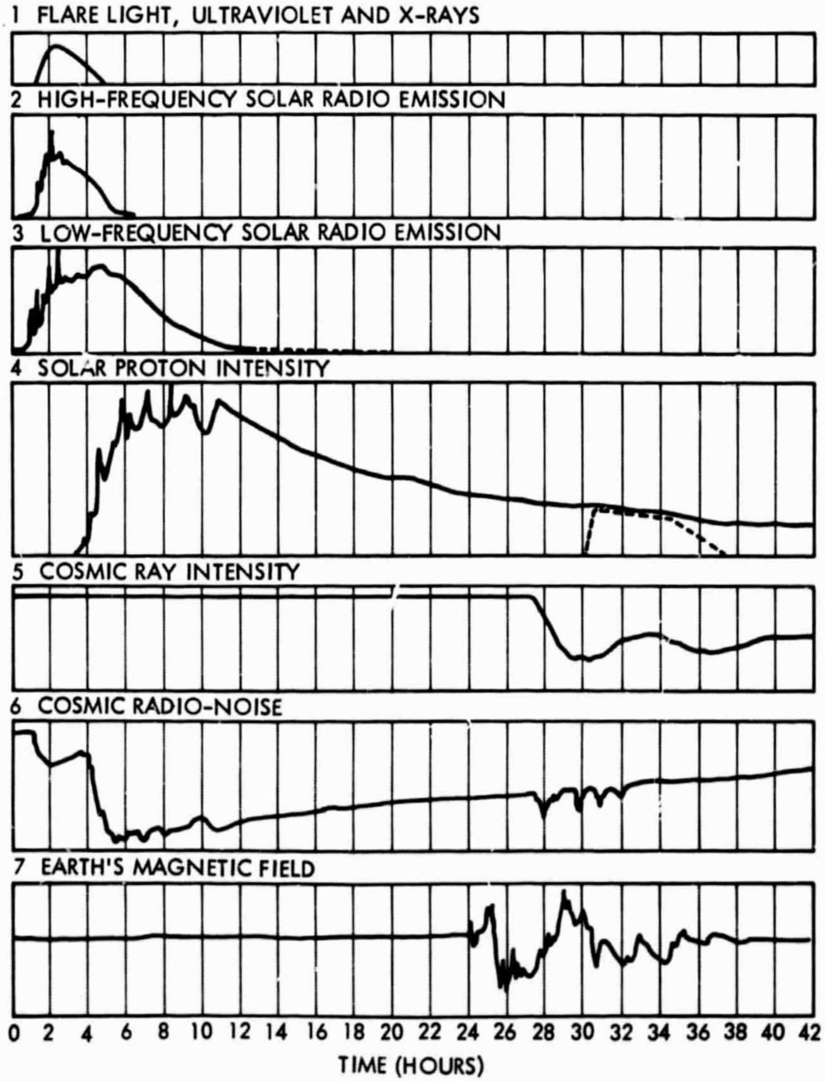


Figure 76. Solar Flare Sequence of Events

emit light in the optical, ultraviolet, and X-ray regions of the electromagnetic spectrum (Profile 1). These emissions rise rapidly to a peak and then die off within a few hours. At the same time, bursts of high frequency radio emissions are observed (Profile 2), exhibiting approximately the same rise and decay profile as the optical emissions. Low frequency emissions (Profile 3) also occur in burst form and with a total intensity rise time similar to the previous effects, but the maximum of the profile is longer, and the decay in time is slower, persisting for as long as 12 hours.

The solar protons (Profile 4) are observed to arrive over the poles of the earth some time after the electromagnetic observations of the flare event, this delay being due to propagation times greater than that required for electromagnetic radiation. The solar proton intensity is observed to have a sharp rise time with irregular fluctuations near the maximum and then to decay slowly with time, the effects still being observable for days afterward.

Profile 7 shows that the earth's magnetic fiel is undisturbed by the previously discussed effects (Profiles 1 through 4). However, at some time which may be 24 hours after the observed flare, the geomagnetic field suffers violent fluctuations. This effect is due to a solar plasma of low energy charged particles (protons and electrons) colliding with the earth's field and causing it to contract, deform, and spring back. The effect of the perturbed geomagnetic field on solar proton arrival is indicated in Profile 4. The structure observed at times 30 through 38 hours depicts the arrival of solar protons over regions on the earth at low latitudes, where they were previously forbidden due to the geomagnetic cutoff of the solar protons. Their arrival is now allowed since the geomagnetic field is deformed, and the particles may penetrate to normally forbidden regions. The propagation delay time for these particles to reach earth is highly variable. This time variation is attributed to the variable conditions of the solar magnetic field structure in interplanetary space.

The galactic cosmic ray intensity, Profile 5, follows the same general form as does the earth's magnetic field. The decrease in cosmic ray intensity is caused by the galactic rays being deflected by magnetic fields "frozen" in the solar plasma. Thus, when the solar plasma has reached the earth's field and perturbed it, many of the galactic cosmic rays in the vicinity of the earth cannot penetrate both the plasma and the field, and thus the previously observed intensity decreases. As the solar plasma passes the earth, both the geomagnetic field and the cosmic ray intensity recover and approach normal levels.

The cosmic radio noise (Profile 6) is affected by arrival of three emissions from the sun, i. e., ultraviolet and X-rays, solar protons, and solar plasma. Cosmic radio noise refers to the emission of radio signals by distant starts which are received and monitored at the earth. Since these radio emissions must penetrate the ionosphere, the conditions of the ionosphere greatly influence their propagation. Coincident with the emission of X-rays from the sun (allowing for a time of flight to the earth of 8 minutes), the reception of cosmic radio noise is decreased (times 1 through 4 hours). This is due to the increase in ionization in the ionosphere caused by the solar ultraviolet and X-rays, making it more opaque to transmission of the radio signals. This opacity is further increased by the arrival of the solar protons (times 4 through 24 hours) which produces additional ionization. The fine structure fluctuations in the recovery of the radio noise reception at time 24 through 32 hours is due to the solar plasma disturbing the field and allowing additional particles to be dumped into the ionosphere, producing more ionization and thus increased opacity.

It can be seen that solar flare events, while complex in themselves, can produce varied geophysical phenomena and influence greatly the space environment. This description of a flare event is typical in terms of the various observed phenomena. However, the occurrence of specific events show great time variations from one flare to the next. This implies that Although visible observation can detect a flare, this is not enough to accurately determine when, or whether higher particles will reach earth. In addition, there have been surprise events. High energy particle fluxes have been observed without a corresponding visible flare, since the specific flares involved were on the far side of the sun.

6.5.3 Effects of Solar Flares

The allowable biological dose rates for crew members or passengers on very high altitude aircraft have not been firmly established, but will probably closely follow the guidelines used by the Atomic Energy

Commission (Table 48) for control of radiation workers. It should be noted that the FAA considers the solar flare hazard to be significant to the extent that they have procured dose monitoring instruments which are currently being flown in high altitude aircraft to investigate the levels of radiation.

