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TEXAS INSTRUMENTS
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FINAL REPORT
FOR
A STUDY OF SATELLITE
EMERGENCY LOCATOR SYSTEMS

Prepared for
NASA-GODDARD
SPACE FLIGHT CENTER

Equipment Group
U1-879110-F
Contract NAS 5-23676

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FOREWORD

The work described in this report was performed by the Space Systems Department, Equipment Group, Texas Instruments Incorporated, Dallas, Texas, under NASA Contract NAS 5-23676 for Goddard Space Flight Center. The reporting period is from August 1976 to May 1977. The technical director of this study activity was J. Leland Langston. He was assisted by John F. Du Bose, Jr., James L. Coates, and Charles E. Orand. Acknowledgment and appreciation are also due James Brown, Daniel Brandel, Earle Painter, and Bernard Trudell, GSFC, for their constructive criticism and suggestions throughout the course of the study.

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A STUDY OF SATELLITE EMERGENCY LOCATOR SYSTEMS

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SECTION 1
INTRODUCTION AND SUMMARY

1.1 INTRODUCTION

This report is the final report on a study of satellite emergency locator systems conducted by Texas Instruments for NASA–Goddard Space Flight Center. The objective of the study was to determine the feasibility and hardware requirements for satellite systems capable of identifying and locating the position of Emergency Locator Transmitters (ELTs) and Emergency Position Indicating Radio Beacons (EPIRBs). Both geosynchronous and near-polar-orbiting satellites were considered. One of the most important aspects of the study was to minimize the cost of the ELT/EPIRB hardware.

The study was done in three distinct areas. The first and perhaps the most extensive was systems analysis, which is presented in Section 2. It considers the requirements of the overall search and rescue problem and the application of satellite technology to this problem. Basic system parameters are derived from a complete system performance analysis.

The second area of study was the ELT/EPIRB hardware design. The results of this effort are presented in Section 3. The study was directed toward achieving a low-cost ELT/EPIRB design capable of meeting the requirements of a satellite emergency locator system. Critical design areas are identified and various potential solutions are discussed.

The final major area of study is discussed in Section 4. This section covers the spacecraft equipment requirements for both low-orbiter satellites and geostationary satellites. It considers block-diagram designs of various methods of implementing the receiver-processor. Key design parameters are identified. Alternate methods of implementing certain functions are discussed where applicable.

Before proceeding with the details of the study, the following subsections summarize the objectives, assumptions, and conclusions of the study.

1.2 ASSUMPTIONS AND CONCLUSION

To place the results of the study in the proper perspective, this subsection presents the assumed axioms and the major conclusions reached in the study.

The key word, indeed the highest priority axiom governing many tradeoffs in this study, is *rescue*. The primary objective of search and rescue is to save lives. Studies have shown that the probability of human survival decreases rapidly with time.* In fact, the probability of survival is less than 50 percent if search and rescue operations exceed 4 hours. Therefore, every effort was made to incorporate technology that reduces response time to less than 4 hours in the primary coverage area, i.e., the contiguous United States (including coastal waters), Alaska, and southern Canada.

An important corollary to the preceding axiom is consideration of the harsh environment in which the system must operate. The ELTs or EPIRBs may be required to transmit in suboptimum situations, e.g., damaged antennas, obstructed viewing angles, reduced power, or extreme temperature ranges. Techniques that are completely adequate for well-defined communications links do not provide sufficient margins for what may be typical operating conditions for an emergency transmitter. Hence, this study has endeavored to select technologies capable of providing margins that might be considered unnecessary in a normal communication system, but that are essential to the success of a search and rescue system.

Key parameters influencing success are system dynamic range, mutual interference, probability of receiving a given message, and probability of obtaining a position "fix." The design goal is to achieve 99 percent probability of detection and location within 4 hours while having as many as 400 transmitters in the field of view (FOV). Location accuracy is to be within 10 km under these conditions. The probability of only detection will be greater than 99.9 percent.

Another high-priority objective of the study was to select technology that would minimize the cost of user equipment. The goal was to provide a system in which ELTs and EPIRBs could be purchased for under $400. The results of the study indicate that the retail price range of ELTs will be $300 to $500. (These figures are based on large production quantities and do not include any allowance for amoritization of development cost.) These cost estimates are for units capable of meeting all the objectives discussed in the preceding paragraphs when used with suitable satellites.

A final major objective of the study was to provide adequate growth potential. It is anticipated that geostationary satellites may be used in future systems. Therefore, technology considerations and tradeoffs were made to ensure that ELTs and EPIRBs designed for a low orbiter satellite would also work with a geostationary satellite. In addition, consideration was given to the orderly expansion of the low orbiter system to provide capacity for additional ELTs/EPIRBs.

All objectives of the study were accomplished. Furthermore, the feasibility of a satellite-aided search and rescue system was confirmed.

*Department of Transportation, *Program for Search and Rescue Electronic Alerting and Locating System,* Final Report DOT-TSC-DST-73-42.

1.3 STUDY REQUIREMENTS

The statement of work for this study listed the following general requirements:

Determination of system performance requirements

Synthesis of a position-location/identification receiver-processor design

Synthesis of ELT/EPIRB devices to be used with the receiver-processor

Overall system analyses, including geographical coverage, probability of success of transmitter identification and location, transmitter to satellite link analysis and other pertinent performance characteristics.

The initial assumptions used included a single satellite in near-polar circular orbit at an altitude of 1,000 km. After contract award, GSFC requested Texas Instruments to assume that the satellite equipment would be used on the TIROS-N satellite system and to use the orbital parameters of this satellite. In addition, the study was to address the problem of using geostationary satellites; the study was to assume that any ELT/EPIRB design should be compatible with such a system.

1.4 MISSION CHARACTERISTICS

Search and rescue (SAR) encompasses a wide area of activity from the local scene usually under the jurisdiction of the local fire and/or police department to the international scene, which may be under no well-defined jurisdiction. This study is concerned primarily with two major categoies: aircraft accidents and maritime accidents. All domestic aircraft are currently required to carry ELTs at all times; inspected marine vessels are required to carry EPIRBs. These devices are activated automatically in an emergency. Unfortunately, there is no well-defined system for monitoring the emergency frequencies. However, when an accident is known to have occurred, the following plan exists for SAR activity:

Inland SAR Region—This region is the contiguous United States and the Commander, Scott AFB, is the designated SAR coordinator.

Maritime SAR Region—This area includes the Atlantic, Pacific, and Gulf coastal region. The U.S. Coast Guard has SAR coordination responsibility.

Overseas SAR Region—The U.S. Military commanders are responsible for SAR coordination in their area. Alaska is an Overseas Region.

The actual SAR operations are conducted by local Civil Air Patrol, Cost Guard, or other services coordinated with the appropriate agency.

The problem has already been stated: no well-defined means of identifying and locating the SAR scene currently exist. Many emergencies go undetected for hours or even days. Once an emergency is known to exist, additional hours and days may elapse before concluding the SAR activities, often unsuccessfully. It is the purpose of this study to evaluate current satellite technology for use in solving the detection and locating problem.

1.5 SYSTEM DESCRIPTION

The detailed analyses completed in this study have shown that the TIROS-N satellite system can be used for a satellite-aided SAR program. The position determining accuracy would be comparable to that being achieved with the random access measurement system (RAMS) currently being used for data collection on Nimbus 6. In addition, the ELTs and EPIRBs described in this report will be capable of operating with geostationary satellites using either interferometer or hyperbolic ranging positioning concepts. The ELTs and EPIRBs can be designed to provide user category identification, craft identification, and user-definable emergency states. The estimated battery operating time for these devices will nominally be 100 hours, including a 121.5-MHz beacon.

A system of four TIROS-N type satellites will provide a mean waiting time of 66 minutes (180 minutes 90th percentile, 240 minutes maximum) for the United States, Alaska, and Canada. With ground stations located in California, St. Louis, and Alaska, these times represent total time from the initial ELT/EPIRB activation to identification and location of emergencies occurring in the contiguous United States, southern Canada, and Alaska. The probability of success on the first attempt is 99 percent or better.

The additional satellite hardware needed for each TIROS-N satellite includes an antenna/receiver, an IF/processor, a power supply, and spacecraft interface hardware. The 406-MHz satellite hardware is estimated to weight 50 to 55 pounds, including the antenna, and to require approximately 50 watts of power. (Exact power and weight requirements will depend on the number of receiver channels made available and the technology selected for implementing the search unit.) The only interface required with the main spacecraft is the primary power and the telemetry system. However, it is desirable to also interface with the spacecraft command system, particularly if meaningful tests are to be conducted during the experimental demonstration phase.

1.5.1 Experimental Demonstration Phase

Once the satellite system is deployed, it will be necessary to evaluate the complete system for functional/operational capability, including a complete test of the ground-based hardware. This phase is termed an experimental demonstration phase and will be used to refine and optimize the identification and positioning methods in the fully operational system. This phase will also be used to refine the ELT/EPIRB design requirements and will demonstrate the system capabilities to the user community. When this phase is completed, the fully operational system will be available for use by the aviation and maritime community and the SAR coordinators.

1.5.2 Fully Operational System

The fully operational system is defined as a system of four TIROS-N satellites (or equivalent) equipped with the appropriate hardware for operating with the 406-MHz ELTs/EPIRBs, three ground stations (one in California, one in St. Louis, and one in Alaska), and a production capability for ELTs and EPIRBs. Once this state is reached, only system maintenance and user cost would be incurred. The study shows that the operational system capabilities can be extended or enhanced by additional ground stations. The system as defined has worldwide coverage capability. However, the penalty paid by not increasing the number of ground stations is increased waiting time outside the primary (real-time) coverage area. A functional description of the operational system for the real-time coverage area follows.

In the event of an aircraft or marine vessel emergency, two modes of ELT or EPIRB activation are possible. If the accident is catastrophic, the transmitter will be activated automatically and the emergency code will default to the general "mayday" alarm. If the emergency is noncatastrophic, the device can be activated manually and the exact nature of the emergency can be delineated through a set of predefined emergency codes. Once activated, the ELT or EPIRB will continue to transmit the coded message at the rate of approximately once each minute. The message will contain the category (aircraft or marine vessel), the craft identification number, and the emergency code. The device will also radiate a 10- to 15-mW CW RF signal at 121.5 MHz. (This signal is to be used by SAR homing equipment to locate the exact scene of the emergency when in the general area of the ELT/EPIRB.)

The signal will continue for a period of approximately 100 hours. However, the signal will be received by a satellite within 4 hours. If the accident is within the primary coverage area, the detection and location will be immediate and the SAR coordination center will know:

The identity of the craft in distress

The location of the accident scene within 10 km

The nature of the emergency.

The SAR coordination center will then notify the appropriate SAR organization in the local area of the emergency. Search and rescue efforts can begin immediately. The identity and position of the emergency will be confirmed within another 4 hours. The exact position of the emergency can be determined via the 121.5-MHz CW homing beacon. (Standard direction-finding equipment can be used by SAR personnel.) The complete cycle from the initial ELT or EPIRB activation to final rescue could be completed in hours.

1.5.3 Alternate System Considerations

The study also considered other means of implementing the satellite-aided SAR program. The study reviewed all potentially applicable satellite positioning technologies. Primary considerations were position location accuracy and ELT/EPIRB cost and complexity. Once the TIROS-N satellite system was defined as the carrier vehicle, the selection of position location technique rapidly converged on the differential doppler method used by RAMS. There are two basic means of using the technique.

The first method uses some onboard signal processing to do the position location. The technique is the one used on the Nimbus 6/RAMS data collection system. A second method was also considered. It requires no onboard signal processing and is referred to as the "bent-pipe" technique. It essentially receives the full ELT/EPIRB spectrum and translates it to another frequency for a downlink. This technique, although technically feasible, has two distinct disadvantages:

It requires approximately 500 kHz of additional spectrum.

It does not provide worldwide coverage capability.

Therefore, very little attention was devoted to this method.

1.5.4 Comparison With RAMS

The main objectives of the study have already been defined. The results of the study define an optimum system based on the criteria set forth in the study definition. However, because of the similarity of the optimum system and the basic RAMS, a comparison between the two is worthwhile.

The major difference between the RAMS system and the system recommended in this study is the choice of modulation. RAMS used phase shift keying (PSK) while the choice for the SOS system is minimum shift keying (MSK). The special characteristics of the latter are that it may be thought of as a type of continuous phase frequency shift keying (FSK) having a narrow channel spectrum yet having the properties of PSK for optimum signal-to-noise characteristics. The transmitters used in the RAMS have a power output of only 600 mW. This low power required the use of PSK to maximize bit error rate (BER) performance. A compensating factor was the relatively small dynamic range requirements of the system. This is very important since PSK produces a wide sideband spectrum having considerable power content.

By comparison, the ELT/EPIRB transmitters have a 5-watt power output, thus enhancing the signal-to-noise ratio. However, the dynamic range requirements are expected to be much greater in the ELT/EPIRB system because of the possibility of widely varying transmitter power. The RAMS transmitter power was well-defined and the antennas were oriented properly, which may not be the case for ELTs or EPIRBs. In this case, PSK would not provide the narrow channel bandwidth required to achieve the mutual interference characteristics desired in the low-orbiter system.

The generation of MSK is relatively simple and is comparable to continuous phase FSK. However, MSK may be demodulated using simple FSK techniques or as a PSK signal using more sophisticated demodulation concepts to take advantage of the improved BER performance of PSK. This technique may be necessary if geostationary satellites are deployed for use in the system.

The baseline system analyzed in this study has a probability of receiving any given ELT/EPIRB message of 90 percent for 400 active devices in the field of view (FOV). The RAMS would have a probability of success of only 60 percent.

One additional hardware change recommended in this study for use with the SOS system is the use of a chirp-Z transform charge coupled device (CCD) in the search unit. RAMS uses a time-compression spectrum analyzer approach. The latter is a viable alternative in the SAR system, but the use of chirp-Z transform (CZT) technique has definite advantages. One important advantage is the speed with which the spectrum can be searched. Another is a major reduction in circuitry and power consumption. The main disadvantage is lack of historical operating data on such devices, since they represent new technology.

The accuracy of the system described in this report will be comparable to that obtained with RAMS. In fact, the RAMS data collection system provides a good baseline for evaluating the minimum performance to the expected from a system of the type described in this study document. This study suggests only one major change to the basic RAMS: the employment of MSK instead of PSK.

1.6 SYSTEM PARAMETER SUMMARY

The system parameters determined during the course of this study are presented in Table 1-1.

TABLE 1-1. SYSTEM PARAMETERS

Item	Parameter
Orbital characteristics (TIROS-N)	
Altitude	834 km
Period	101.5 minutes
Instantaneous geographic coverage area	2.9 × 10^7 km^2
Angular velocity	3.7142 radians/hour
Maximum viewing angle	±27.85-degree latitude
Waiting times (four satellites, real-time)	
Maximum (worst case)	240 minutes
90th percentile	
CONUS	180 minutes
Alaska	90 minutes
Mean	66 minutes
Message characteristics	
Peak power	5 watts ERP
Message length	1 second
Mean period between transmissions	48 seconds
Data rate	128 bits/s
Modulation	MSK
Data encoding	Manchester
Error correction	Triple-error correcting code
Unmodulated carrier preamble	102 ms
Bit sync code	15 bits
Frame sync code	24 bits
User code	2 bits
ID code	48 bits
Emergency code	5 bits
Parity check bits	21 bits
System characteristics	
Signal-to-noise density (worst case)	49.1 dB-Hz
Minimum required E/n_0 for 10^{-3} BER	28.0 dB-Hz
Margin (available for contingencies, suboptimum detection, etc.)	21.1 dB-Hz
Overall probability of success (400 transmitters in FOV)	99 percent
Channel bandwidth	400 Hz
Output (telemetry) data rate	1680 BPS
System error (RMS)	2 to 5 km
Maximum doppler shift	±8.9 kHz
Maximum doppler rate	79 Hz/s
Dynamic range	30 dB

1.7 SATELLITE EQUIPMENT IMPLEMENTATION SUMMARY

The system analysis and satellite equipment design considerations indicate that the full-capability satellite hardware should have the characteristics shown in Table 1-2.

The satellite hardware is therefore expected to be similar to the RAMS hardware. Accordingly, the estimated wieght is 50 to 55 pounds and the estimated power consumption is 50 watts. The exact size and weight will be determined by the implementation technology. (The geostationary satellite equipment will in general be heavier and require more power.)

TABLE 1-2. SATELLITE EQUIPMENT CHARACTERISTICS

Item	Characteristic
Low orbiter	
Antenna beam angle (3 dB)	120 degrees
Antenna gain	2.75 dB
Receiver noise figure	4 dB maximum
Search bandwidth	100 kHz
Search/assignment/time	50 ms, maximum
Receiver dynamic range	30 dB, minimum
Number of channel demodulators	12
Channel bandwidth	400 Hz
Frequency measurement accuracy	±0.5 Hz
Suboptimum data detection allowance	8 dB E/n_0
Output data buffer size	1,024 bits
Output data rate	1,680 BPS
Geostationary, same as low orbiter, except:	
Antenna beam angle (3 dB)	20 degrees
Antenna gain	18 dB
Receiver noise figure	3 dB
Number of channel demodulators	16 to 20
Suboptimum data detection allowance	4 dB

1.8 ELT/EPIRB IMPLEMENTATION SUMMARY

The system requirements for ELT/EPIRB designs were considered in the overall system analysis. Section 3 discusses various design approaches that can be used to implement ELTs and EPIRBs capable of meeting the system requirements. Two approaches to implementing the RF circuitry and two approaches to implementing the logic are presented. However, the only critical design area discussed is the crystal oscillator; the critical parameters are short- and medium-term stability. Another critical design area, the automatic activation mechanism, is not addressed in this study. The ELT/EPIRB characteristics are shown in Table 1-3.

TABLE 1-3. ELT/EPIRB CHARACTERISTICS

Item	Characteristic
406-MHz Transmitter	
*Nominal center frequency (RF output)	406.0285 MHz 406.0715 MHz
*Oscillator stability	
Long term (Gaussian distribution)	±0.003 percent
Medium term (drift rate)	1 Hz/minute maximum at 406 MHz
Short term (fractional frequency deviation)	2×10^{-9} maximum
Power output	5 watts, ERP
Spurious output	50 dB below 5 watts
Antenna gain	3 dB
Polarization	Right-hand circular
VSWR tolerance	Infinite without damage
Modulation	MSK
Data encoding	Manchester
Error correcting code	Triple-error-correcting Bose-Chaudhuri code
Message format	
CW preamble	102 ms
Bit sync code (all ones)	15 bits
Frame sync code (see Subsection 2.6.3)	24 bits
User code	2 bits
ID code	48 bits
Emergency code	5 bits
Parity code	21 bits
Transmission bit rate	128 BPS
Transmission time	1 second
Period between transmissions	48 seconds
121.5-MHz Beacon	
Frequency (RF output)	121.5 MHz ±0.003 percent
Power output	15 mW
Modulation	CW only
Antenna	Whip
Power Source	
Battery	Lithium
Number of cells	6
Battery operating lifetime	100 hours, nominal
Physical Characteristics	
Operating temperature range	−40° to +50°C
Size (including batteries, but not antenna)	5 by 5 by 6 inches
Weight	3 pounds
Other environmental requirements	Same as current devices

*Optional frequency choice can be 406.05 MHz ±0.006 percent with a uniform distribution.

SECTION 2
SYSTEM ANALYSIS

A logical study of the system parameters that determine the feasibility of a satellite emergency locator system are presented in this section.

2.1 CHARACTERISTICS OF AN IDEAL ELT/EPIRB DETECTING AND LOCATING SYSTEM

An obvious prerequisite to conducting a study for determining the feasibility of a satellite ELT/EPIRB detecting and locating system is to determine the requirements for an ideal ELT/EPIRB identifying and positioning system, regardless of the technique used. Then the feasibility of a satellite system can be effectively evaluated in terms of how closely this system approximates the ideal system.

The ideal ELT/EPIRB detecting and locating system can be roughly defined as one that provides all users with the capability of instantaneously alerting SAR personnel of any distress situation and indicating the exact location of the incident, yet costs the user nothing while reducing the cost of SAR operations. Such a system is more specifically characterized by the following features:

1. Instantaneous alert/detection of every distress situation
2. Instantaneous and perfect position determination of each incident
3. Worldwide coverage
4. Automatic alerting
5. Ability to indicate the extent of the emergency and the required aids
6. No false alarms
7. Minimum user investment cost, operating cost, and maintenance cost
8. Minimum system initial cost, operating cost, and maintenance cost
9. Minimum user equipment
10. User equipment survivability
11. Independent of weather/environmental conditions
12. Identification of person and/or craft in distress.

These features were selected as the 12 most desired characteristics for use in an SAR operation after consulting numerous reports published by the U.S. Coast Guard, the U.S. Air Force/Civil Air Patrol, FAA, the U.S. Navy, and NASA.[1, 2, 3, 4] Many of the features are obviously desirable, while others require explanation.

The primary mission of SAR operations is to save lives and/or property. Before SAR activities can be initiated, the following questions must be answered:

Who are you?
Where are you?
What do you need?

Often, since time is extremely important in terms of saving lives (and property), the answer to these questions must be communicated to SAR personnel as soon as possible after the occurrence of a distress situation. Since the person in distress may be incapacitated, the answers must be generated automatically. The answer to the first question could be as general as "I am a person in distress," and would be implied by reception of a distress signal. The ideal system would enable the identity of the person or craft to be established. This could be important in determining the medical history of a person and any special needs (e.g., is insulin required or is the person allergic to penicillin).

The precise initial determination of the position of the person in distress is obviously important. While initial position fixing within a 50-km diameter circle might be completely adequate and make very little difference in total time to reach the scene of the distress at sea or on flat, open land, this accuracy would not be acceptable for a distress scene located in rugged, mountainous terrain where search operations could last several days. Therefore, it is important to precisely locate the scene of the distress incident, and this should be done almost coincidentally with the detection of an alert so that the proper SAR unit could be activated in minimum time. Ideally, this would include any point on the earth's surface.

The answer to the question "What do you need?" could also be very general and implied from determining the identification of the distress signal. For example, if the only information received was an aircraft identification number and the location of the distress site, a worst case emergency would be assumed. Additional information (e.g., number of people aboard and the pilot's name) could be obtained from a flight plan and the equipment needed to perform the search and rescue could be determined from the location of the distress. However, special codes should be available for emergency situations involving manual activation of an ELT or EPIRB. In the case of aircraft, such emergencies would be forced landing, hijacking, loss of NAV/COMM functions, etc. For maritime operations, such emergencies would be no fuel, equipment failure, emergency medical assistance, fire, etc. ELTs carried on potentially hazardous expeditions (e.g., arctic, mountain, or jungle expeditions) should be able to delineate the nature of an emergency.

It is also obvious that the user equipment must survive any environmental stress existing at the site of an emergency as well as any stress caused by the incident that created the emergency. Although this is an important consideration, it is a common factor for almost any system and is not considered in this study.

A final consideration of an ELT/EPIRB detecting and locating system is the false alarm rate. A false alert not only wastes time and money in needless searches, but destroys confidence in the system and is demoralizing to SAR people. Therefore, it is important to ensure that a system will produce few false alarms, ideally zero.

2.2 SATELLITE TECHNOLOGY FOR POSITION LOCATING SYSTEMS

Satellites have often been used as a means of determining position and a wealth of data exists from numerous experiments using varied satellite position locating techniques. Tables 2-1 and 2-2 summarize the major characteristics of the various experimental techniques. Obviously, tradeoffs must be made since none of the applicable systems completely meet ideal SAR requirements. For example, a search through the Position Locating Accuracy column of Table 2-2 indicates that the ATS/PLACE system is definitely the most accurate and would certainly meet the accuracy requirements of any SAR operation. Other advantages include

TABLE 2-1. SUMMARY OF POSITION LOCATION TECHNIQUES

Position Location Technique	Examples of Existing or Proposed System			Number of Necessary S/C for Position Location	Satellite Orbit Altitude Employed	User Package Requires a Receiver?	Collection Interval Per Satellite	Approximate Time Required for Position Location Measurement	Typical Delay Between Measurement and Position Location Calculation
	Name	Satellites Used	Date of Launch						
Radio location via satellite relay	OPLE GRAN	ATS III LES 6	1967	1	Synchronous	Yes	Continuous	3 minutes	Zero
Ranging measurement (dual satellite ranging)	PLACE	ATS and ELF	1969 and 1971	2	Synchronous	Yes	Continuous	1 minute	Zero
Sequential ranging from low-orbiting satellite using store-and-forward capabilities on the satellite	IRLS	Nimbus	Nimbus III, April 1969; Nimbus IV, April 1970	1	Low	Yes	10 hours	4 minutes	Variable[A]
Satellite acting solely as a repeater									
Angle measurement (Interferometer on satellite)	ATS 6 Interferometer			1	Synchronous	No	Continuous	1 minute	Zero
Range rate measurements Doppler with constant frequency source on the package using store-and-forward capabilities on the satellite	RAMS	Nimbus	Nimbus 6, June 1975	1	Low	No	10 hours	5 minutes	Variable[A]
Satellite acting solely as a repeater									Zero

NOTES:

A. This delay is a function of the relative location of the package and the ground station.

* OPLE–Relay OMEGA tones via satellite relay to ground processing facility.
IRLS–Activated upon receipt of interrogation command.

* RAMS–Burst transmission containing ID and sensor data, carrier frequency, time, ID, sensor data is recorded on S/C and dumped to the ground periodically.

TABLE 2-2. SATELLITE APPLICABLE TECHNOLOGY BASE

Satellite Programs	Ground Transmitter Power	Position Location Accuracy	Major Characteristics
OPLE	5 watts	3 to 6 km	56 BPS, 3-minute burst, retransmitted omega
IRLS	6 watts	7 to 10 km	Ranging system, 3-second burst, 1 kbit
EOLE	7 watts	2 km	Range and doppler, 48 BPS, 0.5 second
LANDSAT	5 watts	N/A	5 kBPS (burst)
SMS	5 to 40 watts	N/A	100 BPS
ATS/PLACE	250 watts	80 meters (tone ranging)	Ranging system, 600 to 1200 BPS
NIMBUS-6/RAMS	600 mw	2 to 5 km	100 BPS (burst), 1 second on/60 seconds off, doppler location
ATS	5 watts	2 km	Interferometer

continuous coverage of a major portion of the earth's surface from a geostationary satellite and almost instantaneous identification and location of an ELT. However, this system has two major disadvantages for use as an emergency locating system: (1) the user package (ELT/EPIRB) would require a receiver, and (2) the user transmitter power requirement (250 watts) is not compatible with the cost, size, weight, and operating time requirements for an ELT/EPIRB. The accuracy indicated is based on experiments using the ATS satellite with the power and bit rate indicated and operating in the VHF frequency band. A higher frequency (for a given antenna size) would obviously reduce the power requirements at the expense of beam angle. Data rates could also be lowered to reduce transmitter power while maintaining a given E/n_0 ratio. However, the user equipment would still require a receiver for transponder operation, even though the required transmitter power could possibly be made as low as the 5 to 7 watts required by other methods. The receiver requirement in the user equipment is undesirable for any emergency locating system because of the increased parts count, which increases cost, degrades reliability, and increases power consumption.

The remaining satellite locating techniques have similar accuracies and require similar peak power requirements, except the RAMS transmitter, which only requires 600 mW. If a position accuracy of 10 km or less is presumed adequate, then all systems listed are suitable and other criteria must be used to select a system. Two such parameters are power (both peak and average) and user equipment complexity. If the latter is used as a selection criterion, then the non-requirement of a receiver is important and the consideration of systems is reduced to two. (A technique called unilateral ranging is not listed in the table, but will be discussed later.) These two techniques are the interferometer (angle measurement) technique using a geostationary satellite and the range rate (RAMS) technique using a low-orbiting satellite.

The interferometer technique has the advantage of continuous coverage of a major portion of the earth's surface [e.g., the contiguous United States (CONUS)] from synchronous orbit. The alerting capability is instantaneous and position determination is nearly coincident. Only one satellite is required, provided worldwide coverage is not a major consideration. While experimental data indicates that position determination accuracies of ±2 km are possible at 1600 MHz using a transmitter power of 5 watts, no experiments have been conducted to demonstrate the feasibility of this technique at 400 MHz for field use as a satellite emergency locating system. Also, since position determination is made by precisely measuring the phase difference of received carrier signals, reflections from various surfaces near the transmitter can adversely affect the phase of the received signals and, hence, the accuracy of the position location. However, the advantages definitely make this technique attractive, even if it is a distant future consideration. The design of user equipment (ELT/EPIRB) should be compatible with this technique if possible. The use of geostationary satellites for detecting or locating ELTs is discussed later, independently of measurement technique.

The use of range-rate (doppler) techniques is limited to satellites in low orbit (500 to 10,000 km). The lower orbits provide greater doppler shifts for better accuracy and require less transmitter power. They also have orbital periods that are considerably less than the periods for higher orbits. The disadvantage of the low orbit is the limited geographic coverage at any instant of time. The range-rate technique can be implemented in various ways. One method uses the satellite solely as a repeater (" bent-pipe" technique). An alternate technique uses onboard signal processing in conjunction with store-and-forward capability. Finally, the position location determination can be made using differential doppler in lieu of basic doppler. All of these factors are important considerations in selecting a satellite locator system.

The range-rate technique via low-orbiting satellites has demonstrated positioning accuracies of 5 km or better using the Nimbus satellite (RAMS). The RAMS system has also been used to locate simulated ELTs in various environments. The platform transmitters (simulated ELTs) had a 600-mW RF power output and operated at 401 MHz. The transmitter operating duty cycle was 1/60, thus having an average power drain of only 10 mW. The major disadvantage of this system is the relatively long latency time required for ELT detection.

A tradeoff of the two basic types of satellite systems (low-orbiting versus geostationary) is in order. The criterion will be how well each system approximates the ideal system.

2.3 GEOSTATIONARY VERSUS LOW-POLAR-ORBITING SATELLITES

For comparison, the two orbits will be considered ideal from the standpoint of coverage. The geostationary satellite will be assumed to be over the equator and on a meridian that bisects the contiguous United States. The low-orbiting satellite will be assumed to be in orbit at an altitude of approximately 834 km. The low orbiter will be assumed to use the store-and-forward technique. Single satellites are assumed in both cases.

The first criterion is how close each system comes to providing instantaneous detection of an ELT. The geostationary satellite can "see" any ELT that is within the 1/3 earth-coverage area of the satellite, provided the view of the satellite is not obstructed. The viewing angle (elevation angle) to see the satellite from relative latitudes or longitudes near 70 is very low, and, therefore, detection is highly dependent on terrain. If the satellite is in view, however, detection can be accomplished within 1 minute. A single geostationary satellite will not achieve worldwide coverage, but it can achieve continuous coverage of the continental United States and a major portion of the U.S. maritime regions. The polar-orbiting satellite, on the other hand, can achieve global coverage of each point on the earth at least once every 12 hours. The position accuracy of both satellites (disregarding transponder ranging) is on the order of 5 to 10 km.

The automatic alerting capability, the ability to indicate the extent of the emergency, and the false alarm rate are basically independent of the satellite orbit. This is also true of survivability, identification capability, and weather/environmental conditions.

The use of a geostationary satellite would, in general, require more transmitter power than a low-orbiting satellite. Positioning with a geostationary satellite using the interferometer technique would not increase the complexity of user equipment, but any type of ranging (except unilateral ranging) would increase the complexity of the ELT/EPIRB. The transitory satellite system would require minimum user equipment. Therefore, the low-orbiting satellite system can be assumed to require minimum user investment cost, operating cost, and maintenance cost as well as minimum user equipment. The amount of spacecraft equipment and ground equipment as well as operating maintenance cost of the ground equipment should be about equal for either system.

In summary, a single low-polar-orbiting satellite would provide a system of global coverage and adequate positioning accuracy for ELTs but at the expense of relatively long latency times (up to 12 hours). This system would require minimum user equipment. The geostationary single satellite system would provide almost instantaneous detection and locating of ELTs in major geographic areas, but not worldwide coverage. The user equipment would be slightly more complex and expensive.

2.4 GEOGRAPHIC COVERAGE USING A NEAR-POLAR LOW-ORBITING SATELLITE

Time is of the utmost importance in search and rescue operations. The time to identify and locate an emergency transmission via satellite is directly dependent on the orbital geometry of the satellite. This portion of the study is based on the use of satellites in a near-polar circular orbit at an altitude of 834 km and inclined 99.01 degrees. The following calculations to determine geographic coverage are based on this assumption the use of the mean earth geometry, and nominal physical constants.

2.4.1 Orbital Parameters and Satellite Latency Times

Figure 2-1 is a simple sketch of the earth, an orbiting satellite, and the coverage area of the satellite. The instantaneous coverage area is shown as the shaded area, A_c. This area is the area of a spherical segment on the earth's surface subtended by the solid angle, ω, as measured at the center of the earth. Therefore, the first step in determining geographic coverage is to determine this area.

Figure 2-2 depicts a cross section of the earth-satellite geometry. The earth's radius is R_o. The satellite's altitude is h. The slant range from the satellite to the horizon is d and the surface distance from the satellite's nadir to the horizon is s. The plane angle at the earth's center from nadir to the horizon is θ and the plane angle at the satellite from nadir to the horizon is α. With these parameter definitions and the following constants, the satellite's instantaneous coverage can be determined:

$$R_o = 6.37 \times 10^3 \text{ km (mean radius of earth)}$$
$$h = 8.342 \times 10^2 \text{ km (satellite altitude)}$$

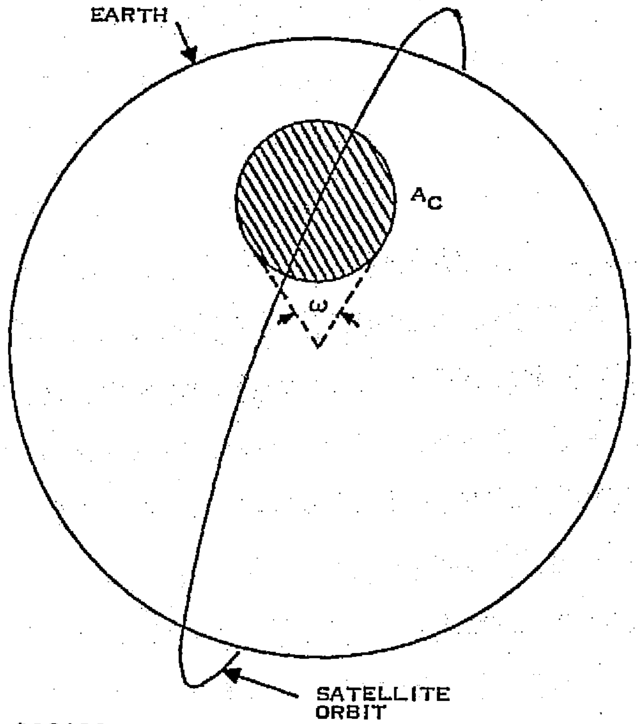


Figure 2-1. Satellite Coverage Area

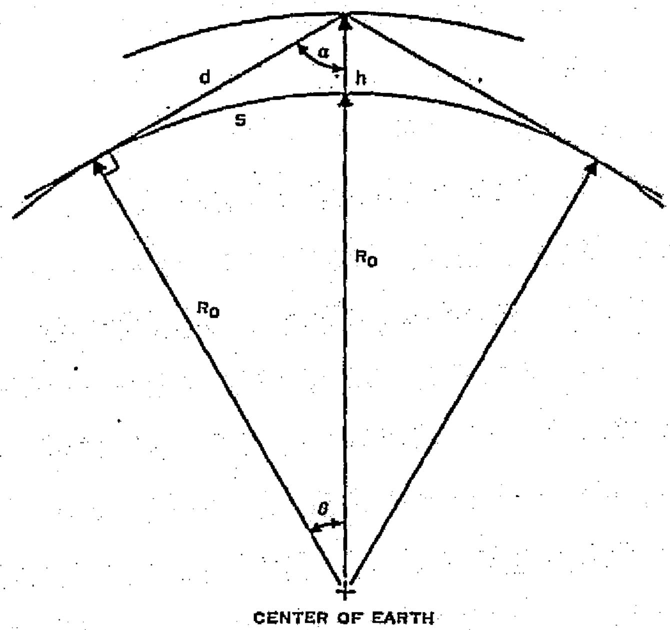


Figure 2-2. Earth-Satellite Coverage

Simple trigonometry yields:

$$\theta = \text{arc cos} \frac{R_o}{R_o + h}$$

$$= 27.85^\circ$$

$$d = (R_o + h) \sin \theta$$

$$= 3365 \text{ km}$$

$$s = \frac{\theta}{360} (2\pi R_o)$$

$$= 3096.3 \text{ km}$$

The solid angle, ω, at the center of the earth that defines this coverage is:

$$\omega = 4\pi \sin^2 \left(\frac{\theta}{2} \right)$$

$$= 0.7278 \text{ steradians}$$

The area of geographic coverage is simply the area of the spherical segment given by:

$$A_c = \omega R_o^2$$

$$= 2.9530 \times 10^7 \text{ km}^2 \quad .$$

The near-polar circular orbit provides complete coverage of all areas north of 71.2 degrees north latitude and all areas south of 71.2 degrees south latitude on each orbital pass. The remaining areas are not covered on each pass. The areas centered at the equator are the least frequently covered, i.e., equatorial regions are worst case. Therefore, the maximum time is determined from the orbital period of the satellite and the rotation rate of the earth.

The period of the satellite is given by:

$$T = 2\pi \sqrt{\frac{(R_o + h)^3}{g_o R_o^2}}$$

where g_o is the gravitational acceleration at the earth's surface and is 0.0098062 km/s². The orbital period in minutes is:

$$T = \frac{2\pi}{60} \sqrt{9.396750 \times 10^5}$$

$$= 101.5 \text{ min}$$

The angular velocity of the satellite is

$$\omega_s = 212.81°/\text{hr} = 3.7142 \text{ rad/hr}$$

The longitudinal displacement of nadir from one orbital pass to the next is:

$$\Delta\phi = \frac{101.5 \text{ min}}{60 \text{ min/hr}} \times 15°/\text{hr}$$

$$= 25.38°$$

where 15 degrees/hour is the earth's rate of rotation.

The next step is to determine the worst-case latency time for detection of an ELT. This would occur when an ELT was activated at the equator 5 degrees east of the orbital track just as the satellite left the field of view. This is calculated from the rotation rate of the earth and the time required for the satellite to rotate from a latitude of 27.85 degrees (limit of satellite FOV) back to the equator. During this time, the ELT would be rotated to a position more than 27.85 degrees of longitude from the orbital track, which is outside the satellite's FOV. The orbit time for this delay is approximately 93.65 minutes.

The appropriate calculations are:

$$\Delta\phi' = 27.85^\circ - \frac{(360^\circ - 27.85^\circ)\ 25.38^\circ}{360^\circ}$$

$$= 4.4^\circ$$

$$\Delta t = \frac{360^\circ - 27.85^\circ}{360^\circ} \times 101.5 \text{ min}$$

$$= 93.65 \text{ min}$$

The next opportunity for the ELT to be in view of the satellite occurs when the earth's rotation moves the ELT to intercept the orbital track on the opposite side of the earth, or after the earth rotates through approximately 152 degrees. The satellite must rotate through an integer number of orbital passes plus one-half pass.

If the ELT is assumed to initially be displaced by 27.85 degrees with respect to the orbital track, after 4½ orbits the ELT is moved:

$$\phi = 4.5 \times 25.38^\circ$$

$$= 114.21^\circ$$

The ELT's position is then 114.21 degrees + 27.85 degrees (or 142 degrees) with respect to the original orbital track. It is yet 38 degrees removed from intercepting the satellite track on the opposite side of the earth and, hence, not yet in the satellite's FOV. The next opportunity occurs after approximately 5½ orbital periods or:

$$\phi = 5.5 \times 25.38^\circ$$

$$= 139.59^\circ$$

The ELT position is now 139.6 + 27.85 (or 167.4 degrees) with respect to the reference track. It is now only 12.6 degrees from the orbital track on the opposite side of the earth and hence is well within view of the satellite. The total elapsed time shall have been:

$$t = 5.5 \times 101.5 + 93.6$$
$$= 651.8 \text{ min or } 10.86 \text{ hr}$$

The satellite can see the ELT before reaching the equator, leading this crossing by almost 8 minutes. Therefore, the maximum latency time for ELT detection is approximately 10 hours 45 minutes.

A similar worst case condition is realized when an ELT is activated 12 degrees east of the satellite track just as the satellite leaves the FOV. Six and one-half orbital periods are required to move the ELT under the antipodal track. However, approximately 15 minutes (length of time the satellite is in the FOV) is deducted from the 660 minutes required for 6½ orbits giving a total of 10 hours 45 minutes. Hence, the worst case latency time for a single satellite exists when an ELT is activated between 5 and 12 degrees east of the satellite's orbital track just as the satellite leaves the FOV. A conservative worst case latency time for a single satellite is, therefore, the time for 6½ orbital periods, which corresponds to 660 minutes or 11 hours.

The possibility of an emergency transmission remaining undetected for 11 hours does not appear consistent with the requirements of search and rescue operations. A viable alternative is the use of additional satellites. The first consideration is the use of two satellites whose orbital planes are at right angles to each other. Again the equatorial regions are worst case. During a zenith crossing at the equator, each satellite has an FOV of ±28 degrees of longitude. Therefore, two satellites will cover 224 degrees of the earth's surface during one orbital pass of both satellites. This leaves 34 degrees uncovered in each quadrant at the equator. Since the earth rotates 15 degrees/hr, more than 2 hours ar required to move an ELT located just out of view of one satellite into view of the second satellite. However, since the orbit period is 101.5 minutes, two orbital periods would be required, or a time of up to approximately 3½ hours. This is in addition to the 94-minute worst case time lost due to ELT activation just after passage of a satellite (as described for a single satellite) Therefore, the worst case latency time for a two-satellite system is approximately 300 minutes or 5 hours.

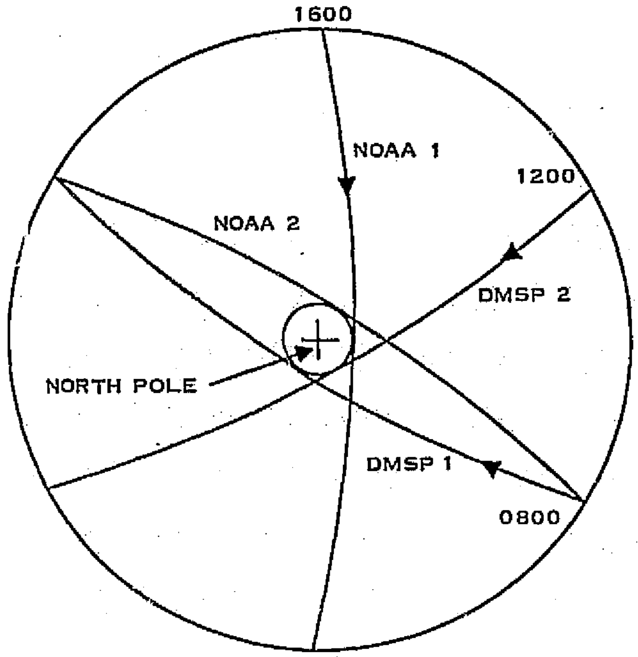
A three-satellite system can be similarly evaluated. Three satellites whose orbital planes are displaced 60 degrees and time phased such that the polar crossings are 60 degrees apart would eliminate the blind areas at the equator, i.e., any point on the equator would always be in view of at least one orbital track. Hence, the latency time is reduced to the satellite rotation period. The latency time for such a system is reduced to approximately 2 hours. However, this is a very marginal worst case time and a more reliable worst case time would be 210 minutes.

Additional satellites will reduce the time to detect an ELT, but the amount of time reduction becomes less and less for each additional satellite. The latency times given have been those times required for satellite detection of an ELT and have not included any additional time required for relay of this data to a central processing location or any processing time. If the data is to be stored until a satellite passes over a data dump point and then relayed to a central processing point, this additional delay can be as much as one orbital period or 101 minutes. Therefore, using additional satellites to reduce time for detection by satellite yet using the data store/dump technique is inconsistent. The use of more than three satellites may be justifiable for other reasons, however. They would also be justifiable if a real-time data link to the central processing facility were used; e.g., relay via a geostationary satellite or using detect and real-time broadcast.

The use of the proposed TIROS-N/NOAA spacecraft and DSMP spacecraft for the SOS program is feasible but requires longer waiting times because of suboptimal orbital characteristics. The latency time for a single satellite is the same as that calculated previously. For two satellites (NOAA1 and NOAA2), the worst case differs since the right-ascensions differ by only 60 degrees instead of 90 degrees. Therefore, there is a 120-degree angle between orbital tracks (see Figure 2-3). If the right-ascensions are time phased such that the eastward orbit lags the other in time by 60 degrees, then the worst case latency time is:

$$4.3 \text{ orbits} \times 101.5 \text{ min/orbit} = 436 \text{ min}$$

If the orbits are not time phased, then up to five orbital passes may be required (500 minutes).



NOTE: DMSP = DEFENSE METEOROLOGICAL SATELLITE PROGRAM

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Figure 2-3. Four Meteorological Satellite Configurations

The four-satellite configuration proposed has the same worst case geometry as the three-satellite system previously analyzed. Therefore, the worst case latency time for suitably time-phased orbits is 210 minutes while the worst case time for randomly time-phased orbits is 240 minutes. A summary of worst case (worldwide) latency times is given in Table 2-3.

The total alert time can be determined by adding the time required for the satellites to pass over a data dump point. If ground stations are located at strategic points (Anchorage, St. Louis, and San Francisco), then the alert time is approximately the same as the latency time for points in Alaska, the contiguous United States, and southern Canada. The real-time coverage area for such a system is shown in Figure 2-4 and the corresponding alert times[5] are summarized in Table 2-4. (Alert times are based on a minimum 4-minute visibility time above a 10-degree elevation angle.)

2.4.2 Satellite Viewing Angles

The previous consideration of geographic coverage was limited to earth coverage as viewed from a transistory satellite. An alternate means of considering satellite coverage is to study the viewing angles required to observe a satellite in polar orbit from various earth locations. This allows the effects of local terrain on satellite coverage to be considered. It also allows one to determine the time that a satellite can be viewed above a specific elevation angle.

The first step is to define the frame of reference. For simplicity, the primary X-, Y-, and Z-axis system has its origin at the center of the earth; the Z-axis is directed through the North Pole; and the X-axis passes through the intersection of the equator and the meridian of longitude on which the observation point (or ELT) is located. The Y-axis is perpendicular to the X-Z plane. The position of an ELT in this frame of reference is then defined by the ELT's (or observer's) latitude. This point (ELT position) on the surface of the earth then becomes the secondary origin, or the translated origin. The X' and Z' axes of the secondary origin are then rotated about the Y' axis so that the Z' axis is normal to the surface of the earth and the X' axis is tangential to the earth and directed toward the south. The polar angles of a vector pointing toward a satellite in the translated and rotated coordinate system are the supplements and complements of the respective azimuth and elevation angles of the satellite as viewed from the ELT position.

By using this technique of translation and rotation, the satellite's earth-central latitude and longitude position can be converted to satellite azimuth and elevation angles referenced to the

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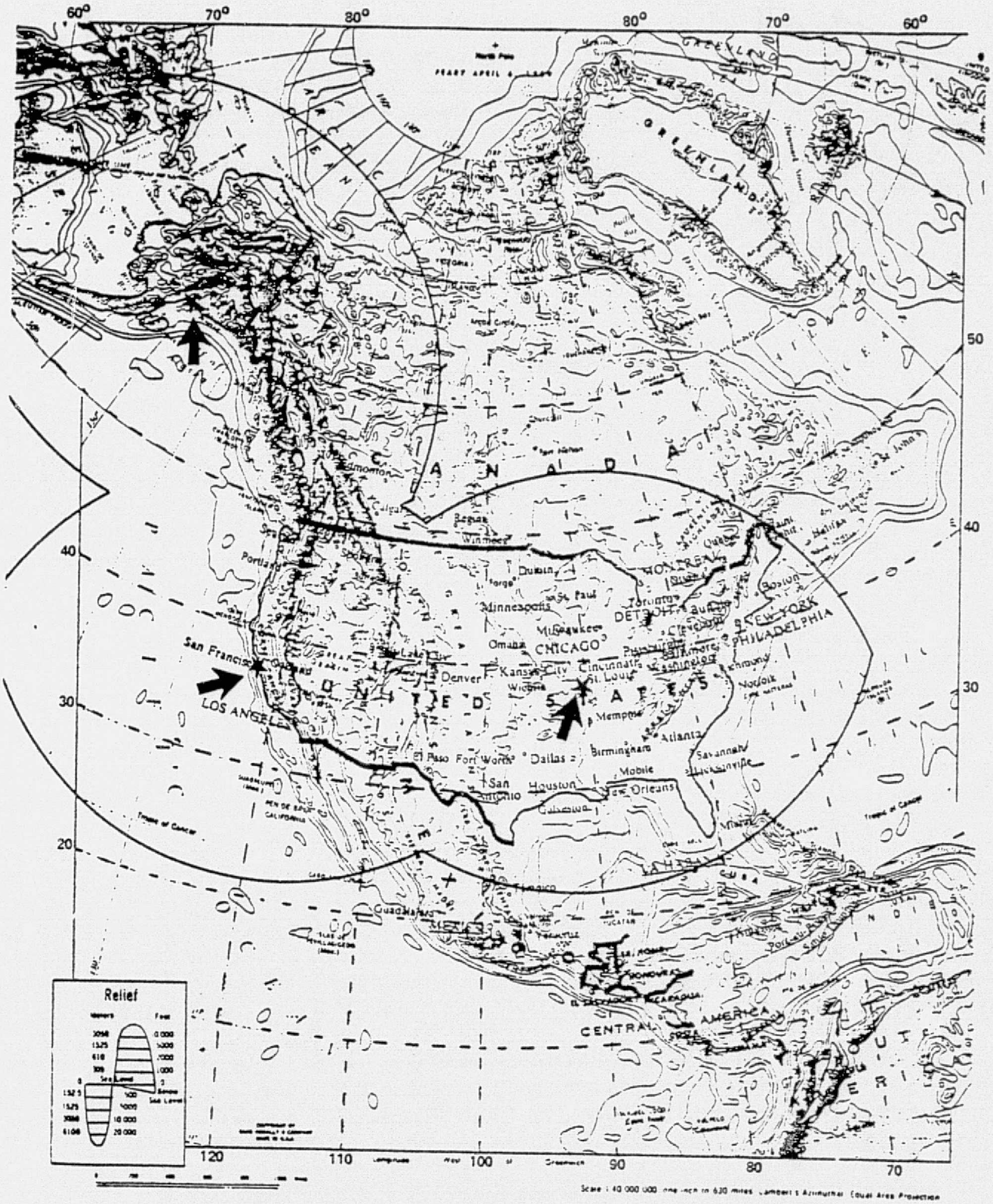

Figure 2-4. Real-Time Coverage Area for TIROS-N Satellite System

TABLE 2-3. SUMMARY OF WORST CASE SATELLITE LATENCY TIME FOR TIROS-N/DMSP SYSTEM

		Number of Satellites		
		1	2	4
Worst Case Satellite Latency Time (minutes)	Satellites optimally time phased	NA	440	210
	Satellites randomly time phased	660	500	240

TABLE 2-4. SEARCH AND RESCUE WAITING TIME SUMMARY (TIME IN MINUTES)

Number of Satellites	Global Coverage (Store and Forward) 1	2	4	Real Time 1	2	3
CONUS	378*	216	142	280	136	66
	880	660	390	630	390	180
	1000	790	480	750	510	240
Alaska	205	117	78	192	98	42
	580	370	290	540	350	90
	730	710	630	730	480	150
Atlantic	514	339	244			
	1000	780	550			
	1140	910	620			
Pacific	415	269	218			
	1000	810	700			
	1310	990	800			

*Key: Mean–378
90th percentile–880
Maximum–1000

ELT's (or observer's) location. If the azimuth and elevation angles are plotted using polar coordinates with elevation plotted as ρ, a curve is obtained for each longitude of a satellite equator crossing. A family of such curves is thus obtained for a given observation latitude. The necessary vector operations are as follows:

$$\vec{R}_o = R_o \; \underline{/A_o, B_o}$$

where

$\vec{R}_o$ = the vector locating the ELT with respect to the center of the earth

R_o = the radius of the earth, 6.37×10^3 km

A_o = the angle R_o makes with the Z-axis in the primary reference system of polar coordinates

B_o = $90° - L_o$, the angle R_o makes with respect to the X-Z plane, 0 degree in this case

L_o = the latitude of the ELT.

The vector locating the satellite is:

$$\vec{R} = R\,\underline{/P,Q}$$

where

$\vec{R}$ = the vector locating the satellite with respect to the center of the earth

R = the radius of satellite's orbit, 7.204×10^3 km

P = the angle R makes with respect to the Z-axis

Q = the angle R makes with respect to the X-Z plane

The satellite may be located with respect to the ELT simply by:

$$\vec{R}' = \vec{R} - \vec{R}_o$$
$$= R'\underline{/A', B'}$$

where $\vec{R}'$ is the vector locating the satellite with respect to the ELT position, R' is the slant range to the satellite, A' is the elevation angle, and B' is the azimuth angle.

The vectors $\vec{R}$ and $\vec{R}_o$ can be represented using the $\vec{i}$, $\vec{j}$, and $\vec{k}$ unit vector notation:

$$\vec{R} = R_x\,\vec{i} + R_y\,\vec{j} + R_z\,\vec{k}$$

$$\vec{R}_o = R_{o_x}\,\vec{i} + R_{o_y}\,\vec{j} + R_{o_z}\,\vec{k}$$

from which

$$\vec{R}' = (R_x - R_{o_x})\,\vec{i} + (R_y - R_{o_y})\,\vec{j} + (R_z - R_{o_z})\,\vec{k}$$

Since the orbital plane of the satellite is inclined with respect to the equatorial plane, the satellite's position is more easily defined in terms of angular displacement in orbit with respect to an equator crossing. This angle is the independent variable C.

The rotation of the earth introduces retrograde motion to the satellite. Therefore, the effective equator crossing continually decreases, making the apparent equator crossing longitude a function of C. Using the following definitions:

P_o = the inclination angle of the satellite's orbital plane with respect to the equatorial plane

C = the satellite's angular displacement above the line of intersection of the equatorial plane and the satellite's orbital plane

E = the relative longitude of the apparent equator crossing

E_o = the relative longitude of the true satellite equator crossing with respect to the ELT (east longitudes are considered positive)

ω_s = the angular velocity of the satellite

ω_e = the angular velocity of the earth's rotation.

The following relationships apply:

$$E = E_o - \omega_e/\omega_s\, C = E_o - 0.070485\, C$$
$$R_x = R(\cos C \cos E - \sin C \cos P_o \sin E)$$
$$R_y = R(\cos C \sin E + \sin C \cos P_o \cos E)$$
$$R_z = R(\sin C \sin P_o)$$
$$R_{o_x} = R_o \sin A_o$$
$$R_{o_z} = R_o \cos A_o$$

The magnitude of R′ is:

$$R' = \sqrt{(R_x - R_{o_x})^2 + R_y^2 + (R_z - R_{o_z})^2}$$

The next step, rotation of the X- and Z-axes through angle A_o, is accomplished by defining a new axis system, X′, Y, and Z′, so that:

$$\vec{i}' = \vec{i} \cos A_o - \vec{k} \sin A_o$$
$$\vec{k}' = \vec{k} \cos A_o + \vec{i} \sin A_o$$

from which:

$$\vec{i} = i' \cos A_o + \vec{k}' \sin A_o$$
$$\vec{k} = \vec{k}' \cos A_o - \vec{i}' \sin A_o$$

$$\vec{R}' = [(R_x - R_{o_x}) \cos A_o - (R_z - R_{o_z}) \sin A_o]\, \vec{i}' + R_y\, \vec{j}$$
$$+[(R_x - R_{o_x}) \sin A_o + (R_z - R_{o_z}) \cos A_o]\, \vec{k}'$$

The polar coordinate angles are:

$$A' = \text{arc cos} \left[\frac{(R_x - R_{o_x}) \sin A_o + (R_z - R_{o_z}) \cos A_o}{R'}\right]$$
$$B' = \text{arc tan} \left[\frac{R_y}{(R_x - R_{o_x}) \cos A_o - (R_z - R_{o_z}) \sin A_o}\right]$$

The proper quadrant for angles A′ and B′ must be determined from the signs of the X, Y, and Z components:

1. If $[(R_x - R_{o_x}) \sin A_o + (R_z - R_{o_z}) \cos A_o] < 0$, then 180 degrees must be added to A′.
2. If $[(R_x - R_{o_x}) \cos A_o - (R_z - R_{o_z}) \sin A_o] < 0$, and
   (a) $R_y < 0$, then 180 degrees must be subtracted from B′
   (b) $R_y > 0$, then 180 degrees must be added to B′.

