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Timation Development Plan

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NAVAL RESEARCH LABORATORY
Washington, D.C.

CONTENTS

ABSTRACT
(Unclassified)

3 TIMATION (Time Navigation) is a technological development program designed to study, identify and develop the critical components and make measurements necessary to describe a navigation system which meets JCS requirements.

4 The purpose of the TIMATION navigation system development ~~plan~~ is to describe the technical parameters, tradeoffs, experiments, and costs encompassing the implementation of a satellite position fixing and navigation system that meets the requirements promulgated by the Joint Chiefs of Staff (JCS) Navigation Study Panel in 1968. This navigation system provides continuous all weather instantaneous readout to an unlimited number and variety of users on a worldwide basis.

PROBLEM STATUS

This is an interim report on a continuing NRL problem.

AUTHORIZATION

NRL Problem R04-16
Project W34-11X

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Section 1

SUMMARY

1.0 INTRODUCTION

A navigation system development plan is presented which describes a satellite navigation system that will meet the Joint Chiefs of Staff (JCS) navigation accuracy requirements, passive user requirements, and Navy ground station location requirements.

1.1 PRINCIPLE OF OPERATION

Only one type of system has been proposed that will meet the requirements for both accuracy and passive users—the Timation system. This system uses high-altitude passive-ranging satellites which transmit signals synchronized by means of internal oscillators which in turn are synchronized to ground station oscillators by a synchronization link.

1.2 GROUND STATIONS

The siting requirement of having all ground stations on U.S. territory affects the system accuracy and the satellite constellation. The four ground stations selected are shown in Fig. 1-1; they encompass the maximum expanse of U.S. territory in the north (Alaska), east (St. Croix, Virgin Islands), west (Guam), and south (Samoa). Figure 1-1 also shows the coverage of these four stations and, more especially, the noncoverage (satellite viewing ground stations) that occurs over the Indian Ocean area.

1.3 SATELLITE CONSTELLATIONS

Figure 1-2 shows the geographical positions of the 3-by-9 constellation made up of satellites having 8-hour, circular, 55° inclination orbits.

Each plane is obtained by launching a group of satellites by means of a single booster from the east coast. The Titan IIID is the booster selected. The Burner II second stage serves to circularize the payload. Figure 1-3 shows the launch sequence. Payloads (two satellites) are replaced by means of a Titan IIIB Burner IIa booster configuration.

1.4 NAVIGATION ACCURACY

The error budget is made up of two parts—a geometrical part, in which the geometry of the satellites present is used as a GDOP (geometrical dilution of position) factor, and the component errors.

In areas where the error budget is constant, the fix error is obtained by multiplying the error budget by the GDOP factor.

Figure 1-4 is an example of an error contour map made for the condition in which the maximum number of satellites have the maximum clock errors and the observer possesses a good clock (1pp 10^{11}). Figure 1-5 is the same case for an observer having a continuous clock correction.

For both Figs. 1-4 and 1-5, the following component errors of the error budget were assumed:

Multipath	2.5 nsec or feet
Ionosphere	2.0 nsec or feet
Troposphere	2.0 nsec or feet
Equipment	5.0 nsec or feet
Satellite position	3.0 nsec or feet
Clock synchronization	5.0 nsec or feet
RSS	8.5 nsec or feet [value used = 10 nsec + clock error]

The error in the clock for Figs. 1-4 and 1-5 is assumed to increase at a rate of 2pp 10^{12} during the time the satellite is out of the view of a ground station.

1.5 SPACE SERVICE IMPLEMENTATION PLAN

The implementation of the space service portion of the development plan is divided into five parts as follows:

1. Technology - Timation II proceeding on schedule

2. System design - Procedures discussed in this report.

3. Design demonstration - The design demonstration phase will consist of a single satellite configured much like Timation II. The principal differences will be that this satellite will operate at 400 and 1600 MHz in contrast to 400 and 150 MHz for the Timation II satellite and will have an orbit altitude of 7500 naut mi. The function of this design demonstration is to measure the navigation error budget. The factors that will be measured are the prediction of satellite position (geodetic constants), clock synchronization multipath, tropospheric and ionospheric refraction, and equipment errors.

4. Navigation demonstration - The navigation demonstration will utilize four satellites in 8-hour orbits. This constellation will be launched with two satellites in each of two planes. The function of this demonstration will be to show that the navigation system meets its specifications.

5. Complete space system - The fifth part consists of launching the complete space portion to provide 27 satellites launched in three planes of nine satellites each, all in 7500-naut mi circular orbits.

1.6 MILESTONES

The milestones are given in Fig. 1-6. Concept formulation extends through the navigation test phase on four satellites and requires 5 years. Contract definition follows the concept formulation for 1 year. The complete space segment is installed in the next 3-year period.

1.7 SPACE SEGMENT COST

The cost of the space segment is shown in Fig. 1-7. In this figure the cost of the space service is shown as approximately 200 million dollars. The backup launch item provides an extra booster and nine satellites to be used if one of the plane boosters fails. If the initial three large boosters are successful, this booster and the satellites will be available for replacement purposes.

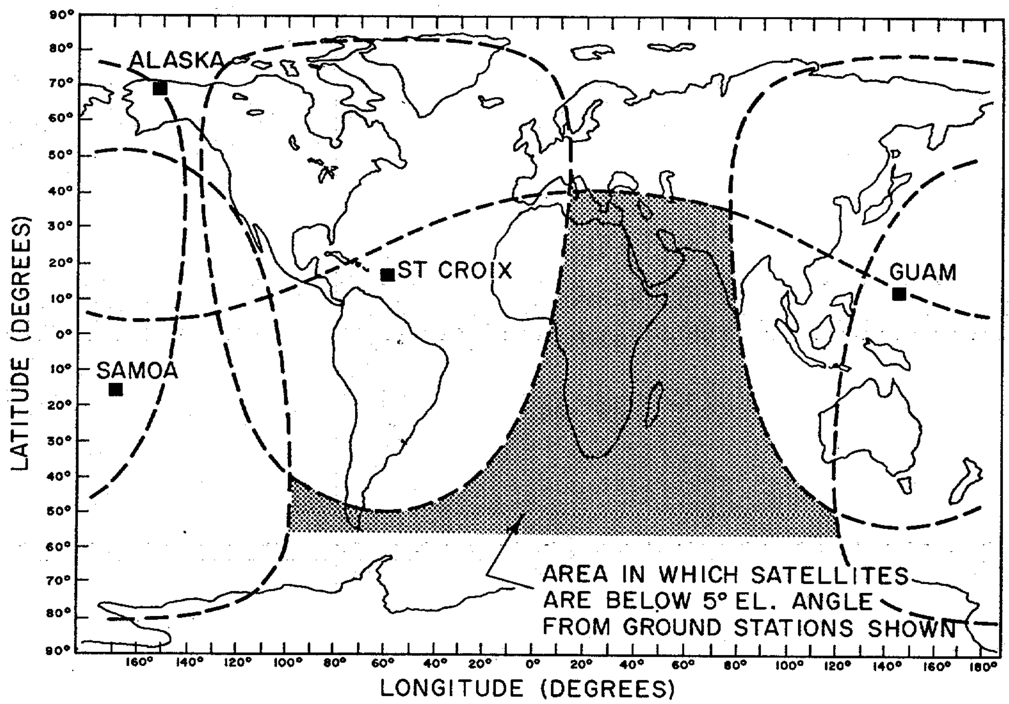

Fig. 1-1 - Chart showing station coverage for stations shown for 8-hr (7500-naut mi), 55-degree orbits

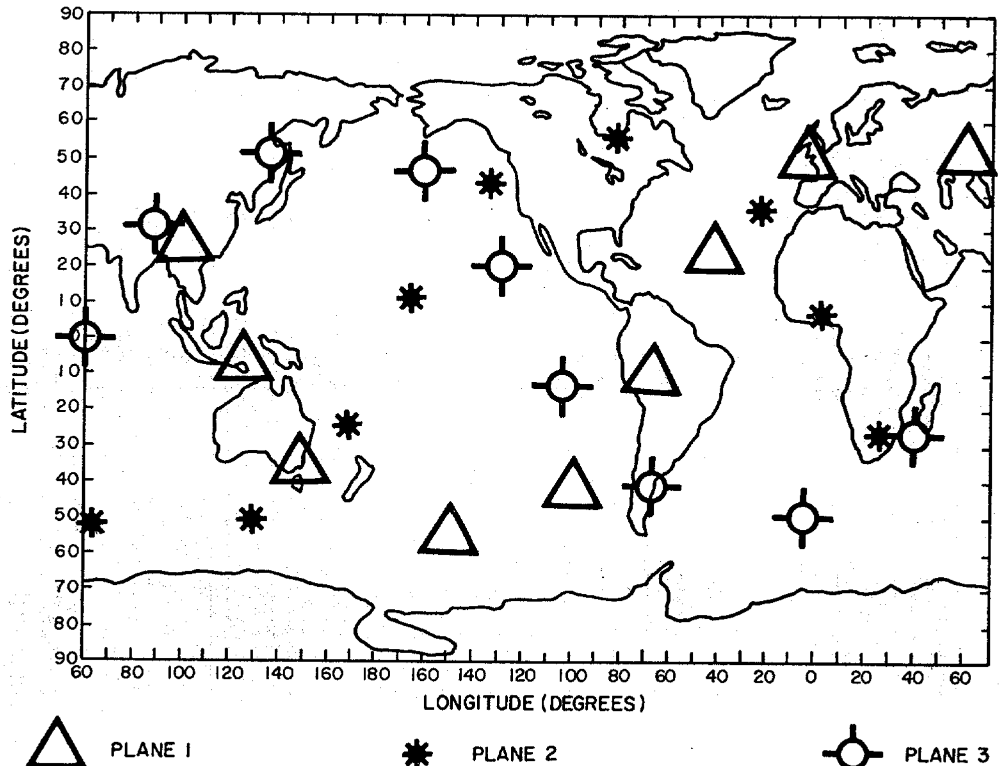

Fig. 1-2 - Geographical positions for a 3-by-9 constellation in 8-hr, 55-degree orbits

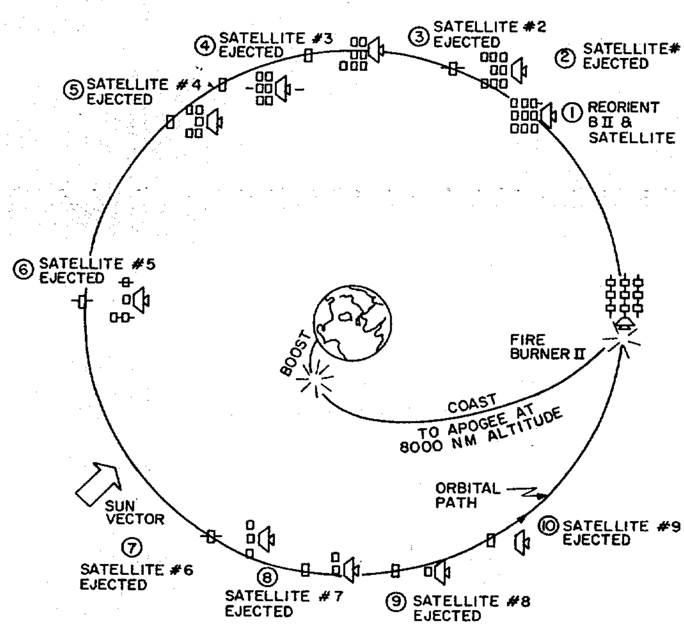

Fig. 1-3 - Launch sequence

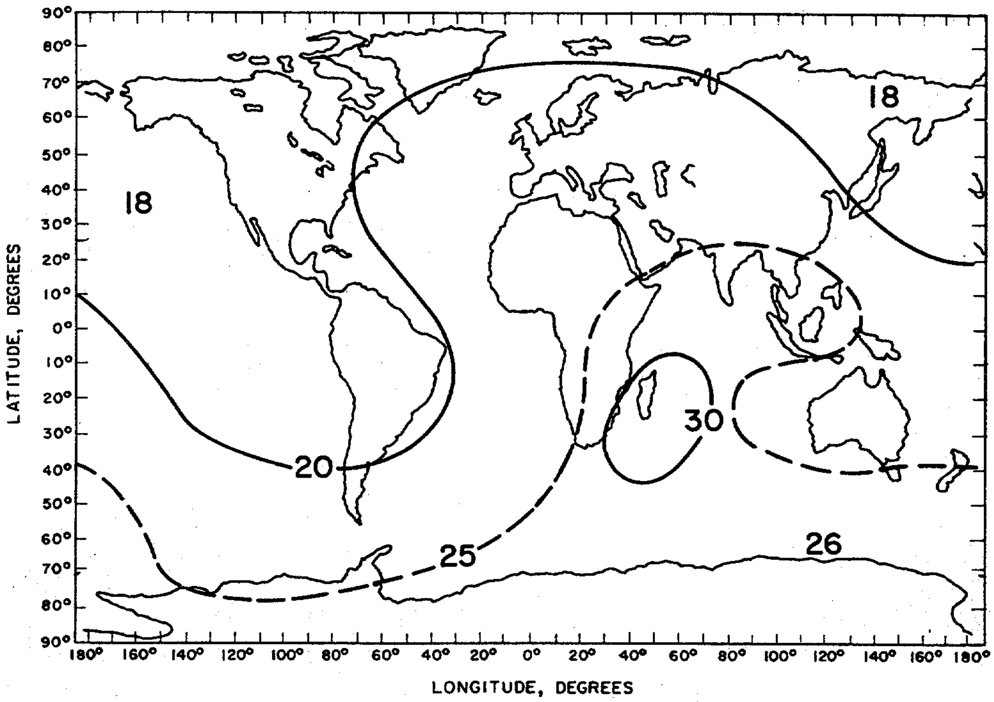

Fig. 1-4 - Error contours (in feet) for a user with a good clock for a 3-by-9, 8-hr, 55-degree constellation

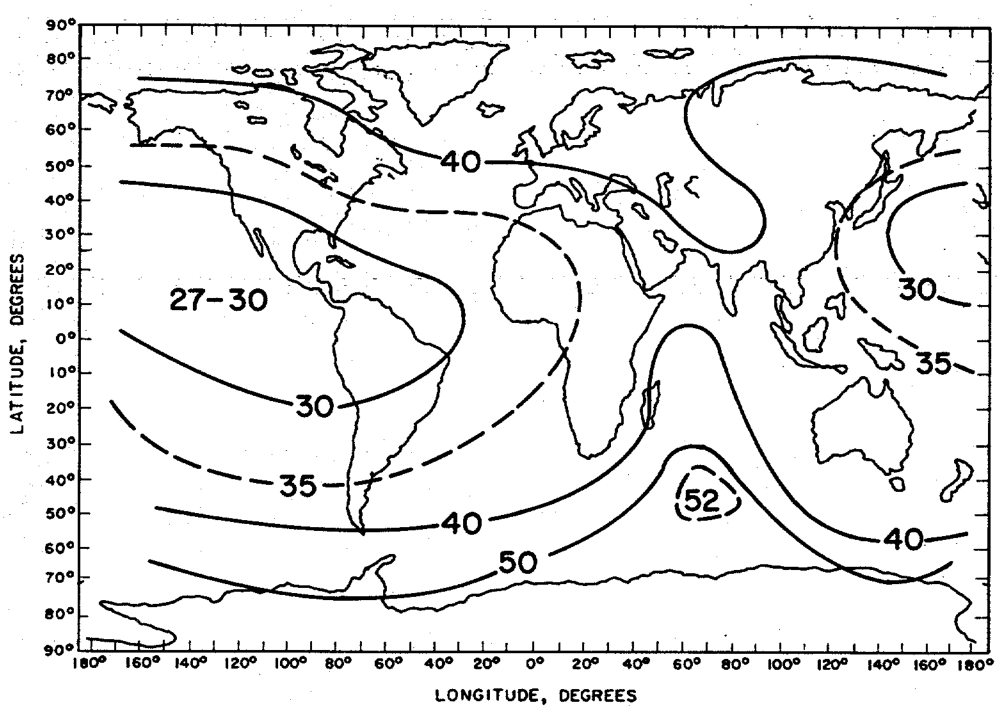

Fig. 1-5 - Error contours (in feet) for a user with a continuously corrected clock for a 3-by-9, 8-hr, 55-degree constellation

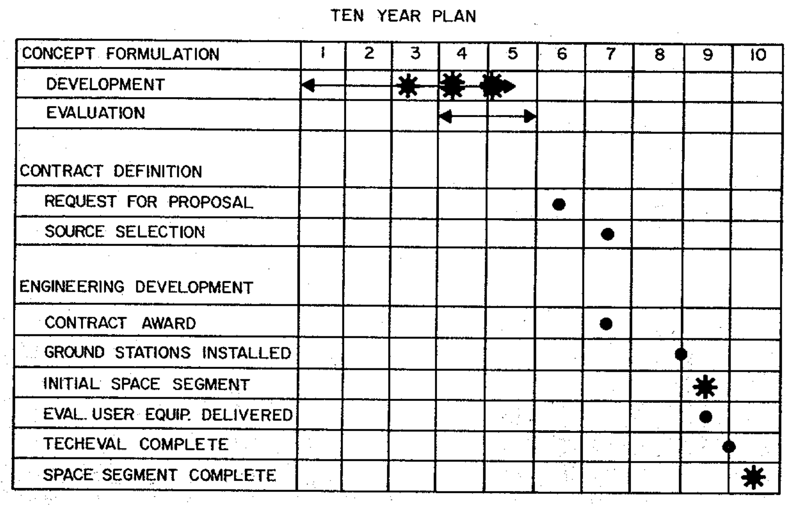

Fig. 1-6 - System implementation

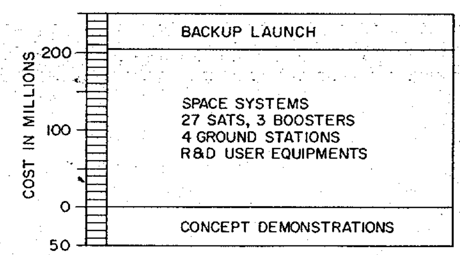

Fig. 1-7 - Estimated cost of Timation navigation service

Section 2

PURPOSE AND GOALS

2.0 INTRODUCTION

The purpose of the navigation system development plan is to describe the technical parameters, tradeoffs, experiments, and costs encompassing the implementation of a satellite position fixing and navigation system that exceeds the requirements promulgated by the Joint Chiefs of Staff (JCS) Navigation Study Panel in 1968. This navigation system provides continuous all weather instantaneous readout to an unlimited number and variety of users on a worldwide basis.

Included in this plan are the near-term experimental effort and the long-term implementation of the results from this effort into a system with the basic simplicity of celestial navigation but with higher accuracy. In the near-term experimental effort this development plan has to:

1. Measure the orbital parameters of satellites at altitudes of 7500 naut mi.

2. Demonstrate the ability to meet the system error budget.

3. Demonstrate position fixing and navigation accuracies that meet the JCS requirements.

4. Demonstrate the ability to measure velocity of a user's platform to 0.05 feet per second.

5. Demonstrate the ability to transfer time to 0.01 microseconds or less to any remote location in the world.

6. Verify the study, previously performed, which indicates that geodetic positioning can be done worldwide with errors of less than 5 feet.

7. Verify the ability to measure azimuth angles to an accuracy of 1 milliradian using a baseline of 60 feet.

The long-term effort is designed to collate and analyze the results of the above experimental goals and to incorporate them into an operational system. The long-term goals are:

1. Design and operate a demonstration system of four satellites and supporting ground stations. (This is the end product of the short-term goals.)

2. Design and procure satellites for use in the operational constellation.

3. Design a family of user equipment that can be applied to the multitude of military users.

4. Design and procure the ground stations and computation center to support the satellite constellation.

5. Determine and provide all necessary logistics and support to facilitate operation and maintenance of the system in service use.

The satisfaction of these goals will provide a navigation service with automated worldwide grid lock and precision position fixing for such applications as targeting and will simultaneously fulfill the general navigation requirement with manually plotted data from a simple receiver.

2.1 ATTAINMENT OF GOALS

The immediate goal of the program to implement this development plan is to exercise the technology now available in the design and performance of experiments in orbit. Proof of the application of this technology will be determined in the test and demonstration phase. Previous development and analysis indicate that this goal may be successfully met. The two experimental spacecraft launched in the exploratory development phase of this project, Timation I and II, were designed, constructed, and preflight tested at the Naval Research Laboratory. These experimental satellites have verified the approach, and have provided a realm of technical data, which has been applied to the design of the operational system. System design studies and critical item studies combined with the actual experimental data have shown that the proposed system is feasible, practical, efficient, and reliable.

The test and demonstration phase of development will empirically prove that the system will meet the JCS requirements. Demonstration user receiving and computation equipment will be designed and built for tests with aircraft, ships, and ground forces. Each type of user need not have specially designed equipment, since the basic equipment subsystems for each are common to all, and only interface equipment and unique software will differentiate between the users. Along with the tests of user equipment the spacecraft design shall be fully tested with experimental ground stations. The spacecraft and ground station will be so configured that on successful completion of the test and demonstration phase, they may phase into full operational status as the satellite constellation becomes established.

The current spacecraft in development is the Timation III. This satellite will be experimental and will be launched to explore and eventually determine the error budget regions for the operational prototypes to follow. Timation III will partly satisfy the near-term objectives and will provide a foundation for the demonstration system with the four prototypes planned to follow. This method of demonstrating the system is conservative in that all systems are vigorously proven before commitment to the entire system. Therefore, the attainment of the long-term goals can be considered as a low-risk venture.

Section 3

SYSTEM DESCRIPTION

SUMMARY

The Timation navigation system would consist of a constellation of 27 satellites, a ground network of four stations to track and command the satellites, an operations or control center to operate the system, and user equipments to satisfy the needs of various classes of users.

The constellation of 27 satellites is made up of three planes of nine equispaced satellites each with plane inclinations of 55 degrees and plane spacings of 120 degrees in right ascension. The satellites transmit ranging and ephemeral data on each of two frequencies for ionospheric refraction correction and by two modes on each frequency. Mode one uses a clear-channel side-tone ranging signal, which satisfies the needs of many classes of users, and mode two uses spread-spectrum modulation techniques, which satisfy those users who require more antijam resistance.

The ground stations are located in Alaska, Guam, Samoa, and the U.S. Virgin Islands. The function of these stations is to track the satellites, to provide orbital and clock performance data, to monitor telemetered satellite system performance, to command and control satellite systems, and to relay the data via satellite transponder links to the control center.

The operations or control center is colocated with the Alaska ground station. Its principal functions are system data collection, orbit computation, clock update, satellite performance analysis, and ephemeral loading.

Six categories of user equipments are addressed, which include high-performance aircraft, surface and subsurface vessels, field or foot soldiers, and general-purpose ships and vehicles.

The system as described meets the accuracy requirements of the JCS and addresses the problems of antijam, denial of access, and radiation hardening.

3.0 SYSTEM DESCRIPTION

This section presents a comprehensive summary of the characteristics of the Timation navigation satellite system. The Timation Satellite System employs a constellation of 27 satellites — each in an 8-hour, 55-degree inclined orbit, ground stations in the vicinity of the Samoan Islands, the Virgin Islands, Guam, and central Alaska, and a variety of user equipments which are graded in cost and performance according to the user's requirements. This system will provide global, all-weather, passive, continuous navigation fixes in three dimensions with accuracies better than 50 feet. Low-cost equipment for general navigation users, at reduced accuracies, will be available for use with the system.

The satellites are identical in form and are equipped to transmit ranging signals, satellite ephemerides, and signals for refraction correction. Ground stations will track the satellites, predict the satellite orbits, maintain the satellite clocks in synchronism with system time, and maintain the satellites' operating status.

The satellites provide the necessary data for the navigator to determine his latitude, longitude, altitude, and time. Not all navigators need to determine all of these parameters. The user with the most stringent requirements will use four satellites in view and the equipment to receive signals from all four satellites; reduced requirements will result in one or more of the four signals being ignored by that user's equipment. This system was configured within the JCS navigation accuracy requirements and with close attention to producing a cost-effective configuration.

3.1 System Concept

The concept of system operation is best described with the aid of the system concept diagram in Fig. 3-1. This figure depicts the relation between the ground station network, the satellite constellation, and representative users.

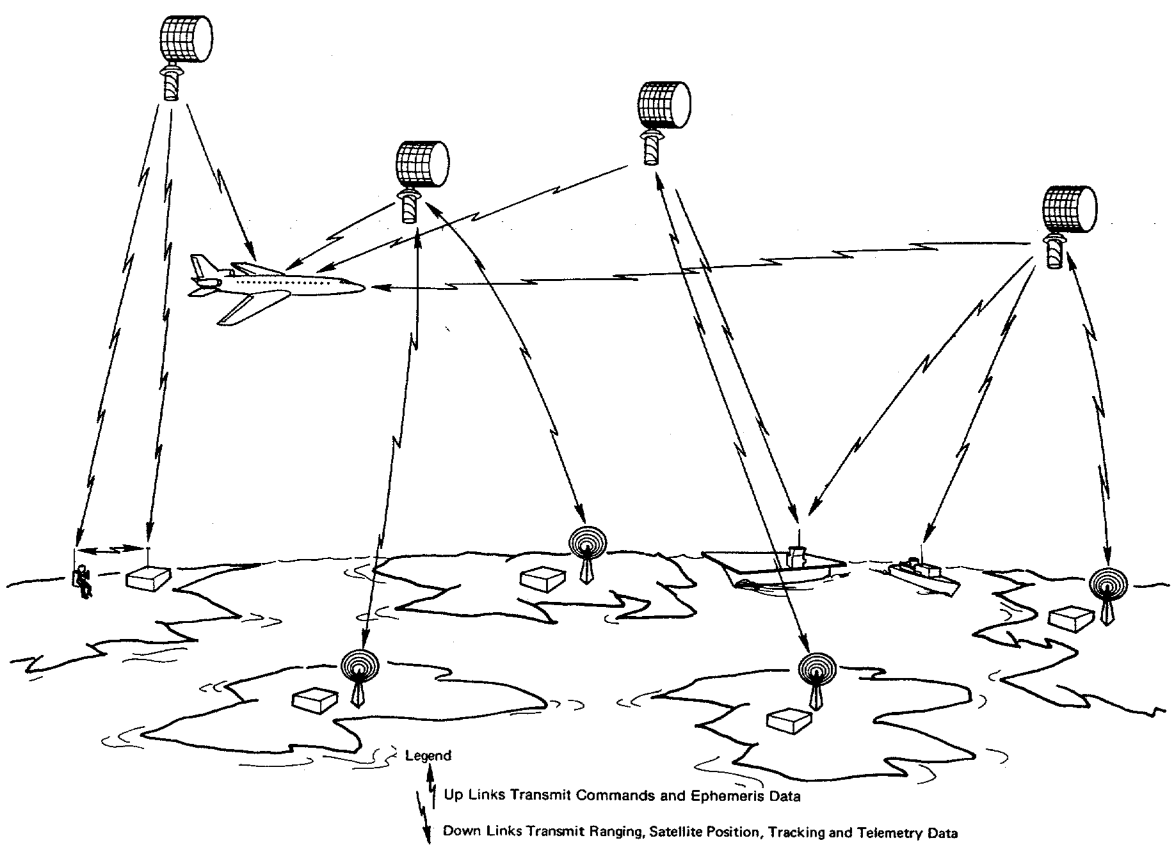

Fig. 3-1 - Timation navigation satellite system

The ground station network consists of four sites, each of which is equipped to track the satellites, monitor the satellite's telemetry, and maintain time synchronization between satellites. Each ground station is equipped with an atomic clock. Synchronization of all ground station clocks is maintained through the satellite links. All satellite clocks are synchronized with system time through the satellite-ground station link.

A separate satellite memory message is generated and transmitted to each satellite after its orbit has been determined by the ground station network. The satellites transmit these ephemerides to the navigators as references for position fixing. Range signals are also transmitted to the navigator and he can use them in either of two modes. If his equipment contains a system-synchronized clock (range-only mode), he can find as many surfaces of position as there are satellites visible, assuming he has sufficient receiver channels. If he does not have a synchronized clock (range-difference mode), he can find surfaces of position from each pair of the satellites present and time information simultaneously. Neither mode requires user transmissions and the satellite transmissions are the same for both. The only difference is in the user equipment.

Both range only and range-difference navigation provides continuous three-dimensional fixes (latitude, longitude, and altitude) over the entire globe, meeting the general navigation requirements of surface users.

Table 3-1 lists typical modes of operation in order of decreasing cost and complexity of user equipment. It should be noted that in categories 3 and 4, a precision clock could be used with two satellites, with little change in accuracy or operation. Category 5 and 6 users make sequential measurements on a set of three satellites, and they require an inexpensive clock to correlate the measurements.

Table 3-1
Modes of Operation

Categories of Potential Users	Number of Satellites Used				
	Range-Only Mode	Range-Difference Mode	Fix Accuracy (feet)	External Data Required	Type of Data From Fix
1. Military (strategic and tactical) aircraft, carrier operations, and gun placement	3	4	50	None	Position and velocity (3 dimensions)
2. Submarines, search and rescue, and logistic support	2	3	50	Altitude	Position (2 dimensions)
3. Land operations (vehicle)	3	4	50	None	Position (3 dimensions)
4. Land operations (foot)	3	4	50	None	Position (3 dimensions) Velocity (3 dimensions)
5. General air navigation	-	3	1000	Altitude	Position (2 dimensions)
6. General marine	-	3	1000	Velocity	Position (2 dimensions)

3.1.1 Space Segment

The space segment is presented in two parts — the constellation configuration and the satellite configuration.

3.1.1.1 Constellation Configuration — The selection of the satellite constellation strongly affects the system accuracy. The criteria for selection of the constellation are:

- continuous availability at all points on the earth of four satellites in a suitable configuration for three-dimensional position fixing;
- the accuracy of the fix;
- economy in system establishment and maintenance cost.

The accuracy of the fix depends in part on the predictability of the satellite ephemerides, which is a function of the constellation, and also on the satellite clock synchronization capabilities, which are a function of both the constellation and the ground station complex.

The constellation selected for the operational system consists of three orbital planes with nine satellites equally spaced in each plane. The nominal constellation parameters are shown in Table 3-2, together with approximate values of the tolerances in the orbital parameters which will be maintained by occasional propulsive maneuvers throughout the satellite lifetime and by satellite replacement as required. The resulting spacial arrangement of satellites is shown in Fig. 3-2. This configuration provides at least four satellites in view in a good geometrical arrangement at all times and at all locations on the globe.

Table 3-2
Nominal Constellation Parameters

Parameter	Value(s)	Tolerance
Mean altitude (naut mi)	7,496	±250
Eccentricity (ratio)	0.0	+0.01
Inclination (degrees)	55.0	±1.0
Right ascensions of ascending nodes (degrees)	X,X + 120, X + 240*	±1.0
In-plane separation (degrees)	40.0	±1.0
Interplane advance† (degrees)	-13 1/3 to +13 1/3	±1.0

*X is arbitrary.
†Referenced to the ascending node crossing.

The constellation would be established by three launches of the Titan IIID/Burner II launch vehicle from the Eastern Test Range. Each vehicle would carry a composite payload of nine spacecraft into the appropriate 55-degree inclined orbit and would approximately dispense the satellites into their required locations. Final phasing of the satellites would be accomplished by the on-board propulsion units. These nine satellites would be deployed and fully operational within 3 weeks. Replenishment of the constellation would be accomplished by launching a two-satellite payload on a Titan IIIB/Burner IIA launch vehicle.

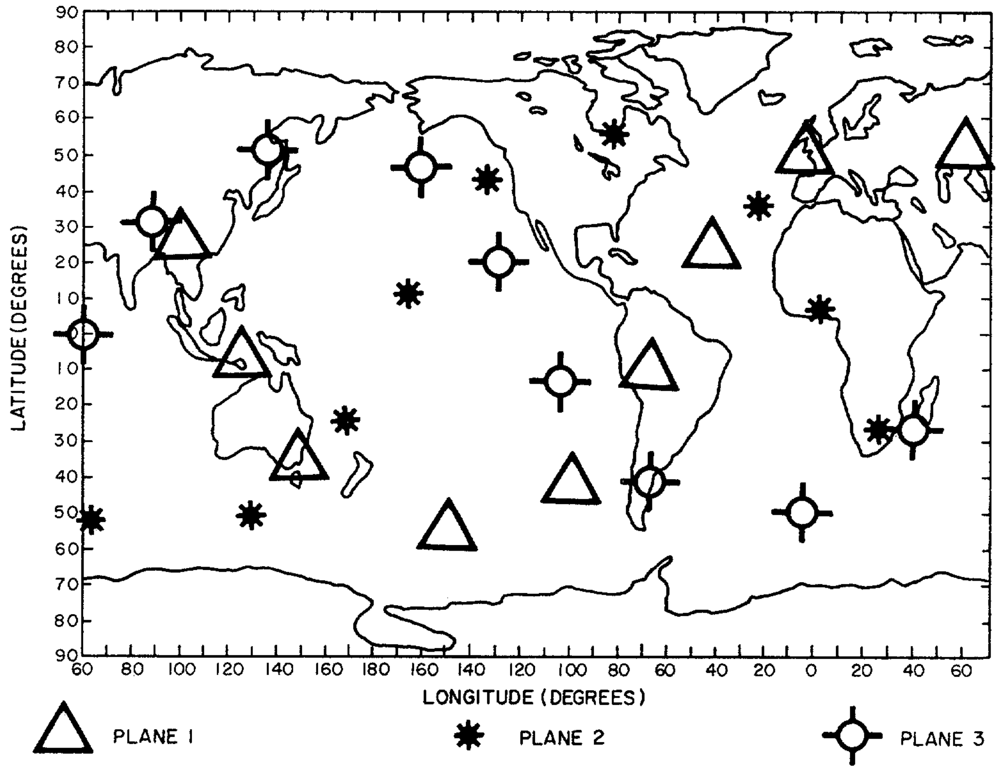

Fig. 3-2 - 27-satellite constellation and ground tracks

Some changes in orbital configuration will occur with time. The perturbations which affect all satellites equally, such as mean motion of ascending nodes and secular inclination changes, can be neglected because the relative position of the satellites within the plane would be unchanged. Perturbations which affect system performance will be small and can be cancelled by the on-board propulsion units.

3.1.1.2 Satellite Configuration — The spacecraft configuration which has been selected for the Timation system is shown in Fig. 3-3. This configuration is designed to function in a medium altitude, 55-degree inclined circular orbit. The spacecraft will transmit navigation information at uhf and L-band frequencies and be tracked and controlled via a two-way S-band link. The basic external elements of the spacecraft are the antennas, the solar array, the scanning mirror for the attitude control assembly and the cut-outs for the station keeping thrustors. In flight the satellite is oriented with its axis of symmetry perpendicular to the orbit plane so that the end of the drum carrying solar cells is sunlit. The entire drum rotates at one revolution per orbit and is phased so that the axes of the earth-covering uhf and L-band antennas coincide with the local vertical (Fig. 3-4). This attitude is maintained by a momentum wheel assembly and magnetic control coils. Positioning of the spacecraft in orbit is accomplished by pulsing the on-board propulsion system using ground command.