Table 48. AEC Radiation Guidelines

	Per Hour	Per Week	Per Year
Radiation Workers	2.5 mrads	100 mrads	5 rads
Nonradiation Workers	0.25 mrads	10 mrads	0.5 rads

Solar flares can affect near-earth communications due to perturbation of the ionosphere by radiation from the flare region. The ionospheric perturbations follows the time profile noted as item 6 (Cosmic Radio Noise) in Figure 76. It can be seen the blackouts consist of three major phases — the first due to flare X-ray bursts, the second due to arrival of flare protons, followed by a third phase which is a slow recovery with additional perturbations at later times due to arrival of the flare plasma perturbating the magnetospheric cavity. The carrier frequencies affected are usually in the VHF range up to 300 MHz.

The radiation at high altitude (80,000 feet) due to galactic cosmic rays is approximately 5 rads per year (Reference 19). Assuming 40 hours of crew flying time per month, not all of which are at 80,000 feet, the maximum accumulated dose would be about 0.27 rads per year. This is about one half the allowed dose for nonradiation workers. The high energy solar protons may present a greater hazard. The magnetic field of the earth provides a protective shield over most of the earth's surface. The unprotected areas consist of those portions in the vicinity of the magnetic poles. The atmosphere provides a second protective shield at 80,000 feet. There are approximately 30 gm/cm^2 of air above that altitude (Reference 20). This requires a proton energy of greater than 200 Mev to penetrate to this altitude (Reference 21). The geomagnetic latitude below which 200 Mev protons cannot penetrate the magnetic field is about 65 degrees. Thus, the danger area to high altitude commercial aircraft from solar flare protons is limited to that defined by a circle 1,500 nautical miles radius centered at either magnetic pole. In the Northern Hemisphere, this circle passes through or near Nome and Seward, Alaska; Seattle, Washington; Minneapolis, Minnesota; Ottawa, Canada; Cartwright, Labrador; and Reykjavik, Iceland. If this region were restricted from SST flights due to solar flare proton radiation, it would affect the polar routes such as New York to Moscow or San Francisco to London.

Assuming a 200 Mev energy cutoff due to the atmospheric shielding, the highest measured solar flare proton flux of this energy or greater is about 2×10^3 protons/cm^2-steradian-sec (Reference 22). If all this flux penetrated to the flying altitude, the dose rate would be on the order of 0.5 to 2 rads per hour. It can thus be concluded that there is a potential hazard to passengers and crew of an SST over the polar regions in terms of exceeding the AEC guidelines for radiation dose. However, it should be noted that the amount of shielding by the atmosphere can be more than doubled by going from 80,000 to 65,000 feet (Reference 22). Secondly, a flare of the magnitude discussed above has only occurred once in more than two decades. Perhaps a more significant effect of the flare is communication blackout at high altitudes, particularly over the magnetic poles. The communication blackout can be a troublesome phenomenon for SST flights.

6.5.4 Prediction

Considerable effort has been exerted to develop a prediction theory, to little success. Perhaps one most worthy of mention was developed by Dr. R. Head of NASA/ERC, which has evidenced promise in long-term prediction using planetary positions as a function of time to predict when gravity gradients might catalize an active region on the sun to produce an eruption. However, even this technique is far from perfect.

It appears quite realistic to monitor solar proton fluxes and energy spectra buildup in real time and process this data using a small ground computer to provide immediate information to air traffic control operations for use with very high altitude aircraft. It is of interest to note that the length of time from first detection of solar protons to the time when the radiation dose hazard is a maximum is on the order of several hours for most flares. Synchronous equatorial orbits are adequate for this mission; however, an inclined synchronous orbit would provide better proton data due to reduced effects of the earth's magnetic field.

6.5.5 Sensing

From a monitoring point of view, two large government networks are currently in operation: the Air Force SOFNET and the NASA SPAN. These nets depend on monitoring H_{α} radiation and microwave frequencies and are not reliable for predicting solar flare particle-producing events. It is these particle events which produce radiation damage and are responsible for the second phase of the radio blackouts. (The X-rays from the burst produce the first phase.) With regard to solar flare monitoring and prediction, Mayfield points out that data on solar flare protons, notably flux and energy spectra, are not available and are extremely important in assessing the magnitude of a solar event. It is therefore recommended that a solar proton detection device be placed on the NAVSTAR spacecraft to provide a real-time monitor of flares which can bedirectly used by aircraft operations groups to control high altitude aircraft flight profiles, and will also be a very welcome input to the SOFNET and SPAN solar monitor networks.

Several instruments to monitor solar protons have been developed and flown by TRW Systems group. Based upon this extensive experience, the appropriate proton monitor would measure the omnidirectional flux of protons in the energy ranges $1 < E < 10$ Mev, $10 < E < 20$ Mev, $20 < E < 50$ Mev, $50 < E < 100$ Mev and $E < 50$ Mev. This instrument would have dimensions of 3 x 5 x 7 inches, weigh 3.5 pounds, consume 0.75 watts of power, and perform with high reliability over a 3 to 5 year period.

The ESSA solar flare warning system utilizes inputs from SOFNET and SPAN as well as selected data from spacecraft such as Pioneer and Vela. The data derived from the Pioneer and Vela spacecrafts are comprised of crude X-ray measurements (Vela), some plasma measurements, and some crude proton measurements which are generally reported as some factor above a background level. This proton data does not contain spectral information, nor is absolute flux readily available for use by the flare networks. Some attempt has been made to make this spacecraft data available to the network in near real time; however, the network still suffers from lack of good proton data available on a timely basis. Again, the present nonspectral proton data is not useable in real time for a projection as to the anticipated intensities of an event, and is not at all useable for relating to biological dose rates.

6.5.6 Collection and Distribution

Although there are a number of spacecraft in orbit with instrumentation capable of monitoring solar flare activity, the primary problem has been acquiring, reducing, interpreting, and disseminating the information within a time span that will effect a meaningful flare warning. An effective flare warning system requires continuous monitoring of the sensor spacecraft, whether or not the spacecraft has a flare-initiated transmitting system. Additionally, the flare warning system must maintain continuous open-line communication among the monitoring ground stations in order that the flare warning be promptly transmitted to the affected air carriers in time for evasive routing. Clearly, the principal effort in establishing an effective solar flare warning network lies in setting up the collection and distribution system, including the communication network.

6.5.7 Navigation/Traffic Control Satellite Application

A desirable alternative or addition to an elaborate solar flare warning system is to incorporate a solar flare monitoring subsystem in the Navigation/Traffic Control Satellite. The warning data could then be transmitted directly to the aircraft communication network for immediate dissemination. The flare monitoring subsystem on the satellite would necessarily require a data processing unit which would be able to discriminate between the characteristics of a potentially dangerous flare and a harmless one. With present knowledge of the characteristics of solar flares, perhaps the most that could be achieved is an indication from the sensor subsystem that a flare has occurred, with some indication of its magnitude. This would warn pilots and airline officials that a significant flare has occurred, and there there is a potential danger. The very important advantage of incorporating the flare sensors in the navigation satellite is that the warning would be immediate and direct to the aircraft, avoiding the warning delays involved with communication networks and agency coordination.