A calculator program was written to solve this set of vector equations and plot the results on polar coordinates. The results of this effort are contained in Appendix A. Figures A-1, A-2, and A-3 are plots for ELT latitudes of 0, 30, and 45 degrees, respectively. The plots depict the locus of the satellite in terms of elevation angle (altitude) above the horizon and azimuth for a specific equator crossing longitude relative to the ELT. This system of locating a satellite is identical to the alt-azimuth method of locating celestial objects. It has the advantage of allowing the occultation caused by local terrain to be considered. The azimuth angles are with respect to north. The elevation angle is the radial distance, ρ, with respect to the center; the center is 0-degree elevation angle and each major circle represents increments of 30 degrees of elevation angle. The plots, when used with Table A-1, can be used to determine the length of time a satellite will be visible above a given elevation angle as a function of relative position.

2.4.3 Mutual Visibility Requirements for Real-Time Coverage of CONUS, Southern Canada, and Alaska

The preceding analyses have dealt only with the problem of the relative position of ELTs or EPIRBs and a satellite. For worldwide coverage, it is assumed that the data is dumped. Obviously, this approach has the disadvantage of increasing the waiting time by at least one orbital period or approximately 100 minutes. This waiting time can be eliminated by having ground stations mutually visible with an ELT or EPIRB and the satellite. Previous studies[5] have shown that such mutual visibility can be achieved for CONUS, Alaska, and southern Canada by locating ground stations at Elmendorf AFB, Alaska; San Francisco, California; and St. Louis, Missouri. The mutual visibility coverage area for a minimum elevation angle of 20 degrees between an ELT/EPIRB and a satellite is shown in Figure 2-4. This information was generated assuming that the satellite would function only as a repeater, i.e., no on-board processing. However, the information is valid for the low-orbiter system being considered except that the coverage area will be *greater* since a minimum elevation angle of 10 degrees is being assumed for this system. Alternatively, the coverage area shown may be interpreted as an area of high accuracy for the waiting times given in Table 2-4.

As previously mentioned, one ground station is assumed to be located at Elmendorf AFB, Alaska. This location is near 60°N latitude and can provide the minimum one-orbit delay 50 percent of the time for ELTs and EPIRBs located outside the mutual visibility area. Two approaches can be used to improve worldwide coverage. One is to build a fourth ground station in northern Finland, Sweden, or Norway. This station would provide the data dump point for the eastern hemisphere. A second approach would be to move the station in Alaska from Elmendorf to Prudhoe Bay. This point would provide single-orbit access 90 percent of the time. (It would still provide mutual visibility coverage of Alaska.) The favored technique would be the fourth station, however.

2.5 ESTIMATED ELT/EPIRB POPULATIONS (UNITED STATES ONLY)

The size of the ELT/EPIRB detecting and locating system is established by the maximum number of such devices that may be expected to be within the satellite's field of view at any given time. This can only be determined from accident statistics. Therefore, a review of maritime search and rescue activities and aircraft accidents is in order.

U.S. Coast Guard statistics[1] indicate that during 1972, approximately 51,000 SAR cases were handled. Of these, 48,500 occurred within 25 miles of the shore. The statistics, however, do not indicate how many of these cases involved marine vessels. For the purpose of determining the maximum system size, it will be assumed that all would be attributable to an EPIRB alert had an operational system existed at the time. The 48,500 annual cases equate to an average (mean) number of 132 cases/day. If the peak demand (or crest factor) is assumed to be three times the mean, the system would be required to service 396 EPIRBs. This represents the maximum demand factor expected on the system assuming full deployment of EPIRBs.

In addition to serving EPIRBs, the system must accommodate the aviation community's ELTs. The statistics for this case are better defined for the actual number of accidents occurring, but are obscured by the huge number of false alarms experienced with current ELTs. For the purpose of this study, it will be assumed that the second generation ELTs will overcome the false alarm problems experienced with current designs.

The statistics for the number of aircraft accidents were taken from the NTSB Aircraft Accident Report[6] for 1975. The total number of aircraft accidents reported for 1975 was 4,239. Of these, 3,569 occurred during daylight hours. Some of these accidents occurred on airports (i.e., aircraft not in flight) and would not have resulted in an ELT alert. These inaccuracies are more than compensated for by the number of false alarms currently experienced, but should be comparable to an acceptable false alarm rate for future ELTs. Therefore, the accident statistics will be taken as representative of ELT alerts.

The daily accident rate was determined for the months of June through December for 1975. The total number of accidents for this period was 2,607 and the peak number was 34 accidents recorded on August 9. The statistics for this period are:

Total	n	= 2607
Mean	$\bar{x}$	= 12.18
Peak	x_{max}	= 34
Standard deviation	σ	= 5.64
Standard error	$\sigma/\sqrt{n}$	= 0.39
Crest factor	$x_{max}/\bar{x}$	= 2.79

The monthly statistics for this period are shown in Table 2-5. The combined (ELT and EPIRB) average demand rate for the system is:

$$\bar{x} = 132 + 12$$
$$= 144$$

The combined peak demand is:

$$x_{max} = 396 + 34$$
$$= 430$$

TABLE 2-5. 1975 AIRCRAFT ACCIDENT STATISTICS

Month	Total Accidents	Mean $\bar{x}$	Peak x_{max}	Standard Deviation σ	Standard Error $\sigma/\sqrt{n}$	Crest Factor $x_{max}/\bar{x}$
June	449	15	27	5.0	0.90	1.8
July	475	15	30	5.3	0.95	2.0
August	487	16	34	5.4	0.97	2.1
September	382	13	25	5.6	1.01	1.9
October	328	11	18	4.3	0.77	1.6
November	271	9	17	4.1	0.75	1.9
December	215	7	14	2.9	0.52	2.0
June ↓ December	2607	12.18	34	5.64	0.39	2.79

It should be noted that while a crest factor of 3 was assumed for the marine case (no detailed statistics were available), the crest factor of 2.79 for aircraft accidents was determined from actual statistics. This tends to substantiate the validity of assuming a crest factor of 3 for the marine cases. However, it is very unlikely that the peak ELT demand would coincide with the peak EPIRB demand. Therefore, the maximum demand rate for the system will be established as 400 per day. This is also the maximum that can be within the field of view since the ELTs are assumed to be within the contiguous United States and the EPIRBs are assumed to be within 25 miles of the shoreline. (The satellite viewing angle of 56 degrees allows it to see the entire United States when over Kansas.)

Before leaving the subject of population, it should be stated that the use of EPIRBs on all boats and ships is not expected to be required before 1990. Therefore, the assumption of a demand rate of 400 may seem excessive. However, this is not generally true since no allowance has been made for false alarms. As the number of users increases, the number of false alarms must decrease. Hence, the use of 400 for setting the system size seems justifiable.

2.6 MESSAGE FORMAT

It is desirable to transmit, as a minimum, a unique address code with each ELT or EPIRB signal. If the signal transmission is based on burst emission, then an ID code is not only desirable but necessary. If the transmission is continuous, then the satellite receiver can lock to and track the signal. If burst transmission is employed, then successive transmissions can only be correlated via a unique address code. A unique code is also desirable since this information can be compared with existing information on the craft, e.g., flight plan information. It is also desirable to be able to identify the class of user, e.g., aircraft, ship, etc., and to enable the user to manually delineate the nature of the emergency.

The processing of received signals for both the doppler position system and the interferometry system requires the transmission of an unmodulated carrier during the initial portion of the burst signal. Therefore, the message format contains a burst of unmodulated carrier at the beginning of the transmission.

The first step in recovering the data is to acquire the bit sync clock. This task is simplified if a sample of the data clock is transmitted before the data. Thus, the unmodulated carrier is followed by a sample of the data clock.

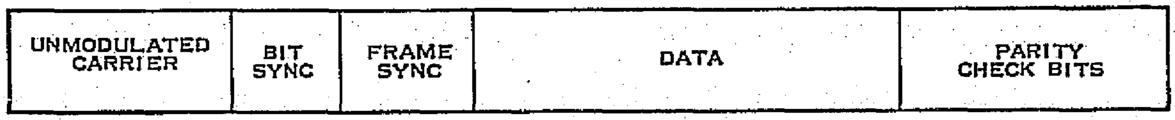

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Figure 2-5. Message Structure

Before data can be properly decoded, it is necessary to delimit the beginning of the data. This is accomplished by transmitting a known frame sync word following the bit sync. Considerable research has been devoted[7] to optimum frame sync codes. In general, the longer the code, the less the likelihood of incorrectly acquiring frame sync.

The desired message bits (and any parity bits) are transmitted following the frame sync code. This message format is similar to that used on RAMS. The relationships of various portions of the message are depicted in Figure 2-5.

2.6.1 ID Numbers

The length of the ID number depends on the information associated with the number. The simplest technique is to arbitrarily assign a unique number to each ELT/EPIRB. A master cross-reference would then be used to identify the craft and/or person with this unique number. The disadvantage of this technique is that a serial number such as this is meaningless without the cross reference, i.e., the number itself is not generally recognizable. Such a system increases the amount of paperwork required at the user end to ensure up-to-date registration information and also increases the chance of erroneous information.

A more desirable alternative is to make the ID code number the same as the craft's registration number (or an individual's social security number in the case of a personal ELT). This technique eliminates the need to establish and maintain a file for unique ELT serial numbers and the corresponding cross-reference. It also enables the identity of the craft to be readily established without the use of a central information file.

The major user categories presently defined for the ELTs/EPIRBs are: aircraft, marine vessels, and individuals. The ID number field in the data format should be capable of accommodating any registration number anticipated for any category. The user code can be defined with 2 bits.

The aircraft category can be broken down into two classes: domestic aircraft and foreign aircraft. The domestic aircraft registration numbers consist of various combinations of numerals and letters that may total as many as six characters. Therefore the ID number field must accommodate up to six alphanumeric characters. Most foreign aircraft registration numbers can also be accommodated with six characters (Appendix B).

Marine vessels are also broken down into two classes. The first class consist primarily of large ships that are commonly referred to as documented vessels. These ships carry a unique registration number called the documentation number. This number is filed with the vessel's registration papers in the Coast Guard office. The unqiue number is a standard 6-digit number.

TABLE 2-6. DATA FORMAT

User Class	User Code	ID Code	Total Number of Bits Required
Undefined	00	Not defined	
Aircraft	01	Requires six 7-bit ASCII characters	44
Marine	10	Requires four 7-bit ASCII characters followed by five 4-bit BCD numerals	50
Individual	11	Requires nine 4-bit BCD numerals	38

Small state-registered boats are not so simply standardized. However, a cursory review of present state registration systems indicates that the general format of the registration consists of two letters, followed by either four or five numerals, followed by two more letters. The first two letters designate the state in which the boat is registered. The last two letters indicate the section of the state in which the boat is registered. The numerals are a unique serial number.

The class of emergency beacons intended for individual use has not been defined. However, two means of assigning a unique number to each unit have been considered. The first is to assign serial numbers to each unit. When a unit is purchased, the serial number would be registered together with the name, address, and any other pertinent information concerning the individual. In the case of shops that may choose to rent such equipment to individuals, the shop would be required to maintain customer information correlated with each rental unit, and ultimately forward such information to a central data file via a data link. A second technique is to use an individual's social security number for identification. In either case, a simple 9-digit number would suffice.

The variety of ID number formats requires that the code generation be general. For those IDs that require combinations of numerals and alphabetical characters, standard 7-bit ASCII characters should be used. For numerals only, BCD codes should be used. To minimize the number of data bits, it will be necessary to utilize the user codes to determine how to interpret the bits received in the ID field. For example,in the case of aircraft, the first 42 bits of the ID code would be interpreted as 7-bit ASCII characters. For marine vessels, the first 28 bits of the ID code would be interpreted as 7-bit ASCII characters while the remaining 20 bits would be interpreted as BCD numerals. (This assumes that the last two letters of a state registered boat are moved to follow the first two letters.) For individuals, the full ID field would be interpreted as BCD. This information is summarized in Table 2-6. The maximum ID code length is 48 bits.

2.6.2 Emergency Codes

In general, the user should be able to indicate the extent and the type of emergency prevailing at the time. (In the case of automatic activation, the general mayday alarm is automatically transmitted; i.e., a worst case emergency is assumed unless manually overridden.) If a 5-bit emergency code is assumed, then 32 distinct emergencies can be delineated. If these codes are grouped with the user category, then 32 distinct emergencies can be defined in each category.

It is beyond the scope of this study to define emergency codes for the system. Consideration has been given to emergency codes only to the extent necessary to define the message format. Candidate emergency codes are:

1. Out of fuel
2. Hijacked
3. Fire
4. Loss of transmitter
5. Loss of transmitter and receiver
6. Loss of NAV/COM
7. Lost
8. Medical emergency
9. Power failure
10. Ice
11. Ditching
12. Foundering
13. Aground
14. Abandoned ship
15. General emergency (mayday)

The 5 bits are allocated to the emergency code. However, 1 bit in the emergency code could be moved to the user code to provide up to eight user categories if other considerations make this preferable.

2.6.3 Error Detecting/Error Correcting Codes

The use of some form of error detecting or error correcting code or both has been considered. The use of error correcting codes in environments subject to interference or noise is desirable. However, the use of such codes has the effect of increasing either the duration of the message or the data rate. The same is true of error detecting codes, but to a lesser degree. For the differential doppler positioning system, which requires a minimum of three signal receptions, the correct determination of the ID code for each transmission is very important. The possibility of widely varying transmitter signal strengths and the rather large path losses associated with a geostationary satellite system make the use of error correcting codes attractive.

The information bits consists of 2 user code bits, 48 ID code bits, and 5 emergency code bits for a total of 55 information bits. The simplest error detecting scheme is the even or odd parity check. This requires the use of 1 additional bit that will detect all single bit errors, and in general the presence of an odd number of errors. It would definitely appear worthwhile to use, as a minimum, a single-bit parity check.

Before evaluating the use of more sophisticated techniques, the characteristics of the communications system and the communications channel must be carefully considered. First, the method of transmission is considered to be of the pulsed type with a low duty cycle. Therefore, once a transmitter is activated, the emitted signals may be considered to be deterministic, stationary, and periodic. (For FSK or MSK, the strict definition of periodic may imply that the signal is nonperiodic.) For a geostationary satellite, the characteristics of the channel and the received signal are not significantly altered. For a low-orbiting satellite, both the amplitude and frequency of the received signals vary; thus the received signal is neither stationary nor periodic. Therefore, the normal connotation of BER does not apply. If the channel noise is considered stochastic, stationary, and ergodic (as is reasonable since the probability of cochannel interference or having two transmissions occupy the same time and frequency are considered separately), then the concept of BER can be used to determine the probability of success in correctly decoding a message and ultimately in determining the location of the transmitter. These ideas can be placed in better perspective by stating them another way: If the message is assumed to be encoded, then a single error in the frame sync code, the user code, or the ID code will result in loss of the message. Since a minimum of three frequency measurements is required in the differential doppler positioning system, the loss of a single message could result in unsuccessful position determination. Thus, a modest BER of 10^{-3} could result in a rather high probability of failure. For example, consider the cases where three and only three messages are received. If the source is to be correctly located, then all 195 bits must be correctly decoded. A BER of 10^{-3} implies that approximately one out of every five signal sources will not be located. Thus, a strong case is made for incorporating some type of error correcting code in the message. Any such code should, for a given BER, result in a high probability of success in decoding the message.

The first step is to assume some refer nce BER. For the remainder of the analysis, the assumed BER is 1×10^{-3} unless otherwise stated. The fundamental equation to be used is:

$$p_n(k) = \binom{n}{k} p^k q^{n-k}$$

where

$p_n(k)$ = the probability of having k errors in n bits

p = the probability of error

$q = (1 - p)$

The first case considered is that for frame sync because without frame sync the data cannot be properly decoded. If a 15-bit frame sync code is assumed and a BER of 10^{-3} exists, the probability of successfully decoding frame sync is:

$$\overline{p_{15}}(k) = \binom{15}{0} \cdot (1 \times 10^{-3})^0 \cdot (0.999)^{15}$$

$$= 0.9851$$

and the probability of not recognizing frame sync is:

$$p_{15}(k) = 1 - \overline{p_{15}}(k) = 0.0149$$

This is a relatively large number. Next consider the case where a single error is allowed. The total probability of success is:

$$p_{15}(0) + p_{15}(1) = 0.999896$$

and the probability of failure is 0.000104. This represents more than two orders of magnitude improvement in the probability of success and results from being able to tolerate one error. Therefore, single error tolerance within the frame sync code would definitely be advantageous.

Next consider the coding of the 55 message bits. There are two basic means of encoding these bits. One is to break the message into words and encode each word. The second is to encode the entire message. Intuitively it would appear that the latter case would allow more flexibility and, thus, would be superior. However, the coding scheme must be considered since the latter would, in general, require additional check bits.

Compare the probability of failure to properly decode a message consisting of 55 information bits and 20 check bits for: (1) a code having five 15-bit words and a single-error correcting code and (2) a code consisting of a single 75-bit word having 20 check bits (assuming such a code exists) capable of correcting three errors. The probability of failure for the first case is:

$$\begin{aligned} p(f) &= 1 - [1 \cdot P_{15}^5(0) + 5 \cdot P_{15}^4(0) \cdot P_{15}(1) + 10 \cdot P_{15}^3(0) \cdot P_{15}^2(1) \\ &\quad + 10 \cdot P_{15}^2(0) \cdot P_{15}^3(1) + 5 \cdot P_{15}(0) \cdot P_{15}^4(1) + P_{15}^5(1)] \\ &= 1 - 0.9427221 \\ &= 0.05727790 \end{aligned}$$

The probability of failure for the second case is:

$$p(f) = 1 - \sum_{i=0}^{k} f(k_i)$$

where

$$f(k_i) = \binom{n}{k_i} p^{k_i} q^{n-k_i}$$

If

$$n = 75$$
$$k = 3$$

then

$$\begin{aligned} p(f) &= 1 - 0.9999988541 \\ &= 0.000001146 \end{aligned}$$

Thus, the results are overwhelmingly in favor of encoding the entire message as a group. In fact, the probability of success in receiving a message even with a BER of 10^{-2} is 0.993 if provision is made to correct three errors in any given message. The limiting factor becomes the probability of successfully receiving frame sync. Previous calculations allowed only a single error in the received frame sync. If two errors are allowed, the probability of failure is reduced to:

$$4.5 \times 10^{-7}$$

The probability of failure to correctly receive a message becomes:

$$\begin{aligned} p\,(\text{failure}) &= 1 - (0.9999988541)\,(0.99999955) \\ &= 1.6 \times 10^{-6} \end{aligned}$$

If the BER falls to 1×10^{-2}, the probability of failure becomes:

$$7.4 \times 10^{-3}$$

The previous considerations have dealt only with the probability of missing frame sync. Another possibility is that of false sync. This possibility exist three ways: (1) random noise can cause false frame sync, (2) a combination of random noise/data can cause false frame sync, or (3) data can cause false frame sync. The probability of the first condition can be practically eliminated by simply not looking for frame sync until bit sync is established. The third condition is virtually bypassed if only a single frame sync word is allowed for any one message. Therefore, the possibility of a false frame sync event is reduced to that produced by a poor BER condition at the time frame sync is being received, i.e., during an overlap condition. This situation has been carefully evaluated in terms of optimum frame sync codes[7] as shown in Figure 2-6. Unfortunately, the curve is valid only for a probability of error of 0.1. However, the probability of false sync occurring can be assumed to decrease as the BER improves. If a 15-bit frame sync code length with a two-error tolerance is assumed, Figure 2-6 indicates that the probability of false sync is 0.01. At first glance this does not appear suitable; however, this corresponds to a 0.01 BER environment. It appears that the probability of false sync is exponentially reduced as BER improves and that a BER of 1×10^{-3} would yield a probability of false sync at least as good as 1×10^{-3}.

The probability of false frame sync should be the same order of magnitude as the probability of incorrectly decoding the message. The probability of false sync for a 24-bit frame sync code (Figure 2-6) is approximately 1×10^{-5}. This allows 2 bits to be in error and corresponds to a BER of approximately 5×10^{-3}. The probability of false frame sync for a BER of 1×10^{-3} should be on the order of 1×10^{-6}. Thus, a 24-bit frame sync code has a probability of false sync that is consistent with the error correction capability of the data code. Before concluding, it is necessary to evaluate the probability of having more than two errors in the frame sync code. This is approximately 2×10^{-6}. Therefore, a 24-bit frame sync code length is consistent for a BER of 10^{-3}. The probability of failure to receive a message is found as follows:

1. Assume a BER of 10^{-3}.
2. Assume a 24-bit frame sync code with a probability of false sync of 1×10^{-6}.
3. Assume a message code length of 76 bits capable of correcting three errors.
4. The probability of failure is approximately the following sum (neglecting joint probability effects):

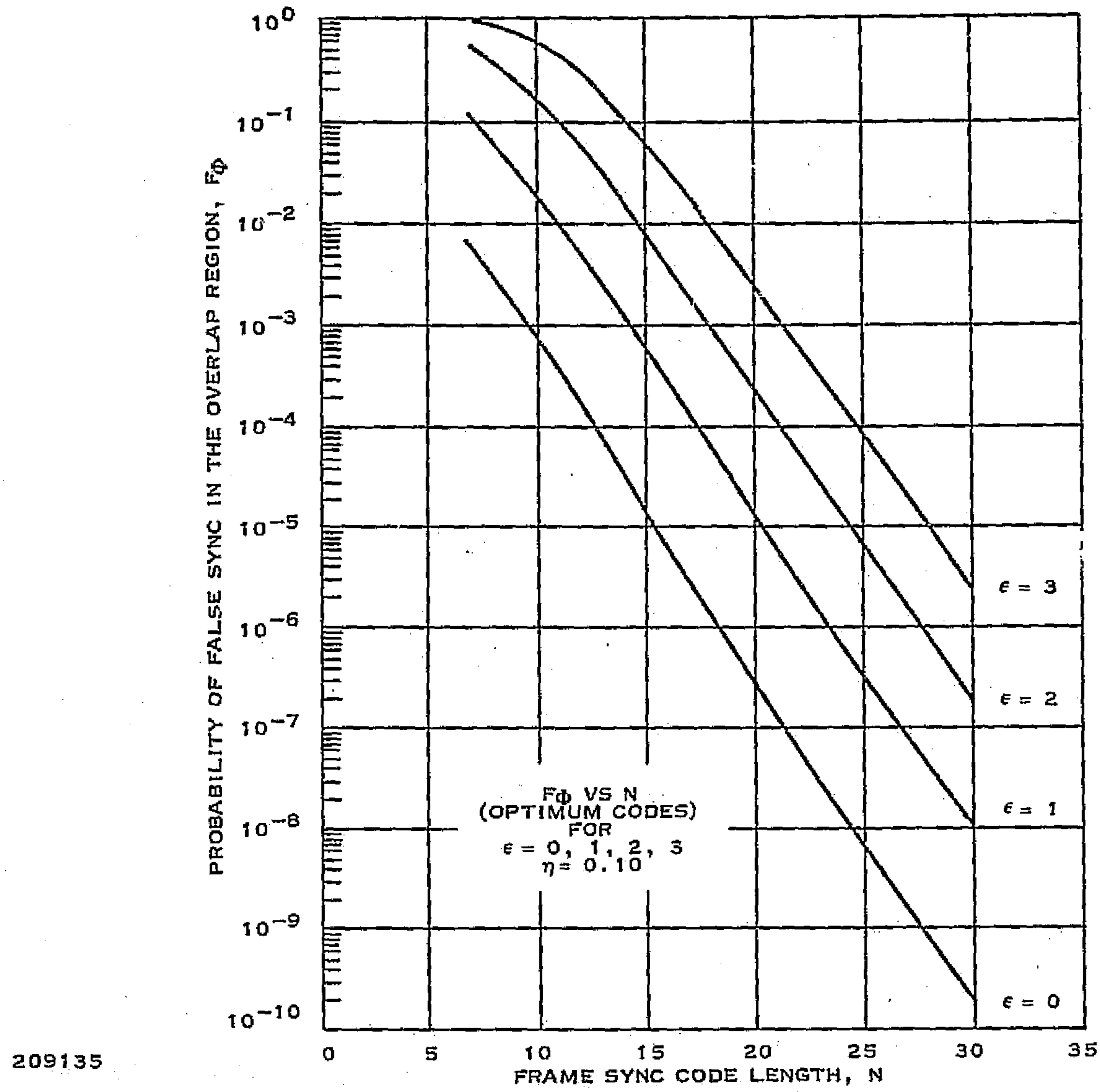

Figure 2-6. Graphic Representation of F_ϕ

$$
\begin{aligned}
P(f) &= P(\text{false frame sync}) + P\ (\text{missed frame sync}) \\
&\quad + P(\text{more than three errors in the data message}) \\
&= 1 \times 10^{-6} + 2 \times 10^{-6} + 1.2 \times 10^{-6} \\
&= 4.2 \times 10^{-6}
\end{aligned}
$$

The final consideration of the coding problem is selection of the code. There are various codes to choose from. The oldest and perhaps best known error correcting code is the Hamming Code.[8] However, it can only correct single errors within a block or word. There are several multiple-error correcting codes, the best known being the Reed-Muller, the Bose-Chaudhuri, and the Fire codes.[8] The last group, the Fire codes, are cyclic codes that are capable of correcting burst errors; i.e., they are best suited to cases where errors are expected to occur as adjacent bit errors. There is no reason to believe that errors in this system will be generated in this manner. Also, because Fire codes generally tend to be very long, they are not considered further.

The Reed-Muller codes and the Bose-Chaudhuri codes meet the error correcting requirements. The Reed-Muller codes have word lengths of 2^S while the latter have length of $2^s - 1$. Since 55 information bits are to be transmitted, $2^s > 55$ or $s > 5$. Since $2^s = 64$ for $S = 6$, only nine check digits would be available for a code of this length. A check of primitive codes indicates that a code of 2^6 bits cannot correct three errors with only 9 check bits. Thus, the minimum value of s is 7; i.e., s = 7. Hence, the code word must be 128 bits for the Read-Muller code or 127 bits for the Bose-Chaudhuri code.

The Reed-Muller code will correct errors in any received message if the number of errors is e or less where:

$$e = 2^{s-r-1} - 1$$

Since it is desired to correct up to three errors within any received word, e = 3. The value of s is taken as 7. Hence r is found to be 4. The number of check bits, m, is given by:

$$\begin{aligned} m &= 1 + {}_sC_1 + {}_sC_2 + \ldots + {}_sC_{s-r-1} \\ &= 1 + \binom{7}{1} + \binom{7}{2} \\ &= 29 \end{aligned}$$

The number of information bits allowed is therefore:

$$\begin{aligned} k &= n - m \\ &= 128 - 29 \\ &= 99 \end{aligned}$$

The code is a (128, 99) code.

The encoding operation for this code requires the storage of s + 1 or eight basis vectors (or an algorithm for generating them) of length 128, the capability of performing modulo 2 multiplication, and the ability to perform modulo 2 addition. The major disadvantage of this code is that it allows 99 information bits, yet only 55 are required. It is desired to shorten the code length by the amount of the unused bits. However, the Reed-Muller code is not a systematic code and does not permit this. Such codes may exist, but are not easily determined.

The final class of codes considered is that of cyclic codes, and only the Bose-Chaudhuri (B-C) group is considered in detail. All B-C primitive codes of length through 255 bits are shown in Table 2-7. The shortest code capable of handling 55 information bits and correcting three errors is the (127, 106) code. However, since only 55 information bits are needed, there are 51 unused bits, suggesting that a shortened cyclic code be used. This code is formed by subtracting 51 from both n and k to yield a (76, 55) code. This code is also a triple-error correcting code. The only constraint is that the code must be systematic.

The advantage of cyclic codes is that they can be implemented with shift reigsters and generally require less hardware than the Reed-Muller codes. There are two methods of implementing cyclic codes with shift registers and achieving a systematic code. One is to use a

TABLE 2-7. BOSE-CHAUDHURI CODES GENERATED BY PRIMITIVE ELEMENTS

n	k	t	n	k	t	n	k	t	n	k	t
7	4	1	63	16	11	127	15	27	255	123	19
15	11	1		10	13		8	31		115	21
	7	2		7	15	255	247	1		107	22
	5	3	127	120	1		239	2		99	23
31	26	1		113	2		231	3		91	25
	21	2		106	3		223	4		87	26
	16	3		99	4		215	5		79	27
	11	5		92	5		207	6		71	29
	6	7		85	6		199	7		63	30
63	57	1		78	7		191	8		55	31
	51	2		71	9		187	9		47	42
	45	3		64	10		179	10		45	43
	39	4		57	11		171	11		37	45
	36	5		50	13		163	12		29	47
	30	6		43	14		155	13		21	55
	24	7		36	15		147	14		13	59
	18	10		29	21		139	15		9	63
				22	23		131	18			

k-stage feedback shift register and the other is to use an n-k stage shift register in conjunction with some additional hardware. Both schemes are based on inserting zeros for the k-i unused information bits before producing the parity-check bits, then discarding the k-i zeros before transmission. This is easily done with a systematic code. These zeros are then inserted in the received code word before the error correction process is applied, and then discarded in the decoded message. Selection of an encoding mechanism is not made at this point, but is deferred to Section 3; it is sufficient to know that such mechanisms exist and that they can be relatively easily implemented.

At this point, it is worthwhile to summarize the assumptions made and the conclusions reached from the preceding analyses.

1. The channel bit-error-rate is assumed to be equal to or better than 1×10^{-3}. This will now be the maximum allowable rate for the remaining system analyses.
2. The number of bits in the frame sync code should be 24 and the code should be the optimum frame sync code:

   11111-01011-11001-10010-0000

   The order of the transmitted bits is right to left.
3. The frame sync code recognizer should allow for two errors in the received code.
4. The message bits or information bits should be accompanied (followed) by parity check bits generated from a primitive B-C (127, 106) code. The resulting code will be a shortened cyclic (76, 55) code capable of correcting three or fewer errors. It can also be used to detect up to seven errors.

5. Such a system will result in a probability of success in receiving any given message of 0.999996, assuming random noise-induced errors.
6. The total length of the combined frame sync code and encoded data is 100 bits.
7. Because of the high probability of success associated with the triple-error correcting code, no error detection capability is needed. However, each received message containing no detected errors should be flagged as such to indicate the high confidence in the validity of the message. In addition, the same parity bits can be used in an error detection algorithm to do both if desired.

These conclusions were reached only after considering whether or not they could be easily implemented in an ELT and only those that are possible are listed.

2.6.4 Bit Sync Code

The characteristics of the bit sync code and its length are more appropriately considered after studying baseband data encoding and modulation techniques. However, in general, it will be n-bits of either the bit sync clock or the all-ones equivalent baseband code.

The required length of the bit-sync code is a function of the desired bit-sync phase-lock loop (PLL) acquisition loop bandwidth, the channel noise, and the number of cycles over which the lock detector is to check for lock. At this point, it will be assumed that the bandwidth of the bit sync loop is greater than the initial frequency error. The acquisition time is then only a function of the transient behavior of the loop and the characteristics of the phase detector. If the phase detector can produce a continuous output as a function of phase error, then acquisition is a function only of the loop parameters ω_n and ζ (disregarding any derogatory effects of noise). If the loop bandwidth is no less than 20 percent of the data rate, the acquisition time will be about 3T or three bit times. If a lock detector is to be used, a minimum of four bit times is needed to determine lock. Therefore, the minimum number of bit times required for the bit sync code is seven. However, because of the importance of establishing bit sync lock, the length of the bit sync code should be doubled. This allows some flexibility in the design of the bit sync PLL (loop bandwidth, damping, phase detector, etc.) and some tolerance for noise. Therefore, the recommended bit sync code is 15 bit periods. The final selected message format is shown in Figure 2-7.

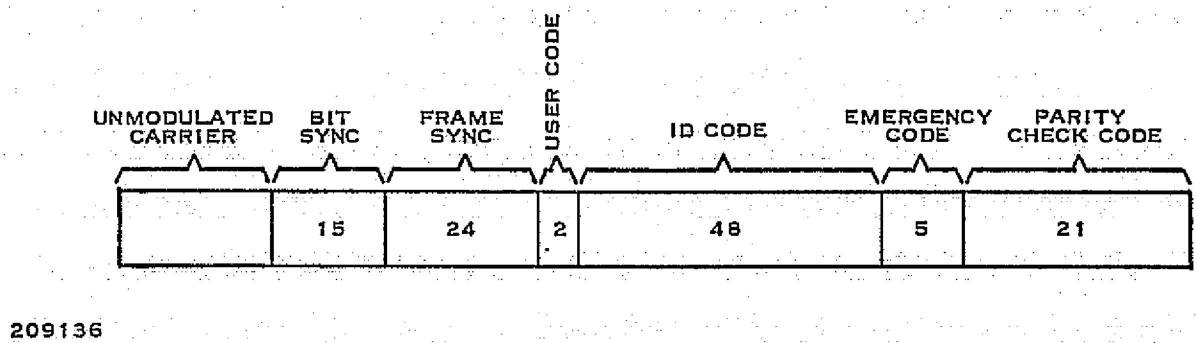

Figure 2-7. Message Format

2.7 SPECTRUM ALLOCATION

The internationally allocated frequency band for mobile-satellite (earth-to-space) EPIRB service is 406.0 to 406.1 MHz. This section treats the uses of this band to achieve the desired service.

The first variable for which spectrum allocation must be made is the frequency tolerance of the ELT/EPIRB transmitter. A review of the specifications for economical commercial crystals suitable for carrier generation for this band indicates that a tolerance of ±0.003 percent is to be expected over the operating temperature range. Therefore, the portion of the spectrum to be allocated to frequency tolerance is:

$$\Delta f = \pm 4.06 \times 10^8 \times 3 \times 10^{-5}$$
$$\cong \pm 12.1 \text{ kHz}$$

for a total of 24.2 kHz.

The second tolerance to be considered is that associated with doppler shift introduced by satellite motion. The maximum shift to be expected here occurs when the satellite subtrack passes through the ELT/EPIRB position and the satellite appears on the horizon. This situation is shown in Figure 2-2. The ELT/EPIRB is located at the intersection of R_o and d; the satellite is located at the intersection of d and h. The appropriate doppler shift equation is:

$$\Delta f_d = f_t \left(\frac{v}{C}\right)$$

where

f_t = the transmitted frequency

C = propagation velocity of light in free space (2.99793 X 10^5 km/s)

v = the relative velocity along the communications path.

The sign of the shift is positive if the satellite is approaching, but negative if the satellite is receding. The tangential velocity of the satellite is found from the previously calculated orbital period and the circumference of the orbit:

$$|\vec{\nu}_\tau| = \frac{2\pi(R_o + h)}{60T}$$
$$= 7.433 \text{ km/s}$$

The component of this velocity vector that lies along the communications path is:

$$|\vec{\nu}_d| = |\vec{\nu}_t| \sin \alpha$$
$$= |\nu_t| \cos \theta$$
$$= 6.572 \text{ km/s}$$

The maximum doppler shift is then:

$$\Delta f_d = 4.06 \times 10^8 \text{ Hz} \left(\frac{\nu_d}{C}\right)$$

$$= 8.9 \text{ kHz}$$

The portion of the spectrum that must be allocated to doppler shift is, therefore, ±8.9 kHz or a total of 17.8 kHz.

The next consideration is that of modulation sidebands. Since the spectral characteristics of the modulation sidebands depend on the form of modulation chosen, the baseband data code, and the data rate (none of which have been discussed to this point), assumptions will be based on preliminary considerations of data rate and worst case modulation characteristics. A data rate of 200 BPS and Manchester PSK data are assumed. Thus, the channel bandwidth is approximately twice the data rate or 400 Hz. However, the sidebands will be assumed to occupy ±0.5 kHz or a total of 1 kHz.

The total portion of the spectrum used to this point is:

$$24.2 \text{ kHz} + 17.8 \text{ kHz} + 1 \text{ kHz} = 43.0 \text{ kHz}$$

which represents only 43 percent of the allocated 100 kHz.

This low percentage suggests the use of two distinct channels, which means the use of two center frequencies. Band 1 and Band 2 will each use 43 percent of the spectrum for a total of 86 percent. The remaining 14 percent should be used as a guard band on each end of the assigned spectrum to act as a buffer against spurious signals generated by adjacent frequency allocations, in particular the fixed/mobile band that begins at 406.1 MHz and the meteorological-aids band that ends at 406.0 MHz. The suggested spectrum allocations are shown in Figure 2-8. The partitioning of each band can be summarized as follows:

Transmitter-frequency tolerance	24.2 kHz
Doppler shift	17.8 kHz
Modulation sidebands	1.0 kHz
Guard band and miscellaneous	7.0 kHz
Total	50.0 kHz

At this point, only worst case frequency variations have been considered. Nothing has been stated concerning the statistical variations in frequency. It may be that the frequency distribution about the mean is not uniform but is similar to a normal distribution. If this is the case, it may be desirable to use a third channel centered at 406.05 MHz. The spectrum assignment then would be:

Band 1	406.0285 MHz
Band 2	406.0715 MHz
Band 3	406.050 MHz

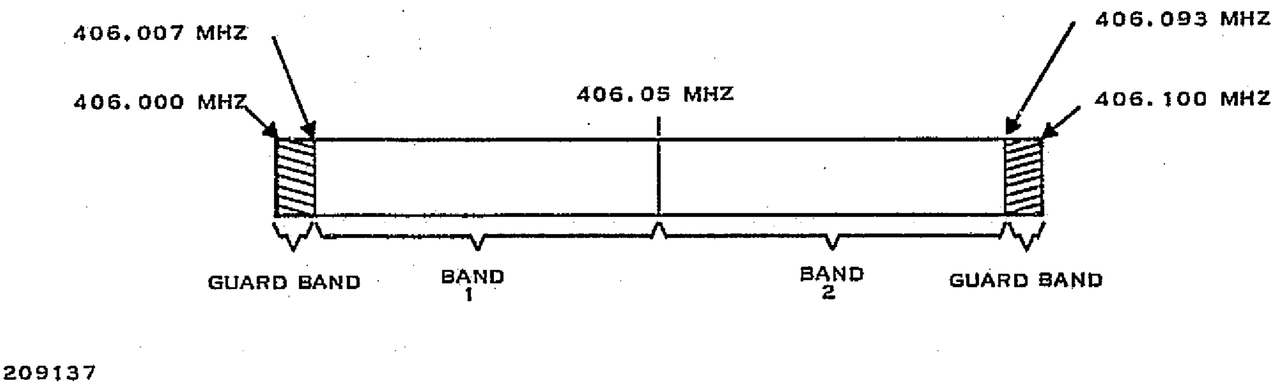

Figure 2-8. Spectrum Allocation for Satellite Emergency Locator System

2.8 ANALYSIS OF ELT/EPIRB-TO-SPACECRAFT LINK

An analysis of the ground-to-spacecraft link from an ELT/EPIRB transmitter is important since this establishes the data rate to maintain the previously determined 10^{-3} BER. Some of the system parameters are well-defined and are common to both a low-polar-orbiting spacecraft and a geostationary spacecraft, but others are defined only for one type of spacecraft; some parameters are selectable and common, while other parameters are selectable only as a function of the spacecraft. Some parameters are constant while others are a function of numerous variables.

To provide some continuity in the analysis, this subsection will define all common parameters (those common to both types of spacecraft) and discuss all variables affecting the transmission path. A numerical analysis of the link for each type of spacecraft will follow in Subsections 2.8.1 and 2.8.2. In each category, a best case will be defined and a worst case condition assumed. The maximum allowable data rate will be calculated based on the worst case link parameters. The minimum available E/n_o will also be calculated for various candidate data rates.

The mean operating frequency is 406.05 MHz and the specified maximum effective radiated power (ERP) is 5.00 watts, or 7 dBW. Since the power is specified as ERP, transmitter antenna gain and transmission line losses are not considered. The receiver thermal noise power density is given by:

$$n_o = k\,T_e$$

where

n_o = energy due to thermal noise

= thermal noise power per unit bandwidth

k = Boltzmann constant, 1.38×10^{-23} J/°K

T_e = effective noise temperature, °K.

(NOTE: This is one-sided power spectral density.)

The noise density can be expressed in decibels referenced to 1 watt (dBW) by:

$$n_o = 10 \log T_e - 228.6$$

The effective noise temperature, T_e, is:

$$T_e = T_a + (LF - 1)\, T_o$$

where

T_a = antenna noise temperature

T_o = standard reference temperature, 290°K

L = transmission line loss factor

F = noise factor of the receiver.

If the receiver noise figure is assumed to be 3 dB and the antenna-to-receiver transmission path loss is assumed to be 0.5 dB, the noise factor and loss factor, respectively, are:

$$F = 2.00$$
$$L = 1.12$$

The radio blackbody temperature of the earth is 254°K. Assuming that the spacecraft antenna beamwidth is sufficiently narrow that it "sees" only the earth, this is also T_a. The effective noise temperature is, therefore:

$$T_e = 254°K + (2.24 - 1)\, 290°K$$
$$= 613.6°K$$

The effective noise density of the receiving system is, therefore:

$$n_o = 10 \log (613.6) - 228.6$$
$$= -200.72 \text{ dBW/Hz}$$

The parameters just described are relatively constant and are common to both the geostationary satellite system and the low-orbiting system. They are summarized in Table 2-8. Parameters common to both systems, but not constant, are refraction, reflection, and absorption. The effects of refraction are noticeable only for very low elevation angles. Since a 10-degree minimum elevation angle is being assumed, refraction will not be considered further.

Absorption is likewise not considered a serious problem at the operating frequency.

TABLE 2-8. ELT/EPIRB-TO-SPACECRAFT LINK PARAMETERS COMMON TO BOTH SYSTEMS

Frequency	406.05 MHz
Transmitter power (5 watts), ERP	7.00 dBW
Effective receiver noise density assuming:	–200.70 dBW
Receiver noise figure = 3 dB	
Receiver line loss = 0.5 dB	
Antenna noise temperature = 254°K	

Precipitation, even heavy precipitation, causes less than 0.001 dB/km attenuation at 400 MHz. Oxygen absorption accounts for only 0.002 dB/km at 400 MHz. Hence, absorption is not considered further.

The multipath reflections are a possible source of signal attenuation. This is particularly true for the geostationary satellite. If a reflection path length differs from the primary path length by exactly $\lambda/2$ and the reflection coefficient is unity, the received signal strength will be zero. However, the reflection coefficient is normally less than unity and the probability of the path length differing by precisely $\lambda/2$ is small. If a reflection coefficient of 0.5 is assumed, maximum reduction in signal strength will be 3 dB. The low-polar-orbiting satellite is constantly in motion, thus producing a pseudospace-diversity effect and, hence, reducing the effects of multipath reflections. However, 3 dB worst case allowance will be made for both systems.

Other sources of loss variation are the transmitter antenna, the receiver antenna, and polarization effects. However, polarization losses can be minimized if circular polarization is used for both the transmitter and receiver. Circular polarization will be assumed and no polarization loss factors are included.

The transmitter antenna gain has been included in the specified maximum ERP. However, antenna gain variations, i.e., gain reductions, must be considered for worst case evaluation. Since the transmitter antenna selection is a function of other parameters, the transmitter antenna characteristics are defined at this point and later refinements will use these definitions. Therefore, the transmitter antenna is assumed to have the following characteristics:

- Hemispherical coverage
- Circular polarization
- 3-dB gain variation, maximum.

The receiver antenna is assumed to be a paraboloid having a design that produces a maximum variation of 3 dB over the usable beam angle. The specific design is a function of the spacecraft type. Therefore, the worst cast antenna variations are:

Transmitter antenna gain variation	3 dB
Receiver antenna gain variation	3 dB

The remaining parameters to be computed are the free space loss and receiver antenna gain. These are both functions of the satellite system.

2.8.1 Link Analysis for Low-Polar-Orbiting Satellite System

The first parameter to be chosen is the required antenna beam angle. This is determined from the geographic coverage calculations for the low-polar-orbiting satellite (Figure 2-2). The horizon-to-horizon angle is simply 2α, which is the required beam angle, β:

$$\begin{aligned}\beta &= 2\alpha \\ &= 2\ (62\ \text{degrees}) \\ &= 124\ \text{degrees}\end{aligned}$$

If a 10-degree minimum elevation angle is assumed, solution of the triangle problem yields a minimum beam angle of 120 degrees. The antenna gain (for a paraboloid) is related to the beam angle by:

$$G(\mathrm{dB}) \simeq 10 \log\left(\frac{27{,}000}{\beta^2}\right)$$

$$= 2.74 \text{ dB (or 1.88)}$$

This is the maximum antenna gain that will permit a 120-degree beam angle.

Antenna gain is related to antenna diameter and operating frequency by:

$$G = k \left(\frac{\pi D}{\lambda}\right)^2$$

where

G = antenna gain

k = antenna efficiency constant $\cong$ 0.54

D = antenna diameter in meters

λ = wavelength in meters.

The diameter of the antenna is:

$$D = \sqrt{\frac{Gc^2}{0.54\ \pi^2 f^2}}$$

$$= 0.44 \text{ meter, or approximately 18 inches}$$

where

c = free space propagation velocity, 3×10^8 m/s

f = frequency, Hz.

Thus, the receive antenna for the low-polar-orbiting spacecraft should have the following characteristics:

Paraboloid diameter	0.44 meter
3-dB beamwidth	120 degrees
Gain	2.75 dB

The free space loss for the low-polar-orbiting system is determined from the slant range from an ELT/EPIRB source to the spacecraft. This is determined by referring to Figure 2-2 and solving the triangle problem. The maximum (assuming a 10-degree elevation angle) slant range, d, is 2500 km while the minimum (also to be considered the best case) distance is 850 km. The free space loss is a function of distance and frequency, and is given by:

$$L_{fs} = 20 \log f + 20 \log d - 87.55 \text{ dB}$$

where

L_{fs} = free space loss, dB

f = frequency, Hz

d = slant-range (line-of-sight) distance, km.

For the low-polar-orbiting satellite,

$$L_{fs} = 152.6 \text{ dB maximum}$$
$$L_{fs} = 143.2 \text{ dB minimum}$$

This completes the determination of all major path gain (loss) elements for the low-orbiting system. The results are listed in Table 2-9.

TABLE 2-9. ANALYSIS OF ELT/EPIRB-TO-SPACECRAFT LINK FOR LOW-POLAR-ORBITING SATELLITE

	Worst Case	Best Case
Frequency (MHz)	406.05	406.05
Elevation angle (degrees)	10	75
Range (km)	2500	850
Transmitter power (W), ERP	2.5	5
Receiver noise figure (dB)	3	3
Transmitter power (dBW), ERP	4.0	7.0
Free space loss (dB)	−152.6	−143.2
Multipath loss (dB)	−3.0	0.0
Receiving antenna gain (dB)	0.0	2.8
Received signal power (dBW)	−151.6	−133.4
Thermal noise density (dBW/Hz)	−200.7	−200.7
Signal-to-noise density (dB-Hz)	49.1	67.3

The worst case signal-to-noise density does not allow any contingency factors such as obscured visibility, low battery power, damaged antenna, etc. If a 10-dB contingency factor is allowed and the minimum required E/n_0 ratio for maintaining a 10^{-3} BER is assumed to be 8 dB, then the maximum bandwidth is found to be:

$$B = \log^{-1}\left[\frac{49.1 \text{ dB} - 10 \text{ dB} - 8 \text{ dB}}{10}\right]$$
$$= 1288 \text{ Hz}$$

Therefore, the maximum bandwidth is 1288 Hz, based on signal-to-noise considerations for the low-polar-orbiting satellite system. The minimum available E/n_0 for various bandwidths is tabulated as follows:

128 Hz	28.0 dB
256 Hz	25.0 dB

512 Hz	22.0 dB
1024 Hz	19.0 dB

2.8.2 Link Analysis for a Geostationary Satellite System

The antenna beam angle for the geostationary satellite can be determined with the aid of Figure 2-2 by letting:

$$h = 3.585 \times 10^4 \text{ km}$$

$$\beta = 2\alpha = 2 \text{ arc sin} \left(\frac{R_o}{R_o + h}\right)$$

$$= 17.4 \text{ degrees}$$

This is the minimum antenna beam angle. However, use of such a minimum angle would be overly restrictive of satellite pointing accuracy. Therefore, a beam angle of 20 degrees is assumed. The antenna gain is:

$$G(\text{dB}) = 10 \log \left(\frac{27{,}000}{\beta^2}\right)$$

$$= 18.3 \text{ dB (or } G = 67.5)$$

and the corresponding antenna diameter is:

$$D = \sqrt{\frac{Gc^2}{0.54\pi^2 f^2}}$$

$$= 2.63 \text{ meters, or approximately 100 inches}$$

The receiving antenna parameters are summarized as follows:

Paraboloid diameter	2.63 meters
3-dB beamwidth	20 degrees
Gain	18.3 dB

The maximum free space loss is for a path length of 42,700 km and the minimum loss is for a path length of 35,850 km. The loss factors computed for the geostationary case are:

$$L_{fs} = 177.2 \text{ dB, maximum}$$

$$L_{fs} = 175.7 \text{ dB, minimum}$$

This completes the determination of the path and system parameters. They are summarized in Table 2-10.

TABLE 2-10. ANALYSIS OF ELT/EPIRB-TO-SPACECRAFT LINK FOR GEOSTATIONARY SATELLITE

	Worst Case	Best Case
Frequency (MHz)	406.05	406.05
Receiver antenna beam angle (degrees)	20	20
Range (km)	42,700	34,850
Transmitter power (W), ERP	2.5	5
Receiver noise figure (dB)	3	3
Transmitter power (dBW), ERP	4.0	7.0
Free space loss (dB)	−177.2	−175.7
Multipath loss (dB)	−3.0	0.0
Receiving gain (dB)	15.3	18.3
Received signal power (dBW)	−160.9	−150.4
Thermal noise density (dBW/Hz)	−200.7	−200.7
Signal-to-noise density (dB-Hz)	39.8	50.3

If the previous assumptions are used for minimum E/n_o and contingency factors, the maximum allowable bandwidth is found to be:

$$B = \log^{-1}\left[\frac{39.8\ \text{dB} - 10\ \text{dB} - 8\ \text{dB}}{10}\right]$$

$$= 151\ \text{Hz}$$

The minimum available E/n_o for various bandwidths is as follows:

128 Hz	18.7 dB
256 Hz	15.7 dB
512 Hz	12.7 dB

2.9 DETERMINATION OF CHANNEL BANDWIDTH

The search spectrum, f, has previously been determined to be 86 kHz (Subsection 2.7). The ultimate minimum value of Δf is set by the maximum rate of change in doppler shift. This is calculated from the rate of change in velocity along the slant-range path (refer to Figure 2-2):

$$a = \frac{Dv}{Dt} = \frac{D^2 d}{Dt^2} = \frac{D^2}{Dt^2}\sqrt{R_o^2 + (R_o + h)^2 - 2\,R_o(R_o + h)\cos\theta}$$

$$= \frac{D}{Dt}\left[\frac{R_o(R_o + h)\,\omega_s \sin\theta}{\sqrt{R_o^2 + (R_o + h)^2 - 2\,R_o(R_o + h)\cos\theta}}\right]$$

$$= \frac{R_o\,(R_o + h)\,\omega_s^2 \cos\theta}{\sqrt{R_o^2 + (R_o + h)^2 - 2\,R_o(R_o + h)\cos\theta}} - \frac{R_o^2\,(R_o + h)^2\,\omega_s^2 \sin^2\theta}{\left[R_o^2 + (R_o + h)^2 - 2\,R_o(R_o + h)\cos\theta\right]^{3/2}}$$

Since

$$\omega_s = \frac{3.7142}{3600} \text{ rad/s}$$

then

$$a \Big|_{\theta = 0} = 5.84 \times 10^{-2} \text{ km/s}^2$$

Therefore, the maximum rage of change in doppler frequency is:

$$\frac{\Delta(\Delta f)}{\Delta t} = f_c \frac{\Delta v}{\Delta tc}$$

$$= f_c \frac{a}{c}$$

$$= 79 \text{ Hz/s}$$

This is the minimum channel bandwidth. Figure 2-9 is a plot of doppler shift as a function of the satellite's latitude relative to an ELT. Figure 2-10 is a plot of the corresponding rate of change.

The portion of the total allocated spectrum occupied by a single ELT or EPIRP transmission at any given time is related to data rate, carrier-to-noise ratio, bit error rate (BER), modulation technique, and data encoding method. The optimization of these various parameters is predicated on system hardware limitations as well as system performance requirements. The following paragraphs are concerned with tradeoffs that are made to achieve the desired system performance.

To maximize the number of ELTs and EPIRBs that can be accommodated within the allocated spectrum, the amount of spectrum occupied by a single ELT/EPIRB transmission should be minimized. Because burst emission will be employed, the duty cycle of the transmission should also be minimized to accommodate the maximum number of ELTs and EPIRBs. The principal equation that states the probability of mutual interference between two transmissions is:

$$P_I = 1 - \left[1 - \frac{2\tau}{T} \cdot \frac{2\Delta f}{F}\right]^{N-1}$$

where

P_I = probability of mutual interference

N = number of ELTs and EPIRBs in the FOV

τ = duration of a burst transmission

T = period between burst transmissions

Δf = total channel bandwidth

F = total search spectrum in which signals may exist.

It is obvious from this equation that for fixed N the probability of mutual interference is minimized by minimizing τ and Δf while maximizing T and F. This assumes that the transmissions are completely random and the frequency distribution is uniform.

Note that Δf is related to τ since τ is simply:

$$\tau = \frac{n}{r}$$

and

$$\Delta f = kr$$

where

n = total number of bits to be transmitted

r = bit rate, BPS

k = channel use constant, Hz/BPS.

This implies that the product $\tau\Delta f$ is a constant set by the form of modulation and encoding (k) and the message length (n).

This leaves only the selection of the message repetition rate, T. Selection of this parameter is also influenced by the satellite visibility time. This is determined from the orbital parameters of the satellite, the minimum number of messages required to provide position location, and the minimum relative satellite displacement allowed between doppler measurements to ensure the desired positioning accuracy. For differential doppler positioning in two dimensions, a minimum of three messages must be received, which implies that the satellite must be visible for approximately three transmission periods, or 3T.