3.1.2 Ground Segment

The ground segment of the system is composed of four ground stations and a data reduction operations center associated with one of the ground stations. The Timation ground station system includes a network of command and tracking stations designed to support and service the constellation of navigation satellites. The network consists of four command and tracking stations located in Alaska, Guam, the Samoan Islands, and the

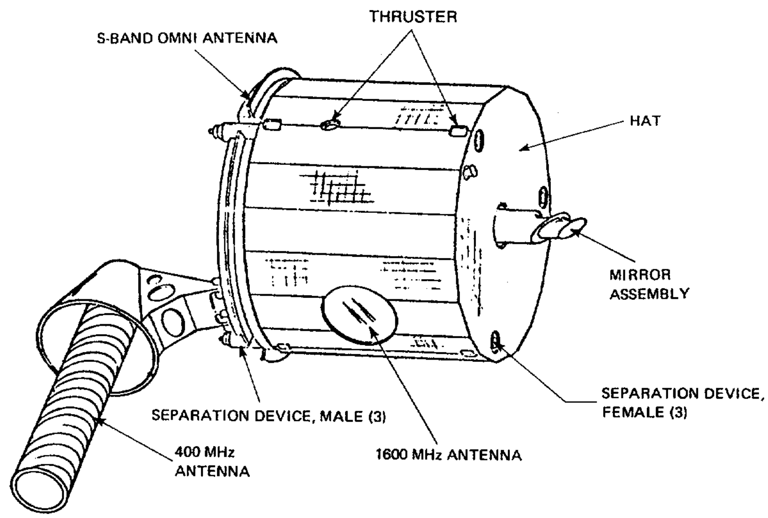

Fig. 3-3 - Spacecraft orbital configuration

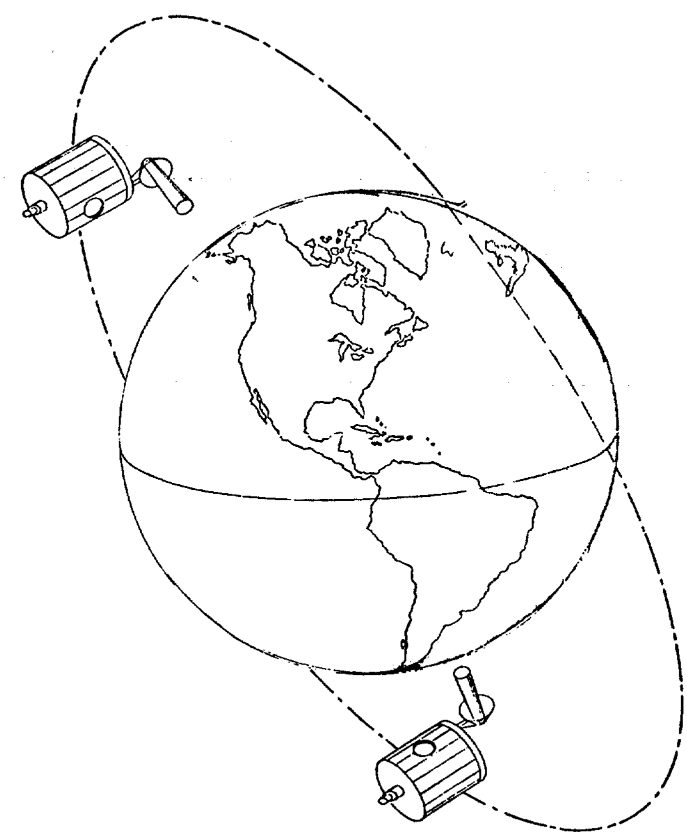

Fig. 3-4 - Spacecraft orbital orientation

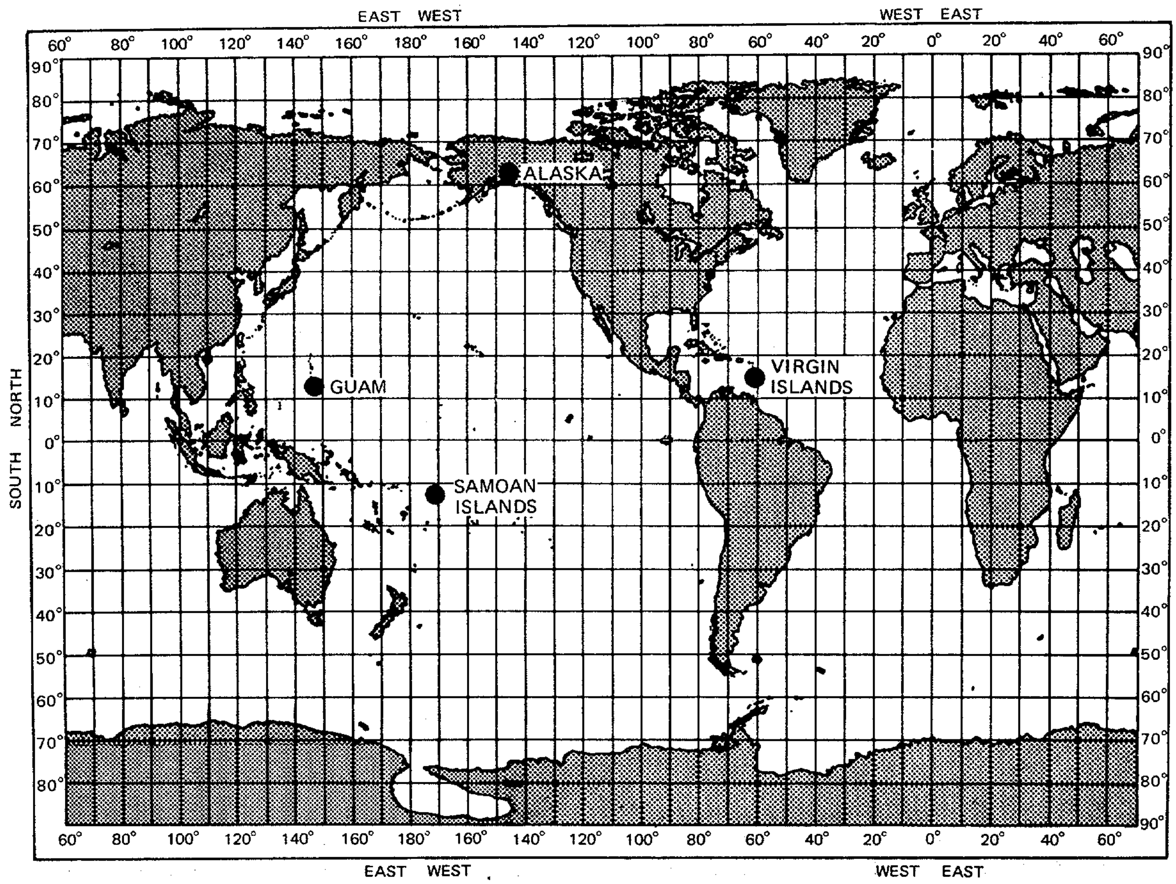

Fig. 3-5 - Ground station network locations

U.S. Virgin Islands (Fig. 3-5). An operations center is associated with the Alaska ground station to act as a collection and distribution point for messages and data. The operations center processes the collected data, generates messages and operational commands, and supervises the navigation satellite system.

The ground system is equipped to perform the following functions:

1. Satellite Tracking - The four ground stations are geographically distributed to generate tracking data covering substantial portions of the satellite orbits. The arrangement is designed to minimize the gaps in ground station coverage while at the same time restricting the sites to U.S. territory and possessions.

2. Satellite Commanding and Programming - Each satellite uses a stored ephemeris to broadcast its position in space. To maintain its accuracy the ephemeris is periodically updated with fresh data transmitted from a ground station. In addition, real-time commands are occasionally required to change the satellite status and the sequences of automatic commands stored in the satellite programmer.

3. Synchronization of the System Clocks - All satellites and ground stations contain time standards which must be correctly synchronized. The satellites carry quartz crystal

clocks that are corrected every few minutes. This correction is accomplished by monitoring and resetting procedures performed by each ground station. The Alaska ground station possesses the reference time standard. The clocks contained in the remote ground stations are periodically compared with the reference by the use of a covisible satellite repeater.

4. Monitoring of Satellite Status - Satellite status and housekeeping are monitored through the periodic reception and evaluation of telemetry data.

5. Monitoring of System Status and Operations - Frequent checks on the navigational performance of the system are needed to ensure that geographic coverage and position fixing accuracy are being maintained. System operations must also be supervised to ensure proper functioning of communications links, station equipment, and data processing facilities.

6. Message and Data Transfer - Messages and data are exchanged between ground stations and fed into and out of the operations center to implement the daily operations of the ground system. The operations center is connected to the Alaska ground station by land lines. The Alaska station functions as the focus for network communications and utilizes covisible satellites as microwave repeaters for the transfer of data to and from the remote ground stations.

The ground stations are equipped with readily available standard components. The stations will be patterned after the operationally proven NavSat ground stations and will include provisions for compatibility with other similar stations.

3.1.3 Data Links

The system must be capable of permitting several data links to function, not necessarily all at the same time. Data must be transmitted from the satellites to the remote ground stations, from the remote ground stations to the satellites, from the satellites to the central ground station, from the central ground station to the satellites, between the central ground station and the operations center, between the remote ground stations, and from the satellites to the users. These links are shown in Fig. 3-6.

Radio frequency links are required between (a) the satellite and the user, (b) the satellite and the ground station, and (c) the central and the remote ground stations.

One-way links are used between the satellite and the user to allow passive operation. The transmissions and frequencies are as follows:

1. Satellite to normal user ranging and ephemerides data link operating at 400 MHz.

2. Satellite to normal user ionospheric refraction error correction link operating at 1600 MHz.

3. Satellite to user antijam ranging link operating at 400 MHz.

4. Satellite to user antijam ranging and ephemeris data link operating at 1600 MHz.

To avoid interference between adjacent satellites each orbit plane is allotted a separate frequency and transmissions from satellites within a specific plane are time shared. The antijam signals are code multiplexed.

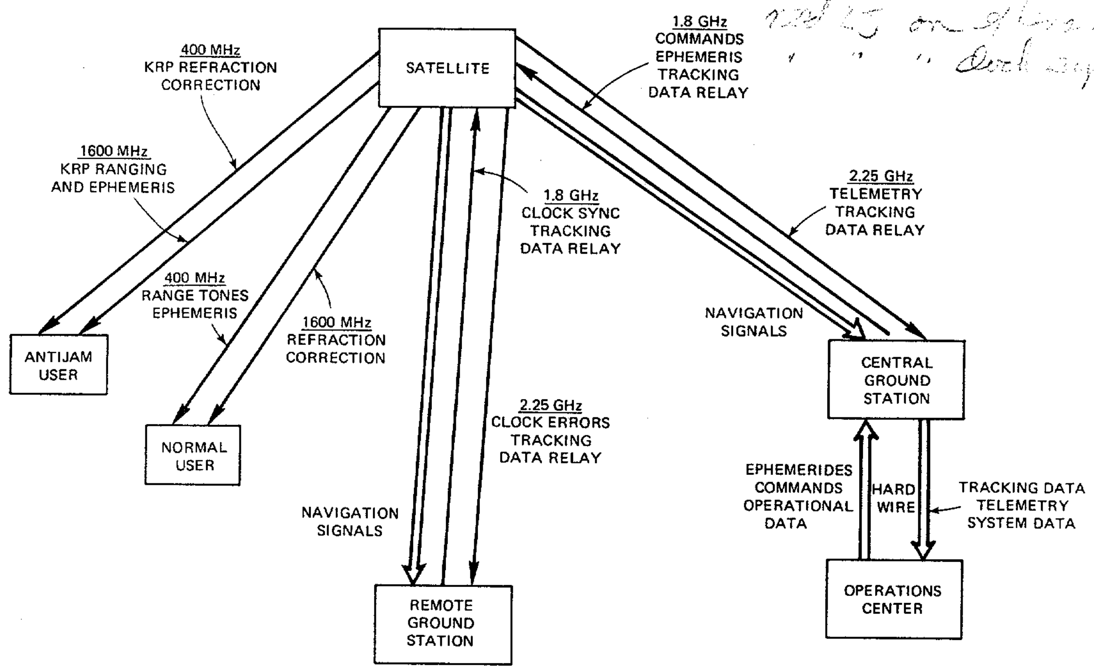

Fig. 3-6 - Timation data links

For satellite-to-ground-station data transfer the navigation transmissions are supplemented by a two-way tracking, telemetry and command link operating at S band. Its functions are defined as follows:

1. Ground station-to-satellite uplink operating at 1.8 GHz; carrying ranging modulation, spacecraft commands, and ephemeris messages for the satellite memory.

2. Satellite-to-ground station downlink operating at 2.25 GHz; carrying transponded ranging modulation and telemetry data.

The system satellites function as data repeaters between the widely spaced Timation ground stations. Since the spacecraft tracking transponder already repeats ranging modulation, it is a simple matter to modify the design to receive and retransmit an additional digitally modulated subcarrier. The satellite coverage circle has a diameter of approximately 8000 naut mi and, since none of the remote ground stations are more than 5000 naut mi from the Alaskan ground station, interstation communication can be effectively and economically accomplished by transmitting the required data over the S-band uplink to a satellite visible to the remote station, thus enabling that station to receive the data on the S-band downlink. There are always at least eight satellites visible to the Alaskan station, and one of these will be suitably positioned for data transfer to any remote ground station.

3.2 System Performance

The principal objective of the Timation navigation satellite system is to achieve the best possible performance consistent with equipment reliability, system longevity, and low overall cost. Ultimately the performance of any navigation system is measured in terms of the accuracy with which a user can determine his position in the operating environment. This environment may include hostile jamming signals.

To meet the requirements specified by the Navigation Panel of the Office of the Joint Chiefs of Staff the preferred navigation system must:

1. Provide continuous worldwide, all-weather navigation coverage.

2. Permit high accuracy (50 ft rms) position fixing by land, sea, and airborne users.

3. Inhibit tampering by unauthorized agencies.

4. Guarantee a high level of operational reliability.

5. Provide adequate protection from jamming, spoofing, and other forms of radio interference.

6. Provide passive nonsaturable access for all users.

7. Possess potential for growth in capability and performance in response to advances in technology and development of new and more sophisticated user requirements.

After an intensive study of the characteristics of low, medium, and synchronous altitude satellite constellations, the arrangement employing 27 satellites orbiting at 8000 naut mi was selected for its excellent geometry, worldwide coverage, and moderate number of satellites required to guarantee passive three-dimensional position fixing.

System accuracy depends on many factors, such as instrumentation errors, propagation effects, system geometry, radio interference, and the properties of the data links. The most significant factors are discussed in the following paragraphs.

The principal sources of range errors are listed in Table 3-7. For a worst-case estimate of the total measurement error accumulated in the direct ranging mode of navigation, the individual contributions are root-sum-squared.

3.2.1 Link Analyses

The results of the navigation link analyses are summarized in Table 3-3. The rf power budgets are given for both 400 MHz and 1600 MHz. Worst-case tolerances have been used in the design of each link to ensure reliable link performance The effective satellite antenna gains and the space losses apply to a satellite altitude of 7496 naut mi.

Table 3-3
Satellite-to-User Sequenced Tone RF Link

Parameter	400-MHz Carrier Frequency	1600-MHz Carrier Frequency
Modulation	SSB	SSB/SC
Duty cycle (%)	CW	0.7
Satellite transmitter power (dBm)	40.0*	40.0*
Satellite losses (dB)	-1.2	-1.4
Satellite antenna gain (dB)	10.4	10.4
Space loss (dB)	-170.6	-182.7
Received power at user (dBm)	-121.4	-133.7
User antenna gain (dB)	-3.0	-3.0
Circuit losses (dB)	-0.4	-0.6
KT (dBm/Hz)	-170.2	-170.5
c/KT (dB-Hz)	45.4	33.2
Required c/KT (dB-Hz)	32.0	28.3
Margin (dB)	13.4	4.9

*Peak power.

The performance characteristics for a typical high-accuracy receiver designed to work with the isotropic rf power available are shown in Table 3-4.

Power budgets for the tracking telemetry and command links are summarized in Table 3-5.

3.2.2 Signal Acquisition and Reacquisition

Two situations exist for a user employing the satellite navigation system: the first concerns the user's initial signal acquisition from the satellite and the second concerns subsequent signal reacquisition after a temporary loss of contact. Both types of acquisition occupy a finite time, the length of which depends on the amount of preliminary searching needed to (a) acquire the carrier and (b) track the ranging modulation. This preliminary search time depends on the quality of the apriori satellite-to-user range and the user velocity information available to the receiver. Table 3-6 contains the range and range rate uncertainties for both the range tone and antijam links.

3.2.3 Antijam Performance

The Timation system, like any other system employing electromagnetic sensors, can be jammed by a sufficiently determined enemy. The best that can be done in antijamming measures is to force the jamming system to be expensive. Ideally the expense of the jammer should be unreasonably burdensome to the enemy, while the added expense of the antijamming system should be modest.

Table 3-4
Satellite-to-User Antijam Link Characteristics

Parameter	400-MHz Carrier Frequency	1600-MHz Carrier Frequency
Modulation	KRP*	KRP
Duty cycle (%)	CW	10
Satellite transmitter power (dBm)	40	40†
Satellite losses (dB)	-1.2	-1.4
Satellite antenna gain (dB)	10.4	10.4
Space loss (dB)	-170.6	-182.7
Received power at user (dBm)	-121.4	-133.7
User antenna gain (dB)	-3.0	-3.0
Circuit losses (dB)	-0.4	-0.6
KT (dBm/Hz)	-170.2	-170.5
c/KT (dB-Hz)	45.4	33.2
Required c/KT (dB-Hz)	32.0	28.3
Margin (dB)	13.4	4.9

*Keyed random phase.
†Peak power.

Table 3-5
Tracking, Telemetry, and Command Links

Telemetry Link (2250 MHz)	Value
Transmitter	+20.0 dBm
Satellite antenna gain	+9.0 dB
Space loss	-185.3 dB
Other losses	-3.7 dB
Receiver antenna gain	+37.4 dB
Received signal power	-122.6 dBm
Receiver noise temperature	290°K
kT	-174.0 dBm/Hz
Available c/kT	51.4 dB-Hz
Command Link (1800 MHz)	**Value**
Transmitter	+50 dBm
Ground station antenna gain	+34.9 dB
Space loss	-182.8 dB
Other losses	-4.7 dB
Satellite antenna gain	-6.0 dB
Received signal power	-108.6 dBm
Receiver noise temperature	2600°K
kT	-164.4 dBm/Hz
Available c/kT	55.8 dB-Hz

Table 3-6
Receiver Acquisition Performance

Range Tones
Range uncertainty = 2000 naut mi/4200 ft Range rate uncertainty = ±7500 ft/sec/±19 ft/sec
Antijam System (KRP* Code)
Range uncertainty = 4200 ft Range rate uncertainty = ±19 ft/sec

*Keyed random phase.

3.2.3.1 Jamming — Electronic warfare is a game. There are several levels of sophistication in signal jamming, and there are corresponding levels of sophistication in signal design and receiver design to mitigate the effects of jamming. It is the purpose of this section to identify these levels and to suggest a choice of the radio navigation system signal design that makes effective jamming difficult while at the same time placing only a bearable burden on the navigation system user.

Level 1. The lowest level of jamming, in terms of effectiveness, is produced by radiating a jamming signal whose power spectrum is roughly uniform over the spectrum occupied by the desired signal. Such jamming is effectively countered by user receiver filtering that matches, in response spectrum, the power spectrum of the desired signal.

Level 2. The next level of jamming is produced by radiating a jamming signal whose power spectrum is matched to the power spectrum of the desired signal without regard to the phase of the spectral components. The simplest of radio navigation systems are resistant to "incoherent" jamming of this kind since they employ highly self-correlated transmitted signals — pulse trains, tone modulated carriers, etc. — and corresponding receivers that respond best to signals whose spectral components have prescribed phase relationships.

Level 3. A better jammer is one that radiates signals that precisely imitate, in both amplitude and phase, the characteristics of the signals of the radio navigation transmitter. Standard jammers that exist in the inventories of the armies, navies, and air forces of the world are capable of imitating AM, FM, swept, and unmodulated cw signals, pulsed signals, and other conventional emissions that have classically been used by radars and navigation systems in the past. An evident countermove open to the navigation system designer is to use unconventional signals, that is, signals that cannot be duplicated by standard equipment.

Level 4. The repeater jammer is one that precisely duplicates the desired signal by receiving, delaying, and reradiating the desired signal. The jamming can be effective if the desired signal is, like a sinusoid or a sinusoidally modulated sinusoid, of a kind whose sum with a delayed replica cannot be distinguished from the original signal, save for an indeterminable phase change. A countermove open to the system designer is to recognize that if the user is in motion with respect to the navigation system frame of reference, usually there will be a differential doppler shift between the received navigation signals and the received jammer signals; the navigation receiver, if sufficiently narrow in acceptance band and if aided by auxiliary measures of user movement (e.g., from inertial devices), can be arranged to reject doppler-shifted jamming signals that are encountered.

Obviously, however, such a counter does not apply to the user whose movement can be so slow as to prevent discrimination between desired and jamming signals on the basis of differential doppler shift. The countermove that has more general application is the use of signals which, when delayed by a jammer and added to the desired signal, remain distinguishable from the desired signal over most of the navigation system coverage area. Consider, for example, the transmission of an arbitrary pseudorandom binary biphase modulated sequence of length n and repetition period $n\tau$, where τ is the duration of a single binary digit. Assume that the navigation receiver locally generates an identical signal adjustably displaced in time and correlates it with the received radio navigation signal to derive a measure of transit time. Then, for any differential arrival time greater than τ between receipt of the transmitted and the delayed repeater-jamming signals, two distinguishable correlation peaks will be obtained: the earliest when the locally generated receiver sequence corresponds with the desired signal and the latest when the receiver sequence corresponds with the delayed jamming signal. Thus, the desired signal, because it arrives earlier, can be distinguished from the jamming signal. If the interval over which the correlator output is summed is made small, the spatial region in which the desired and delayed signals correspond is made correspondingly small; and if the interval over which the correlator output is summed is made large, noiselike perturbations introduced by the presence of the jammer into the measurement of correlation with the desired signal will be made small.

Level 5. A highly sophisticated enemy whose object was to jam a radio navigation system transmitting a repetitive pseudorandom sequence of the type described would depart from repeater jamming. Instead, his strategy would be to observe the repetitive sequence and duplicate it precisely, and to advance in time, rather than delay, his transmitted sequence relative to the desired sequence as observed at the jammer location. Thus, he could achieve a spatial correlation between the jamming and the desired signal in several regions in the coverage area. Where such spatial correlation was achieved, the user could not distinguish between desired and jammer except on the basis of differential doppler shift. By adjusting his synchronization, the jammer can sweep the region of confusion through the navigated system coverage area. The regions in which the jammer can achieve spatial correlation are hyperboloids of revolution — constant — time-difference surfaces — whose axes of symmetry lie on the line connecting the navigation transmitter and the jammer, whose thickness is $c\tau/2$ on axis (thicker off axis), and whose separation is $cn\tau/2$ on axis (greater off axis), where c is the velocity of light.

Thus, the navigation system designer can reduce the number of user-confusion regions to one (per jammer) by making $n\tau$ sufficiently large so that $cn\tau/2$ is greater than the distance between the jammer and the navigation transmitter. And by making τ very small, he can reduce the thickness of the confusion region so that the volume of the region is very small compared to the volume of navigation system coverage.

3.2.3.2 Recommended Signal Design — It is recommended that the antijam link transmit a repetitive pseudorandom sequence of repetition period $n\tau$. The duration τ of a single binary digit should be sufficiently small as to make negligible the confusion region that a jammer transmitting the same signal would produce at any instant in time, but at the same time sufficiently large that reasonably expectable maximum noise jamming-to-signal power (J/S) ratios (40 dB to 60 dB in the worst case) will not unlock the delay lock tracking loop in a navigation receiver. A τ of 100 nanoseconds, with an integration time of 0.1 second, will give an unlock J/S threshold of between 50 and 60 dB incoherent jamming. Figure 3-7 is a plot of J/S ratio versus range for a 1 kW effective radiated power (ERP) jammer.

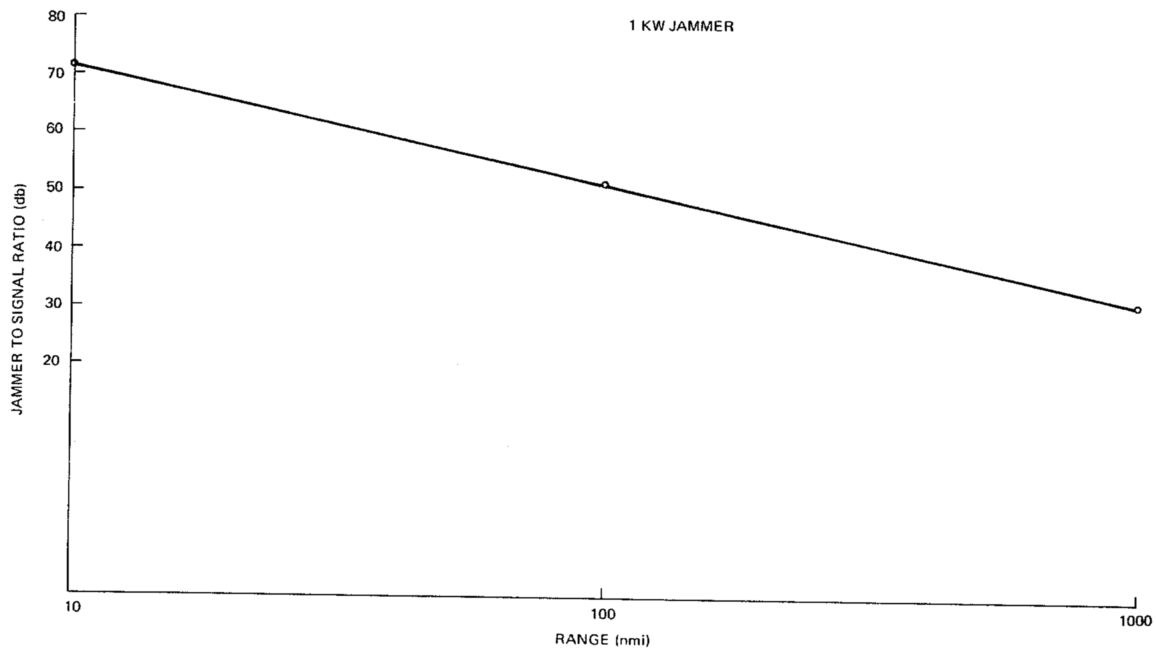

Fig. 3-7 - Jammer power vs jammer-to-user distance

3.2.3.3 Designing the Side-Tone Ranging for Antijam Performance — It cannot be expected that the side-tone ranging modulation will be as jam-proof as more sophisticated systems (such as were just described). The side-tone ranging modulation may be modified so as to increase the difficulty of jamming it by the provision of several changes in the receiving equipment and the frequency usage.

The first modification requires that the receiver first measure the satellite range when in a nonjammed environment. After the range is once measured, it is updated by tracking the doppler signal with a narrowband filter. The local oscillator that controls the filter can be in turn controlled by an inertial platform to reduce the required bandwidth still further. In addition this combination can be programmed to show the presence of spoofing and jamming signals.

For antijam performance the satellite doppler signals would be spread over the allocation band (rather than having each plane on the same transmitted frequency). The spacing between frequencies might be approximately 30 kHz. To disallow the use of a 30-kHz modulation jamming all channels the actual spacing would vary with some separations being 20 kHz, other 24 kHz, 28 kHz, 30 kHz, etc.

The objective of using varying separations is to force the enemy into using a noise jammer. Once he is forced to do this the cw system is competitive with the more sophisticated brand of antijam, when using the narrowband technique described above.

3.2.4 Orbit Determination and Ephemeris Prediction

The position fix accuracy available to the users depends in part on the accuracy of the broadcast ephemerides. The accuracy of the broadcast ephemerides in turn depends on the accuracy of orbit determination and the accuracy of prediction of the ephemerides over the period from the end of a given tracking span through the orbit determination and satellite memory injection to the end of the applicable period of a complete ephemeris message (26 hours worst case). The selected constellation is excellent for both orbit determination and ephemeris prediction. Accurate orbit determination is assisted by the high-percent visibility time from the four ground stations. Prediction accuracy is good because the satellites are above the atmospheric drag region and other forces of uncertain magnitude (variations in solar pressure, propulsion system leakage, etc) are small.

The quality of ephemeris predictability is therefore principally determined by the sophistication of the orbital dynamic model used in both the orbit determination and ephemeris prediction computer programs, and by the quality of the tracking data. It is necessary to model all known forces, including gravitational forces of the sun and moon and solar pressure effects.

Tracking data from the remote sites are preprocessed to remove the random errors and to compact the data before transmittal through the system satellites to the central processing facility. The orbits are determined, using prior estimates of orbital parameters and recent tracking data obtained with standard programs developed by the Naval Weapons Laboratory (NWL) and modified for application to the selected orbits. From the orbits thus determined, the predicted orbital elements for up to 1 day are generated and put into a form suitable for interpolation by the satellite calculator. This process is carefully scheduled to ensure utilization of the most recent tracking data and to minimize the overall prediction period.

A sample contact time profile for the Samoan station is shown in Fig. 3-8. Similar schedules apply to the Guam and Virgin Island stations. Each satellite is in view of a given near-equatorial site for about 25% of the orbit. The percent time from Alaska is about 18%. Each satellite is in view of at least one tracking station 80% of the time, and the maximum continuous period during which any satellite is out of view is about 220 minutes.

The method of tracking relies mainly on two-way range techniques similar to and compatible with those used in the operationally proven Timation ground station. Doppler data are particularly useful for providing accurate short-term range difference values used for satellite clock synchronization from the remote sites. The navigation signals provide ionospheric refraction corrections and a fully redundant means of tracking the satellites.

3.2.5 Navigation Accuracy

A measurement error model was assumed to permit computer modeling for estimating the navigation accuracy available with the system; the values assumed were typical and are shown in Table 3-7. The estimates derived were based on the assumption that the user takes full advantage of the system capabilities but without using an inertial navigator or other sensors. Figure 3-9 shows the error contours for the high-accuracy user both with and without a good clock. Figure 3-10 is a plot of geometrical dilution of precision (GDOP) for every 10 degrees of latitude for the chosen constellation. The error contours of Fig. 3-9 are obtained by multiplying the error budget of Table 3-7 by the GDOP values of Fig. 3-10. The values for the "good clock" contours correspond to the lower envelope of the GDOP curves. The values for the continuously corrected clock correspond to the upper envelope of the GDOP curves. In areas where the satellites are out of view of the ground stations, the error values are caused by the maximum clock errors for the interval the satellites are out of view.

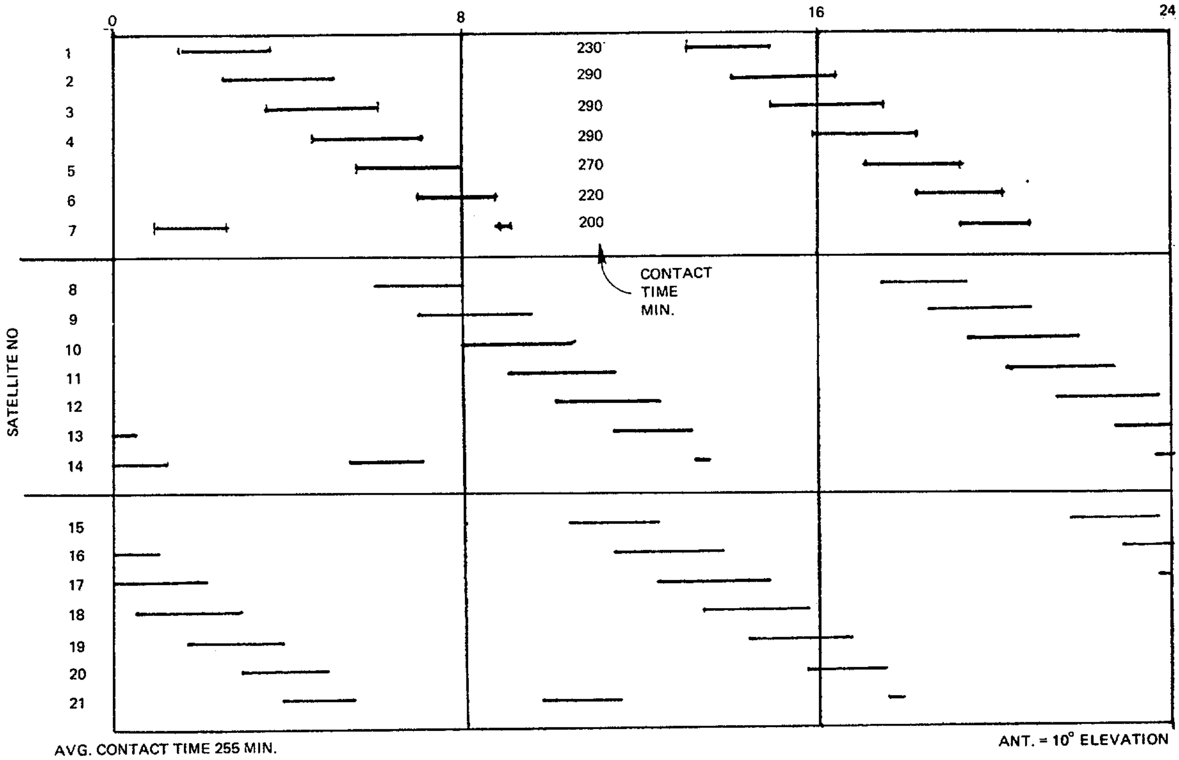

Fig. 3-8 - Samoan ground station contact time profile

Table 3-7
Typical ~~Maximum~~ Accuracy Error
Budget for a User

Multipath	2.5 nsec or feet
Ionosphere	2.0 nsec or feet
Troposphere	2.0 nsec or feet
Equipment	5.0 nsec or feet
Satellite position	3.0 nsec or feet
Clock synchronization	5.0 nsec or feet
RSS	8.5 nsec or feet (value used = 10 nsec + clock error)

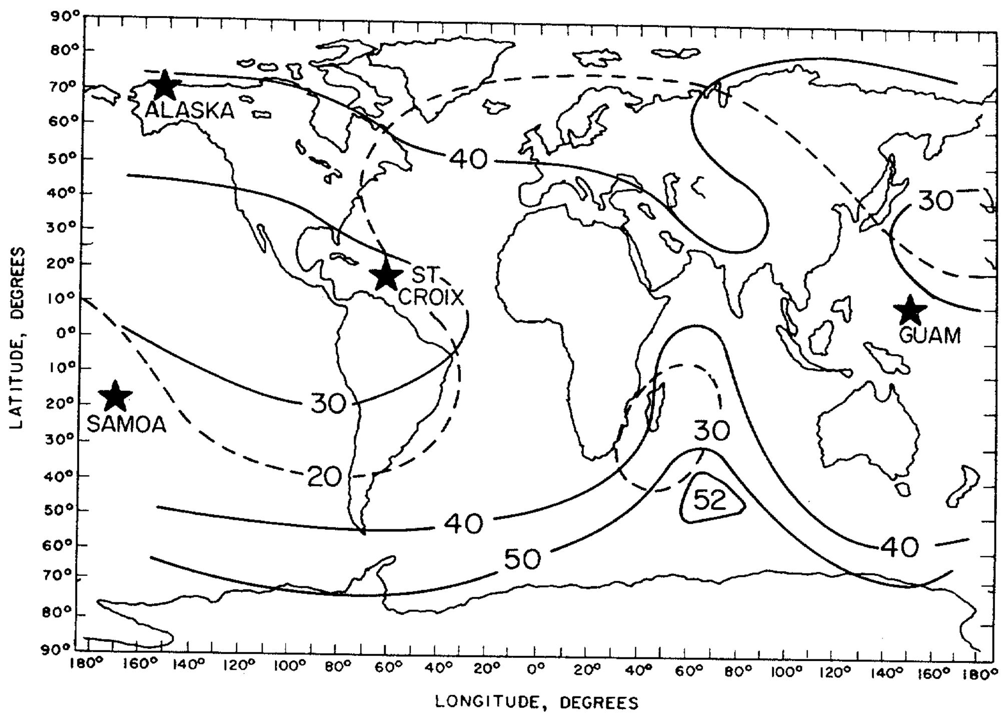

Fig. 3-9 - Error contours in feet for ground stations shown; 3-by-9, 55-degree inclination, 8-hr constellation. Solid lines show fixes with continuously updated clock; dashed lines correspond to user with good clock.

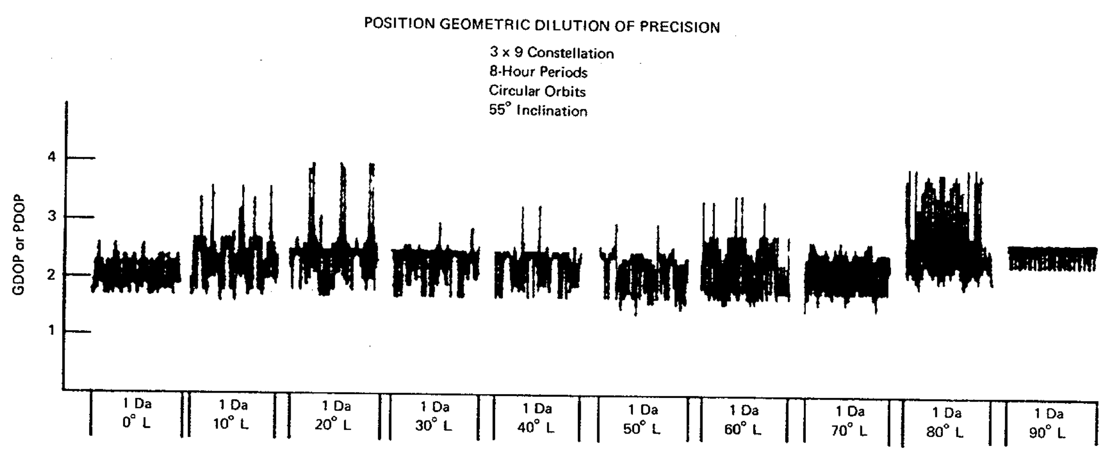

Fig. 3-10 - Position geometric dilution of precision

3.3 Space Segment

The spacecraft employed in the Timation system are designed to fulfill the primary mission of transmitting ranging and ephemeris information to system users. Four radio links are provided to pass this information. The principal link operates at 400 MHz, employs fixed-range-tone modulation, and carries the ephemeris data in the form of a binary coded subcarrier. A subsidiary link operating at 1600 MHz determines the range differential caused by dispersive ionospheric refraction; this link also employs fixed-tone modulation. Two additional links have been provided for a substantial degree of ECM resistance. These protected links operate at 400 MHz and 1600 MHz and employ KRP* modulation techniques. Ephemeris data are also transmitted via these links.

Tracking, telemetering, and control of the spacecraft is accomplished through an S-band communications link which also carries messages for storage in the spacecraft memory. The spacecraft is equipped with a maneuvering capability to facilitate acquisition and retention of its proper station.

To support its mission the configuration of the spacecraft includes the following subsystems: (a) structure, (b) communications, (c) power supply, (d) thermal control, (e) attitude control, and (f) propulsion. A block diagram illustrating the functional characteristics of the active subsystems is shown in Fig. 3-11.

Hohmann-type ascent profiles are used to maximize the payload weight transfer into the required Timation orbits. In the operational payload configuration of nine satellites the Burner II vehicle and payload are boosted by the Titan IIID into a ballistic trajectory with an apogee at 8000 naut mi. The Burner II motor will ignite at this apogee and provide the impulse needed to reach the circular orbit velocity. As soon as the required velocity is attained, dispensing of the satellites into their appropriate stations will proceed.

The replenishment operation will be effected in a similar manner using the Titan IIIB and Burner IIA combination to place the two satellites in specified orbits.

3.3.1 Spacecraft Structure

The spacecraft is a simple cylindrical structure (Fig. 3-12). The cylinder is 4 feet long and 5 feet in diameter, the sides and one end are covered with solar cells while the remaining end is a high thermal conductivity baseplate which supports most of the subsystem components. A number of access ports have been provided for stabilization sensors and propulsion nozzles.

In flight the cylinder is oriented with its axis normal to the orbit plane and the end with solar cells toward the sun. The 1600-MHz antenna fitted into the side the cylinder always faces the earth; the 400-MHz cupped helix is attached to the rim of the baseplate and hinged to permit deployment from a prelaunch position alongside the cylinder to an earth-oriented position in-orbit.

An annular omnidirectional S-band antenna attached to the rim of the baseplate completes the complement of antennas.

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*Keyed random phase.