The 52.5 degree inclined configuration B satellite when it is near apogee will, because of its high latitude and altitude, be near the boundary of the earth's magnetosphere and will therefore be in the best position to detect the X-ray and proton events. In addition, this is an ideal location from which to transmit this data to aircraft flying at the more dangerous high latitudes. False alarms based on ground-based optical flare detection techniques and surprise events — an increase in particle flux values in the absence of observed solar flare activity — would both be accounted for with this system.

Specifically, such a flare monitor subsystem would include as sensors:

- Bremsstrahlung X-ray sensors (20 to 100 kev), and
- High energy proton sensors (E > 200 Mev).

The data from these sensors would be processed by an onboard processor which would evaluate the sensor outputs and provide an indicator signal. This signal could be transmitted to the aircraft in conjunction with the navigation data. This should be interpreted as a potential danger, and an immediate evaluation of the magnitude and characteristics of the flare could be made by the more elaborate flare warning systems, or the signal from the satellite could be transmitted to the air traffic control centers, where it would simply become a part of the air traffic control function. The associated change in flight plan and instrument flight rules clearance could be a part of or closely tied to the warning itself.

6.6 THE POTENTIAL APPLICATION OF THE NAVIGATION/ TRAFFIC CONTROL SATELLITE SYSTEM TO SPACE NAVIGATION

6.6.1 Introduction

Studies show that earth synchronous satellites could provide accurate navigation fixes to ships, aircraft, and other terrestrial vehicles. The knowledge (and stability) of the locations of synchronous satellites admits a number of methods of making the fixes through use of ranges (trilateration), range differences (hyperbolic navigation), or angular measurements. It is possible to extend the use of the Navigation/Traffic Control Satellite System to space missions, both earth-orbital and lunar; such utilization of NTC Satellites would provide manned spacecraft with a simple yet precise onboard navigation system superior to an optical system and independent of an earth-based tracking and computing system*. A synchronous satellite navigation system, in fact, can achieve some significant advantages over the ground-based navigation (i. e., tracking) system for manned spacecraft because of the more extensive geometry provided by the 22,000 mile orbital radius of the satellites. Put very simply, any system which can provide accurate ranging to a spacecraft from a wide-stance geometry while requiring small, light onboard components will be a very attractive solution to the onboard space navigation problem.

6.6.2 Error Analysis

A simple error analysis of a ranging NTCS which would provide onboard navigation data to an Apollo spacecraft returning from the moon was made. Many simplifying assumptions have been made, but the potential applicability of the approach will be apparent.

A trajectory was generated (using an Apollo 8 reference trajectory point) which began at a distance of about 190,000 n mi from the earth, on a return from a lunar mission. The trajectory point was propagated forward, under the assumption of earth gravity only, for 65 hours to a point

*The satellites themselves must of course be supported by such a system for ephemeris determination but it is much simpler than, say, the Manned Space Flight Network (MSFN) now used for Apollo tracking.

at 3650 n mi altitude (latitude 25° S). This point represented trajectory termination, although further segments (into reentry) could of course have been examined. We assume that while on the return trajectory, the spacecraft was capable of receiving signals from a system of equatorial synchronous satellites which were being used primarily for earth-based navigation of commercial vehicles. Since the antenna patterns of these satellites would cover only the disc of the earth (about 16 degrees) special modification ot the NTC Satellite to provide broad coverage in the other direction would be required; possible satellite reconfigurations of the NTC Satellites are discussed in Paragraph 6. 7. 3. We see from Figure 77 that the baseline provided by two satellites on opposite sides of the earth is much larger than the baseline achievable on the earth; furthermore the earth-rate motion of only one satellite provides substantial swings in geometry during the long flight time. At the initial point (190, 000 miles) for example the earth (and tracking stations on it) subtends only 2. 1 degrees, while the synchronous satellite orbit circle subtends 13. 4 degrees. Rather spectacular variations in geometry are achieved as the spacecraft nears and passes through the orbital radius of the synchronous satellites. Lastly, we note that an earth-based tracking station is guaranteed half-day periods of no-tracking each day as the earth rotates, but that the synchronous satellites will have appreciably shorter periods of nonvisibility to a spacecraft at long range.

In the trajectory modelled, the initial (190, 000 mile point) earth centered velocity is only 2250 fps; velocity increases to over 25, 000 fps at the terminal (3650 mile altitude) point. Since the velocity of a synchronous satellite is about 10, 000 fps, we can expect to see very high changes in doppler frequencies during the earth return flight, particularly if we are near the satellite orbit plane. If the navsats use a ranging code for example, proper receiver design will be required to cope with these high doppler values; for the primary navsat function of earth-based navigation, the maximum doppler rates are always restricted to about vehicle speed, since the satellites (equatorial ones at least) are earth stationary.

The reference trajectory remained about 20 degrees south of the earth's equatorial plane for most of the 65 hours. The result was that the earth subtended a smaller angle than the minor axis of the ellipse representing the spacecraft's view of the satellite orbit plane for most of the

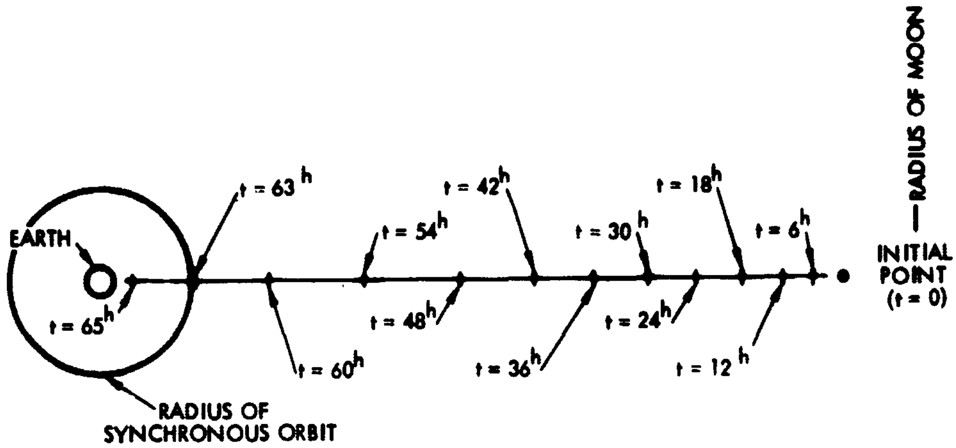


Figure 77. To Scale Sketch of Tracjectory

trajectory; that is, the entire NTC Satellite orbit was continuously in view.

Assuming a data rate of one point (measurement) per hour over selected intervals of the flight, computations on the accuracy with which the spacecraft positions and velocity could be computed were made. The measurements were assumed to be ranges with an unknown (but recoverable) bias of 100 feet (1 σ) and a random error of 50 feet (1 σ); these numbers represent the general capability of NTC Satellite systems now under study.