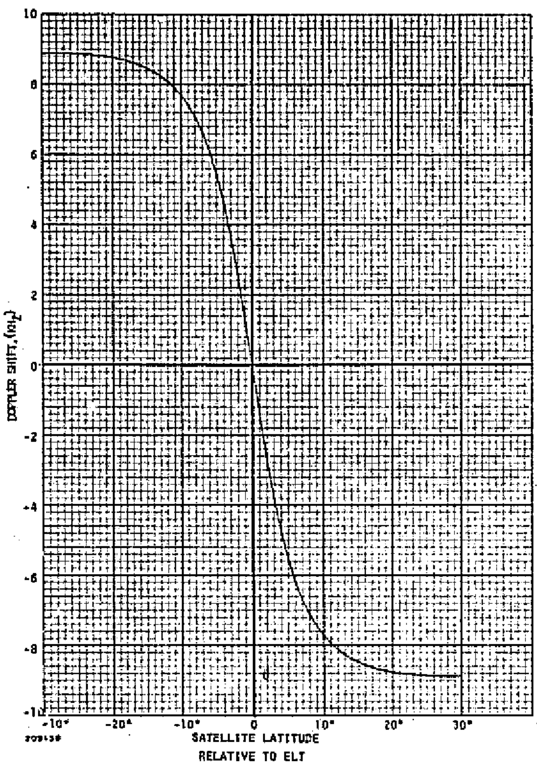

Figure 2-9. Plot of Doppler Shift Versus Relative Latitude With ELT on Satellite Subtrack

Previous considerations of geographic coverage have assumed a minimum visibility time of 4 minutes. Thus, the maximum value of T is 1.33 minutes. Conversely, the minimum time separation between doppler measurements is 40 seconds (discussed in Subsection 2.11) and the minimum value of T is 40 seconds.

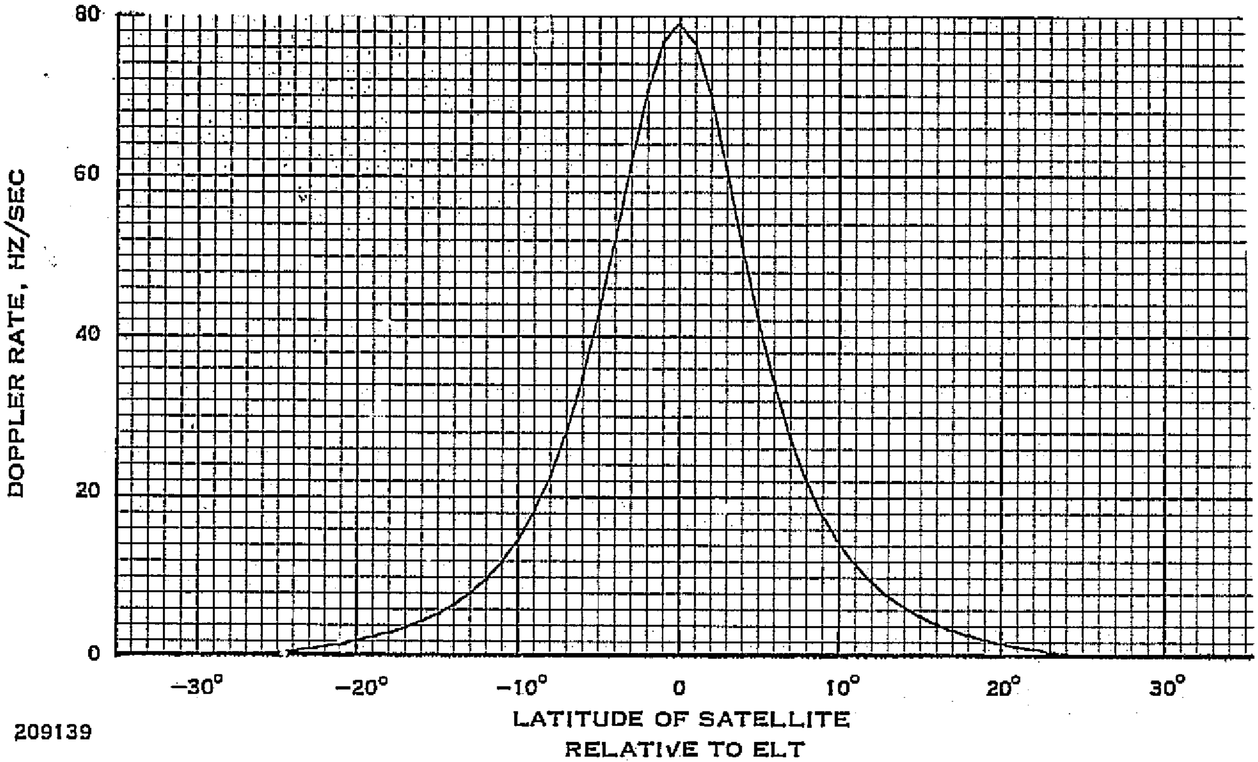

Figure 2-10. Rate of Change in Doppler Shift With ELT on Satellite Subtrack

The final consideration is the choice of making T as large as possible and requiring reception of three consecutive messages or making T smaller and only requiring reception of at least three messages in 4 minutes. This will be made by fixing the following parameters:

$F = 86{,}000$ Hz

$k = 1$ Hz/BPS

$n = 115$ bits

$N = 201$ ELTs and/or EPIRBs in the FOV.

For T = 1.33 seconds then, three out of three messages must be received. The probability of interference is:

$$P_I = 1 - \left[1 - \frac{(4)(1)(115)}{(80)(86{,}000)}\right]^{200}$$
$$= 0.0132835$$

The probability of receiving three out of three transmissions is:

$$P_s = (1-0.0132835)^3$$
$$= 0.96068$$

Now consider the probability of success for receiving three out of four transmissions with T = 60 seconds:

$$P_I = 1 - \left[1 - \frac{(4)(1)(115)}{(60)(86{,}000)}\right]^{200}$$

$$= 0.0176722$$

The probability of receiving three out of four messages is:

$$P_s = \sum_{m=3}^{4} \binom{4}{x} \cdot (1 - P_I)^x \cdot P_I^{4-x}$$

$$= 0.99817$$

It is obviously better to reduce T and increase the number of messages transmitted but to require the reception of only three.

Next, consider the case of transmitting five 48-second messages in 4 minutes.

$$P_I = 1 - \left[1 - \frac{(4)(1)(115)}{(48)(86{,}000)}\right]^{200}$$

$$= 0.02204152$$

The probability of receiving three out of five messages is:

$$P_s = \sum_{x=3}^{5} \binom{5}{x} \cdot (1 - P_I)^x \cdot P_I^{5-x}$$

$$= 0.9998964$$

These results are summarized in Table 2-11. The technique could be extended to other values of T, but this is not practical for periods less than 40 seconds since the period between doppler measurements becomes too small for adequate position accuracy.

The preceding calculations have established the minimum channel bandwidth for the system as 79 Hz. They also indicate that the data rate does not influence the mutual interference probability for a fixed message length. Only the type of message encoding, the modulation technique, the repetition rate, and the available spectrum influence the mutual interference characteristics. Therefore, the channel bandwidth should be minimized to maximize the noise immunity (best BER) and the channel use constant should also be minimized to minimize both the channel bandwidth and the probability of interference. However, since the minimum resolvable channel assignment frequency may limit the useful channel bandwidth, the analyzing bandwidth is considered next. This is followed by a study of modulation effects, baseband clock/data encoding techniques, and finally an optimum data rate. The problem of multiple access and probability of success are reviewed in Subsection 2.12 after considering other BER factors and system errors.

TABLE 2-11. DETERMINATION OF REPETITION RATE, T

T (seconds)	Number of Messages Transmitted in 4 Minutes	Probability of Interference in a Given Message *	Probability of Successfully Receiving Three Messages in 4 Minutes *	Probability of Successfully Receiving Five Messages in 4 Minutes *
80	3	0.0132835	0.960680	NA
60	4	0.0176722	0.998170	NA
48	5	0.0220415	0.999896	0.894545
40	6	0.0263915	0.999993	0.990266

*Values calculated for F = 86 kHz, k = 1 Hz/BPS, n = 115 bits, and N = 201 messages/period, T.

2.9.1 Analyzing Bandwidth

The analyzing bandwidth is the minimum frequency resolution to which the received signal frequency can be determined. This frequency is determined by the characteristics of the spectrum search circuitry and is related to channel bandwidth simply because the double-sided channel bandwidth cannot be less than the analyzing bandwidth. Thus, the analyzing bandwidth should be made as small as practical for noise and mutual interference considerations.

The two practical techniques for performing the search routine are discussed in Section 4. One proven technique is the time-compression spectrum analyzer approach as used in the Random Access Measurement System (RAMS). The second technique, using a chirp-Z transform (CZT) approach, is possible through recent technology. Some of the information relating to the practical limitations for these methods is used in this subsection.

The state of development for the CZT permits the resolution of a 100-kHz spectrum into 250 400-Hz lines. This is via the use of a 500-point CZT chip. The 500-point unit can resolve 250 real spectral lines. For a 100-kHz spectrum, the spectral resolution is:

$$\Delta f = \frac{100\ \text{kHz}}{250} = 400\ \text{Hz}$$

The length of time required to perform the analysis is:

$$t = \frac{1}{400\ \text{Hz}} = 2.5\ \text{ms}$$

Therefore, the minimum period of unmodulated carrier transmission is twice this or 5 ms. (Subsection 4.1 shows that the unmodulated carrier should exit for 100 ms.) The clock frequency for the CZT is determined from the fact that 500 coefficients (500 points) must be determined in 2.5 ms, or:

$$\tau = \frac{2.5 \text{ ms}}{500} = 5 \mu s$$

The clock frequency for a practical CZT unit being considered is:

$$f_{ck} = \frac{1}{\tau} 200 \text{ kHz}$$

Thus, the resolution of the 100-kHz band into 250 spectral lines of 400-Hz bandwidth requires a 500-point CZT with a clock rate of 200 kHz. This is near the practical limit of current technology. The resolution frequency should be 400 Hz and the analyzing bandwidth and channel bandwidth should correspond.

Having selected the minimum practical analyzing bandwidth, the determination of the best modulation technique, encoding scheme, and data rate can proceed.

2.9.2 Modulation Effects

The choice of modulation is a key factor in determining the channel bandwidth use constant, k. (This factor is discussed in Subsection 2.9.) The modulation methods considered are FSK, PSK, and MSK.

The relative spectral dispersion characteristics of these types of modulation are shown in Figure 2-11. Note that the absolute value of k in this figure is valid only for NRZ data; however, the relative characteristics are valid for any type of data (NRZ, Manchester, Miller, etc.). The amount of power outside the allocated channel determines the amount of adjacent channel interferences and the minimum received signal level that can be detected in an adjacent channel. From Figure 2-11, it is apparent that the worst adjacent channel interferences will occur with PSK modulation and the least will occur with MSK modulation. Continuous phase FSK is a close second best.

The effects of modulation sidebands on system dynamic range/mutual interference is further demonstrated by Figure 2-12. The spectral characteristics of MSK and continuous phase FSK are shown to be almost identical. It is clear that MSK or continuous phase FSK can provide a dynamic range of at least 20 dB for a channel use factor (k = bandwidth occupied/data rate) of k = 6 and can provide a dynamic range of 30 dB for k = 10. PSK requires a value of k = 12 to achieve a dynamic range of 20 dB and a value of k = 32 to achieve a 30-dB dynamic range. This implies that MSK can accommodate over three times the number of transmitters that PSK would permit.

A second consideration in the choice of modulation is the susceptibility of the system to noise, i.e., the BER performance characteristics. Figure 2-13 shows the relative performance characteristics for various modulation and detection methods. The characteristics of MSK are not shown, but are known to be only slightly worse (0.5 dB) than coherent PSK, provided coherent MSK detection is employed. From a noise performance standpoint, it is obvious that coherent PSK is superior, MSK is a close second, and FSK is inferior to PSK by 3 dB. A compromise must be made between best noise performance and minimum adjacent channel interference. PSK provides the best noise performance but the worst adjacent channel interference. MSK is slightly worse than PSK for noise performance, but superior for minimum interference. FSK is inferior in

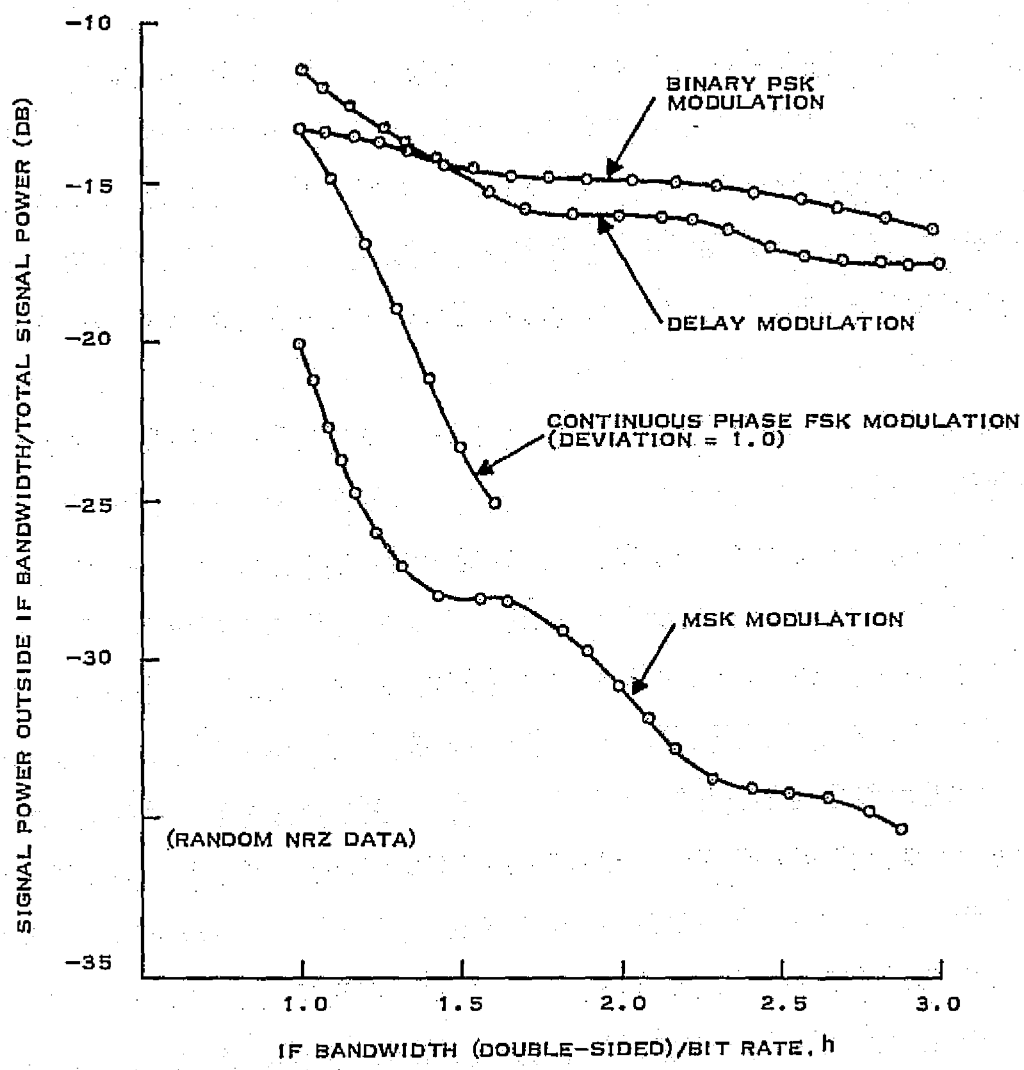

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Figure 2-11. Fraction of Total Modulated Signal Power Outside IF Bandwidth Versus IF Bandwidth

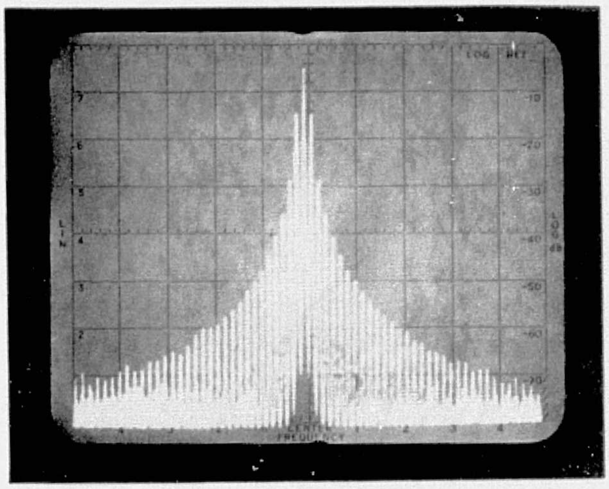
noise performance, but similar to MSK for adjacent channel interference. Therefore, it appears that MSK is the best compromise between the two performance criteria.

2.9.3 Baseband Data Encoding Techniques

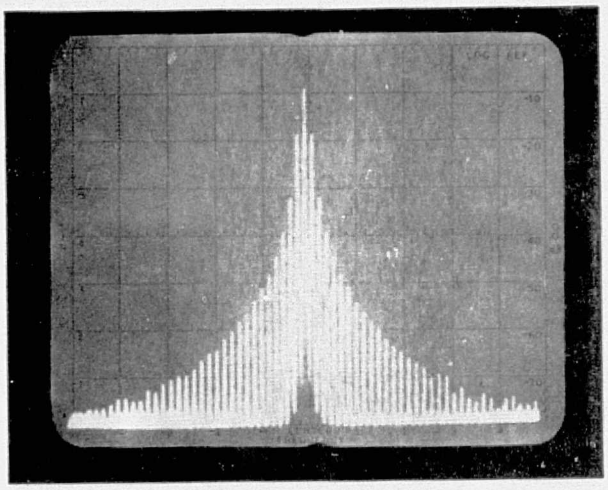
The choice of baseband data code is an important factor in determining channel bandwidth use. Figure 2-14 depicts the power spectral density of various codes. It is obvious that NRZ data requires the minimum channel bandwidth. However, it has a very high dc component for random data and can have infrequent transitions for some data patterns. The high dc content means that the phase displacement in MSK can be substantial, resulting in potentially low carrier content. The major problem in using NRZ data is the potential for long periods of time without a data transition. This lack of data transitions makes the problem of holding bit sync lock a major

320Hz SQUARE WAVE
MODULATION

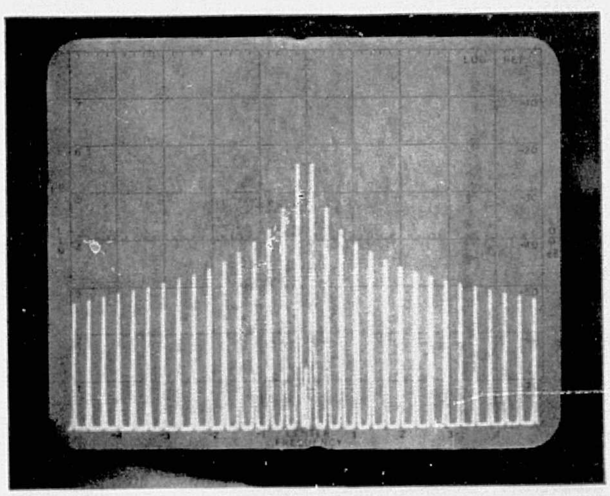
2KHz/Division-X Axis
10 dB/Division-Y Axis



MSK MODULATION
(AT 20 KHz-
DEVIATION = ±160 Hz)



CONTINUOUS PHASE FSK
(10.7 MHz-VCXO
DEVIATION = ± 160 Hz)



±90° PSK
(AT 10.7 MHz)

Figure 2-12. Comparison of Spectral Characteristics of MSK, Continuous Phase FSK, and PSK

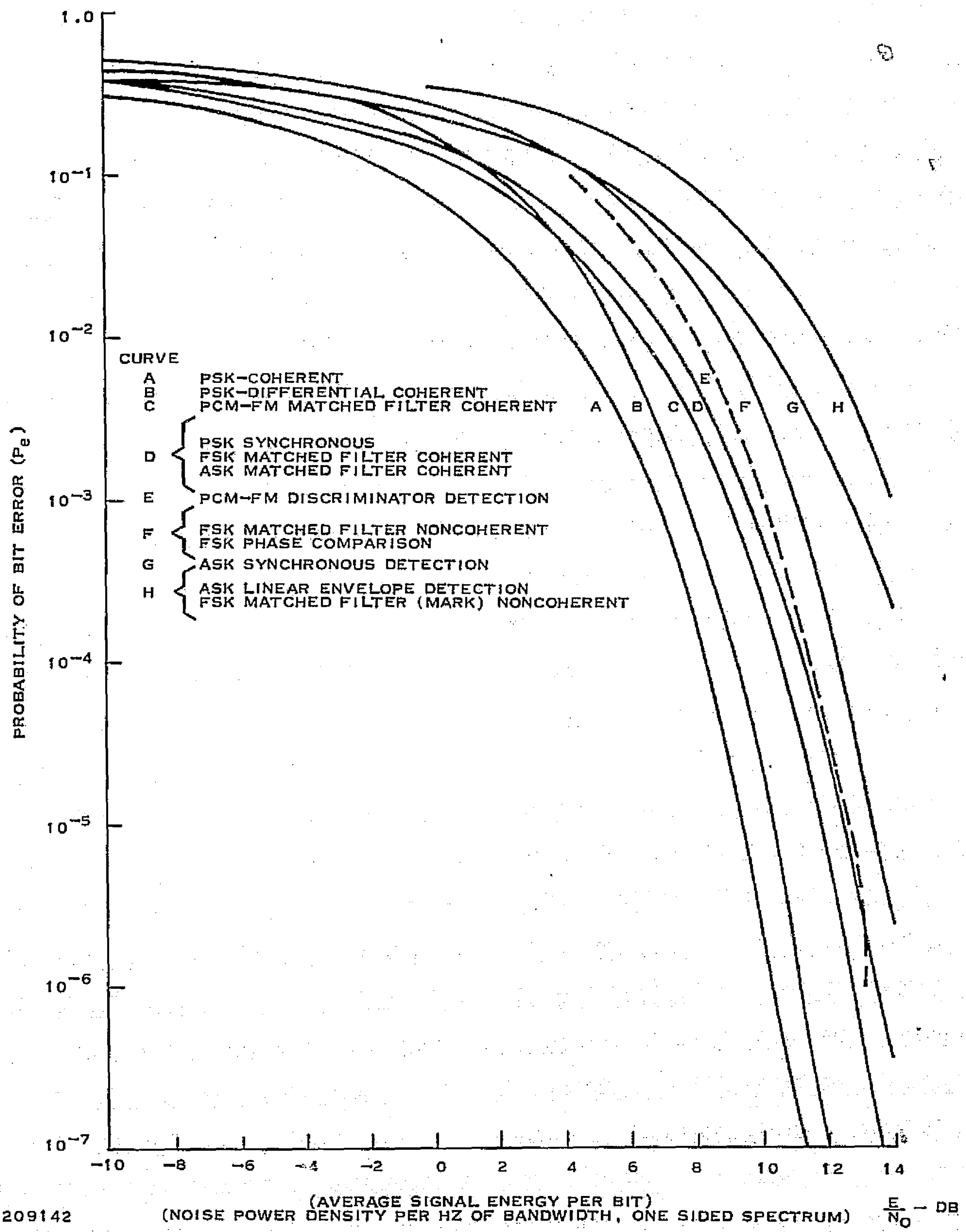


Figure 2-13. Performance Comparison of Various Modulation/Detection Techniques

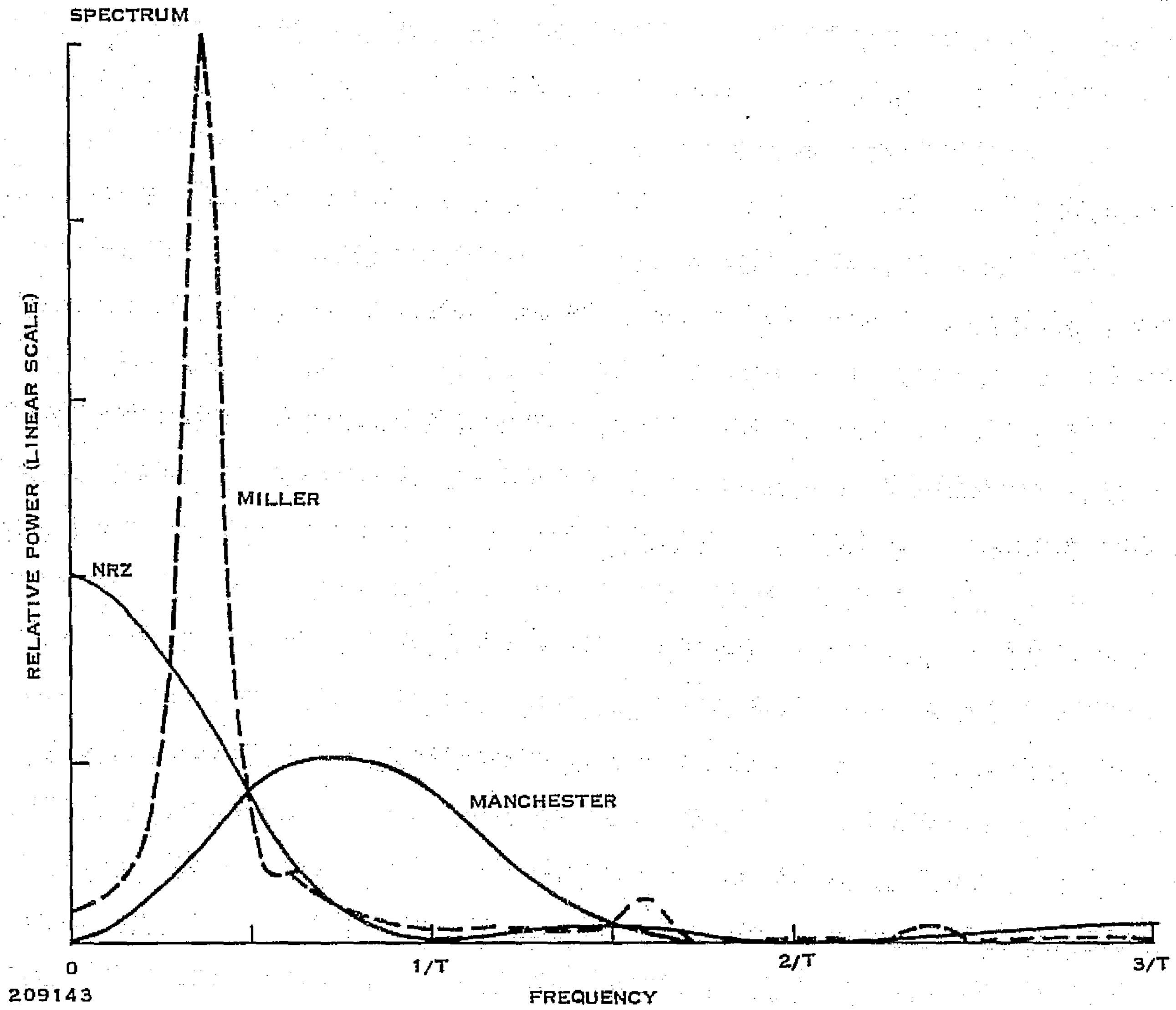


Figure 2-14. Comparison of Power Spectral Density of Various Codes

concern. In addition, if the hyperbolic ranging method is to be employed in a geostationary satellite, a continuous clock is desirable and basic NRZ data is not suitable.

A second popular encoding scheme is Manchester or biphase data. This data/clock code has no dc component and contains at least one transition in each bit time. However, the data bandwidth is about twice that of NRZ. This has the effect of increasing the amount of bandwidth required for a given data rate, i.e., it increases the value of k.

A third useful data code, even though not as popular as NRZ or Manchester, is the Miller or delay code. This code has a small dc component for random data, but not nearly as large as that for NRZ. It has a very dense power spectrum as shown in Figure 2-14. The bandwidth requirements are comparable to NRZ data, yet this code guarantees at least one transition every other bit time. Although the optimum data code appears to be Miller code, the recovery of bit sync clock from Miller encoded data is a more difficult task than recovering a clock from Manchester data. The generation of the Miller data is not a problem; Manchester data is generated and used as a clock input to a flip-flop, the output of which is Miller data. The use of the Miller code requires the reception of a unique 101 pattern before the proper clock phase can be established, and the bit sync code must include a 101 sequence before the frame sync code. Since 15 bit-times have been allocated to the bit sync code, the bit sync code pattern should become: 111111111010101. This does not end in a 111 . . . pattern for which the "optimum" frame sync code is optimized. However, since bit sync must be established before an attempt is made to decode frame sync, the requirement for the reception of two consecutive 101 patterns before bit sync lock is recognized will negate the bad effects of this code.

The major advantages of Manchester data are the relative simplicity with which the bit sync clock can be recovered and the fact that no special data pattern is required to establish clock phase. A minor advantage for the hyperbolic ranging system is the more numerous time marks guaranteed by Manchester data. However, because the use of Manchester data requires twice the bandwidth compared to Miller or NRZ data, the bandwidth use constant is doubled and the number of ELTs and EPIRBs serviced is halved.

One final comparison between NRZ, Miller, and Manchester data should be made. The major information-containing portion of NRZ data is concentrated near dc. When modulated with a carrier, the data sidebands will be located very close to the carrier, and the spectral purity of the generated carrier must be almost perfect. This can create a definite problem in designing a low-cost ELT or EPIRB transmitter. Both Miller and Manchester codes translate the major energy content of the information sidebands to a band away from the carrier. Manchester data has a rather broad spectrum centered at 3/4 of the clock frequency; Miller data has a narrow, high-density spectrum centered at 3/8 of the clock frequency. Therefore, the Manchester data sidebands and the Miller data sidebands will be located away from any carrier noise.

The final selection of data encoding method is deferred until some breadboard test data can be accumulated. At this point, Miller encoded data is preferable. However, the system impact of using Manchester data will be considered. The value of k for Manchester data will be assumed to be twice the value of k for Miller data.

By referring to Figure 2-11, it will be observed that for MSK, a value of $k = 1$ (for either NRZ or Miller data) ensures that out-of-band power will be at least 15 dB down with respect to the in-band power. This limits the dynamic range of the system to 15 dB. Since previous link analyses have shown an 18-dB variation in received signal strength, a 15-dB dynamic range is inadequate. Additionally, if a 10-dB loss contingency is to be considered, the required dynamic range is 28 dB. Therefore, the minimum value of k is 2.5 for either NRZ or Miller data; the minimum value of k is 5 for Manchester data. If only a 5-dB contingency factor is allowed (instead of 10 dB), the value required for k is only 1.5 for NRZ or Miller data and only 3 for Manchester data. Therefore, such a value of k will provide adequate dynamic range under normal conditions of signal strength for adjacent channel usage and will provide dynamic range equal to the limit of receiver noise only one channel removed. These values of k for the two data codes seem reasonable and the data rate can now be determined.

2.9.4 Data Rate Determinations Based on Channel Bandwidth

Both the receiver analyzing bandwidth and the channel bandwidth (double-sided) were previously determined to be 400 Hz. The value of k was determined to be 1.5 for Miller encoded data and MSK modulation and 3.0 for Manchester encoded data and MSK modulation. Since k is related to channel bandwidth, B, and data rate, r, by

$$B = kr$$

the maximum data rate is:

$$r = \frac{B}{k} = \frac{400 \text{ Hz}}{1.5 \text{ Hz/BPS}}$$

$$= 266 \text{ BPS for Miller data}$$

and

$$r = \frac{400}{3}$$

$$= 133 \text{ BPS for Manchester data}$$

By anticipating the desirability of using a binary data clock frequency, the two selected data rates are:

256 BPS for Miller data and MSK modulation

128 BPS for Manchester data and MSK modulation.

This concludes the determination of the channel bandwidth, modulation method, data encoding method, channel use constant, and data rate. The results are summarized in Table 2-12.

TABLE 2-12. COMPARISON OF MANCHESTER AND MILLER ENCODED DATA

Data Encoding Method	Channel Bandwidth (B)	Channel Use Constant (k)	Data Rate (r)	Relative Power Outside Channel (maximum)	Effective Noise Bandwidth
Miller data	400 Hz	1.5	256 BPS	–23 dB	≈ 200 Hz
Manchester data	400 Hz	3.0	128 BPS	–23 dB	128 Hz

2.10 SYSTEM NOISE MARGINS BASED ON DATA RATE, EFFECTIVE NOISE, BANDWIDTH, AND A SPECIFIED BER

It is now possible to evaluate the system noise margins related to a fixed BER for both the geostationary satellite system and the low-orbiting system. For these analyses, the following parameters are defined:

Minimum acceptable BER is 10^{-3}

MSK modulation is to be used and a coherent MSK demodulator will be employed

The data rate will be 128 BPS for Manchester data and 256 BPS for Miller data

Minimum signal-to-noise density is 49 dB-Hz for the low-polar-orbiting satellite and 40 dB-Hz for the geostationary satellite.

Since the performance of coherent MSK is similar to coherent PSK, curve A of Figure 2-13 indicates the minimum E/n_o required to ensure a BER of 10^{-3} is 7 dB. These data will be used to evaluate both satellite systems.

2.10.1 Geostationary Satellite System

The aforementioned minimum signal-to-noise density for this system is 40 dB-Hz. If optimum data detection is employed, Manchester data can be integrated for one bit time, τ. The minimum available energy-to-noise density ratio is:

$$E/n_o = 40 \text{ dB-Hz} - 10 \log \frac{1}{\tau}$$

$$= 40 \text{ dB-Hz} - 10 \log r$$

$$= 18.93 \text{ dB}$$

Since the minimum E/n_o required to maintain a BER at least as good as 10^{-3} is 7 dB, the margin is 11.9 dB. Some of this margin will be lost by suboptimum data detection. Part of the margin, however, will be available as a contingency factor for reduced ELT or EPIRB radiated power. Therefore, Manchester data transmitted at 128 BPS will provide adequate E/n_o margin for a geostationary satellite system.

As previously discussed, the use of Miller data can double the data rate and halve the channel use factor. This is important for increasing the number of ELTs and EPIRBs that can be serviced. It does decrease the E/n_o margin by 3 dB, however. It is also difficult to implement a near-ideal detector for Miller data. The maximum integration time (for a simple detector) is half a bit period. This further degrades the E/n_o margin by 3 dB [$\tau = (1/2 \cdot (1/256)$]. The minimum available E/n_o for Miller data is, therefore:

$$E/n_o = 40 \text{ dB-Hz} - 10 \log \frac{1}{\tau}$$

$$= 12.93 \text{ dB}$$

The maximum margin for Miller data is:

$$12.93 \text{ dB} - 7 \text{ dB} = 5.93 \text{ dB}$$

Part of this margin will be lost in the suboptimal detecting process. The actual final margin for contingencies will be less, and every effort should be made to improve the signal detection process for the goestationary satellite if Miller data is used. Comparison of E/n_o margins (maximum) for Manchester and Miller data in a geostationary satellite system is summarized as follows.

Manchester	11.9 dB E/n_o margin, maximum
Miller	5.9 dB E/n_o margin, maximum

2.10.2 Low-Polar-Orbiting System

The worst case signal-to-noise density for this system was found to be 49 dB (Table 2-9). Since this is 9 dB better than the geostationary case, the margin is correspondingly increased for both Miller and Manchester data. It can be safely stated that an adequate noise margin exists for the low-polar-orbiting satellite system. The margins are:

Manchester	20.9 dB E/n_o margin, maximum
Miller	14.9 dB E/n_o margin, maximum

With such large margins, the ELT and EPIRB power requirements can be reduced, provided the units do not have to operate with a geostationary satellite. If the power output is maintained, a large margin is available to allow for any contingencies such as low power, broken antenna, poor satellite visibility, etc. The low-polar-orbiting satellite system should not have a noise problem, but caution must be exercised in the geostationary satellite system design.

2.11 POSITION ERROR ANALYSES

Major sources contributing to positioning inaccuracies include ELT/EPIRB oscillator instabilities, system noise, spacecraft oscillator instabilities, spacecraft hardware/measurement limitations, and spacecraft ephemeris inaccuracies. Because system measurement errors tend to be constant while the measured position resolution varies with respect to satellite-ELT relative longitude, the magnitude of the error is also a function of satellite position. Hence, system measurement errors tend to produce worse position errors as the doppler lines of position for a given set of measurements tend toward being parallel. The extreme case occurs when the ELT is located on the satellite subtrack from where the position error increases without bound for even a minute measurement error. Therefore, when specifying allowable error, the minimum off-track ELT displacement must also be specified.

Other error sources include transmission path induced errors, interference, and upper atmosphere anomalies. These errors can be minimized by making more than the minimum three doppler measurements. In addition, these errors tend to be injected fortuitously and should be accounted for statistically.

2.11.1 System Errors

System errors are defined as those arising from sources of measurement error, i.e., those errors induced by imperfections in the ELT/EPIRB frequency and/or the satellite frequency measurement technique. The magnitude of the instantaneous position error is a function of the satellite-ELT/EPIRB geometry at the time of the measurements as well as the magnitude of the measurement error. This definition is valid only for a system defined as an ELT/EPIRB and the satellite receiver-processor. It excludes errors induced by imperfect determination of satellite position and time with respect to earth-referenced coordinates. In addition, the errors caused by variation of transmission path are considered separately in Subsection 2.11.3.

The parameters that affect system error can only be identified by considering the methodology of determining position. For the differential doppler technique, the key measurement parameter is the rate of change in the doppler shift of the received signal. The advantage of this technique is that the exact frequency of the ELT carrier does not need to be known. This is illustrated by the following equations:

$$f_r = (f_o + \Delta f)(1 + v/c)$$
$$= f_o + \Delta f + f_o (v/c) + \Delta f(v/c)$$

where

f_r = the received frequency

f_o = the nominal transmitter frequency

Δf = the unknown frequency offset of the transmitter

v = the relative velocity component along the transmission path

c = speed of light.

The true doppler shift is given by:

$$f_d = f_o (v/c) + \Delta f (v/c)$$

However, since $\Delta f \ll f_o$ ($\Delta f/f_o \cong 10^{-5}$), the product $\Delta f(v/c)$ can be assumed to be zero with negligible error in f_d. The measured doppler shift, however, is

$$f_d' = (f_r - f_o) = f_o(v/c) + \Delta f$$

and the true doppler shift differs from the measured shift by Δf. The magnitude of Δf is approximately the same as the magnitude of the actual doppler shift and, hence, is intolerable.

If Δf is assumed to be constant, this source of error can be eliminated by taking the derivative (thus the term "differential doppler") of the shift:

$$\frac{d(f_r - f_o)}{dt} = \frac{d}{dt}\left(\frac{f_o v}{c}\right) + \frac{d}{dt}(\Delta f)$$
$$= \left(\frac{f_o}{c}\right)\frac{dv}{dt}$$
$$= \left(\frac{f_o}{c}\right) a$$

A plot of the rate of change in doppler shift as a function of ELT position relative to satellite position for the one-dimensional case where the ELT is on the satellite subtrack is shown in Figure 2-10. It is obvious that a measurement of doppler acceleration can be used to determine position regardless of any steady-state frequency error existing. Use of this direct method causes problems.

The first problem is one of hardware implementation. One approach (Figure 2-15) is the use of a phase-lock loop. The doppler rate ($\ddot{\theta}$) output is direcred to an A/D converter. The dynamic range of this converter must yield accuracy comparable to the ±1 Hz/40 s = ±0.025 Hz/s accuracy of the integral-hertz measurement system as used on RAMS. The full-scale range must be approximately 100 Hz/s. Therefore, the dynamic range is 100/0.025 = 4,000, or 12 bits. Monolithic 12-bit A/D converters capable of 25-μs conversion times exist, and are not a

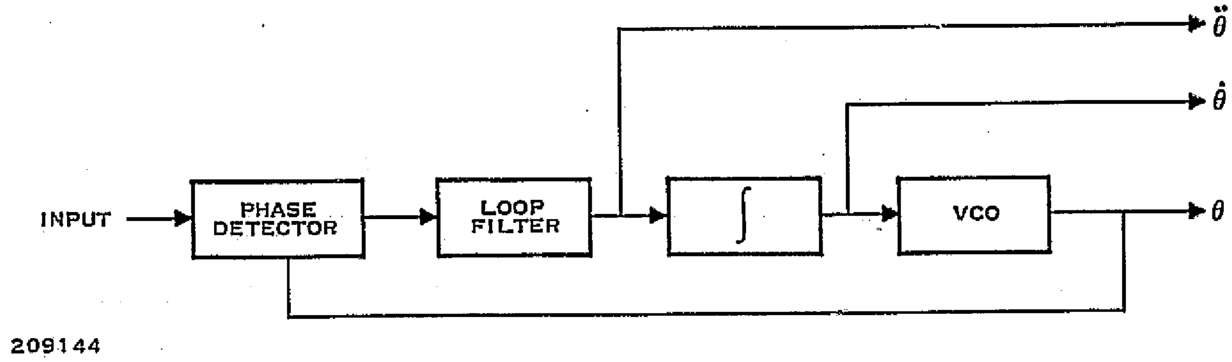


Figure 2-15. Phase-Lock Loop Approach to Measuring Doppler Rate

limitation. However, there are inherent noise problems with systems designed for such accuracies. For example, if 100 Hz/s corresponds to a 10-volt input to the A/D, the maximum noise level cannot exceed ±1 mV to preserve the resolution. In addition, the absolute error in the phase-lock loop design must not produce an error exceeding ±1 mV at this point. Finally, the overall loop design must provide rapid acquisition and fast settling. These are difficult requirements. An abbreviated analysis is appropriate.

The phase-lock loop shown in Figure 2-15 is essentially a third-order loop whose block transfer functions are:

k_d = phase detector gain constant, volts/radian

$$F(s) = \frac{(s\tau_2+1)^2}{s\tau_1} = \text{loop filter transfer function}$$

$$\frac{1}{s\tau_1} = \text{intergrator transfer function}$$

k_o = VCO gain constant, radians/sec/volt

For overall loop performance, the integrator is assumed to be part of the loop filter such that:

$$F(s) = \frac{(s\tau_2 + 1)^2}{(s\tau_1)^2}$$

The closed-loop transfer function is:

$$H(s) = \frac{k_o k_d F(s)}{s + k_o k_d \; F(s)}$$

$$= \frac{k_o k_d \; (s\tau_2 + 1)^2/\tau_1^2}{s^3 + s^2 \; k_o k_d \; (\tau_2/\tau_1)^2 + s \; 2k_o k_d \tau_2/\tau_1^2 + k_o k_d/\tau_1^2}$$

Typical values for k_o and k_d are:

$$k_d = 1 \text{ volt/rad}$$

$$k_o = 100\,\pi \text{ rad/s/volt (assuming a VCXO)}$$

It is desired to make the maximum value of $\ddot{\theta}$ (100 Hz/s or 200 π rad/s^2) correspond to a voltage of 1 volt at the $\ddot{\theta}$ output. Hence,

$$\ddot{\theta} = \frac{k_o\, V(\ddot{\theta})}{\tau_1}$$

$$\tau_1 = \frac{k_o\, V(\ddot{\theta})}{\ddot{\theta}}$$

$$= \frac{(100\,\pi)(1)}{200\,\pi}$$

$$= 0.5$$

The value of τ_2 is the only undefined parameter. Its value is determined from loop stability and damping considerations, i.e.,

$$3\sqrt[3]{\frac{\tau_1^2}{4k_o k_d}} \geqslant \tau_2 \geqslant \sqrt[3]{\frac{\tau_1^2}{2k_o k_d}}$$

$$0.175 \geqslant \tau_2 \geqslant 0.0736$$

For this analysis, $\tau_2 = 0.1$ is assumed. The loop bandwidth is

$$\omega_3 = \sqrt[3]{\frac{k_o k_d}{\tau_1^2}}$$

$$= 10.8 \text{ rad/s or} \approx 2 \text{ Hz}$$

This is a narrow bandwidth and may not provide adequate settling time.

To summarize, the third-order loop shown in Figure 2-14 has the following characteristics:

k_d = 1 volt/rad

k_o = 100 π rad/s/volt (50 Hz/volt)

τ_1 = 0.5

τ_2 = 0.1

ω_3 = 10.8 rad/s (2 Hz)

$v(\ddot{\theta})$ = 10 mV/Hz/s

To determine the amount of noise present at the $\ddot{\theta}$ output with the minimum expected noise/power density ratio of −50 dB/Hz, it is assumed that a two-pole filter which cancels the two zeros of F′(s) is inserted in series with $\ddot{\theta}$ output. This transfer function is:

$$G(s) = \frac{\ddot{\theta}(s)}{\theta_i(s)} = \frac{s}{s + k_o k_d \; F(s)} \times \frac{F'(s)}{(s\tau_a + 1)^2}$$

$$= \frac{1/\tau_1}{s + k_o k_d F(s)}$$

The equivalent noise bandwidth is:

$$B_L = \int_0^{\infty} |G(j\omega)|^2 \, df \text{ (Hz)}$$

$$= 0.5 \text{ Hz}$$

The phase spectral density for the low orbiter is:

$$\Phi = \frac{\omega_i}{P_s} = 10^{-5} \text{ rad}^2/\text{Hz (worst case)}$$

The rms noise voltage at the output $\ddot{\theta}$ is:

$$v_{n\ddot{\theta}} = k_d \sqrt{\Phi \, B_L}$$

$$= 2.3 \text{ mV rms}$$

When compared with the minimum desired measurement voltage of 10 mV corresponding to 1 Hz/s, this induces a large measurement error. The only way to reduce this error is to reduce the noise bandwidth. The noise bandwidth for this circuit is given by:

$$B_L = \frac{\tau_2}{2\tau_1^2 \left(2\, k_o k_d \; \tau_2^3 - \tau_1^2\right)}$$

The bandwidth can be reduced by increasing τ_2. For $\tau_2 = 0.15$ instead of 0.1, the noise bandwidth becomes $B_L = 0.16$ Hz instead of 0.5 Hz and the rms noise can be reduced to approximately 1 mV. This corresponds to an rms error of 0.1 Hz/s or 6 Hz/min, which is greater than the expected drift rate of the ELT oscillator and may not be tolerable. In addition, this error level does not consider any contingency factor that might reduce the signal-to-noise density by 10 dB. Such a degradation would increase the error to approximately 20 Hz/min or 0.3 Hz/s. Hence, the indirect method of obtaining doppler rate remains preferable. The indirect method measures the received frequency at two points in time and estimates the rate of change over the period between measurements.

The second problem is the ambiguity associated with the use of differential doppler. This problem is inherent in differential doppler since there are two values of the independent variable (position) that produce the same rate of change in doppler frequency (Figure 2-10). This

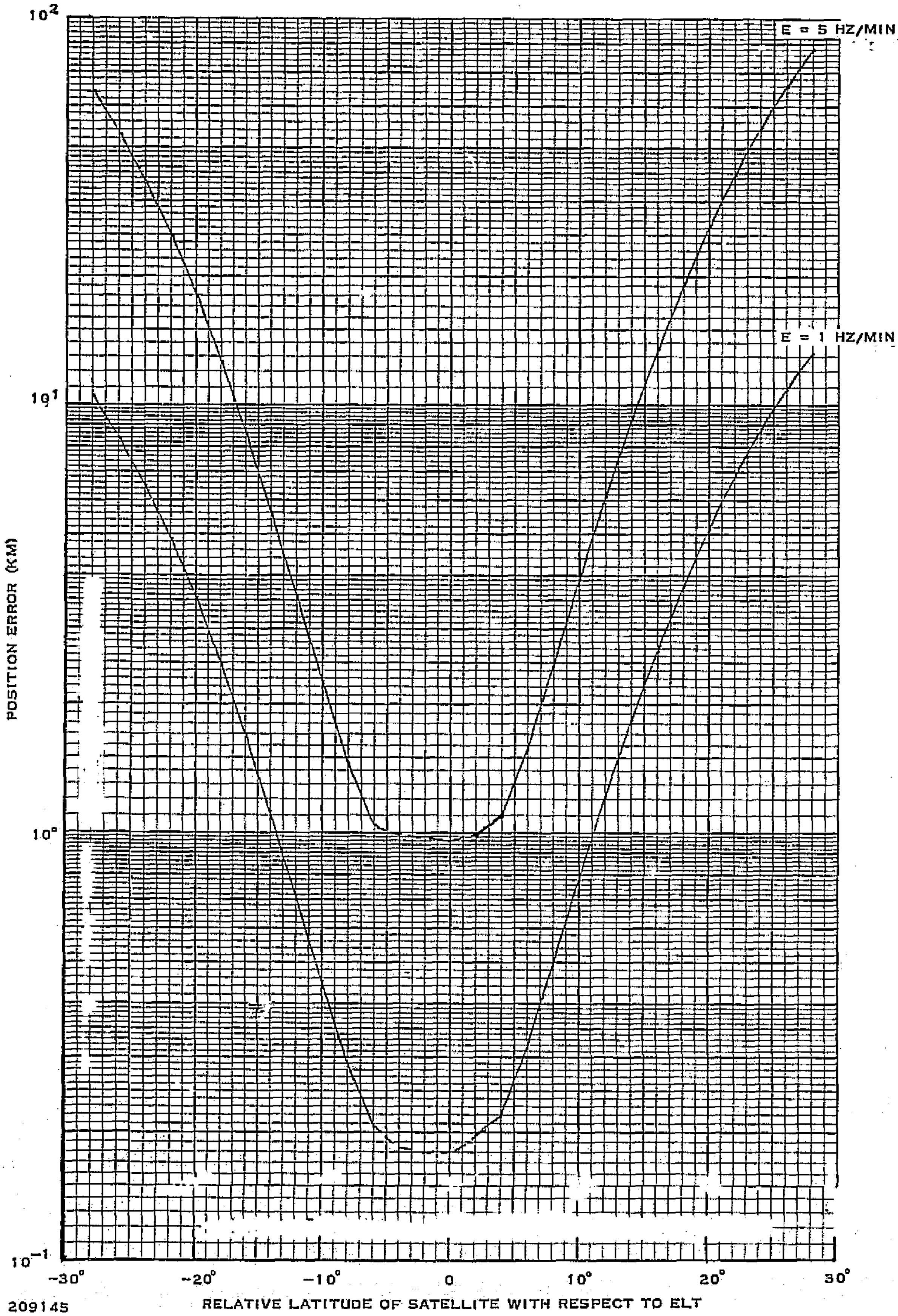


Figure 2-16. Position Error as a Function of Satellite Position Relative to ELT Using Two Consecutive Transmissions (N = 1)

ambiguity can be resolved by making two such measurements on the same side of the curve. (Three measurements may be required.) The side of the curve is determined from whether the magnitude increased or decreased.

The next step in the error analysis is to evaluate the geographic error as a function of measurement error. Once again, only the one-dimensional case will be considered. (The two-dimensional error includes not only measurement error but the effects of the nonorthogonal intersection of the lines of position.)

The one-dimensional error analysis used a computer program to calculate the doppler shift as a function of geographic angle (relative latitude) between the satellite and an ELT located on the subtrack. The frequency difference for two measurements determined the true doppler shift. A known doppler rate error was introduced and the "measured" geographic angle was computed using an iterative technique. The absolute value of the difference between the actual angle and the measured angle was used to compute surface error in kilometers. This surface error was plotted as a function of the relative latitude of the satellite and ELT.

The program was used to compute and plot position error as a function of relative latitude for an error rate of 5 Hz/min and 1 Hz/min, assuming that two consecutive transmissions would be used. The results are shown in Figure 2-16. A similar plot was made for an error rate of 5 Hz/min and for $N = 1$ and $N = 5$. ($N = 1$ is for using the first transmission following the initial transmission; $N = 5$ is for the fifth transmission following the initial transmission.) These results are shown in Figure 2-17. This figure indicates that the mean measurement error is not greatly dependent on the time between frequency measurements although there is some skew in the error. Figure 2-16 indicates that the doppler rate error cannot exceed 1 Hz/min if the position error is to be maintained within ±5 km over the full field of view of the satellite. With a total error rate of 5 Hz/min, the expected position error can be as great as ±25 km. These errors will be increased somewhat for the two-dimensional case; in particular, the error becomes infinite when the ELT is on the subtrack because of the distortion caused by the nonorthogonal intersection of the lines of position.

There are two primary sources of doppler rate error. The first is the drift rate of the ELT oscillator. The second major source of error is in the measurement of doppler rate. The first source is discussed in Subsection 2.11.2, and is expected to be on the order of 1 Hz/min. The second source depends on the measurement technique. If the integral-hertz method of frequency measurement is used, then the frequency error between measurements can be expected to be ±1 Hz, which corresponds to a rate error of ±1.5 Hz/min. If the direct method of doppler rate measurement is used, the expected error can be as high as 6 Hz/min. Hence, if the integral-hertz frequency measurement approach is used, the total rate error may be 2.6 Hz/min, which corresponds to a maximum position error of approximately ±10 km. If the direct measurement of doppler rate is used, the error can be expected to be as much as ±30 km under nominal worst case signal-to-noise conditions.

The estimated error range and error budget can be summarized as follows:

Integral-hertz measurement

ELT oscillator stability	1 Hz/min
Measurement accuracy	1.5 Hz/min
Location accuracy	1 to 10 km

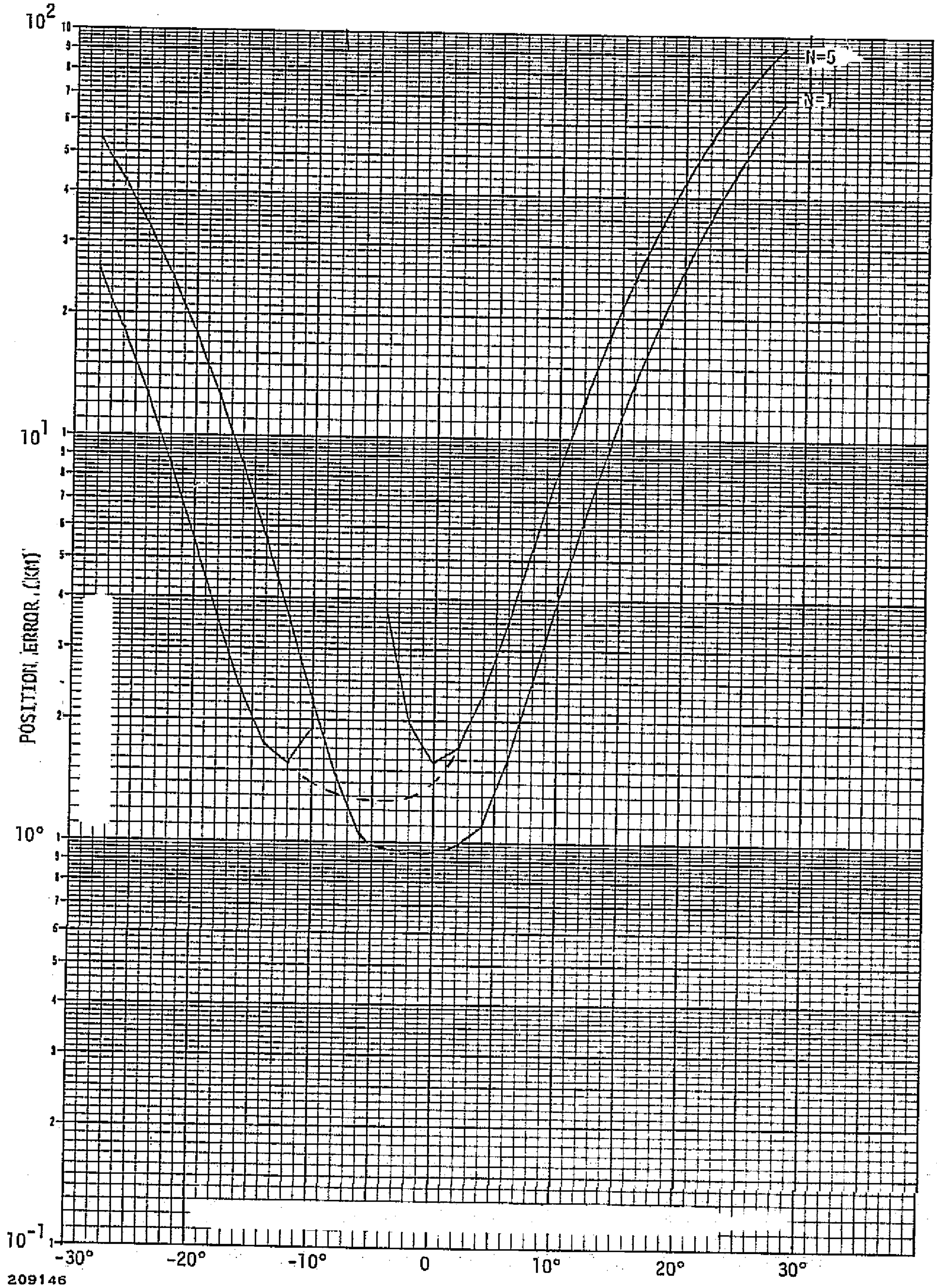


Figure 2-17. Position Error as a Function of Satellite Position Relative to ELT for a Constant Error Rate of 5 Hz/Min

Direct doppler-rate measurement

ELT oscillator stability	1 Hz/min
Measurement accuracy	6 Hz/min
Location accuracy	1 to 30 km

Therefore it seems the correct primary measurement choice would be the indirect method. The ELT oscillator stability will then be the primary accuracy limitation. The design goal for oscillator stability should be 1 Hz/min or less if ±5-km accuracy is desired.