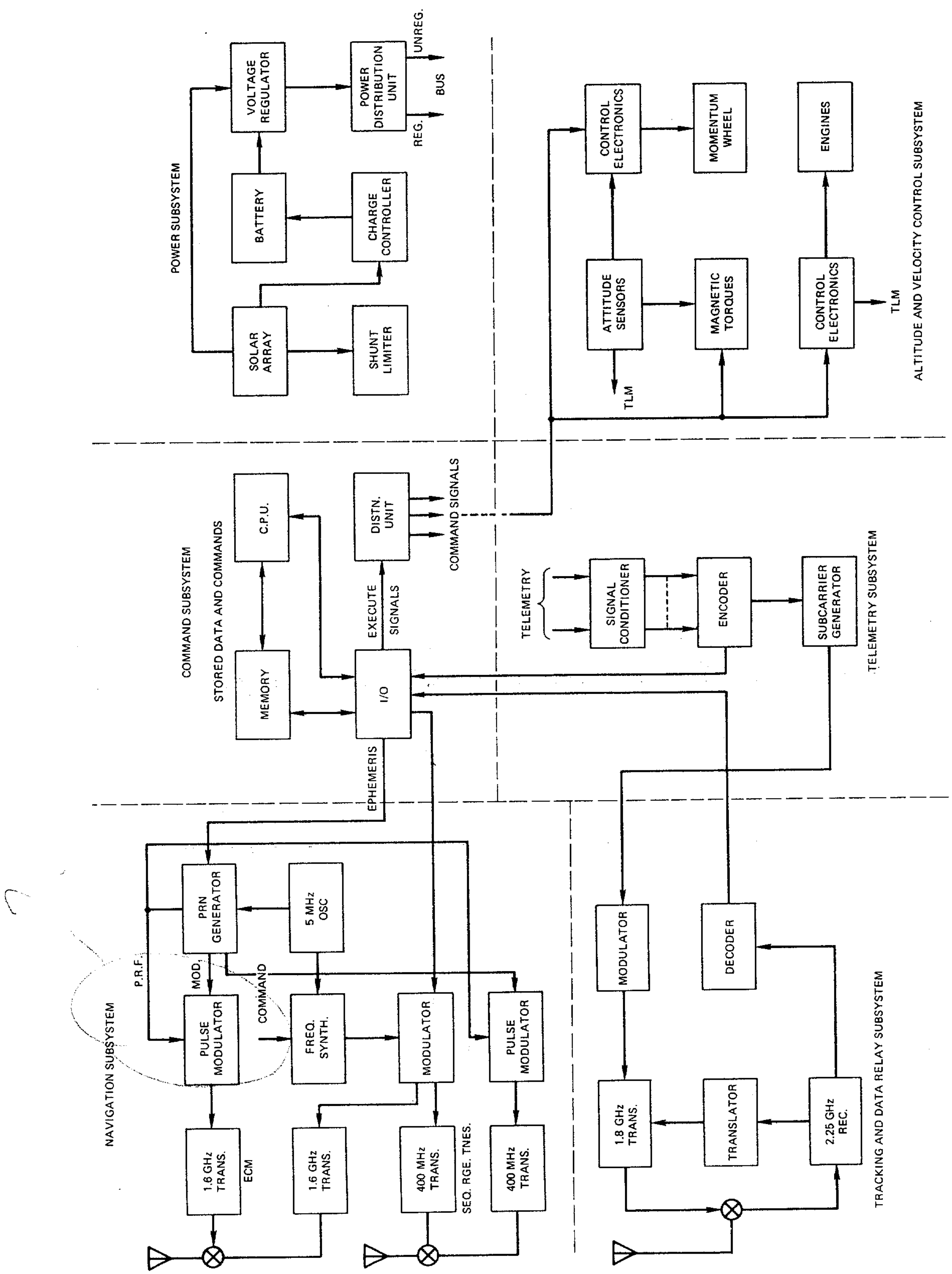


Fig. 3-11 - Timation satellite functional block diagram

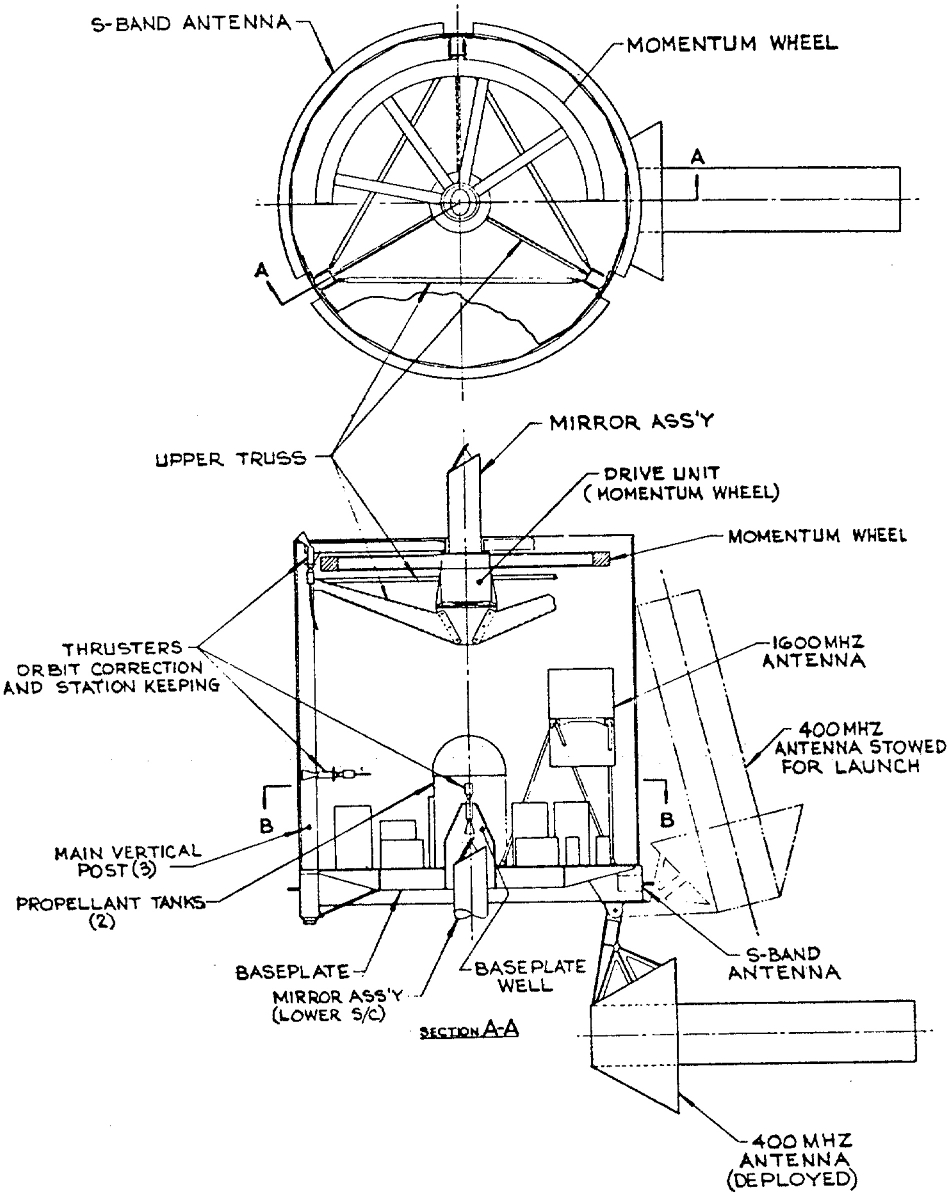


Fig. 3-12 - Spacecraft mechanical configuration

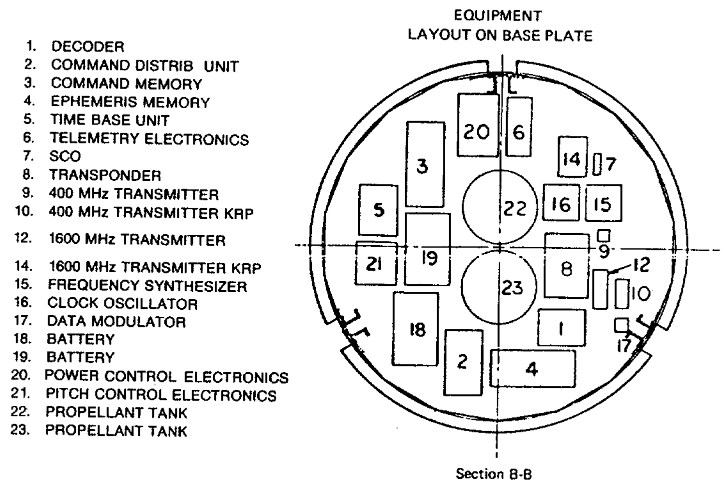


Fig. 3-13 - Spacecraft component layout

The components mounted on the baseplate are located and grouped as shown in Fig. 3-13 to facilitate integration and testing. The momentum wheel assembly which provides altitude control is supported on three columns from the baseplate and positioned directly beneath the top of the solar array hat. Propellant storage tanks are located in the space between the baseplate and the momentum wheel and disposed symmetrically about the cylinder axis.

The structure is designed to permit the stacking of nine spacecraft into a single payload. A representative stacking configuration is shown in Fig. 3-14. A summary of the weights of the various subsystems and major components is listed in Table 3-8.

3.3.2 Data Package

The spacecraft data package performs all those functions required to implement the data links between satellite and user and satellite and ground station. The packages contain the navigation subsystem, the telemetry subsystem, the command subsystem, and the tracking transponder. A functional block diagram of the data package is shown in Fig. 3-15.

The fundamental element of the navigation subsystem is a precision 5-MHz oscillator with a stability of 1 to 2 parts in 10^{12} and a very low drift rate. This oscillator is housed in multiple-temperature controlled ovens and is the source of all satellite timing signals. The oscillator output supplies a frequency synthesizer which produces the 400-MHz and 1600-MHz carrier frequencies and the ranging modulations. The frequency and phase of the oscillator is adjustable by ground command.

The command, tracking telemetry, and data relay subsystems function via a two-way S-band communications link. The uplink operates in the frequency band 1.7 to 1.8 GHz and carries commands, ephemeris data, and ranging modulation, at a data rate of 256 bits per second (bps) using intermediate subcarriers where appropriate. Data transfer from remote tracking stations is handled by the tracking transponder modified to function as a repeater. The downlink operates in the band 2.2 to 2.3 GHz and carries the returned ranging signal and telemetry. The command subsystem employs frequency shift keying of three command tones which together with the ranging signals and repeater subcarrier

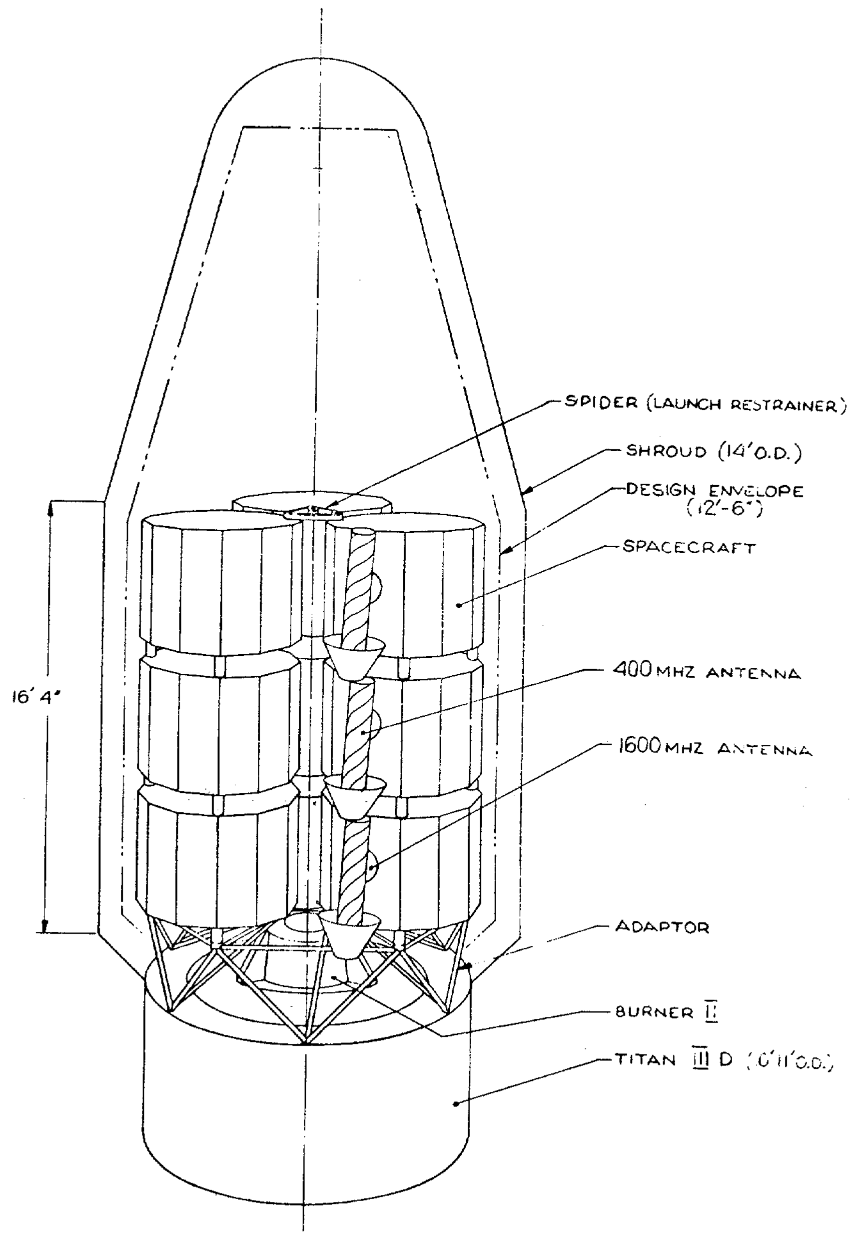


Fig. 3-14 - Nine-satellite launch configuration

Table 3-8
Weight Summary

Subsystem	Weight (lb)
Structure	85
Harness	20
Balance weights	6
Misc. hardware	5
Navigation electronics*	41
Tracking and data relay*	31
Telemetry*	11
Command and control*	40
Attitude control*	53
Propulsion	35
Power Supply	125
Total	452

*Includes redundant components.

are phase modulated on the uplink rf carrier. Similarly, the telemetry is phase modulated on a separate subcarrier, summed with the repeater sub-carrier and ranging modulation, and phase modulated on the downlink rf carrier. Satellite ephemeris data and programmed commands are stored in the spacecraft computer. This computer performs the following functions:

1. Interpolation of stored ephemeris data points and conversion to a form convenient to the navigator.

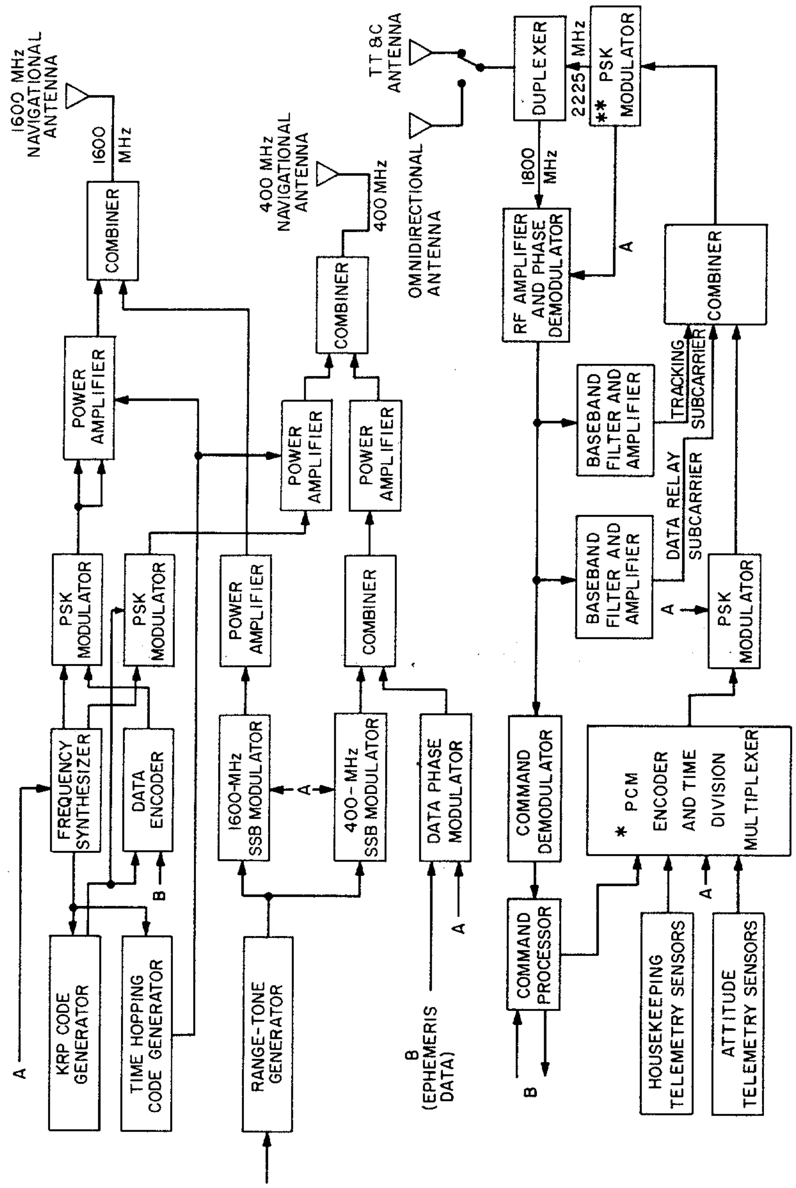
2. Programmed control of the attitude control and propulsion subsystems.

3. Adjustment of the spacecraft clock.

All the antennas associated with the communications package are circularly polarized. The L-band antenna is a flush-mounted, cavity-backed spiral and is designed to radiate both 1600-MHz navigation signals and the S-band downlink carrier. The 400-MHz antenna is a helix with a cup reflector and transmits only the 400-MHz navigation signals. The omnidirectional coverage required for command reception is provided by the circumferential waveguide-fed dipole array.

3.3.3 Power Generation and Thermal Control

The power subsystem energy will be derived from the solar array, and the excess available during the sunlit portion of the orbit will be stored in nickel cadmium batteries. The subsystem utilizes a direct energy transfer configuration employing two



NOTE IN THE DIAGRAM "A" INDICATES "FROM FREQUENCY STANDARD"
"B" INDICATES "FROM COMPUTER"

* PCM = PULSE CODE MODULATION
** PSK = PULSE SHIFT KEYER

Fig. 3-15 - Data package block diagram

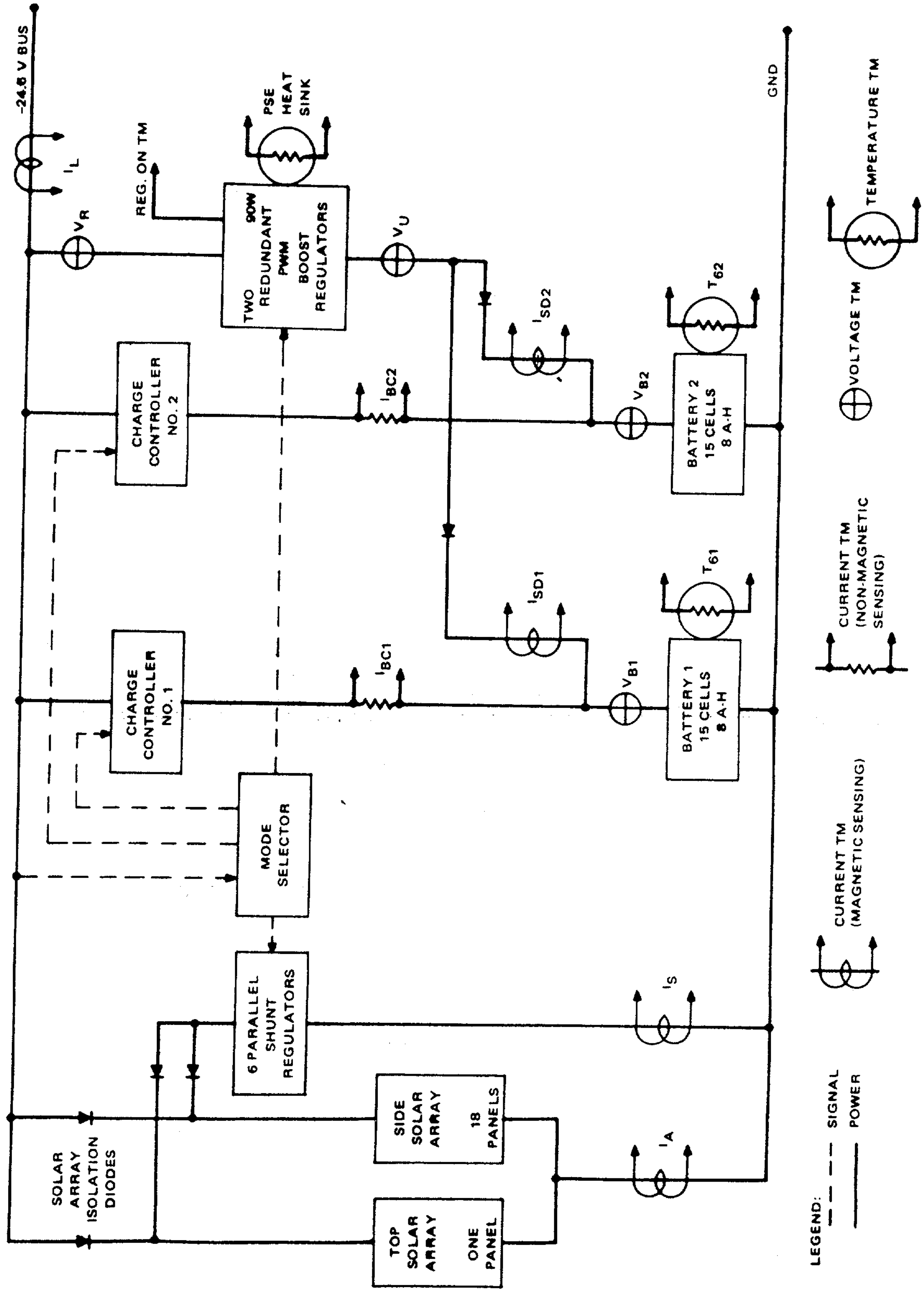


Fig. 3-16 - Power subsystem block diagram

Table 3-9
Spacecraft Electrical Loads

Subsystem	Average DC Power (watts)
Navigation electronics	50
Tracking and telemetry	12
Command and control	10
Attitude control	11
Total	83

batteries, a control and conditioning module, a shunt regulator assembly, and a solar array which is attached to the hat of the spacecraft. The subsystem, shown in Fig. 3-16, is designed to supply the electrical load for 5 years. The satellite load requirements are shown summarized in Table 3-9.

The thermal control subsystem has the baseplate as the principal heat exchanger providing radiative heat transfer into space. The temperature of the baseplate can be adjusted by covering the outer surface with a radiation stopping blanket and cutting holes to produce the proper radiation coupling.

3.3.4 Attitude and Velocity Control

The principal characteristics required of the attitude control system are (a) long life, (b) low weight, power consumption, and cost, and (c) accurate pointing (±3°). The system employs a dual-spin stabilization concept (Fig. 3-17). It provides precise three-axis attitude control and features long operational life, low weight and power drain, and adaptability to spacecraft of any size. The angular momentum needed for altitude stabilization is provided by a small fly wheel spinning concentrically with the principal axis of the spacecraft. This fly wheel provides roll and yaw stiffness. Control of the pitch axis is obtained through reaction of the spacecraft body to the fly wheel. Pitch, roll, and yaw attitude errors are derived from two infrared sensors which generate reference pulses when their fields of view intercept the earth's horizons. The subsystem electronic assembly processes the attitude error data and generates appropriate magnetic and inertial torques that will correct the spacecraft attitude. As discussed previously, the spacecraft is cylindrical in shape, and oriented so that its principal (pitch) axis is normal to the orbit plane. By suitable distribution of the angular momentum between the structure and the fly wheel, the body of the spacecraft is made to rotate around the pitch axis at one revolution per orbit to maintain the required earth-oriented attitude.

The attitude control subsystem performs another important function during its operational life. Every 6 months when the sun angle reaches 90° the spacecraft pitch axis is reversed. This maneuver ensures that the baseplate and the solar array are always properly oriented.

To acquire and maintain station within the Timation constellation, individual satellites require auxiliary propulsion systems capable of correcting orbit injection errors and general orbital drift occurring during the lifetime of the spacecraft. The propulsion system has been sized to be compatible with the Titan III/Burner II series of launch vehicles and

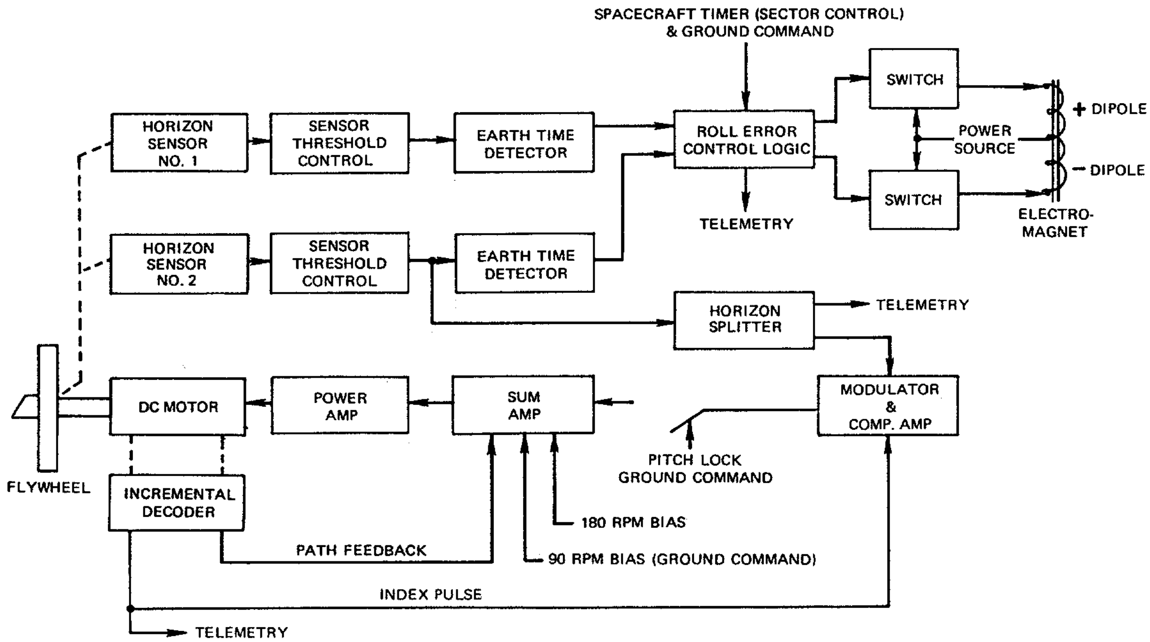


Fig. 3-17 - Attitude control subsystem block diagram

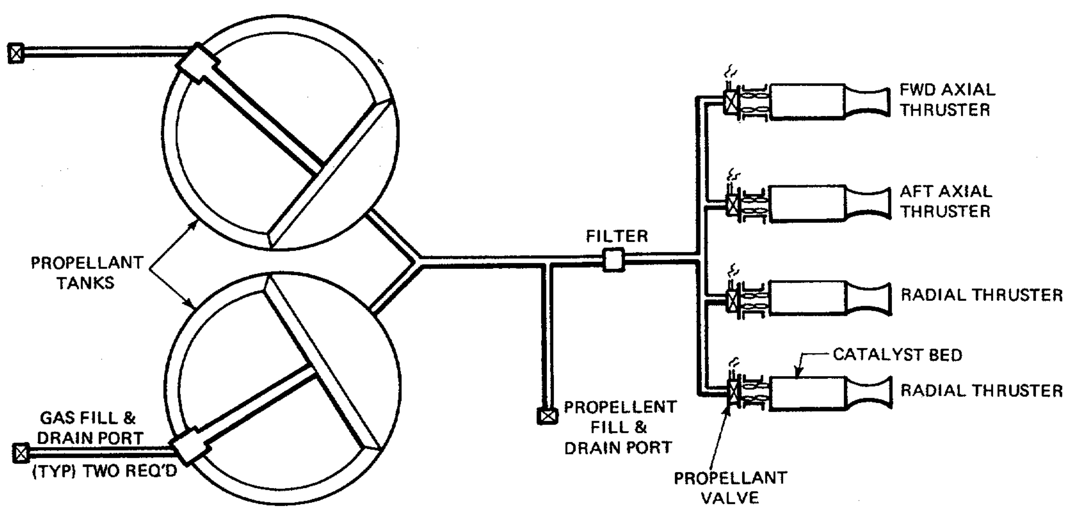


Fig. 3-18 - Propulsion subsystem diagram a spacecraft lifetime of 5 years. A diagram of the propulsion subsystem is shown in Fig. 3-18. The subsystem consists of four thrusters and a propellant feed system, each thruster incorporates a valve, a catalyst bed, a decomposition chamber, and an expansion nozzle. The hydrazine propellant is fed from the tank under the pressure of dry nitrogen. On command the appropriate propellant valve opens, the pressurant gas forces fuel into the catalyst bed. Decomposition of the fuel develops pressure at the throat of the nozzle and produces thrust by expansion through the nozzle. The propulsion subsystem has an overall velocity correction capability of 450 feet/sec.

Control of both the attitude and propulsion subsystems is accomplished either by programmed commands stored in the spacecraft computer or by real-time ground commands.

3.4 Ground Segment

The ground station network required to support the Timation navigation satellite system is described in this section. The functions of the ground segment were previously identified; they are summarized according to the signal flow in Fig. 3-19.

The principal facilities of the ground system are (a) the three remote ground stations (RGS), (b) the central ground station (CGS), and (c) the operations or control center.

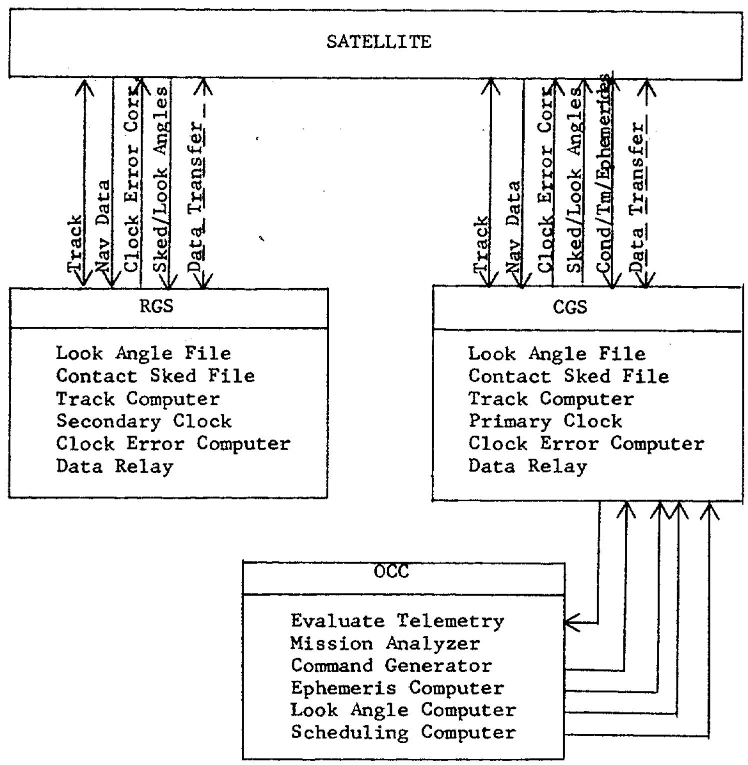
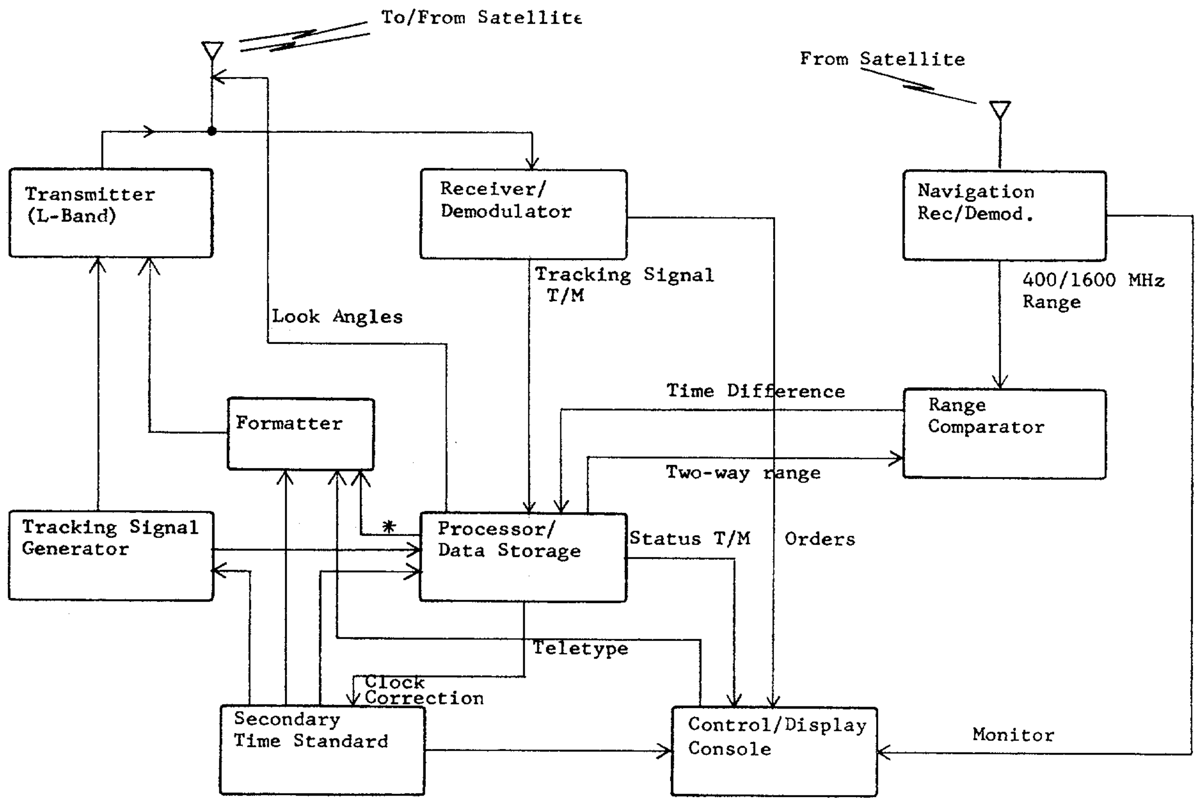


Fig. 3-19 - Ground station complex functional flow diagram



*Compacted tracking data satellite clock corrections

Fig. 3-20 - RGS functional block diagram

The preferred locations for the facilities are (a) the central ground station and operations center in Alaska and (b) the remote ground stations in Guam, Samoa, and the U.S. Virgin Islands. The distribution of the ground stations is shown in Fig. 3-5. This arrangement provides adequate tracking and clock synchronization opportunities combined with politically acceptable siting.

3.4.1 Remote Ground Stations

The remote ground stations perform the following functions as illustrated in Fig. 3-20.

1. Track the satellites.

2. Preprocess the received tracking data and prepare it for transmission to the central ground station.

3. Correct the epoch of the local time standard by comparing it with the system standard via a covisible satellite.

4. Transmit data to, and receive from, the central ground station via a covisible satellite.

5. Correct the clock errors of all satellites in view as necessary.

6. Monitor the navigation signals.

The remote stations are designed to maintain contact with up to six satellites simultaneously through six channels. The hardware includes (a) a 30-foot-diameter parabolic steerable S-band antenna, (b) 400-MHz and 1600-MHz omnidirectional antennas, (c) 400-MHz and 1600-MHz user-type receivers, (d) S-band transmitter, receiver, command telemetry, and tracking equipment, and (e) a data transfer terminal.

In addition, the following items service all six channels:

1. Two atomic frequency standards from which all operating frequencies and timing signals are derived. These oscillators function as local standards.

2. A general purpose digital computer for preprocessing the tracking data, data formatting and control, antenna pointing, fault isolation, and system monitoring.

3. Status display and control panel.

3.4.2 Central Ground Station

The central ground station is the principal communications terminal for the Timation navigation satellite system. It is designed to perform the following functions in addition to those performed by a RGS:

1. Transmit command and ephemeris data to the satellites.

2. Receive telemetry data from the satellites.

3. Transmit data to, and receive data from, the remote ground stations via covisible satellites.

4. Transfer all tracking and telemetry data to the operations center.

5. Perform all functions of a RGS.

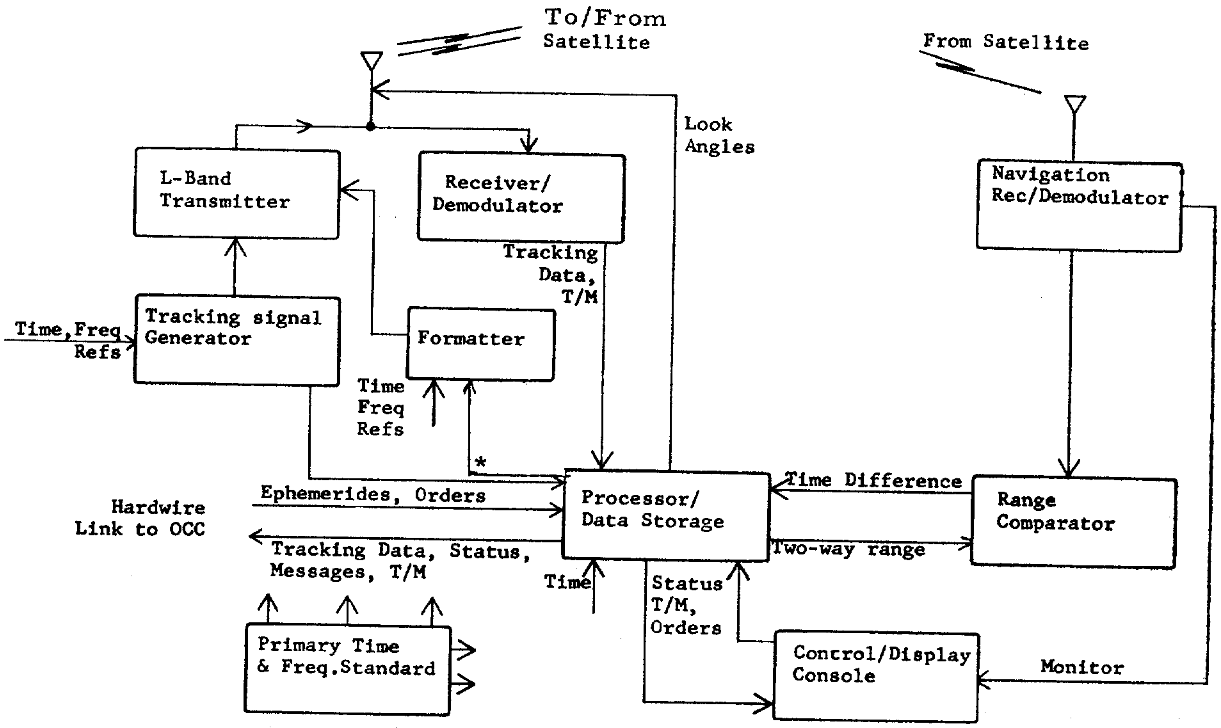
A functional block diagram of the CGS is shown in Fig. 3-21.