Initial point spacecraft position/velocity uncertainties were set at 1000, 000 feet in three orthogonal directions (each) and 250 fps in each direction.

In the filtering calculations which computed position and velocity undertainties, the assumptions were made that there were no gravity model errors, solar perturbations, spacecraft venting or maneuvering impulses, nor any other of the many minor disturbances which may complicate trajectory determination. However, uncertainty in satellite location was included in the model. The four error analysis cases following show the effects of using one or two satellites and varying the measurement intervals.

6.6.2.1 Case 1

- Ranging to one satellite only (the same satellite)
- One measurement per hour through entire trajectory
- Satellite position uncertainty (1 σ): 1000 feet (x, y) 250 feet (Z radius)

The 1000 feet horizontal (each direction) and 250 feet radial satellite position uncertainties are much worse than a ground tracking site would have, and can probably be bettered for the satellites particularly if more effort were applied to locating them during a mission that would be warranted for their day-to-day navigation duties. Improvement of the position uncertainty is rapid for this case, as shown in Figure 78. After 12 hours from the initial point, while the spacecraft has come about 20, 000 n mi closer, the rss position uncertainty has been reduced from the initial 28. 5 miles to 10 miles. At 24 hours, this value reaches 7800 feet. Thereafter, the reduction are less rapid and reach a low of about 1000 feet. Velocity uncertainty reaches 1 fps after 12 hours, 0. 1 fps after 24 hours, and stabilizes at about 0. 035 fps. The initial 1000 feet bias uncertainty in the ranging measurement is only slightly improved, to 93 feet, by the filtering procedure.

6. 6. 2. 2 Case 2

- Improved satellite position uncertainty

This case duplicated Case 1 (one satellite on all the time at one hour intervals) except that the uncertainty in satellite position was placed at 100 feet in each axis. The position uncertainties from the filter for this case is shown in Figure 79. Substantially better estimates are made in this case, and the 100 foot range bias is now estimated to 31 feet instead of the 93 feet of Case 1.

Since the NTC Satellites would radiate continually for the benefit of their earth-based users, the ranging data would be available to the spacecraft at any time a satellite was not occluded by the earth. There is no reason then why, hardware/electronic design considerations having been satisfied, ranges could not be taken continually, up to the capacity of the navigation computer to handle new inputs. However, even a small number of measurements could provide navigation information, as Case 3 illustrates.

6. 6. 2. 3 Case 3

- One satellite – 1000 feet (x, y)/250 feet (z) position uncertainty
- Measurement periods as shown in Figure 80.

This case used only short intervals of data (at one per hour) about 1 day apart, until 10 hours before trajectory end when the 1/hr data rate