2.11.2 ELT/EPIRB Errors

The effects of the ELT/EPIRB oscillator drift on system error have already been discussed, and details of the oscillator are discussed in Section 3. At this point, it is appropriate to consider some of the oscillator error sources.

The oscillator frequency error can be broken into three categories:

- Long-term drift (±0.003 percent of f_o over days)
- Medium-term drift (±1 Hz/minute over minutes)
- Short-term (primarily oscillator phase noise).

The use of differential doppler essentially eliminates concern for the long-term drift. Medium-term drift and its effects on position accuracy have been discussed previously. The short-term stability is now considered.

Previous analyses have shown that the indirect method of differential doppler measurement requires that that frequency error rate should not exceed 1 Hz/min. This implies that the frequency measurement accuracy should be on the order of 1 Hz, since the frequency measurements are made at approximately 1-minute intervals. Because the carrier frequency is 406.05 MHz, the fractional frequency deviation must be on the order of 1 Hz/4.06 $\times 10^8$ or approximately 10^{-9} over the measurement interval. The measurement interval will be between 100 ms and 1 second. Therefore, the short-term stability of the ELT oscillator should be specified as follows:

$$\langle \sigma\, \Delta f/f\, (2, T, \tau)\rangle \leqslant 2 \times 10^{-9},\ T = 1\ \text{s},\ \tau = 100\ \text{ms}$$

where

$\sigma\, \Delta f/f$ = the fractional frequency deviation

T = interval between measurements

τ = measurement interval.

This noise level, although not easily obtainable, can be realized with a well-designed low-noise, high-level crystal oscillator. The noise level is intrinsically dependent on the circuit design as well as the crystal.

If the direct approach of doppler rate measurement is to be implemented, then a more meaningful short-term frequency stability specification would be in terms of single-sideband-to-carrier (SSB/C) phase noise power ratio. Previous calculations have shown that the effective phase spectral density is 10^{-5} rad^2/Hz. The total phase noise-to-carrier density should be less than -50 dB over the expected bandwidth of the phase-lock loop. The previously discussed third-order loop bandwidth was found to be approximately 2 Hz. The phase noise will be attenuated 60 dB below 0.2 Hz. (This implies that the phase-lock loop VCXO must also have good noise characteristics above 0.2 Hz.) Therefore, the SSB/C phase noise density should be $\leqslant -60$ dB for $0.2 \leqslant f_m \leqslant 10$ Hz where f_m is the frequency offset from the carrier.

The ELT/EPIRB oscillator characteristics are summarized as follows:

Accuracy f_o ± 0.003 percent

Drift $\leqslant 1$ Hz/min

Fractional frequency deviation $\leqslant 2 \times 10^{-9}$, $\tau = 100$ ms

SSB/C phase noise $\leqslant -60$ dB, 0.2 Hz $\leqslant f_m \leqslant 10$ Hz.

2.11.3 Transmission Path Errors

The errors considered here may be produced in the system independent of the equipment. They are important in determining system designs that reduce any adverse effects they might cause. The two primary error sources are reflection and refraction.

The reflection of ELT/EPIRB signals can be caused by numerous terrain features, in particular, water or metalic objects. The magnitude of these reflected signals varies considerably with respect to location of an ELT/EPIRB. The location of the peak magnitude varies considerably along the satellite's path, and the reflected signal will also undergo a different doppler shift that will normally move it outside the bandwidth of the frequency measurement loop. Erroneous measurements would be made only in the improbable event that both the amplitude and the frequency of the reflected signal equaled those of the primary signal. Even in this case, the error would only be on the order of the loop bandwidth or 2 Hz. Therefore, multipath reflections are not considered a problem; however, multipath can possibly result in two simultaneous position measurements.

Refractions are caused by the variation in the speed of light in the atmosphere. These refractions essentially alter the apparent arrival angle of the received signal, resulting in an alteration in the perceived doppler shift, hence introducing some location error. The effect is analogous to the position distortion of objects viewed below water. However, because the angle of refraction at this frequency is small, the amount of position error created is also small; thus refraction is not considered a problem for position location in this system.

2.12 MULTIPLE ACCESS/PROBABILITY OF SUCCESS

This subsection brings all previously discussed parameters to focus in a single system metric, the probability of successfully detecting and locating an ELT or EPIRB within the maximum waiting period. The key elements that determine the overall probability of successfully receiving a given message are:

Mutual interference

Signal detection by the search unit

Availability of a receive channel

Probability of recognizing frame sync

Probability of receiving good data.

The probability of mutual interference was discussed in Subsection 2.9, but only to the extent necessary to compare the relative probability of successfully receiving m messages in n trials as a function of message repetition rate. Now that the major parameters affecting mutual interference have been more completely defined, mutual interference as a function of the number of ELTs and EPIRBs being serviced can be evaluated for various repetition rates and k factors. The equation for mutual interference is:

$$P_I = 1 - [1 - 4kn/TF]^{N-1}$$

where

P_I = probability of mutual interference

N = number of ELTs and EPIRBs requesting service

T = period between burst transmissions

F = total search spectrum in which signals may exist

k = channel use factor

n = number of bit times each message lasts.

The range of N was determined to be 0 to 500. The range of T was limited to 40 to 80 seconds. The value of F was found to be 86 kHz. The channel use constant was found to be 1.5 if Miller data is employed and to be 3.0 if Manchester data is employed. If a 100-ms unmodulated carrier preamble is assumed, the effective number of message bits is 128. A plot of the probability of mutual interference as a function of N is shown in Figure 2-18 for four values of T. A similar plot is shown in Figure 2-19 for T = 40 seconds and for k = 1.5 and 3.0.

The probability of the search unit detecting a signal is an important consideration and will now be evaluated. The principal parameters to be determined are the probability of false alarm, the probability of detection, and the minimum carrier-to-noise (C/N) ratio to establish these limits. A family of curves showing these parameters is depicted in Figure 2-20. Since previous calculations have established the probability of failure to detect frame sync or receive a message as being on the order of 10^{-6}, a consistent value for the false alarm rate would be 10^{-6}. Thus, selecting the 10^{-6} false alarm rate curve, the minimum peak signal-to-noise ratio is 19 dB for a probability of detection of 0.9999. This corresponds to a minimum E/n_o value of 16 dB. If the analyzing bandwidth is 400 Hz and the signal-to-noise density is 49 dB-Hz for the low orbiter or 40 dB-Hz for the geostationary satellite, the maximum available E/n_o is:

$$\begin{aligned} E/n_o &= 49 \text{ dB-Hz} - 10 \log 400 \\ &= 23.0 \text{ dB} \quad \text{(low orbiter)} \end{aligned}$$

$$\begin{aligned} E/n_o &= 40 \text{ dB-Hz} - 10 \log 400 \\ &= 14.0 \text{ dB} \quad \text{(geostationary)} \end{aligned}$$

Therefore, a 7-dB margin exists for the low-orbiting system, but the geostationary satellite system will not be able to maintain the desired performance. If the threshold is set to achieve the false alarm rate of 10^{-6}, the probability of detection will only be 0.98 and some means of

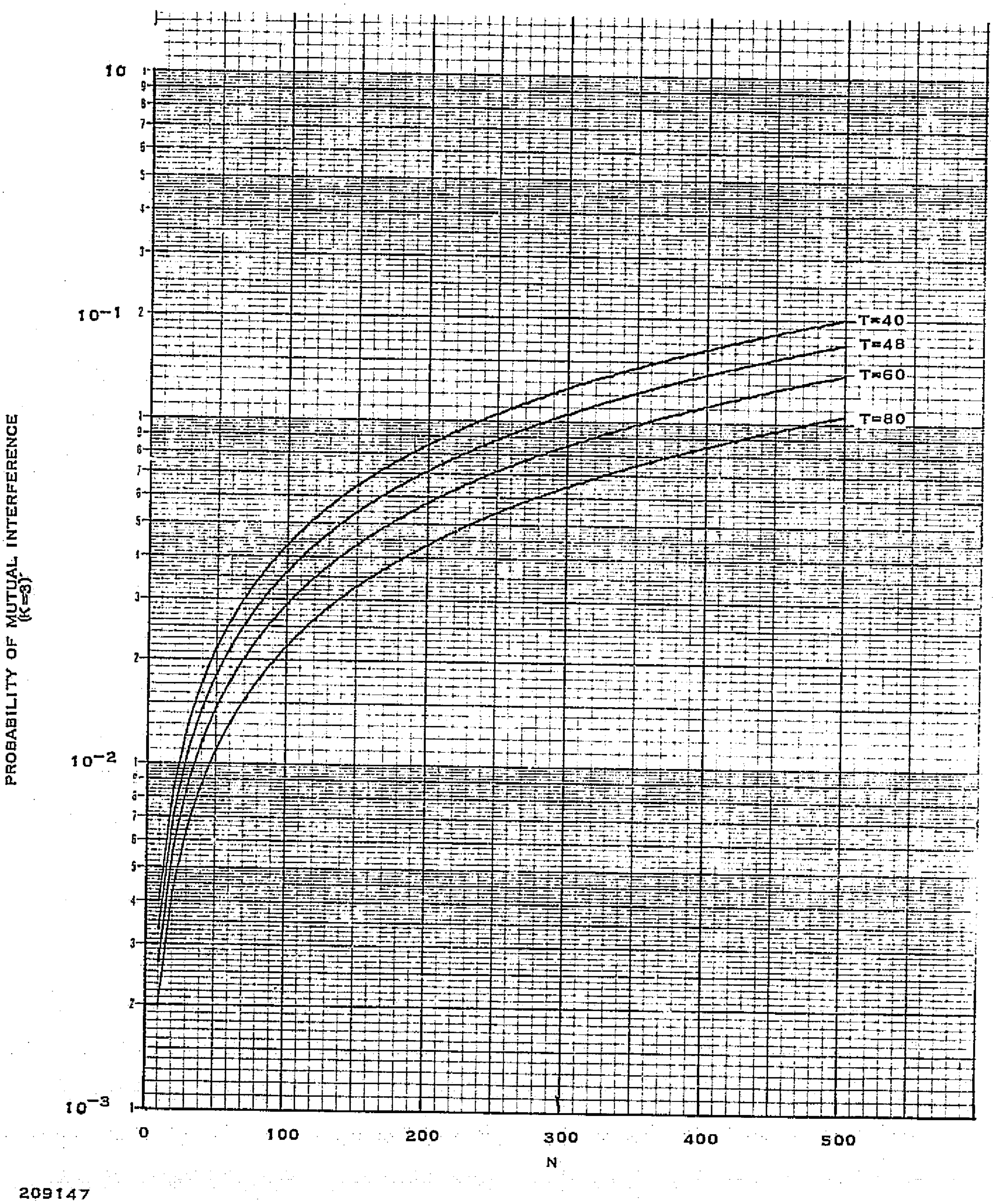


Figure 2-18. Probability of Mutual Interference, k = 3

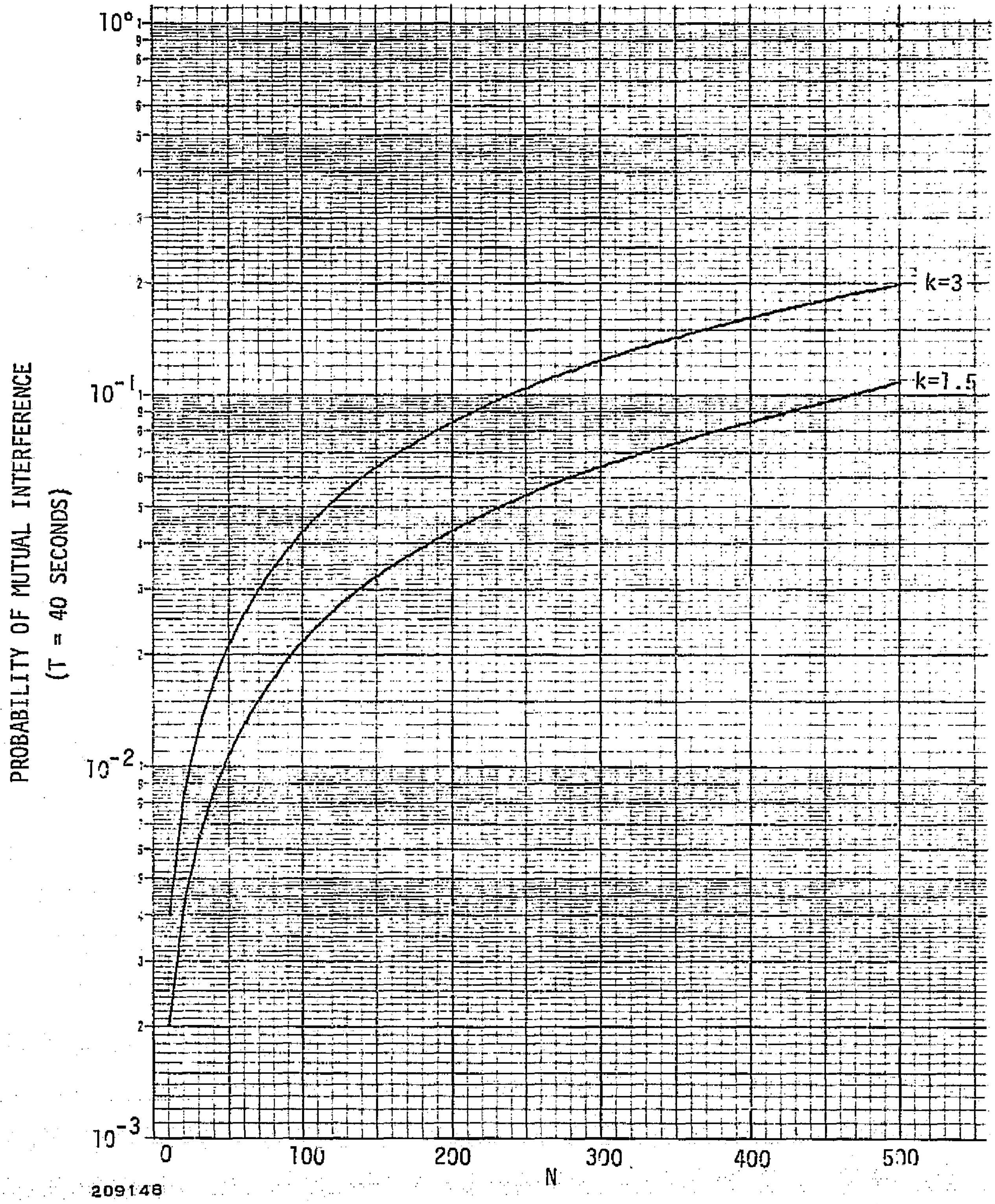


Figure 2-19. Probability of Mutual Interference, T = 40 Seconds

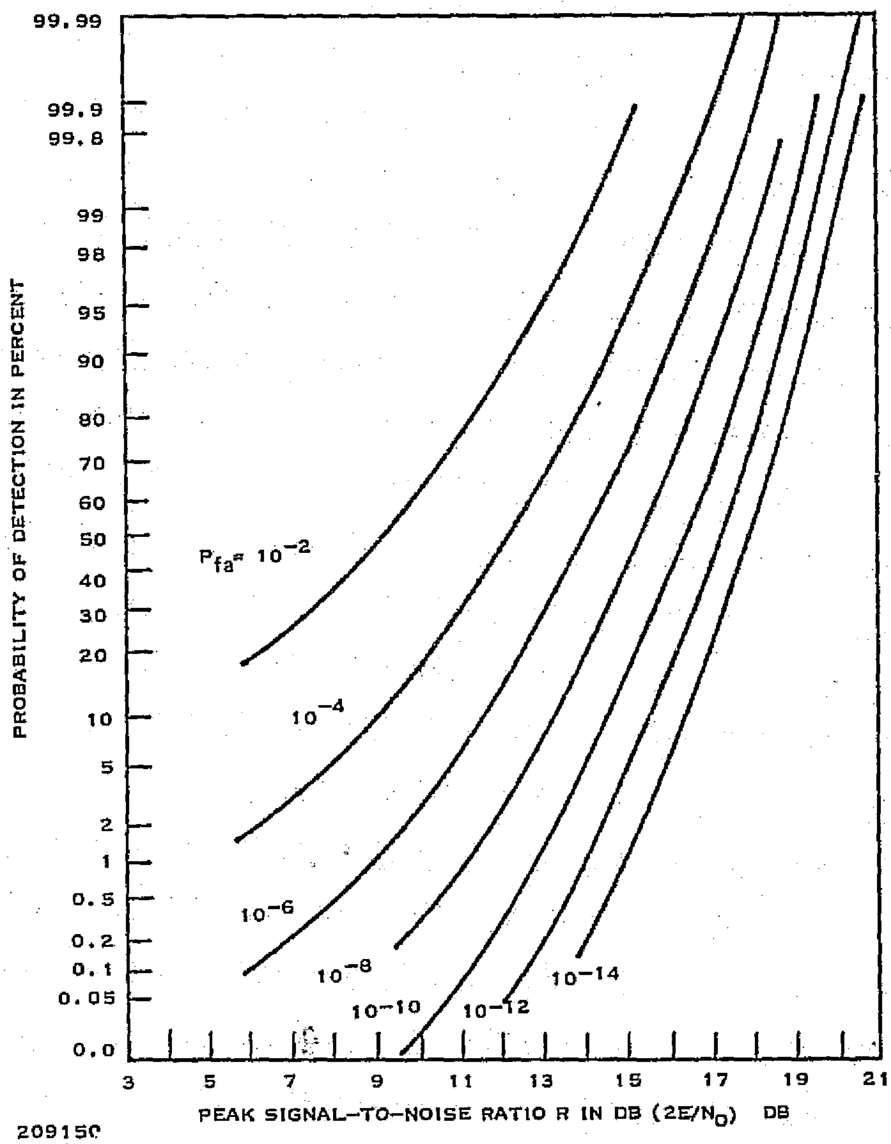


Figure 2-20. Detection Characteristics for Signal of Unknown Phase

signal enhancement will be needed in the geostationary satellite system to achieve a 0.9999 probability of detection. For either system to achieve a comparable margin with respect to the data detection margin, a 14-dB enhancement is needed. Such an enhancement can be achieved by averaging several frequency measurements during the 100-ms unmodulated carrier preamble. The minimum value for probability of detection will be 0.9999.

A second parameter not yet determined is the probability that a receive channel is available. This is solely a function of the number of frequency-agile receive channels available and the number of signals requesting service. The fundamental equation is the cumulative binomial distribution equation:

$$P_L = \sum_{x=0}^{L} \binom{n}{x} p^x (1-p)^{n-x}$$

where

P_L = probability that a channel is available

L = number of frequency-agile channels

n = number of ELTs/EPIRBs requesting service (corrected for mutual interference)

p = probability that a transmission occurs within a given time slot, assuming uniform time distribution.

From the previous parameter considerations, if N = 400, T = 60 seconds, and k = 3 are assumed, then the probability that a signal is not interfered with is 0.8877. The number of signals requesting service is:

$$\begin{aligned} n &= 0.8877\ N \\ &= 0.8877\ (400) \\ &= 355 \end{aligned}$$

If the number of channels available is 12, then:

$$P_L = \sum_{x=0}^{12} \binom{355}{x} \left(\frac{1}{60}\right)^x \left(1 - \frac{1}{60}\right)^{355-x}$$

$$= 0.9926$$

This number gives the probability of a signal being serviced by a system having 12 channels, but gives no insight into the adequacy (or inadequacy) of 12 loops. This is best depicted by calculating the probability of success as a function of the number of loops available and the number of ELTs and EPIRBs requesting service. Before performing this calculation, the results presented to this point will be summarized.

The probability of successfully receiving a message is given by:

$$P_s = (1 - P_I)\ P_d P_L P_{fs} P_{DG}$$

where

P_s = probability of successfully receiving a message

P_I = probability of mutual interference

P_d = probability of signal detection by search unit

P_L = probability of having an available channel

P_{fs} = probability of correctly recognizing frame sync

P_{DG} = probability that the data is good.

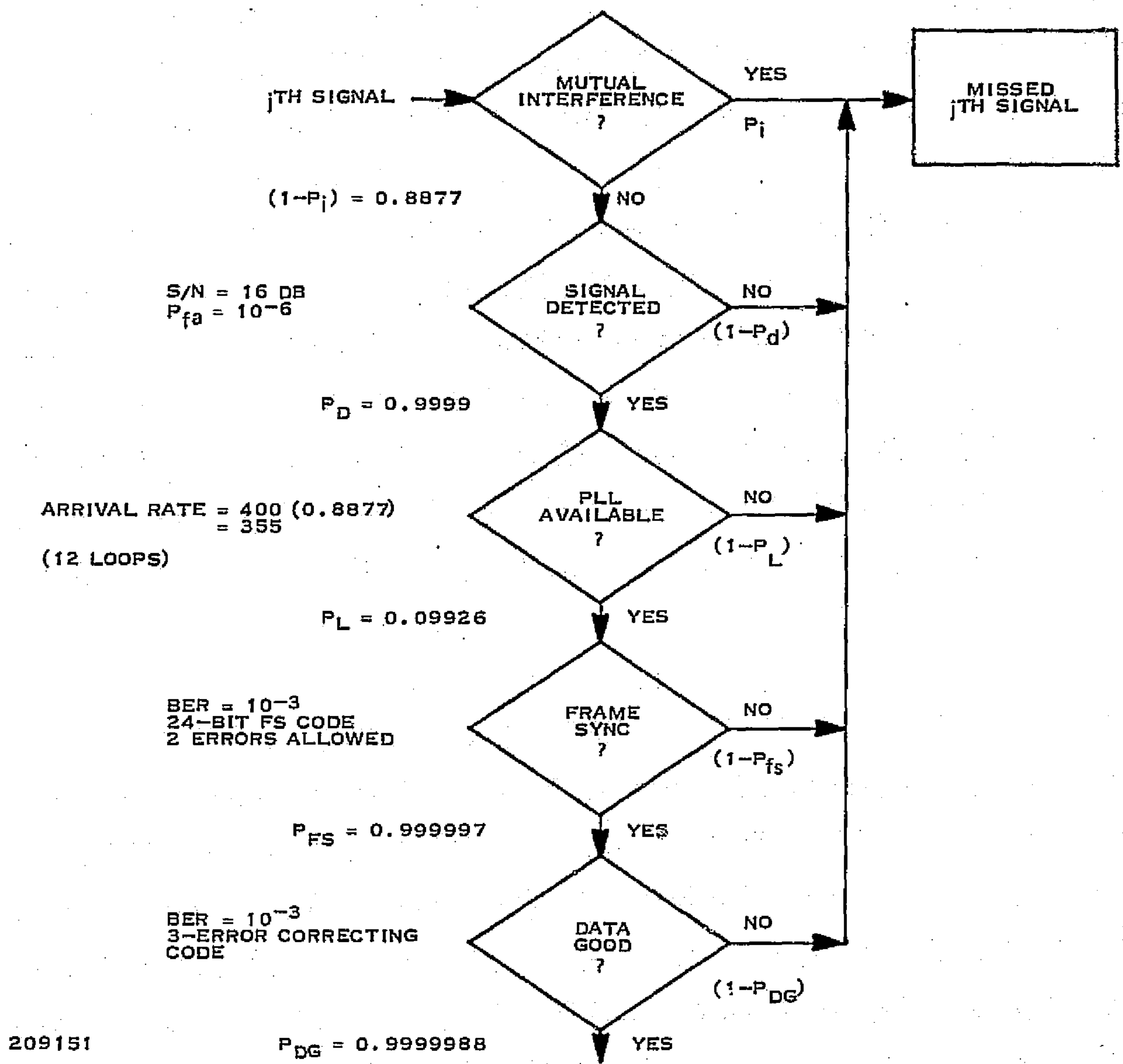


Figure 2-21. Event/Probability Flow Chart for Signal Detection

The relationships between the various parameters are depicted in Figure 2-21. The conditions, in addition to those indicated for each probability determination, are: number of ELTs/EPIRBs = 400, MSK modulation/Manchester data (k = 3), channel BW = 400 Hz, message length = 128 bits, data rate = 128 BPS, and period between transmissions = 60 seconds. This set of parameters is termed the baseline system. The probability of success for this set of parameters is:

$$
\begin{aligned}
P_s &= (0.8877)(0.9999)(0.9926)(0.999997)(0.9999988) \\
&= 0.881
\end{aligned}
$$

It is obvious that the probability of success is dominated by the probability of no mutual interference for N = 400. The probability of failure as a function of the number of ELTs/EPIRBs for this baseline system is shown in Figure 2-22.

If the minimum viewing time is 4 minutes, then an ELT or EPIRB can transmit a minimum of four messages (assuming T = 60 seconds) during one pass. The probability of receiving three good messages (the minimum required to locate a transmitter) is:

$$
{}^4p_3 = \sum_{x=3}^{4} \binom{4}{x} p_s^x (1 - p_s)^{4-x}
$$

$$
= 0.9279
$$

If the satellite is visible for a longer period, the probability of receiving three good messages increases:

$$
{}^5p_3 = 0.986
$$

$$
{}^6p_3 = 0.9975
$$

Returning to the problem of the number of available channels, the probability of failure is plotted as a function of N with L as a parameter for Manchester data (k = 3) and Miller data (k = 1.5). The plots are shown in Figure 2-23. The other probabilities and parameters remain as before. Observe that for Manchester data, eight loops are inadequate for N > 300, but there is little difference between L = 12 and L = 16 for N < 500. Therefore, a 12-channel system is both necessary and sufficient for a system serving 400 to 500 ELTs if Manchester data is used. With Miller data, 12 loops are inadequate for N > 400 if full advantage of Miller data is to be obtained. Hence, the number of channels required to serve 400 ELTs and EPIRBs is 12.

Another item to be observed in Figure 2-23 is the comparison of Miller data and Manchester data. Although Miller data offers an improvement in probability of success, this improvement is not sufficient to encourage the use of Miller data. This is particularly true when the use of Miller data will degrade the E/n_0 performance by 6 dB.

A final comparison plot is presented in Figure 2-24, which shows the difference between the baseline system and a baseline system without an error correcting code. Also, a comparison between RAMS (which used PSK and no error correcting code) and the baseline system is shown.

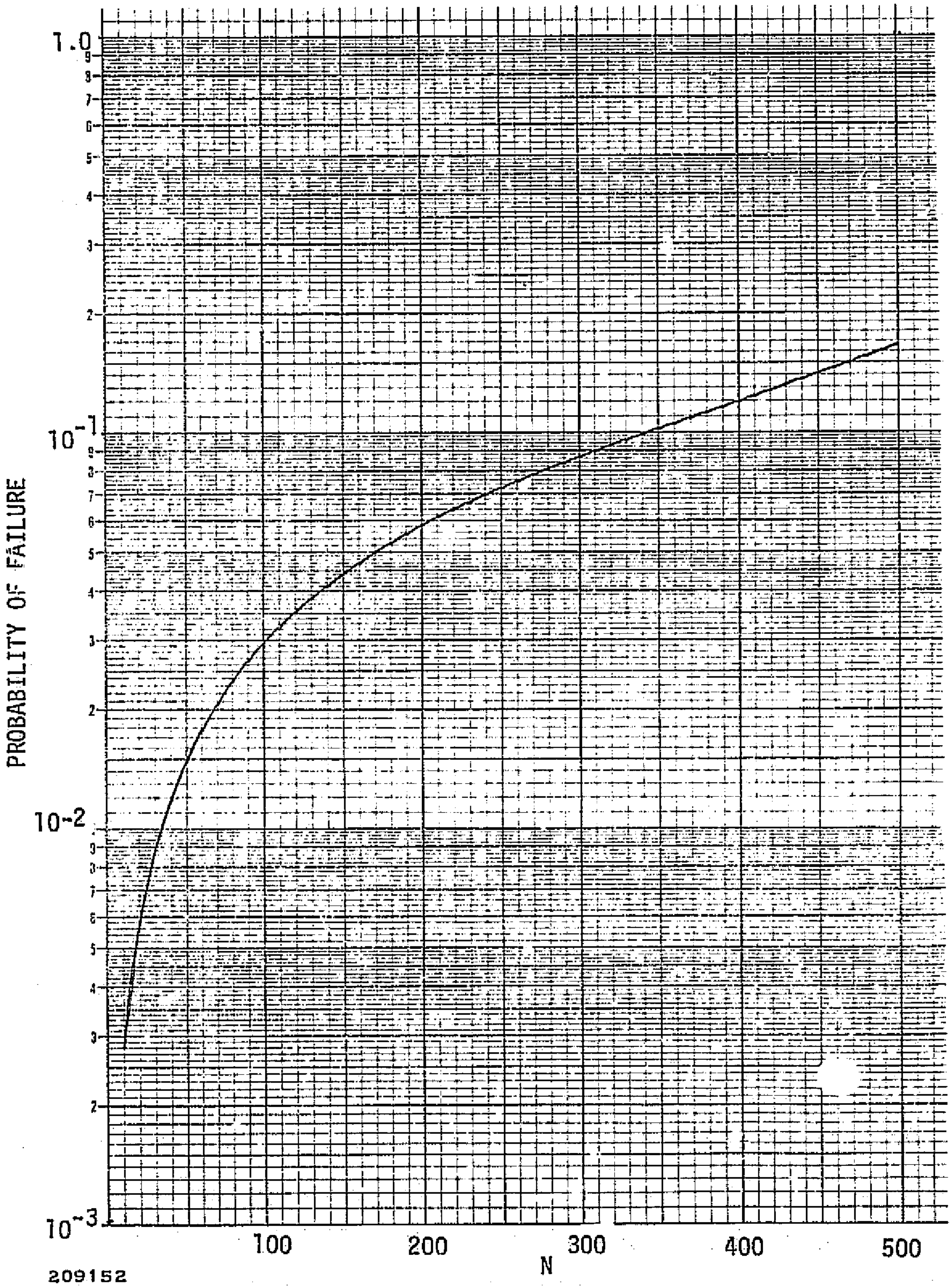


Figure 2-22. Probability of Failure to Receive a Given Message Versus the Number of Transmitters in the FOV for the Baseline SOS System

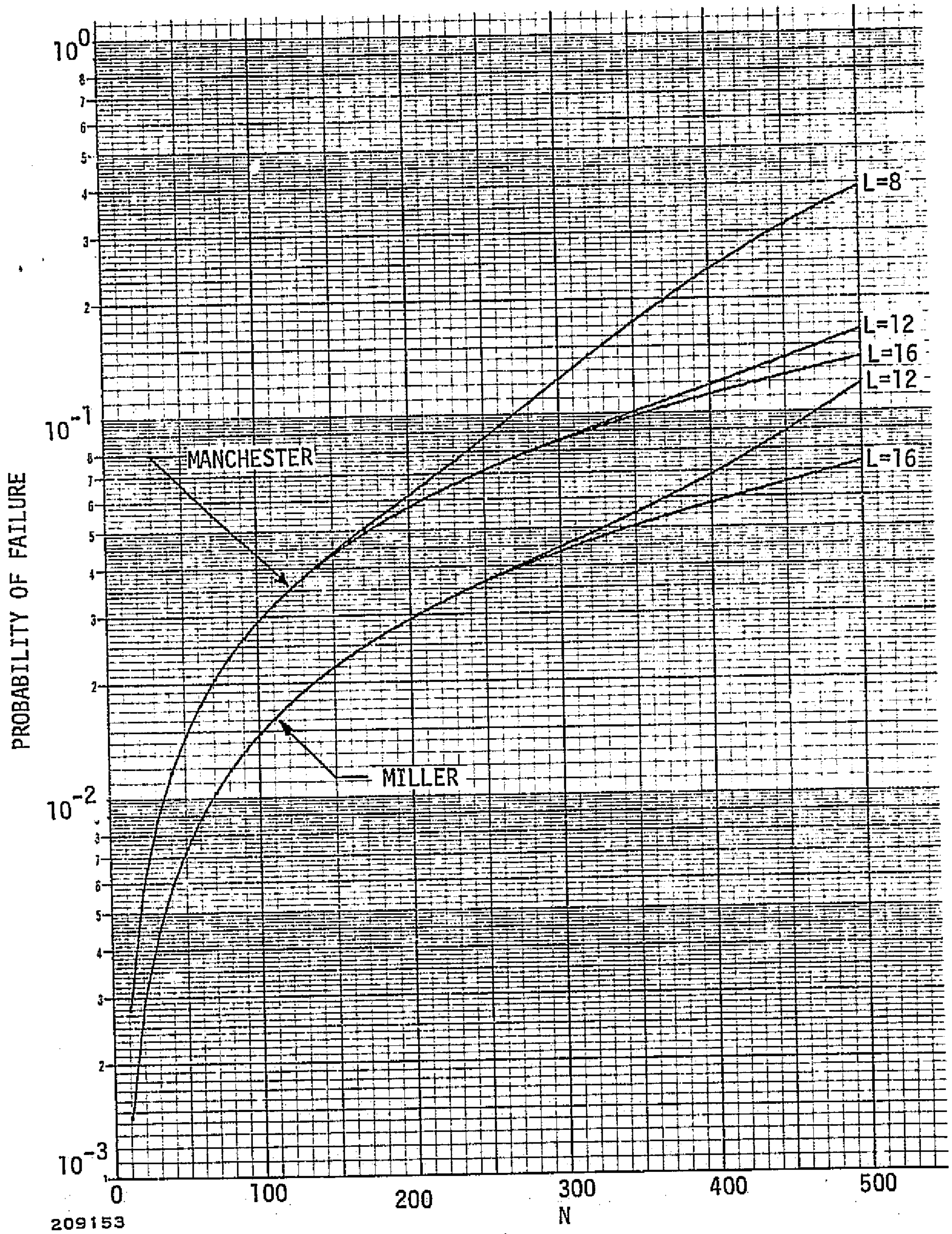


Figure 2-23. Comparison of Manchester and Miller Encoded Data as a Function of Transmitters Being Served Versus Probability of Failure (T = 60 Seconds)

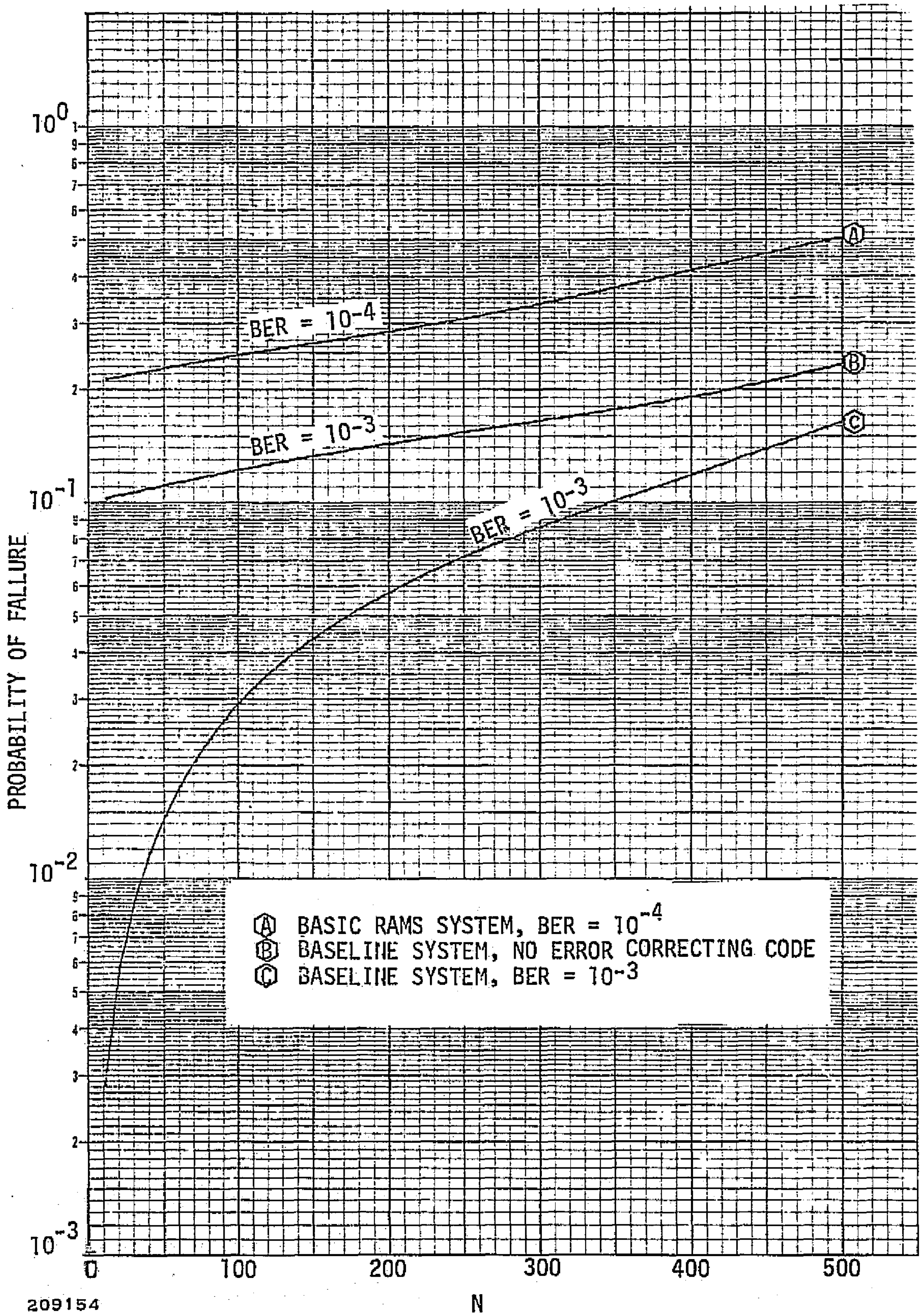


Figure 2-24. Comparison of Various Types of Systems as a Function of the Number of Transmitters Being Serviced Versus Probability of Failure

A review of the algorithms used to obtain position information within a ±5-km accuracy has indicated the need to obtain more than the minimum number of messages. The minimum number of messages believed to be necessary to achieve the desired accuracy is 4, and the period between transmissions should be reduced to 48 seconds. This allows the transmission of at least 6 messages within the 4-minute viewing time. For this case and with N = 400:

$$
\begin{aligned}
1 - P_I &= 0.86 \\
P_L &= 0.9698 \\
P_s &= (0.86)(0.9999)(0.9698)(0.999997)(0.9999988) \\
&= 0.8339
\end{aligned}
$$

Note that the net effect of reducing the period between transmissions from 60 seconds to 48 seconds is to reduce the probability of receiving any given message from 0.88 to 0.83. The corresponding probability of successfully receiving three messages (the minimum number) in 4 minutes of satellite viewing time is 0.965. The probability of receiving four messages in 4 minutes (five attempts) is:

$$^5P_4 = 0.8048$$

This is acceptable only when it is recognized that this is a worst case situation. If the satellite is visible for 4 minutes and 48 seconds, then six tries are permitted and the overall probability of receiving four messages is:

$$^6P_4 = 0.9383$$

Given a satellite visibility time of 5.5 minutes, the probability of obtaining location accuracies of ±5 km becomes:

$$^7P_4 = 0.9826$$

Therefore, the probability of success is not a threshold number, but rather the system provides a graceful degradation in accuracy as the system limits of viewing angle, number of devices, and service rate are approached.

One final comment needs to be made concerning the repetition rate of each transmitter. The 48-second interval is intended to define the mean interval. The actual interval should be "dithered" to ensure a random (uncorrelated) timing relationship between transmitters. Such a precaution is necessary to prevent transmitters that suffer time interference on one transmission from undergoing such interferences on succeeding transmissions. The optimum amount of dither required has not been determined, but a ±5-second pseudorandom time variations should be sufficient.

It is worthwhile to determine the probability of success as a function of the number of transmitters being serviced for a system having the characteristics selected in this section:

Transmission period	1 second
Period between transmissions (mean)	48 seconds
Usable spectrum	86 kHz

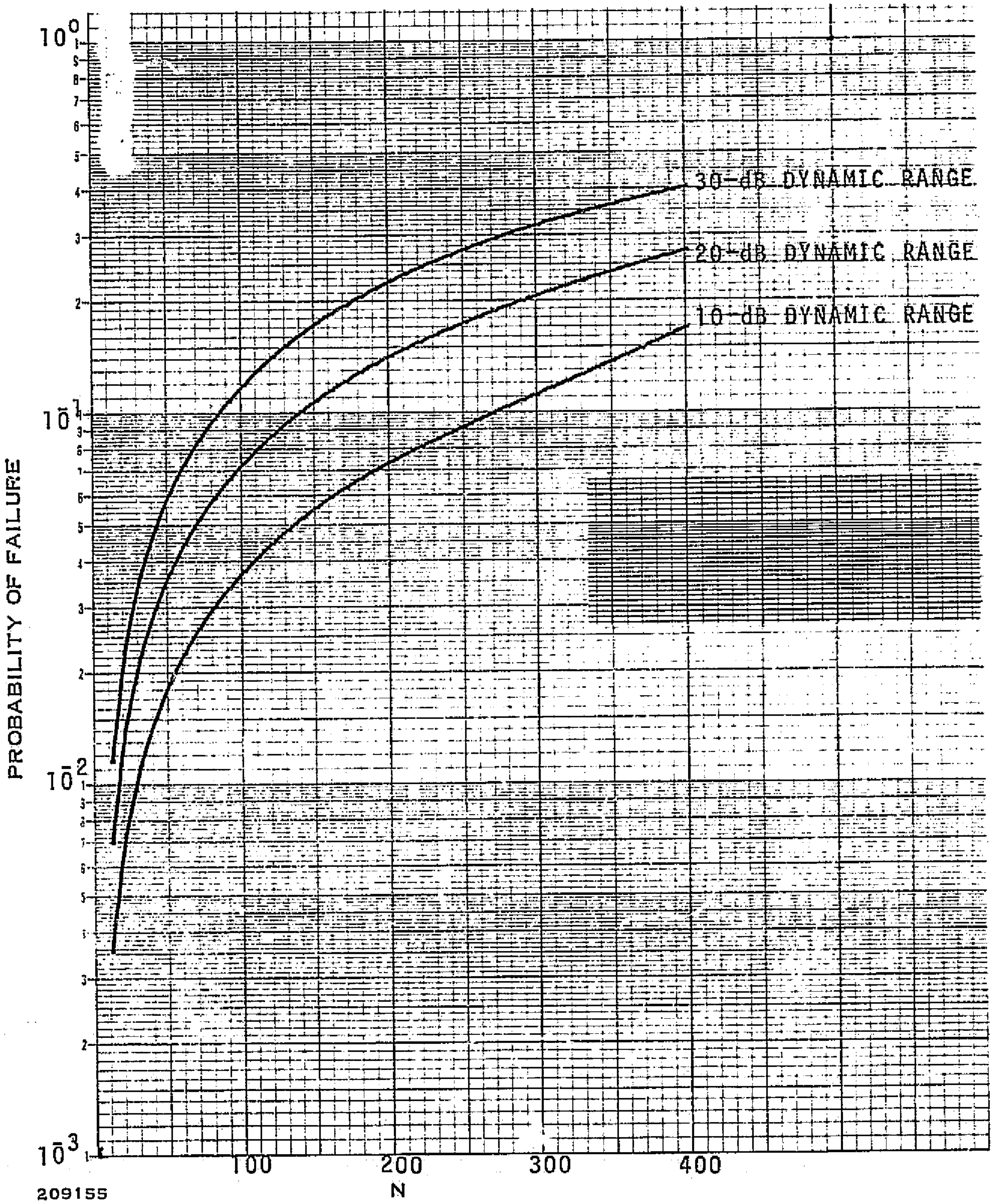


Figure 2-25. System Performance as a Function of Dynamic Range

Number of phase-lock loop receivers	12
Modulation	MSK
Data/clock coding	Manchester
Forward error correcting code	three-error correcting.

Although dynamic range was not listed, its relationship to system performance will be illustrated for three distinct dynamic range levels of 10, 20, and 30 dB. This is accomplished by using three corresponding channel use factors determined in Subsection 2.9 (Figure 2-12), i.e., $k = 3$, $k = 6$, and $k = 10$, respectively. A BER of 10^{-3} is assumed. The results are depicted in Figure 2-25. The weaker signals are more likely to suffer mutual interference, with some signals requiring the minimum (10-dB) dynamic range and others requiring the maximum. If 20-dB dynamic range is used as a mean, then the system will provide an acceptable performance level (99-percent probability of location to within ±5 km) for up to 300 ELTs and/or EPIRBs within the FOV.

No consideration has been given to the geostationary satellite system. The probability of mutual interference increases for such a system because the amount of frequency dispersion is reduced since no doppler shift is introduced. The total spectrum available for a geostationary satellite system is approximately 40 kHz instead of 86 kHz. Thus, the probability of receiving any given message is reduced in the geostationary satellite system, but this is compensated by the lack of constraints on satellite viewing time and satellite waiting time. Because several attempts are permitted and only one good message is required to obtain location information, the mutual interference problem is less severe for a geostationary satellite. The number of attempts (number of messages transmitted) necessary to achieve a 99-percent probability of detection as a function of the number of transmitters in the FOV is shown in Figure 2-26. The plot was made for a worst case repetition rate of once each 40-second interval. Therefore, any system capable of meeting the mutual interference requirements of a low-orbiter satellite will function with a geostationary satellite.

2.13 ANALYSIS OF AN INTERFEROMETER TECHNIQUE USING A GEOSTATIONARY SATELLITE

The basic underlying principle of operation for this technique is the existence of a unique relationship between the electrical phase angle of a received carrier and the mechanical or geometrical angle of a transmitter and a pair of receivers. A fundamental assumption is that the received signals from a given source are spatially parallel, i.e., the distance between the source and the two receivers is very great with respect to the displacement of the two receivers. For such a system to be mechanically feasible for spacecraft application, the wavelength of the carrier must be small with respect to the antenna baseline displacement.

The geometry for such an interferometer is shown in Figure 2-27. One receiver/antenna is located at point A; a second is located at point B. The distance between the center of antenna A and the center of antenna B is D. Point Q is the center of the baseline AB. The received signal arrives along a path parallel to line IQ; the angle of incidence is θ. Line PQ is perpendicular to baseline AB. The angle between baseline AB and the plane of the wavefront ac is also θ. Hence, the signal arriving at receiver B must travel an additional distance, s:

$$s = D \sin \theta$$

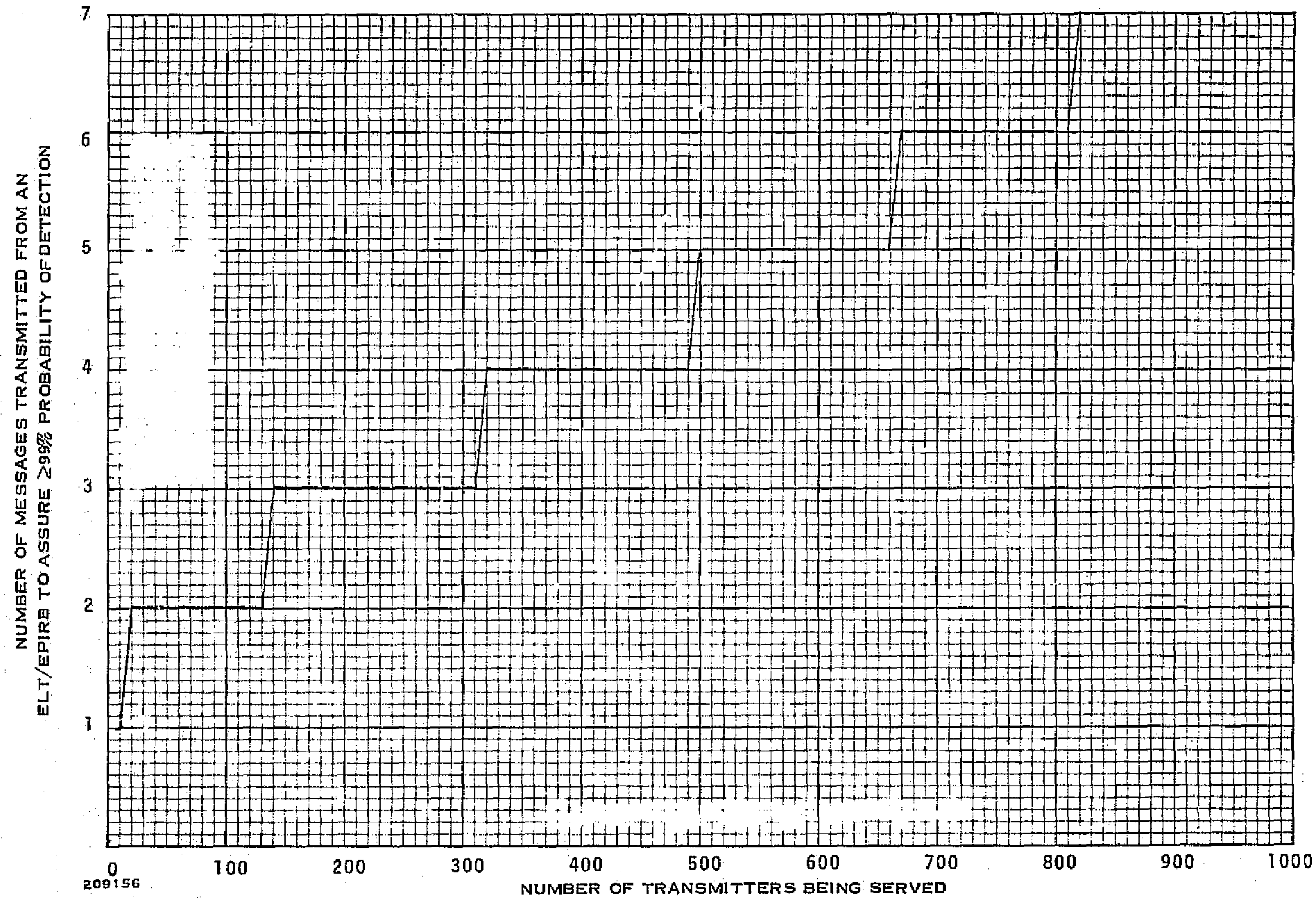


Figure 2-26. Number of Attempts Necessary Versus Number of ELTs/EPIRBs in FOV of a Geostationary Satellite to Assure 99-Percent Probability of Detection (T = 40 Seconds)

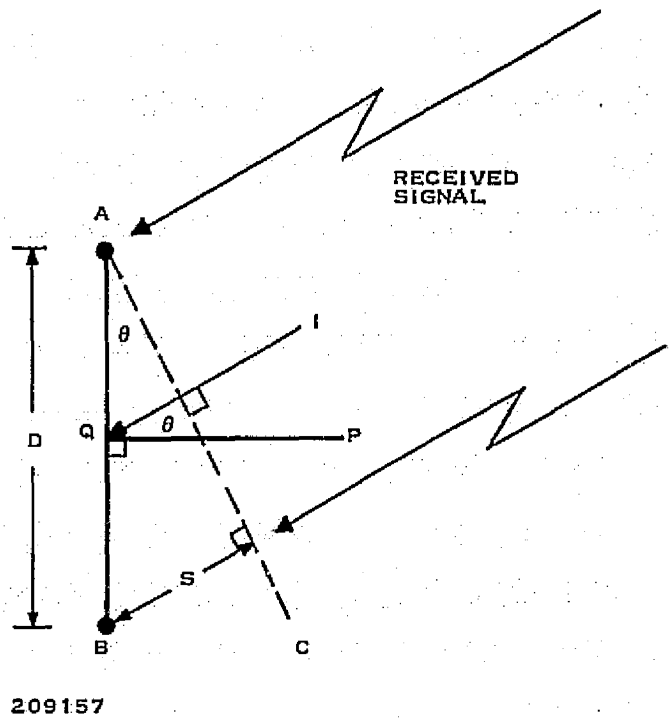


Figure 2-27. Geometry for the Basic Interferometer System

Since the signals were transmitted from a common source and traveled along virtually parallel paths, the signal arriving at point B is displaced in phase with respect to the signal at point A by precisely:

$$\phi = \frac{2\pi}{\lambda} s$$

where

ϕ = the phase displacement of the signal received at point B with respect to the signal received at point A in radians

λ = the wavelength of the received signal.

A simplified sketch of an interferometer system is shown in Figure 2-28. The phase angle of the received carrier is accurately determined or both receivers. The difference in the electrical phase angle, ϕ, is determined, and the angle of incidence, θ, is computed:

$$\theta = \text{arc sin}\left(\frac{\lambda\phi}{2\pi D}\right)$$

where

λ = the wavelength of the received signal

D = the length of the antenna baseline

ϕ = the difference in the phase of the two received signals in radians.

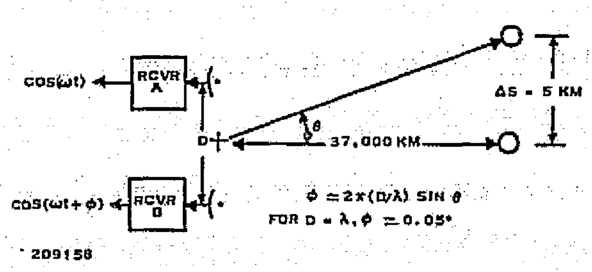


Figure 2-28. Interferometer Position Locating System

Because the value of ϕ can only be measured unambiguously to $\pm\pi$, the following must be true if θ is to be unique:

$$-\pi < \frac{2\pi D}{\lambda} \sin\theta < \pi$$

Since the value of θ can be ± 10 degrees for a geostationary satellite, then $\sin\theta \leqslant 0.17$. Hence,

$$D \leqslant 3\lambda$$

For a carrier frequency of 406 MHz, this corresponds to a baseline distance of 2 meters. However, previous calculations have shown that the antenna diameters alone must be approximately 2.8 meters and it is therefore impossible to have an unambiguous system without some method of coarse phase measurement.

One technique for deriving a coarse angle measurement would be the use of a fifth antenna. (A second pair of orthogonal antennas is required for two-dimensional position locating.) The fifth antenna could perform the coarse angle measurement by using a technique similar to that used to position tracking antennas, i.e., relative signal strength measurement. A second technique would use an off-focus sensing antenna in one of each pair of main antennas. No further consideration is given to the ambiguity problem.

The final consideration for this technique is accuracy. For the antenna size required (2.8 meters) and wavelength (0.75 meters), the minimum D/λ ratio is approximately 3.75. For practical consideration, the ratio will be assumed to be 10, which corresponds to a displacement of 7.5 meters. The positioning should be performed to within ± 5 km. At an altitude of approximately 37,000 km, this corresponds to an angular displacement of

$$\begin{aligned} \delta\theta &= \text{arc tan} \frac{5}{37{,}000} \\ &= 0.0077 \text{ degree} \end{aligned}$$

from which

$$\begin{aligned} \delta\phi &= 2\pi \frac{D}{\lambda} \sin(\delta\theta) \\ &= 0.0085 \text{ radian} \\ &= 0.486 \text{ degree} \end{aligned}$$

From these calculations, it is apparent that the mechanical alignment of the system must be calibrated and maintained to within ± 0.007 degree or approximately ± 30 seconds of arc. The electrical phase measurement must be made accurate to within ± 0.4 degree. This error includes any phase error introduced by multipath signals.

The effect of multipath signals is evaluated by

$$\frac{V_r}{V_i} \max \leq \tan 0.4 \text{ degree}$$

$$= 0.007$$

Hence, any reflected signal must not exceed −43 dB with respect to the primary signal. Because reflected signal strengths will probably exceed this value, it is expected to be a major source of error.