Appropriate equipment is provided to perform these tasks and permit simultaneous contact with up to nine satellites through nine separate channels. The hardware for each channel is similar to that for a RGS channel.

In addition, the following items service all nine channels:

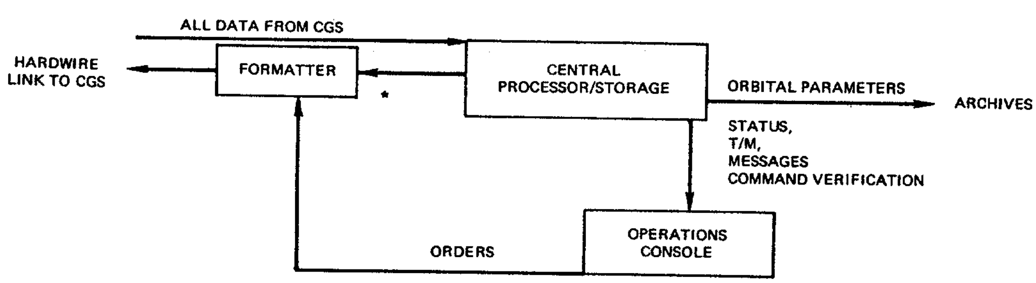
1. Two hydrogen maser oscillators from which are derived all operating frequencies and timing signals. The system time and frequency reference so derived is synchronized to Naval Observatory time.

2. A general purpose digital computer for tracking data processing, data formatting and control, antenna pointing, fault isolation, and system monitoring.



*Clock Corrections (RGS and satellite), Ephemerides, Status, Orders

Fig. 3-21 - CGS functional block diagram



*EPHEMERIDES, SCHEDULES, COMMANDS

Fig. 3-22 - Operations center block diagram

3.4.3 Control and Data Processing Center

The control center is the main system computational facility and focal point of all system operations. The center may be remote from or colocated with the central ground station; in either case the two facilities are hardwire connected.

The operations center performs the following functions as shown in Fig. 3-22:

1. Reception of tracking and telemetry data from the central and remote ground stations.

2. Analysis and evaluation of the satellite telemetry data.

3. Analysis of the satellite tracking data, updating of the satellite orbits, and generation of predicted ephemerides.

4. Transmission of command, ephemeris, and other data to the central ground station.

5. Display of information required to support command and control operations.

6. Monitoring of system operations and timekeeping.

The equipment to implement these functions is:

1. Communications equipment for interfacing with the central ground station.

2. Digital multiprocessor for orbit determination, telemetry analysis, ephemeris prediction, system analysis, status update and display, data formatting, and operations control.

3. Man/machine interface equipment (control consoles, displays, etc.).

All major Timation system computing activities are concentrated in the operations center. Maximum advantage is taken of the opportunity to automate routine functions to free the human controllers for the more important tasks of monitoring and supervision. The system is designed to make rapid and flexible responses to routine or unscheduled events. The operating staff retains full control at all times and can intervene whenever necessary.

3.5 User Equipment

One of the primary objectives of user equipment design is to develop an interface with the other system elements (satellites and ground stations) that will meet the needs of a wide variety of users. Some users will require maximum performance irrespective of cost while others will prefer to pay lesser amounts for reduced capabilities, i.e., lower accuracy, longer fix acquisition times, or fewer fix parameters.

Users having approximately similar position-fixing objectives have been grouped into specific categories. This effort limits the number of separate user equipment configurations which must be developed to satisfy the wide range of requirements indicated by previous studies of the navigation problem. Equipment configurations designed to satisfy each category have been conceived. Each configuration contains similar elements, such as a single or multichannel receiver, a clock, a digital processor, and a display. The complexity and cost of each element depends on the accuracy of the position fix, the rate of fixing, the degree of automation, and the number of parameters to be displayed.

Specific categories of user equipment are identified, the characteristics of the corresponding equipment configurations are discussed, and representative examples of the receiver, data processor, and display are described in the remainder of this section.

Each user category has the option of antijam or sequenced range-tone receivers, according to the requirements for operation in hostile environments.

3.5.1 Categories of User

Six categories of user have been provisionally identified, covering both military and civilian operations.

Category 1 - Combat Aircraft. Category 1 users are tactical military aircraft, such as fighter bombers and ground support helicopters, which need continuous maximum accuracy and three-dimensional position information.

Category 2 - Marine Operations. Category 2 users are submarines, aircraft carriers, mine layers, and other naval vessels requiring high-accuracy, two-dimensional position fixes. Because of the lower operating speeds and different mission characteristics, fixes are required less frequently than for category 1 users, particularly where inertial navigators are available.

Category 3 - Land Operations (Vehicle). Category 3 users are Army tanks, jeeps, field headquarters, and other mobile vehicles who require high-position-fixing accuracy relative to base headquarters or other reference location.

Category 4 - Land Operation (Foot). Category 4 users are military foot patrols engaged in reconnaissance, advanced area operations, and search and destroy missions.

Category 5 - General Air Navigation. Category 5 users are military, commercial air military, and commercial air transports operating over long distances which require infrequent three-dimensional position fixes with accuracies in the 1-naut mi range. For these users, equipment costs must be competitive with existing navigation systems.

Category 6 - General Marine Navigation. Category 6 users are mainly merchant ships and military sea transports. For these users, equipment cost is of prime importance, and the cheapest device which meets their low-accuracy, two-coordinate position fixing criteria will be favored.

3.5.2 Equipment Characteristics

To satisfy the requirements identified with the user categories discussed in the previous section, corresponding equipment configurations have been developed. These configurations are described in the following paragraphs and figures and summarized in Table 3-10.

The category 1 configuration employs a four-channel ranging receiver along with a single-channel ionospheric refraction correction receiver operated sequentially. For operation in a jamming environment an alternative four-channel receiver designed to

Table 3-10
Timation User Equipment Characteristics

Equipment Category	No. of Range-Tone Channels	No. of KRP Channels	Ionospheric Refraction Correction Channel	Timation Display Required	System Processor	External Sensor Inp. Req'd	Clock Precision
1	4	4	Yes	No	Digital Computer	None	1pp10^9
2	3	3	Yes	No	Digital Computer	Altitude	2pp10^{12}
3	3	3	Yes	Yes	Digital Computer	Altitude	2pp10^{12}
4	4	4	Optional	Yes	Digital Computer	Altitude	2pp10^{12}
5	3	0	No	Yes	Digital Computer	Altitude	
6	1	0	No	Yes	Electronic Calculator	Velocity	

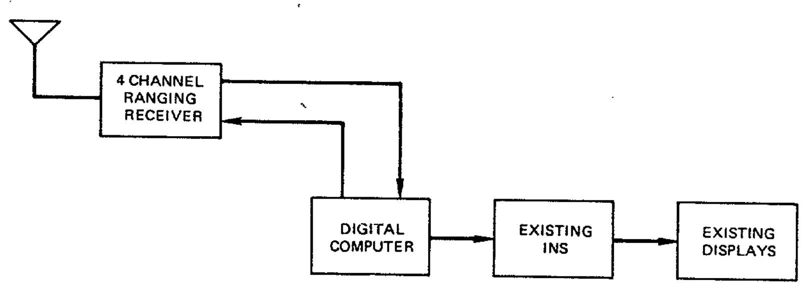


Fig. 3-23 - Category 1 equipment

accept coded spread spectrum signals is available. The digitized receiver outputs are processed by a digital computer and used to derive required navigation data, such as position, speed heading, ETA, etc. The computer also selects the optimum satellite configuration and pretunes the receivers. The navigation data are displayed in the form most convenient to the navigator under jamming conditions. An inertial platform is assumed available to provide the continuity between fixes needed to guarantee continuous high accuracy. Figure 3-23 diagrams the category 1 configuration.

The category 2 configuration employs a three-channel ranging receiver along with the single-channel ionospheric refraction correction receiver. The use of three navigation

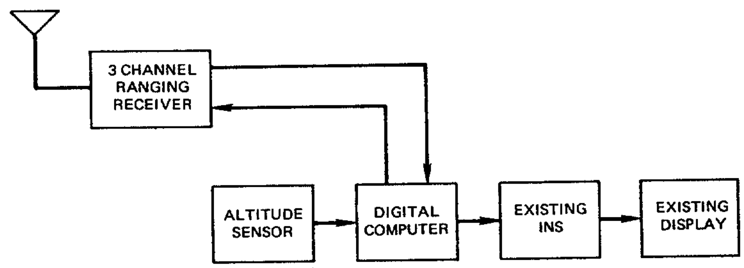


Fig. 3-24 - Category 2 equipment

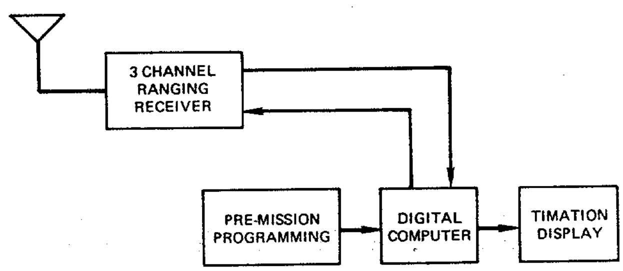


Fig. 3-25 - Category 3 equipment

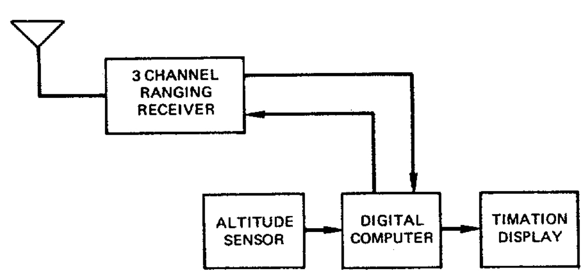


Fig. 3-26 - Category 4 equipment

channels requires, for a three-dimensional position fix, an independent measurement of altitude by the user, which contributes to a simpler computer. The computer selects the optimum three-satellite configuration, pretunes the receivers, and processes the receiver outputs to provide navigational data. A user's existing inertial platform is updated by the navigational data, thereby providing continuous position readout on a display designed to meet the specific needs of the navigator; in many cases a user's existing display will suffice. Category 2 includes an alternative three-channel KRP receiver for use in a jamming environment. A block diagram of category 2 equipment is shown in Fig. 3-24.

Category 3 equipment uses three satellite ranging channels and one channel for ionospheric refraction correction. A three-channel user will not possess an inertial navigation system. Hence, equipment for this category includes a display updated by a computer which must interpolate satellite ephemerides in order to provide a continuous position readout. Ephemeris extrapolation is provided in addition to the standard computer functions of constellation selection, receiver tuning, and position determination data processing. Category 3 equipment is shown in Fig. 3-25.

The category 4 equipment is built into a manpack unit for foot patrol use. Category 4 equipment consists of three 400-MHz receiver channels corrected for ionospheric refraction by an additional channel at 1600 MHz. Alternatively, the 1600-MHz channel can be omitted and corrections for refraction can be applied later at a nearby location where refraction had been measured.

Position fixes will be obtainable every 2 minutes using a small computer built into the manpack. The computer will be programmed with receiver tuning information at the start of each mission based on the known satellite orbits. The manpack can be operated as a self-contained, portable navigation station requiring no external computational equipment or data links to remote facilities. The category 4 equipment block diagram is shown in Fig. 3-26.

Category 5 equipment is designed for commercial aircraft or military aircraft in a nonjamming environment. This equipment category uses a three-channel, 400-MHz ranging receiver. The computer thus requires an independent user altitude input. The computer is not required to interpolate satellite ephemeris data; therefore, provisions are made for position location updates to the display once every 2 minutes. The computer can be simplified in the interest of reduced cost by a preflight programming of satellite constellations to be used during the flight. Figure 3-27 diagrams category 5 equipment.

Category 6 consists of a single-channel, 400-MHz ranging receiver which is time shared among three satellites. The digital computer of the other categories is replaced in category 6 by a simpler electronic calculator and the receiver is manually pretuned. Satellites are selected on the basis of their rf carrier frequency without regard for constellation optimization. The occurrence of poor constellation geometries could be minimized by pretuning the receiver to widely separated satellite frequencies and perhaps by using a fourth satellite as a check. The category 6 user must account for his ship's motion, if any, during the fix determination period. The category 6 equipment block diagram is given in Fig. 3-28.

3.5.3 Receiver

Although the exact hardware complement comprising the Timation receiver varies with the user category, constituent equipment falls into two general types: a coherent range-tone receiver subsystem and a noncoherent KRP receiver system. The range-tone system is employed by all user categories in the absence of jamming. The KRP system is employed only by military user categories.

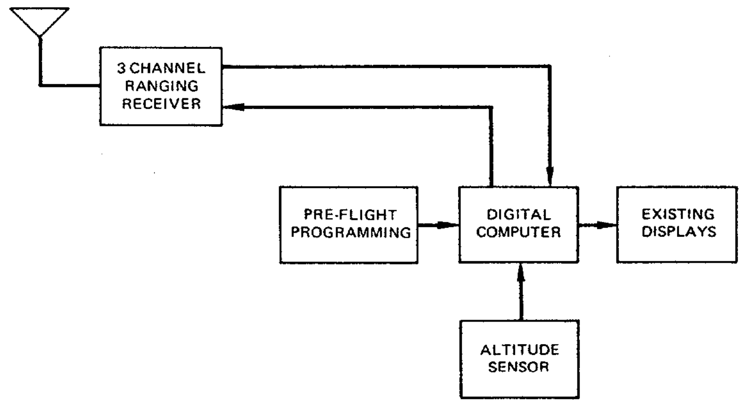


Fig. 3-27 - Category 5 equipment

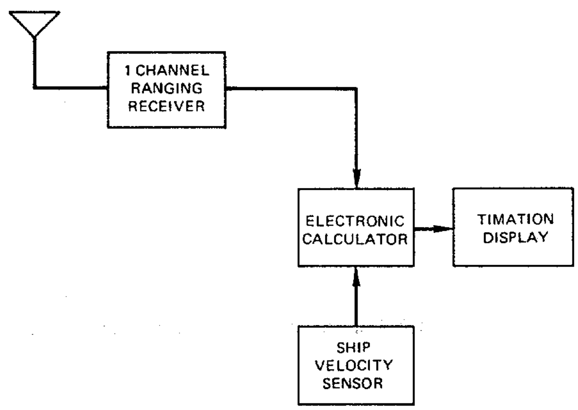


Fig. 3-28 - Category 6 equipment

Each system is generally a multichannel device with the exact number of channels depending on the number of satellites to be simultaneously received in order to meet user category navigation requirements. However, within each system, channels are identical, hence, it suffices to describe a single channel in each of the system discussions presented below.

3.5.3.1 Range-Tone Receiver — The range-tone receiver coherently processes those satellite transmissions which consist of a carrier with a single-sided adjacent spectrum of a subcarrier signal and several sequential range tones. The highest frequency tone is determined by the desired navigation accuracy, but the sequence begins at a tone with a wavelength long enough to ensure an unambiguous, although coarse, initial phase measurement. Other tones are intermediaries between these two. The use of a subcarrier results in a cancellation of post front-end delays in the receiver processing to be described.

The range-tone receiver processes the satellite range-tone transmissions so as to measure the propagation phase delay of the satellite signals by comparison with reference signals synchronized with the satellite clock. This phase delay is easily converted to range and is automatically done so within the receiver providing a digital range measurement as the final receiver output. The receiver consists of two basic phase-locked loops: the rf carrier loop and the range-tone loop as shown in Fig. 3-29. With the carrier loop

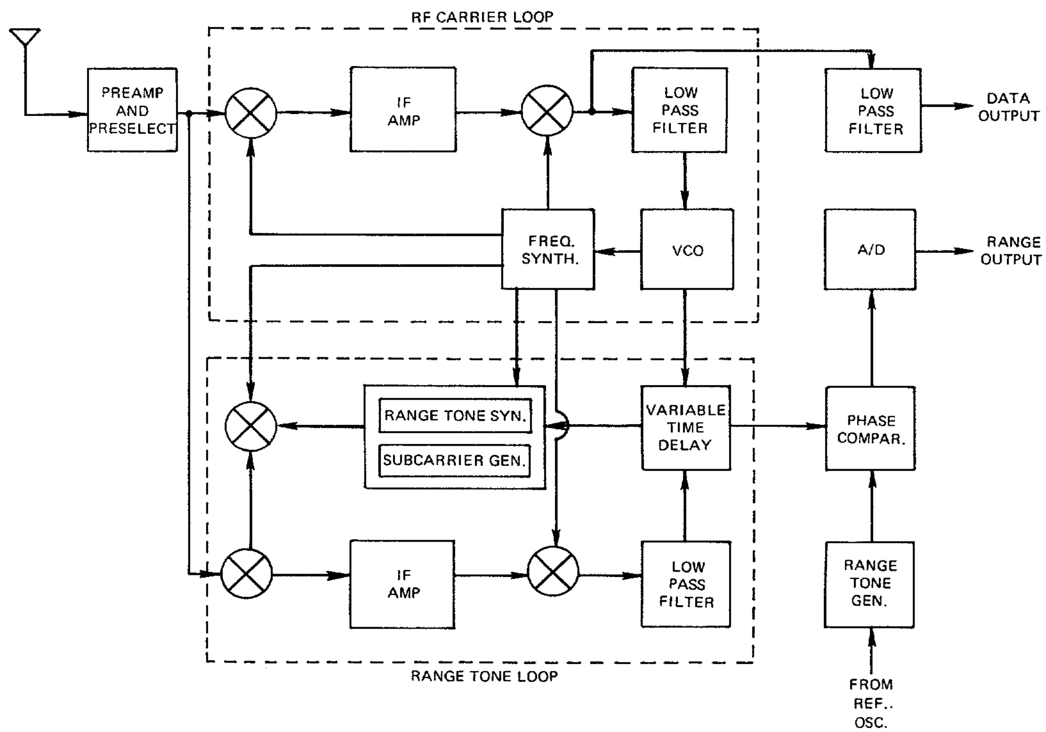


Fig. 3-29 - Range-tone receiver block diagram

locked, the voltage control oscillator (VCO) frequency reflects the receiver doppler shift percentage. Since the VCO is used to synthesize both the carrier frequency in the carrier loop and the subcarrier and range-tone frequencies in the range-tone loop, the carrier-subcarrier and carrier-range tone frequency ratios remain constant regardless of doppler shift.

The range-tone loop by means of the variable time delay makes the synthesized tone sequence phase-coherent with the incoming satellite tone sequence. Phase comparison of the synthesized tone sequence with a set of reference tones derived from the receiver clock yields the information of interest since the phase difference is a function of the transit time of the satellite transmission. Provided carrier lock is maintained, this range-tone receiver implementation allows ranging at any time, even though range tones are not continuously transmitted. Satellite data which are phase shift keyed onto the subcarrier are automatically available to the system computer in the output of the carrier loop phase detector.

3.5.3.2 KRP Receiver — A simplified block diagram of the KRP receiver is given in Fig. 3-30. The input signal to this receiver consists of the product of a ranging carrier and an ephemeris subcarrier and a KRP code. The receiver demodulates this signal by correlating (multiplying) it with a locally generated KRP reference signal as shown. The i-f output is the ranging carrier and the ephemeris data subcarrier. A phase-locked loop tracks the ranging carrier and a simple demodulator recovers the ephemeris data.

Range is measured by noting the delay between the local KRP signal and the transmitted KRP signal when maximum correlation (ranging tracking) is obtained.

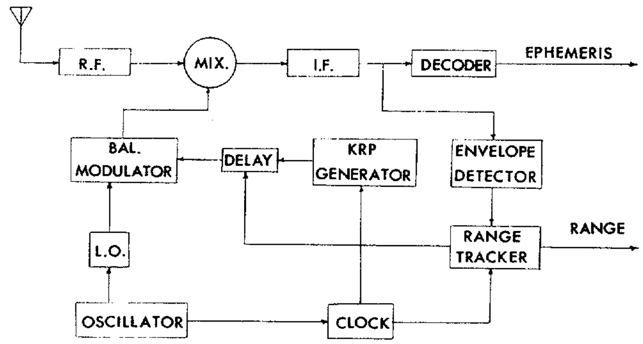


Fig. 3-30 - KRP receiver block diagram

Table 3-11
Timation-Related Computer Functions

Function	Category					
	1	2	3	4	5	6
1. Align and update INS* from timation data	Yes	Yes	No	No	No	No
2. Process data from other navigation aids, including INS	Yes	Yes	Yes	No	Yes	No
3. Cross calibration of other sensors	Yes	Yes	Yes	No	No†	No
4. Data editing	Yes	Yes	Yes	Yes	No	No
5. Detect jamming	Yes	Yes	Yes	Yes	No	No
6. Control receiver	Yes	Yes	Yes	Yes	Yes	No
7. Self-test	Yes	Yes	Yes	Yes	Yes	Yes
8. Interpolate ephemeris data‡	Yes	No	Yes	No	No	No

*INS inertial navigation system.
†More sophisticated equipment than envisaged here could be available automatically to perform this function also.
‡Only required by users needing fixes more often than the ephemeris update frequency (one every 2 minutes).

3.5.4 Data Processor

In general, the data processor consists of the computer for Timation-related functions and its interfaces with the display, the inertial navigation system where applicable, the Timation receiver, and the other sensors. The processor should be designed, where feasible, as a functionally integrated unit.

Specific processor functions for each user category are indicated in Table 3-11. The interface between the computer and the Timation receiver is considered as part of the processor. Generally, the conversion of the received navigation signal data to digital values of range is accomplished in the receiver. Also, both sampled and integrated range-rate values are available at the receiver output in all user equipment categories where fixes are required while the equipment is in motion.

The computer characteristics common to the various categories are listed below:

- Word length: 32 bits
- Memory type: scratch pad and nondestructive
- Hardware multiply and divide
- Full instruction repertoire
- Rugged
- Miniature (microminiature for manpack)
- Military specifications (except possibly for categories 5 and 6)
- Mean-time between failures approximately 6000 hours (military only)

The input/output characteristics depend on the data required for display and the particular interface requirements. However, all processors must be capable of accepting manually operated switchpanel-type inputs.

3.5.5 Display

The Timation system displays will be designed to present navigational data in forms most convenient to the navigator. Aircraft which will benefit from Timation and military vessels (categories 1, 2, and 5) will already possess displays for navigational information, and these displays are generally adequate for use with a Timation-aided navigation system. Other user categories in general do not possess navigation displays and an installation of Timation equipment must include the appropriate display.

In those cases where existing displays are utilized, the Timation system, in its normal mode of operation, is used as an accurate and periodic update for the user's inertial platform. It is also possible to directly couple the Timation computer output to the existing displays through appropriate interfaces. Normally, this scheme is a backup for inertial system failure and position information can be offered to the display continuously only if the Timation system computer is equipped to interpolate satellite ephemerides. This will be the case for computers of categories 1 and 3.

The F-4 fighter-bomber is a typical category 1 user. It possesses digital readouts, a vertical director indicator, and a horizontal situation display. In a Timation-only mode, these latter two devices, seen in Fig. 3-31, can be interfaced with the Timation navigation system using digital-to-analog signal converters between the computer and the displays.

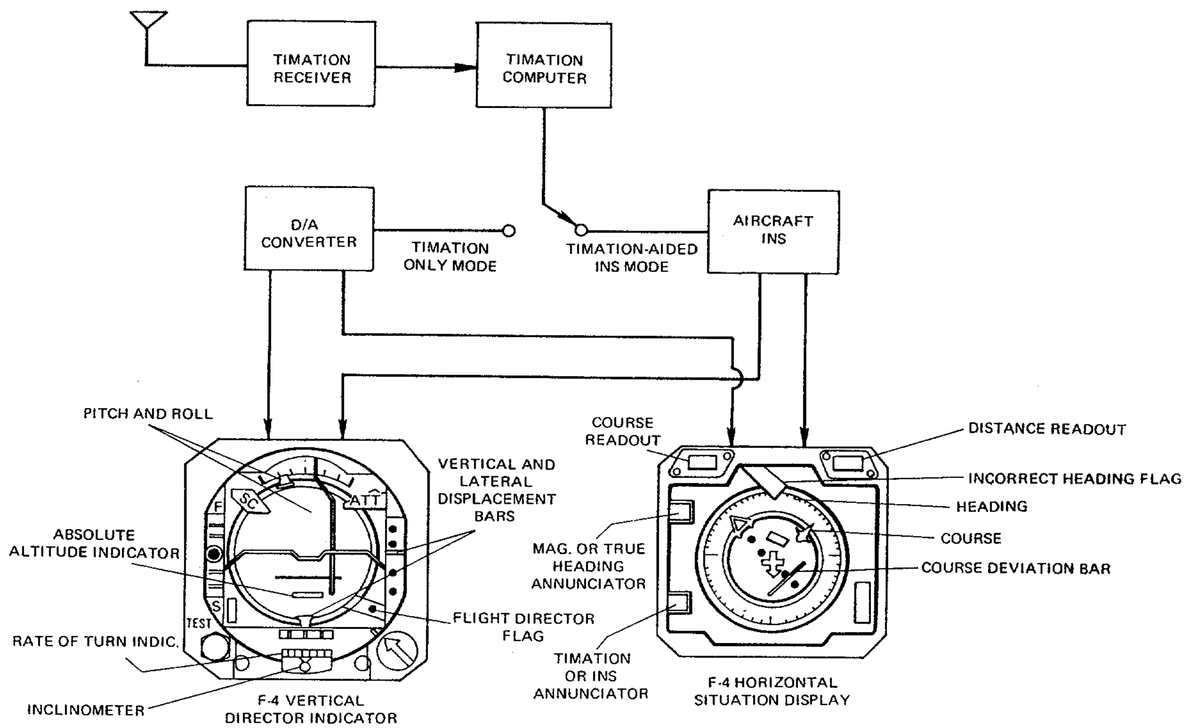


Fig. 3-31 - Display implementation for category 1 user

Table 3-12
User Category Display Hardware Requirements

Category	Display	Additional Display-Assoc. Hardware
1 - Combat aircraft	Existing	D/A Converter
2 - Marine operations	Existing	None
3 - Land operations (vehicle)	Timation digital	None
4 - Land operations (foot)	Timation Miniature digital	None
5 - General navigation	Existing	None
6 - General marine navigation	Timation digital	None

Category 4, the manpack user, will possess a small digital readout of latitude and longitude. The commercial aircraft of category 5 will use existing inertial navigation system displays with the inertial platform updated every 2 minutes by the Timation navigation data. Category 6 will possess a Timation digital display of latitude and longitude. Table 3-12 summarizes display equipment requirements by user category. All users will require a small control panel for Timation system test and initialization.

Section 4

TECHNICAL APPROACH AND TRADE-OFF CONSIDERATIONS

SUMMARY

The purpose of this section is to provide a detailed discussion of the Timation navigation satellite system and its components and to give a description of the tradeoffs that are possible. The topics are grouped under the following headings:

- System Design. This portion deals with the interfaces between major subsystems, such as the ground station network, the satellite constellation, communications links, and overall navigational performance.
- Spacecraft Design. This portion discusses the selection of a spacecraft configuration designed to function at the altitude selected and to meet the communications, attitude control, and power supply requirements of the overall system.
- Ground Station Design. This portion discusses requirements for satellite tracking, communications, data processing, and system control in the ground stations.
- User Equipment Design. This portion describes the design of the receiver subsystems which are common to all user equipments and considers the manner in which the accuracy requirements of the individual user categories are met. The equipment implementations designed to serve each category are described together with the options available.

Where the satellite altitude of 8000 naut mi is referenced, the specific altitude is 7496 naut mi.

4.1 SYSTEM DESIGN

The technical approach to an operational advanced navigation system was based on the following requirements and constraints imposed by the most demanding category of users:

- Three-dimensional instantaneous position fix capability must be globally available at all times.
- The fix capability must be supplied without any external aid, such as an altimeter or an inertial navigation system.
- The instantaneous fix accuracy must be better than 50 feet.
- The system must not require radio emission.
- Ground stations should be located on U.S. territory.

Note: The Summary was formerly denoted Subsection 4.0.

- Other users with lesser requirements and/or constraints must be able to obtain position fixes with equipment for their requirements.

- User equipment warm-up time must be compatible with mission operating requirements.

The selection of a navigation mode is fundamental to the system design. Four basic multiple satellite modes of navigation are listed in Table 4-1. Swept fan-beam systems and interferometer systems are eliminated from consideration because these do not meet the accuracy requirements.

Table 4-1
Passive Navigation Method Tradeoff Matrix

Tradeoff Factors	Range Only	Range Difference	Doppler Only	Range and Doppler
Required number of satellites in view for rapid fix	3	4	3	2
Precise clock required in user equipment?	Yes	No	Yes	Yes
Number of receiver channels	3	4	3	2
Velocity information potentially available	Yes	Yes	No	No
External inputs required	None	None	Velocity	Velocity

NOTE: Three-dimensional position fix requirement.

The doppler-only and range-doppler systems are also eliminated from further consideration due to their dependence on knowledge of user velocity for fix determination. The two remaining candidate modes, range-only and range-difference, are compared below.

The principal difference between user equipments and satellite constellations for the range-only and range-difference modes are:

- The range-difference mode requires one more satellite in view than is required for range-only navigation.

- The range-difference mode requires a less stable clock than does the range-only mode.

- The range-only mode (with a better user clock) offers improved navigation over range-difference navigation.

For a given orbital configuration (e.g., circular polar orbits at a fixed altitude), the number of satellites required for a range-difference system is about 1/3 greater than for range-only navigation because the former requires four satellites in view of any user (for three-dimensional navigation) while the latter requires three in view. The constellation design is somewhat more difficult for the range-difference system because of the greater effect of the relative geometrical arrangement of the satellites on the fix accuracy. Detailed studies have shown that the 1/3 increase in number of satellites is close to the correct value and will result in an equivalent increase in satellite costs. However, since the satellites can be launched on the same number of boosters of the same type for the two systems, the launch cost does not increase significantly. Three boosters are sufficient to launch a complete operational system for either range-only or range-difference navigation modes. Nonrecurring system support costs will be somewhat higher for a larger constellation because more antennas and receiver/transmitter channels will be required at each tracking station. However, the number of ground stations is unchanged, and the staffing and data processing requirements are only marginally affected by the increase in number of satellites.

Studies have indicated that, for all satellite constellations considered, if the range-difference requirements were met, the range-only requirements would also be met. Since the increase in system complexity and cost are not excessive and the range-difference mode allows a less sophisticated user equipment to be employed if desired, it was decided to configure the system to satisfy the requirements of range-difference navigation.

The requirement for worldwide navigational coverage combined with the preference for locating the system ground stations within U.S. territory and possessions has led to the selection of a 27-satellite constellation with altitudes in the range of 8000 naut mi to 20,000 naut mi and an inclination of the orbit planes in the range of 55 to 90 degrees. In the interests of economy it was decided to launch the satellite complement (9) of each orbit plane simultaneously as a single booster payload. The launch vehicle which is most cost-effective for this purpose is the Titan IIID combined with the Burner II. This vehicle has a payload of 3700 lb when launched into an 8000-naut mi, 90-degree inclined orbit from the Western Test Range and 5100 lb when launched into an 8000-naut mi, 55-degree orbit from the Eastern Test Range. These payloads allow a maximum per satellite weight of 400 lb and 550 lb, respectively.

The present spacecraft weight estimate is 452 lb, which, together with supporting trusses and separation devices, suggests a final payload weight exceeding 4500 lb but less than 5000 lb. These considerations indicate an 8000-naut mi, 55-degree inclined orbit with launches from the Eastern Test Range.

The effect of altitude on satellite radiation environment has been considered. Estimates of the number of trapped high-energy protons and electrons show that the numbers are appreciably reduced as altitude increases from 8,000 to 12,000 naut mi. However, at altitudes above 6,000 naut mi the effects of solar flare positions become significant.

The spacecraft solar array is the component most affected by particle bombardment and its power output must be sufficient to offset the degradation caused by a lifetime of exposure to the radiation environment. In a tradeoff of spacecraft weight versus altitude it was found that the reduction achieved by increasing the spacecraft altitude from 8,000 naut mi to 12,000 naut mi was more than offset by the loss of booster payload capability.

4.1.1 Constellation Selection

The selection of the range-difference navigation mode requires that the constellation used provide a minimum of four satellites in view, in good geometrical configuration at all times, and from all points of the earth's surface.

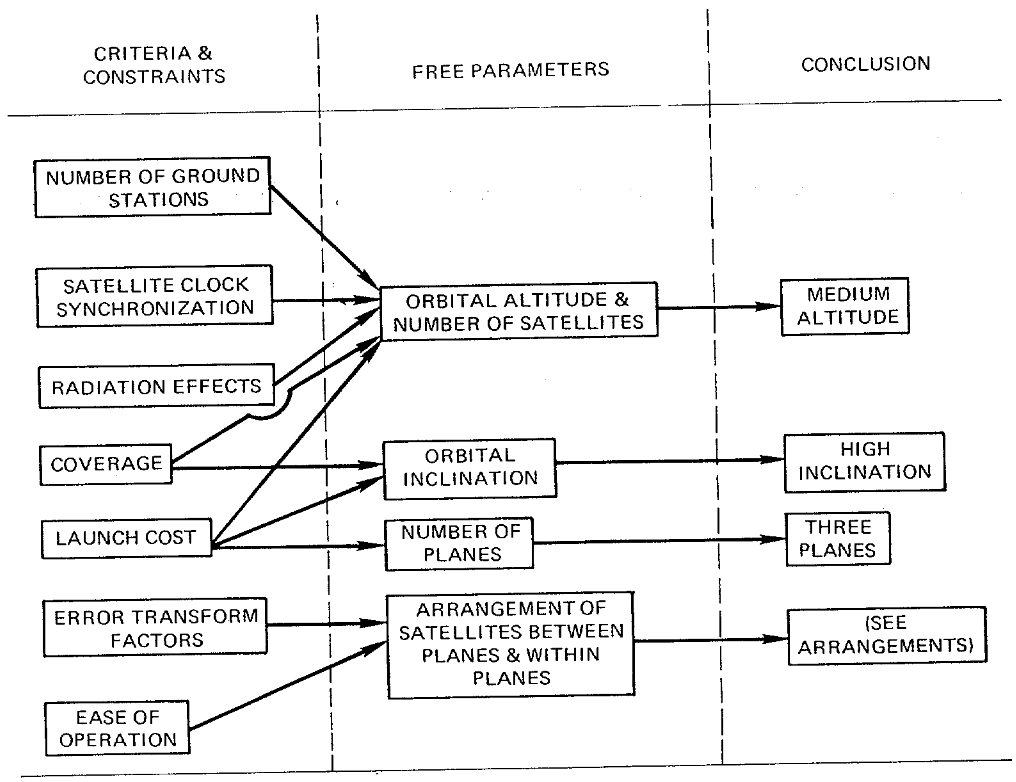
The criteria which enters into constellation selection are:

- The effect of nonobservation time on the accuracy of synchronization of system clocks,
- the accuracy of prediction of ephemerides,
- the quality of geometrical arrangement provided,
- the interaction with ground station selection, and
- the ease of system operation.

The principal parameters which may be varied are the number of satellites, the orbital altitude, the orbital inclination, and the orbital eccentricity.

The interactions between these criteria and the parameters are shown in Table 4-2, and the constellations are examined in detail in Table 4-3.

Table 4-2
Factors in Constellation Selection



One of the main parameters that can be varied is the mean orbital altitude. Clearly, coverage per satellite increases with altitude; therefore, the required number of satellites

Table 4-3
Constellations Examined

No. of Planes	Orbital Period (Hours)	Orbital Inclination (degrees)	Number of Satellites Per Plane							
			5	6	7	8	9	10	11	12
3	4	90					X			
3	5	90					X			
3	6	53								X
3	6	90					X			
3	8	40					X			
3	8	45					X			
3	8	50		X	X	X	X	X		
3	8	53			X	X	X	X	X	X
3	8	55					X			
3	8	60		X	X	X	X	X		
3	8	70		X	X	X				
3	8	80		X	X	X				
3	8	90		X	X	X	X	X	X	X
3+1	8	90(3)+0(1)		X	X					
4+1	8	90(4)+0(1)	X							
4	8	90	X	X	X					
3	12	40					X			
3	12	45					X			
3	12	53				X	X			
3	12	90				X	X			
3	24	53			X	X	X			
3	24	90			X					

decreases with altitude. However, it is more difficult to orbit satellites at higher altitudes. Another problem that arises at higher altitudes is an increase in the period of time during which the satellites are not visible from any of the ground stations. These gaps in ground station contact cause degradation in satellite clock synchronization.

The effect of altitude on coverage is shown in Fig. 4-1. Note that increasing altitude from the 8-hour orbit to the 12-hour orbit achieves the same increase in coverage radius as the increase from the 12-hour to synchronous altitude. Further, increase of elevation mask from 5 degrees to 10 degrees requires an increase of satellite altitude from the 8- to the 12-hour altitude, or from the 12- to the 24-hour altitude, for equivalent coverage. Higher-altitude satellites move more slowly and therefore take longer to cross areas where they are not in contact with any ground station. During this time, errors in synchronism of the satellite clock with system time tend to increase on the average as shown in Fig. 4-2. Therefore, to economize in the number of ground stations and to avoid the need to provide atomic time standards in the satellites, high-altitude constellations (synchronous and near-synchronous) are not favored. Furthermore, synchronous systems require ground stations located outside U.S. territory. The coverage and synchronization criteria taken together favors medium-altitude satellites, i.e., between about 6- and 12-hour orbits.

Figure 4-3 illustrates the effect of altitude (and inclination) for a specific booster.