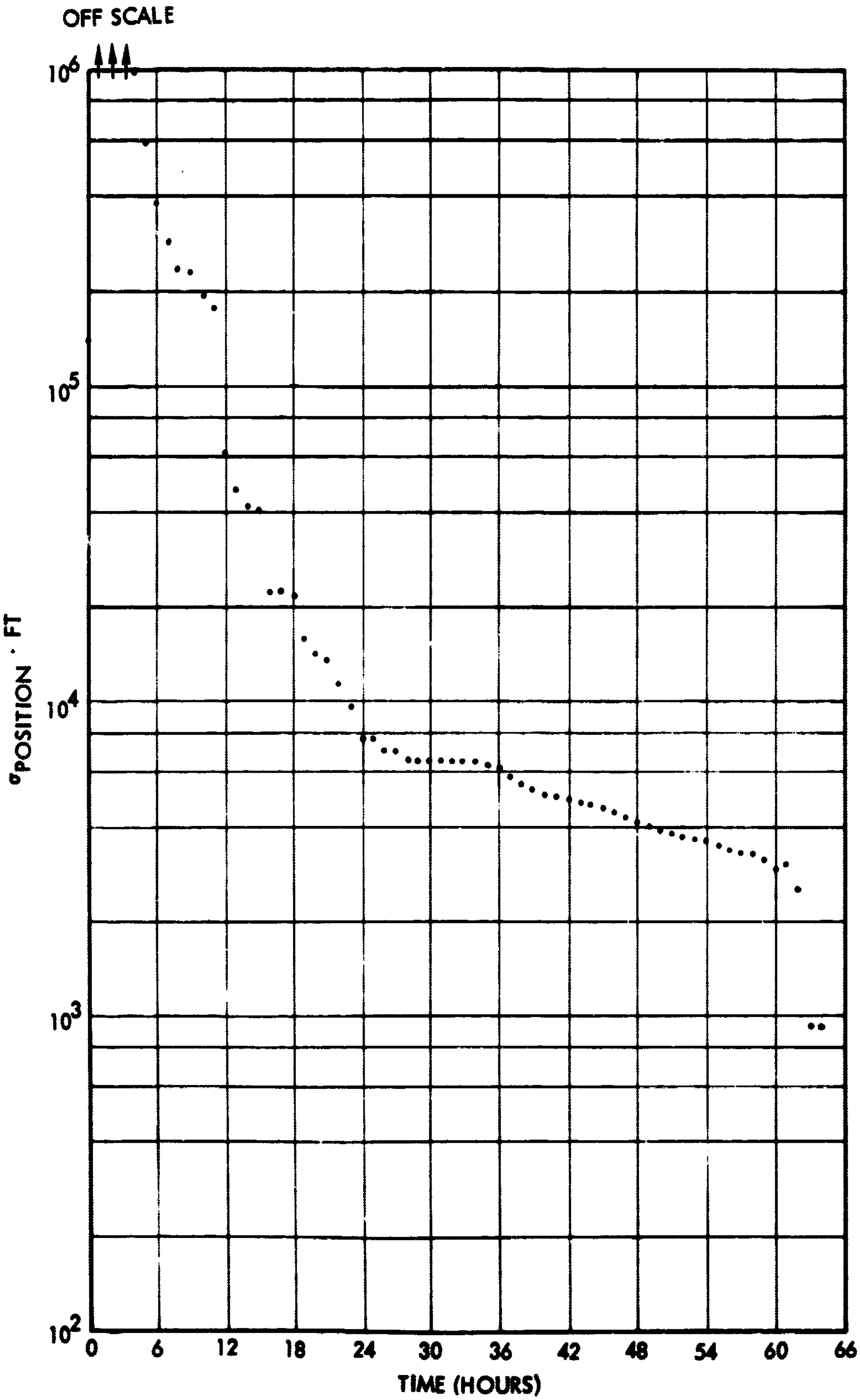


Figure 78. **Case 1 – One Satellite On All the Time**

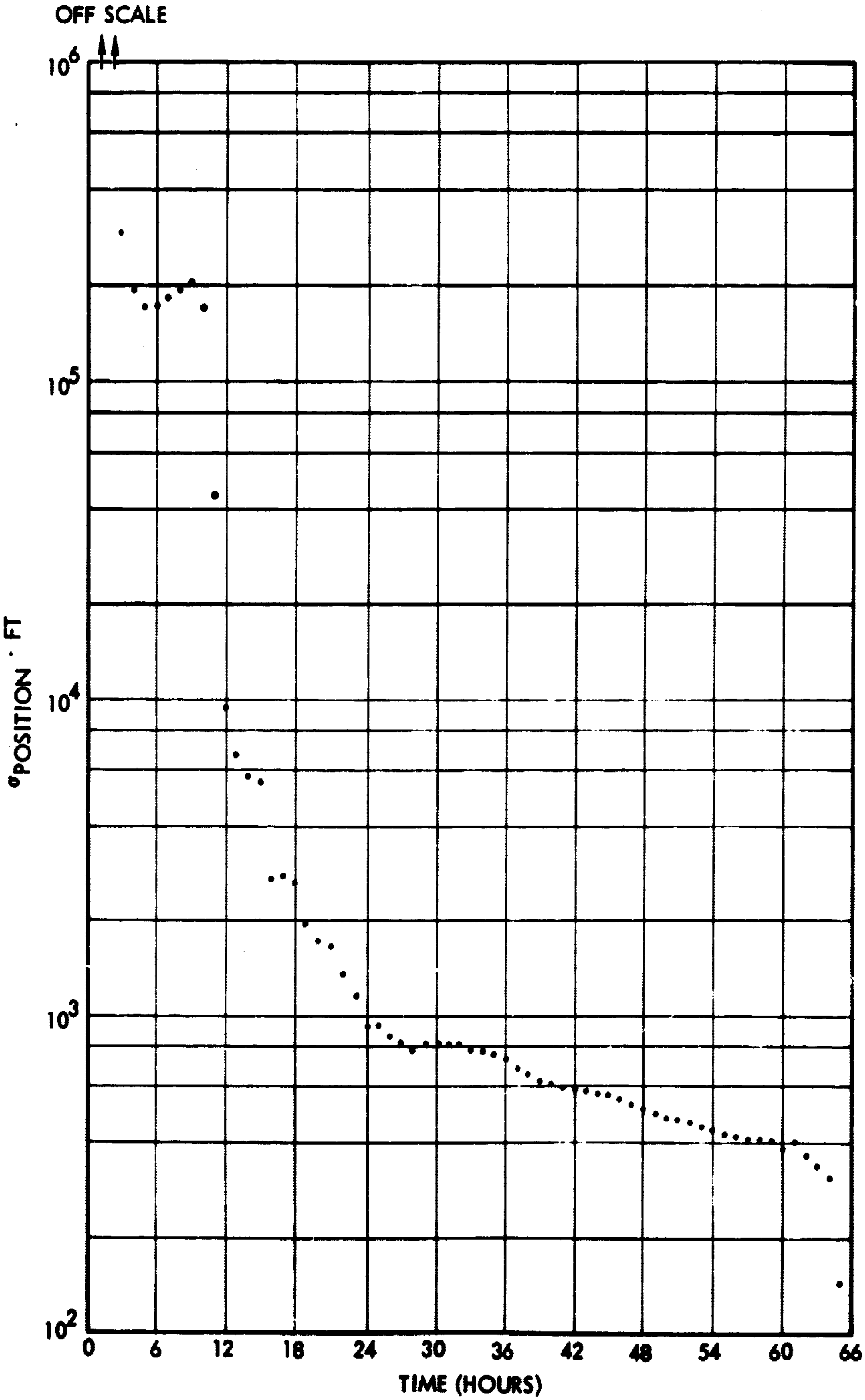


Figure 79. Case 2 — One Satellite, Improved Satellite Position

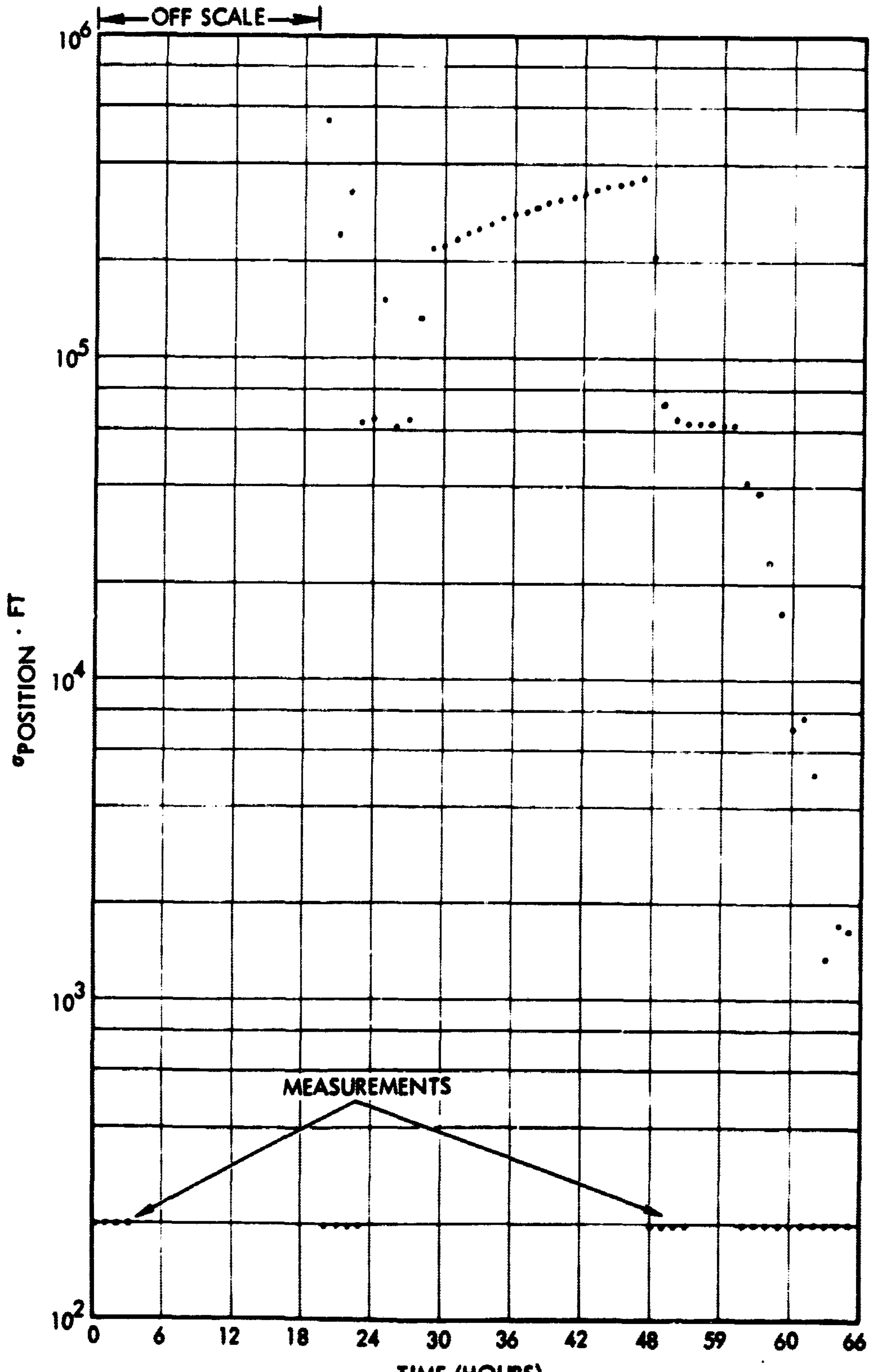


Figure 80. Case 3 — One Satellite, Intermittent Measurements

was resumed. The three four-point measurement intervals (12 observations in all) reduced the position error to 52, 000 feet at the initiation of the final measurement interval, which begins at an altitude of about 60, 000 miles. As the spacecraft closes from this range, significant variations in geometry (to the satellites) occurs and the uncertainties are further reduced. Bias recovery was 98 feet.