In summary, the interferometer can be used to do angle measurement and therefore positioning via a geostationary satellite. The major design problems with this concept are:

1. A coarse angle measurement is required to eliminate ambiguity problems generated in the fine angle measurement technique.
2. The mechanical alignment of the system is critical and the accuracy requirements approach those of an optical system.
3. Multipath reflections may impair system accuracy.

2.14 ANALYSIS OF UNILATERAL RANGING (HYPERBOLIC RANGING) USING GEOSTATIONARY SATELLITES

The hyperbolic system referred to here is a time-difference system. Although its salient feature is being able to determine range without requiring a transponder, the system does require a minimum of three geostationary satellites to achieve the desired ranging. (A transponder ranging system requires only two satellites.) The principle of operation is the same as that employed in loran. It is based on the accurate determination of the time of arrival of received signals. The difference between the arrival times at any pair of satellites establishes a unique hyperbolic line of position in a plane or a unique hyperboloid in space. By assuming a known earth geometry, the intersection of the surface of the hyperboloid and the surface of the earth establishes a unique line of position on the surface of the earth. A second such line of position is formed from the second pair (one is common to the other measurement) of time measurements. If the three satellites are within the equatorial plane, the two lines of position will intersect in only two points: one in the northern hemisphere and one in the southern hemisphere. Thus, it will be necessary to somehow resolve this ambiguity. In addition, since the three satellites will be co-planar, the position accuracy will degrade toward the equator. This can be better evaluated by considering the transverse error as a function of a longitudinal error.

The transverse error associated with a given measurement error along the longitudinal path is a function of the off-axis angular displacement of a transmitter. This is illustrated by Figure 2-29 and the following calculations. The amount of angular error associated with a surface displacement error is:

$$\delta\theta_e = \frac{360\ \delta s_e}{2\pi\ (6370)}$$

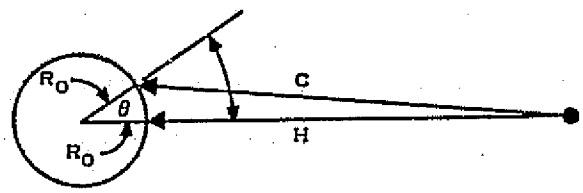


Figure 2-29. Geometry for Determining the Minimum Value of θ Allowable for Obtaining a 5-km Position Accuracy Using a Geostationary Satellite and Hyberbolic Ranging

where

$\delta\theta_e$ = the angular error in degrees

δs_e = the position error in km.

If the maximum allowable value for δs_e is 5 km, then:

$$\delta\theta_e \text{ max} = \frac{(360)(5)}{2\pi\ (6370)}$$

$$= 0.044973 \text{ degree, maximum}$$

For an initial value of θ = 10 degrees, the value of C (the slant-range) is found by the solution of the triangle problem:

$$C_1 = 3.596379 \times 10^4 \text{ km}$$

For an error of $\delta\theta_e$ = 0.044973 degrees, then:

$$C_2 = 3.596481 \times 10^4 \text{ km}$$

hence

$$\delta C_e = 1.021 \text{ km, maximum}$$

As the value of θ increases, the maximum allowable value of δC_e increases. Therefore, the system measurement accuracy should be sufficient to resolve slant-range distances to within ±1 km with the restriction that the minimum off-axis angle be 10 degrees; i.e., the minimum usable conic half-angle displacement from a line formed by the satellite and the earth's center is 10 degrees. If the satellite's orbit lies within the equatorial plane, then reasonably accurate results can be expected in latitudes exceeding ±10 degrees.

Since the maximum allowable range error is to be ±1 km, the maximum time of arrival error to be allowed is determined from

$$\delta t_e = \frac{\delta C_e}{\nu} = \frac{1\ \text{km}}{3 \times 10^5\ \text{km/s}} \quad (\nu = \text{velocity of light})$$

$$= 3.33\ \mu s$$

Therefore, a design goal should be to determine the time of arrival within ±1 μs with ±3 μs being the maximum allowable. If the bit clock is to be used to establish the precise time of arrival (after the determination of frame sync, of course), the bit clock jitter should be no more than ±1 μs. If the bit clock is 128 Hz, this corresponds to a ±0.013-percent jitter, maximum. This is equivalent to a phase error of ±0.045 degree p-p or 0.016 degree rms. This is obviously an extremely tight requirement and can only be achieved with a phase-lock loop having high loop gain and a narrow bandwidth. In addition, the VCO must have very good short-term stability.

Before further consideration of system accuracy, the technique for determining time of arrival should be discussed. The method under consideration uses a coarse time measurement and a fine time measurement. The coarse time measurement is established by the frame sync code; it is the time at which frame sync is recognized. This time will be accurate to within ±¼ bit time. A narrow bandwidth phase-lock loop locks to the data clock and tracks this frequency for a predetermined number of bit times. After this period, a well-defined clock transition marks the precise time of arrival. The key parameter that limits system accuracy is jitter of the highly processed bit clock. This jitter (assuming a phase-lock loop approach) is related to system signal-to-noise ratio and loop bandwidth. These relationships are well documented for phase-lock loops, Subsection 2.9.

The minimum practical loop bandwidth is thought to be approximately 5 Hz for a satellite system. The minimum loop noise bandwidth is achieved for a loop damping factor of 0.5, in which case the noise bandwidth is related to loop bandwidth* by:

$$B_L = 0.5\ \omega_n$$

where

B_L = loop noise bandwidth, Hz

ω_n = loop bandwidth, rad/s

from which:

$$B_L = 0.5\ (2\pi)\ (5)$$
$$= 15.71\ \text{Hz}$$

The rms output phase jitter is:

$$\sqrt{\theta^2_{n_o}} = \sqrt{\frac{W_i\ B_L}{P_s}}$$

*This assumes a second-order loop.

where

$$\theta^2_{n_o} = \text{the mean-square output phase jitter, rad}^2$$
$$P_s = \text{the input signal power, watts}$$
$$W_i = 2n_o$$
$$= n_o \text{ (dB)} + 3 \text{ dB}$$

where

$$n_o = \text{input noise density.}$$

Since the worst case signal-to-noise density for a geostationary satellite was found to be approximately:

$$\frac{n_o}{P_s} = -40 \text{ dB}$$

then

$$10 \log \frac{W_i}{P_s} = -40 \text{ dB} + 3 \text{ dB}$$

and

$$\frac{W_i}{P_s} = 2 \times 10^{-4}$$

The rms phase jitter is

$$\sqrt{\theta^2_{n_o}} = \sqrt{2 \times 10^{-4} \; B_L}$$
$$= 5.6 \times 10^{-2} \text{ rad}$$
$$= 3.2 \text{ degrees}$$

which is many times the maximum allowable jitter needed to obtain ±5-km accuracy. This corresponds to a root-mean-square timing error of 0.89 percent, or 70 μs for a data rate of 128 BPS, which is approximately 20 times the limit required to maintain a ±5-km error. The rms error at a latitude of 10 degrees can be expected to be approximately 100 km or 60 miles. This error will decrease as the distance from the equator increases. At 30-degree latitudes, the error will be reduced to 35 km or 20 miles.

Therefore, it is seen that the system accuracy will be poor near the equatorial regions with a moderate error above 30 degrees north latitude. The error is sufficient to indicate that the system is not optimized for this measurement technique. Improved performance would result if a higher frequency side tone could be used. This is not possible, however, since such a tone would increase the channel bandwidth occupied by an ELT, thus increasing mutual interference. A second alternative would be improving the received signal-to-noise ratio. This could possibly be done by using an antenna with reduced beamwidth and higher gain. Also the receiver noise figure could be improved. A 10-dB improvement in signal-to-noise performance would improve accuracy by a factor of 3. Such a sys em would approach the desired performance goals.

2.15 SECTION 2 REFERENCES

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SECTION 3

ELT/EPIRB IMPLEMENTATION

Section 2 has analyzed the overall system problem and established the requirements for both the ELT/EPIRBs and the spacecraft equipment. The purpose of this section is to examine the implementation problems in the ELT/EPIRB and to provide a better understanding of the ELT/EPIRB equipment.

The three major considerations for the ELT/EPIRB are reliability, performance stability, and cost. In light of the reliability and performance considerations, the design analyses presented in this report are conservative and are based on experience at Texas Instruments with military equipment programs. Rigorous design would, no doubt, simplify the circuitry and bring down the cost.

3.1 ELT/EPIRB REQUIREMENTS

Before beginning a detailed discussion of the ELT/EPIRB design, a brief summary of the requirements is given. As shown in Table 3-1, the ELT/EPIRB is a 5-watt (effective radiated power, ERP) transmitter at 406.05 MHz with a relatively low duty cycle (on for 1 second every 48 seconds). This low duty cycle is employed to lengthen the transmit lifetime (battery lifetime) to about 100 hours and to allow system time division multiplex operation. A digital data message included with each transmission has a unique identification code for each ELT/EPIRB; the capability for specifying the type of emergency has also been included.

The beacon transmitter is a low-powered (15-mW) CW transmitter operating at 121.5 MHz. Its function is to provide a signal for detection by direction-finding equipment on board rescue craft when in proximity to the ELT/EPIRB.

3.2 ELT/EPIRB DESIGN

3.2.1 ELT/EPIRB Design Approach

As shown in Figure 3-1, the ELT/EPIRB consists of the following basic functions:

406.05-MHz transmitter

121.5-MHz beacon

Battery

Activator.

The 406.05-MHz transmitter provides 2.5 watts of RF power into a 3-dB gain antenna which results in a 5-watt ERP. Pulsed operation is employed with a duty cycle of 1 second in every 48 seconds, lowering the power requirements from the battery power supply.

TABLE 3-1. ELT/EPIRB CHARACTERISTICS

Frequency	406.05 MHz
Stability	
Long-term	±0.003 percent (over temperature & life)
Medium-term	±1 Hz/min
Short-term (fractional frequency deviation)	2×10^{-9} maximum
Power output (ERP)	5 W
Antenna gain	3 dB peak
Spurious output	50 dB below 5 W
VSWR	Infinite without damage
Modulation	Minimum shift keying (MSK)
Transmit lifetime (battery life)	100 hours
Operating temperature	−40° to +50°C
Bit rate	128 BPS
Data coding	Manchester
Message length	115 bits
CW preamble	102 ms
Transmission time	1 s
Repetition rate	48 s
Codes in message	
Emergency	32
User	4
ID codes	48 bits*
Beacon	
Frequency	121.5 MHz ±0.003 percent
Power output	15 mW
Modulation	none, CW only

*See text, Subsection 2.6.

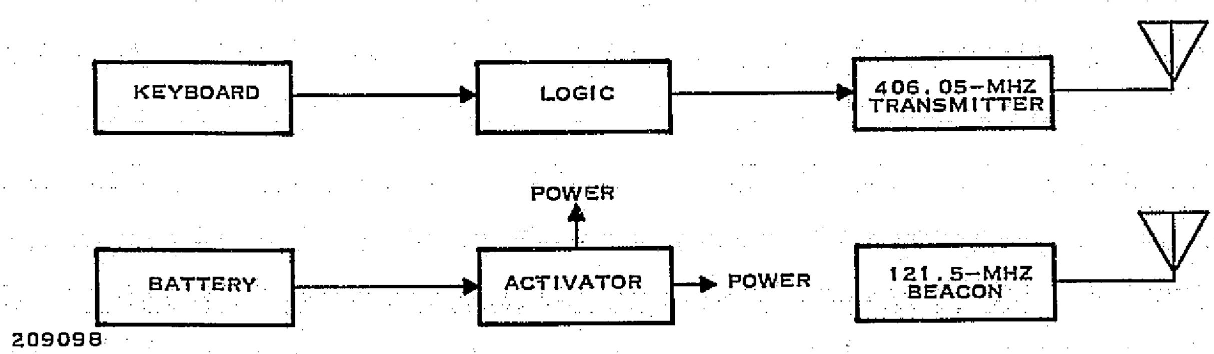

Figure 3-1. ELT/EPIRB Block Diagram

The logic generates the data stream for modulating the 406.05-MHz transmitter and provides the following functions:

- Timing
- ID code generation
- Keyboard interface/emergency code generation
- Bit sync/frame sync code generation
- Error correcting encoding
- Manchester encoding.

For the more complex EPIRB installations, a keyboard will be used by the operator to input emergency codes.

As previously mentioned, the 121.5-MHz beacon provides a 15-mW carrier for homing purposes by rescue craft in proximity to the ELT/EPIRB.

Power for the ELT/EPIRB will be supplied by internal lithium batteries which, with the use of pulsed transmitter operation, will provide operation for approximately 100 hours. In EPIRB installation, power could be supplied externally, but standby operation from battery power would always be desirable.

The activator that turns on the ELT/EPIRB will be a "g" switch, an oil-pressure monitor, etc., for the aircraft ELT. In the EPIRB, the activator will be a manual or a water-contact switch. The study of the activator is beyond the scope of this study and it is not discussed further in this report.

3.2.2 Antenna Considerations

Five types of antennas were studied for application as the ELT or EPIRB transmit antenna. Factors used in evaluating antennas included manufacturing cost, radiation pattern, suitability for the application, size, and weight. While a detailed antenna evaluation is beyond the scope of this study, a brief description of the basic requirements and potentially suitable design concepts is included.

The general requirements for the ELT/EPIRB antenna are that it:

- Produce a circularly polarized signal
- Provide a 3-dB gain
- Provide a donut-shaped hemispherical radiation pattern with a maximum gain variation of 3 dB over 360 degrees of azimuth angle and 10 to 70 degrees of elevation angle
- Have suitable aerodynamic properties (ELTs) and be resistant to damage
- Be economical to build/install.

Since it is possible that antenna orientation in an emergency situation may not be as desired and, in addition, the antenna is subject to damage, the antenna selected should be capable of providing a usable signal in such suboptimum situations.

The first antenna type considered is the whip, which is the dominant type used in current ELTs. Its major problems are that it does not permit much flexibility in shaping the radiation pattern and it is subject to damage. However, the whip is a logical choice for the 121.5-MHz beacon to be used with the second-generation ELTs. It may also be a reasonable choice for EPIRBs.

A group of antenna types collectively referred to as conformal antennas offer more flexibility. The first is the stripline antenna. It is constructed using printed circuit stripline methods, with a resonant cavity behind the stripline antenna. This design has mechanical/reliability problems and is not an economical approach. The stripline antenna also requires a considerable amount of mounting space for 406-MHz versions.

A second type of conformal antenna is the microstrip antenna. It is a generally feasible approach but has the intrinsic microstrip problem of being sensitive to surrounding conductors. This causes interaction between adjacent antenna elements and has the potential for severe alteration in the antenna pattern by the surrounding medium.

A third antenna in the family of conformal antennas is the slotline antenna. It is essentially the opposite of a microstrip antenna, having conductor gaps in the place of conductors. The major difficulty in using this technique is matching it to a transmission line.

A final antenna design which overcomes many of the above design problems is the coplanar stripline antenna. Essentially, it combines the microstrip and slotline techniques to provide a well-behaved low-cost fabrication method for conformal antennas. The E-field is contained in the gaps between conductors. Antenna patterns can be etched in the plane with little coupling between elements. The construction is simple and the antenna feed is easily implemented. Circular polarization is easily achieved using two adjacent elements fed from a 90-degree hybrid; the hybrid is part of the antenna.

A sketch of a single-element coplanar stripline antenna is shown in Figure 3-2. The antenna is etched on a double-sided printed circuit board (PCB) on which one side is a solid ground plane. The antenna can be fed directly from the rear side. However, for the ELT application, additional elements would be used in conjunction with a hybrid to produce the desired circularly polarized radiation pattern. A sketch of such a pattern is shown in Figure 3-3.

The coplanar stripline antenna has most of the desired electrical features for the ELT application. Being planar and conformable, it can be mounted flush with an aircraft surface. It is simple and can be made of lightweight materials. Before final selection, however, it must be experimentally evaluated to ensure that it produces a usable signal in spite of abnormal orientations. The antenna could be used for EPIRBs, but a modified whip might be more desirable for this application. The whip is still the best candidate for the 121.5-MHz beacon.

3.2.3 Power Source Considerations

Although power for the ELT/EPIRB could be supplied externally in large installations, the use of internal batteries is the only reasonable configuration from a reliability standpoint for ELTs. Both primary (nonrechargeable) and secondary (rechargeable) batteries can be used. However, the reliability and shelf life (time that battery maintains full charge) rule out the use of secondary batteries.

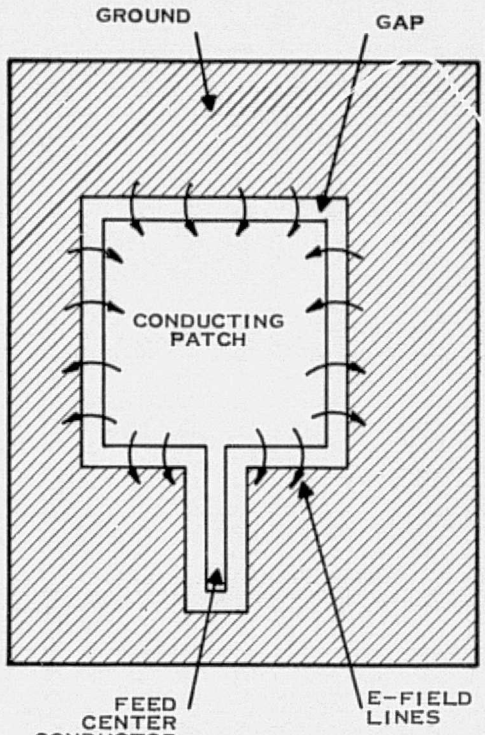

Figure 3-2. Coplanar Stripline Antenna/Feed (Fundamental Model)

In the last few years, developments in primary battery construction have vastly improved performance. As discussed in numerous articles,[1, 2] lithium batteries provide outstanding performance in terms of capacity, flat voltage discharge, shelf life, and operating temperature.

A summary of the characteristics of the various primary battery types is shown in Table 3-2. Examination of the operating temperature range quickly reduces the choice to only two types, alkaline and lithium. The required operation at –40°C can only be met by lithium.

Figure 3-4 compares the relative capacity (watt-hours) for the various types of primary D-cells at 70°C and 1-ampere load. The lithium D-cell's capacity is equivalent to four mercury-zinc D-cells, five alkaline manganese D-cells, seven magnesium D-cells, or thirty carbon-zinc D-cells.

The cell voltage of lithium is nominally 2.95 volts (unloaded) and exhibits relatively flat voltage discharge characteristics, as shown in Figure 3-5. During storage, the lithium cell develops a passive film which greatly reduces any parasitic side reactions and results in a shelf life of several years. The passive film is quickly removed when load is applied to a cell.

The major obstacle to the lithium cell is that it outgasses hydrogen and sulfur compounds. In cases of shorts, extreme pressures are built up within the cells and a venting system is required to prevent explosions. Sealing of the cells has improved over the past few years and fusing of battery packs is normally done to avoid the short-circuit problem. The outgassing problem is serious since it can cause corrosion of printed circuit boards and result in malfunctioning equipment.

In spite of these problems, the lithium battery remains the only viable power source for future ELTs and is now used in a large number of current ELTs. Improvement in battery sealing and housing the batteries in separate compartments will substantially lower the risk of corrosion for future ELTs.

The battery pack for the ELT/EPIRB would consist of five lithium cells in series. With a nominal cell voltage of 2.6 volts (loaded) the battery pack voltage will be 2.6 X 5 = 13.0 volts. D-cells of 10 ampere-hour capacity would be used. One additional D-cell would be used to supply power to the logic circuits.

[1] "Battery Technology–Packaging More Muscle Into Less Space," *Electronics*, April 3, 1975, pp. 75–82.

[2] "In Power, Longevity, and Voltage, Batteries Are Reaching New Peaks," *Electronic Design*, May 24, 1974, pp. 28–34.

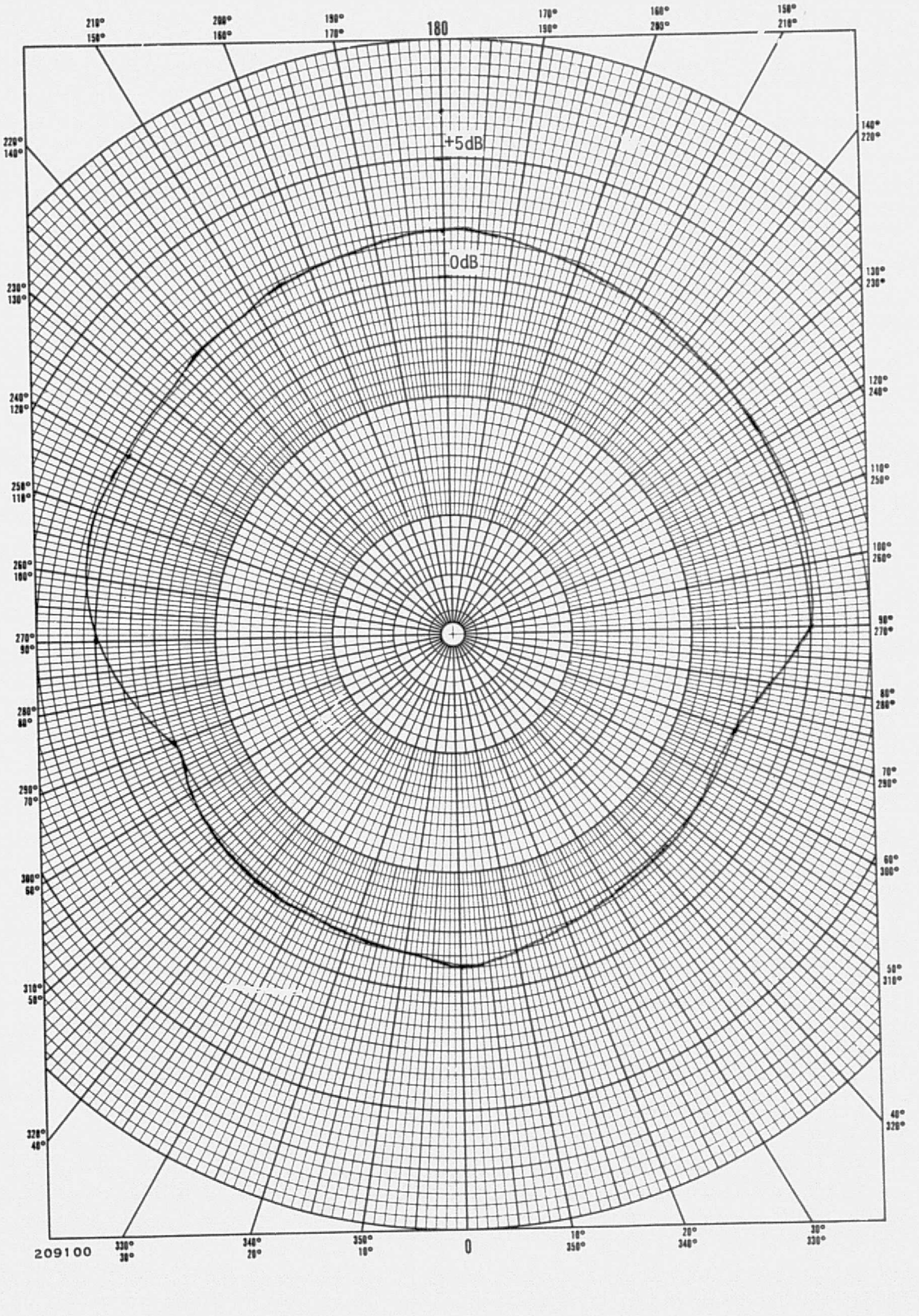

Figure 3-3. Antenna Pattern for Coplanar Stripline Antenna

TABLE 3-2. ELECTRICAL CHARACTERISTICS OF PRIMARY BATTERIES

	Leclanche	Zinc-Chloride	Alkaline	Magnesium	Mercury-Oxide	Silver-Oxide	Divalent Silver-Oxide	Lithium
Energy output								
Watt-hours per lb	20	44	20 to 35	40	46	50	70	100 to 150
Watt-hours per in^3	2	3	2 to 3.5	4	6	8	14	8 to 15
Nominal cell voltage	1.5	1.5	1.5	2.0	1.35 or 1.4	1.50	1.5	2.8
Practical drain rates								
Pulse	Yes	Yes	Yes	No	Yes	Yes	Yes	Yes
High (>50 mA)	100 mA/in^2	150 mA/in^2	200 mA/in^2	200 to 300 mA/in^2	No	No	No	Yes
Low (<50 mA)	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes
Impedance Z_i	Low	Low	Very low	Low (Delay on startup)	Low	Low	Low	Less than 1 ohm
Temperature range								
Storage	–40° to +120°F	–40° to +160°F	–40° to +120°F	–40° to +160°F	–40° to +140°F	–40° to +140°F	–40° to +140°F	–65° to +160°F
Operating	+20° to +130°F	0° to +160°F	–20° to +130°F	0° to +160°F	+32° to +130°F	+32° to +130°F	+32° to +130°F	–40° to +130°F
Temperature versus capacity	Poor at low temperature	Good at low temperature compared to Leclanche	Fair to good at low temperature	Fair at low temperature	Good at high temperature Poor at low temperature	Poor at low temperature	Poor at low temperature	Excellent
Shelf life at 68°F to 80 percent initial capacity (in years)	2 to 3	2 to 3	3 to 5	2 to 3	2 to 3	2 to 3	2 to 3	3 to 5 (estimated)
Shape of discharge curve	Sloping	Sloping	Sloping	Fairly flat	Flat	Flat	Flat	Flat

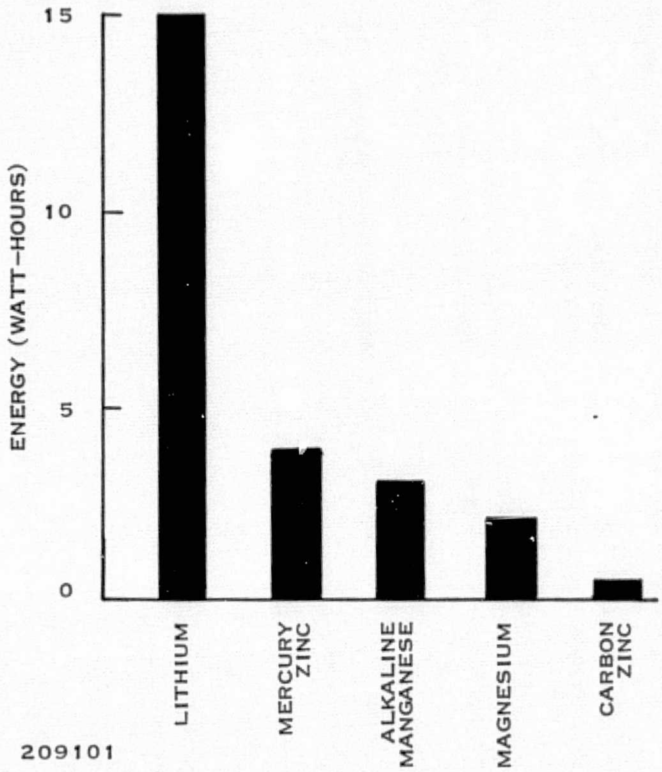

Figure 3-4. Comparison of Energy of Primary D-Cells

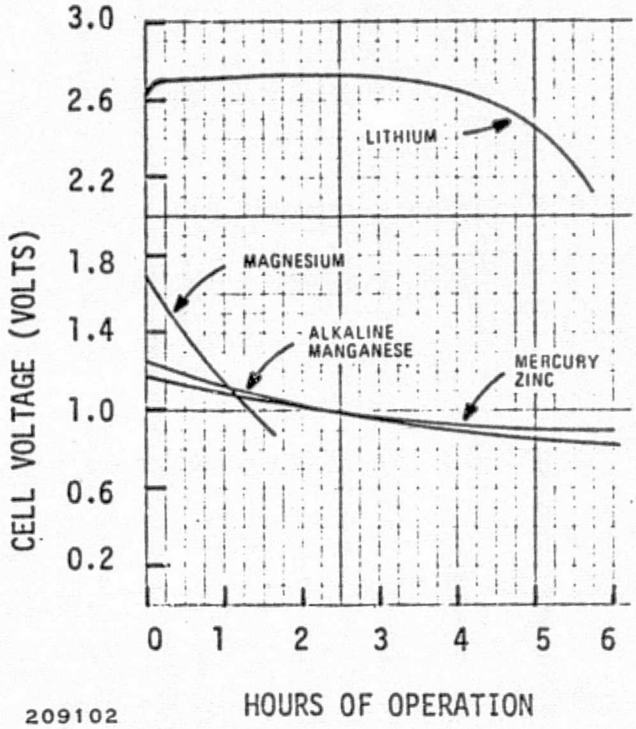

Figure 3-5. Voltage Versus Time for D-Cells at 1A Load

The goal of 100 hours of operation from the batteries is possible only if pulsed operation is employed in the ELT. An estimate of transmitter lifetime is made and discussed in Subsection 3.2.7.

3.2.4 Logic Considerations

The logic circuitry performs all timing, data generation, encoding, and control functions in the ELT or EPIRB. Each device will contain a unique ID number and user code stored in some form of read-only memory (ROM). The logic circuitry will provide data to the modulation circuitry of the RF section. In general, the logic can be implemented two ways.

The first method considered is a microprocessor approach. This approach may seem like an overkill at first glance, but some devices are so inexpensive that they may provide an economical approach. In particular, devices that contain ROM, random-access memory, and processor on a single chip are attractive. Current microcomputers of this type are capable of performing the logic functions, but appear to require more power than is desirable.

Another possible choice is the custom LSI chip. An approach using integrated injection logic (I^2L) to provide all logic functions except ID storage may be feasible. Power requirements appear to be less than 25 mW including an oscillator for the clock; only a single 2.5-volt source would be needed. The chip area required for the logic is only 25 percent of what is currently possible. This implies that yields would be good and costs would be low. This approach would probably yield the most efficient logic design since the circuitry could be customized.

3.2.4.1 *Data Format*

The data format has been discussed in Section 2. The ELT/EPIRB logic must generate the data format in the correct sequence and at the proper data rate. The sequence is:

1. Initiate transmission.
2. Provide 102 ms unmodulated preamble (13 bit times).
3. Output an all-1's data pattern for bit sync (15 bits).
4. Output frame sync (24 bits).
5. Output user code/ID/emergency code (55 bits).
6. Produce/output parity check code (21 bits).
7. Terminate transmission.
8. Provide 48-second delay (randomly timed).

The output data rate is 128 bits per second and the transmit period is 1 second. The data is in Manchester format.

3.2.4.2 *Logic Functions*

The logic circuitry must control the ELT/EPIRB. It provides the following functions:

Timing

ID code generation

Keyboard interface/emergency code generation

Bit sync/frame sync generation

Error correcting encoding

Manchester encoding.

The use of custom large-scale integration (LSI) technology to perform these functions makes the circuit design method essentially the same as that used for standard transistor-transistor logic (TTL) random logic. The use of a microprocessor approach implies that the design is primarily one of software generation. Both of these implementation methods are discussed in Subsection 3.2.4.3. This subsection considers circuit functions without regard to implementation technique where possible; the hardware approach is assumed where necessary.

Timing is critical for the ELT/EPIRB operation. There are two timing operations to be performed in the ELT/EPIRB. First, the period between transmissions must be controlled. It is desired that the mean time between transmit pulses be 48 seconds. However, it is also important that this interval not be periodic. It should be of a random nature or at least have pseudorandom characteristics. This is important in reducing mutual interference between transmitters. For example, if the period between transmissions of an ELT is precisely controlled and is periodic, then two ELTs or EPIRBs that interfere with each other on one transmission will probably interfere with each other on the following transmission. Hence, it is desirable to introduce some dither in the timing of ELT/EPIRB transmissions. This random time variation should cause the period between transmissions to vary between 40 and 60 seconds with a mean of 48 seconds. Such a technique will help ensure the desired Poisson distribution of transmissions assumed in the analysis.

The second timing operation is sequencing of the transmission interval and control of the output data rate. This timing must be precise, especially if hyperbolic ranging capability is to be retained. If optimum MSK is desired, then the data clock must be coherent with the RF carrier; therefore, the data clock should be derived from the RF crystal oscillator. Otherwise, the data clock can be generated from a stable source such as a watch crystal.

The data timing consists of producing a precise output data clock of 128 Hz. Multiples of this frequency and possibly other phases of the 128-Hz or 256-Hz clock are needed in the LSI logic approach. From this data clock, it is necessary to provide the precise timing signals that are needed to furnish 102 ms (13 bit periods) of unmodulated carrier at the beginning of transmission, to furnish the modulation data, and to terminate the transmission interval. If a microprocessor is used, it may be necessary to provide an additional clock for processor timing.

The event sequence that the timer must control is shown in Figure 3-6. The start pulse is generated from the 48-second timer already discussed. The sequence shown is executed once each time a start pulse is produced. The sequence may be controlled either by a custom random-logic chip or by a suitable microprocessor. The timing oscillator and RF power amplifier are held off except for the 1-second transmit interval. The crystal oscillator remains on to maintain frequency stability. If a countdown chain is used to derive the bit-time signal from this oscillator, it is also kept off. If a microprocessor is used, it can be turned off except during the transmit interval provided the proper initiating sequence can be performed without disturbing the required sequence.

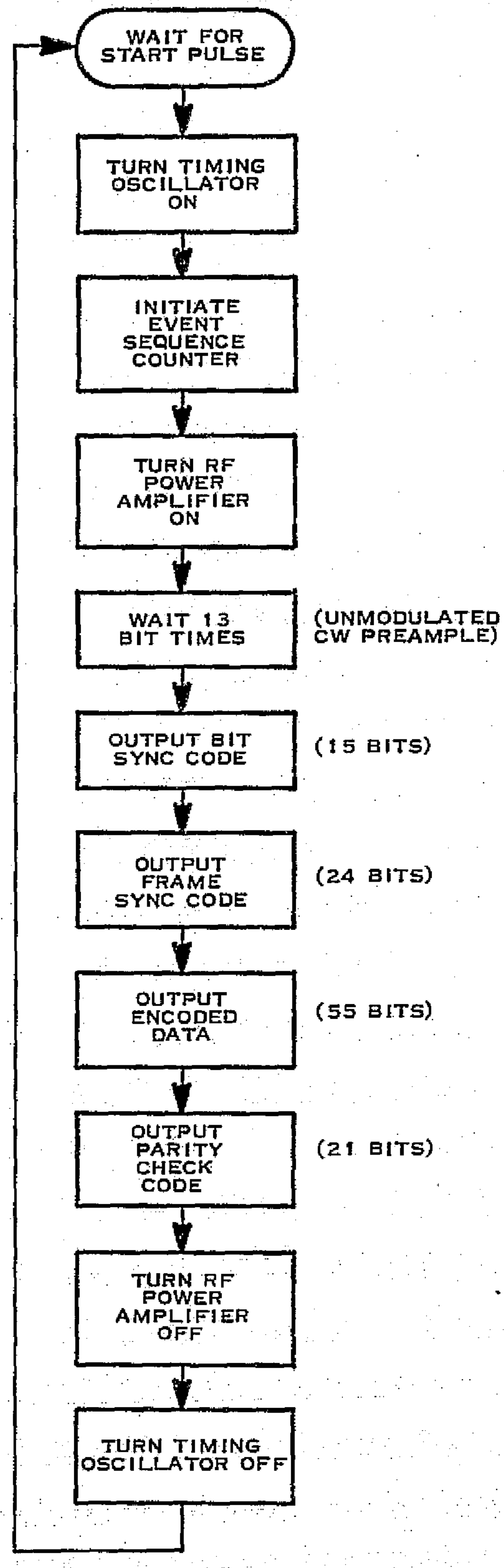

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Figure 3-6. ELT/EPIRB Event Timer Flow Chart

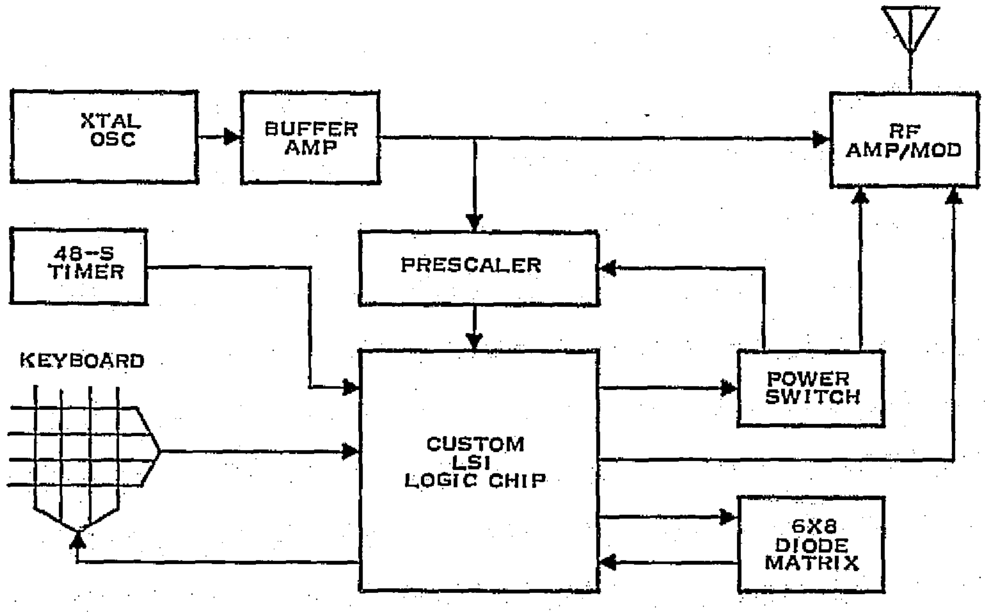

Figure 3-7. Generalized Block Diagram for Coherent MSK Generation Using a Custom LSI Chip

3.2.4.3 Logic/Timing Implementation

The previous subsection has discussed the functional requirements of the logic and timing circuits and suggested ways they can be implemented. This subsection considers three methods of implementation.

The first method suggests a custom LSI chip using I^2L technology, a discrete-part 48-second timer, and a prescaler from the RF oscillator to derive the data clock. A block diagram is shown in Figure 3-7. The RF oscillator and buffer oscillator are continuously powered once the ELT/EPIRB is activated. The 48-second timer and LSI chip are also continuously powered. The 48-second timer pulses set a latch in the LSI chip which turns on the power switch supplying power to the prescaler and RF amplifier/modulator. The sequence of events necessary to execute the transmission of a single message is begun.

The emergency code from the keyboard is entered and stored in the LSI chip. Either a debounced X-Y address mode or a scanner mode can be used to interface with the keyboard.

The ID code is stored in the 6 X 8 diode matrix. This code is transferred to the LSI chip by sequentially addressing six address lines and loading data via eight data lines.

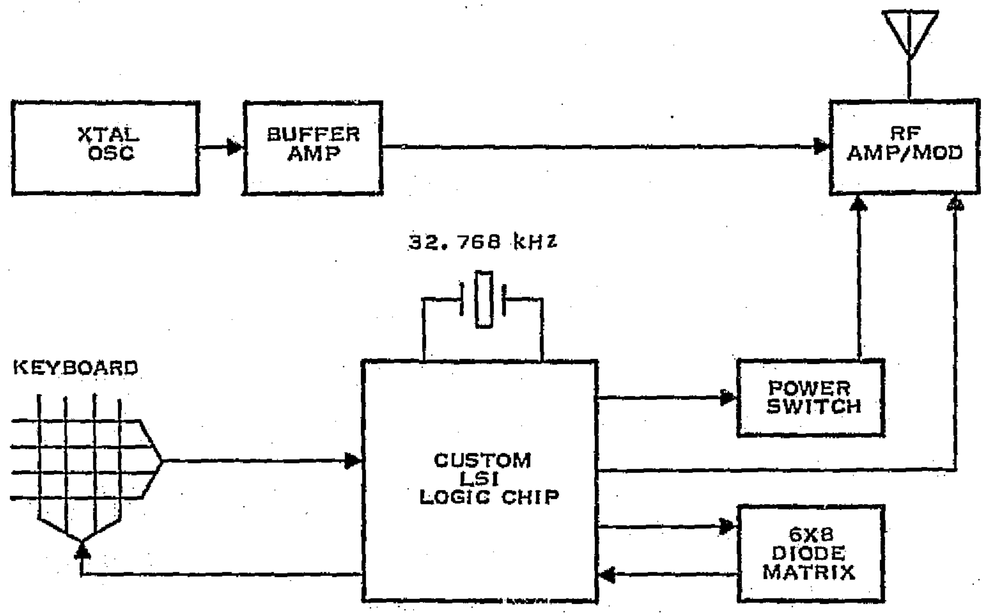

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Figure 3-8. Block Diagram for a Simple Two-Chip Logic Implementation

The LSI chip contains all logic necessary to generate the proper data which is output to the modulator at the proper rate. This circuit requires three chips in addition to the custom chip:

A 6 X 8 diode matrix

A 48-second timer oscillator

A prescaler for the clock.

This circuit meets the requirements for generating coherent MSK; i.e., the data clock is derived from the RF carrier generator. If this is not a primary design requirement, the circuitry can be simplified.

A simplified design is shown in Figure 3-8. This chip design would include circuitry for an oscillator similar to that used for watches having a 32.768-kHz crystal. It would also provide circuitry for synthesizing the pseudorandomly varying 48-second transmit-start pulse. This technique eliminates both the prescaler and the 48-second timer circuits. The only logic circuit required in addition to the LSI chip is the diode matrix. This two-chip approach would probably be the least expensive method of implementing the design in large quantities. All timing and logic functions are performed by the single LSI chip.

The final method studied for implementing the ELT/EPIRB logic is the microprocessor approach shown in Figure 3-9. A microprocessor such as the TMS 1000 containing a processor, ROM, and random access memory on a single chip may provide an economical approach to the

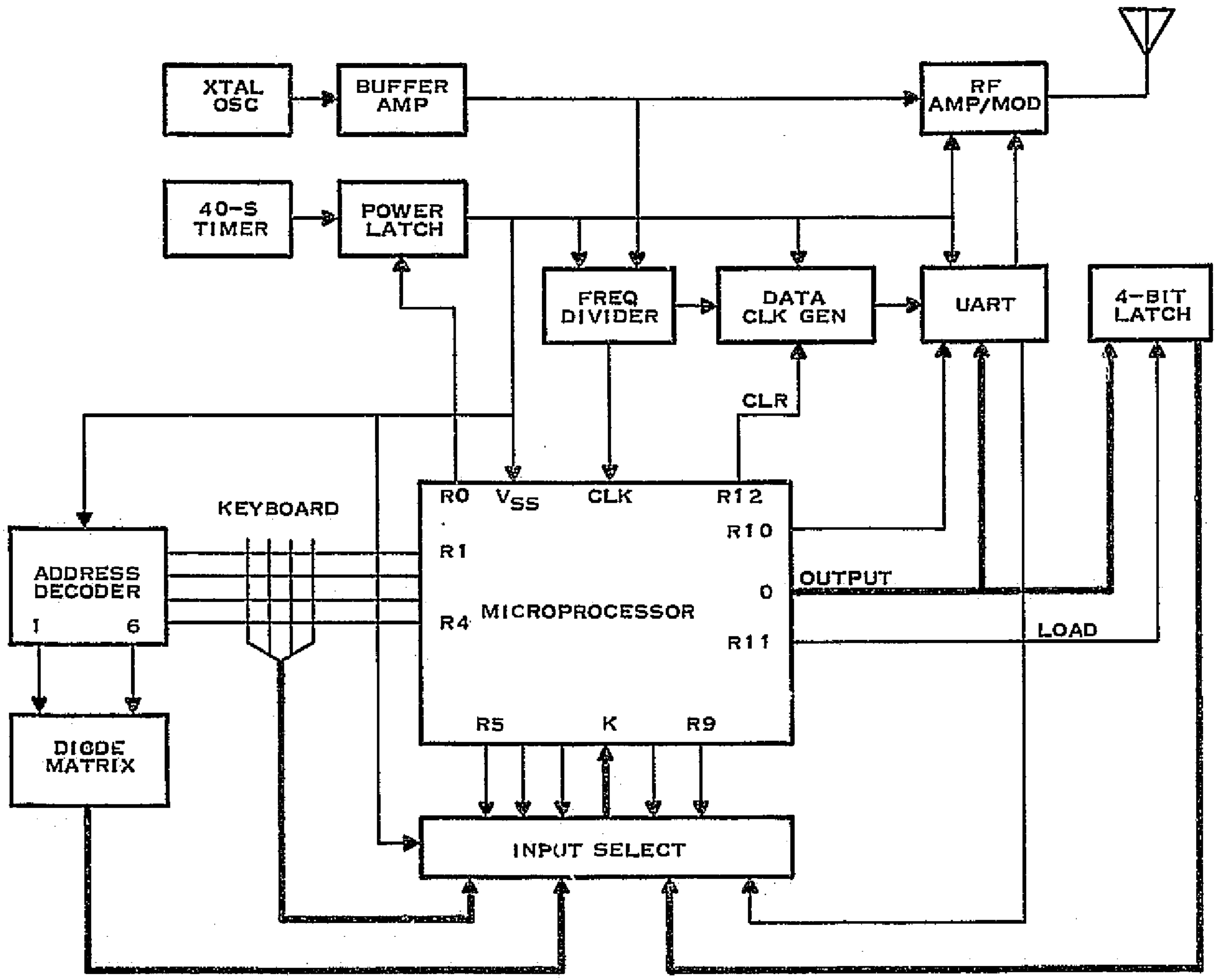

Figure 3-9. Microprocessor Approach to Logic Implementation

logic design. It is only necessary to generate a simple software program and construct a program mask. This cost (assuming a correct functional requirement is defined) will be nominal, probably no more than a few thousand dollars.

In addition to the microprocessor, a universal asynchronous receiver/transmitter (UART), a countdown chain, a 6 X 8 diode matrix, a 4-bit external latch, and a 48-second timer are required. The microprocessor is powered only during the transmit interval because of the power demand (100 to 200 mW). Since the emergency code keyboard is controlled by the microprocessor, it will be necessary to apply power to the processor when a key is pressed. The 4-bit emergency code must also be stored in an external latch.

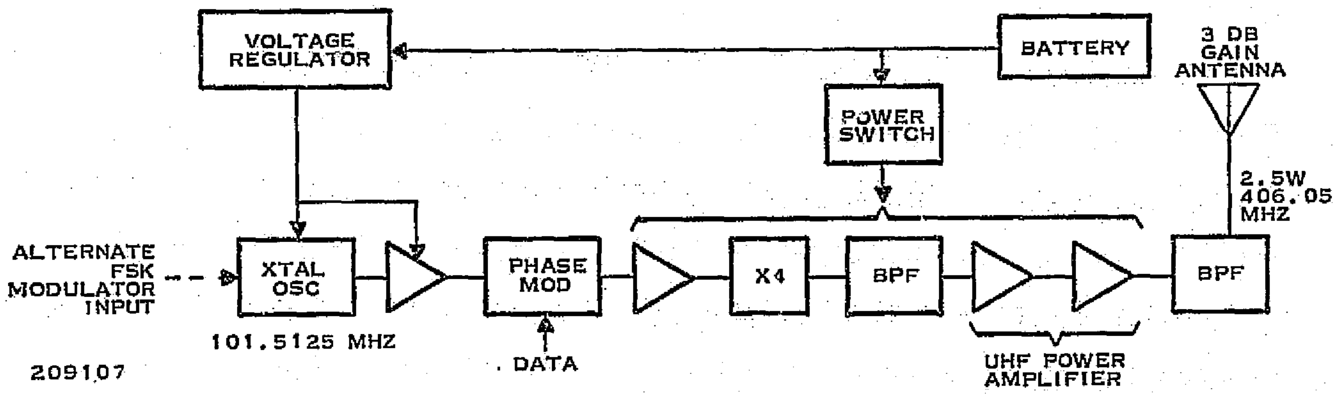

Figure 3-10. Oscillator-Multiplier-Amplifier Concept

The general operating sequence will be to execute all instructions necessary to produce the desired data message and to store these bits in the random access memory each time a transmit-start pulse is generated. The message bits will be transferred to the UART on a demand basis until the complete message is transmitted. The power latch will then be reset. The sequence will repeat each time a transmit-start pulse is generated.

Other approaches to implement the ELT/EPIRB logic are available in addition to the three described in this subsection. The choice depends primarily on economics, which may change with time. It may be that the microprocessor provides the best choice initially when production quantities are small or for the experimental demonstration phase. However, the custom chip may provide the best approach later when quantities justify the higher tooling cost.

3.2.5 RF Circuit Considerations

The 406.05-MHz output signal from the ELT/EPIRB can be generated in several ways. Two of the most feasible techniques are the classical oscillator-multiplier-amplifier concept and the power oscillator concept as shown in Figures 3-10 and 3-11.

The oscillator-multiplier-amplifier concept is straightforward and starts with a crystal oscillator at one-fourth the output frequency (101.5125 MHz), which is subsequently multiplied by four and amplified to a 2.5-watt output. The antenna gain of 3 dB results in an ERP of 5 watts. Modulation is provided by a phase modulator at 101.5 MHz. Modulation could also be done by frequency shift keying (FSK) the crystal oscillator.

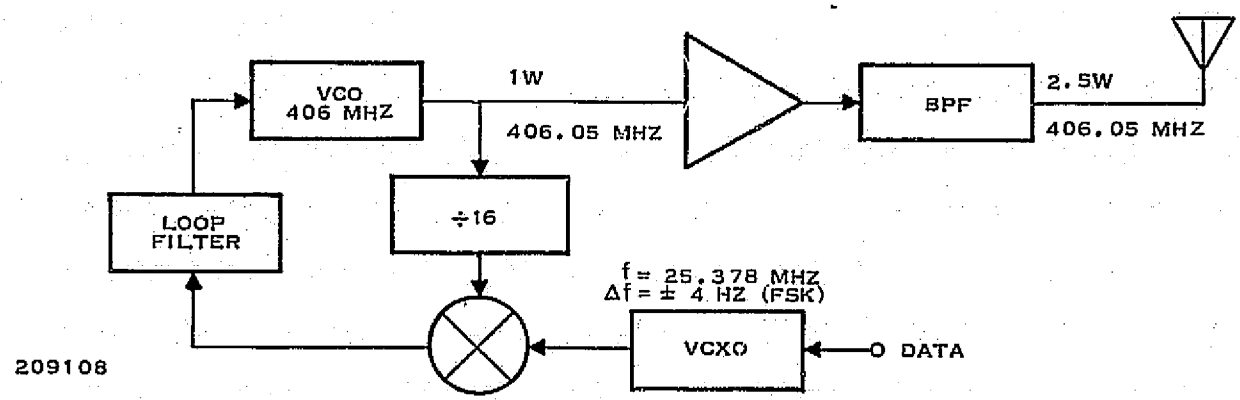

Figure 3-11. Power Oscillator Concept

The power oscillator consists of a phase-lock loop that generates a 1-watt, 406-MHz signal from a 25.378-MHz reference oscillator. The output of the phase-lock loop is then amplified and filtered to provide the required 2.5-watt output. Modulation is provided by frequency shift keying the 25-MHz crystal oscillator.

Both concepts have advantages and disadvantages, with the power-oscillator concept being the simplest and lowest cost approach. Both of these concepts are discussed in some detail below so that an insight into the design problems may be obtained. However, before starting these discussions, a brief review of the MSK concept is given.

3.2.5.1 Modulation Concept

As discussed in Subsection 2.9, the use of MSK modulation is recommended for the ELT/EPIRB because of its narrow spectral occupancy. MSK modulation is in reality FSK with a modulation index of 0.5 and with the bit rate coherent with the carrier. It can be shown[3] that the resulting signal can be visualized as a continuous-phase FSK signal or as a phase-modulated signal with the phase linearly advanced or retarded by $\pi/2$ radians during a bit interval. Figure 3-12 is a map showing which paths the phase can take. Note that at odd bit intervals the phase is always a multiple of $\pi/2$ and at even bit intervals the phase is always a multiple of π. The advantage of MSK from a spectral content is illustrated vividly in Figure 2-12.

[3]Texas Instruments Incorporated, "A Study of the Application of MSK Modulation and Chirp-Z Transforms to Data Collection, "Final Report, NASA/GSFC Contract NAS 5-22501, October 1975.

The use of Manchester encoding of the NRZ data (before MSK modulation) is recommended in Subsection 2.9 in order to ease the bit synchronization process in the satellite receiver. Manchester encoding of the data results in the phase map shown in Figure 3-13. The phase is advanced or retarded by $\pi/2$ radians during one-half an NRZ bit time, depending on whether the Manchester data is a 1 or a 0. Note that the maximum phase excursion from the starting phase is $\pm\pi/2$. In contrast, the phase of the MSK signal using NRZ data (Figure 3-12) grows without bound. A comparison of the effect of the two data trains (NRZ and Manchester) on phase shift is shown in Figure 3-14.

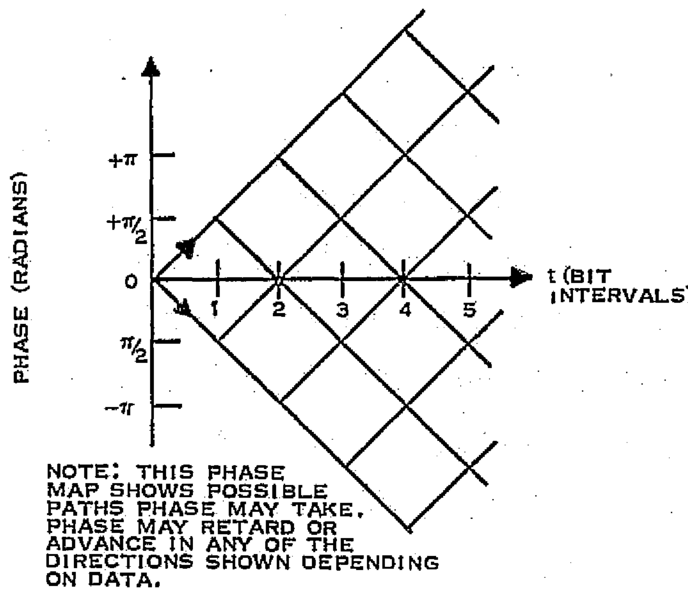

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Figure 3-12. Phase Versus Time for MSK Signal–NRZ Data

The true MSK signal as defined above can be coherently demodulated with the achievable E/N_0 performance being virtually equivalent to that for phase shift keying (PSK). This level of performance is required for a system using geosynchronous satellites. However, for the polar orbiting satellite system, considerable margin exists in the RF link and an FSK form of modulation can be used. As long as a deviation ratio of 0.5 is maintained and the FSK is phase continuous (no phase discontinuities at bit transitions), the signal spectrum is identical to that for MSK. This has been confirmed experimentally at Texas Instruments on NASA/GSFC Contract NAS 5-23599.