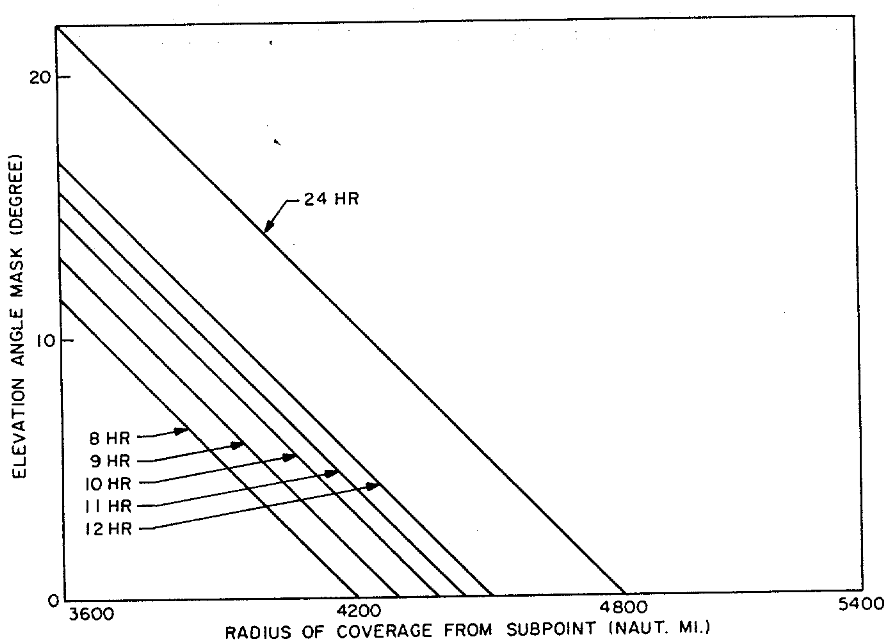


Fig. 4-1 - Coverage radius for various elevation angles and orbit periods

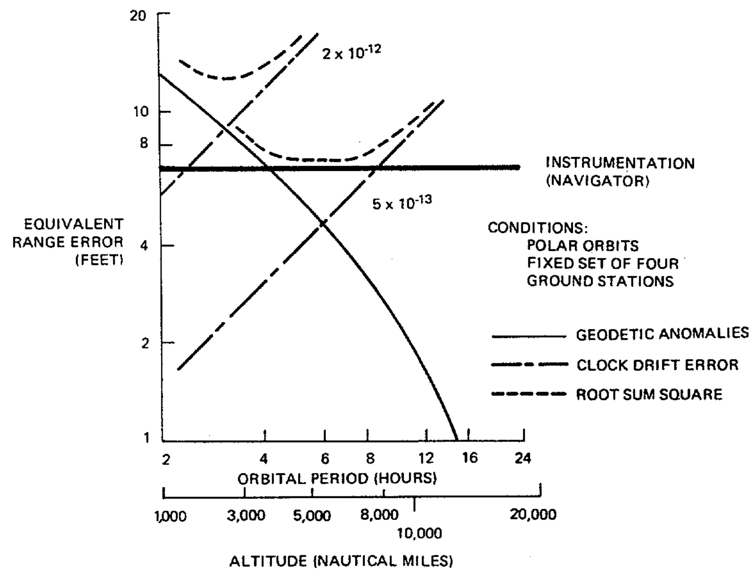


Fig. 4-2 - Effects of various sources of error on range measurement accuracy

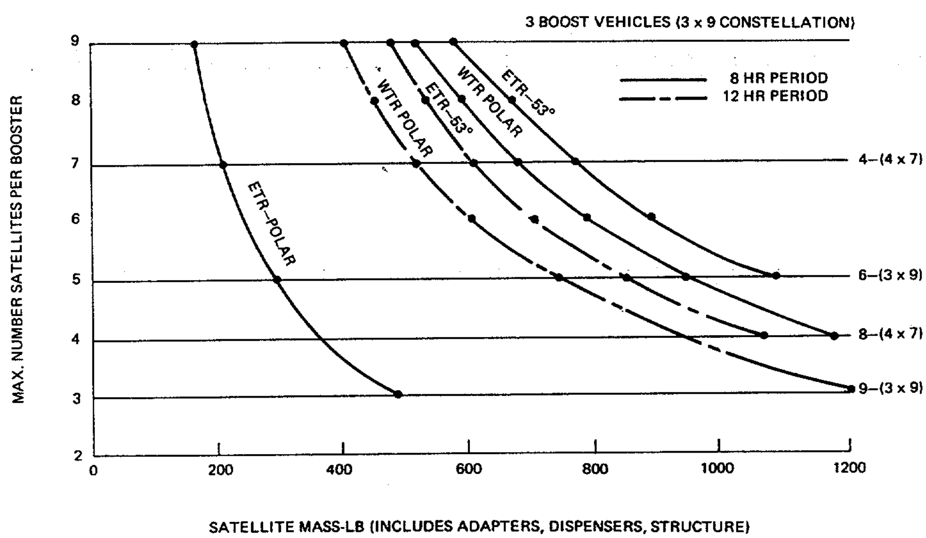


Fig. 4-3 - Boosters required for various 8- and 12-hour orbit constellations (Titan IIID/Burner II-2300)

Finally, the tradeoff resolves itself into the problem of finding a match between the boosters available and the number of satellites required to provide the coverage (with adequate geometrical quality). Cost is minimized by observing the following ground rules:

- use the smallest possible number of orbit planes,
- use medium-altitude orbits,
- avoid significant "dog-legs" in the launch trajectory, and
- launch eastward.

Applying these ground rules, the minimum number of orbit planes is three. In 8-hour orbits, nine satellites per plane provide four-satellite coverage with good geometry. Satellites can be launched directly into orbits with a 53-degree inclination from ETR. The dog-leg to obtain 55 degrees costs less than 100 feet/second. With present estimates of satellite mass (less than 500 lb) the available booster (Titan IIID/Burner II) can achieve a 55-degree inclination or higher when loaded with nine satellites.

A computer program was used to generate contour maps of error transform factors for each constellation listed in Table 4-3. Several constellations were found to yield satisfactory transform factors continuously over the globe. However, the 3-by-9 constellation at a 55-degree inclination (see Fig. 4-4) meets all the requirements and constraints and is cost-effective.

4.1.2 Satellite Orbit Determination and Ephemeris Prediction

This subsection discusses the orbit determination system tradeoffs and the ephemeris prediction tradeoffs. The rationale for the selected system is given in each case.

The orbit determination subsystem must meet the system accuracy budget. Orbit determination accuracy, in general, depends on

- the number and distribution of the tracking sites,
- the accuracy to which the coordinates of the tracking sites are known,
- the accuracy to which the external forces (e.g., gravitational, radiational, aerodynamical, control jet, outgassing) can be determined,
- the quantity and quality of the tracking data, and
- the accuracy of the corrections made for tropospheric and ionospheric refraction effects.

Careful attention has been given to each of the above factors.

During the past 10 years, a number of different organizations have expended considerable effort to develop highly accurate orbit determination programs. The most outstanding of these organizations is the Naval Weapons Laboratory. As a result, there

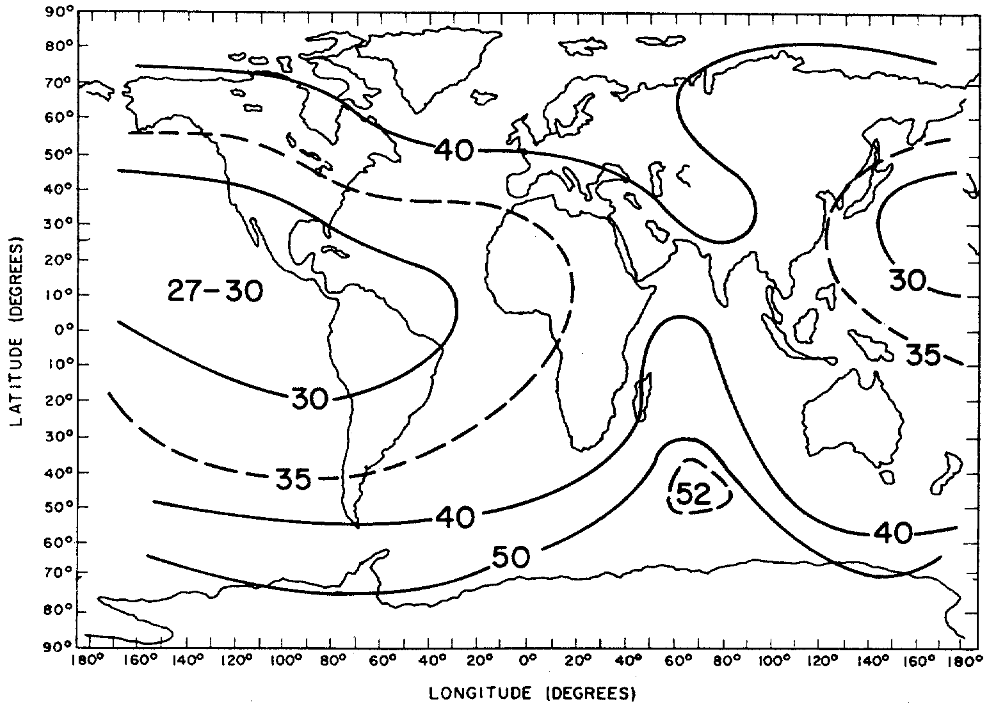


Fig. 4-4 - Typical position fix errors for 3 x 9 inclined orbit constellation using four ground stations

exists today a highly accurate orbital determination program which can meet the needs of Timation. From the known characteristics of the program, it is possible to conclude that availability of a sufficiently accurate orbit determination program will not be a problem.

Recent and still-continuing efforts to determine the coordinates of tracking sites are achieving accuracies of a high order. This ensures that accuracy of the tracking site coordinates will not be a source of difficulty for orbit determination. In addition, the number and distribution of the four sites selected for Timation have been carefully considered during the selection process.

The altitude of each Timation satellite implies that the only external forces of significance, at least for intervals between tracking data, will be the geopotential and luni-solar gravitational forces.

The selection of the tracking method is important for orbit determination. It is not a question of the type of tracking data provided by the different methods but rather the quality and quantity which are of greatest importance. The need for highly accurate tracking data rules out the radar skintrack method. One- and two-way doppler tracking methods offer significant advantages in potential accuracy, cost, and simplification of ground-based equipment. However, in spite of the fact that one-way doppler has been successfully used in the present Navy navigation system, its potential for successful utilization in the Timation system appears doubtful because of the reduced doppler-frequency shift. The accuracy of two-way range, together with its relatively modest demands in complexity of both satellite and ground equipment, and the lack of any major disadvantages make its selection clear for this mission. Both the quantity and quality of the tracking data from a transponder-aided, two-way range tracking system should suffice to meet the basic orbit determination accuracy requirement of 5 feet.

Experience already gained on geodetic and navigational satellites indicates that the removal from the Timation tracking data of tropospheric and ionospheric refraction effects is feasible. Ionospheric errors are minimized by using correction factors from the navigation signals. Similarly, experience gained to date indicates that the performance of the spacecraft and ground-station equipment should allow attainment of the required orbit determination accuracy.

Following the initial orbit determination, the continuous updating of the orbit will be accomplished by an accurate projection of the satellite ephemeris and the transfer of this predicted data to the satellite. Associated with this requirement is an important tradeoff consideration, i.e., the form of the predicted ephemeris data transmitted to the satellite. It must be decided whether (a) to transmit and store in memory all of the data to be broadcast by the satellite to users or (b) to transmit a compacted form of such data and expand it onboard the satellite before broadcasting it.

The first alternative mentioned has the advantage of minimizing the complexity of the ephemeris-associated function which must be performed onboard the satellite. The second alternative has the advantage of requiring less data storage in the satellite memory and sharply reducing the communication capacity needed of the command link. Because of the great compressibility of Timation ephemeris data and the existence of simple but highly accurate interpolation algorithms, the advantage of the second alternative substantially outweighs that of the first. On this basis the onboard expansion approach has been selected.

4.1.3 Modulation Techniques

Signal design and modulation studies have been performed to ensure effective and accurate range measurement, reliable command and data transmission, and satisfactory telemetry reception. The objective has been to maintain the required link performance and to limit the complexity of the transmitting, receiving, and signal processing equipment. It is particularly important that the user equipment remain cost-effective within the constraints imposed by its environments.

A ranging signal is modulated on both a 400-MHz carrier and a 1600-MHz carrier. A study was made to determine the best ranging waveform and type of modulation that meet the requirements of the Timation navigational system. Ranging waveforms that were considered include sequentially transmitted fixed tones, simultaneously transmitted fixed tones, and a KRP digital code. Navigation systems employing these ranging waveforms were compared as to range measurement accuracy, acquisition time, bandwidth requirement, satellite equipment complexity, and user equipment complexity.

In a fixed-tone ranging system, range is measured by comparing the phases of the demodulated tones received from the satellite with those of corresponding tones derived from the user clock, which are in phase with the tone oscillator in the satellite. Ambiguity in the range measurement is resolved by measuring the phases of eight lower tones. Only the four highest frequency range tones need to be sent on the 1600-MHz link.

In the sequentially transmitted, fixed-tone ranging system, one range tone is transmitted at a time from the satellite. On the 400-MHz navigation link, each tone is transmitted as a single side tone to the 400-MHz carrier. Because the tones are not transmitted simultaneously, modulation loss due to dividing the signal power among the range tones is avoided. The acquisition time for the range tones and the subcarrier is the sum of the individual acquisition times. On the 1600-MHz link, only a subcarrier and the four range tones are transmitted. Satellite ephemeris data are transmitted by phase-shift keying the ranging subcarrier during the time that range tones are not being transmitted.

The only satellite equipment needed to generate the ranging waveform is a frequency synthesizer that switches its output frequency each time a new range tone is transmitted. The user receiver is also quite simple, consisting of amplitude and phase demodulators. Satellite transmissions are displaced from each other in frequency or in time to prevent interference between transmissions of satellites simultaneously in the line of sight of a user. A user, therefore, only has to change his frequency or the timing of his receiver when it acquires a new satellite.

In the simultaneously transmitted, fixed-tone ranging system, the range tones are all phase modulated simultaneously onto the respective carriers—nine tones on the 400-MHz carrier and four tones on the 1600-MHz carrier. The simultaneous transmission reduces the power per range tone as compared to the sequentially transmitted fixed-tone system, but the range code acquisition time is equal to the longest tone acquisition time instead of the sum of the tone acquisition time. Because all of the tones must be processed simultaneously, the user receiver is more complex than the sequential fixed-tone receiver. However, the satellite transmitter, the frequency and time multiplexing scheme, the transmission of data, and the frequency plan are essentially the same as those described for the sequential fixed-tone system.

In the KRP digital code ranging system, KRP-noise codes are phase modulated onto the 400-MHz and 1600-MHz carriers in the satellite. When the codes are acquired, the received codes are compared with locally generated codes. The length of the codes determines the ambiguity resolution possible, and the bit rate determines the range accuracy. A bit rate of 7.5 megabits/second is used for both codes. Code lengths of 93,310 bits for the code transmitted on the 400-MHz link and of 70 bits for the codes transmitted on the 1600-MHz link are used. Data are transmitted by changing the phase of the KRP code at 150 times/second. Mutual interference between satellites is avoided by transmitting the code at different bit rates.

The information presented in the previous paragraphs is summarized in Table 4-4.

The KRP ranging system is complex. The sequential system is preferable because of the simpler user receiver and the greater system margin on the 400-MHz link.

Table 4-5 reviews the requirements for a jam-resistant, satellite-to-user link and describes briefly a concept for meeting these requirements.

The following assumptions were established to define a set of evaluation criteria for the antijam concept.

1. The antijam link must impose a significant burden on the enemy attempting to jam it during an engagement.

2. The concept employed for jam protection must be readily improvable as technology improves with a minimum impact on the user equipment investment.

It is further desired to keep the cost of the user equipment and satellites as low as possible while fulfilling these requirements.

A fundamental question of whether or not a separate jam-resistant link is required should be addressed. The side-tone ranging signal described for the nonjam resistant link is vulnerable in some degree to cw jamming. Since some level of dependence must be placed on the availability of the system to make it a portion of a weapon system complex, a greater amount of jam resistance is implied. Therefore, a more jam-resistant link has been investigated.

The fundamental technology used in the jam-resistant link concepts is a keyed-random-phase spectrum-spreading technique. The benefit of this technique is that cw jamming techniques have poor effectivity against the receiving equipment. The measure of the ability to reject this type of jamming is often referred to as processing gain; the larger the processing gain, the greater the tolerance to a given level of jamming for a fixed-link radiated power.

The implementation concept employs a continuous transmission of the spread-spectrum waveform which is received in a coherent receiver. Bandwidth compression is accomplished by synchronous demodulation governed by a low bandwidth tracking loop. Range measurement is accomplished by alignment of locally generated codes with received codes from the satellite. Processing gain is determined by the ratio of spread spectrum width to

Table 4-4
Ranging Waveform Summary

Ranging Waveform	Sequential Fixed Tones		Simultaneous Fixed Tones		Swept Tone		KRP Code		BINOR Code	
Carrier Frequency (MHz)	400	1600	400	1600	400	1600	400	1600	400	1600
Modulation	SSB	SSB/SC	PM	PM	SSB	SSB/SC	PM	PM	PM	PM
Acquisition Time (Seconds) (at 10° Elevation Angle)	3	3	3	3	3	3	0.91	0.99	1.40	1.94
RMS Range Error (Feet) (at 30° Elevation Angle)	4	4	4	4	4	4	4	4	4	4
Satellite RF Power (dBW)	4.8	10	4.8	10	4.8	10	4.8	10	4.8	10
System Noise Temperature (°K)	692	654	692	654	692	654	692	654	692	654
System Operating Margin (dB)	8.2	4.9	7.1	8.0	0.4	-6.4	0.3	3.4	8.0m	12.1
Frequency Spectrum	Discrete frequencies in 3.7-MHz band		Discrete frequencies in 3.75-MHz band		Carrier plus 570-kHz band		15-MHz band		15-MHz band	
Satellite Equipment Complexity	Simple		Simple		Simple		Complex		Simple	
User Equipment Complexity	Simplest		Less Simple		Complex		Most Complex		Simple	

Table 4-5
Jammer Characteristics Summary

Jammer Type	Jammer Characteristics Coherent Continuous Wave Concept		Jammer Characteristics Noncoherent Pulsed Concept		Relative Jammer Effectivity for Continuous Wave vs Pulsed Concepts, Given Equal Jammer Complexities
	Complexity	Effectivity	Complexity	Effectivity	
Spoofer	Highest	Moderate	Highest	Moderate	Higher
Continuous Noise	Low	Good	Low	Good	Higher
Pulsed Noise	Moderate	Poor	Moderate	Fair	Lower
CW	Lowest	Poor	Lowest	Poor	Higher

tracking loop width which, for practical systems, can be on the order of 50 to 60 decibels. The length of the code used in the KRP modulator is then selected to deny effectiveness to a spoofer-type jammer, with the general observation that the longer the codes, the less effective a spoofer jammer will be. A tradeoff to the use of long codes is the length of time to acquire the signal track in the presence of jamming. The accuracy of the tracking during jamming will depend on the duration of tracking loop integration for a fixed level of jamming and received useful power. A practical limit of loop bandwidth is approximately 10 Hz (as bound by user acceleration) and provides an integration time of approximately 100 milliseconds. Data received beyond this time interval are useful in (a) maintaining track once locked-on and (b) allowing the user to make a range measurement at will.

In the Timation navigation satellite system, the following links for tracking, telemetry, and command functions are required:

- Transmission of spacecraft housekeeping data, attitude telemetry data, and command verification to the ground stations.
- Transmission of commands and ephemeris data from the ground stations to the satellite.
- A link for data transmission between ground stations.
- A link for satellite tracking by two-way range measurement.

Frequencies in the S-band range were chosen because of the availability of off-the-shelf equipment designed to provide S-band tracking, telemetry, and command capabilities. Operation of the link at C band was considered in view of the virtual elimination of ionospheric refraction error at this frequency. This option was rejected because of the additional power requirement when compared to the equivalent S-band link and the availability of ionospheric refraction correction data from the navigation links. The frequencies adopted were 1800 MHz for the uplink and 2225 MHz for the downlink.

The telemetry information is modulated onto a 1.24-MHz subcarrier that is in turn phase modulated on the 2225-MHz carrier. The PCM/PSK modulation scheme was chosen for the telemetry link because this method requires less satellite power for a particular probability of bit error than with pulse amplitude modulation. A bit rate of 256 bits/second is employed in the data transmission. To preserve the command code security, the command verification signal indicates only if a command is received and if it is received correctly.

The commands from the ground stations to the satellites are modulated onto the command subcarrier, which is in turn phase modulated on the 1800-MHz carrier. The command format consists of a synchronizing word, address word, command word including parity bit, the complement of the command word, and a spacing word. A command link bit rate of 256 bits/second was chosen. With this bit rate, the satellite computer receives one ephemeris data point out of 15 and computes the remainder. The computer also stores commands from the ground and causes a series of commands to be performed upon the receipt of an execute command from the ground.

Data to be sent from one ground station to another through the satellite is PSK modulated on a subcarrier, which is in turn phase modulated on the 1800-MHz uplink carrier. The subcarrier is demodulated from the 1800-MHz carrier, amplified, and phase modulated on the downlink carrier in the satellite with no further processing. The data are transmitted at 256 bits/second, a bit rate that requires some processing of satellite tracking data at the ground stations before the data are transmitted to the central ground station. The data link will also be used to maintain synchronization of the ground station clocks.

4.1.4 System Error Analysis

The navigation system was examined to determine the various sources of error and to compare these against the system requirements. The following are the navigational requirements:

- Ranging will employ a 400-MHz signal and a 1600-MHz ranging signal for compensation of ionospheric reduction;
- Initial range ambiguity of 400-MHz ranging is to be 2000 naut mi;
- Initial range ambiguity of 1600-MHz ranging is to be 1 naut mi;
- Rms range resolution is to be 5 feet or less.

Error budgets were determined (over a range of satellite elevation angles) for the case of a high-accuracy user employing a satellite for ranging. The variation of rms phase-measurement error with satellite elevation angle is shown in Fig. 4-5; the results indicate that angles from 5 to 90 degrees can be used. A typical error budget for a high-accuracy user ranging from a satellite at an elevation angle of 30 degrees is presented in Table 4-6. The errors are given in nanoseconds, where each nanosecond error causes a range error of 1 foot. Briefly, the error sources are:

1. Spacecraft Clock Error - The error is caused by changes in the frequency of the satellite frequency standard. The magnitude of the error depends on the rate of frequency drift, the amount of time since the last correction of the clock by a ground station command, and the amount of error in the correction.

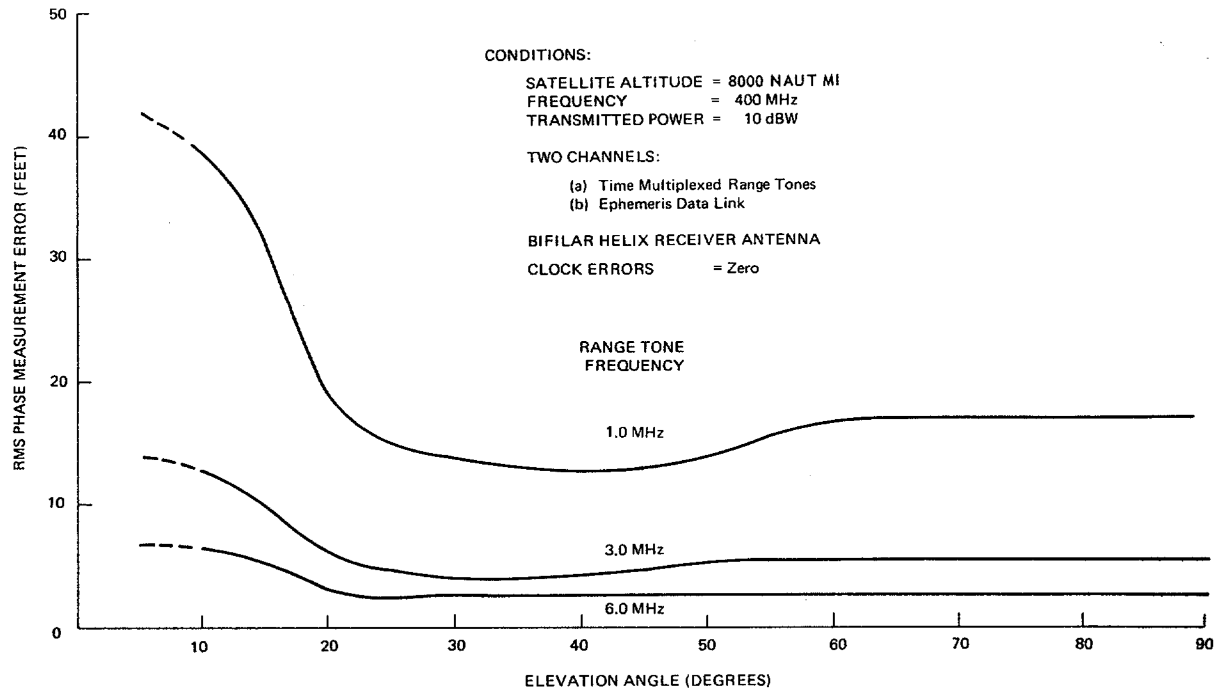


Fig. 4-5 - Range error due to phase measurement error

2. Ephemeris Measurement and Prediction - This error depends on the error in characterizing the satellite orbit and the error in measuring the satellite range at the controlling ground stations.

3. Phase Measurement - This error arises from phase-detector uncertainty and corruption of the ranging signals by thermal noise and multipath distortion. Phase measurement error varies primarily with satellite elevation angle.

4. Equipment Delay Uncertainty - Certain delays in the user receiver cannot be calibrated, resulting in a phase error in the signals compared in range measurement.

5. Ionospheric Effects - The refraction of rays passing through the ionosphere from the satellite to the user contributes to the range error. These errors are largely compensated by the use of two ranging frequencies.

6. Tropospheric Effects - Refraction of signals in the atmosphere causes a range error depending on the altitude of the user and the local weather conditions.

4.2 SATELLITE DESIGN

4.2.1 Introduction and Summary

This section of the development plan contains the tradeoff and performance studies which were carried out to develop the specific requirements and capabilities of the Timation system baseline satellite (Section 3.3).

Table 4-6
Error Budget for High-Accuracy Users

Multipath	2.5 nsec or feet
Ionosphere	2.0 nsec or feet
Troposphere	2.0 nsec or feet
Equipment	5.0 nsec or feet
Satellite position	3.0 nsec or feet
Clock synchronization	5.0 nsec or feet
RSS	8.5 nsec or feet (value used = 10 nsec + clock error)

The systems studies previously reported provide fundamental requirements for the preliminary design of the operational satellite. The conclusions having the major impact are:

- The preferred configuration of the system space segment is a constellation of 27 satellites, nine satellites in each of three 8000-nautical mile, 55-degree inclined, posigrade orbit planes, the right ascension of the ascending node of each plane being separated by 120 degrees.

- The cost-effectiveness considerations which led to the selection of this constellation are based in part on the availability of the Titan IIID/Burner II booster, which can place nine satellites into the selected orbit plane at one time.

- Each satellite is required to provide ranging and ephemeris data at 400 MHz and 1600 MHz, to permit the derivation of ionospheric refraction and error-correction data and to provide operational capability in an ECM environment.

- The satellite tracking is accomplished by an S-band ranging transponder onboard the satellite.

The altitude and inclination of the constellation orbits determined the satellite orbital environment for analyses of (a) radiation degradation effects, (b) sun angle range, (c) attitude and velocity disturbance effects from which the station-keeping requirements are derived, and (d) the effective radiated power (ERP) required for navigation, tracking, telemetry, and command. The attitude, velocity, and ERP requirements were particularly significant in determining the electrical load, the type of solar array, the power subsystem configuration, the choice of antennas, and the attitude control requirements. The conclusion was reached that earth-coverage antennas for the basic navigation, tracking, and telemetry functions conbined with a body-mounted solar array formed the basis for a satisfactory satellite configuration.

The selection of the Titan IIID/Burner II booster for launching nine satellites at a time imposed constraints of fairing size, i.e., satellite stacking arrangement, initial orbit correction requirements, and satellite dispersal. A bulbous (Viking) fairing was specified to provide the maximum diameter for accommodating nine satellites. The stacking requirement affects the shape of the satellite and the structural support design.

The frequencies specified for the navigation links determined the design of the navigation antennas and the choice of the modulation techniques. The S-band transponder included for satellite tracking can also demodulate spacecraft commands on the uplink and modulate telemetry on the downlink.

Studies of the Timation satellite internal components indicate that all satellite functions can be met by state-of-the-art techniques and hardware. In many instances, several alternate approaches are feasible, and many tradeoffs can be made. Some of the more significant tradeoffs are reported herein.

4.2.2 Operations Plan

A description is given of the operations plan covering the time-line sequence of events from lift-off to the final mission mode configuration of the Timation constellation of satellites. A preliminary plan covering the five stages of boost, deployment, dispersal, station acquisition, and station maintenance keeping has been developed which:

- Identifies the critical launch vehicle/spacecraft interface and support requirements and
- Identifies the satellite maneuver requirements.

Two different launch vehicles and launch profiles are required for the demonstration (two satellites) and the operational (nine satellites) configuration, but the same Burner stages can be used.

Boost phase events for the operational mode are shown in Table 4-7. The Titan has its trajectory shaped for a ballistic coast to an 8000-nautical-mile apogee at a 50-degree inclination for this ETR launch at a 44-degree azimuth. Firing of the Burner II is guided to result in a fully circularized 8000-nautical-mile orbit inclined 55 degrees to the equator.

The deployment method for the operational configuration is shown schematically in Fig. 4-6. The method is similar for the demonstration configuration with the exception that two spacecraft are deployed instead of nine.

Table 4-7
Boost-Phase Events for the Operational Mode

Sequence	Event
1	Titan IIID launches at 44-degree launch azimuth from ETR
2	Strap on Solid Rocket motors cutoff and separate
3	1st booster engine cutoff
4	1st stage separation
5	2nd stage ignition
6	Nose fairing jettison
7	2nd stage engine cutoff
8	Titan IIID/Burner II separation - injecting Burner II and payload in an elliptical orbit with apogee at the design altitude, 50-degree inclination
9	Attitude maneuver for Burner II firing at apogee
10	Burner II fires at apogee bringing Burner II and satellite payloads into design circular orbit and inclination
11	Attitude maneuver to orient spacecraft momentum wheel axis into the orbit plane subject to the constraint that the sun angle be between 0 and 90 degrees.
12	Satellite 1 momentum wheel spin-up, Burner II satellite 1 separation
13	Retrothrust of Burner II, H_2O_2 system to clear satellite 1
14	Repeat 11 and 13 for satellites 2 through 9*

*Satellites 1, 2, 3, and 4 are separated near the North Pole, 5 near the equator, and 6, 7, 8, and 9 near the South Pole. The Burner II must maintain its initial inertial orientation at separation of spacecraft 1 until the separation of the last spacecraft. This orientation will be in the orbit plane perpendicular to the radius vector at the North Pole.

To assure satisfactory solar exposure to the body-mounted solar array, the spacecraft sun angle (the angle between the sun vector and the pitch axis) must be between 0 and 90 degrees. During deployment, this constraint requires maneuvering of the Burner II and payloads prior to separation as shown in Fig. 4-6.

The dispersal method involves changing the satellite orbit periods selectively by optimal firing of onboard thrusters. All of the satellites of a given constellation are clustered together in essentially the same orbit immediately following deployment. To adjust the nine satellites within their common orbit plane to their proper position, two of them must be phase shifted by ±160 degrees, two by ±120 degrees, two by ±80 degrees, two by ±40 degrees, and one unadjusted. The velocity correction increment, and hence propellant requirement, is a function of the waiting time and phase shift, as shown in Fig. 4-7. Assuming a twenty-day total adjustment time, a 72-feet/second velocity correction increment is required for the maximum phase adjustment of 160 degrees.

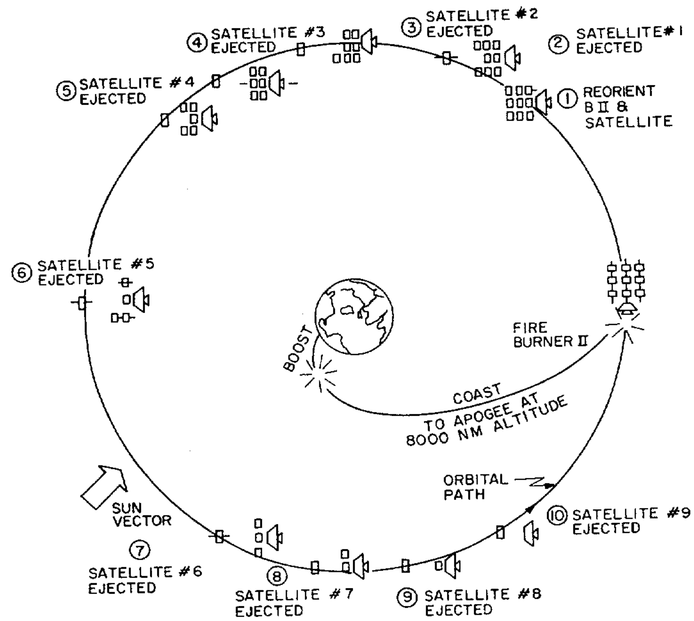


Fig. 4-6 - Timation satellite system deployment method for operational configuration

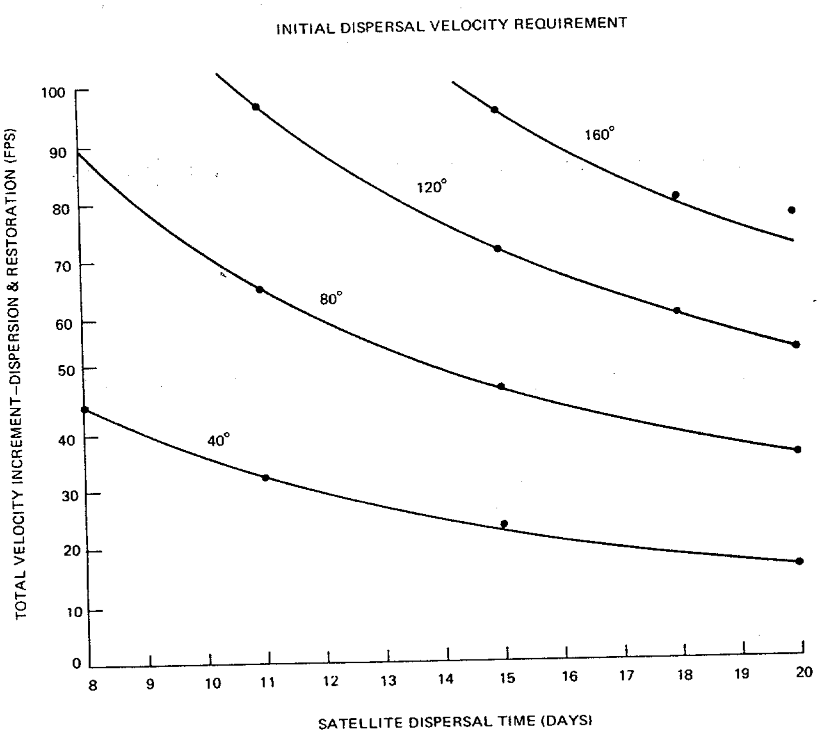


Fig. 4-7 - Timation satellite system initial dispersal velocity requirement

The satellite axial thrusters, i.e., thrusters with a force vector parallel to the spin axis, are used for dispersal. By spinning the satellite during the maneuvers, thrust errors arising from misalignments are minimized. The actual firing will occur when the spin axis is nearly perpendicular to the radius vector so that the change in velocity of the satellite will have the maximum effect on the orbit period. Axial thrusters are provided at both ends of the spacecraft to permit maneuverability, and to assure that the spacecraft sun angle of from 0 to 90 degrees is maintained during these maneuvers.

Inclination and circularity launch errors are corrected by thrusting at apogee and/or perigee during satellite dispersal. To correct orbital-inclination errors, the appropriate axial thruster is fired at the nodal crossing, with the spacecraft maneuvered such that the spin axis is parallel to the orbit normal. The maximum 3σ orbit-injection inclination error is expected to be less than 0.25-degree. This error requires a velocity correction of 63.5 feet/second.

Station acquisition is completed when the spacecraft is oriented with its spin axis parallel to the orbit normal, the main body is despun to a rate of one revolution per orbit, and with the antenna axis is locked on the earth. Additional periodic trimming of the orbit will be performed using both the axial and radial thrusters.

Two station-keeping functions are required, i.e., attitude control and position control.

Periodic correction of the satellite spin-axis orientation is required to adjust for the effects of solar pressure and residual magnetism. In addition, the satellite spin axis must be reversed every 6 months by 180 degrees, in a single maneuver, to maintain sun angles of from 0 to 90 degrees. All of these spacecraft attitude maneuvers, including those required during station acquisition, are accomplished by electromagnets onboard the spacecraft.

Infrequent position control is planned for eccentricity and orbit-period (spacing) adjustments. The major orbital perturbations are inclination changes from lunar and solar attraction and nodal regression of nearly 0.1 degree/day westward due to the earth's oblateness. Since these perturbations equally affect all satellites in the constellation, minimal correction for these disturbances is anticipated. Additional station keeping is expected periodically, to correct the phase changes within the constellation resulting from imperfect orbit corrections and orbit disturbances.

Satellite replacement may be accomplished using the Titan IIIB/Burner IIA Boost vehicle for a two-satellite launch. One spacecraft is positioned in the failed spacecraft slot and the other is placed in a standby position.

4.2.3 Structure

The factors most influential in determining the Timation spacecraft configuration and structural design are the orbit, the navigation, tracking, telemetry, and command requirements, the power subsystem design, the attitude-control subsystem requirements, and the multiple-launch and booster-fairing constraints.

The 55-degree-inclined, 8000-nautical-mile orbit has a direct impact on the ERP requirements and on the power subsystem requirements, related to solar array degradation and operational sun angles. The angle between the orbit plane and the sun vector changes cyclically each year between maximum and minimum limits of ±78.5 degrees. The solar array is designed to accommodate this sun-angle range. Antennas with earth coverage (40-degree beamwidth) were selected to provide a sufficient margin of received power. A tradeoff between a configuration utilizing an earth-oriented antenna and one with an equivalent toroid antenna or omniantenna showed that the former provided a weight advantage of between 50 and 100 pounds over the toroid and omniantenna configurations, respectively. The selection of an earth-oriented antenna on this basis then led to the requirement for accurate earth pointing.

Two other tradeoff areas were considered before selecting the preferred configuration. The first area was concerned with the choice of an attitude-control subsystem. A completely despun mainbody using a momentum-biased flywheel for the basic spin stabilization versus a spinning mainbody with a mechanically despun antenna developed as the primary candidates. The biased flywheel approach was selected on the basis of mechanical simplicity and maximum flexibility.