6.6.2.4 Case 4

- Two satellites

Case 3 was repeated except that an additional satellite 180 degrees in longitude from the first was hypotehsized (a 1000 feet/250 feet uncertainties in position). Bias recovery improved hardly at all, to 96 feet, but improved early position and velocity uncertainties were seen. Results are shown in Figure 81.

6.6.3 NTCS Modifications for Manual Spacecraft Navigation Support

The hardware modifications needed in a conventional NTC network to support Apollo are primarily confined to modifications of the NTC Satellites to broadcast their normally earth-oriented signals toward the spacecraft.

It is assumed that computer design for the spacecraft would be such as to permit the inclusion of the satellite-to-spacecraft ranges in a normal navigation computation which could also handle, say optical data. The significant system hardware design problems should be in the areas of:

- Radiating power "outward" from the satellite orbit rather than toward the earth only, as in the satellite's normal navigation mode.

- Orienting a spacecraft antenna to receive the signal.

- Acquiring the signal at high velocities and during high accelerations as well if measurements into re-entry are required. Longer acquisition times may result if apriori doppler value is not available.

- Modifying ranging code design to provide larger ambiguity resolution.

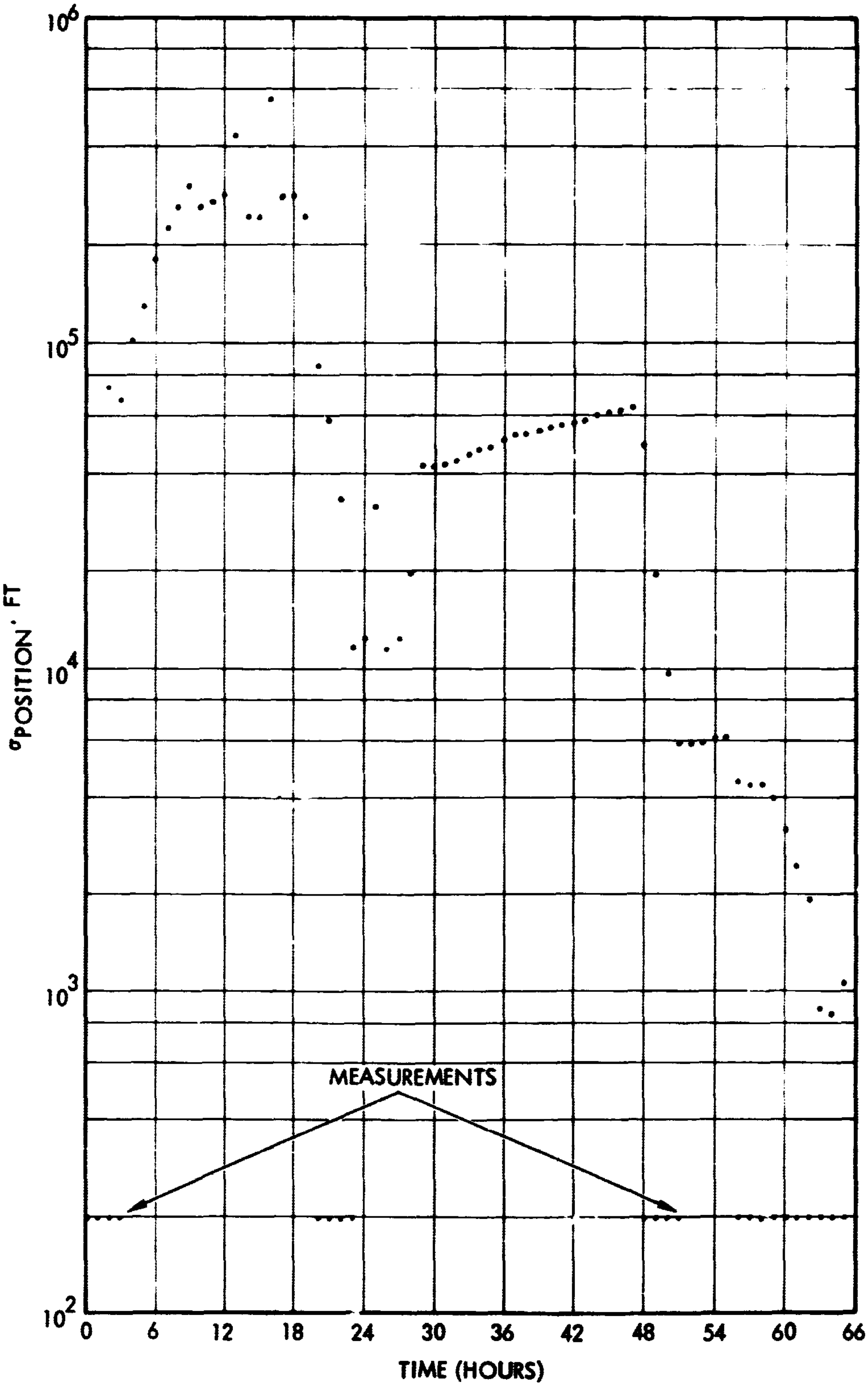


Figure 81. Case 4 — Two Satellites

6.6.3.1 Power Requirements and Satellite Reorientation

We look first at the satellite radiated power. If the satellite antenna is normally covering the earth, the beam width will be about 16 degrees. One solution would be to orient the satellite main antenna toward the returning spacecraft for the duration of the return flight at the expense of providing earth coverage; however, active control of the satellite orientation would be required. The power budget for a 50 watt satellite is used as a baseline; it is for the BINOR code ranging system. All else being equal for the spacecraft applications, the items which may be adjusted easily in design tradeoffs are:

- Satellite transmitter power (50 w) : 47 db
- Satellite antenna gain: 16 db
- Space loss at 22,000 m (1560 MHz): 188.5 db
- User antenna gain (earth user): 0 db

We will leave such items as receiver noise temperature and circuit losses unchanged and hold the user margin fixed. If we could direct the primary satellite transmission toward the spacecraft, the signal could be received at the 190,000 mile range at which our reference trajectory started, at the same margin, if the user (spacecraft) antenna gain were increased by the additional space loss at this range.

Space loss at 190,000 miles = 207 db (at L-band)

Increase in space loss: From 22,000 miles = 18.5 db

To provide this gain increase (18.5 db) would require only a 2-1/2 foot diameter dish antenna on the spacecraft, easily compatible with the sizes of the current Apollo service module antenna now mounted.