This fact makes it possible to use simple modulators to produce the FSK signal. The simplest modulator would use a small (fractional picofarad) varactor to deviate the 101.5-MHz

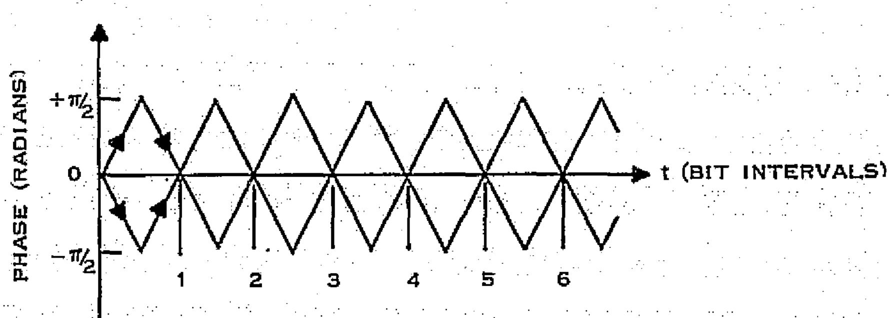

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Figure 3-13. Phase Versus Time for MSK Signal–Manchester Data

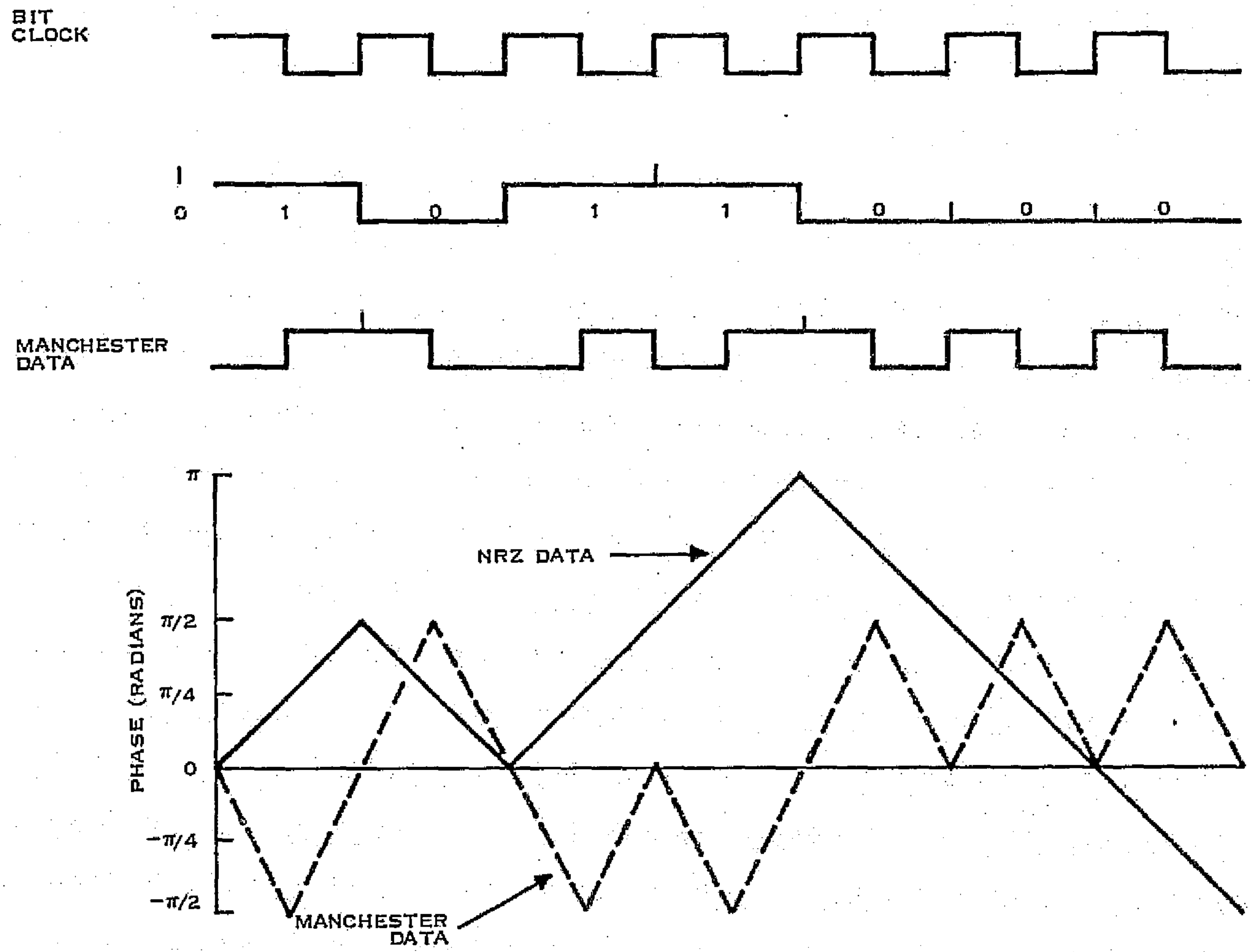

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Figure 3-14. MSK Signal Comparison of Phase Versus Time for NRZ Data and Manchester Data

crystal oscillator (Figure 3-10) by only ±16 Hz. The modulation could also be accomplished by the use of an RF phase shifter after the 101.5-MHz crystal oscillator, where the phase would be linearly retarded or advanced by $\pi/2$ radians during one bit interval (Manchester). Both of these modulators are simply realized and are duscussed further in the following subsections.

3.2.5.2 Oscillator-Multiplier-Amplifier Concept

As shown in Figure 3-15, the major elements of the oscillator-multiplier-amplifier are the following:

Crystal oscillator (101.5 MHz)

Modulator

101.5-MHz power amplifier

X4 multiplier

406-MHz power amplifier.

In addition, the battery voltage will be regulated before being supplied to the crystal oscillator. Power will be switched to all other RF circuits by a solid-state switch in order to take full advantage of the low transmitter duty cycle. Each of the major elements is discussed below.

3.2.5.2.1 Oscillator

The oscillator is the most critical circuit in the ELT/EPIRB. Its stability performance is key to the position location accuracy achieved by the overall system.

An oscillator circuit suitable for use in the ELT is shown in Figure 3-16. It is a field-effect transistor (FET) oscillator followed by a grounded-gate amplifier stage for load isolation from the remainder of the amplifier chain. The supply voltage to both these stages is supplied through an integrated circuit regulator selected for low power drain, good performance, and low cost.

The oscillator uses an FET for the good noise performance typical of that device. In addition to good AM and FM noise performance, an FET oscillator does not squedge or jump to unwanted crystal overtones as easily as one using a bipolar transistor. The oscillator has the crystal in the source circuit and has variable capacitors for adjusting the impedance presented by the following stage. A variable capacitor is also used to adjust the feedback voltage presented to the crystal to account for unit-to-unit variations in the crystal parameters.

The buffer amplifier following the oscillator stage is a grounded-gate FET design. The input impedance of this stage is high and therefore reduces the impedance transformation ratio necessary to match the high impedance drain circuit of the oscillator. The isolation provided by this stage and the pad following it will help reduce frequency modulation caused by impedance variations of the phase modulator, and the succeeding amplifier is slightly overdriven so that its RF power output will be constant as the oscillator power output varies because of temperature changes. The next amplifier stage will thus be provided a constant RF drive level over temperature extremes to stabilize the amplifier/multiplier chain performance.

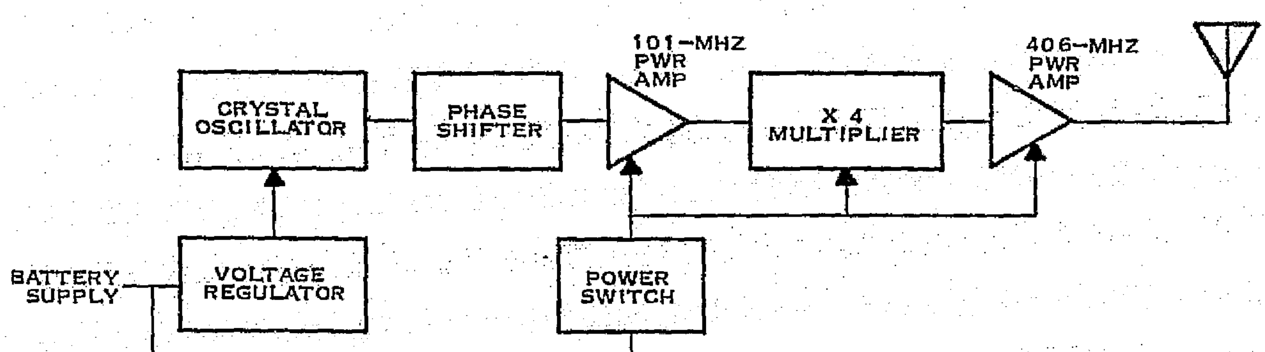

Figure 3-15. Major Elements of the Oscillator-Multiplier-Amplifier Concept

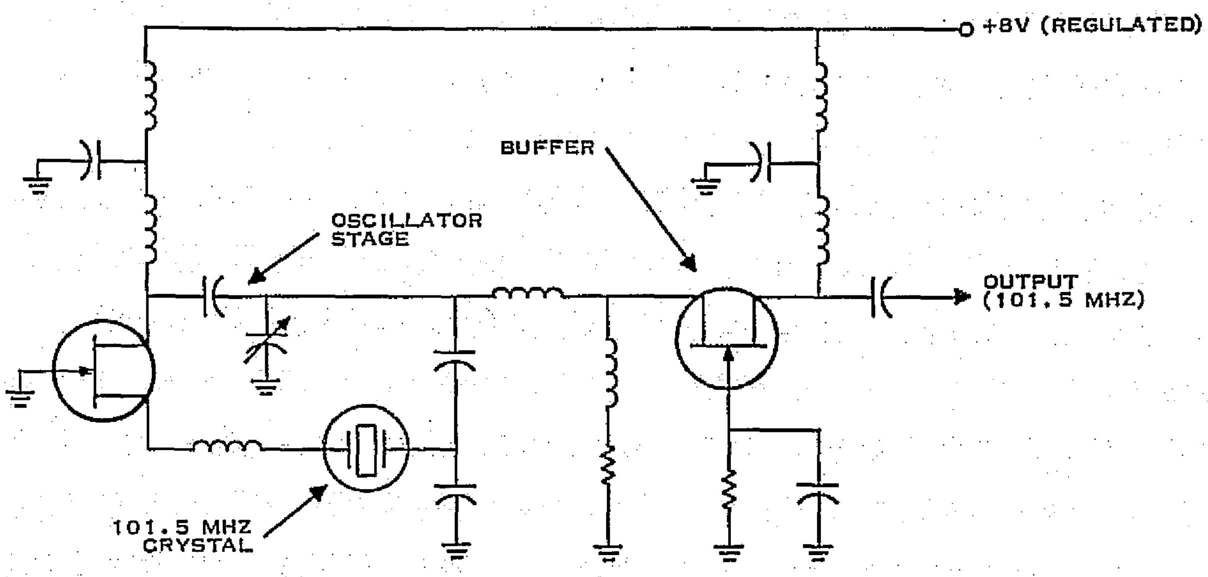

Figure 3-16. 101.51250-MHz Crystal Oscillator

In addition to the electrical design considerations just discussed, the amplifier will be grounded to its shielding cavity and will have B+ filtering to prevent the multiplier or power amplifier from generating noise currents in the B+ or ground circuits of the oscillator. Careful packaging of the oscillator will be necessary to prevent short-term thermal gradients and attendant frequency drifts. When the ELT is designed, considerable effort will be required to ensure stability and reproducibility.

3.2.5.2.2 Modulator

As previously discussed, two low-cost modulator techniques are suitable for the ELT/EPIRB, the FSK modulator and the phase shift modulator.

The FSK modulator can easily be implemented by adding a varactor circuit to the crystal oscillator as shown in Figure 3-17. The input data would be a positive voltage for a logic 1 and a zero voltage for a logic 0. This data input voltage is attenuated and then applied as bias to varactor CR1. The change in voltage caused by the data varies the bias voltage on the varactor and results in a change of capacitance to ground on the FET drain. A fractional picofarad voltage-variable capacitor will produce the desired FSK of ±16 Hz at 101.5 MHz. Subsequent multiplication by 4 results in a deviation of ±64 Hz at 406 MHz, which is proper for a deviation ratio of 0.5. Continuous phase operation (during bit transitions) is provided by the high Q of the crystal.

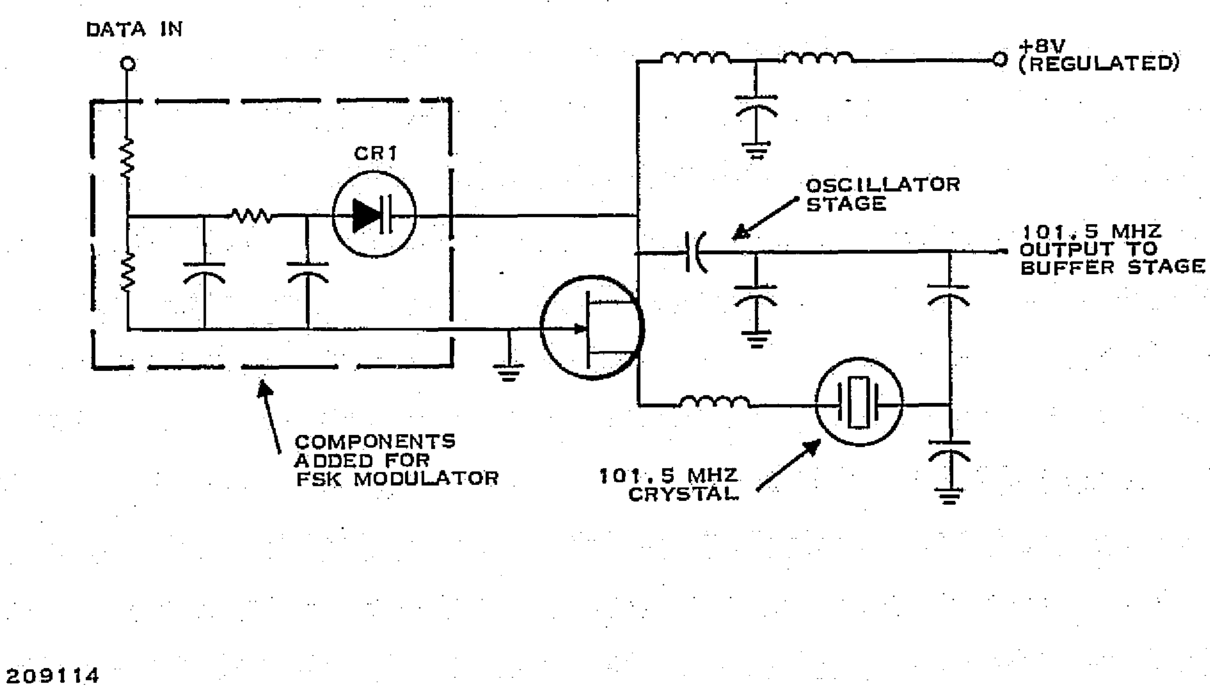

Figure 3-17. Crystal Oscillator With Added Components for FSK Modulation

The FSK modulator is very simple; its only drawback may be that it can possibly degrade the crystal oscillator stability. However, it is believed that the small deviation required (±16 Hz) will not degrade the oscillator's inherent stability.

An alternate modulator can be made using an RF electronic phase shifter. These devices are commercially available and consist of a quadrature hybrid in conjunction with an inductor capacitor (LC) network to produce phase shift. Varactor diodes are used in the LC network so that a dc bias voltage can be used to vary the capacitance and, in turn, produce phase shift. Figure 3-18 shows the components of the phase shifter.

The phase shifter can be designed for use in the 101.5-MHz chain or in the 406-MHz chain of the transmitter. However, the optimum frequency is the lowest possible for the least phase shift. Placing the phase shifter at 101.5 MHz reduces the total phase shift requirement to only one-fourth that required at 406 MHz. As discussed in Subsection 3.2.5.1, the required phase shift is $\pm\pi/2$ radians, or 90 degrees, at 406 MHz. The phase shift at 101.5 MHz is only $\pm\pi/8$ radians, or ±22.5 degrees.

As shown in Figure 3-19, the transfer characteristics of a typical phase shifter are linear on the lower portion of the curve. The required ±22.5 degrees of phase shift can be accomplished by driving the phase shifter with a ramp of voltage that linearly increases for logic 1 data and decreases for logic 0 data. The slope of the driving waveform would be such that it would produce a phase change of 22.5 degrees in one bit time (Figure 3-14). A phase shift of 22.5 degrees can be obtained with an input change of less than 2 volts so that the 8-volt regulated supply voltage can be used to power the driving circuits.

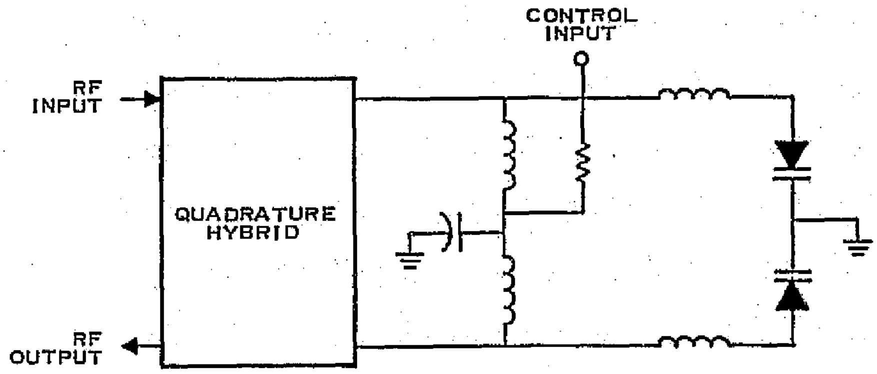

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Figure 3-18. Phase Shifter

The phase shift modulator has the advantage over the FSK modulator of not disturbing crystal oscillator stability. Its main disadvantage is that it is slightly more complicated.

Both the FSK modulator and the phase shift modulator can be modified to produce true coherent MSK by deriving the bit clock from the 101.5-MHz oscillator output. This can be done by counting down the 101.5-MHz oscillator, by using high-speed logic, to the bit rate clock frequency. Since this counter has to be powered only during the transmitter on-time, the average power consumption would be only a few milliwatts.

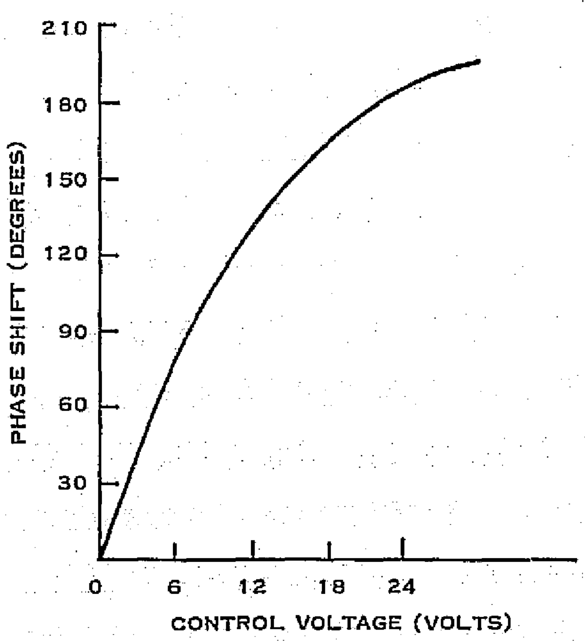

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Figure 3-19. Typical Phase Shift Characteristics

3.2.5.2.3 VHF Amplifier

The 101.5-MHz VHF RF amplifier provides amplification of the signal from the phase shifter and drives the X4 multiplier. Three common emitter stages (two class A and one class C) will provide an overall gain of over 30 dB and a drive to the multiplier of approximately 1.375 watts. To ensure stability, RF pads will be used at the input of the amplifier and between the output stage and the multiplier. Adjustable components will be used as necessary to allow rapid alignment of the amplifier to account for production tolerances and to minimize test costs. RF feedback will be incorporated to stabilize the transistor parameters as temperature or signal levels are varied and to account for transistor production tolerances.

3.2.5.2.4 Frequency Multiplier

The frequency multiplier multiplies the 101.5-MHz signal to the transmit frequency of 406 MHz. The most straightforward type of multiplier is a step recovery diode, which is used extensively and is economical and reliable. As shown in Figure 3-20, isolation between the basic X4 multiplier and adjacent circuits is achieved through the use of 2-dB pads at the input and output ports. These pads are necessary to prevent parametric oscillations between the multiplier and the VHF amplifier on the input side and between the multiplier and the bandpass filter on the output side. The last circuit making up the multiplier is a bandpass filter used to reduce the unwanted multiplier harmonics. It will use high-Q parts for low loss and stability and will have two adjustable capacitors so that it can be rapidly aligned. The output power will be approximately 0.17 watt to the UHF power amplifier.

At Texas Instruments, a multiplier such as this has been tested from –55° to +100°C with good performance results. The keys to the stable performance were the proper choice of component types and component values, proper circuit layout and grounding, and the padding used to prevent parametric oscillations.

3.2.5.2.5 UHF Power Amplifier

The output of the frequency multiplier is amplified by the UHF power amplifier to provide the output transmit level.

This two-stage ampliifer has an RF pad between the first stage and the output of the multiplier to achieve RF stability and prevent generation of spurious signals. It is followed by a bandpass filter to reduce the second harmonic power levels to meet the ELT requirements of –50 dB for spurious signals. The power transistors used in these two stages are operated in stress regions where they will accept infinite VSWR loads. Common emitter configurations are incorporated for RF stability and to reduce the generation of spurious signals caused by load and temperature variations. These last two design considerations will ensure stable, reliable operation in the field and will prevent the transistors from being stressed during factory alignment.

These stages are operated Class C for maximum power supply efficiency and a switched B+ is supplied to the transistors to prevent any chance of oscillation caused by extreme conditions in the field. Without a switched power supply, even a Class C amplifier can be made to oscillate with a high VSWR load after it receives a pulse of signal to start RF currents in the circuits. This oscillating condition would rapidly drain the batteries in the ELT/EPIRB.

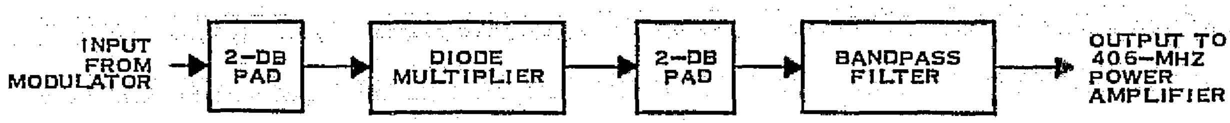

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Figure 3-20. Times-Four Multiplier Block Diagram

3.2.5.3 Power Oscillator Concept

The power oscillator concept has been previously mentioned as an alternative technique for the ELT/EPIRB (Figure 3-11). The 406-MHz signal is derived using a PLL that is locked to the 16th harmonic of a reference crystal oscillator at 25.378 MHz. Modulation is provided by operating the crystal oscillator as a VCXO and deviating it only ±4 Hz. Operating the crystal at 25 MHz, as opposed to 101.5 MHz, should result in a more stable and producible oscillator.

The phase-lock loop (PLL) can use readily available integrated circuits for the VCXO, the divide-by-16 circuit, the phase detector, and the loop filter. These circuits are now available from several manufacturers. Power to all PLL components would be switched off between transmissions. Power to the 25-MHz VCXO would be left on continuously.

The output of the PLL would be at a substantial 1-watt power level at 406 MHz eliminating the frequency multiplier in the other concept. Amplification and filtering of the signal would be accomplished as before but only one Class C UHF power amplifier stage would be required.

The advantages of the power oscillator are summarized below:

- Improved crystal oscillator stability
- Better power stability over time/temperature
- Fewer parts
- Less assembly time
- Less alignment time
- Lower cost
- Potentially lower power consumption and increased battery life.

The disadvantage in using the power oscillator concept is that it may be more difficult to reduce the noise close in to the carrier because of the high multiplication rate. However, the design of similar power oscillators for radar equipment has shown that reduction of close-in noise is possible.

It appears that the power oscillator has the advantage over the oscillator-multiplier-amplifier concept. In order to make a choice, both approaches need to be designed and tested.

3.2.5.4 121.4 MHz Beacon

As shown in Figure 3-21, the 15-mW beacon transmitter will consist of a crystal oscillator and a buffer amplifier. Power for the oscillator will be obtained from the 8-volt regulator used in the main ELT/EPIRB transmitter section.

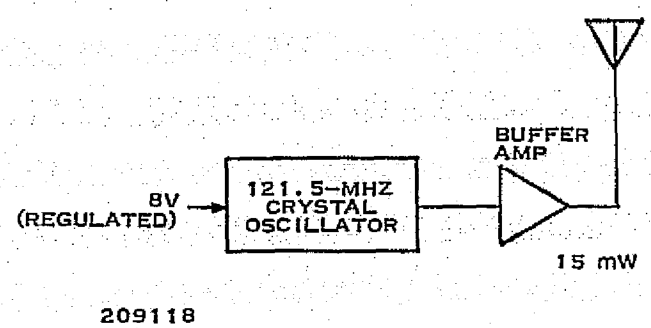

Figure 3-21. Beacon Transmitter Block Diagram

The circuit will be almost identical to that for the 101.5-MHz crystal oscillator with FETs being used for both the oscillator and buffer. Since no frequency multiplication is involved, very little filtering of the output will be required.

Power will be applied continuously to the beacon. Because of the low (15-mW) power output level, the power consumption is quite small, but it is still an appreciable part of the total power consumption.

3.2.6 Power Consumption Estimates

As previously mentioned, the battery pack for the ELT/EPIRB will consist of five lithium D-cells in series for the RF circuits. Power for the logic IC will be provided by a separate lithium cell. Since the logic power consumption is very low, a D-cell with a rating of 3.5 ampere-hours (AH), at –40°C, will be more than adequate.

In the RF circuitry, power must remain on the crystal oscillator and its buffer continuously in order to achieve stability. Although the power consumption of the oscillator and its buffer are small (24.5 mA), this power is only approximately one-half the power consumed by the ELT/EPIRB.

Tables 3-3 and 3-4 contain the power estimates for the two ELT/EPIRB concepts. Power consumption is broken down into two categories, continuous load and switched load. The duty cycle of 1 in 40 is used to adjust the switched peak current to an average current.

Discussions with lithium battery manufacturers show that the rating of the 10 AH D-cells at the peak current of approximately 1.0 A should be considered to be 6 to 7 AH at +25°C and only 3 to 4 AH at –40°C. At +50°C, the ratings are between the +25° and –40°C ratings.

Hours of operation are obtained by dividing the AH rating by the average load current at +25°C. The operating time for both concepts exceeds 100 hours. At –40°C, both concepts have about 70 hours of operating time. Keep in mind that the above estimates are conservative; with careful design, the power consumption can be lowered and the 100-hour goal met even at –40°C.

3.2.7 ELT/EPIRB Packaging

In keeping with the considerations of reliability, performance stability, and low cost, the ELT/EPIRB must be packaged in a rugged, but not expensive, enclosure. The case must be sturdy enough to survive shock environments and impervious to corrosion from the lithium batteries. A separate battery compartment must be provided in order to isolate the PCBs from the batteries and minimize corrosion.

With the advances over the past few years in plastics, the use of a polycarbonate plastic such as nylon appears most appropriate for the ELT/EPIRB case. A plastic case can be molded in two or three parts at a minimum cost and provide a sturdy, durable enclosure.

In those cases where a keyboard is included, the keyboard will be on the top surface of the enclosure. An adhesive metal background such as those in use for low-cost calculators can be used to label the keys. The keyboard area and other openings can be configured to provide sealed operation.

TABLE 3-3. ELT POWER ANALYSIS–OSCILLATOR-MULTIPLIER-AMPLIFIER CONCEPT

I. Continuous Load (all currents in amperes)

Regulator	0.0045
Oscillator/buffer	0.0100
Beacon oscillator/amplifier	0.0100
Total	0.0245

II. Switched Load

Output amplifier	0.560
Driver	0.156
Amplifiers	0.236
Power switch	0.100
Total	0.952

$$\text{Average current} = \frac{0.952}{40} = 0.0238 \text{ ampere}$$

III. Operating Time

Total average current = 0.0245 + 0.0238 = 0.0483 ampere

Battery rating

At +25°C 6 to 7 AH (6.5)

At –40°C 3 to 4 AH (93.5)

Operating time

$$\text{At } +25°\text{C} = \frac{6.5 \text{ AH}}{0.0483 \text{ A}} = 135.6 \text{ hours}$$

$$\text{At } -40°\text{C} = \frac{3.5 \text{ AH}}{0.0483 \text{ A}} = 72.5 \text{ hours}$$

TABLE 3-4. ELT POWER ANALYSIS–POWER OSCILLATOR CONCEPT

I. Continuous Load (all currents in amperes)

Voltage regulator	0.0045
VCXO	0.0100
Beacon oscillator/amplifier	0.0100
Total	0.0245

II. Switched Load

Output amplifier	0.560
VCO	0.250
÷16	0.060
Phase detector	0.040
Loop filter	0.020
Power switch	0.100
Total	1.030

$$\text{Average current} = \frac{1.030}{40} = 0.026 \text{ ampere}$$

III. Operating Time

Total average current = 0.0245 + 0.026 = 0.0505 ampere

Operating time

$$\text{At } +25°\text{C} = \frac{6.5 \text{ AH}}{0.0505 \text{ A}} = 128.7 \text{ hours}$$

$$\text{At } -40°\text{C} = \frac{3.5 \text{ AH}}{0.0505 \text{ A}} = 69.3 \text{ hours}$$

Printed circuit boards will be used for the logic and RF circuits for the ELT/EPIRB. Two PCBs are envisioned: one for the power amplifier and one for the other functions.

Figure 3-22 shows the two PCBs. The figures is a cross-sectional view depicting the two housing cavities separated by a wall for RF isolation. The cavity containing the logic, the oscillators, the low-frequency amplifiers, and the multiplier is separated from the high-frequency power amplifier cavity to provide shielding so that low-frequency subharmonics are not radiated at the high-frequency port. Some additional shielding may be required between the beacon and the 101-MHz oscillator and multiplier. If so, an additional wall may be riveted to the oscillator cover. Figure 3-22 also shows the PCBs on which the circuit components will be mounted. The use of stamped aluminum parts that can be assembled quickly and PCBs that provide interconnection paths ensures inexpensive parts and lowers the cost of mounting the top assembly parts. Flow solder techniques will be used during the PCB assembly process to reduce test times and labor costs.

The ELT/EPIRB should be approximately 5 by 5 by 6 inches in size and weigh approximately 3 pounds. Units without a keyboard would be slightly smaller (5 by 4½ by 6 inches).

3.2.8 ELT/EPIRB Costs

Low cost for the ELT/EPIRB is one of the key goals in the search and rescue system where voluntary use of the transmitters is anticipated. With considerable design effort, the ELT/EPIRB can be optimized for producibility and low cost and still maintain the required level of performance. The list of steps to minimize labor is almost endless, but some of the most obvious are as follows:

PCBs with as many inductors printed (part of the etch) as possible

Flow solder

Molded case parts

Automated test equipment

Integrated circuits used where possible.

A brief analysis of both the ELT/EPIRB concepts was done assuming that 10,000 units would be built in 1 year. It was also assumed that a pilot production run had already been completed so that all design errors would be corrected and assembly and test procedures optimized. The keyboard was assumed to be included. Deletion of this feature would save a few dollars. Amortization of preproduction costs was not included in the estimate.

The cost estimates should be treated only as rough estimates. In 1977 dollars, the following cost estimates (actually sales prices) resulted:

Oscillator-multiplier-amplifier	$400–500
Power oscillator	$300–450

These prices may vary depending on the manufacturer's overhead/pricing structure as well as the distribution arrangements.

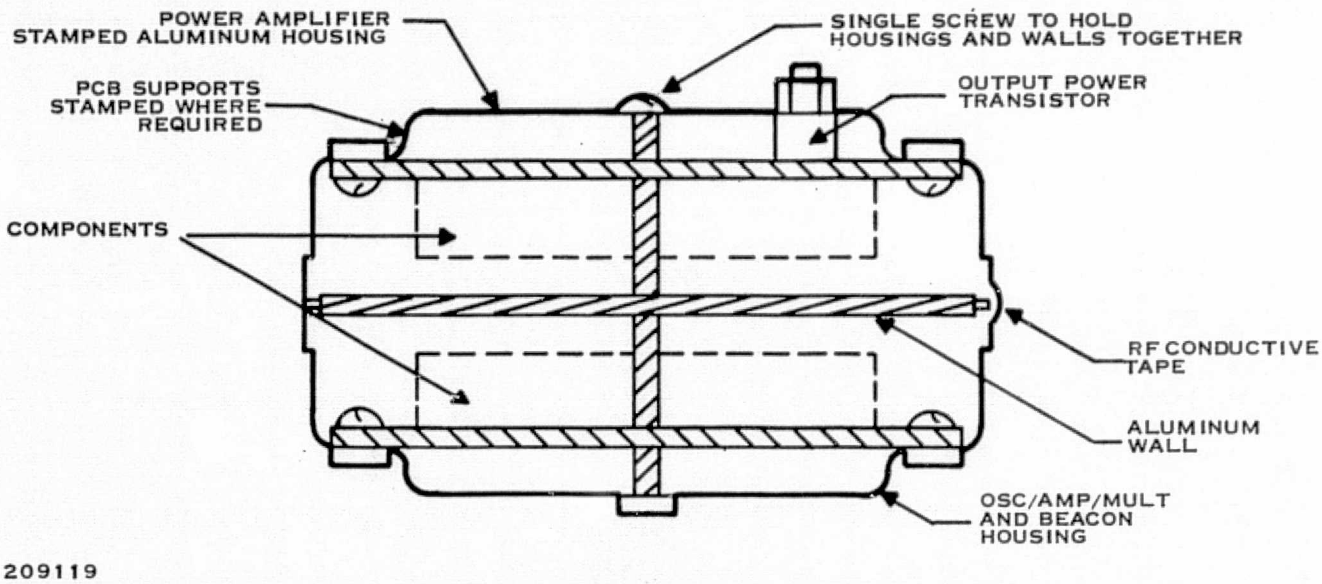

Figure 3-22. RF Module Cross Section

In summary, a conservative design estimate has shown that the price of the ELT/EPIRB should be in the \$300 to \$500 range. With rigorous design, the complexity of the ELT can be reduced resulting in even lower costs.

SECTION 4
SPACECRAFT EQUIPMENT

The type of equipment to be used on board a satellite for detecting and locating ELTs and EPIRBs is a function of the type of satellite and the amount of signal processing to be done by the satellite. If no onboard processing is to be done (i.e., "bent-pipe" technique), the satellite equipment need be little more than a transponder system. The satellite would receive the broadband spectrum of emergency transmissions, translate them to another frequency, and relay them to a ground station. The satellite position and time information would also be transmitted. The entire signal processing would be accomplished on the ground. The major disadvantage of this technique is that the ground station and the ELTs/EPIRBs must be mutually visible. Such a restriction precludes worldwide coverage with a low-orbiting satellite. Therefore, no further discussion of the bent-pipe technique is included in this report.

The equipment required for the low-polar-orbiting satellite system is not the same as the equipment required for a geostationary satellite system. For example, it is not possible to use a doppler technique for the geostationary satellite system. Conversely, it is not practical to use an interferometer technique for a low-orbiting satellite system. For reasons such as these, the equipment required for each type of satellite system is considered separately in this section.

4.1 LOW-POLAR ORBITER SATELLITE EQUIPMENT

The use of satellites in low near-polar orbits in a system for detecting and locating emergency transmitters is discussed in Section 2. The only positioning technique that appears reliable for this system is the differential doppler method used on the random-access measurement system (RAMS). A block diagram of the satellite equipment is shown in Figure 4-1. The receiver block represents the circuitry that accepts the signal from the antenna, filters and amplifies the desired signal, converts the received spectrum to an IF spectrum, and provides a broadband AGC function. The receiver performs the RF and IF signal-processing function.

The search and assignment circuitry is basically a spectrum analyzer. The incoming received spectrum (100 kHz) is analyzed to determine the frequency of any signals contained within the bandwidth of interest. Once a signal is detected and its frequency determined, the assignment logic finds an available phase-lock loop (PLL) receiver and slews its VCO to the approximate frequency of the detected signal. The search function can be implemented by at least two methods.

The PLL receivers are essentially phase-lock loops that lock to and track the incoming signal. Once phase lock is established, the signal can be demodulated. The resulting data are available for further processing. In addition, the exact frequency of the received signal is determined by precisely measuring the frequency of the PLL VCO. The frequency and time data are combined with the recovered data, properly formatted, and returned to an earth station.

The data recovery circuitry is dependent on the type of data chosen to be transmitted by an ELT or EPIRB and the type of data to be returned to earth. The basic function of this circuitry, however, is to properly identify the transmitter and its emergency code and combine

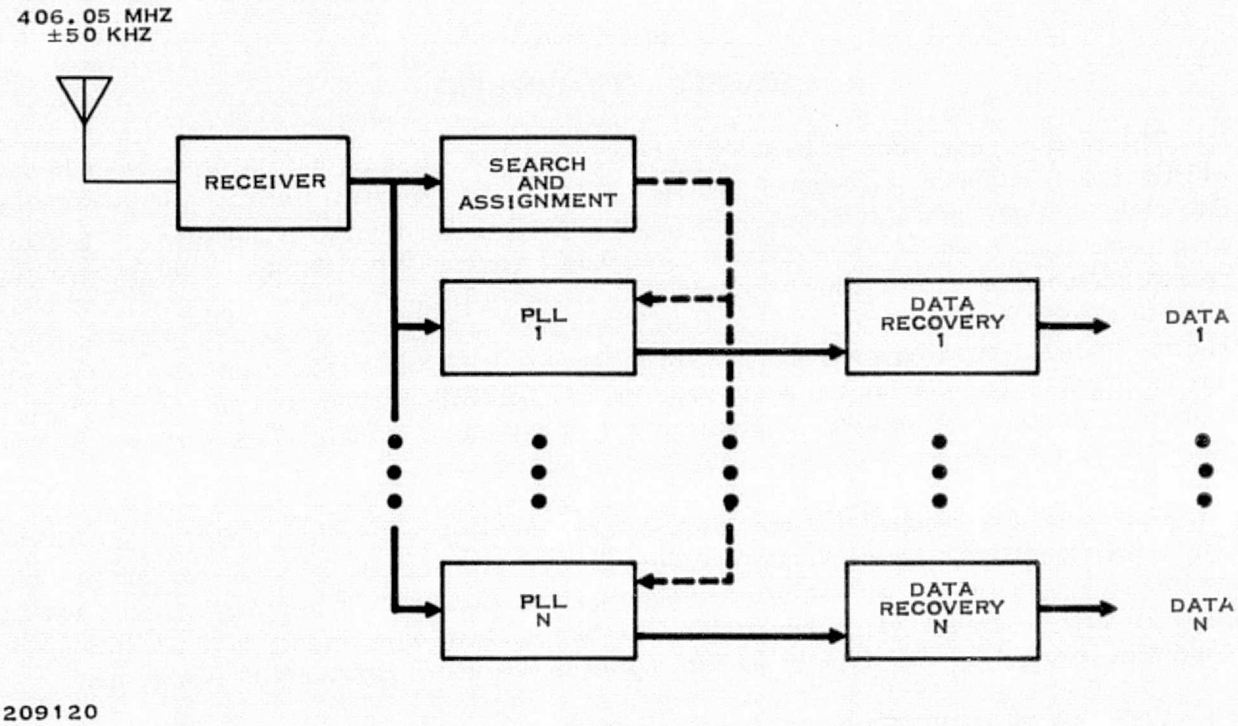

Figure 4-1. Generalized Block Diagram of Common Satellite Receiver and Data Recovery Circuitry

this information with the corresponding frequency and time measurement for relay to a ground station. A major function of this circuitry is data compression.

A final requirement of the satellite equipment is to provide for reception of emergency beacons operating on 121.5 MHz and 243 MHz. (The details of the equipment requirements for this function are discussed in a separate study.) This reception requirement is necessary to provide ELT detection for current ELT designs. A very general discussion of this equipment is given in Subsection 4.1.6. The emphasis is on the relationship of the 121.5/243-MHz equipment to the 406-MHz equipment.

4.1.1 Receiver

The receiver block diagram is shown in Figure 4-2. The purpose of the receiver is to translate the 406.0- to 406.1-MHz spectrum to a baseband spectrum of 0 to 100 kHz, of which only 6 to 94 kHz will be processed. (The baseband spectrum is required by the search circuitry for maximum frequency resolution.) The receiver uses a dual conversion process to reach baseband. However, the output of the first IF is used by the PLL receivers. The system performance requires a minimum 3-dB noise figure for a geostationary satellite system. A 3-dB noise figure was also assumed for the low-polar orbiting satellite receiver, but this is not an absolute requirement.

The RF preselector and preamplifier accepts signals from the antenna, provides some gain for the desired signals, and attenuates undesired signals (especially the image frequencies). The

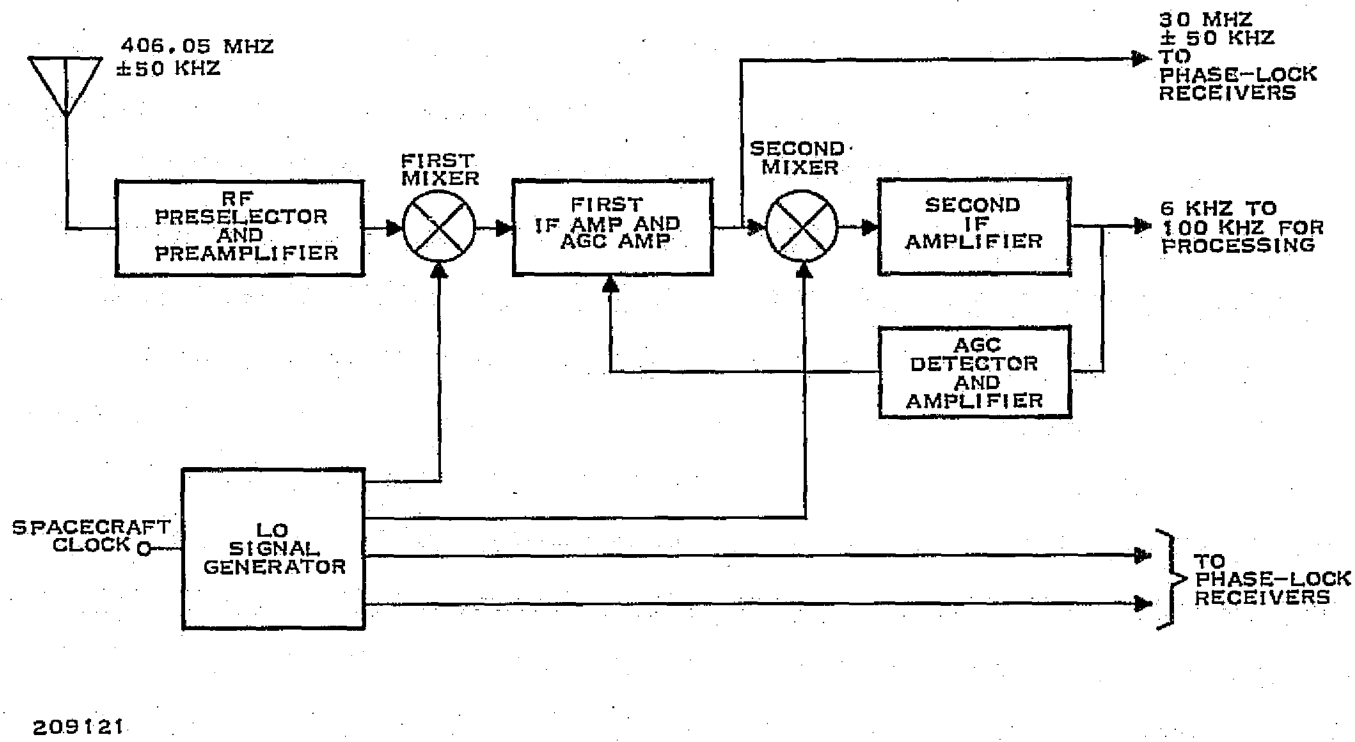

Figure 4-2. Satellite Receiver Block Diagram

design of the preselector and preamplifier circuitry is critical since it will determine the noise figure of the receiver. In addition, the design must provide a wide linear dynamic range; i.e., the preamplifier must not introduce any intermodulation distortion products over the range of signal levels presented to the amplifier.

The frequency translation must be done accurately so that the doppler shift (measured by the signal processor after the conversions) can be accurately determined. Therefore, the local oscillator injection signals should be derived from a common stable source. This implies some type of frequency synthesis for LO signal generation. Four such signals are needed. The frequencies chosen should provide good image signal rejection and a practical IF bandpass filter capable of good selectivity. The mixers should produce very low cross-modulation components and should have good noise figures.

The first IF amplifier should have a bandpass sufficient to pass the entire 100-kHz ELT/EPIRB spectrum, but reject all out-of-band signals. It should also have variable gain capability, yet produce very low intermodulation distortion. The total gain control should be a minimum of 30 dB, and preferrably 40 dB. The gain control voltage is derived by the automatic gain control (AGC) detector/amplifier circuit. The time constant should be selected to provide operation consistent with the overall receiver-processor design. The general design should have a fast-attack, slow-release characteristic, but the time constant should not permit a change in gain over at least two analyzing periods.

The second mixer should have basically the same characteristics as the first, but it must have low intermodulation distortion with the input levels expected from the first IF amplifier. This mixer design can be critical in determining the maximum dynamic range of the search unit, particularly if the output level of the first IF is high. (For this reason, the output level of the first IF should be kept low.) The total intermodulation products produced in the receiver should not exceed −40 dB with respect to the maximum expected desired signal output, or the search unit will "find" numerous false signals.

The second IF amplifier should have low intermodulation distortion, high gain, and good rejection of the higher mixer products. While this circuitry has been termed a second IF, it may be considered a baseband amplifier. The amplifier should have flat gain over the range of 6 to 94 kHz. The combined bandpass characteristics of the two IF amplifiers should produce a minimum attenuation of 40 dB for output frequencies above 100 kHz.

The first IF output of the receiver should provide a low source impedance and a moderate signal level capable of driving the required number of phase-lock receivers. As an alternate, multiple outputs may be desirable. The second IF output must interface with the search unit. The key parameter for the entire receiver design is linearity. The first receiver gain stage must be optimized for good noise figure, but the remainder of the circuitry (particularly the mixers) must be designed for good linearity to prevent the generation of false signals.

4.1.2 Search and Assignment Circuitry

The circuitry that detects the presence of signals and provides a coarse frequency measurement of these signals is basically a spectrum analyzer. As briefly discussed in Subsection 2.9.1, there are two basic techniques for implementing this function. The chirp-Z transform approach is described in general terms; a detailed description of this method is presented in this subsection. The second practical approach, the time compression spectrum analyzer method, is also described.

The RAMS used the time compression spectrum analysis technique. This approach obtains samples of the time domain baseband received signal spectrum, stores a sufficient number of these samples to provide the required resolution and bandwidth, and recirculates the stored samples at a much higher rate than the rate of acquisition. The resulting signal is compressed in time and expanded in frequency, enabling the spectrum analysis to be performed at a higher frequency and in a correspondingly shorter time. Ideally, the time compression should be sufficient to enable the analysis to be accomplished in real time, i.e., the analysis cycle should occur within the period of one sample acquisition time. For the resolution of the 100-kHz spectrum into 250 spectral components, this time is 2.5 ms. This means that the ratio of acquisition time to analysis time is 250:1. Since the sample rate must be approximately 300K samples/second, the output word rate must be $250 \times 3 \times 10^5 = 7.5 \times 10^7$ or 75 MHz. A 75-MHz clock rate is possible but would require a prohibitive amount of power and board space to provide the logic necessary to store the 750 digital samples. A major problem would be the construction of a digital-to-analog converter (DAC) that would operate at 75 MHz. Therefore, it seems unlikely that a time compression ratio of 250:1 is currently realizable. A more realistic ratio would be 10:1 or possibly 16:1. A compression rato of 16:1 would require logic speeds of approximately 5 MHz. Such a system using digital storage would require a DAC with a conversion time less than 200 ns. For a dynamic range of 40 dB, the minimum word length per sample would be 7 bits. For 256 spectral lines and a minimum sampling rate of 300 kHz, the amount of storage required is 768 seven-bit words or 5,376 bits. This is realizable, but would require substantial circuitry and board space. (In fact, the total storage capacity required is twice this, or 10,752 bits, since one set of registers must be reading data while another set is writing data.) The digital approach also requires a sample-and-hold circuit, an A/D converter (ADC), and a D/A converter (DAC).

Another technique is possible using recent advances in analog CCD shift register technology. This approach eliminates the need for the sample-and-hold circuitry and the A/D and D/A conversions. It also eliminates the requirement for storing 7-bit words; instead, 768 analog samples are stored in each of the two sets of CCD shift registers. This has the obvious advantage of reduced circuitry and onboard space utilization.

Using either the digital storage or the analog storage technique to achieve a 16:1 time compression ratio would require 16 samples to resolve the input spectrum into 256 lines. Since the minimum sample acquisition time is the reciprocal of the resolution frequency, 1/400 Hz or 2.5 ms, the total time to analyze the 100-kHz spectrum is 16 X 2.5 ms or 40 ms. In order to have two tries at detecting a signal, the minimum unmodulated carrier duration is 80 ms with 100 ms preferred. The spectrum analysis using a narrowband filter following a mixer with a stepped VCO injection signal is the same for either system. A block diagram of the digital storage system is shown in Figure 4-3; a block diagram for analog CCD storage is shown in Figure 4-4. Both are for a time compression ratio of 16:1 and a 256-line analyzer.

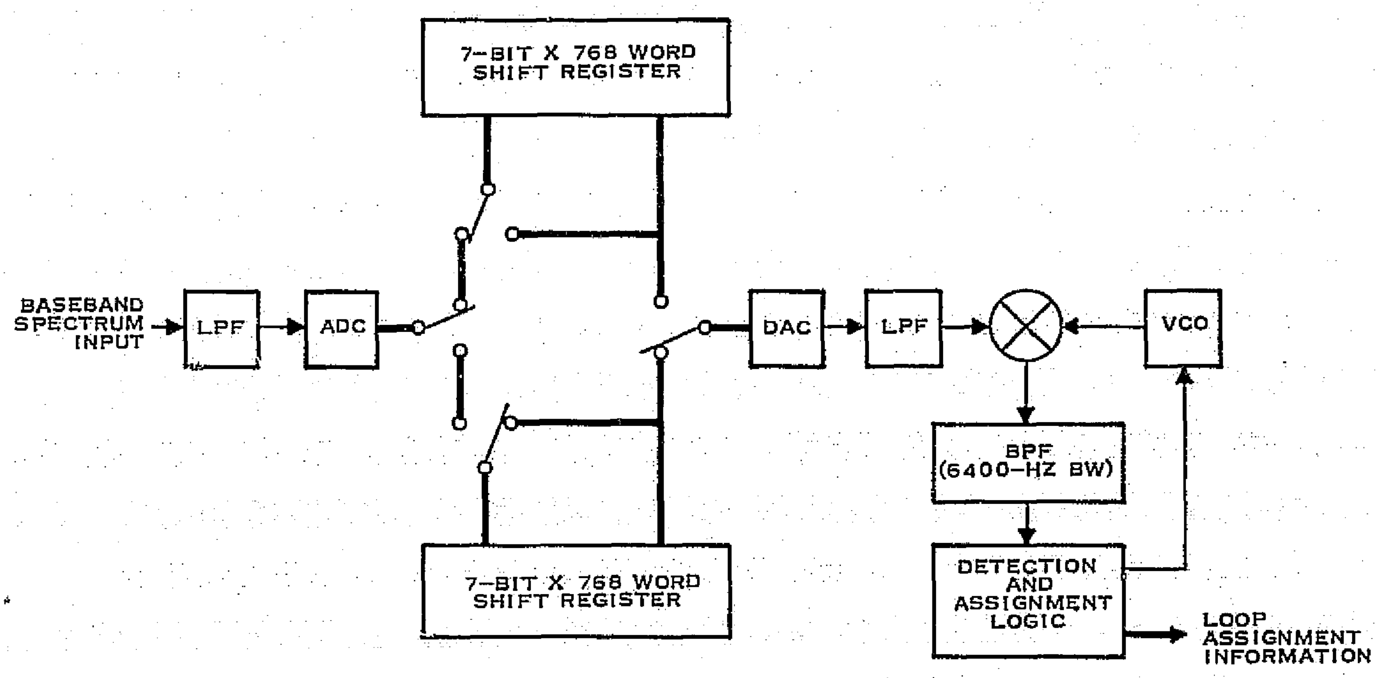

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Figure 4-3. Block Diagram for Digital Time Compression Spectrum Analyzer for Signal Detection and Channel Assignment Function

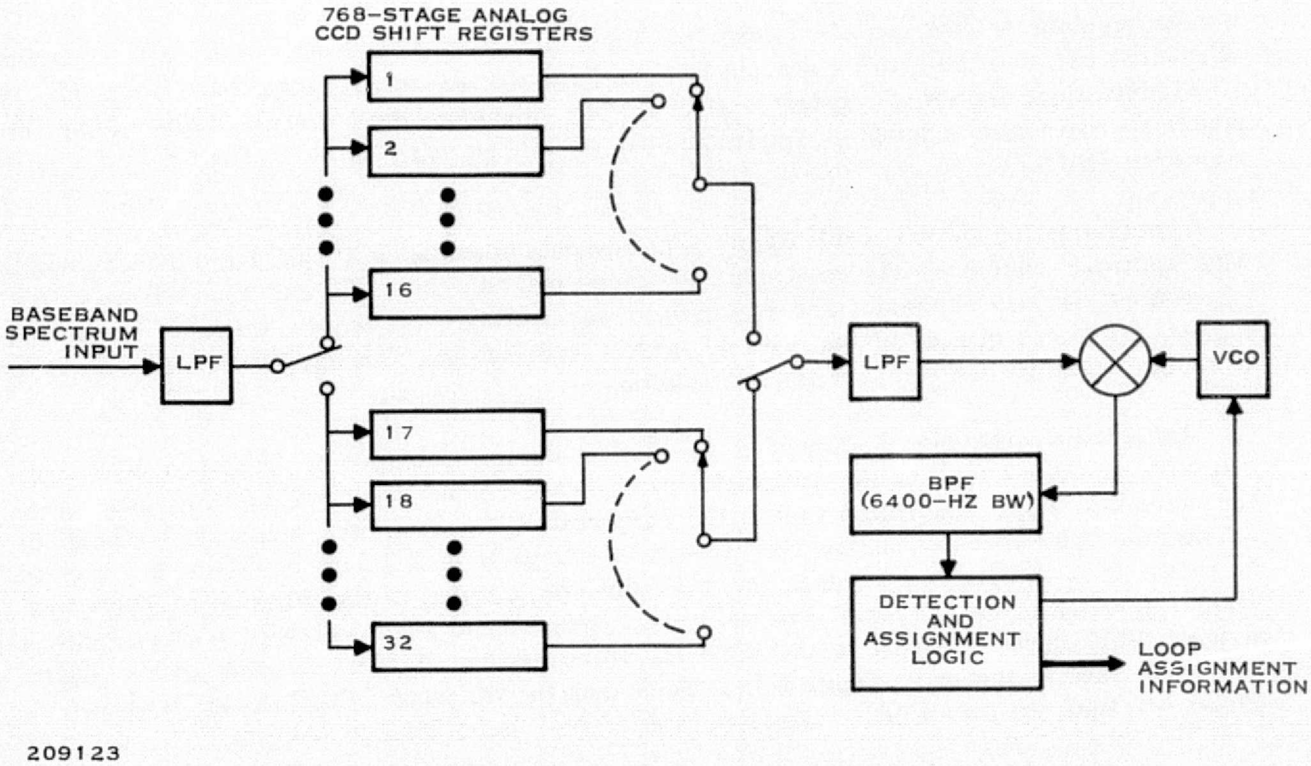

Figure 4-4. Block Diagram for Analog CCD Time Compression Spectrum Analyzer for Signal Detection and Channel Assignment Function

The digital time compressor operation is discussed in more detail in the following paragraphs. The input baseband spectrum is limited to dc to 100-kHz by the antialiasing lowpass filter. The filtered signal is sampled using a sample-and-hold amplifier followed by an 8-bit analog-to-digital conversion. The sampling rate should provide approximately 3 samples of the highest frequency component, i.e., the sampling frequency should be approximately 300 kHz. This provides 300,000 eight-bit words per second from the ADC. Of these 8 bits, only the 7 most significant bits are retained. (The use of certain window functions is desirable to minimize spectral sidelobes.) The output data are routed to a digital steering gate. For the position shown in Figure 4-3, data are shifted into the upper set of shift register memory elements. The registers are loaded in word-serial, bit-parallel fashion. When 768 samples have been stored, the input is switched to the lower set of shift registers while the output is taken from the upper set of registers. Figure 4-3 shows the output being taken from the lower registers. Since the output data rate is 16 times faster than the input rate, the contents of the serial shift registers are circulated 16 times. The digital words are applied to the DAC, the output of which is filtered by a lowpass filter (LPF). The frequency-expanded signal is mixed with a VCO signal. The VCO frequency is held constant for 16 samples of data. Following the readout of the 16 samples of stored data, the VCO is stepped to a new frequency until the VCO has been tuned to 256 distinct frequencies. The mixer output is amplified and filtered by a bandpass filter with a bandwidth of 6400 Hz. Any signal appearing in the passband of the filter above a predetermined threshold is detected. The frequency of the incoming signal is determined and a PLL receiver is assigned.