The second tradeoff related to the solar-array configuration; fixed paddles, oriented paddles, or a body-mounted solar array were the main options. At the power levels required for this mission, and considering the long lifetime requirements, the body-mounted cells provided the best approach. This choice was further reinforced by the requirements for stacking, stowage, and deployment.

Stacking of the nine satellites into the Titan IIID/Burner II booster is also required. The largest (400-MHz) antenna, approximately 9 inches in diameter by 50 inches long, must be folded during launch and then deployed in such a manner as to avoid shadowing the solar array. The preferred stacking arrangement, shown in Section 3.3, requires each satellite to possess a separation mechanism for the satellite attached to it, and a mechanism for its own release. Also, the structure of each satellite must carry the loads of the satellite(s) above it. The bulbous (Viking) Titan fairing was selected to maximize the available stacking envelope. The final requirement to be satisfied by the satellite is the accommodation of the propulsion system for velocity correction and control.

The preferred preliminary satellite configuration resulting from the tradeoffs discussed above is basically a closed cylinder, as previously shown in Fig. 3-12. The axis of the cylinder is oriented along the orbit normal. The sides and one end of the cylinder (hat) form the substrate for the solar cells. The other end of the cylinder (baseplate) provides the primary structural mounting surface for the satellite components and the hat.

Three vertical posts, located 120 degrees apart, have been provided for satellite stacking. These posts extend from the baseplate to the top of the array and provide separation devices for the upper and lower satellites. A truss located at the top of the posts is used to support the momentum-wheel assembly and add ridigity to the vertical members.

The solar array is a hollow cyclinder with cells on the sides and at one end of the array. The hat is lowered over the structural skeleton, bolted to the baseplate at its periphery, and attached to the upper end of the three vertical posts. The main structural components are shown in Fig. 4-8. Arrays of this configuration have been fabricated from riveted sheet metal (aluminum) or bonded aluminum/plastic honeycomb. Of the three, aluminum honeycomb offers the lightest substrate design.

The three vertical posts are the primary load-carrying members. Signals from the baseplate, hat, truss, and stacked-satellite loads are all transmitted through the posts to the spacecraft/booster interface structure. Shear loads resulting from lateral and torsional inputs are carried by the hat substrate. Each post provides mounting for the

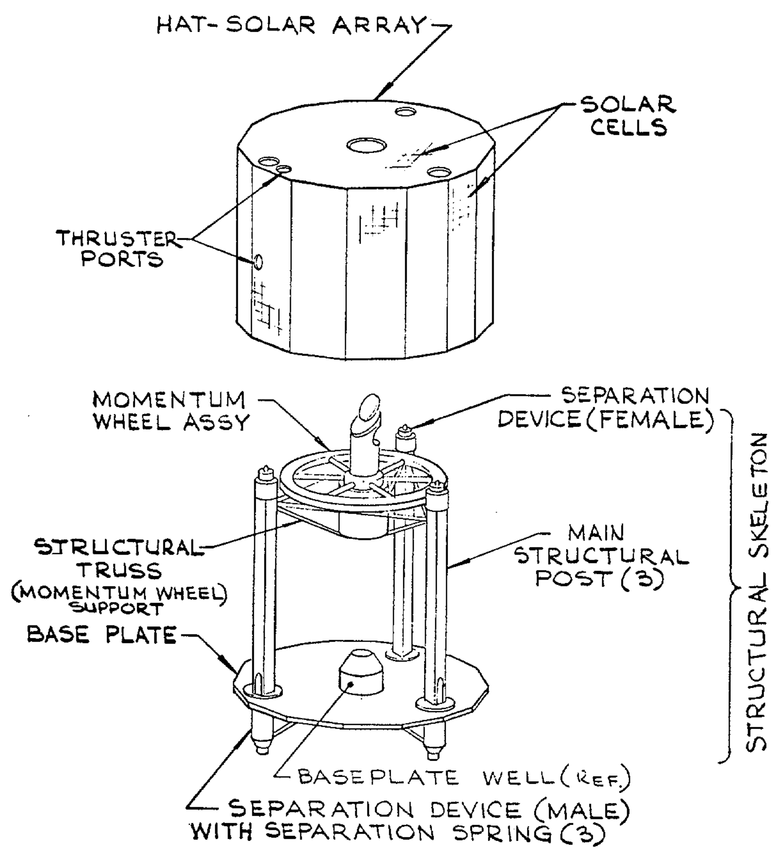


Fig. 4-8 - Timation satellite (main structural components)

male and female half of the separation devices, with the female half mounted on the base-plate end of the post. These "ball-pin" devices, shown in Fig. 4-9, contain integrated separation springs and are operated pneumatically in orbit. The posts will be fabricated from titanium tubing.

4.2.4 Navigation Subsystem

The navigation subsystem is the key element of the Timation satellite design. All other subsystems will support this primary function of the satellite.

Many of the key tradeoffs relating to the specification of requirements for the navigation subsystem have been derived earlier in the systems analysis section. For completeness, the major requirements are summarized here.

- Ranging signals will be modulated onto a 400-MHz carrier and a 1600-MHz carrier to determine ionospheric delay effects.

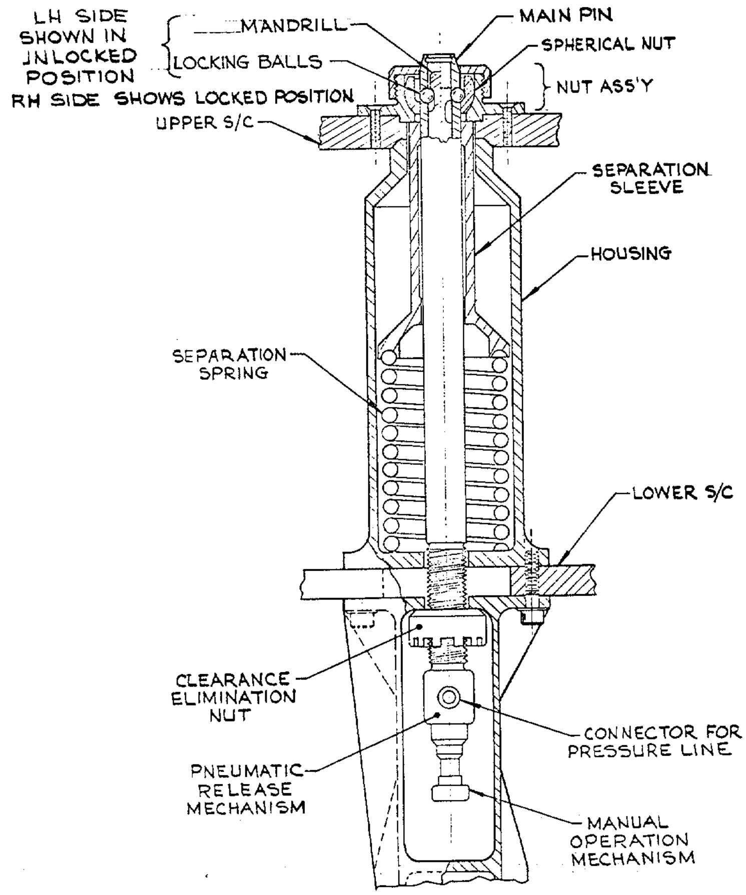


Fig. 4-9 - "Ball-pin" separation device

- Ephemeris data transmitted on the 400-MHz link will be approximately 400 bits every 30 seconds, with a bit error rate no greater than 10^{-6}. Data will be transmitted immediately after range-tone transmission. The unmodulated carrier is available for tracking.

- A sequential fixed-tone modulation technique is used to provide maximum user equipment simplicity.

- Nine range tones and a subcarrier are modulated on the 400-MHz carrier with the lowest tone, RT-1, less than 81 Hz and the highest tone, RT-9, greater than 3.75 MHz; only the four highest tones and the subcarrier are modulated on the 1600-MHz carrier.

- Satellites in the same orbit plane will use the same set of frequencies because of the separation of each group of range tones into time slots. Three sets of frequencies are required for the three different orbit planes.

- A spacecraft clock will be provided that has a stability of 2 parts in 10^{12}. The clock will be capable of correction for accumulated error on command from a ground tracking station.

Design studies show that the link requirements will be met by a 10-watt, 400-MHz carrier, operating continuously with 10 watts in the sideband tones at a 25-percent duty cycle and by a 10-watt, 1600-MHz carrier at a 7-percent duty cycle.

The oscillator and frequency synthesizer are shown in Fig. 4-10. The oscillator frequency is set at (4/3) (RT - 9) or 4.893 MHz. It is divided by 4096 and multiplied by 16N to give the desired carrier frequency (f_c). The factor N is either 20899, 20900, or 20901 depending on which orbit the spacecraft occupies. The range tones and subcarrier are synthesized as shown.

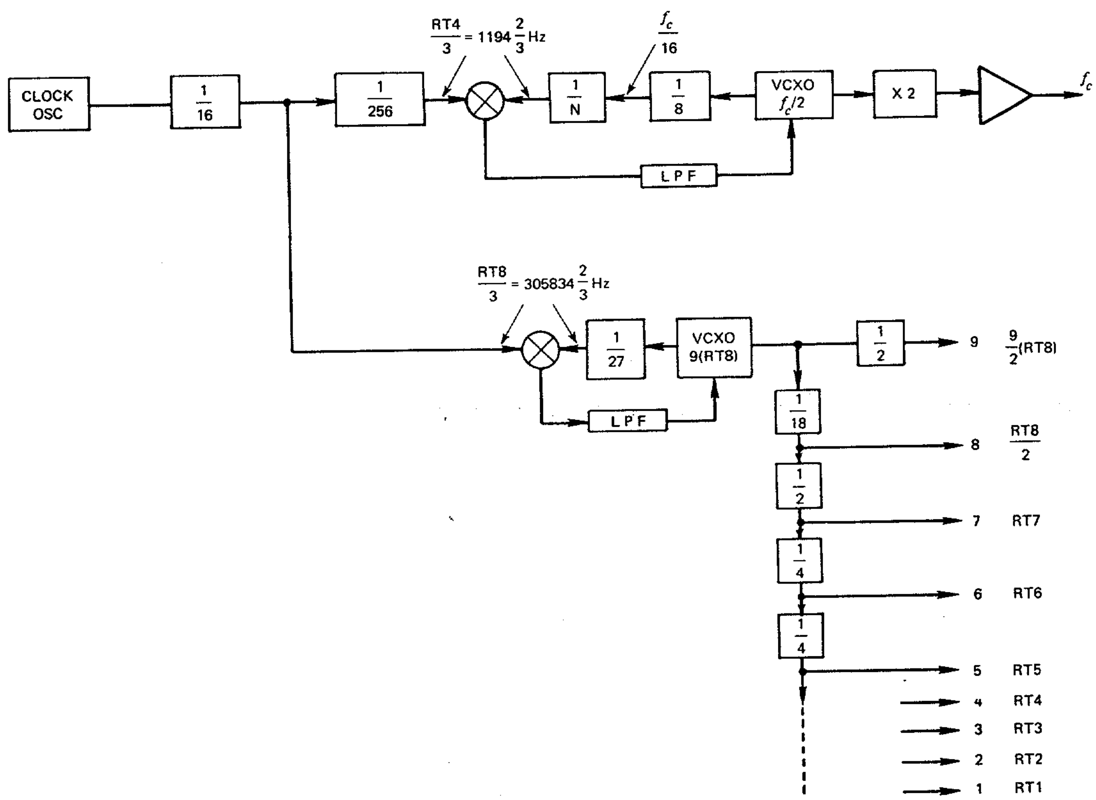


Fig. 4-10 - Clock oscillator and frequency synthesizer, functional schematic diagram

The oscillator has high stability characteristics which approach those of an atomic standard. The quartz crystal-controlled oscillator is the spacecraft model in which special measures are taken to obtain maximum long-term stability. Such measures include a triple proportional oven, ultraclean crystals operated at the zero-temperature coefficient point, and effective regulation and filtering of input voltages to eliminate ac and dc transients. Such models have been tested at NRL. Figure 4-11 shows a schematic of the two transmitters, antennas, and data modulator. A switching network driven by the command subsystem determines which of the nine ranging tones is transmitted. A single-sideband modulator modulates each range tone onto the 400-MHz and 1600-MHz carriers. The 400-MHz carrier is amplified, combined with the amplified sideband, and radiated from the appropriate antenna. The 1600-MHz range-tone sideband is radiated from its own antenna.

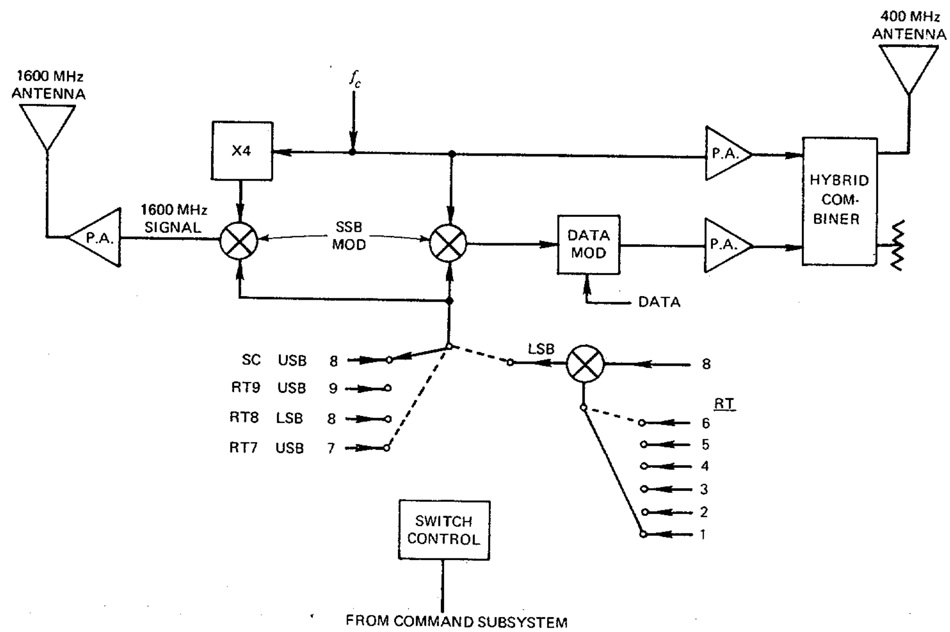


Fig. 4-11 - Navigation transmitters

The 400-MHz navigation link requires a narrowband earth-oriented antenna with circular polarization. For the required 40-degree beamwidth, antenna length may be traded off for antenna cross section. In the case of the 400-MHz unit with its large wavelength and physical dimensions, the best approach is to utilize a design which derives its gain from antenna length because such a form-factor lends itself to easier deployment in orbit. The preferred design is a helix with a cup reflector; the use of a cup reflector reduces backlobes below levels available from the equivalent plane reflector. Dimensions for this unit are shown in Fig. 4-12. The antenna is stowed along the earth-facing panel and then deployed when earth orientation is achieved. The helix and reflector will be copper bonded to a fiberglass cylinder and a fiberglass cup. Both cylinder and cup are foam filled.

The 1600-MHz link likewise requires an earth-oriented antenna with circular polarization. The approach which best combines shallow depth and reasonable cross section with a 40-degree beamwidth is the cavity-backed helix. As shown in Fig. 4-13, the dimensions are 14 inches in diameter by 9 inches in depth. The antenna is flush mounted on the earth-oriented panel. The helix and cavity are aluminum with a fiberglass

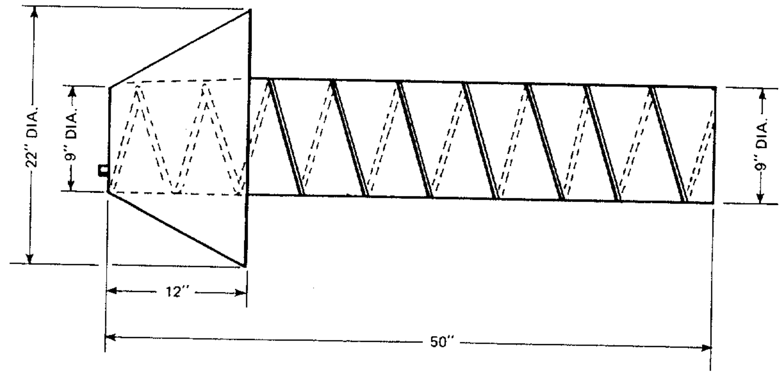


Fig. 4-12 - 400-MHz navigation links (antenna design)

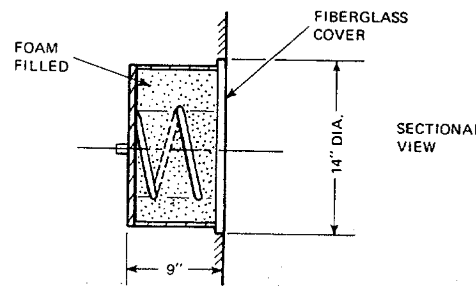


Fig. 4-13 - 1600-MHz navigation links (antenna design)

cover at the aperture. A polyfoam type of material would fill the cavity. The location is at the center of the spacecraft to avoid pattern nonsymmetries due to wall currents.

For improved antijam performance the satellite will radiate the same KRP code signals, but the signals will be at sufficiently distinct bit rates such that of those in view, no two satellites will transmit at the same code rate. The nominal rate is 10 megabits/sec. Both satellite transmitters will radiate 10 watts. Figure 4-14 shows the block diagram of a satellite KRP L-band transmitter. The output of the multiplier chain at 1600 MHz is (phase) modulated by +90 degrees by the KRP digital signal, by a doubly balanced ring modulator. The assigned bit rate is determined by a bit rate synthesizer, as shown in Fig. 4-14. A 250-Hz output is provided to drive a biphase data modulator and then is combined with KRP using an exclusive "OR" gate. Each data transition is simultaneous with a KRP transition.

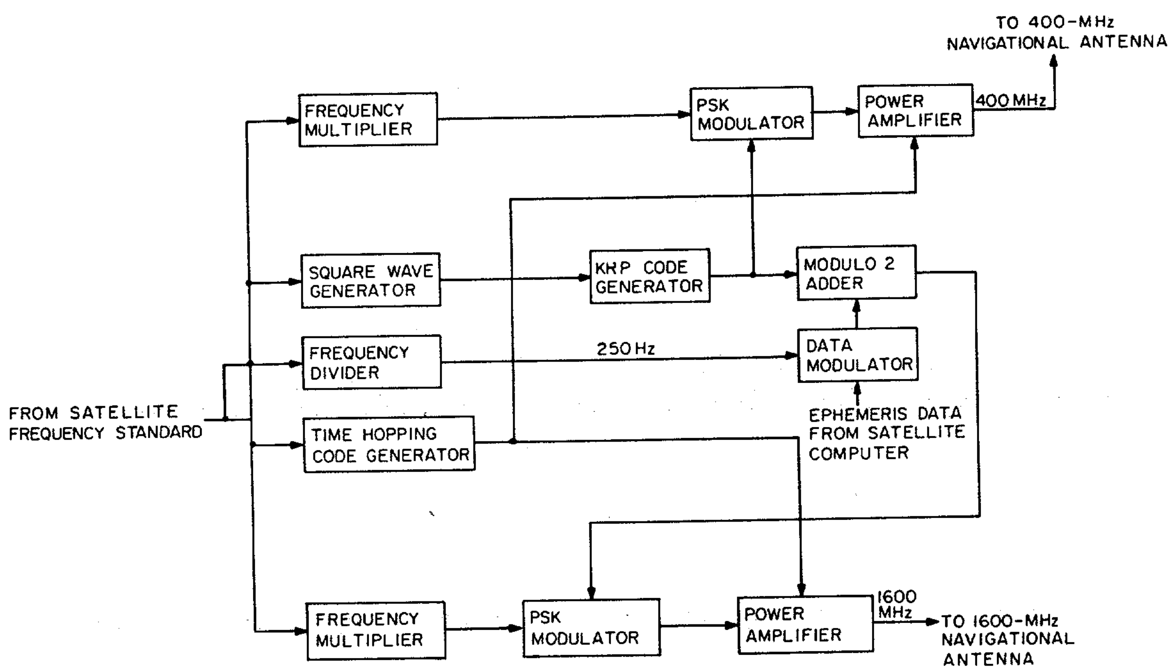


Fig. 4-14 - Satellite KRP L-band transmitter

4.2.5 Tracking, Telemetry, and Command Subsystem

The tracking, telemetry, and command receiver portion of the spacecraft is required to perform the following functions:

- Transmit housekeeping telemetry and satellite attitude data to the ground stations;
- Receive commands and ephemeris data from the ground stations;
- Provide a data relay between ground stations;
- Receive and retransmit tracking signals from ground stations to permit measurement of spacecraft range and range rate;
- Provide means for synchronizing ground-station clocks and spacecraft clocks.

To allow ground tracking stations to measure range and range rate, the satellite carries a coherent turn-around transponder.

Two major decisions have been made for the overall Timation system, the specific rf frequency, and the type of modulation for the tracking function. Selection of C-band frequencies or higher for the transponder has the advantages that the ionospheric correction would be quite small, and spectrum allocations would be more readily available than with lower frequency bands. However, use of the L or S band leads to other advantages. Existing tracking systems use these bands, and equipment development problems are eased because flight-proven equipments are presently available. Also, the use of lower frequencies leads to more efficient transmitters with lower-transmission line losses, thereby reducing the dc power requirements. Correction can be made for the ionospheric effects on range measurement at S band or lower by making measurements with the 400-MHz and 1600-MHz navigation links. Therefore, S-band tracking capability is recommended.

Time and frequency error measurements will be performed by observing the satellite clock as received at a ground station and correcting for the time delay due to the distance between the satellite and ground station. This will use the tracking transponder for high-precision measurement. Corrections to the satellite clock setting are sent via the command link. A functional block diagram of the satellite transponder is shown in Fig. 4-15. Figure 4-16 shows the additional circuitry needed to determine the transponder delay in orbit. Such a device is now being developed by NASA and will be available for use in the desired timeframe.

The command antenna pattern is omni-directional about the satellite spin axis. Since launch constraints prohibit the use of a boom, the only approach is a "flush-mounted" concept. Bandwidth is a severe requirement because this antenna is used to receive at 1800 MHz and transmit at 2250 MHz.

The satellite will transmit the following types of telemetry to ground stations: housekeeping data, command verification, and attitude-sensing data. At a nominal frequency of 2250 MHz and a bit rate of 256 bits/sec, the telemetry link is designed to carry the following signals:

- Pulse code modulation (PCM) signals representing analog housekeeping functions measured in the satellite, such as temperatures and voltages;

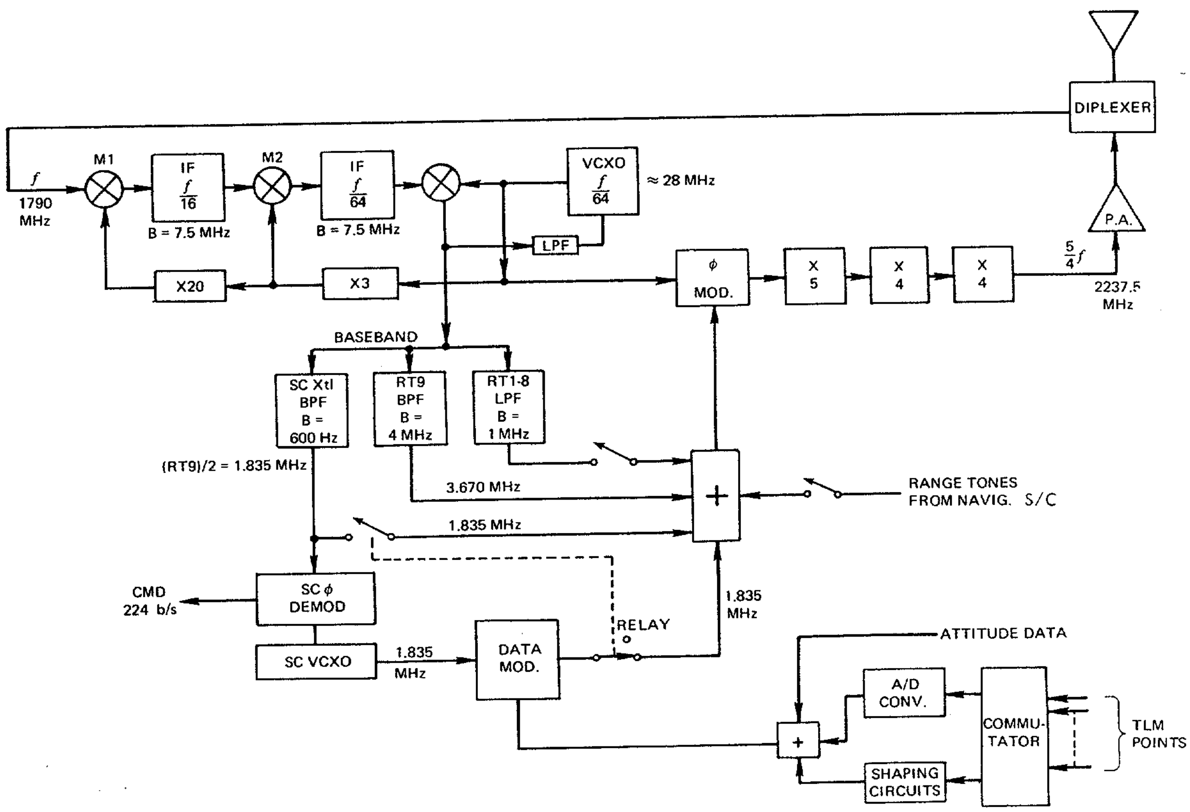


Fig. 4-15 - Satellite transponder

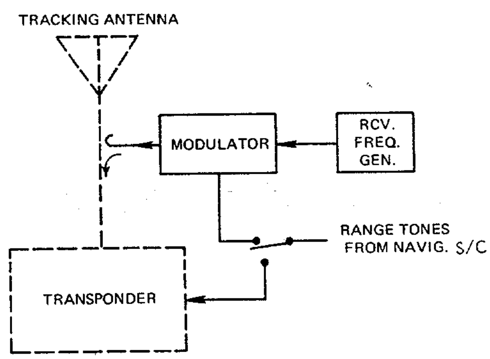


Fig. 4-16 - Satellite transponder (delay calibration circuitry)

- Digital on/off indications of satellite electronic equipment status;
- PCM signals containing the satellite attitude sensing data;
- Coded digital words indicating receipt of a certain command or asking for repeat of a command;
- The downlink tracking signal; and
- The downlink interstation data signal.

Figure 4-17 is a block diagram of the satellite equipment used in the telemetry link. Redundant equipment is not shown.

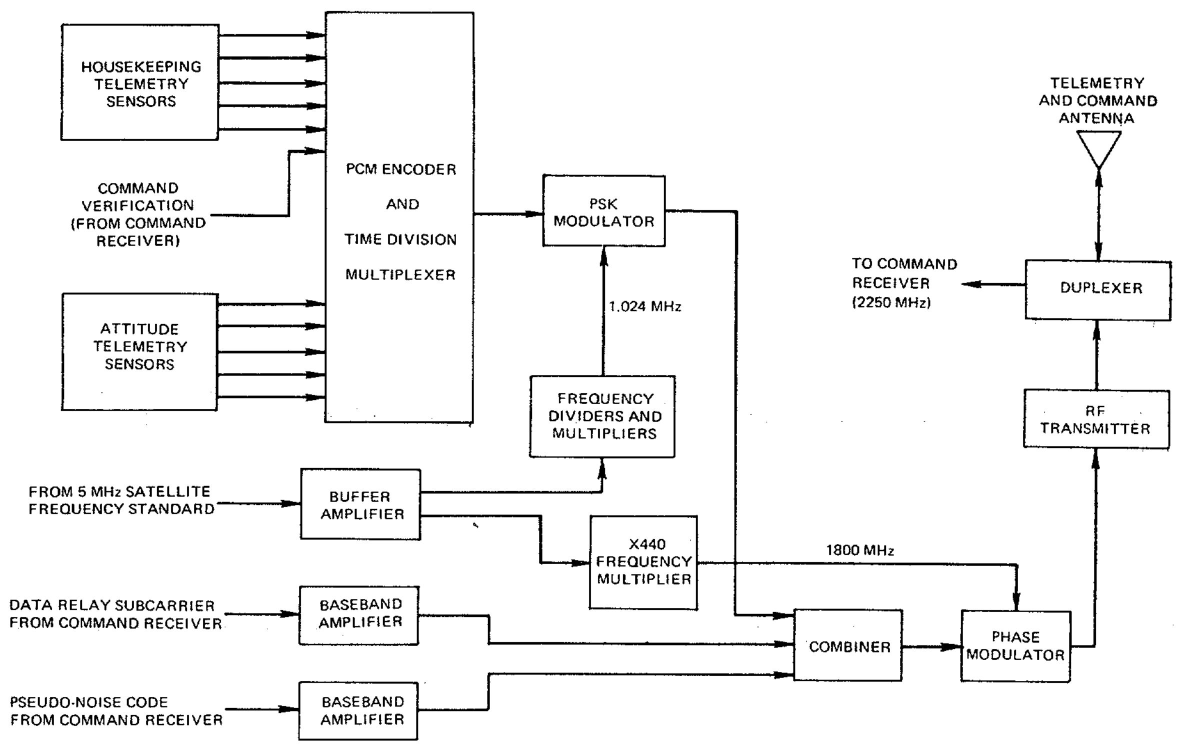


Fig. 4-17 - Timation satellite (telemetry encoder and transmitter)

Each housekeeping telemetry sensor in the satellite puts an analog voltage into the PCM encoder. This voltage is converted to an 8-bit digital signal, which is time-division multiplexed with the other housekeeping telemetry data and command reception indication. A maximum of 115 analog telemetry sensors will provide the ground stations with the status of satellite equipment.

To maintain the security of the command link, the satellite indicates, with one of two telemetry bits, whether a command was or was not recieved. If the satellite computer detects an error in the received command, the second bit indicates to the ground station that it should repeat the command.

The satellite data-relay for use between ground stations will provide the following functions:

- Relay of satellite telemetry and attitude data to CGS;
- Relay of commands;
- Transmission of operating schedules (e.g., satellite contact times) to telemetry, tracking, and command (TT&C) stations;
- Transmission of system status information;
- Communication of data necessary for synchronization of clocks; and
- Transmission of tracking data from TT&C stations to the Central Ground Station (CGS).

The command subsystem must provide real-time and remote satellite operations and storage of ephemeris data for transmission on the navigation link. The major elements are the command antenna, the command receiver, the command demodulator, the clock, and a computer, consisting of a central processing unit, memory, and an input/output unit.

Discrete commands for switching redundant equipment, turning equipment on or off, or correcting the frequency and phase of the satellite oscillator are processed immediately. Remote commands, such as the initiation of time-sequenced operations (e.g., some of the propulsion and attitude-control maneuvers) are stored in the memory for later actuation. Ephemeris data are received over the command link and stored in the memory. The command uplink is also used with the satellite transponder for satellite tracking, clock calibration, and data transmission from one ground station to another.

The design of the command equipment and the parameters of the command link are subject to antijam and security requirements. The provision of antijam capability requires a spread-spectrum receiver in the satellite. Link security is achieved through the use of an encryptor on the ground and a decryptor in the satellite.

Figure 4-18 is the basic block diagram of the satellite command receiver; the demodulator is shown in Fig. 4-19. Decoding devices for spread-spectrum signals and a decryptor must be added to the diagram subsystem for antijamming and security operation. Redundant equipment is not shown.

Although an ultrastable oscillator is available for navigation functions, a separate lower stability clock will be used during data-loading operations. This approach was selected to minimize the number of possible loading errors.

The satellite computer will accomplish the following data processing tasks:

- Store ephemeris data transmitted from the ground stations;
- Interpolate additional ephemeris data points between those received for transmission to the user; and
- Store programmed commands for execution according to schedule.

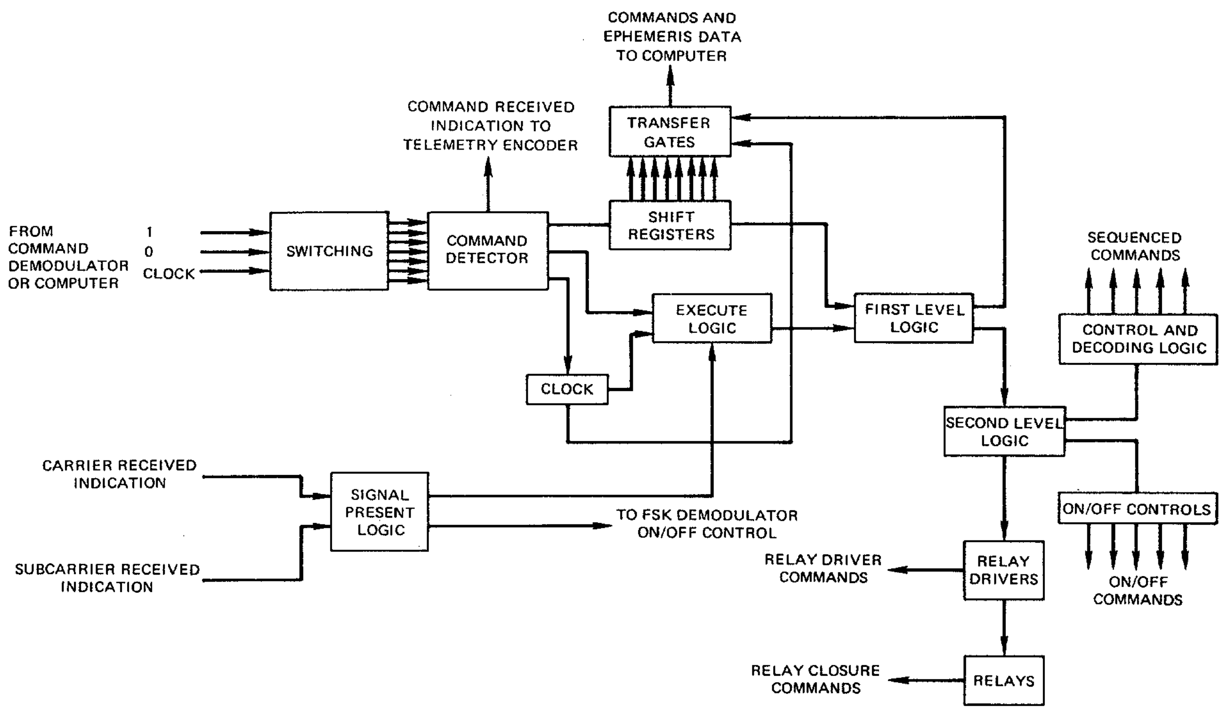


Fig. 4-18 - Timation satellite command receiver

Space-qualified digital computers complete with memory, a central processing unit, and input/output units are available to meet these requirements, the necessary computations can be accomplished by a single processor of moderate speed linked to a 40,000-bit memory. The required computational accuracy will be achieved through the use of 32-bit words. The computer design will employ integrated circuits and MSI technology throughout to ensure low-power and weight characteristics.

The commands when received are processed according to their type. Each command sent to the satellite causes one of the following actions to occur:

- Logic pulses are generated to initiate the loading of the satellite memory with the satellite ephemeris data and commands to be stored for future action.
- Sequences of commands stored in the computer memory are initiated. After the command which starts the sequence is received, the sequence continues without further instructions from the ground.

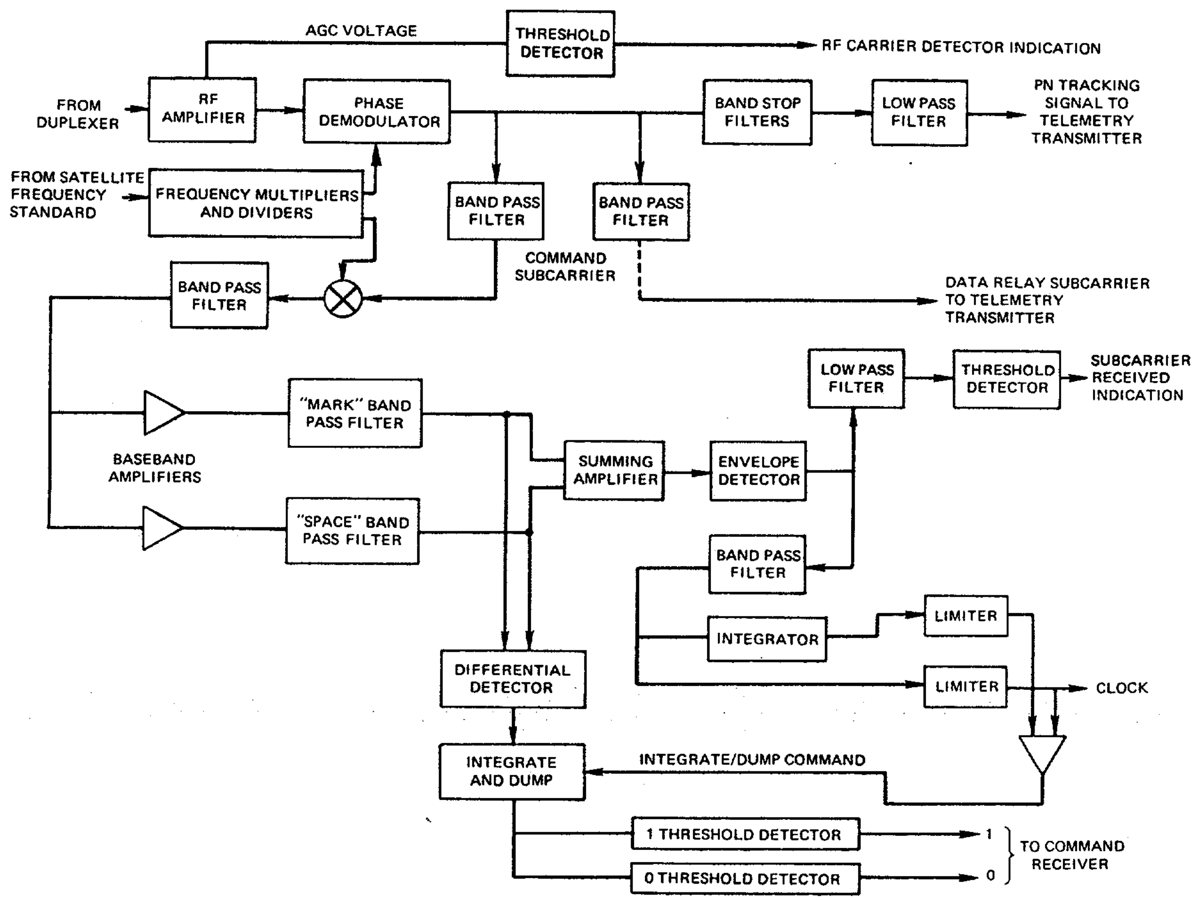


Fig. 4-19 - Timation satellite command demodulator

4.2.6 Power Subsystem

The power subsystem components (solar array, rechargeable batteries, and power conditioning electronics) are arranged in an efficient configuration to provide power for spacecraft subsystems and mission operational equipment. Tradeoffs of several alternate configurations were considered and are discussed in the following paragraphs. The effects of operation at orbital altitudes of 8000 and 12,000 naut mi and mission lifetimes of 5 to 8 years on subsystem components were also considered.