The beam - width of this spacecraft antenna would be about 20 degrees, more than the 13.4 degrees subtended by the synchronous orbit at the initial point in the return trajectory. Thus, orienting the antenna initially would involve pointing it at the earth and all satellites would be within the beam. As the spacecraft moved closer, the antenna would require orientation to the transmitting satellites. As the spacecraft moves through the satellite orbital radius, rather extensive spacecraft antenna orientation would be required.

It may be thought undersirable to reorient the satellite's main antenna, as earth service would be curtailed and satellite altitude control would be required. However note that only two satellites are sufficient to yield reasonably good accuracies; if the NTC system has redundancy in numbers of satellites, then two or three committed to an Apollo mission for a few days may not be too many to spare and may be cheaper than adopting some of the following suggestions for special modifications to the satellite which would avoid satellite reorientation. With these modifications, earth users would still be provided for; however, in view of the emphasis placed on space missions (not to mention the cost of the MSFN), the priorities may be in favor of using some of the satellites for Apollo alone during a mission.

6.6.3.2 Other Techniques for Providing Spacecraft Target Coverage

Secondary Broad Beam Antenna Fixed Back-to-Back with the Primary Antenna Power Shared in Time Multiplexing. The satellites could alternate full power between earth and spacecraft antenna. Assuming a zero db satellite antenna with essentially hemispherical coverage, we find at the initial point in the earth-return trajectory, the required spacecraft antenna gain is 34.5 db; this would require a 16 foot dish, probably prohibitive in Apollo-sized spacecraft.

Satellite Radiates a Fraction of Power in Twin Antenna Back-to-Back with Primary Antenna — Earth Users Suffer Somewhat During Lower Periods. Assume 1/4 of total 50 watt power is radiated out of a twin 16 db antenna. The remaining 3/4 (37.5 watts) goes to the primary antenna to support the normal earth navigation functions. The 6 db drop in full power radiated from the satellite requires a 24.5 db spacecraft receiving antenna; a 4-1/4 foot dish would be required, which is probably compatible with Apollo capabilities. On still using the 2-1/2 foot spacecraft antenna, a reduction in maximum range to 95,000 miles would be obtained. This would still provide useful navigation data to the spacecraft although mid-course correction requirements would be increased because of the later initiation of the data.

These are two examples of the kind of tradeoffs that can be made if reorientation of the satellite is not desirable. They show however that a 50 w primary power levels, NTC Satellite could work with the Apollo spacecraft at near-lunar distances using spacecraft antennas not incompatible in size with the current S-band antennas. Some spacecraft antenna orientation would be required but is not a much more severe problem than that of orienting the high gain S-band antennas now in use.

6.6.?.3 Acquisition

The doppler frequencies will at the spacecraft receiver be much higher than the conventional earth user would see. Although the spacecraft is moving at about the same speed initially (near the moon) as a fast supersonic aircraft (2000 to 3000 fps) the satellites are moving inertially at about 10,000 fps. Early in the return flight then the spacecraft will see a range of Doppler frequencies of about ±15 KHz. As the spacecraft nears the earth its velocity increases and the mean Doppler will increase (although the satellite velocity will still cause a ±10 KHz daily variation) to 37 KHz at the 3650 mile reference trajectory terminal point (V = 25,000 fps). If we assume that there is simple no a priori knowledge of the Doppler rate, then the acquisition loop must sweep rather large frequency ranges initially of up to 30 KHz. Reference 23 shows a typical value for two sided acquisition loop bandwidth of about 3300 Hz.

Since acquisition time is

$$= \frac{5\ F}{{B_1}^2}$$

a 1650 Hz (one sided) loop would require about 0.06 second to acquire.

At lower values of carrier power-to-noise in acquisition of, say 34 db, acquisition times of about 1.6 seconds would be required. As spacecraft velocities rise, a larger band must be swept if no a priori assumptions on velocity are made. Thus, at the 3650 n mi altitude (velocity 25,000 fps), the total Doppler uncertainty could reach 55 KHz, requiring acquisition times of several seconds. However, we assume that the crew knows whether they are coming or going, and that a priori uncertainty in the spacecraft velocity is small compared to the satellite velocity relative to a fixed frame, 10,000 fps. Of course, if the crew knows what satellites they wish to receive and where they are, the a priori Doppler uncertainty might be even further reduced. However, if the estimated spacecraft velocity is used as an a priori Doppler offset for acquisition, the acquisition times could exceed one second for low carrier power to noise ratios in sweeping the ±15 KHz band. If these times exceed the normal satellite carrier broadcast duration for conventional users, a special, somewhat extended carrier signal may be required for the Apollo missions. Note that even much longer acquisition times (i. e., smaller loop bandwidths) would be acceptable to the spacecraft, because it will not be required to receive different satellites in rapid (2 second) succession, as the conventional user must do to obtain a fix. The spacecraft will track one satellite for more extended periods of time (probably fractions of hours at least) and rapid acquisition is not a critical problem; however, the satellite must of course transmit the carrier long enough for the spacecraft to acquire.

Once acquisition is complete, the loop bandwidth will be decreased for tracking. The system is then sensitive to Doppler variation due to acceleration, which may cause loss of lock if the loop cannot track fast enough. Reference 23 (pages 173 - 174) shows that with a twosided tracking loop bandwidth of 50 Hz, an acceleration along the range direction of over 6 g's can be accommodated. Since the spacecraft and the satellites are both in free fall (prior to reentry) it is easy to specify the acceleration history; we see that the relative range acceleration cannot exceed 2 g's (for both satellite and spacecraft near to, and on opposite sides of, the earth) and in general will be less. Hence the tracking loop will maintain track in an environment which is actually more passive than a maneuvering aircraft will experience. Tracking loop bandwidth could be reduced in fact to reflect the lower acceleration.

As the vehicle begins to reenter, acceleration rises substantially. With the limitation of crew survivability however, the accelerations are not too high. Figure 82. shows Apollo load factor (g's) and altitude versus time for the reference trajectory on the circumlunar Apollo 8 mission. Peak values are only about 6 g's and some widening of the tracking loop for s afety factor could accommodate these accelerations. We might note in passing that even if the acquisition were performed using the two-sided loop bandwidth of 50 Hz, the entire 30 KHz Doppler uncertainty region could be swept in 240 seconds, not prohibitive for spacecraft operations if the satellite carrier transmission times were made this long.