The analog CCD time compressor shown in Figure 4-4 performs the same function, but in a different manner. The baseband signal is frequency limited by a lowpass filter. The filtered signal is applied to either CCD shift registers 1 through 16 or to shift registers 17 through 32, depending on the "position" of the analog multiplex switch. The signal is clocked into one set of registers while data are read from the other set. Each CCD shift register must store 768 analog samples. Sixteen such registers are required in each set because the data cannot be recirculated as in the case of the digital data. The data is read out of a set of shift registers in multiplexed fashion, i.e., the contents of register 1 are read out, then register 2, etc. The readout rate is 16 times faster. Thus, the contents of all 16 registers are extracted from one group while 16 sets of identical data are loaded into the other set. The output is filtered and analyzed in the same way as in the digital system.

The chirp-Z transform approach is described generally in Subsection 2.9.1. The advantage of this approach is the speed with which the spectral analysis may be performed. A measure of the magnitude of each of 250 spectral components of the baseband can be determined in one acquisition interval (2.5 ms). This is possible because of the parallel nature of the coefficient-determining structure of the chirp-Z transform; a new coefficient is determined each time the data are sampled.

A simplified block diagram of a chirp-Z transform spectrum analyzer is shown in Figure 4-5. A simple antialiasing filter precedes the chirp-Z transform network which extracts 250 real coefficients and 250 imaginary coefficients. The value of each pair may be squared and added to represent the magnitude of any real frequency component. The spectrum is "scanned" once each 2.5 ms. The detection and assignment functions are the same as described for the time compressor technique.

For low signal-to-noise conditions, the probability of detection can be improved by averaging the magnitude of ensembles of data. Such a technique is useful for a geostationary satellite where signal strength will be low. A block diagram is depicted in Figure 4-6. The frequency magnitude coefficients are obtained by the chirp-Z transform network as previously described. However, the magnitudes are digitized using the ADC. The magnitudes are divided by the number of ensembles to be averaged. For binary numbers, this is nothing more than a shift register. The contents of memory (initially zero) are added word by word to the scaled outputs from the ADC and the results returned to memory. This process continues until 16 measurements (ensembles) have been averaged. Each memory location represents one average value for each of 250 frequency magnitude components. The contents of memory are then extracted word by word to ascertain the average magnitude. A digital "detector" assigns a loop to frequencies that exceed certain predetermined values. Such a technique provides a signal-to-noise improvement of 10 log N dB, where N in this case is 16. Hence, the improvement or enhancement is 12 dB. Such averaging is possible because each analysis can be performed in only 2.5 ms. One ensemble average can be performed in only 16 X 2.5 ms or 40 ms. However, in order to guarantee one complete ensemble average, the carrier burst must be present for at least twice this time (80 ms). Thus, this system can easily operate with a 100-ms unmodulated carrier preamble.

The four methods of implementing the search and assignment function presented in this section offer alternatives that are functions of the equipment design, technology development, and system application. Final selection will depend on the satellite equipment design and the quality of the system desired as well as the state of the technology for the chirp-Z transform method. While the single-ensemble-average chirp-Z transform method requires only a 5-ms carrier

Figure 4-5. Block Diagram of a Single-Ensemble Chirp-Z Transform Frequency-Determining Circuit

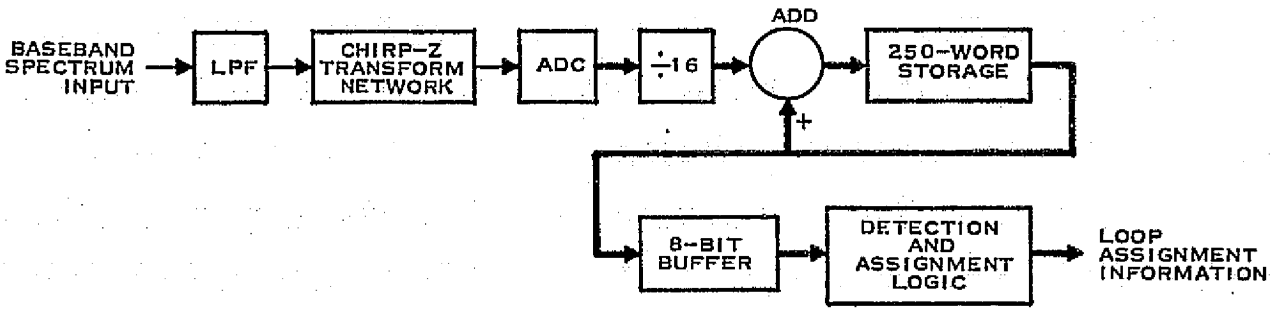

Figure 4-6. Block Diagram for 16-Ensemble Averaging Chirp-Z Transform Spectrum Analyzer

preamble, it is recommended that a 100-ms (minimum of 80 ms) carrier burst be used to allow the time-compressor approach and/or the ensemble averaging method to be used.

4.1.3 Phase-Lock Loop Characteristics

The primary function of the PLL receivers is to provide single-channel selection for individual ELT or EPIRB signals. The PLL approach provides the necessary frequency agility to accommodate the random frequency and time characteristics of the system. The PLL also provides a convenient means for measuring the frequency of the received signal, thus enabling the doppler shift to be determined.

The use of MSK necessitates a modification to the PLL approach used on RAMS because of the basic difference in MSK and PSK. There are two methods of implementing this function, one of which is a frequency-lock loop approach. For simplicity (as well as the mathematical similarity of the loops), the circuitry will continue to be referred to as PLL receivers. The frequency-lock approach uses a frequency comparator or discriminator followed by an integrator to provide an effective phase detector with a steady-state frequency error. The error can be made arbitrarily small with sufficient loop gain. The second approach uses a PLL for carrier tracking and a second PLL for demodulation.

A frequency-lock loop circuit is depicted in Figure 4-7. When the search circuitry detects a signal and assigns a loop, the control line for the VCXO is grounded. This provides a nominal injection frequency to be mixed with the output of the digital oscillator. The nominal center frequency of the VCXO must be sufficiently stable, so that the error of the frequency output

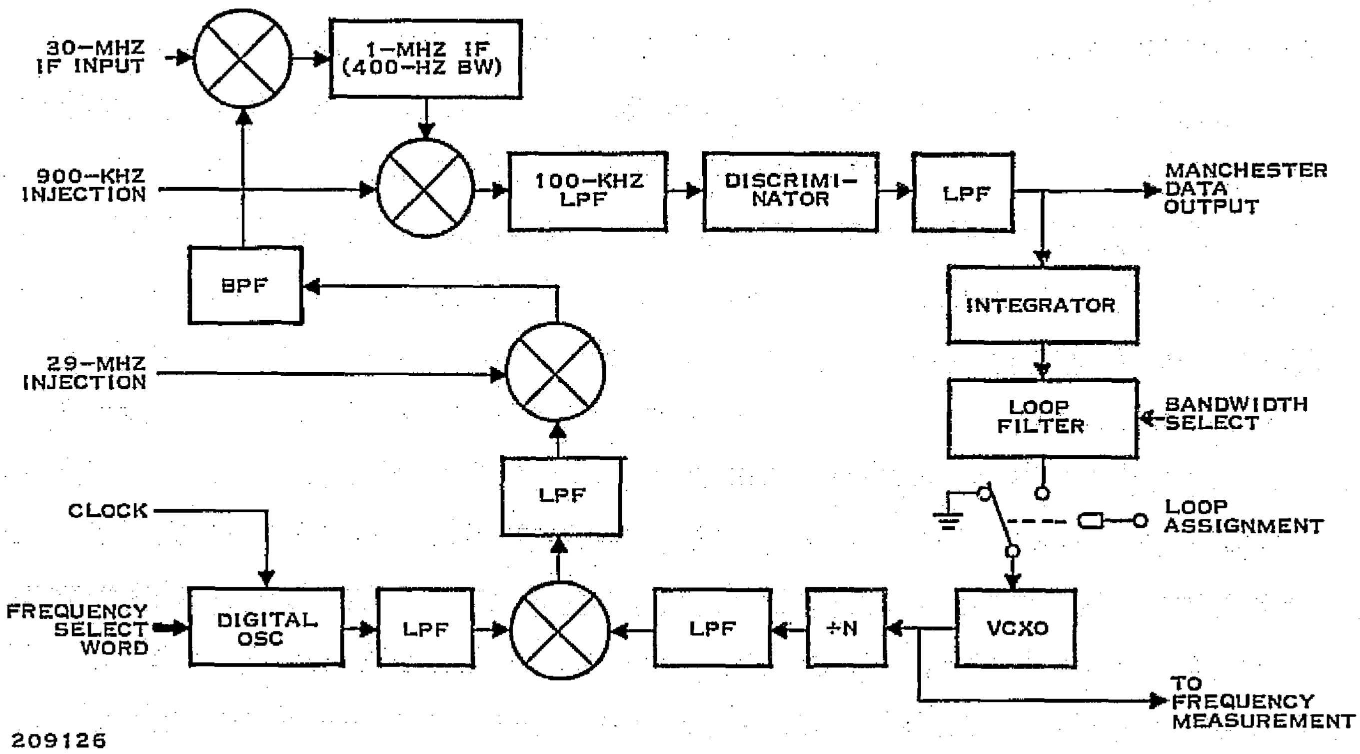

Figure 4-7. Frequency-Lock Loop Demodulator and Carrier Acquisition Circuitry

from the ÷N counter is negligible compared to 400 Hz, i.e., the frequency error output of the ÷N stage should be less than 40 Hz. This restriction is necessary to ensure that the detected spectral component will fall within the passband of the channel filter.

The difference frequency (digital oscillator and the VCXO/N frequency) is selected by an LPF. This difference frequency is mixed with an injection signal, shown as 29 MHz in Figure 4-7. (This is not a final choice for the injection frequency, but rather a choice to illustrate the neighborhood frequency.) Either the sum or difference frequency is selected by a bandpass filter (BPF), the choice depending on design details. This translated injection signal is mixed with the 30-MHz IF spectrum. (The 30 MHz value is also an illustrative choice, not a final choice.) The output of this mixer is the input to a channel filter with center frequency approximately 1 MHz. The bandwidth of this IF filter is approximately 400 Hz.

The 1-MHz filter specifications are critical. The center frequency stability must be ±0.003 percent at 1 MHz. This filter determines the channel bandwidth (400 Hz) and adjacent channel rejection (the filter shape factor should be 1.2 at the 30 dB points on the skirt). Since data are to be passed, the differential delay of this filter is an important consideration. This filter is in a feedback system, so that the absolute delay and phase shift are important parameters. The postfilter gain of this IF amplifier should be high. (This is possible since cross-modulation is no longer a problem.)

A final translation is made to a frequency suitable for performing the discriminator function. The frequencies shown in Figure 4-7 are intended to be illustrative values, not a final choice. The 100-kHz lowpass filter is noncritical, and is intended only to remove the higher frequency mixer components.

Once the discriminator has derived a frequency error voltage, the output of the loop filter is connected to the control line of the VCXO. The VCXO frequency is then slewed until the detected signal is centered within the passband of the 1-MHz IF and the discriminator error signal is near zero. This is possible since an integrator is employed in the loop, thus reducing the steady-state frequency error of the VCXO to a few hertz. When the loop has acquired lock, the loop bandwidth is reduced to preclude data modulation of the VCXO frequency.

The LPF following the discriminator removes the carrier component from the data. The demodulated data are available at this point for data detection and bit sync recovery. The output of the LPF is integrated and filtered to reduce the steady-state frequency error and remove the data modulation, respectively. The output of the VCXO is applied to a frequency counter which precisely measures the frequency of the VCXO. This information is correlated with the digital oscillator frequency to provide an accurate determination of the relative received signal frequency.

The absolute frequency of the injection signals is not highly critical. However, the frequency of all mixer injection signals (except the VCXO) must be phase-locked to a common reference having good short-term stability. The digital oscillator clock must also be derived from this source.

The physical size of the circuit is an important consideration since 12 loops are to be used. The key design parameters have been discussed. However, to achieve the ultimate desired capability, all design aspects must be evaluated and optimized.

4.1.4 Doppler Frequency Measurement

The low orbiter satellite determines position by accurate frequency measurement of the receiver carrier. Since differential doppler is to be used, a small steady-state frequency error is tolerable. However, the measurement error must not change over the period of one satellite pass, approximately 15 minutes. In addition, since the doppler frequency is continually changing as a function of satellite position, it is necessary that the precise time of the frequency measurement be determined and that the rate of change of frequency over the measurement interval be essentially linear. These problems and their resolutions are now considered.

Figure 4-7 indicates that frequency measurement of the VCXO is to be made. However, this frequency represents only a difference frequency or fine frequency. The coarse frequency measurement is obtained from the search unit. Since the incoming signal spectrum is resolved into 250 discrete frequencies, each can be defined by a unique "bin number." Hence, the coarse frequency measurement can be represented by a single 8-bit binary number. This binary number is generated by the search unit and represents a unique frequency, accurate to within ±200 Hz with respect to the reference clock.

The fine frequency measurement is derived from measurement of the VCXO frequency after the channel demodulator achieves lock. The measurement interval begins either upon recognizing frame sync or at a specified point in time following frame sync. The duration of the measurement interval is selected as a function of the frequency of jitter imposed on the VCXO signal. The frequency measurement determines the mean frequency over the measurement interval. Thus, the longer the measurement interval, the more closely the measured mean frequency will approach the actual mean frequency. Any drift over the measurement interval will introduce a bias error. However, if the drift is at a linear rate, its effects can be compensated for by taking the time of the measurement as that time corresponding to the mean time of the interval. Therefore, the frequency measurement interval should be as long as possible, i.e., approximately 500 ms for a message duration of 1 second with a 100-ms carrier preamble.

The least significant digits of the VCXO frequency are the only ones of interest since the frequency variations are small. (The maximum frequency variation is expected to be only ±200 Hz.) Therefore, 9 bits are required for this measurement. These bits, combined with the bits for the coarse frequency measurement, yield a minimum word length of 17 bits for the frequency measurement information. However, practical constraints indicate that the frequency error from the search unit may be as much as ±600 Hz, thus requiring 11 bits of fine frequency information for a total of 19 bits of frequency information.

The method of performing the fine-frequency measurement can assume various forms. The straightforward method involves measuring the rest frequency of the VCXO (control line grounded) and measuring the VCXO frequency with the loop locked. The difference is a precise measure of the fine frequency. However, this technique requires either the transmission of the rest frequency information to the ground station or a means of extracting the difference between two numbers on the spacecraft. An easier approach to implement uses the system master clock (which is phase-locked to the LO frequencies and search clock) to control the count interval of a counter measuring the VCXO frequency. Thus, the VCXO frequency is obtained relative to the search-unit-derived coarse frequency, and the need to measure the VCXO rest frequency is eliminated. In addition, since only the least significant bits of the VCXO frequency measurement are to be used, the counter length needs to be no greater than the number of stages given by:

$$
\begin{aligned}
l_c &= \log_2 2048 + \log_2 N\tau \\
&= 11 + \log_2 N\tau
\end{aligned}
$$

where τ is the measurement interval in seconds. (The product $N\tau$ must be rounded to the next largest binary number if it is not a binary number.) If only ±1-Hz accuracy is desired, only the 11 most significant bits are used.

As previously stated, the precise time at which the mean of the frequency measurement interval occurs must be determined. This time must be referenced to time used to establish spacecraft ephemeris data that are collected and transmitted to the ground station along with the frequency measurement data. The mean time of the measurement interval is delineated by using the next to the most significant bit of a counter used to measure the count interval, τ; this corresponds to a time mark at $\tau/2$. This time mark is used to fetch spacecraft time. The number of bits required for time depends on how often spacecraft time is transmitted. If the spacecraft time is transmitted once every minute via a separate channel, then the data message needs only enough bits to transmit time accurate to ±0.01 second for up to 60 seconds. This corresponds to 13 bits.

The data clock recovery is shown to be implemented using a phase-lock loop approach. This is not the only method that can be used but is certainly an attractive method for optimum data detection. The design of this circuitry should proceed with the objective of rapid acquisition, small noise bandwidth, and minimum clock jitter. Although not indicated in Figure 4-8, it may be desirable to inhibit the bit sync recovery circuit until the channel demodulator PLL receiver has acquired lock. This approach would not only shorten acquisition time, but would prevent random clock pulses from creating undesirable results. The bit synchronizer should be able to sense the correct clock phase since the bit sync code consists of an all-1s data pattern representing a sample of the data clock.

The lock detector should compare the recovered clock phase with the incoming sample of the data clock (bit sync code) and sense when lock has been established. A critical design parameter is the time constant of the lock detector. If the time constant is too great, lock will not be detected within the period of the bit sync code. If the time constant is too small, the probability of false lock increases. The lock signal is used to enable the frame sync recognizer to initiate its search for frame sync.

The decoder is nothing more than an exclusive-OR circuit followed by a single storage register. The decoder combines the detected data and recovered bit sync to produce NRZ data that are shifted into the single register.

The frame sync recognizer can be implemented in numerous ways. It must, however, simultaneously compare 24 bits of data with the known frame sync pattern. Frame sync will be indicated when 22, 23, or 24 of the 24 bits check true. The data will be checked once each bit time until frame sync is recognized.

Once frame sync is established, the next 76 bits of data are loaded into a buffer storage register. The loading and unloading of data is accomplished using control logic which interfaces with the output data buffer control. The extent of the control logic and the length of the buffer register are, therefore, a function of the design of the output buffer. The phase-lock loop is retained until all data have been removed, at which time it is released.

To summarize, the doppler frequency measurement requires the determination of three parameters. First, the coarse-frequency measurement is made directly from the bin number used to assign the frequency of the loop. Second, the fine-frequency measurement is obtained from a counter gated precisely by a signal derived from the system clock. Third, the spacecraft time of day is fetched at precisely the midpoint of the fine-frequency measurement interval. These data consist of 8 bits of coarse-frequency information, 9 bits of fine-frequency information, and 13 bits of time information which are combined with the received data in a suitable format for transmission to ground equipment. This formatter is discussed in Subsection 4.1.6.

4.1.5 Data Recovery

The demodulated Manchester data (Figure 4-7) from the channel demodulator must be filtered and the bit sync clock extracted before the data can be converted to NRZ data. Following this conversion, frame sync can be determined and the desired information and parity bits can be extracted. These bits are combined with the doppler information for transmission to the ground equipment.

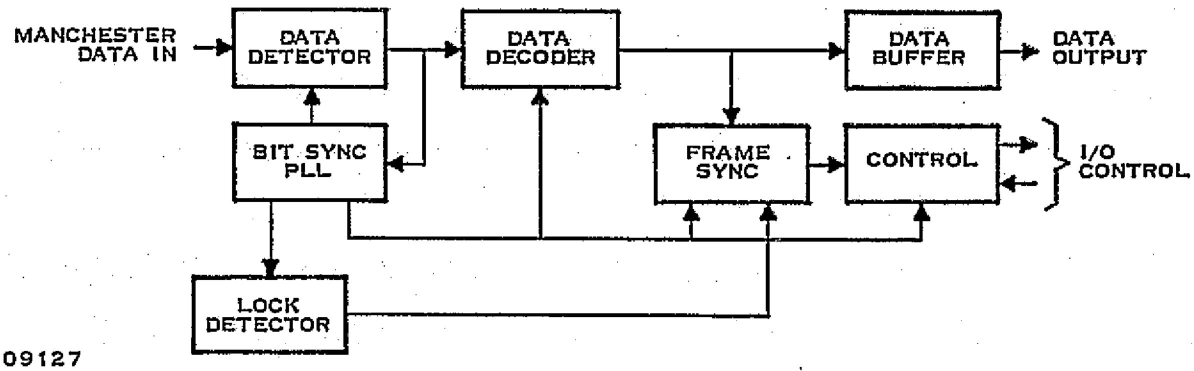

Figure 4-8. Data Recovery Block Diagram

Figure 4-8 is a block diagram of the data recovery circuitry. The data detector is a matched filter that also serves as a phase detector for the bit sync phase-lock loop. The bit sync phase-lock loop extracts the data clock from the Manchester data. A lock detector senses when the bit sync PLL is in lock; this signal enables the frame sync recognizer. The recovered data clock is combined with the detected Manchester data in the data decoder. The decoded data are in NRZ format. The frame sync detector, when enabled, begins to check the data for frame sync. When the frame sync pattern is recognized, the data buffer register's input shift control is enabled by the control circuitry. The enable is maintained for precisely 76 bit times. This allows the 55 information bits and the 21 parity check bits to be loaded into the data buffer. When this task is completed, the control logic signals that data are ready. A signal from the data formatter/buffer initiates data transfer. Upon completion of data transfer, the control logic releases the phase-lock loop channel demodulator for reassignment.

The data detector can be implemented using a reset integrator approach if the demodulated data are essentially square-wave data. However, if the channel filtering alters the waveform significantly, the detector may need to include a filter that equalizes the data waveform. Both phase and amplitude equalization may be required.

4.1.6 Data Formatter/Telemetry Buffer

Once the 76 information and parity bits and the 30 frequency and time bits have been compiled, they must be combined into a composite 106-bit message.* A typical format is shown in Figure 4-9. If the data are to be identified on a word basis, then seven 16-bit words or 14 bytes of data would accommodate a message.

The telemetry buffer must store data obtained from each of the 16 phase-lock loop channel demodulators. This storage must be sufficient to combine the 16 channels of data into a single output data stream for storage and/or transmission. The maximum message rate is 16 messages per second. However, the average message rate will be well below this rate. Therefore, the telemetry data rate can be reduced by having buffer storage sufficient to accommodate peak loads.

*The fine frequency or time may require additional bits, depending on individual system design.

If the maximum number of transmitters in the field of view is 400, each transmitting one message every 40 seconds, the average arrival rate is:

$$\alpha = \frac{400}{40} = 10 \text{ messages/second}$$

If the transmissions occur randomly in time, their time distribution is a Poisson distribution. The Poisson density function is:

$$f(x) = \frac{(\alpha T)^x}{x!} e^{-\alpha T}$$

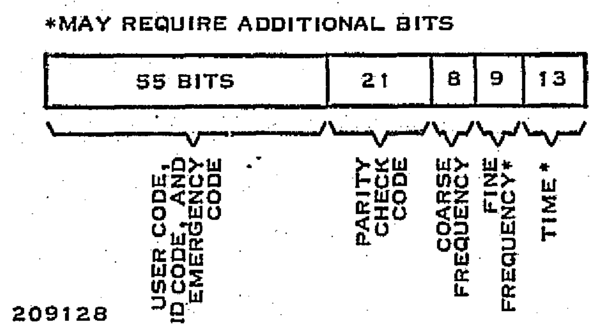

Figure 4-9. Telemetry Data Format for a Single Message

The probability that k or fewer messages arrive in T seconds is given by the cumulative distribution:

$$p(x \leqslant k) = \sum_{x=0}^{k} f(x)$$

For $\alpha = 10$ and T = 1 second,

$$p\,(x \leqslant 16) = 0.973$$

Hence, the ratio of peak-to-average arrival rate is equal to or less than 16/10, 97.3 percent of the time during any given second. Minimum buffer size is capable of storing six messages (16–10). This assumes a service rate of 16 messages per second for a probability of service of 97 percent. This minimum buffer size is predicated on a constant service rate of one message every 16 ms. Obviously, if the period between services increases, the buffer size must also increase.

The above method of determining queue size circumvents a basic mathematical problem of calculating the probability density function for the number in queue in a system having random arrival and fixed service rates. Works[1,2], on queueing theory consider the worst case of random arrival rates (Poisson distributed) and random service time (exponential time distribution). The probability density function and the probability distribution function for the number, n, in queue are respectively:

$$p(n) = (1 - \rho)\rho^n$$

$$p(n > N) = \rho^{N+1}$$

[1] D.R. Cox and W.L. Smith, *Queues*, Methuen and Company, Ltd. (London, 1967).
[2] Phillip M. Morse, *Queues, Inventories, and Maintenance*, John Wiley and Sons (New York, 1962).

where

$\rho = \alpha/\sigma$ = traffic intensity in erlangs

α = the mean arrival rate

σ = the mean service rate.

For a probability of 0.027 that the number in queue is greater than N, the value of N is found to be:

$$N = \frac{\log 0.027}{\log \rho} - 1$$

$$= 6.68$$

Since N must be an integer, the minimum value of N is 7. This is only slightly greater than the value obtained for a constant service time. Therefore, for a traffic intensity (ρ) value of 0.63, the buffer can be sized by assuming random service times. If a 10-message buffer is assumed, the probability of accommodating peak demands is:

$$p_B = 1 - \rho^{11}$$

$$= 0.994$$

Hence, a 1,024-bit output buffer should be adequate. The buffer size was calculated on a service rate of 16 messages per second. If each message consists of 14 bytes, the data rate is:

$$\text{service rate} = 16 \times 14 \times 8 = 1{,}792 \text{ bits/second}$$

Since the TIROS-N spacecraft has only 1,920 BPS of spare data capacity, the use of this data rate would leave only 128 BPS of spare telemetry capacity. Therefore, it may be desirable to reduce the service rate and increase the buffer size. If the output data rate were reduced to 14 messages/second (1,568 BPS), then a residual spare telemetry rate of 352 bits/second would be maintained. The buffer size is then:

$$N = \frac{\log p_n}{\log \rho} - 1$$

$$= 14.2 \text{ messages}$$

Such a system would require a 1,600-bit buffer. If a 2,048-bit buffer is assumed, then 18 messages can be stored. The value of ρ is:

$$\rho = \log^{-1} \left\{ \frac{\log [1 - p\,(n > 18)]}{19} \right\}$$

$$= 0.764 \text{ for } p\,(n > 18) = 0.006$$

The minimum service rate is:

$$\sigma = \alpha/\rho$$
$$= 13 \text{ messages/second}$$
$$= 1{,}456 \text{ bits/second.}$$

This leaves a 464-BPS spare telemetry data rate.

The TIROS-N telemetry channel structure consists of one major frame each 32-second interval. Each major frame consists of 320 minor frames. Each minor frame consists of 104 eight-bit words. A minor frame is transmitted each 0.1-second interval. Therefore, a minimum of 18.2 eight-bit words must be transmitted during each minor frame. This corresponds to 1.3 messages per minor frame. This does not provide an easy means of formatting the data since the meassage should correspond to an integral number of bytes of data and the message division should be a rational number. There are two alternatives: (1) transmit 1.5 messages per minor frame corresponding to 1,680 BPS, or (2) transmit one idle byte in each message and 1.333 . . . messages per minor frame corresponding to 1,600 BPS. The latter case adds only 80 BPS to the spare telemetry rate but wastes 120 BPS of usable space. It seems that the optimum output data rate is 1,680 BPS, leaving a residual spare data rate of 240 BPS. Therefore, the service rate will be 15 messages/second. The resulting traffic intensity is:

$$\rho = \frac{10}{15} = 0.666\ldots$$

If a 1,024-bit buffer is used, the maximum number of messages that can be stored is:

$$N = \frac{1024}{14 \times 8} = 9.14$$

$$= 9 \text{ complete messages}$$

$$p(n > 9) = \rho^{N+1}$$

$$= 0.01734$$

The probability of service is 1 − 0.01734 or 0.9827. However, since the calculations are based on random service times while in actuality the service time is constant, the probability of success will be greater.

The output buffer and telemetry characteristics can be summarized as follows:

Service rate = 15 messages/second

Traffic intensity = 0.6667 erlang

Output buffer size = 1,024 bits or 9 messages

Number of 8-bit words required during each minor frame = 21

Probability that the buffer is adequate $\geqslant$0.9827.

4.1.7 121.5/243-MHz Transponder

The requirements for a satellite system capable of operating with existing ELTs are studied in a separate report. This section relates only the basic concepts for the sake of continuity.

The equipment required for detecting and locating existing ELTs is essentially a satellite transponder system that uses a "bent-pipe" technique. The satellite equipment for this task is assumed to be a completely separate package or satellite subsystem. It will have nothing in common with the differential doppler system discussed in this report. Even though such a system eventually uses the same technique for performing the position locating function, the implementation is considered to be totally different because the problems associated with existing ELTs necessitate a different approach. Major system differences include continuous-broadcast ELT characteristics, undesirable FM sidebands, low power, and basically poor signal quality from existing ELT designs.

4.2 GEOSTATIONARY SATELLITE EQUIPMENT

The satellite equipment required for a geostationary satellite system is similar to that required for the low orbiter system. However, there are design variations peculiar to geostationary satellite equipment.

The most obvious difference is that the received signal strength is much less than for a low orbiter. In general, the receiver front-end design is more critical and the design should use techniques to improve the receiver noise figure. This is particularly true in the case of the hyperbolic ranging concept where the limiting factor for accuracy becomes the signal-to-noise ratio of the receiver.

Another major difference is the absence of doppler dispersion which increases the probability of mutual interference. A compensating factor, however, is the fact that the satellite is continuously visible. This permits the problems of mutual interference to be overcome by offering unlimited tries with the only constraint being waiting time. Such a system obviously requires more channel demodulators (phase-lock loop receivers) than a low-orbiter system since the number of transmitters requesting service is greater.

Finally, the individual requirements of the intended use of such a system necessitate unique design characteristics. Therefore, a discussion of possible applications of a geostationary satellite system is in order.

The simplest system is one which is used for detection only. Such a system requires only one geostationary satellite (stationed over Brazil) with a single receiver. The receiver functions the same as that described for a low orbiter, but does not require frequency measurement circuitry. The only requirement for the receiver is to detect and decode ELT signals, strip the information bits from the data, and relay the information bits to a ground station. Such a satellite provides one-third earth coverage for detection of ELT signals.

An extension of the basic detection circuitry to four receivers provides the basis for the interferometry technique. Such a system may also require a special-purpose receiver that provides coarse angle information. The other four receivers are identical. The position information is extracted by accurate measurement of the carrier phase. The phase measurement is made at the channel demodulator carrier frequency. The four phase measurements are digitized and combined with the ID of the received signal. The information returned to a ground station is similar to that from a low orbiter. The basic system simplicity is the requirement for only a single satellite, but this is offset by the relative difficulty of maintaining alignment between four antennas and receivers.

The last implementation considered is the hyperbolic ranging concept. This system requires a minimum of three satellites. The absolute time and position of each spacecraft must be determined. While this is possible, it is not an easy task to keep three satellites properly located. The three spacecraft receivers are basically the same as those described for the low orbiter with the addition of a narrow-bandwidth phase-lock loop capable of tracking the bit clock. Spacecraft time is taken at a predetermined bit time and combined with the ELT data. This information is transmitted to ground equipment. (The ID of each spacecraft is also included, either implicitly or explicitly.)

The use of a geostationary satellite system as a sole means of providing position information does not appear practical with ELTs meeting the design criteria for the low-orbiter system. However, as supplemental information systems, they can provide invaluable aid in reducing the total emergency reaction time. Thus, while a range accuracy of 100 km for geostationary satellite position information may not be adequate for final search-and-rescue activity, it would be extremely helpful in initializing search activity in the proper area. Once such activity was initiated, final accurate position information could be obtained from the low orbiter and forwarded to the initial search operation. The use of geostationary satellites to obtain initial detection and a rough estimate of ELT location provides search-and-rescue personnel with the highly desirable feature of nearly instantaneous alert.

4.2.1 Location Using Interferometry

The satellite equipment for each of the four receivers is essentially the same as the equipment described for the low orbiter. The primary difference is the implementation of the channel demodulator since the key parameter to be measured is carrier phase. This places some stringent requirements on the phase coherence and phase stability of the injection signals, however. Therefore, the design requirements for the carrier generation circuitry must be more tightly specified for phase stability and steady-state phase error. In addition, provision must be made for precise calibration for equal phase delay through each pair of antenna/receiver sets.

The channel demodulator must contain a phase-lock loop that forces the VCXO into phase lock with the received carrier (refer to Figure 4-7). The loop characteristics must provide rapid acquisition ($t_a < 50$ ms to 0.05-percent accuracy). The phase measurement is then made with respect to the paired receiver. The proper carrier frequency is obtained from the search unit and appropriate channel demodulators are paired accordingly. The phase measurement must be completed within 50 ms. The IDs from the paired demodulators should be compared to validate the phase measurement. The phase data are combined with the ID and parity bits for transmission to ground equipment.

As previously discussed, it will be necessary to obtain a coarse phase measurement to eliminate ambiguities. One method of obtaining this information is to have a fifth receiver/antenna combination. This system could be structured to sense amplitude differences in received signals arriving from different directions, thus providing coarse angle information, or a short-base line interferometer technique could be used. Either technique would require essentially the same equipment as the precision measurement system. It may be desirable to operate the equipment in a dual mode: part of the time the system would obtain coarse angle information, and part of the time it would obtain fine angle information.

Full assessment of the problems and design considerations of an interferometry system is beyond the scope of this study. Only basic compatibility problems have been evaluated and general design considerations outlined. The basic conclusion is that such a system is technically feasible although some severe problems may be encountered in the implementation.

4.2.2 Location Using Unilateral Ranging

The circuitry required for implementing this approach is basically the same as that for the low orbiter. It is assumed that the time clocks for all three spacecrafts are synchronized and maintain the same absolute time. It is further assumed, of course, that the satellite positions are accurately known.

The major change in the spacecraft equipment design is in the data recovery circuitry. A block diagram of the modified circuitry is depicted in Figure 4-10. The object is to determine precisely the arrival time of a predetermined bit of data. The coarse time is obtained from frame sync recognition. The precision phase-lock loop locks to the data clock. The bandwidth is narrowed sufficiently to provide a minimal mean-squared phase error in the data clock transition, i.e., to reduce phase jitter to an acceptable level. This refinement should occur over almost the entire message duration. After a specified number of bit times, a clock transition marks the precise time a bit arrives. This time mark is used to delimit spacecraft time which is combined with the received data and forwarded to ground equipment.

The design requirements for the precision PLL have been discussed. The primary design parameters are short-term stability and minimum noise bandwidth. In addition, the system signal-to-noise ratio is critical to system accuracy and every effort should be made to enhance the receiver noise figure. Finally, it is not necessary to measure carrier frequency, so that circuitry for measuring frequency is not required.

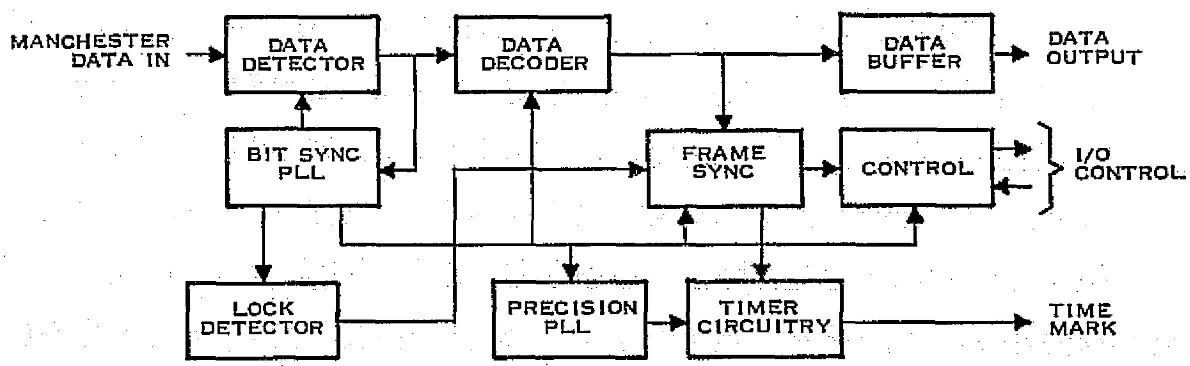

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Figure 4-10. Data Recovery Circuitry Modified for Unilateral Ranging

The brief study of a geostationary satellite system has been included to show that it can operate with ELTs in a low-orbiter system, although some of the spacecraft equipment design may pose some technical challenges.

APPENDIX A
GEOGRAPHIC COVERAGE DATA

APPENDIX A

GEOGRAPHIC COVERAGE DATA

A calculator (HP 9830) was programmed to solve the geographic coverage vector equations presented in Subsection 2.4. The results were plotted in polar coordinates for ELTs located at 0-, 30-, and 45-degree latitudes. This data is depicted in Figures A-1 through A-3. Each family of curves plots azimuth (θ in polar coordinates) versus elevation angle (ρ in polar coordinates) for various satellite equator crossing longitudes relative to the longitude of the ELT. The 0-degree elevation angle (horizon) is at the origin while the 90-degree elevation angle (zenith) is the third major concentric circle. Each curve in a family defines the set of azimuth and elevation points traced out by a satellite having the indicated relative longitude at the time it crosses the equator. The plots include the earth's rotation effects and the inclination of the satellite's orbital plane. (The "butterfly" patterns are tilted because of the plane of the satellite's orbit.) These curves are helpful in evaluating the effects of terrain on viewing angle, the variation of elevation angle as a function of equator crossing longitude, and amount of time the satellite is visible above a given elevation angle.

The same data is presented in tabular form immediately following Figure A-3. An estimate of the time that the satellite will be visible above a 10-degree elevation angle is computed for each set of values. The time is not exact because only discrete points were used. A copy of the program listing follows this table.

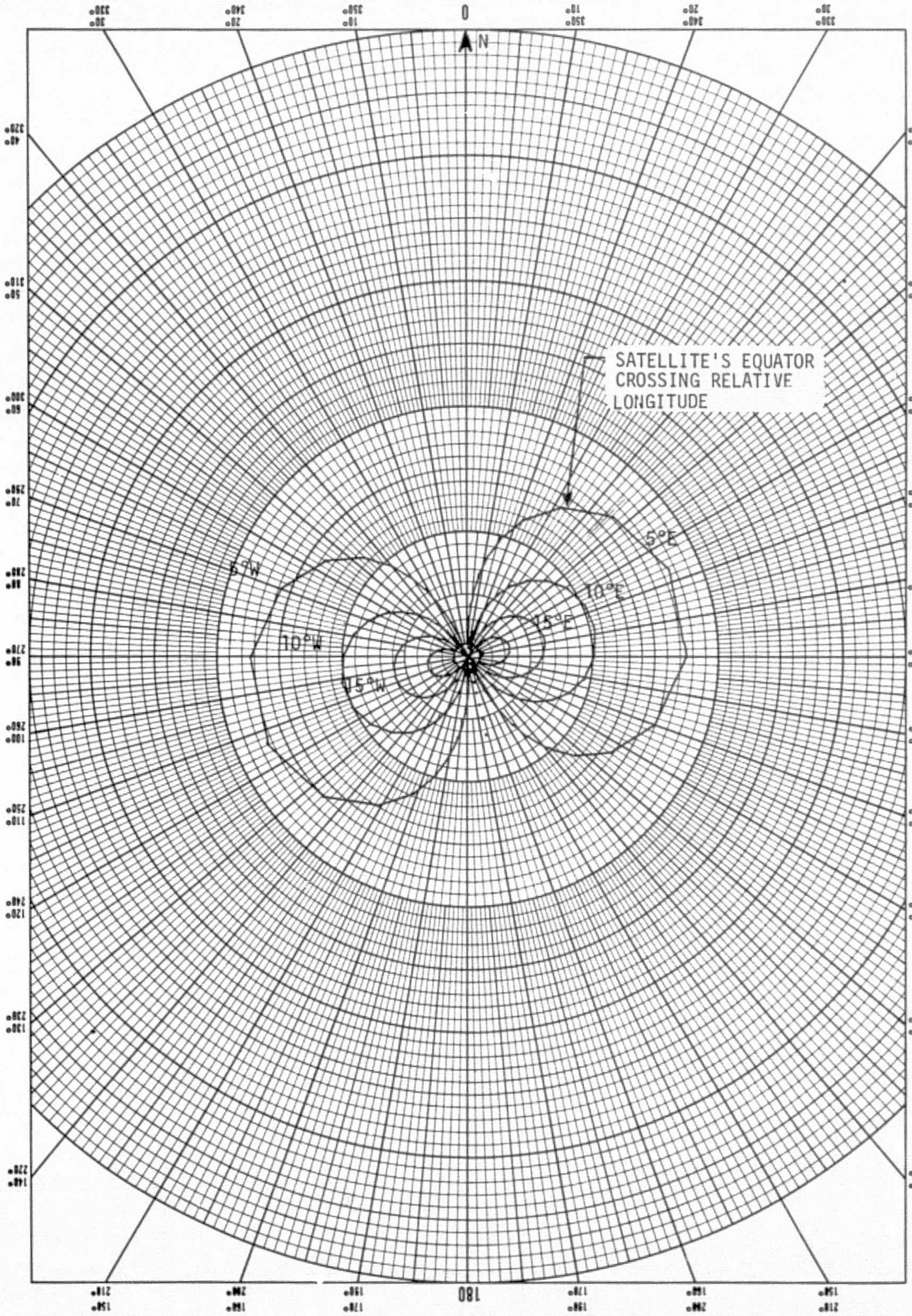

Figure A-1. Satellite Viewing Angles for a 0-Degree Latitude

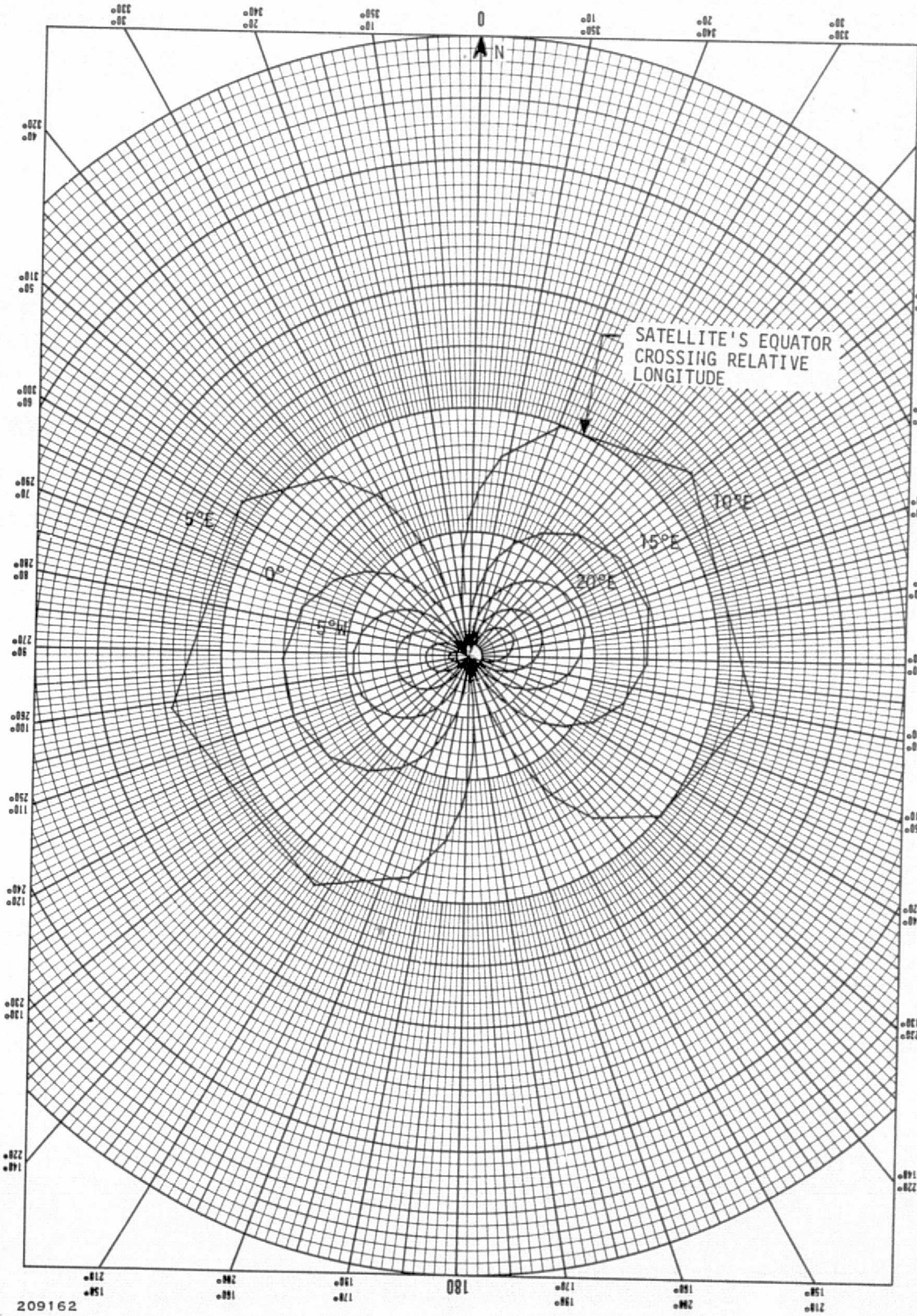

Figure A-2. Satellite Viewing Angles for a 30-Degree Latitude

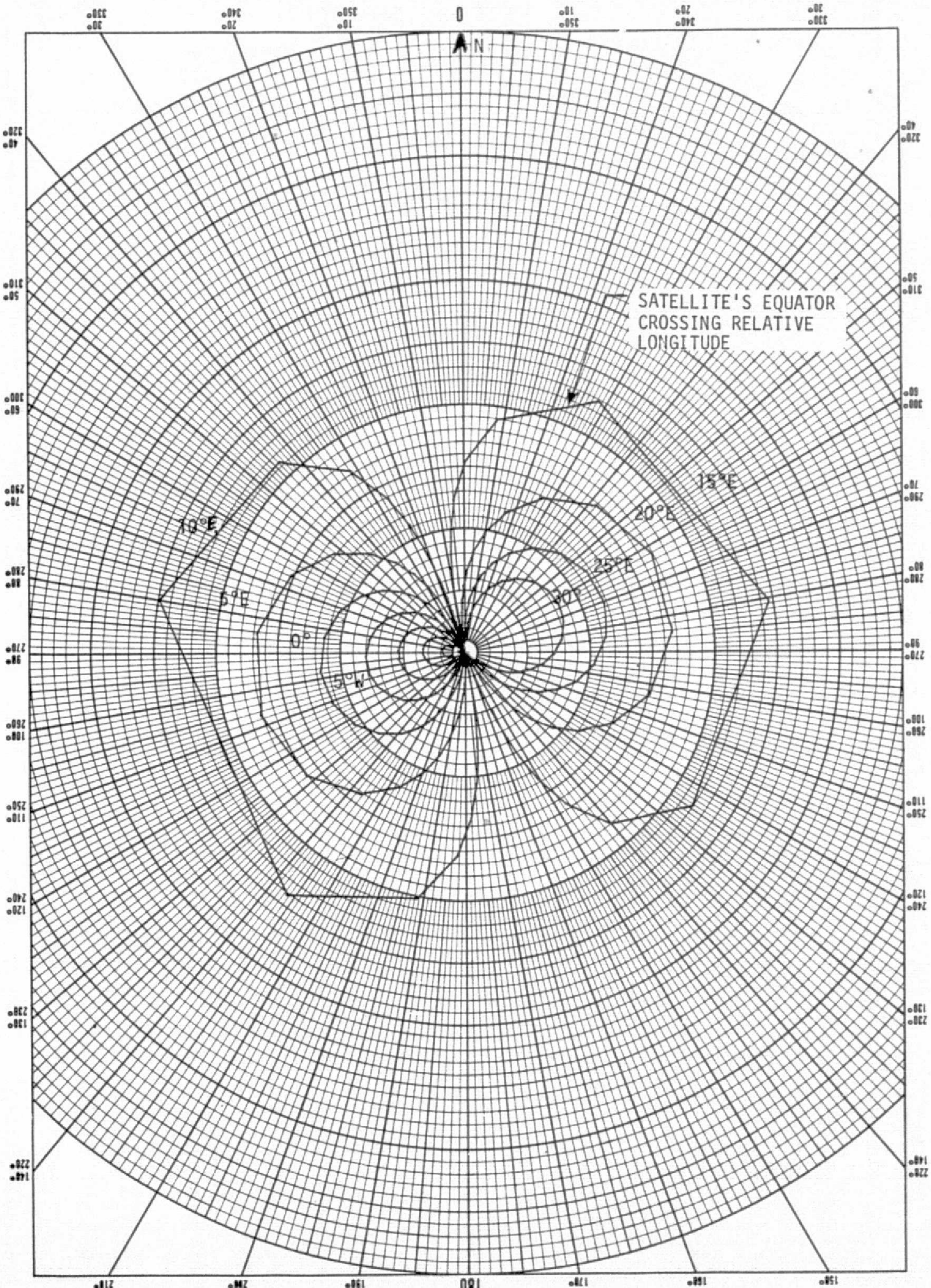

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Figure A-3. Satellite Viewing Angles for a 45-Degree Latitude

OBSERVER LATITUDE IS 0.01 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
-18.00	-25.00	3284.44	227.95	0.73
-16.00	-25.00	3193.93	232.07	1.58
-14.00	-25.00	3117.60	236.43	2.31
-12.00	-25.00	3056.60	240.99	2.91
-10.00	-25.00	3011.89	245.71	3.36
-8.00	-25.00	2984.22	250.56	3.64
-6.00	-25.00	2974.08	255.46	3.75
-4.00	-25.00	2981.62	260.38	3.67
-2.00	-25.00	3006.66	265.23	3.41
0.00	-25.00	3048.69	269.98	2.99
2.00	-25.00	3106.95	274.57	2.41
4.00	-25.00	3180.44	278.96	1.70
6.00	-25.00	3268.00	283.14	0.88

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 0.00 MINUTES

OBSERVER LATITUDE IS 0.01 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
-24.00	-20.00	3323.62	209.41	0.37
-22.00	-20.00	3171.11	212.72	1.79
-20.00	-20.00	3028.27	216.37	3.20
-18.00	-20.00	2896.68	220.37	4.55
-16.00	-20.00	2778.11	224.76	5.84
-14.00	-20.00	2674.40	229.54	7.03
-12.00	-20.00	2587.45	234.70	8.07
-10.00	-20.00	2519.08	240.20	8.92
-8.00	-20.00	2470.89	245.99	9.54
-6.00	-20.00	2444.10	251.98	9.89
-4.00	-20.00	2439.41	258.05	9.95
-2.00	-20.00	2456.92	264.09	9.72
0.00	-20.00	2496.10	269.97	9.21
2.00	-20.00	2555.87	275.61	8.46
4.00	-20.00	2634.73	280.93	7.50
6.00	-20.00	2730.89	285.88	6.37
8.00	-20.00	2842.46	290.45	5.13
10.00	-20.00	2967.53	294.64	3.81
12.00	-20.00	3104.29	298.46	2.44
14.00	-20.00	3251.08	301.95	1.04

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 0.00 MINUTES

OBSERVER LATITUDE IS 0.01 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
-26.00	-15.00	3275.07	197.67	0.82
-24.00	-15.00	3089.34	200.26	2.59
-22.00	-15.00	2909.84	203.20	4.41
-20.00	-15.00	2738.03	206.54	6.29
-18.00	-15.00	2575.65	210.35	8.21
-16.00	-15.00	2424.80	214.70	10.14
-14.00	-15.00	2287.96	219.66	12.05
-12.00	-15.00	2167.92	225.30	13.87
-10.00	-15.00	2067.76	231.63	15.51
-8.00	-15.00	1990.58	238.64	16.87
-6.00	-15.00	1939.19	246.20	17.82
-4.00	-15.00	1915.70	254.13	18.27
-2.00	-15.00	1921.14	262.14	18.16
0.00	-15.00	1955.21	269.96	17.52
2.00	-15.00	2016.40	277.34	16.40
4.00	-15.00	2102.23	284.11	14.93
6.00	-15.00	2209.69	290.19	13.22
8.00	-15.00	2335.66	295.59	11.37
10.00	-15.00	2477.13	300.34	9.46
12.00	-15.00	2631.42	304.52	7.54
14.00	-15.00	2796.20	308.18	5.64
16.00	-15.00	2969.51	311.42	3.79
18.00	-15.00	3149.72	314.28	2.00
20.00	-15.00	3335.48	316.83	0.26

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 7.33 MINUTES

OBSERVER LATITUDE IS 0.01 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
-28.00	-10.00	3361.34	186.22	0.03
-26.00	-10.00	3151.76	187.86	1.98
-24.00	-10.00	2944.96	189.75	4.05
-22.00	-10.00	2741.86	191.94	6.25
-20.00	-10.00	2543.61	194.51	8.61
-18.00	-10.00	2351.70	197.55	11.14
-16.00	-10.00	2168.07	201.20	13.87
-14.00	-10.00	1995.26	205.61	16.78
-12.00	-10.00	1836.54	211.00	19.86
-10.00	-10.00	1696.09	217.61	23.00
-8.00	-10.00	1578.96	225.67	26.03
-6.00	-10.00	1490.78	235.30	28.62
-4.00	-10.00	1437.00	246.34	30.37
-2.00	-10.00	1421.54	258.19	30.90
0.00	-10.00	1445.64	269.94	30.08
2.00	-10.00	1507.34	280.71	28.11
4.00	-10.00	1602.20	290.00	25.39
6.00	-10.00	1724.62	297.74	22.33
8.00	-10.00	1869.04	304.08	19.19
10.00	-10.00	2030.57	309.26	16.15
12.00	-10.00	2205.26	313.53	13.29
14.00	-10.00	2390.02	317.08	10.61
16.00	-10.00	2582.45	320.06	8.13
18.00	-10.00	2780.73	322.60	5.81
20.00	-10.00	2983.44	324.78	3.65
22.00	-10.00	3189.51	326.69	1.62

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 9.59 MINUTES

OBSERVER LATITUDE IS 0.01 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
-28.00	-5.00	3347.10	176.70	0.16
-26.00	-5.00	3125.59	177.51	2.23
-24.00	-5.00	2905.07	178.46	4.47
-22.00	-5.00	2686.08	179.58	6.89
-20.00	-5.00	2469.35	180.92	9.56
-18.00	-5.00	2255.82	182.56	12.52
-16.00	-5.00	2046.81	184.59	15.87
-14.00	-5.00	1844.15	187.19	19.70
-12.00	-5.00	1650.47	190.59	24.13
-10.00	-5.00	1469.61	195.21	29.29
-8.00	-5.00	1307.16	201.74	35.25
-6.00	-5.00	1171.03	211.38	41.83
-4.00	-5.00	1071.49	225.86	48.18
-2.00	-5.00	1019.42	246.38	52.34
0.00	-5.00	1022.16	269.89	52.10
2.00	-5.00	1079.26	289.85	47.61
4.00	-5.00	1182.79	303.79	41.18
6.00	-5.00	1321.76	313.07	34.65
8.00	-5.00	1486.09	319.40	28.77
10.00	-5.00	1668.12	323.90	23.69
12.00	-5.00	1862.46	327.25	19.32
14.00	-5.00	2065.42	329.82	15.55
16.00	-5.00	2274.45	331.87	12.25
18.00	-5.00	2487.78	333.53	9.32
20.00	-5.00	2704.15	334.91	6.68
22.00	-5.00	2922.63	336.09	4.28
24.00	-5.00	3142.53	337.10	2.07
26.00	-5.00	3363.31	337.98	0.01