The load power profile for operational Timation orbits and the duty cycle for each load are shown in Fig. 4-20. Other infrequently applied power loads associated with the propulsion system and the attitude control system are not shown, but their influence has been included in the sizing of the power subsystem.

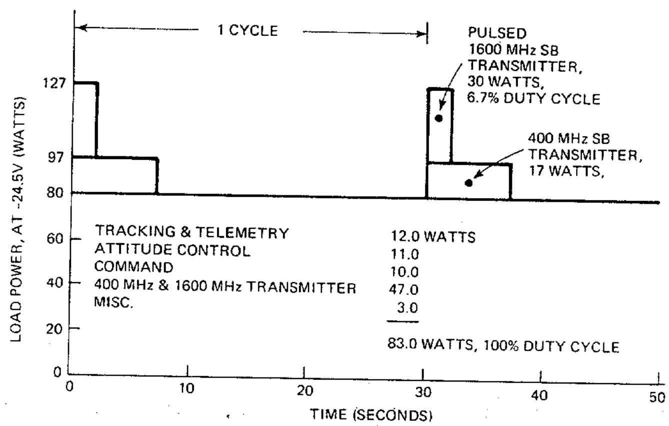


Fig. 4-20 - Load power profile

Orbital parameters, the spacecraft structural shape, and the attitude-control constraints produce sun angles varying from 8.5 to 90 degrees (zero to 90 degrees were used for design parameters) and a solar-array daytime-operating-temperature range of -50° to 50°C, with a minimum nighttime solar-array temperature of -110°C.

4.2.6.2 Candidate Power-Subsystem Configuration — Four power subsystem configurations were examined by using conditions for tradeoff studies that are sufficiently close to the operating conditions to permit meaningful comparisons. The configurations considered are shown in Fig. 4-21. They fall into two main categories: tracking and nontracking. Each category has series and parallel forms. In the tracking system, the maximum power point of the solar array is continuously tracked (electronically), and the power is transferred to the load through power-conditioning electronics. In the nontracking systems, on the other hand, the solar array operates at an essentially fixed voltage. Although mission conditions often determine which of these categories of subsystems is more acceptable, it can be stated generally that the colder the array, the more preferable is the tracking system.

The series/parallel form indicates the relative position of electronics relative to power source and load; i.e., a series system has electronic elements in series with the load whereas the elements of a parallel system are in parallel with the load. Generally, the higher the fractional sun time per orbit, the more desirable is the parallel form. This nontracking, parallel system is commonly called the direct energy transfer (DET) system.

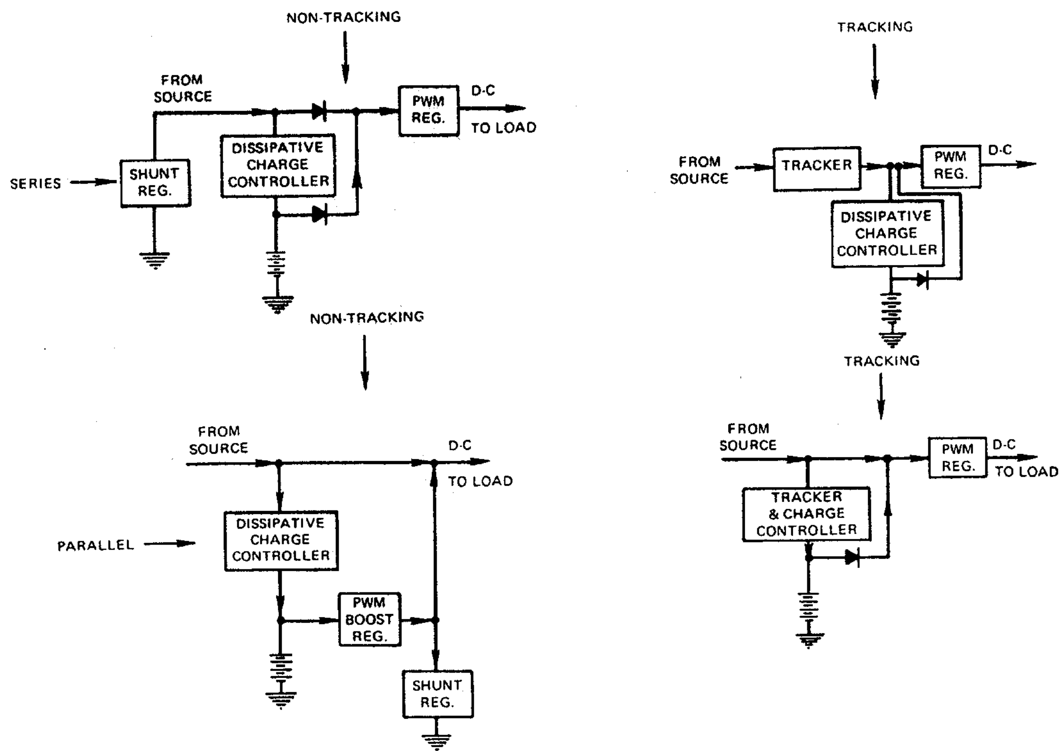


Fig. 4-21 - Comparison of power subsystems

The results of an analysis (Table 4-8) show that the minimum weight and most efficient configuration (weight/average load power) is the DET arrangement. A description of the major components of the DET system and the influence of lifetime and orbit altitude is given in Table 4-9. The table shows that the power subsystem weight is 11 pounds more for the 12,000-naut mi orbit than for the 7500-naut mi orbit. This increase is caused primarily by a longer eclipse time at the higher altitude and the resulting higher battery weights.

Table 4-8
Weight Estimates

Power Subsystem Configuration	Normalized Weight Estimates (pounds)				Weight/ Average Load Power
	Solar Array	Battery	Electronics	Total	
Tracking					
Series	1.26	1.18	1.20	1.21	2.55
Parallel	1.12	1.18	1.20	1.12	2.36
Nontracking					
Series	1.08	1.18	1.00	1.07	2.25
Direct energy transfer (DET)	1.00	1.00	1.00	1.00	2.11

Table 4-9
Power Subsystem Components

Characteristic	7500 naut mi Altitude		12,000 naut mi Altitude	
	5 Year	8 Year	5 Year	8 Year
Maximum average load P (external)	101.5 W	95.8 W	117.6 W	110.8 W
Estimated weight of electronics	15 lb		17 lb	
Estimated weight of batteries	44 lb		53 lb	
Estimated weight of solar array	107 lb			
TOTAL WEIGHT	166 lb		177 lb	
Number of batteries	2			
A-H size of each battery	12 A-H		15 A-H	
Number of series cells in battery	15			
Composite battery A-H size	24 A-H		30 A-H	
Total Number of solar cells	8868			
Number of series cells on top panel	73			
Number of parallel cells on top panel	30			
Number of series cells on side panels	63			
Number of parallel cells on side panels	5 or 6			

The solar array converts solar radiation to the electrical power needed to supply daylight electrical loads, including recharging of the nickel-cadmium batteries. These batteries are used for nighttime operation and for daylight peak loads.

The array size is determined by the spacecraft shape, the stabilization method, sun angle, solar cell efficiency (as influenced by accumulated radiation dose, ultraviolet radiation, etc.), inefficiency of batteries, and other electrical control and distribution equipment and by the requirements of the spacecraft loads. For a worst case, the end-of-life (EOL) conditions are combined with an unfavorable sun angle. The array dimensions are based on these conditions and sufficient power is made available to supply all electrical loads at any sun angle during the entire mission life. Some of the conditions influencing the array design for the spacecraft are shown in Table 4-10.

Table 4-10
Solar Array Design Conditions

Spacecraft	Cylindrical, Body-Mounted Array on Despun Part of Spacecraft
Attitude control and orbit parameters	Spin axis normal to orbital plane, sun angle variation 0 to 90 degrees, and suntime variation 84 to 100%
Solar cells	Beginning-of-life (BOL efficiency: 11.07% End-of-life (EOL) efficiency: 6.0% Dimensions: 2 × 4 cm
Battery	Nickel-cadmium system
Array temperature (operational)	-50° to +50°C
Spacecraft electrical load	See power profile

The battery system selected (nickel-cadmium) is used extensively in long-life spacecraft. Some considerations used for determining the size of the battery defined in Table 4-9 are given in Table 4-11.

The power subsystem electronics (shown in Fig. 4-22) conditions, controls, and distributes the electrical power as needed by the various loads.

The primary function of the control electronics is to regulate the -24.5-volt bus to ±2%. This is accomplished during the satellite day by having the charge controller and shunt regulator circuits combine their action with the load to maintain the array output at this fixed voltage. During satellite night, the -24.5-volt power at ±2% is provided by batteries discharging through the pulse-width modulator (PWM) boost regulator. Control of the boost regulator, charge controller, and shunt regulators is the function of the mode selector.

The mode selector circuits measure small voltage differences at the regulated bus and control the operations of the system as shown in Fig. 4-23.

The direct energy transfer (DET) system includes a nondissipative boost regulator by which the dc input to the regulator is switched (chopped) to produce a square-wave-voltage pulse train of constant frequency. By varying the duty cycle of the pulse train (i.e., modulating its pulse width), the voltage appearing at the load is controlled. Since the duty cycle is less than unity and this term appears in the denominator of the voltage equation, an output voltage greater than the input voltage can be obtained.

Redundant regulators are provided which can be switched on automatically by a regulated bus comparator circuit or by ground command.

The four modes in which the charge controller can operate are:

1. Current regulator mode 1 (C/15 charge rate — current limit mode)
2. Current regulator mode 2 (C/40 trickle charge rate — current limit mode)
3. Voltage limiting mode (tapered-charge current mode)
4. Voltage regulation mode (modified current regulator mode).

Table 4-11
Battery Design Conditions

Power subsystem configuration	DET, using a battery boost regulator
Type	Nickel-cadmium system
System voltage	-24.5
Depth-of-discharge	25%
Number of charge-discharge cycles	1200 (5-year life)
Temperature	13° to 26°C
Charge rate	C/15
Trickle charge rate	C/40
Charge efficiency factor*	0.62 to 0.55

*Based on capacity in ampere-hours.

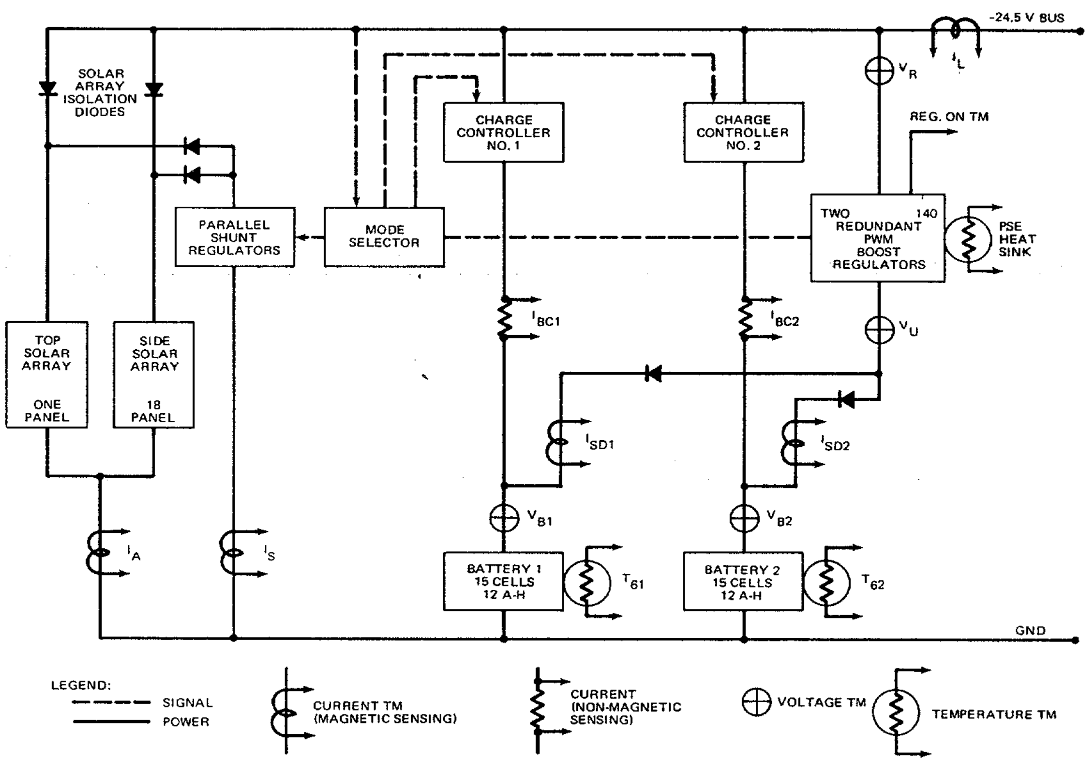


Fig. 4-22 - Functional block diagram of the power subsystem

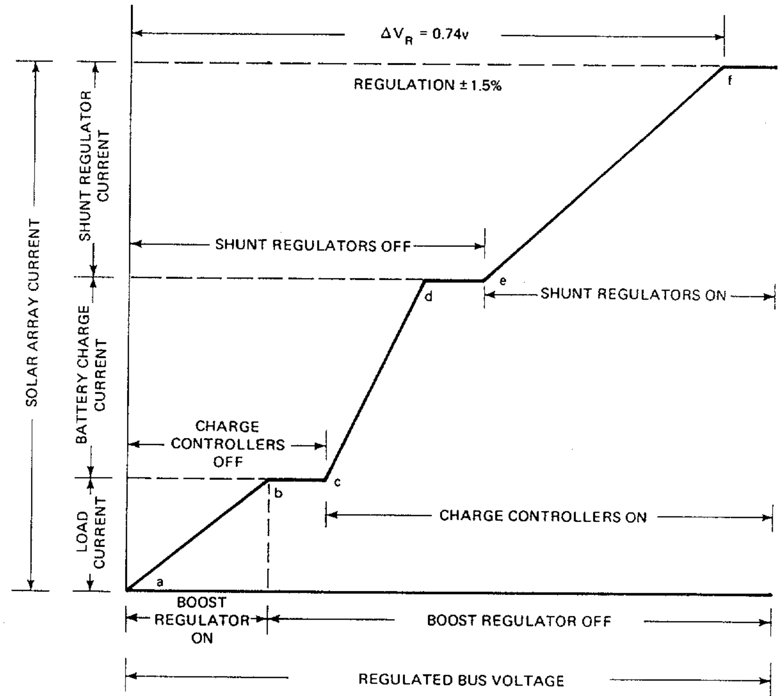


Fig. 4-23 - Mode selector operation (bus voltage profile)

For mode 1, a closed loop, feedback, current regulator is used. With some modification, this circuitry is also used for mode 2. Limiting the voltage to a predetermined value, i.e., mode 3, the controller reduces the current (tapered-charging) and operates essentially as a voltage regulator. In mode 4, the controller varies the current to the battery to regulate the power bus. This condition occurs when the battery is being charged and the power-load demand increases. For modes 1, 2, and 3, excess array current is bypassed through the shunt regulators.

Shunting excess array current is brought about by programmed load variations and temperature and age-induced variations in array output capability.

The pass element of the shunt regulator is a transistor in series with a resistor. Several parallel circuits are used to distribute the substantial heat uniformly throughout the spacecraft.

4.2.7 Thermal Control Subsystem

Thermal control of a spacecraft can be achieved by active or passive techniques, or by a combination of both techniques. The simplicity and reliability of the passive design makes it the choice for the spacecraft design. Factors influencing passive design include equipment arrangement, power dissipation profiles, and internal and external thermal coupling factors.

The primary considerations and constraints for the thermal control system are (a) the spacecraft mission lifetime of at least 5 years, (b) a sun angle variation (determined by orbital parameters) of 6 to 90 degrees, (c) a variation of 1.7 to 1 in the power-producing capability of the solar array, (d) the battery-temperature constraint, and (e) the requirement that the thermal subsystem be relatively insensitive to radiation damage.

The spacecraft subsystems and instruments are mounted on a structural baseplate which serves as a primary thermal radiator. The outside surface of the baseplate is covered with a multifoil radiation blanket in which holes are cut to provide the proper radiation coupling to space.

The components mounted on the baseplate are covered with a radiation blanket. The blanket thermally isolates the baseplate equipment from the inner surfaces of the array which contain high thermal dissipation shunt regulator circuitry.

To complete the control system the exterior surfaces of the spacecraft have second surface mirrors distributed among the solar cells. Fused silica glass is used throughout to improve the radiation resistance of the mirrors.

A thermal profile (shown in Fig. 4-24) of the spacecraft has been determined as a function of sun angle. In addition, temperatures have been computed for one complete orbit where the sun angle is 90 degrees (Fig. 4-25). To maintain thermal equilibrium, the insulation blanket (assumed effective emittance of 0.01) covering the outside surface of baseplate requires a hole area of 173 square inches. It can be seen that the temperatures obtained will meet mission requirements.

4.2.8 Attitude Control

The subsystem performance requirements, which dictated the design selection for the attitude control subsystem, are:

- The antenna axis is to be parallel to the local vertical vector within 3 degrees.
- The (circular) orbit altitude will be 8000 naut mi at 55 degree inclination.
- Closed-loop pitch and roll/yaw control must be incorporated.
- Performance must be consistent with the spacecraft operational lifetime of at least 5 years.

The selected design for this subsystem incorporates a single momentum-wheel assembly, connected to a stable platform containing the antenna complement. In this concept, the spacecraft's main body is despun. Angular momentum is stored in a flywheel. Continuous earth orientation is achieved by placing the spin axis (pitch axis) along the orbit normal and rotating the platform at one revolution per orbit. Earth-horizon sensors operating in the 14 to 16μ (CO_2) band provide pitch error information which is used to transfer momentum between the platform and flywheel and thereby reduce the pitch error.

Spacecraft roll/yaw control (i.e., maintenance of the flywheel spin axis parallel to the orbit normal) is provided by magnetic control. Roll/yaw errors sensed by the same CO_2-band earth sensors are used with spacecraft orbit position information to energize an electromagnet. The magnetic field of the coils reacts with the earth's magnetic field to provide controlled torques to the spacecraft to correct spin-axis misalignments.

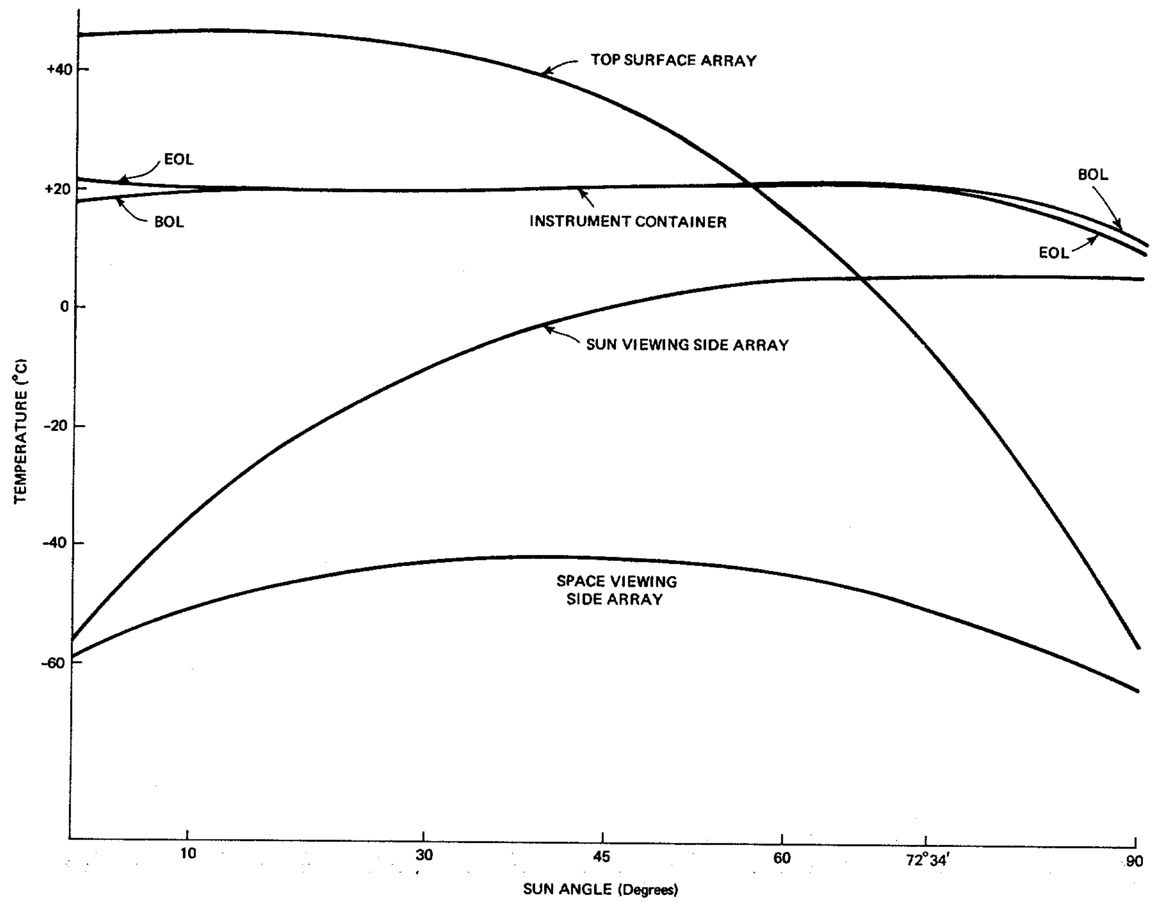


Fig. 4-24 - Steady-state temperature during spacecraft daylight

Several stabilization system techniques were investigated for the given mission requirements. Table 4-12 presents the important data on control subsystems commonly used. Note that all of these are potentially applicable to this navigation satellite mission.

Weight considerations eliminated the use of reaction jets. Even assuming zero leakage, a 5-year mission life would require an unacceptably large propellant load requirement.

Control moment gyros were eliminated because of weight, power, and life considerations. Their system dynamics tend to be more complex because of the highly nonlinear behavior of the attitude control loops. In addition, nonperiodic disturbances eventually saturate the gyros, and momentum dumping techniques are required.

A purely spin-stabilized configuration was rejected because of the obvious implications on power requirements. A spinning spacecraft (with a mechanically despun antenna platform) offers the second most acceptable stabilization concept to the proposed momentum wheel design. The spinning body offers the advantage of processing an inherently

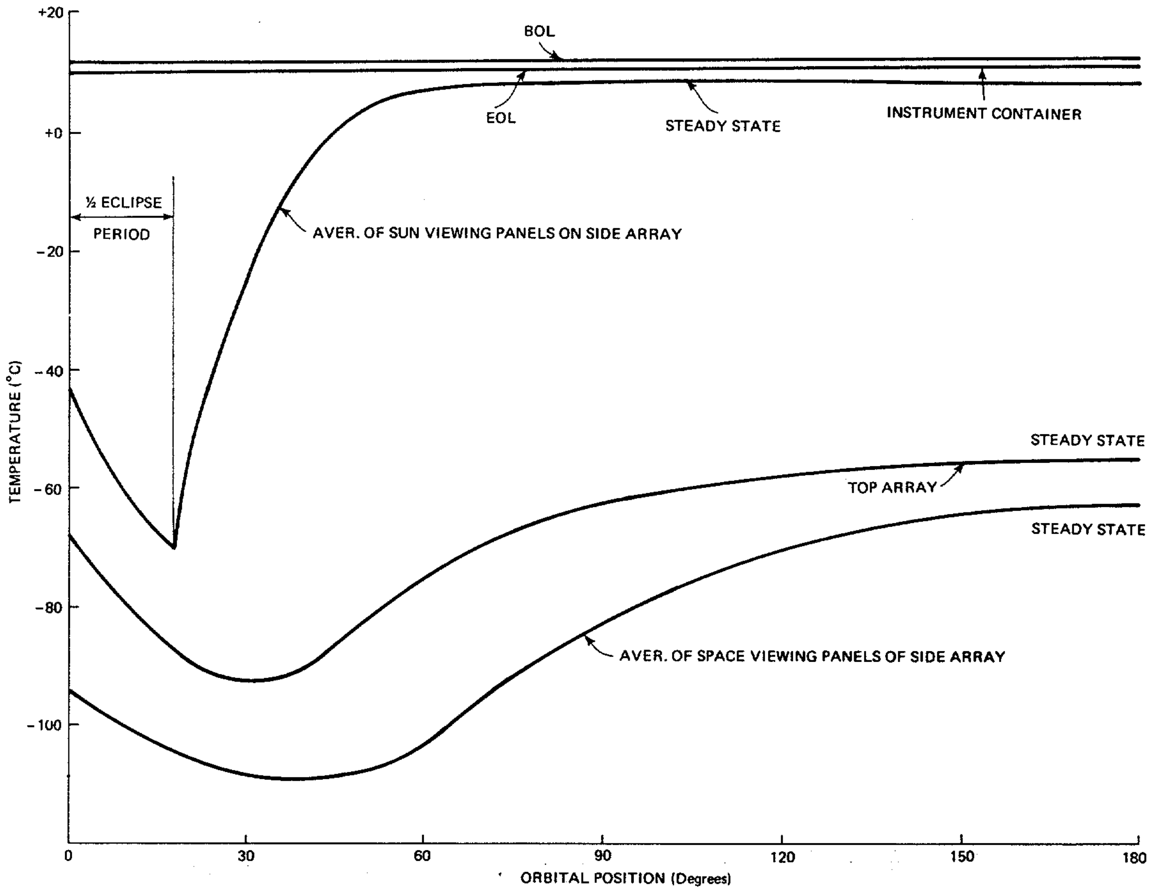


Fig. 4-25 - Temperature variation during orbit for which sun angle is 90 degrees

large rotating inertia, thereby eliminating the need for a flywheel. Some array weight is saved because spinning the array results in lower operating temperatures. The primary disadvantage of this configuration is the need for transmitting power across the rotary joint to the antenna and its equipment complement. Precision spin balancing of the large rotating mass also presents problems.

Two attitude control candidate systems remain—the momentum wheel assembly (MWA) and the gravity gradient.

The MWA proposed is an existing design. It is a single package containing a flywheel attached to a motor and encoded rotor. The motor and encoder stator are attached to the MWA housing, which is fixed relative to the spacecraft despun platform. A mirror assembly on the flywheel provides earth scanning. Two stationary thermistor bolometers produce the pulses at the sky/earth and earth/sky transitions which are used to provide pitch and roll/yaw error information. Local vertical (a generated reference pulse) is defined by the geometric bisector of the sky/earth, earth/sky horizon pulses. This new reference

Table 4-12
Spacecraft Stabilization Techniques

Classification	Momentum Control Torque Sources (Typical)	Onboard Power Requirements	Attitude Sensor Requirements	Control Logic Requirements	Significant Features
Passive	Gravity gradient (g-g) Solar Spin Magnetic Aerodynamic	None	None	None	Limited maneuverability
Semipassive	G-G, momentum G-G, control moment gyro (CMG)	Small	None	None	Internal torquing requires stored or accumulated power; open-loop maneuvers
Semiactive	Spin Dual spin G-G, momentum Stabilite	Yes	Minimum number	Some	More degrees of attitude control than are sensed; Pseudoclosed loop maneuvers
Active	Jets Wheels, jets Wheels, magnetics Wheels, g-g Dual spin, zero momentum Stabilite with closed roll/yaw loop CMG, jets	Yes	Sensors for all degrees of freedom	Significant	Closed-loop maneuvers
Hybrid	Several of Above	Yes	Sensors for more degrees of freedom than required for stabilization	Significant	Complete orientation flexibility and greater potential for precise pointing

pulse is compared with an encoder generated index pulse which defines the spacecraft yaw axis. The time difference between the two reference signals defines the pitch error at a rate of once per flywheel rotation. The signals are then used to control the rotation of the spacecraft body to achieve a near-zero pitch error. The flywheel stores 350 in.-lb-sec of momentum at an angular rate of 180 rpm.

An electromagnet is used to generate a magnetic dipole moment along the spacecraft pitch axis. This dipole moment reacts with the earth's magnetic field to provide a controlled spin-axis procession rate. The coils must be energized at the proper time in the orbit as determined by the direction of the earth's field vector to precess the spin axis in the proper direction to effect a zero roll error. Closed-loop control is achieved by sensing the magnitude and direction of the roll error and energizing the electromagnet at the proper time (in the orbit), and for the proper duration, to obtain the attitude correction.

A similar electromagnet is used to control the magnitude of the total system momentum. The magnetic dipole of this device is oriented perpendicular to the pitch axis such that, when energized, a pitch-axis torque is developed which can increase or decrease the total system momentum. Momentum losses at the 8000-naut mi altitude will be small, and the frequency of operation of this control coil will be less than once per week.

A passive, liquid nutation damper is used to control spin-axis nutation continuously. The damper consists of a 1-in.-diameter, closed tube, bent in a toroid with approximately a 40-in. diameter. The enclosed fluid is silicone oil of a kinematic viscosity selected to produce maximum energy loss per cycle of oscillation at the nutation frequency. A pressurized bellows ensures that no voids will occur in the fluid over the operating temperature range.

A digital sun sensor is used to provide backup information for attitude determination prior to station acquisition. It generates signals for the closed-loop, 180-degree attitude maneuver required when the sun angle approaches 90 degrees.

Gravity gradient (g-g) stabilization techniques offer a number of advantages over other types. Once the booms are deployed the technique requires no spacecraft power. The single moving element (the damper) is suspended in a friction-free environment. It rotates once every 8 hours. Once proper stabilization is obtained the gravity gradient technique requires no power nor adjustments; it should have a near infinite life. Such a stabilization device would not require precession of the spacecraft to eliminate undesirable sun angles.

The disadvantages of gravity gradient installations are that they do not provide pointing accuracies much better than $\pm 5°$ and they are sensitive to control jet forces.

At present the favored system is the MWA. Further experience will be available from both types before a satellite design is frozen.

4.2.9 Propulsion Subsystem

4.2.9.1 Requirements — Each satellite requires a propulsion subsystem for station acquisition, correction of orbit-injection errors, and station keeping maintenance during the life of the satellite.

The specific impulsive velocity requirements for an operating life of 5 years are listed in Table 4-13 for the selected launch vehicles (Titan IIID/Burner II and Titan IIIB/Burner IIA).

While bipropellant, monopropellant, plasma, resistojet, and electrostatic propulsion systems were considered as candidates for this mission, a Hydrazine Shell 405 catalyst monopropellant propulsion system was selected as the baseline system based on high performance combined with extensive flight applications, high reliability, excellent long-term storage capability, and low power demands.

4.2.9.2 Subsystem Description — The propulsion subsystem consists of four rocket engine assemblies (REA) and a propellant feed system. Each REA contains a propellant valve, a thermal standoff, a Shell 405 catalyst bed, a decomposition chamber, and an expansion nozzle. Propellant feed is achieved by an integral blowdown pressurization system utilizing the surface tension and capillary forces of screens in the tank for propellant management in the zero gravity environment. Tradeoffs with other (gravity-free)

Table 4-13
Mission Velocity Requirements

Item	ΔV (ft/sec)	Comments
Orbit-injection errors		
Inclination	63.5	–
Circularity	216	–
Trim	20.4	–
Subtotal	300	–
Station acquisition	72	160-degree change
Station keeping	125	25 ft/sec annually
Total	497	

feed systems were made, including those using elastomeric metallic and nonmetallic diagrams, bladders, preinduced settling by thrusting or spin-induced centrifugal force. Capillary screening was selected on the basis of superior operation under long-term storage conditions with modest weight and cost penalties. The propellant feed system also includes fill and drain valves, pressurization and propellant lines, and filters. Gas pressure and temperature transducers, for monitoring the available propellant supply, and a tank heater complete the system.

Dry nitrogen gas was chosen as the pressurant instead of helium on the basis of superior leakage characteristics. This factor assumes great significance for long-term storage applications. The nitrogen vs helium weight penalty incurred is negligible (0.5 lb) for the small tankage volumes used.

A functional schematic of the propulsion subsystem is shown in Fig. 4-26. Functional characteristics are shown in Table 4-14. Five-pound thrusters have been selected because the dynamics of the spacecraft are compatible with this thrust level. A total impulse of 6900 lb fuel-sec is required to provide a total velocity increment of 500 ft/sec. Although 32 lb of N_2H_4 is required for the total maneuvers, an additional 1.2 lb is allowed for expulsion inefficiency and thruster misalignment. A conservative specific impulse of 215 sec has been used for system sizing. This value is approximately 15 seconds lower than test measurements.

Nitrogen is initially loaded to a pressure of 400 psia. The final pressure after full propellant depletion is approximately 143 psia (a 2.8-to-1 expansion ratio), yielding a thrust blowdown range of 5.0 to 2.4 lb.

4.2.9.3 Control Operations — Thruster operation (selection of one of the four thrusters and the burn time) is initiated to obtain the optimal impulsive correction as determined by satellite tracking. A period of 20 days is allocated for dispersal. During this time, orbit plane inclination errors due to booster inaccuracies will be corrected. A final, orbit-period trim will complete the station acquisition.

Real-time telemetry readings monitor propulsion system status and integrity. Heaters are provided for protection under cold conditions.

Table 4-14
Propulsion Subsystem Characteristics

Characteristic	Type/Value
Blowdown propellant feed	Zero "g" capillary feed
Shell 405 catalyst	Bed
Total impulse	6900 lb fuel-sec
Thrust	5 lb fuel
Min. specific impulse, lb fuel-sec/lb mass	215 sec
Max. burn time	11.75 sec
Propellant/catalyst	N_2H_4/Shell 405
Pressurant	N_2
Number of thrusters (50:1 area ratio)	4
Number of propellant tanks	2
Propellant weight (including residuals)	33.2 lb
Pressurant weight	0.4 lb
Unloaded weight (dry)	14 lb

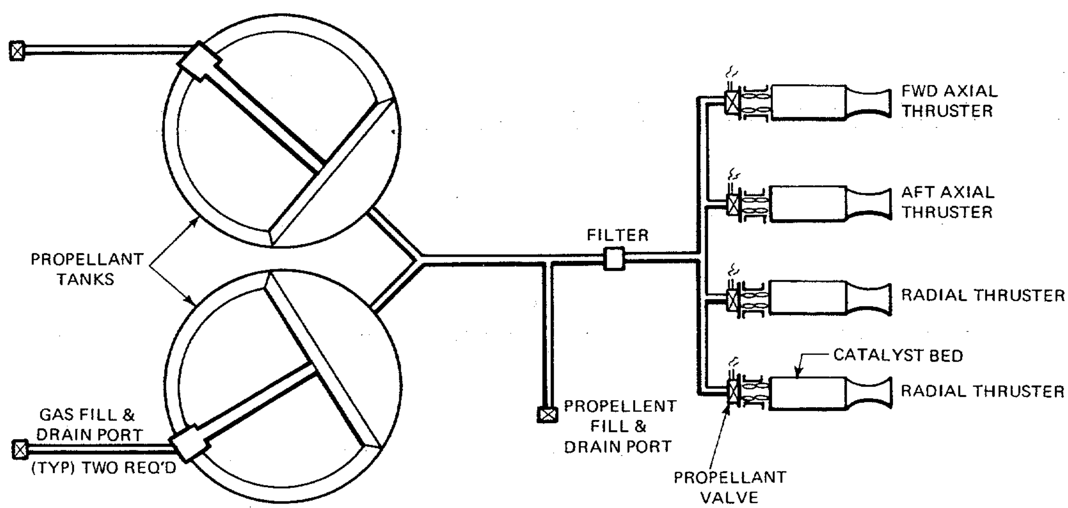


Fig. 4-26 - Functional schematic diagram of the propulsion subsystem

The axial thrusters, which are aligned with the momentum axis of the spacecraft, correct orbit injection errors and effect satellite dispersal*. The radial thrusters provide station keeping. Final trimming may be performed by either type of thruster.

*For the large ΔV's associated with these maneuvers, the thruster is fired while the main body rotates to reduce the effects of thrust misalignment torques.

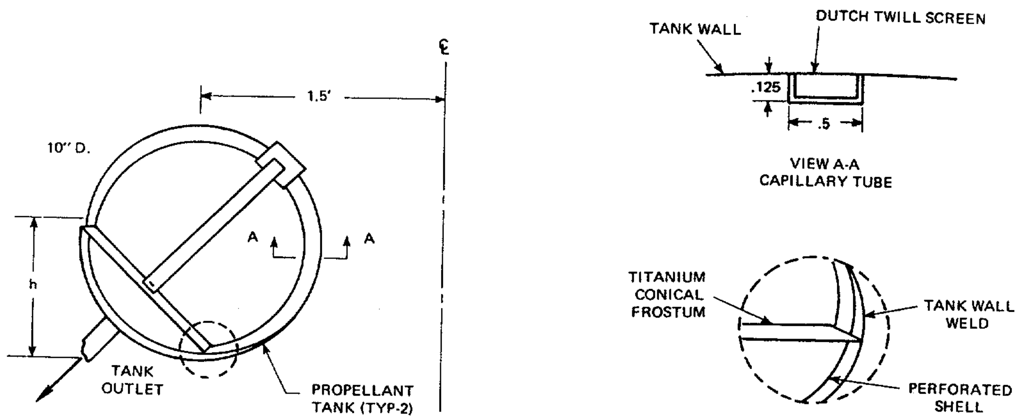


Fig. 4-27 - Capillary propellant-management tank

An advantage of the use of four REA's is that redundancy is provided in the event of a nonleakage-type failure of one or even two thrusters.

Thrusters may be operated in either a continuous or intermittent (i.e., pulsed) mode subject to the limitations of attitude-control capability and required velocity change. The continuous mode will be used for the correction of orbit-injection errors and satellite dispersal. Pulsed operation will be used during station keeping. Pulse bits of approximately 0.25 lb-sec are compatible with the 5-lb thruster. This bit produces a minimum velocity change of 0.075 ft/sec.