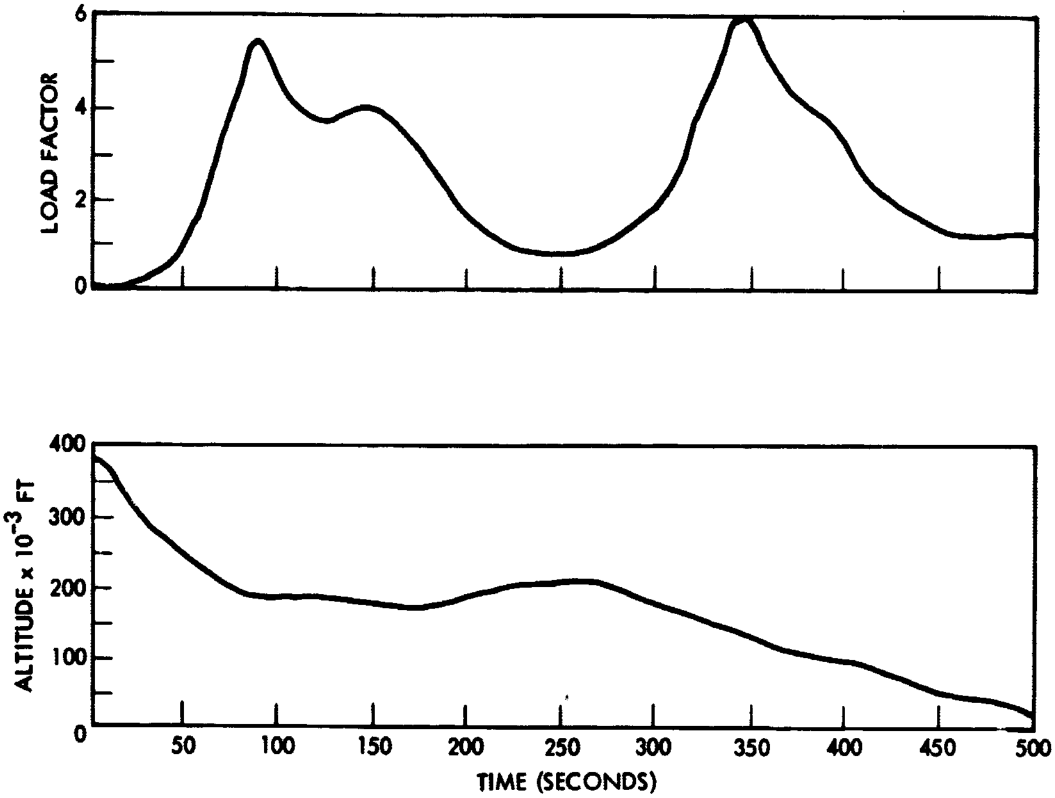


Figure 82 Apollo Circumlunar Return: Reentry Load Factor and Altitude Versus Time

However, use of the NTC System into reentry presents problems in addition to the acceleration (doppler rate) problem. One of these is the antenna problem; 2 to 4 foot dishes would be jettisioned with the service module and the spacecraft would be required to use a flush or stub antenna. Secondly, as Figure 83 shows, blackout is expected to be a problem starting at relatively high altitudes. These problems probably preclude use of the navsat system as an active reentry navigation system although it would be useful if, for example, the spacecraft missed the landing area by a wide margin (e. g., aborted mission) and wished to report its position to recovery forces.

6.6.3.4 Ambiguity

One possible problem area not studied closely is that of providing larger ambiguity levels than the 2000 miles used in Reference 23, which requires a lowest-code component of 81 Hz, while still using code acquisition loop bandwidths of reasonable size. If full lunar distance is required,

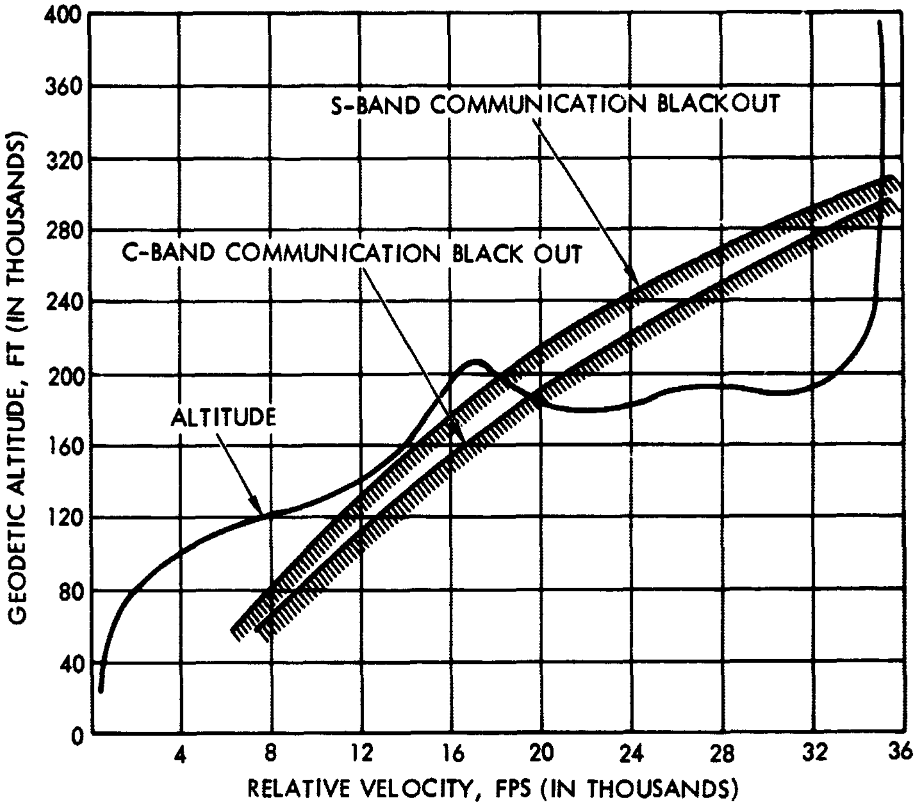


Figure 83. Communications Blackout

1 Hz values for lowest component would be required and it would be necessary to get the tracking bandwidth lower than this with consequent increases in code acquisition time. Some a priori spacecraft positions would be required if a shater code were to be used. However, proper design of a long code with unambiguous ranges larger than the current design of Reference 23 must be examined. The extended code length will increase code correlation times, if the highest code frequency (around 300 KHz) remains fixed; these times could approach the acquisition times of 1 to 2 seconds and modifications of the satellite transmitting pattern might be required.

6.6.4 Concluding Remarks

Synchronous navigation satellites with precision ranging links could provide the same advantages for spececraft that are promised to more conventional users: an onboard receiving and computing system with good coverage and high accuracy capable of rapid position computation.

The large synchronous orbit radius provides a better baseline and better variation in geometry for a lunar mission than is obtainable with earth-based trackers. Also, freedom from atmospheric effects may be enjoyed. The satellite location will not be as exactly known as are the locations of a fixed tracking site; however, the requirements on NTC satellite location knowledge for precision terrestial navigation should also be satisfactory space navigation.

The NTC Satellite configuration modification will be required to radiate the navigation (ranging) signals toward a spacecraft returning from the moon. No significant modification to the satellites would be required for low earth orbiting spacecraft (e.g. MOL). The satellite modification, primarily in the area oa antenna configuration, is a problem somewhat alleviated by the large increase in receiving antenna gain (over the low gain terrestial user antenna) with the antennas even now carried on the Apollo vehicle.

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