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 10.15 MINUTES

OBSERVER LATITUDE IS 0.01 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
-26.00	0.00	3199.15	167.34	1.53
-24.00	0.00	2974.16	167.30	3.75
-22.00	0.00	2749.76	167.26	6.16
-20.00	0.00	2526.43	167.23	8.82
-18.00	0.00	2304.79	167.20	11.81
-16.00	0.00	2085.73	167.17	15.21
-14.00	0.00	1870.47	167.15	19.16
-12.00	0.00	1660.81	167.13	23.87
-10.00	0.00	1459.50	167.11	29.62
-8.00	0.00	1270.83	167.10	36.83
-6.00	0.00	1101.61	167.10	46.07
-4.00	0.00	962.43	167.10	57.97
-2.00	0.00	868.06	167.13	72.85
0.00	0.00	834.00	180.00	89.91
2.00	0.00	867.42	347.00	73.00
4.00	0.00	961.28	347.04	58.10
6.00	0.00	1100.11	347.05	46.17
8.00	0.00	1269.10	347.07	36.91
10.00	0.00	1457.62	347.09	29.68
12.00	0.00	1658.83	347.11	23.92
14.00	0.00	1868.42	347.13	19.20
16.00	0.00	2083.64	347.16	15.24
18.00	0.00	2302.67	347.18	11.84
20.00	0.00	2524.29	347.22	8.85
22.00	0.00	2747.61	347.25	6.19
24.00	0.00	2972.00	347.29	3.77
26.00	0.00	3196.98	347.33	1.55

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 10.71 MINUTES

OBSERVER LATITUDE IS 0.01 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
-24.00	5.00	3144.58	157.11	2.05
-22.00	5.00	2924.66	156.10	4.26
-20.00	5.00	2706.15	154.93	6.66
-18.00	5.00	2489.74	153.55	9.29
-16.00	5.00	2276.36	151.89	12.22
-14.00	5.00	2067.27	149.86	15.52
-12.00	5.00	1864.23	147.29	19.29
-10.00	5.00	1669.77	143.96	23.65
-8.00	5.00	1487.57	139.47	28.72
-6.00	5.00	1323.01	133.16	34.60
-4.00	5.00	1183.72	123.92	41.13
-2.00	5.00	1079.77	110.04	47.58
0.00	5.00	1022.16	90.11	52.10
2.00	5.00	1018.88	66.59	52.39
4.00	5.00	1070.46	46.00	48.25
6.00	5.00	1169.62	31.46	41.91
8.00	5.00	1305.47	21.79	35.32
10.00	5.00	1467.74	15.24	29.35
12.00	5.00	1648.48	10.61	24.18
14.00	5.00	1842.07	7.20	19.74
16.00	5.00	2044.68	4.60	15.91
18.00	5.00	2253.65	2.56	12.56
20.00	5.00	2467.16	0.92	9.59
22.00	5.00	2683.88	359.58	6.92
24.00	5.00	2902.85	358.46	4.49
26.00	5.00	3123.37	357.51	2.25

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 10.15 MINUTES

OBSERVER LATITUDE IS 0.01 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
-22.00	10.00	3191.37	146.71	1.60
-20.00	10.00	2985.26	144.81	3.63
-18.00	10.00	2782.48	142.63	5.79
-16.00	10.00	2584.14	140.09	8.11
-14.00	10.00	2391.62	137.12	10.59
-12.00	10.00	2206.76	133.58	13.27
-10.00	10.00	2031.92	129.32	16.13
-8.00	10.00	1870.21	124.14	19.17
-6.00	10.00	1725.58	117.82	22.30
-4.00	10.00	1602.89	110.09	25.38
-2.00	10.00	1507.71	100.82	28.10
0.00	10.00	1445.64	90.06	30.08
2.00	10.00	1421.15	78.31	30.91
4.00	10.00	1436.23	66.44	30.40
6.00	10.00	1489.67	55.39	28.66
8.00	10.00	1577.56	45.74	26.07
10.00	10.00	1694.47	37.66	23.04
12.00	10.00	1834.75	31.04	19.90
14.00	10.00	1993.34	25.64	16.82
16.00	10.00	2166.06	21.22	13.90
18.00	10.00	2349.62	17.57	11.17
20.00	10.00	2541.48	14.52	8.64
22.00	10.00	2739.70	11.95	6.27
24.00	10.00	2942.78	9.76	4.07
26.00	10.00	3149.56	7.87	2.00

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 9.59 MINUTES

OBSERVER LATITUDE IS 0.01 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
-20.00	15.00	3337.11	136.85	0.25
-18.00	15.00	3151.27	134.31	1.98
-16.00	15.00	2970.98	131.45	3.78
-14.00	15.00	2797.57	128.22	5.63
-12.00	15.00	2632.67	124.56	7.52
-10.00	15.00	2478.24	120.39	9.44
-8.00	15.00	2336.60	115.64	11.36
-6.00	15.00	2210.44	110.25	13.21
-4.00	15.00	2102.75	104.17	14.92
-2.00	15.00	2016.67	97.41	16.40
0.00	15.00	1955.21	90.04	17.52
2.00	15.00	1920.85	82.22	18.17
4.00	15.00	1915.13	74.20	18.28
6.00	15.00	1938.33	66.27	17.84
8.00	15.00	1989.47	58.70	16.89
10.00	15.00	2066.43	51.69	15.53
12.00	15.00	2166.40	45.34	13.89
14.00	15.00	2286.28	39.70	12.08
16.00	15.00	2423.00	34.73	10.17
18.00	15.00	2573.75	30.37	8.23
20.00	15.00	2736.05	26.56	6.32
22.00	15.00	2907.80	23.22	4.44
24.00	15.00	3087.26	20.28	2.61
26.00	15.00	3272.95	17.68	0.84

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 7.33 MINUTES

OBSERVER LATITUDE IS 0.01 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
-14.00	20.00	3252.25	121.99	1.03
-12.00	20.00	3105.35	118.50	2.43
-10.00	20.00	2968.45	114.68	3.80
-8.00	20.00	2843.23	110.49	5.13
-6.00	20.00	2731.50	105.93	6.37
-4.00	20.00	2635.15	100.98	7.49
-2.00	20.00	2556.09	95.66	8.45
0.00	20.00	2496.10	90.03	9.21
2.00	20.00	2456.69	84.14	9.72
4.00	20.00	2438.96	78.10	9.96
6.00	20.00	2443.43	72.03	9.90
8.00	20.00	2470.00	66.04	9.55
10.00	20.00	2517.99	60.25	8.93
12.00	20.00	2586.18	54.74	8.08
14.00	20.00	2672.97	49.58	7.04
16.00	20.00	2776.54	44.79	5.86
18.00	20.00	2894.99	40.40	4.57
20.00	20.00	3026.48	36.39	3.21
22.00	20.00	3169.24	32.74	1.81
24.00	20.00	3321.68	29.43	0.39

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 0.00 MINUTES

OBSERVER LATITUDE IS 0.01 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
-6.00	25.00	3268.51	103.17	0.88
-4.00	25.00	3180.79	99.00	1.70
-2.00	25.00	3107.13	94.61	2.41
0.00	25.00	3048.69	90.02	2.99
2.00	25.00	3006.47	85.28	3.42
4.00	25.00	2981.25	80.42	3.67
6.00	25.00	2973.53	75.51	3.75
8.00	25.00	2983.48	70.60	3.65
10.00	25.00	3010.97	65.75	3.37
12.00	25.00	3055.52	61.03	2.92
14.00	25.00	3116.38	56.47	2.32
16.00	25.00	3192.56	52.11	1.59
18.00	25.00	3282.95	47.97	0.74

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 0.00 MINUTES

OBSERVER LATITUDE IS 30.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
12.00	-20.00	3363.65	234.15	0.01
14.00	-20.00	3252.85	238.00	1.02
16.00	-20.00	3154.18	242.10	1.96
18.00	-20.00	3068.91	246.44	2.79
20.00	-20.00	2998.26	251.02	3.50
22.00	-20.00	2943.33	255.80	4.06
24.00	-20.00	2905.05	260.73	4.47
26.00	-20.00	2884.09	265.77	4.69
28.00	-20.00	2880.81	270.85	4.72
30.00	-20.00	2895.25	275.91	4.57
32.00	-20.00	2927.09	280.89	4.23
34.00	-20.00	2975.70	285.73	3.73
36.00	-20.00	3040.17	290.38	3.08
38.00	-20.00	3119.42	294.82	2.29
40.00	-20.00	3212.22	299.02	1.40
42.00	-20.00	3317.30	302.98	0.43

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 0.00 MINUTES

OBSERVER LATITUDE IS 30.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
8.00	-15.00	3287.50	218.92	0.70
10.00	-15.00	3136.11	222.29	2.13
12.00	-15.00	2994.20	225.99	3.54
14.00	-15.00	2863.35	230.06	4.91
16.00	-15.00	2745.26	234.51	6.21
18.00	-15.00	2641.79	239.35	7.41
20.00	-15.00	2554.79	244.57	8.47
22.00	-15.00	2486.08	250.15	9.34
24.00	-15.00	2437.27	256.02	9.98
26.00	-15.00	2409.58	262.10	10.35
28.00	-15.00	2403.73	268.26	10.43
30.00	-15.00	2419.87	274.40	10.21
32.00	-15.00	2457.49	280.39	9.71
34.00	-15.00	2515.56	286.13	8.96
36.00	-15.00	2592.60	291.55	8.00
38.00	-15.00	2686.85	296.61	6.88
40.00	-15.00	2796.42	301.28	5.64
42.00	-15.00	2919.45	305.56	4.31
44.00	-15.00	3054.12	309.48	2.94
46.00	-15.00	3198.79	313.06	1.53
48.00	-15.00	3351.97	316.32	0.11

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 1.69 MINUTES

OBSERVER LATITUDE IS 30.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
6.00	-10.00	3198.25	206.21	1.54
8.00	-10.00	3018.02	209.03	3.30
10.00	-10.00	2844.63	212.21	5.11
12.00	-10.00	2679.60	215.80	6.97
14.00	-10.00	2524.77	219.89	8.85
16.00	-10.00	2382.32	224.52	10.72
18.00	-10.00	2254.75	229.77	12.54
20.00	-10.00	2144.87	235.68	14.24
22.00	-10.00	2055.62	242.25	15.72
24.00	-10.00	1989.86	249.41	16.88
26.00	-10.00	1950.01	257.04	17.62
28.00	-10.00	1937.69	264.92	17.85
30.00	-10.00	1953.40	272.78	17.55
32.00	-10.00	1996.42	280.36	16.76
34.00	-10.00	2064.96	287.46	15.56
36.00	-10.00	2156.47	293.95	14.05
38.00	-10.00	2268.03	299.79	12.34
40.00	-10.00	2396.69	304.98	10.52
42.00	-10.00	2539.68	309.56	8.66
44.00	-10.00	2694.52	313.61	6.79
46.00	-10.00	2859.09	317.20	4.96
48.00	-10.00	3031.60	320.37	3.16
50.00	-10.00	3210.54	323.21	1.42

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 7.33 MINUTES

OBSERVER LATITUDE IS 30.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
4.00	-5.00	3231.11	193.24	1.23
6.00	-5.00	3027.53	195.24	3.20
8.00	-5.00	2827.68	197.52	5.30
10.00	-5.00	2632.67	200.17	7.52
12.00	-5.00	2443.90	203.26	9.89
14.00	-5.00	2263.18	206.91	12.41
16.00	-5.00	2092.83	211.24	15.09
18.00	-5.00	1935.79	216.44	17.88
20.00	-5.00	1795.77	222.69	20.72
22.00	-5.00	1677.18	230.17	23.46
24.00	-5.00	1584.97	238.97	25.86
26.00	-5.00	1524.03	249.01	27.61
28.00	-5.00	1498.21	259.87	28.39
30.00	-5.00	1509.33	270.88	28.05
32.00	-5.00	1556.53	281.31	26.66
34.00	-5.00	1636.61	290.64	24.49
36.00	-5.00	1744.93	298.65	21.86
38.00	-5.00	1876.45	305.38	19.04
40.00	-5.00	2026.48	311.00	16.23
42.00	-5.00	2191.04	315.69	13.51
44.00	-5.00	2366.88	319.63	10.93
46.00	-5.00	2551.46	322.97	8.51
48.00	-5.00	2742.80	325.83	6.24
50.00	-5.00	2939.33	328.31	4.11
52.00	-5.00	3139.84	330.48	2.09
54.00	-5.00	3343.38	332.39	0.19

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 9.02 MINUTES

OBSERVER LATITUDE IS 30.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
4.00	0.00	3166.82	182.06	1.84
6.00	0.00	2949.76	183.33	4.00
8.00	0.00	2734.60	184.79	6.33
10.00	0.00	2522.12	186.52	8.88
12.00	0.00	2313.36	188.59	11.69
14.00	0.00	2109.70	191.11	14.81
16.00	0.00	1913.05	194.24	18.32
18.00	0.00	1726.08	198.24	22.29
20.00	0.00	1552.53	203.46	26.78
22.00	0.00	1397.65	210.47	31.74
24.00	0.00	1268.50	220.06	36.94
26.00	0.00	1173.74	233.08	41.68
28.00	0.00	1122.23	249.71	44.73
30.00	0.00	1119.97	268.13	44.87
32.00	0.00	1167.23	285.03	42.04
34.00	0.00	1258.36	298.39	37.40
36.00	0.00	1384.58	308.26	32.22
38.00	0.00	1537.13	315.48	27.22
40.00	0.00	1708.78	320.86	22.70
42.00	0.00	1894.14	324.98	18.69
44.00	0.00	2089.34	328.22	15.15
46.00	0.00	2291.65	330.83	12.00
48.00	0.00	2499.09	332.99	9.17
50.00	0.00	2710.26	334.80	6.61
52.00	0.00	2924.09	336.35	4.26
54.00	0.00	3139.78	337.70	2.10
56.00	0.00	3356.73	338.88	0.07

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 10.15 MINUTES

OBSERVER LATITUDE IS 30.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
4.00	5.00	3197.01	170.77	1.55
6.00	5.00	2972.83	171.19	3.76
8.00	5.00	2749.39	171.68	6.16
10.00	5.00	2527.20	172.26	8.82
12.00	5.00	2306.94	172.96	11.78
14.00	5.00	2089.54	173.82	15.14
16.00	5.00	1876.29	174.90	19.05
18.00	5.00	1669.11	176.31	23.66
20.00	5.00	1470.85	178.25	29.25
22.00	5.00	1285.94	181.08	36.16
24.00	5.00	1121.29	185.61	44.79
26.00	5.00	987.30	193.97	55.34
28.00	5.00	898.00	212.95	66.79
30.00	5.00	867.39	258.94	73.01
32.00	5.00	901.49	303.33	66.20
34.00	5.00	993.61	321.37	54.72
36.00	5.00	1129.51	329.42	44.27
38.00	5.00	1295.34	333.85	35.75
40.00	5.00	1480.90	336.64	28.93
42.00	5.00	1679.44	338.57	23.41
44.00	5.00	1886.66	340.00	18.84
46.00	5.00	2099.76	341.11	14.97
48.00	5.00	2316.88	342.00	11.63
50.00	5.00	2536.73	342.75	8.70
52.00	5.00	2758.41	343.38	6.06
54.00	5.00	2981.27	343.93	3.67
56.00	5.00	3204.79	344.43	1.47

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 10.71 MINUTES

OBSERVER LATITUDE IS 30.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
4.00	10.00	3318.87	159.99	0.41
6.00	10.00	3094.32	159.63	2.54
8.00	10.00	2870.27	159.22	4.84
10.00	10.00	2647.20	158.72	7.35
12.00	10.00	2425.68	158.12	10.13
14.00	10.00	2206.50	157.39	13.27
16.00	10.00	1990.75	156.45	16.86
18.00	10.00	1780.03	155.23	21.07
20.00	10.00	1576.64	153.58	26.09
22.00	10.00	1384.14	151.21	32.23
24.00	10.00	1208.06	147.57	39.85
26.00	10.00	1056.89	141.34	49.27
28.00	10.00	942.93	128.84	60.27
30.00	10.00	880.90	99.08	69.98
32.00	10.00	881.91	49.41	69.78
34.00	10.00	945.75	20.45	59.92
36.00	10.00	1061.06	8.26	48.95
38.00	10.00	1213.14	2.16	39.59
40.00	10.00	1389.81	358.59	32.02
42.00	10.00	1582.66	356.28	25.93
44.00	10.00	1786.26	354.68	20.93
46.00	10.00	1997.11	353.51	16.75
48.00	10.00	2212.91	352.63	13.17
50.00	10.00	2432.09	351.95	10.05
52.00	10.00	2653.58	351.41	7.27
54.00	10.00	2876.59	350.98	4.77
56.00	10.00	3100.55	350.63	2.48

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 11.28 MINUTES

OBSERVER LATITUDE IS 30.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
6.00	15.00	3302.54	149.24	0.56
8.00	15.00	3083.98	148.13	2.64
10.00	15.00	2866.90	146.85	4.87
12.00	15.00	2652.00	145.34	7.29
14.00	15.00	2440.15	143.55	9.94
16.00	15.00	2232.55	141.37	12.87
18.00	15.00	2030.80	138.68	16.15
20.00	15.00	1837.13	135.27	19.85
22.00	15.00	1654.70	130.87	24.02
24.00	15.00	1487.92	125.01	28.71
26.00	15.00	1342.88	117.04	33.80
28.00	15.00	1227.56	106.11	38.87
30.00	15.00	1151.09	91.54	42.97
32.00	15.00	1121.56	73.95	44.77
34.00	15.00	1142.67	56.08	43.47
36.00	15.00	1211.76	40.91	39.66
38.00	15.00	1321.26	29.41	34.67
40.00	15.00	1461.99	21.02	29.54
42.00	15.00	1625.73	14.88	24.77
44.00	15.00	1806.06	10.30	20.50
46.00	15.00	1998.30	6.78	16.73
48.00	15.00	2199.13	4.03	13.38
50.00	15.00	2406.17	1.82	10.39
52.00	15.00	2617.72	0.02	7.70
54.00	15.00	2832.55	358.53	5.24
56.00	15.00	3049.71	357.28	2.98

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 10.15 MINUTES

OBSERVER LATITUDE IS 30.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
10.00	20.00	3164.17	136.93	1.86
12.00	20.00	2959.87	134.93	3.89
14.00	20.00	2759.55	132.62	6.05
16.00	20.00	2564.41	129.92	8.35
18.00	20.00	2376.04	126.74	10.81
20.00	20.00	2196.46	122.95	13.42
22.00	20.00	2028.29	118.40	16.19
24.00	20.00	1874.84	112.91	19.07
26.00	20.00	1740.26	106.26	21.96
28.00	20.00	1629.44	98.28	24.67
30.00	20.00	1547.67	88.93	26.92
32.00	20.00	1499.82	78.42	28.34
34.00	20.00	1489.27	67.31	28.67
36.00	20.00	1516.82	56.38	27.82
38.00	20.00	1580.46	46.34	25.99
40.00	20.00	1676.03	37.61	23.49
42.00	20.00	1798.35	30.26	20.67
44.00	20.00	1942.25	24.17	17.76
46.00	20.00	2103.15	19.15	14.92
48.00	20.00	2277.28	15.00	12.21
50.00	20.00	2461.65	11.54	9.66
52.00	20.00	2653.92	8.63	7.27
54.00	20.00	2852.29	6.17	5.03
56.00	20.00	3055.35	4.06	2.92

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 9.02 MINUTES

OBSERVER LATITUDE IS 30.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
12.00	25.00	3324.75	126.59	0.36
14.00	25.00	3136.59	124.10	2.13
16.00	25.00	2954.12	121.29	3.95
18.00	25.00	2778.71	118.10	5.84
20.00	25.00	2612.08	114.46	7.77
22.00	25.00	2456.24	110.29	9.73
24.00	25.00	2313.63	105.52	11.68
26.00	25.00	2187.05	100.09	13.57
28.00	25.00	2079.64	93.93	15.31
30.00	25.00	1994.65	87.07	16.79
32.00	25.00	1935.18	79.59	17.90
34.00	25.00	1903.73	71.67	18.50
36.00	25.00	1901.75	63.57	18.54
38.00	25.00	1929.35	55.61	18.01
40.00	25.00	1985.30	48.06	16.96
42.00	25.00	2067.24	41.12	15.52
44.00	25.00	2172.15	34.88	13.80
46.00	25.00	2296.79	29.37	11.92
48.00	25.00	2437.99	24.54	9.97
50.00	25.00	2592.91	20.33	8.00
52.00	25.00	2759.07	16.66	6.06
54.00	25.00	2934.38	13.45	4.16
56.00	25.00	3117.10	10.64	2.32

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 6.77 MINUTES

OBSERVER LATITUDE IS 30.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
18.00	30.00	3214.86	111.61	1.38
20.00	30.00	3058.65	108.25	2.89
22.00	30.00	2912.21	104.53	4.39
24.00	30.00	2777.32	100.41	5.85
26.00	30.00	2655.95	95.87	7.24
28.00	30.00	2550.22	90.88	8.53
30.00	30.00	2462.31	85.46	9.65
32.00	30.00	2394.33	79.65	10.56
34.00	30.00	2348.13	73.52	11.19
36.00	30.00	2325.09	67.18	11.52
38.00	30.00	2325.95	60.78	11.51
40.00	30.00	2350.70	54.45	11.16
42.00	30.00	2398.61	48.34	10.50
44.00	30.00	2468.28	42.56	9.57
46.00	30.00	2557.88	37.18	8.43
48.00	30.00	2665.30	32.24	7.13
50.00	30.00	2788.38	27.75	5.73
52.00	30.00	2925.01	23.68	4.26
54.00	30.00	3073.24	20.02	2.75
56.00	30.00	3231.31	16.73	1.22

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 3.38 MINUTES

OBSERVER LATITUDE IS 45.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
36.00	-25.00	3316.97	261.67	0.43
38.00	-25.00	3259.92	265.97	0.96
40.00	-25.00	3217.11	270.39	1.36
42.00	-25.00	3189.16	274.91	1.62
44.00	-25.00	3176.45	279.49	1.74
46.00	-25.00	3179.14	284.08	1.72
48.00	-25.00	3197.17	288.64	1.55
50.00	-25.00	3230.23	293.13	1.23
52.00	-25.00	3277.80	297.52	0.79
54.00	-25.00	3339.16	301.76	0.23

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 0.00 MINUTES

OBSERVER LATITUDE IS 45.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
28.00	-20.00	3320.69	239.83	0.40
30.00	-20.00	3204.38	243.66	1.48
32.00	-20.00	3099.82	247.76	2.48
34.00	-20.00	3008.33	252.13	3.40
36.00	-20.00	2931.24	256.76	4.19
38.00	-20.00	2869.76	261.61	4.84
40.00	-20.00	2824.96	266.66	5.33
42.00	-20.00	2797.67	271.83	5.63
44.00	-20.00	2788.40	277.08	5.73
46.00	-20.00	2797.31	282.32	5.63
48.00	-20.00	2824.18	287.50	5.33
50.00	-20.00	2868.45	292.54	4.85
52.00	-20.00	2929.25	297.40	4.21
54.00	-20.00	3005.48	302.03	3.43
56.00	-20.00	3095.87	306.41	2.52
58.00	-20.00	3199.10	310.53	1.53
60.00	-20.00	3313.83	314.38	0.46

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 0.00 MINUTES

OBSERVER LATITUDE IS 45.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
24.00	-15.00	3286.43	225.14	0.71
26.00	-15.00	3135.92	228.52	2.13
28.00	-15.00	2994.74	232.24	3.53
30.00	-15.00	2864.44	236.31	4.90
32.00	-15.00	2746.68	240.75	6.20
34.00	-15.00	2643.29	245.59	7.39
36.00	-15.00	2556.11	250.81	8.45
38.00	-15.00	2486.92	256.38	9.33
40.00	-15.00	2437.31	262.24	9.98
42.00	-15.00	2408.52	268.30	10.36
44.00	-15.00	2401.30	274.47	10.46
46.00	-15.00	2415.83	280.62	10.26
48.00	-15.00	2451.66	286.62	9.79
50.00	-15.00	2507.81	292.39	9.06
52.00	-15.00	2582.86	297.85	8.12
54.00	-15.00	2675.10	302.94	7.02
56.00	-15.00	2782.70	307.66	5.79
58.00	-15.00	2903.79	311.99	4.48
60.00	-15.00	3036.59	315.96	3.11
62.00	-15.00	3179.48	319.58	1.71
64.00	-15.00	3330.96	322.89	0.30

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 1.69 MINUTES

OBSERVER LATITUDE IS 45.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
22.00	-10.00	3204.84	213.13	1.47
24.00	-10.00	3030.40	216.09	3.17
26.00	-10.00	2863.24	219.41	4.91
28.00	-10.00	2704.90	223.14	6.67
30.00	-10.00	2557.22	227.34	8.44
32.00	-10.00	2422.30	232.07	10.18
34.00	-10.00	2302.55	237.36	11.84
36.00	-10.00	2200.58	243.24	13.36
38.00	-10.00	2119.06	249.71	14.66
40.00	-10.00	2060.47	256.68	15.64
42.00	-10.00	2026.87	264.01	16.22
44.00	-10.00	2019.50	271.52	16.35
46.00	-10.00	2038.62	278.98	16.01
48.00	-10.00	2083.45	286.18	15.25
50.00	-10.00	2152.30	292.93	14.12
52.00	-10.00	2242.85	299.15	12.72
54.00	-10.00	2352.47	304.78	11.13
56.00	-10.00	2478.47	309.82	9.44
58.00	-10.00	2618.32	314.33	7.69
60.00	-10.00	2769.75	318.33	5.94
62.00	-10.00	2930.76	321.91	4.20
64.00	-10.00	3099.67	325.10	2.49
66.00	-10.00	3275.04	327.96	0.82

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 6.77 MINUTES

OBSERVER LATITUDE IS 45.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
20.00	-5.00	3215.07	200.75	1.38
22.00	-5.00	3019.53	203.07	3.28
24.00	-5.00	2828.70	205.72	5.28
26.00	-5.00	2643.83	208.74	7.39
28.00	-5.00	2466.49	212.24	9.59
30.00	-5.00	2298.65	216.31	11.90
32.00	-5.00	2142.76	221.06	14.27
34.00	-5.00	2001.79	226.62	16.66
36.00	-5.00	1879.26	233.11	18.99
38.00	-5.00	1779.13	240.60	21.09
40.00	-5.00	1705.44	249.05	22.78
42.00	-5.00	1661.78	258.28	23.85
44.00	-5.00	1650.56	267.92	24.13
46.00	-5.00	1672.41	277.47	23.58
48.00	-5.00	1726.02	286.49	22.29
50.00	-5.00	1808.46	294.64	20.45
52.00	-5.00	1915.91	301.81	18.26
54.00	-5.00	2044.27	308.00	15.92
56.00	-5.00	2189.71	313.30	13.53
58.00	-5.00	2348.86	317.84	11.18
60.00	-5.00	2518.94	321.74	8.92
62.00	-5.00	2697.66	325.11	6.76
64.00	-5.00	2883.20	328.04	4.70
66.00	-5.00	3074.11	330.62	2.74
68.00	-5.00	3269.21	332.91	0.87

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 8.46 MINUTES

OBSERVER LATITUDE IS 45.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
18.00	0.00	3321.20	188.66	0.39
20.00	0.00	3109.93	190.23	2.39
22.00	0.00	2901.00	192.03	4.51
24.00	0.00	2695.22	194.11	6.78
26.00	0.00	2493.64	196.54	9.24
28.00	0.00	2297.63	199.43	11.91
30.00	0.00	2109.01	202.90	14.82
32.00	0.00	1930.19	207.13	17.99
34.00	0.00	1764.39	212.37	21.42
36.00	0.00	1615.81	218.92	25.03
38.00	0.00	1489.82	227.14	28.65
40.00	0.00	1392.69	237.32	31.92
42.00	0.00	1330.87	249.43	34.28
44.00	0.00	1309.40	262.81	35.16
46.00	0.00	1330.24	276.20	34.30
48.00	0.00	1391.43	288.33	31.97
50.00	0.00	1487.91	298.54	28.71
52.00	0.00	1613.22	306.80	25.10
54.00	0.00	1761.06	313.39	21.49
56.00	0.00	1926.06	318.66	18.07
58.00	0.00	2103.99	322.94	14.90
60.00	0.00	2291.64	326.45	12.00
62.00	0.00	2486.57	329.38	9.33
64.00	0.00	2686.97	331.86	6.88
66.00	0.00	2891.48	333.99	4.61
68.00	0.00	3099.03	335.85	2.49
70.00	0.00	3308.80	337.48	0.51

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 9.59 MINUTES

OBSERVER LATITUDE IS 45.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
18.00	5.00	3297.34	178.29	0.61
20.00	5.00	3077.35	179.27	2.71
22.00	5.00	2858.50	180.39	4.96
24.00	5.00	2641.39	181.70	7.42
26.00	5.00	2426.78	183.25	10.12
28.00	5.00	2215.68	185.12	13.13
30.00	5.00	2009.50	187.42	16.53
32.00	5.00	1810.19	190.32	20.41
34.00	5.00	1620.57	194.11	24.91
36.00	5.00	1444.73	199.22	30.11
38.00	5.00	1288.56	206.41	36.05
40.00	5.00	1160.22	216.88	42.44
42.00	5.00	1069.94	232.28	48.29
44.00	5.00	1027.89	253.09	51.61
46.00	5.00	1039.97	275.63	50.61
48.00	5.00	1104.34	294.20	45.89
50.00	5.00	1212.62	307.23	39.61
52.00	5.00	1354.18	316.08	33.36
54.00	5.00	1519.58	322.25	27.74
56.00	5.00	1701.68	326.73	22.87
58.00	5.00	1895.47	330.11	18.66
60.00	5.00	2097.50	332.76	15.01
62.00	5.00	2305.36	334.88	11.80
64.00	5.00	2517.37	336.64	8.94
66.00	5.00	2732.33	338.13	6.36
68.00	5.00	2949.33	339.40	4.00
70.00	5.00	3167.72	340.51	1.83

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 10.71 MINUTES

OBSERVER LATITUDE IS 45.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
18.00	10.00	3344.37	167.99	0.18
20.00	10.00	3119.82	168.34	2.29
22.00	10.00	2895.71	168.74	4.56
24.00	10.00	2672.45	169.19	7.05
26.00	10.00	2450.60	169.72	9.80
28.00	10.00	2230.87	170.36	12.90
30.00	10.00	2014.29	171.13	16.44
32.00	10.00	1802.30	172.11	20.58
34.00	10.00	1597.02	173.40	25.53
36.00	10.00	1401.72	175.19	31.59
38.00	10.00	1221.48	177.87	39.17
40.00	10.00	1064.24	182.36	48.71
42.00	10.00	941.80	191.45	60.41
44.00	10.00	869.18	216.49	72.58
46.00	10.00	859.16	279.90	75.20
48.00	10.00	913.80	315.86	64.25
50.00	10.00	1022.71	327.75	52.05
52.00	10.00	1170.71	333.19	41.85
54.00	10.00	1344.79	336.30	33.73
56.00	10.00	1535.92	338.32	27.26
58.00	10.00	1738.27	339.76	22.01
60.00	10.00	1948.14	340.85	17.65
62.00	10.00	2163.09	341.72	13.95
64.00	10.00	2381.52	342.43	10.73
66.00	10.00	2602.29	343.04	7.89
68.00	10.00	2824.62	343.57	5.33
70.00	10.00	3047.90	344.03	3.00
72.00	10.00	3271.69	344.45	0.85

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 10.71 MINUTES

OBSERVER LATITUDE IS 45.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
20.00	15.00	3234.09	157.91	1.20
22.00	15.00	3009.39	157.64	3.39
24.00	15.00	2785.33	157.31	5.76
26.00	15.00	2562.38	156.92	8.38
28.00	15.00	2341.20	156.43	11.29
30.00	15.00	2122.65	155.80	14.60
32.00	15.00	1907.96	154.99	18.42
34.00	15.00	1698.90	153.91	22.94
36.00	15.00	1498.16	152.38	28.39
38.00	15.00	1309.86	150.12	35.14
40.00	15.00	1140.48	146.46	43.60
42.00	15.00	999.92	139.69	54.11
44.00	15.00	901.97	124.09	66.12
46.00	15.00	861.42	80.45	74.57
48.00	15.00	886.22	28.14	68.93
50.00	15.00	971.35	8.17	56.99
52.00	15.00	1102.83	0.07	45.99
54.00	15.00	1266.16	355.89	37.04
56.00	15.00	1450.44	353.39	29.92
58.00	15.00	1648.47	351.75	24.18
60.00	15.00	1855.64	350.60	19.46
62.00	15.00	2068.98	349.77	15.49
64.00	15.00	2286.53	349.14	12.07
66.00	15.00	2506.95	348.67	9.07
68.00	15.00	2729.31	348.30	6.39
70.00	15.00	2952.90	348.01	3.96
72.00	15.00	3177.22	347.78	1.74

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 10.71 MINUTES

OBSERVER LATITUDE IS 45.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
22.00	20.00	3190.59	147.51	1.61
24.00	20.00	2969.92	146.58	3.79
26.00	20.00	2750.66	145.49	6.15
28.00	20.00	2533.51	144.19	8.74
30.00	20.00	2319.38	142.61	11.60
32.00	20.00	2109.48	140.65	14.81
34.00	20.00	1905.54	138.17	18.47
36.00	20.00	1710.00	134.93	22.67
38.00	20.00	1526.39	130.59	27.54
40.00	20.00	1359.81	124.53	33.14
42.00	20.00	1217.55	115.76	39.36
44.00	20.00	1109.20	102.80	45.57
46.00	20.00	1045.54	84.38	50.16
48.00	20.00	1034.91	62.22	51.02
50.00	20.00	1078.91	41.89	47.64
52.00	20.00	1171.34	26.91	41.81
54.00	20.00	1301.84	16.74	35.48
56.00	20.00	1460.09	9.80	29.60
58.00	20.00	1637.90	4.91	24.45
60.00	20.00	1829.39	1.32	20.01
62.00	20.00	2030.49	358.62	16.16
64.00	20.00	2238.39	356.51	12.78
66.00	20.00	2451.13	354.84	9.80
68.00	20.00	2667.31	353.48	7.11
70.00	20.00	2885.93	352.36	4.67
72.00	20.[illegible]	3106.22	351.43	2.42

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 10.15 MINUTES

OBSERVER LATITUDE IS 45.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
24.00	25.00	3212.63	137.18	1.40
26.00	25.00	2999.81	135.65	3.48
28.00	25.00	2789.64	133.87	5.71
30.00	25.00	2583.08	131.77	8.12
32.00	25.00	2381.37	129.27	10.73
34.00	25.00	2186.16	126.25	13.58
36.00	25.00	1999.63	122.54	16.70
38.00	25.00	1824.74	117.91	20.10
40.00	25.00	1665.42	112.07	23.76
42.00	25.00	1526.79	104.65	27.53
44.00	25.00	1415.16	95.27	31.12
46.00	25.00	1337.46	83.77	34.01
48.00	25.00	1299.92	70.56	35.06
50.00	25.00	1306.05	56.80	35.30
52.00	25.00	1355.28	43.98	33.32
54.00	25.00	1443.13	33.07	30.16
56.00	25.00	1563.03	24.27	26.48
58.00	25.00	1708.12	17.35	22.71
60.00	25.00	1872.38	11.90	19.12
62.00	25.00	2051.04	7.58	15.80
64.00	25.00	2240.47	4.11	12.75
66.00	25.00	2437.96	1.28	9.97
68.00	25.00	2641.46	358.94	7.42
70.00	25.00	2849.48	356.98	5.06
72.00	25.00	3060.85	355.33	2.87

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 9.59 MINUTES

OBSERVER LATITUDE IS 45.00 DEGREES
SATELLITE ALTITUDE IS 834.00 KM
INCLINATION OF SATELLITE ORBITAL PLANE IS 99.00 DEGREES
PERIOD OF SATELLITE IS 101.51 MINUTES
EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS 27.84 DEGREES
SATELLITE ANGULAR VELOCITY IS 3.55 DEGREES PER MINUTE

SATELLITE ANGULAR POSITION	SATELLITE EQUATOR CROSSING	SLANT RANGE TO SATELLITE	AZIMUTH ANGLE	ELEVATION ANGLE
26.00	30.00	3294.39	127.29	0.64
28.00	30.00	3092.09	125.28	2.56
30.00	30.00	2893.82	122.98	4.58
32.00	30.00	2700.75	120.31	6.72
34.00	30.00	2514.38	117.21	8.98
36.00	30.00	2336.57	113.55	11.36
38.00	30.00	2169.71	109.23	13.84
40.00	30.00	2016.75	104.10	16.40
42.00	30.00	1881.33	98.00	18.94
44.00	30.00	1767.68	90.81	21.34
46.00	30.00	1680.39	82.50	23.38
48.00	30.00	1623.87	73.19	24.82
50.00	30.00	1601.44	63.25	25.41
52.00	30.00	1614.59	53.23	25.06
54.00	30.00	1662.46	43.72	23.83
56.00	30.00	1742.14	35.13	21.92
58.00	30.00	1849.45	27.65	19.59
60.00	30.00	1979.79	21.29	17.06
62.00	30.00	2128.78	15.93	14.50
64.00	30.00	2292.64	11.44	11.98
66.00	30.00	2468.22	7.65	9.57
68.00	30.00	2653.01	4.44	7.28
70.00	30.00	2845.01	1.71	5.11
72.00	30.00	3042.62	359.36	3.05

SATELLITE VIEWING TIME ABOVE 10.00 DEGREES IS 8.46 MINUTES

```
10 REM THIS PROGRAM COMPUTES AND PLOTS THE AZIMUTH AND ELEVATION ANGLES
20 REM NECESSARY TO LOCATE A SATELLITE IN A CIRCULAR ORBIT. THE FOLLOWING
30 REM PARAMETERS MUST BE INPUT:OBSERVER LATITUDE, L0, (0<ANGLE<90); INITIAL
40 REM EQUATOR CROSSING LONGITUDE RELATIVE TO OBSERVER, E0, (EAST IS +), ANGLE
50 REM BETWEEN ORBITAL PLANE AND EQUATORIAL PLANE, P0, AND THE
60 REM ALTITUDE OF THE SATELLITE IN KM, H.  THE FOLLOWING DATA IS PRINTED:
70 REM THE PERIOD OF THE SATELLITE, THE ANGULAR VELOCITY OF THE SATELLITE
80 REM (DEGREES PER MIN), OBSERVER LATITUDE, SATELLITE EQUATOR CROSSING
90 REM LONGITUDE, SLANT RANGE TO SATELLITE(KM), AZIMUTH, AND ELEVATION.
100 REM THE INDEPENDENT VARIABLE IS THE SATELLITE'S ANGULAR DISPLACEMENT
110 REM ABOVE THE EQUATORIAL PLANE.  THE INITIAL VALUE FOR THIS ANGLE IS
120 REM COMPUTED FROM SATELLITE ALTITUDE AND OBSERVER LATITUDE.  THE AZIMUTH
130 REM AND ELEVATION CAN BE PLOTTED ON POLAR PAPER. THE LOWER LEFT CORNER
140 REM IS 225 DEG AZIMUTH, 127 ELV; THE UPPER RIGHT CORNER IS 45 DEG AZ,
150 REM 127 ELV. THE MINIMUM ELEVATION ANGLE IS DESIGNATED A9.
160 DEG
170 FIXED 2
180 SCALE -90,90,-90,90
190 PEN
200 DISP "L0 EQUALS";
210 INPUT L0
220 DISP "E0 EQUALS";
230 INPUT E0
240 DISP "P0 EQUALS";
250 INPUT P0
260 DISP "H EQUALS";
270 INPUT H
280 DISP "A9 EQUALS";
290 INPUT A9
300 R0=6370
310 G0=0.0098062
320 W1=0.25
330 E1=-E0
340 R=R0+H
350 T=2*PI/60*SQR(R↑3/(G0*R0↑2))
360 O=ATN(SQR(1-(R0/R)↑2)/(R0/R))
370 W=360/T
380 PRINT "OBSERVER LATITUDE IS"L0" DEGREES"
390 PRINT "SATELLITE ALTITUDE IS"H" KM"
400 PRINT "INCLINATION OF SATELLITE ORBITAL PLANE IS "P0" DEGREES"
410 PRINT "PERIOD OF SATELLITE IS"T" MINUTES"
420 PRINT "EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS"O" DEGREES"
430 PRINT "SATELLITE ANGULAR VELOCITY IS"W" DEGREES PER MINUTE"
440 PRINT
450 PRINT
460 PRINT "SATELLITE      SATELLITE      SLANT RANGE    AZIMUTH        ELEVATION"
470 PRINT "ANGULAR        EQUATOR            TO          ANGLE          ANGLE   "
480 PRINT "POSITION       CROSSING       SATELLITE                              "
490 PRINT
500 F=0
510 A0=90-L0
520 X0=R0*SIN(A0)
530 Z0=R0*COS(A0)
540 C=10*(INT((L0-O)/10))
550 E=E0-W1/W*C
560 X=R*(COS(C)*COS(E)-SIN(C)*COS(P0)*SIN(E))
570 Y=R*(COS(C)*SIN(E)+SIN(C)*COS(P0)*COS(E))
580 Z=R*(SIN(C)*SIN(P0))
590 X1=X-X0
600 Z1=Z-Z0
610 R1=SQR(X1↑2+Y↑2+Z1↑2)
620 X2=X1*COS(A0)-Z1*SIN(A0)
630 Z2=Z1*COS(A0)+X1*SIN(A0)
640 A1=ATN(SQR(1-(Z2/R1)↑2)/(Z2/R1))
650 IF Z2<0 THEN 810
660 A2=90-A1
670 B1=ATN(Y/X2)
680 IF X2>0 THEN 730
690 IF Y>0 THEN 720
700 B1=B1-180
710 GOTO 730
720 B1=B1+180
730 B2=180-B1
740 B3=90-B2
750 IF (A2-A9)<0 THEN 770
760 F=F+1
770 M=A2*COS(B3)
780 N=A2*SIN(B3)
790 PRINT C,E0,R1,B2,A2
800 PLOT M,N,-2
810 C=C+2
820 IF C>(L0+O) THEN 840
830 GOTO 550
840 J=2*F/W
850 PRINT
860 PRINT "SATELLITE VIEWING TIME ABOVE"A9" DEGREES IS"J" MINUTES"
870 PRINT
880 PRINT
890 PEN
900 E0=E0+5
910 IF E0>E1 THEN 930
920 GOTO 380
930 END
```

```
10 REM THIS PROGRAM COMPUTES AND PLOTS THE AZIMUTH AND ELEVATION ANGLES
20 REM NECESSARY TO LOCATE A SATELLITE IN A CIRCULAR ORBIT. THE FOLLOWING
30 REM PARAMETERS MUST BE INPUT:OBSERVER LATITUDE, L0, (0<ANGLE<90); INITIAL
40 REM EQUATOR CROSSING LONGITUDE RELATIVE TO OBSERVER, E0, (EAST IS +), ANGLE
50 REM BETWEEN ORBITAL PLANE AND EQUATORIAL PLANE, P0, AND THE
60 REM ALTITUDE OF THE SATELLITE IN KM, H.  THE FOLLOWING DATA IS PRINTED:
70 REM THE PERIOD OF THE SATELLITE, THE ANGULAR VELOCITY OF THE SATELLITE
80 REM (DEGREES PER MIN), OBSERVER LATITUDE, SATELLITE EQUATOR CROSSING
90 REM LONGITUDE, SLANT RANGE TO SATELLITE(KM), AZIMUTH, AND ELEVATION.
100 REM THE INDEPENDENT VARIABLE IS THE SATELLITE'S ANGULAR DISPLACEMENT
110 REM ABOVE THE EQUATORIAL PLANE.  THE INITIAL VALUE FOR THIS ANGLE IS
120 REM COMPUTED FROM SATELLITE ALTITUDE AND OBSERVER LATITUDE.  THE AZIMUTH
130 REM AND ELEVATION CAN BE PLOTTED ON POLAR PAPER. THE LOWER LEFT CORNER
140 REM IS 225 DEG AZIMUTH, 127 ELV; THE UPPER RIGHT CORNER IS 45 DEG AZ,
150 REM 127 ELV. THE MINIMUM ELEVATION ANGLE IS DESIGNATED A9.
160 DEG
170 FIXED 2
180 SCALE -90,90,-90,90
190 PEN
200 DISP "L0 EQUALS";
210 INPUT L0
220 DISP "E0 EQUALS";
230 INPUT E0
240 DISP "P0 EQUALS";
250 INPUT P0
260 DISP "H EQUALS";
270 INPUT H
280 DISP "A9 EQUALS";
290 INPUT A9
300 R0=6370
310 G0=0.0098062
320 W1=0.25
330 E1=-E0
340 R=R0+H
350 T=2*PI/60*SQR(R↑3/(G0*R0↑2))
360 O=ATN(SQR(1-(R0/R)↑2)/(R0/R))
370 W=360/T
380 PRINT "OBSERVER LATITUDE IS"L0" DEGREES"
390 PRINT "SATELLITE ALTITUDE IS"H" KM"
400 PRINT "INCLINATION OF SATELLITE ORBITAL PLANE IS "P0" DEGREES"
410 PRINT "PERIOD OF SATELLITE IS"T" MINUTES"
420 PRINT "EARTH-CENTRAL CONIC HALF-ANGLE OF EARTH COVERAGE IS"O" DEGREES"
430 PRINT "SATELLITE ANGULAR VELOCITY IS"W" DEGREES PER MINUTE"
440 PRINT
450 PRINT
460 PRINT "SATELLITE      SATELLITE      SLANT RANGE      AZIMUTH        ELEVATION"
470 PRINT "ANGULAR        EQUATOR            TO           ANGLE            ANGLE   "
480 PRINT "POSITION       CROSSING         SATELLITE                               "
490 PRINT
500 F=0
510 A0=90-L0
520 X0=R0*SIN(A0)
530 Z0=R0*COS(A0)
540 C=10*(INT((L0-O)/10))
550 E=E0-W1/W*C
560 X=R*(COS(C)*COS(E)-SIN(C)*COS(P0)*SIN(E))
570 Y=R*(COS(C)*SIN(E)+SIN(C)*COS(P0)*COS(E))
580 Z=R*(SIN(C)*SIN(P0))
590 X1=X-X0
600 Z1=Z-Z0
610 R1=SQR(X1↑2+Y↑2+Z1↑2)
620 X2=X1*COS(A0)-Z1*SIN(A0)
630 Z2=Z1*COS(A0)+X1*SIN(A0)
640 A1=ATN(SQR(1-(Z2/R1)↑2)/(Z2/R1))
650 IF Z2<0 THEN 810
660 A2=90-A1
670 B1=ATN(Y/X2)
680 IF X2>0 THEN 730
690 IF Y>0 THEN 720
700 B1=B1-180
710 GOTO 730
720 B1=B1+180
730 B2=180-B1
740 B3=90-B2
750 IF (A2-A9)<0 THEN 770
760 F=F+1
770 M=A2*COS(B3)
780 N=A2*SIN(B3)
790 PRINT C,E0,R1,B2,A2
800 PLOT M,N,-2
810 C=C+2
820 IF C>(L0+O) THEN 840
830 GOTO 550
840 J=2*F/W
850 PRINT
860 PRINT "SATELLITE VIEWING TIME ABOVE"A9" DEGREES IS"J" MINUTES"
870 PRINT
880 PRINT
890 PEN
900 E0=E0+5
910 IF E0>E1 THEN 930
920 GOTO 380
930 END
```

APPENDIX B
AIRCRAFT NATIONALITY MARKS

APPENDIX B

AIRCRAFT NATIONALITY MARKS

This appendix lists all such Nationality Marks listed by ICAO as of 31 May 1974. Section A lists the marks in alphabetical order of States. Section B lists the marks in alphabetical order of the Nationality Marks.

List of Aircraft Nationality Marks Notified to ICAO up to 31 May 1974

A. Arranged alphabetically in order of States

Afghanistan	YA
Algeria	7T
Argentina	LQ,LV
Australia[2]	VH,P2
Austria	OE
Bangladesh	S2
Barbados	8P
Belgium	OO
Bolivia	CP
Botswana	A2
Brazil	PP,PT
Bulgaria	LZ
Burma	XY,XZ
Burundi	9U
Canada[1]	C,CF
Central African Republic	TL
Chad	TT
Chile	CC
China	B
Colombia	HK
Congo, People's Republic of	TN
Costa Rica	TI
Cuba[3]	CU
Cyprus	5B
Czechoslovak Socialist Republic	OK
Dahomey	TY
Democratic Yemen	7O
Denmark	OY
Dominican Republic	HI
Ecuador	HC
Egypt, Arab Republic of	SU
El Salvador	YS
Equatorial Guinea	3C
Ethiopia	ET
Fiji	DQ
Finland	OH
France	F
Gabon	TR
Germany, Federal Republic of	D
Ghana	9G
Greece	SX
Guatemala	TG
Guinea	3X
Guyana	8R
Haiti	HH
Honduras	HR
Hungary	HA
Iceland	TF
India	VT
Indonesia	PK
West Irian	PK
Iran	EP
Iraq	YI
Ireland	EI,EJ
Israel	4X
Italy	I
Ivory Coast	TU
Jamaica	6Y
Japan	JA
Jordan	JY
Kenya	5Y
Khmer Republic	XU
Korea, Republic of	HL
Kuwait	9K
Laos	XW
Lebanon	OD
Lesotho	7P
Liberia	EL
Libyan Arab Republic	5A
Liechtenstein[2,4]	HB plus national emblem
Luxembourg	LX
Madagascar	5R
Malawi	7QY
Malaysia	9M
Mali	TZ
Malta	9H
Mauritania	5T
Mauritius	3B
Mexico	XA,XB,XC
Monaco	3A
Morocco	CN
Nauru	C2
Nepal	9N
Netherlands, Kingdom of the	PH
Netherlands Antilles	PJ
Surinam	PZ
New Zealand	ZK,ZL,ZM
Nicaragua[2]	AN
Niger	5U
Nigeria	5N
Norway	LN
Oman	A4O
Pakistan	AP

Panama	HP	Tanzania, United Republic of	5H
Paraguay	ZP	Thailand	HS
Peru	OB	Togo	5V
Philippines(2)	RP	Trinidad and Tobago	9Y
Poland	SP	Tunisia	TS
Portugal	CR,CS	Turkey	TC
Qatar	A7	Uganda	5X
		Union of Soviet Socialist Republics(2)	CCCP
Romania	YR	United Kingdom	G
Rwanda	9XR	Colonies and Protectorates	VP,VQ,VR
		United Republic of Cameroon	TJ
Saudi Arabia	HZ	United States	N
Senegal	6V,6W	Upper Volta	XT
Sierra Leone	9L	Uruguay	CX
Singapore	9V		
Somalia	6O	Venezuela	YV
South Africa	ZS,ZT,ZU	Viet-Nam, Republic of	XV
Spain	EC		
Sri Lanka	4R	Western Samoa	5W
Sudan	ST		
Swaziland	3D	Yemen	4W
Sweden	SE	Yugoslavia	YU
Switzerland(2,4)	HB plus national emblem	Zaire, Republic of	9Q
Syrian Arab Republic	YK	Zambia	9J

B. Arranged alphabetically in order of Nationality Marks

AN	Nicaragua(2)	F	France
AP	Pakistan		
A2	Botswana	G	United Kingdom
A7	Qatar		
A40	Oman	HA	Hungary
		HB +national emblem	Switzerland(2,4)
B	China		
		HB +national emblem	Liechtenstein(2,4)
C,CF	Canada(1)		
CC	Chile	HC	Ecuador
CCCP	Union of Soviet Socialist Republics(2)	HH	Haiti
		HI	Dominican Republic
CN	Morocco	HK	Colombia
CP	Bolivia	HL	Korea, Republic of
CR,CS	Portugal	HP	Panama
CU	Cuba(3)	HR	Honduras
CX	Uruguay	HS	Thailand
C2	Nauru	HZ	Saudi Arabia
D	Germany, Federal Republic of	I	Italy
DQ	Fiji		
		JA	Japan
EC	Spain	JY	Jordan
EI,EJ	Ireland		
EL	Liberia	LN	Norway
EP	Iran	LQ,LV	Argentina
ET	Ethiopia	LX	Luxembourg
		LZ	Bulgaria

N	United States
OB	Peru
OD	Lebanon
OE	Austria
OH	Finland
OK	Czechoslovak Socialist Republic
OO	Belgium
OY	Denmark
PH	Netherlands, Kingdom of the
PJ	Netherlands Antilles
PK	Indonesia
PK	West Irian
PP,PT	Brazil
PZ	Surinam
RP	Philippines (2)
SE	Sweden
SP	Poland
ST	Sudan
SU	Egypt, Arab Republic of
SX	Greece
S2	Bangladesh
TC	Turkey
TF	Iceland
TG	Guatemala
TI	Costa Rica
TJ	United Republic of Cameroon
TL	Central African Republic
TN	Congo, People's Republic of
TR	Gabon
TS	Tunisia
TT	Chad
TU	Ivory Coast
TY	Dahomey
TZ	Mali
VH,P2	Australia (2)
VP,VQ,VR	United Kingdom Colonies and Protectorates
VT	India
XA,XB,XC	Mexico
XT	Upper Volta
XU	Khmer Republic
XV	Viet-Nam, Republic of
XW	Laos
XY,XZ	Burma
YA	Afghanistan
YI	Iraq
YK	Syrian Arab Republic
YR	Romania
YS	El Salvador
YU	Yugoslavia
YV	Venezuela
ZK,ZL,ZM	New Zealand
ZP	Paraguay
ZS,ZT,ZU	South Africa
3A	Monaco
3B	Mauritius
3C	Equatorial Guinea
3D	Swaziland
3X	Guinea
4R	Sri Lanka
4W	Yemen
4X	Israel
5A	Libyan Arab Republic
5B	Cyprus
5H	Tanzania, United Republic of
5N	Nigeria
5R	Madagascar
5T	Mauritania
5U	Niger
5V	Togo
5W	Western Samoa
5X	Uganda
5Y	Kenya
6O	Somalia
6V,6W	Senegal
6Y	Jamaica
7O	Democratic Yemen
7P	Lesotho
7QY	Malawi
7T	Algeria
8P	Barbados
8R	Guyana
9G	Ghana
9H	Malta
9J	Zambia
9K	Kuwait
9L	Sierra Leone
9M	Malaysia
9N	Nepal
9Q	Zaire, Republic of
9U	Burundi
9V	Singapore
9XR	Rwanda
9Y	Trinidad and Tobago

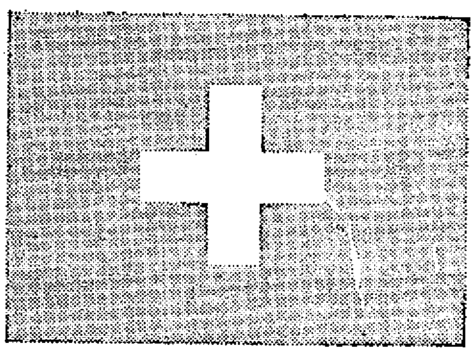
NOTES:

1. (a) Nationality Mark "CF" will be phased out in early 1980s.
   (b) Under Nationality Mark "C" allocation of registration marks is limited to the series C-FAAA to C-KZZZ.

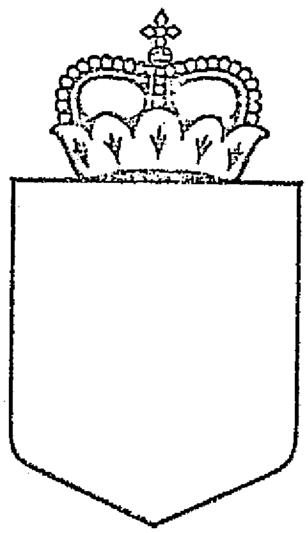
2. This mark differs from the provision in 2.3 of this Annex.

3. This mark is not yet officially confirmed.

4. For national emblems of Liechtenstein and Switzerland, see below.
   The nationality marks appearing on the aircraft consist of a group of characters and the national emblem.
   The "group of characters of the nationality marks" appears in lieu of the nationality marks on the wings and either on the fuselage or on the upper halves of the vertical tail surfaces. In addition,
   - in applying 3.2.2, the national emblem which forms part of the nationality marks is attached to the basket suspension cables.
   - in applying 3.3.2, the national emblem which forms part of the nationality marks appears on both sides of the vertical tail surfaces or on the outboard sides of the outer tail surfaces.



NATIONAL EMBLEM OF SWITZERLAND



NATIONAL EMBLEM OF LIECHTENSTEIN