4.2.9.4 Implementation — The system includes a capillary propellant-management device in each of two tanks. This design, shown in Fig. 4-27, consists of a metal plate which divides the tank into two compartments interconnected by two screen-lined feed tubes. The smaller of the two compartments contains a screen liner over the tank outlet. The larger volume is sized for the spacecraft dispersal and initial orbit correction propellant requirements (about 80%). The secondary compartment contains the propellant required for station-keeping maneuvers. The capillary barriers on the tubes (woven wirecloth) are wetted by the propellant and serve to prevent pressurant from entering the secondary compartment until the larger volume is emptied. This design assures a propellant full compartment for station keeping.

All propulsion subsystem hardware has been either flight proven or, as in the case of the zero-gravity capillary screening, is generically similar to flight design.

The rocket engine is typical of the 5-lb-thrust, monopropellant hydrazine engines that are available. Each engine is installed such that the nozzle is aligned with the spacecraft's center of gravity to within 1/4 in.

The propellant tanks are required to provide a total volume of 1500 cubic inches. The tanks are symmetrically installed about the spin axis, 180° apart, to minimize the axial cg travel along the spin axis during expulsion.

4.3 GROUND STATION DESIGN

The selection of the number and siting of the ground stations are intimately related to the constellation selection and has been covered in the discussion of Subsystem Tradeoff Studies. The approaches to satisfying the functional requirements of each ground station are discussed in this section.

4.3.1 Remote Ground Station

The technical approaches and tradeoff studies relating to the remote ground station, as differentiated from the equipment and functional requirements (presented earlier), are presented here.

The necessity for transmitting significant quantities of tracking data from the remote ground stations (RGS) to the central ground station (CGS) warrants consideration of data compaction. A range of options exists between the two alternatives of transmitting all of the raw tracking data from the RGS to the CGS without any preprocessing and transmitting the tracking data to the CGS only after performing a maximum amount of preprocessing and compaction at each remote station. The first alternative imposes a maximal load on the central ground station and the data communication channel while minimizing the processing function of the remote stations. The second alternative distributes the processing load among the central and remote stations while minimizing the required capacity of the communication link. The second alternative has been selected for the design approach because a relatively low-capacity communication link is a distinct advantage and the distribution of some data-processing functions from the central to the remote stations results in no loss of useful information.

The tracking data preprocessing to be performed at each station will include (a) time tagging, (b) metric conversion, (c) range ambiguity correction, (d) removal of known instrument calibration biases, (e) corrections for tropospheric and ionospheric refraction effects, (f) editing, smoothing, and compacting of the tracking data, and (g) formatting. The sixth of these seven preprocessing procedures is the most time consuming and most complex mathematically. Numerous methods exist for performing this procedure. A generic method has been selected which can accomplish the required editing, smoothing, and compacting. This method uses prior knowledge of the orbit to determine the basis of a linearized method of optimal data filtering. Since the method permits varying the relative emphasis or importance given to both fidelity to the observed data and prior knowledge, a few iterations of the method with different relative emphasis will sequentially effectuate the editing, smoothing, and information. These approaches are described briefly in the following paragraphs.

For the active approach, a two-way, S-band cw ranging link has been specified for satellite tracking. This link comprises a transponder in the satellite and tracking equipment (antenna, transmitter, and receiver) at the ground station. Figure 4-28 shows the related clock update method. This method also provides tracking data (range).

Satellite clock synchronization is accomplished as follows: The satellite transmits a ranging waveform synthesized from an ultra stable clock. This waveform is received by the ground station and compared with a similar waveform synthesized from the station time standard. The resulting time (phase) difference is equal to the transit time (τ) plus the time difference (δ) between the satellite and ground station clocks (i.e., $\tau + \delta$). The locally generated ranging waveform is now advanced in time (phase) by $\tau + \delta$ and transmitted to the satellite, where it is translated into frequency and transponded back to the ground station. The round-trip delay is $2\tau - (\tau + \delta) = \tau - \delta$ with respect to the ground

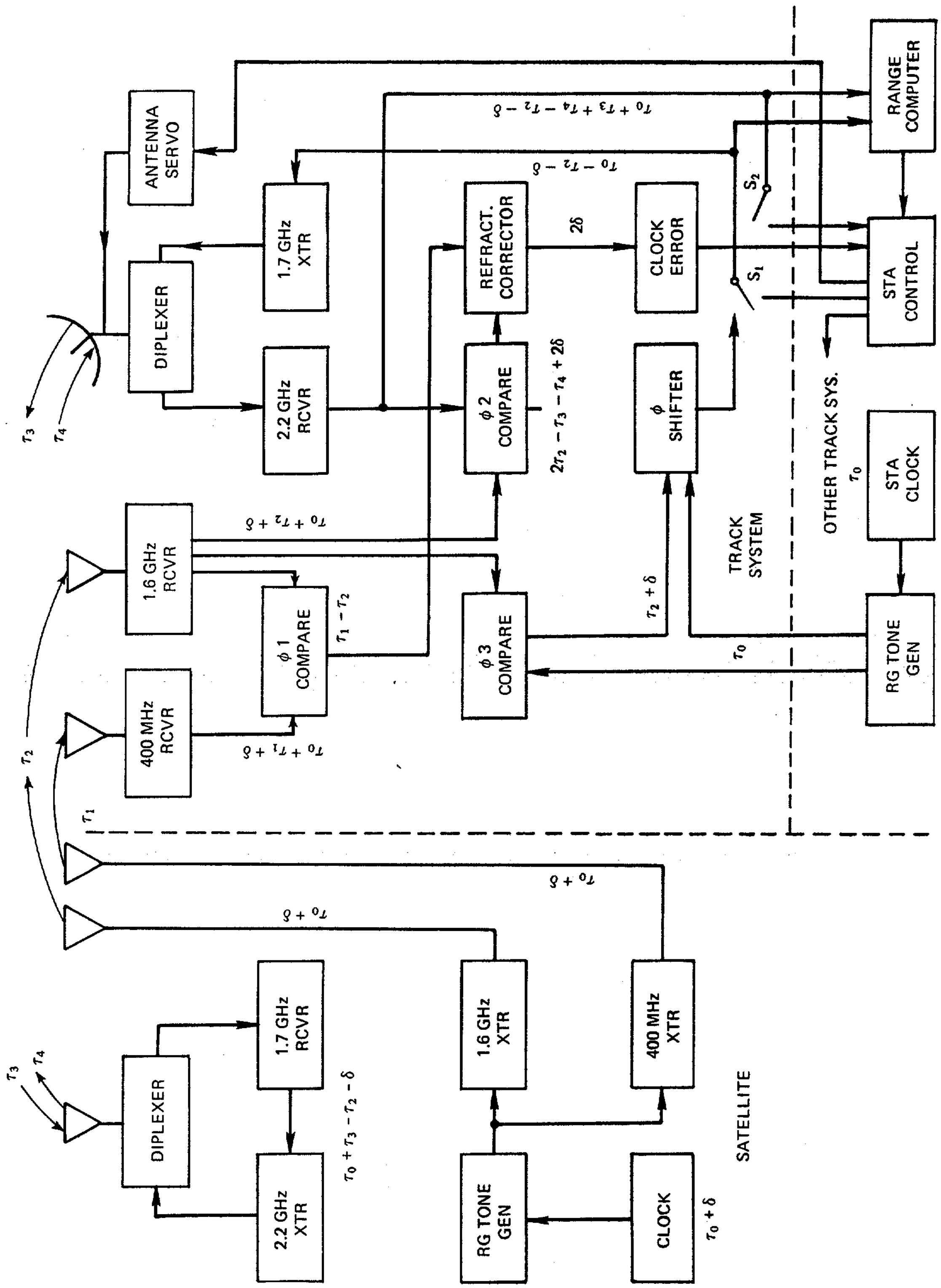


Fig. 4-28 - Active approach of satellite clock synchronization and range tracking (functional block diagram) station clock. If the transponded waveform is compared with that synthesized from the satellite clock, the time (phase) difference is $\tau + \delta - (\tau - \delta) = 2\delta$ or twice the satellite clock error. Thus, tracking data and clock error data are obtained.

Refraction correction is required and is obtained from the 400-MHz and 1600-MHz signals. These signals are compared to obtain a time difference between the received signals at these frequencies. This time difference is due to refraction; hence, it can be used in compacting the tracking data.

Each ground station monitors the status of satellite clocks within range, determines their time and frequency errors relative to standards located in the station, and commands the appropriate corrections to obtain satellite clock synchronization. In addition, the stations are periodically synchronized with one another as discussed in section 4.3.1. Synchronization with Naval Observatory time can be provided to give the subsidiary facility worldwide time dissemination.

Satellite clocks may be synchronized by either a remote ground station or the central ground station. A satellite in view of both the CGS and an RGS will always be synchronized by the CGS and will frequently be used as a time transfer relay between the CGS and an RGS.

All satellite clocks are synchronized to within 5 nanoseconds.

There are two basic approaches to satellite clock synchronization. The first approach assumes the availability of two-way ranging equipment in the ground stations and will be termed the active approach. The second approach is termed passive and assumes no independent ranging link. This approach relies on appropriate processing of the navigation signals received from the satellite to provide both tracking data and clock update to the refraction corrector to obtain the correct estimate of satellite clock asynchronism.

The principal difference between the active and passive approach is the speed of implementation. In the active approach, the satellite clock error can be determined in real time and the appropriate correction transmitted over the command link. In contrast, the passive approach requires that a number of samples of range and doppler data be assembled and processed before a satellite clock correction can be computed. These samples must be distributed over a long enough time span to ensure the required accuracy in satellite position and clock error determination.

A block diagram of the correction process for the passive approach is shown in Fig. 4-29. Range and doppler data are received from the satellite, sampled, digitized, and fed into the computer. The range and doppler data are corrected for atmospheric refraction and used to derive satellite position, clock time, and frequency error. The error quantities thus obtained are suitably coded and transmitted to the satellite over the command link. The commands are converted to phase and frequency adjustments to the satellite clock.

The active approach was selected for incorporation in the Timation system, primarily because of the speed with which satellite clock time can be measured and corrected. This speed is an important factor in limiting the worst-case navigation performance: correction of the satellite clocks after long periods of no contact with a ground station. No additional receivers or transmitters are required for the active approach. The 400- and 1600-MHz receivers serve two other functions: monitoring the navigation subsystem and providing information on ionospheric refraction. Uplinks and downlinks to the satellite

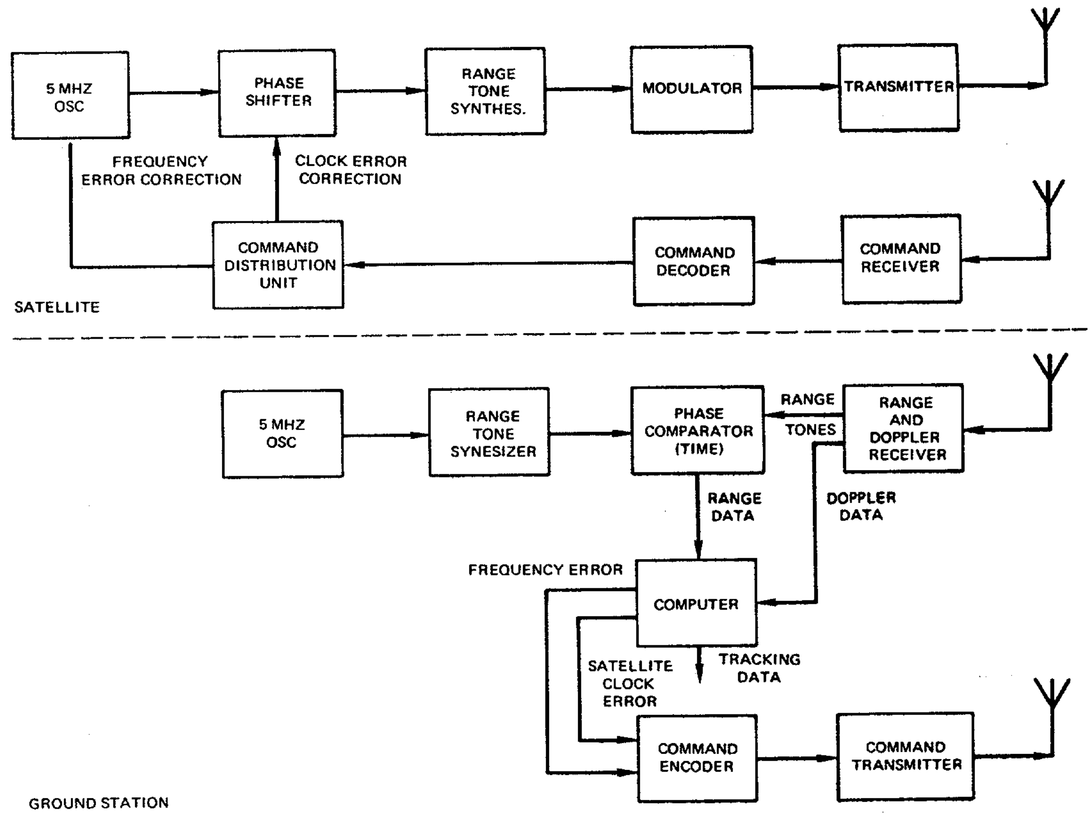


Fig. 4-29 - Passive approach to satellite clock synchronization and range tracking (functional block diagram)

are required for the various communications functions. Also, the active approach provides redundancy, in that the active-configuration equipment will also function in the passive mode to provide (degraded) backup in an emergency.

4.3.2 Central Ground Station

In addition to satellite clock synchronization, which is accomplished by both the central and remote ground stations, the system time standard needs to be synchronized in the CGS and the secondary standards in the RGS. A satellite clock is used as a time transfer agent. A satellite which has a high angle of elevation when viewed from both stations is selected for the interstation clock synchronization operation. Initially, the satellite clock is compared with the central ground station standard and corrected in the manner described in subsection 4.3.1. As soon as the satellite clock correction procedure is complete, the participating remote ground station makes its own time comparison against the satellite clock and nulls the time difference by appropriately adjusting the epoch of its secondary standard.

4.3.3 Operations Center

The primary data-processing functions of the operations center (Op Cen) are (a) the conversion of the preprocessed tracking data received from the remote stations into an updated orbit determination for each satellite and (b) the subsequent generation of predicted ephemerides for transmission and injection into the satellite memory. Since the utility of the results deteriorates with increasing age, these functions must be performed rapidly.

The requirement for high speed and high accuracy in operational orbit determination for the multisatellite system necessitates the implementation of a large-scale computer of an integrated software system (preferably of modular construction) at the operation center. The second function mentioned above implies that such a system must also be capable of performing the necessary ephemeris generation. The ASTRO program of the Naval Weapons Laboratory would perform fast enough to meet the requirements.

It appears that a single multiprocessor computer, such as the CDC 3800 (with backup provisions), can accomplish all the data processing functions required at the operations center.

4.3.4 Data Transfer

The data flow required between ground stations for maintaining the system in operational condition was shown in Fig. 3-19 of Section 3. Communications between the CGS and Op Cen are over hardwire links. Communications from the CGS and RGS to the satellites are accomplished on a 1800-MHz uplink. The corresponding downlink is at 2250 MHz. These links enter into the ground station data flow because the selected method of accomplishing communication between the several RGS and the CGS is through the system satellites.

This approach was selected for the following reasons:

- The links are too long for reliable and secure cable or microwave channels.
- The system satellites can provide the necessary interstation contacts without compromising the constellation from the navigation viewpoint.
- The uplink to the satellite (required for clock synchronization, tracking, ephemeris injection, commands, etc.) combined with the downlink (required for command verification, tracking, telemetry, etc.) forms a natural communication channel of approximately the correct bit-rate capability.

A data rate of 256 bits/second on the uplink and downlink is adequate to satisfy all functions, with the preprocessing provisions discussed in section 4.3.2 for the tracking data and with the compact representation of the ephemeris data mentioned in section 4.1.2.

It is estimated that the volume of tracking data required is about 500 suitably distributed range values per satellite per station per day to determine the orbits to the required accuracy. A single data point can be obtained in about 4 seconds. About 20 seconds is required to acquire the satellite. Tracking data can be obtained in about nine spans for a long duration pass. A minimum of 25 data points per span would be taken, requiring 120 seconds per span. Only about 18 minutes of a pass is therefore necessary for tracking. An antenna would normally be used to perform satellite clock time measurement and synchronization and data relay functions and tracking while locked to a satellite. Each operational channel will service two satellites on about a 40-percent duty cycle each. A complement of six channels therefore permits two channels to be down for maintenance or backup purposes, since eight satellites are normally in view from the near equatorial sites. About 7 seconds are required to communicate the compacted tracking data for a single pass and about 4 seconds to communicate a clock synchronization message to the spacecraft.

The tracking schedule would be similar on a pass by the central ground station. Commands will average 25 per message (1600/bits/message). The ephemeris message (24 hours) requires about 60 seconds for injection. The system is capable of transferring predicted ephemerides to an RGS for injection in the event of failure to inject from Alaska. Although 8 hours is the normal update period, operation of a satellite is possible for up to 24 hours between injections.

An average of 1 minute is normally required for the tracking data relay mode.

4.3.5 System Operations

Approximate data rates, volume, and contact times will be given in this section to indicate the feasibility of setting up a noninterfering tracking and communications schedule. Maintenance of 27 satellites in continual operational status will require detailed scheduling of all operations. RGS operations planning is required to ensure that no conflicts arise between communications requirements from various ground stations to or through each satellite. Scheduling of processing functions to ensure timely updating of satellite ephemerides will be a factor in the system accuracy.

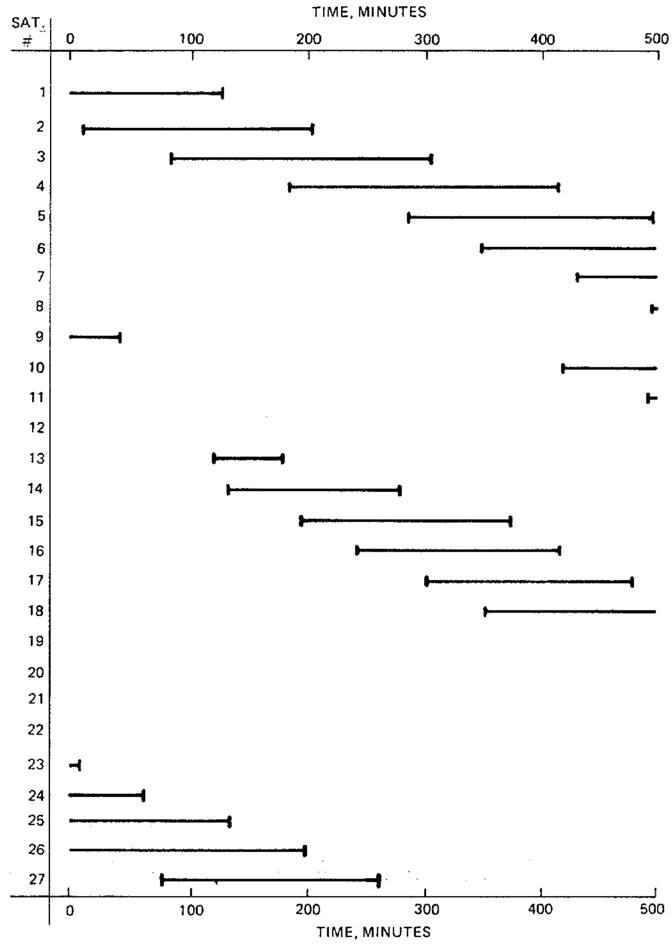
Each satellite will be in view of the Alaskan station on every orbit. The minimum contact time with this station is about 60 minutes, and the maximum elevation angle on this pass is 13 degrees. Adjacent passes of any one satellite over the Alaskan site last for about 170 minutes. Most satellites have two contacts per day with three remote ground stations. Some satellites have only one contact per day (about 240 min) with one of the RGS, but the ground station distribution is such that such satellites have two long contacts per day with the other RGS sites.

A typical contact time chart for an RGS is shown in Fig. 4-30. Up to nine satellites may be in view at any time. Tracking does not need to be continuous throughout all passes. On high-elevation passes, a satellite would be tracked on about a 50-percent duty cycle, distributed through the pass in four or more spans of 20 minutes each. This technique ensures adequate tracking data and permits time for antenna slewing. Scheduling is established in advance using approximate (old) orbital parameters. Each satellite is in contact with one other ground station approximately 40 percent of the time during which it is in contact with a specific RGS.

4.4 USER EQUIPMENT DESIGN

4.4.1 Introduction

This section deals with the selection of the user equipment outlined in section 3.5 and discusses in further detail the operation and performance of the equipment selected. Subsection 4.4.2 presents the advantages of coherent range-tone processing and gives precise implementation and a detailed description of ranging as accomplished in the noncoherent pulsed-KRP receiving equipment. Subsections 4.4.3 through 4.4.8 separately establish each user category's operational navigation requirements and prescribe Timation equipment configurations for their satisfaction. Subsection 4.4.9 is a review of peripheral controls required for the operation of each equipment configuration.



Note: Schedule repeats every eight hours with satellite index incremented by 9 modulo 27.

Fig. 4-30 - Typical contact time schedule for a remote ground station

4.4.2 Receiver Design

The six classes of users of the navigational satellite system all employ the same four basic navigational signal demodulators. The differences between the receivers of different classes are in the number of navigation signal channels, the stability of the user frequency standard, and the complexity of the user computer and navigation display. The four types of navigational signal demodulators are 400-MHz range-tone receiver, 1600-MHz range-tone ionospheric-correction receiver, 400-MHz KRP antijamming receiver, and 1600-MHz KRP antijamming receiver. These demodulators are described in detail in the following paragraphs.

4.4.2.1 400-MHz Range-Tone Receiver — The two generic implementations of 400-MHz range-tone receivers are the noncoherent and coherent. A noncoherent receiver consists of an rf front end, local oscillator generator, a broadband i-f (sufficiently wide to pass the carrier, subcarrier, and highest range tone without appreciable phase shift), detector, and narrowband low-pass filters for each received range tone. Since the receiver is broadband, it is susceptible to rf interference. The delay through the entire receiver (from the rf front and through the low-pass range-tone filters) must be calibrated as a function of received frequency and must remain stable.

The coherent range-tone receiver implementation has two separate narrowband i-f channels — the carrier loop and the range-tone loop. The bandwidth of the carrier loop can be made narrow since it passes only the carrier. The bandwidth of the range-tone loop can also be made narrow because all the range tones are derived from the carrier frequency.

The range-tone frequencies are synthesized in frequency coherence (but not phase coherent) with the received range tones after the carrier phased-locked-loop has acquired the carrier. As the range tones are transmitted, the phase of the internally synthesized range tones are brought into phase coherence with the received range tones. The internally synthesized range tones will remain in both frequency and phase coherence so long as carrier lock is not broken. This characteristic allows range measurements to be made even when the subcarrier and range tones are not being transmitted and permits hardware reduction for receiving multiple satellites. The subcarrier range-tone loop can be time multiplexed between the satellite range-tone transmissions while simultaneous range measurements are made from the several satellites.

Since the phase lock of the carrier and the range-tone loops is accomplished at the input mixers, the delays through the i-f amplifiers do not contribute to the range measurement. Only the delay through the broadband rf amplifier (preamplifier), which is small and stable, contributes to the range measurement.

The 400-MHz range-tone receiver, shown in Fig. 4-31, demodulates the nine sequential single-sideband-modulated range tones from the 400-MHz subcarrier and determines range by comparing the phases of the demodulated range tones with the phases of tones generated in the user receiver. Satellite ephemeris data are received between range-tone transmissions by demodulating the phase-shift keying on the ranging subcarrier.

The circularly polarized hemispherical coverage antenna is at the front end of the receiver. It is followed by an rf amplifier. This amplifier has a bandwidth wide enough to pass the 400-MHz range-tone frequency from any satellite. The amplifier is designed to have a small, premeasured phase variation with frequency.

Depending on the class of user, the output of the rf amplifiers is split into from one to four channels and is sent to the range-tone demodulators. Distance to a satellite is determined by comparing the phases of the range tones generated by the range demodulator with the phases of the outputs of a frequency synthesizer driven by the user clock.

In a ranging signal transmission from the satellite, each tone is transmitted for 0.4 second. Phase-adjusting network 1 in Fig. 4-31 is driven by the dc error voltage from mixer M3 such that the output of the phase-adjusting network is in phase with the received ranging subcarrier. Phase-adjusting network 2 is driven by the dc output of M3 so that the range-tone generator output is in phase with the range tones received from the satellite. The frequency-mixing technique in the demodulator causes the frequency of the signal that passes through the i-f filter between M2 and M3 to be the same for the subcarrier transmission and all nine range-tone transmissions.

The phase between each range tone from the range demodulator and the corresponding output of the user range-tone generator is measured by counting the number of cycles of a high-frequency signal that occurs between the zero crossing of the local range tone and

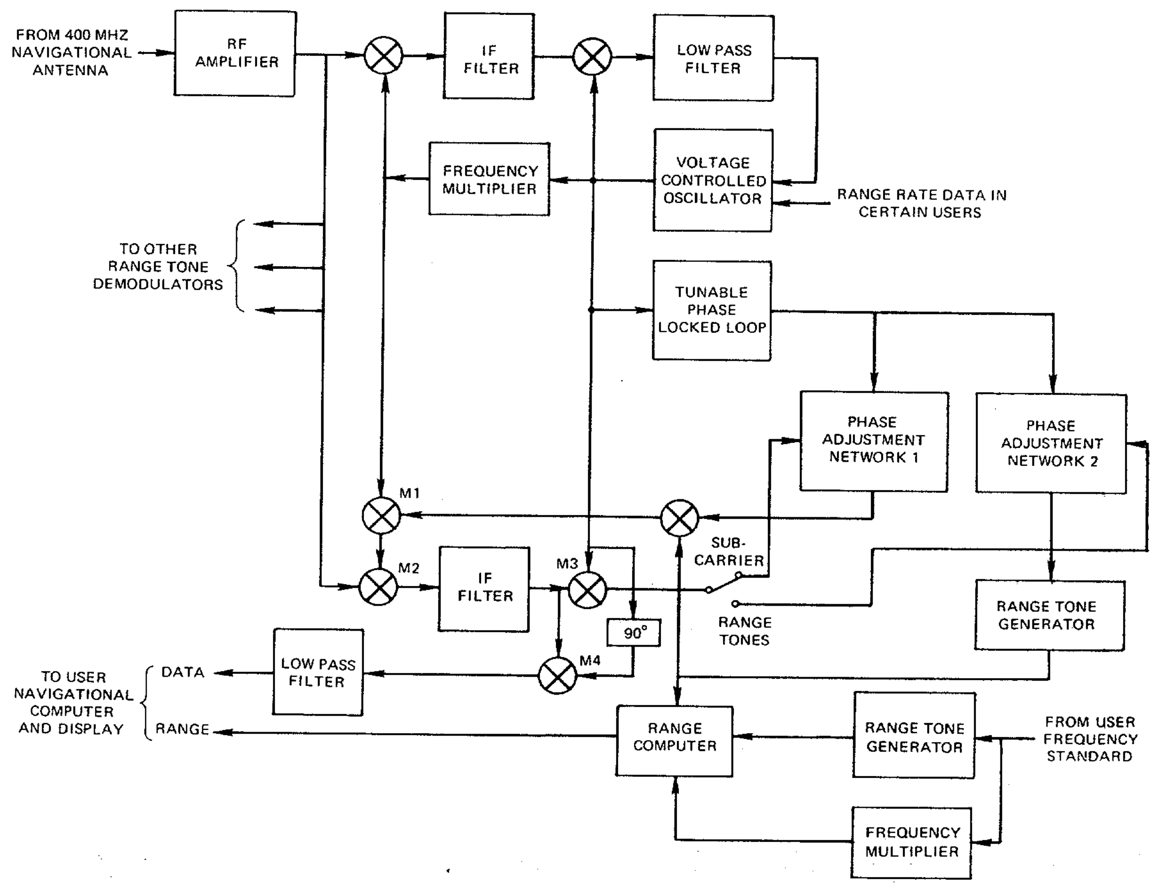


Fig. 4-31 - Block diagram for the 400-MHz range-tone receiver

that of the demodulated range tone. The results of the phase measurements made on the nine range tones are either put into a computer for fast range determination (in the case of a high-performance user) or employed in the computations done by the user of simpler equipments.

A 400-MHz range-tone receiver consists of one rf amplifier, one local range-tone generator, and one frequency standard, plus a range-tone demodulator, counter, and portion of computer memory for each simultaneous range-tone measurement. When a satellite is selected, the frequencies of the two voltage-controlled oscillators, the i-f filter between M2 and M3, and the switching timing of a range-tone demodulator are selected by depressing a pushbutton corresponding to the satellite desired.

Phase-shift-keyed satellite ephemeris data are demodulated from the ranging sub-carrier by comparing the phases of the two inputs of mixer M3 while the phase shift of phase-adjusting network 1 is kept constant. Depending on the user equipment, these data, which are updated every 2 minutes, are stored in the computer memory or are recorded on paper.

4.4.2.2 <u>1600-MHz Range-Tone Receiver</u> — The 1600-MHz range-tone receiver provides ionospheric corrections to the range measurements made with the 400-MHz

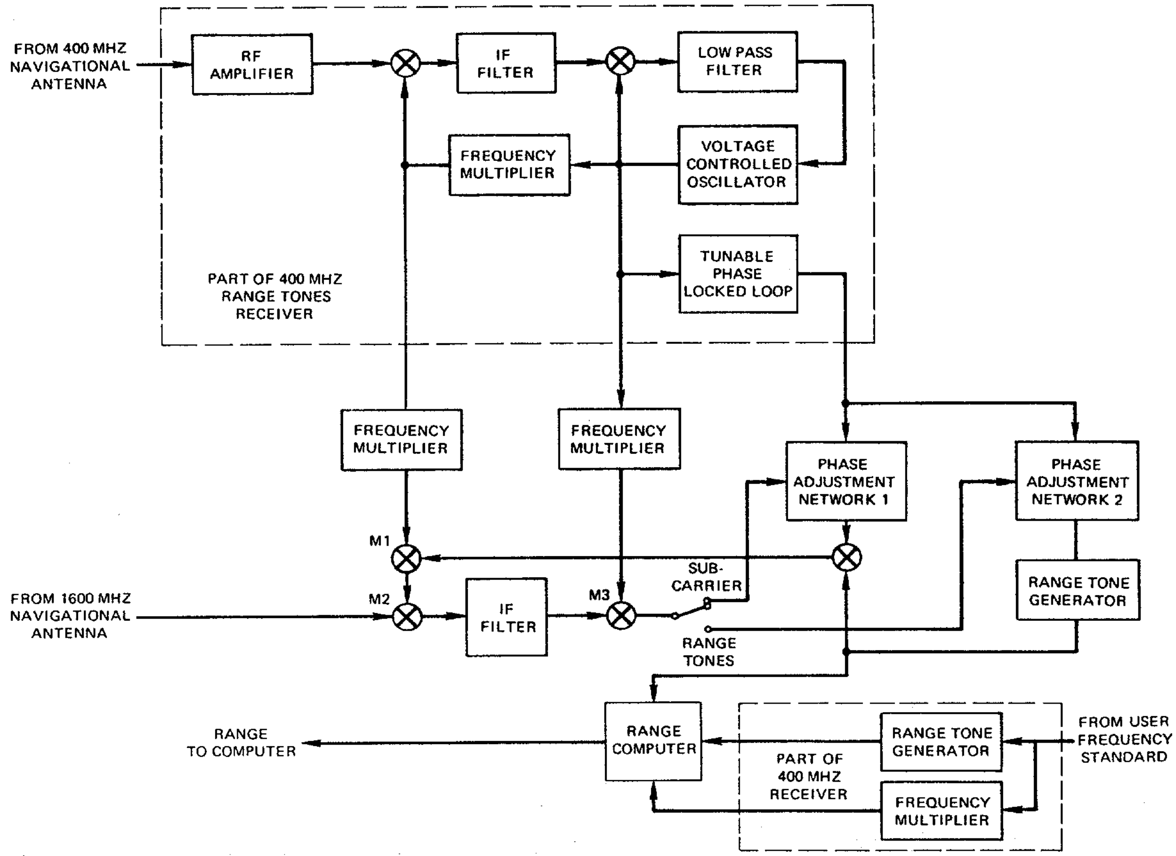


Fig. 4-32 - Block diagram of the 1600-MHz range-tone receiver

range-tone receiver. The receiver, shown in Fig. 4-32 with a portion of the 400-MHz range-tone receiver, demodulates four range tones from the 1600-MHz subcarrier. Three inputs to the receiver from the corresponding 400-MHz range-tone demodulator compensate for doppler shift of the rf signal.

A 1600-MHz antenna with a hemispherical receiving pattern and an rf amplifier provide the rf input to the 1600-MHz range-tone demodulator. This demodulator operates exactly like the 400-MHz demodulator. One phase-adjusting network is automatically set so that its output is in phase with the received ranging subcarrier. The other phase-adjusting network is set such that the outputs of the range-tone generator in the demodulator are in phase with the tones received from the satellite. Range is measured by comparing the phases of the range tones from the range-tone demodulator with the phases of the outputs from the range-tone generator. This phase comparison is made by measuring the time between zero crossings of the two signals with a counter, as was described for the 400-MHz receiver. The same user-frequency standard and range-tone generator is used for the 1600-MHz range-tone receiver as for the 400-MHz receiver.

The 1600-MHz range-tone receiver is always connected to the 400-MHz range-tone receiver if the user has only a single 400-MHz range tone demodulator. Otherwise, the three doppler-correcting inputs to the 1600-MHz range-tone are switched from one range-tone demodulator to another when the ionospheric corrections to the ranges measured

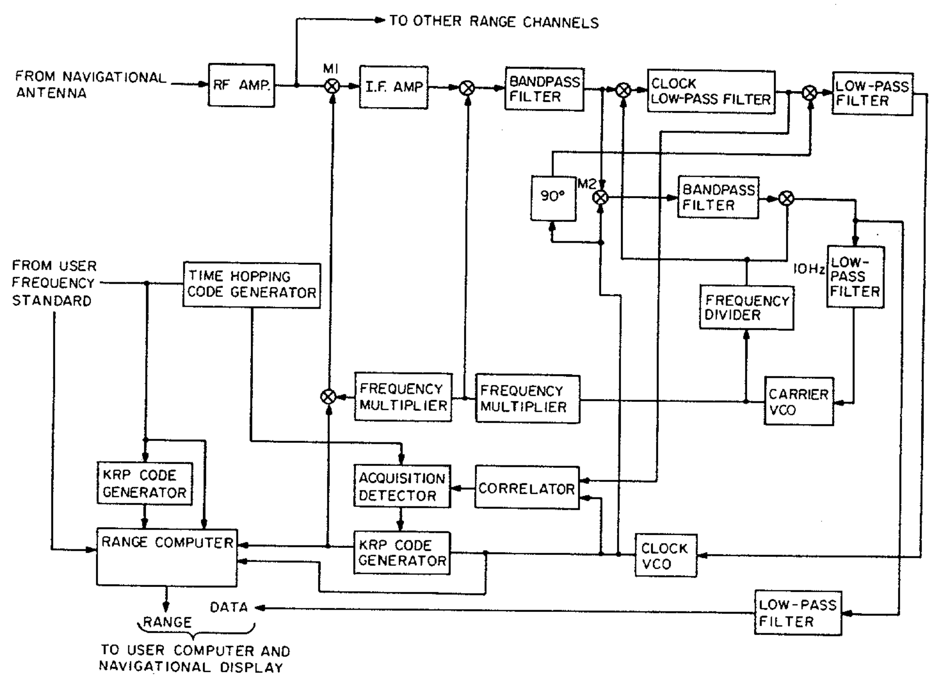


Fig. 4-33 - Block diagram of the 1600-MHz KRP receiver

between the user and the different satellites are computed. The range output of the 1600-MHz range receiver is switched to the appropriate section of the user computer memory when a range measurement is made. When range measurement on a new satellite is made, the i-f bandpass filter and the timing of the receiver switching are changed, either automatically or by depression of a pushbutton switch. Because the 1600-MHz receiver is not used until the 400-MHz range demodulator is locked onto the navigational carrier, there is no acquisition mode of operation for the receiver.

4.4.2.3 1600-MHz KRP* Receiver — The two KRP receivers are used as a backup for the range-tone receivers for a user operating in a jamming environment. The 1600-MHz receiver, shown in Fig. 4-33, computes range and demodulates ephemeris data from the satellite in the presence of a jamming signal with up to 47 dB more power than the average ranging signal power. The 1600-MHz navigational signal passes to the KRP receiver from the 1600-MHz navigational antenna that is also used to receive the 1600-MHz range-tone transmission from the satellite.

The KRP code used consists of the following length subcodes: 43 bits, 39 bits, 35 bits, and 2 bits.

The bit subcode is the clock component (CL). The product of the subcodes gives the total KRP code of 117,000 bits having a period of 0.012 sec, or 1900 naut mi ambiguity at 7.5 Mb/s.

*Keyed-random phase.

The subcodes 43, 39, and 35 bits will be referred to as KRP in the system configuration. Then KRP + CL is the complete code. This distinction is made because the 2-bit CL is acquired before the KRP subcodes are acquired. Different bit rates are required when simultaneous signals are present to permit extracting the CL component and acquisition of the corresponding subcodes.

The rf signal modulated with KRP + CL is fed to balanced modulator M1. The local oscillator driving M1 is modulated ±90 degrees with KRP alone from the receiver KRP generator. When the receiver KRP generator is phased to match the modulation on the rf signal, the i-f signal is then modulated with only the CL signal. The CL loop synchronizes the clock VCO which drives the receiver KRP generator. Receiver jamming protection arises because jamming signal energy is spread over the i-f band of twice the bit rate by M1, whereas the KRP + CL signal is compressed into an i-f band of 20 Hz by use of the 10-Hz postdetection loop filters.

When the KRP code is acquired, the CL modulation is removed at M2 leaving a cw carrier which is passed through a narrowband filter.

The received CL is correlated with the VCO clock while stepping subcode A along a bit at a time until maximum correlation is achieved. The same process is repeated for subcodes B and C. Maximum initial acquisition time depends on the signal-to-noise (S/N) and for this code is 28,000 N_0/C for 99.9% probability of acquisition, where N_0 is noise spectral density and C is total signal power.

The 1600-MHz link calculation gives a value of C/N_0 = 33.3 dB with no jamming. The S/N ratio in a 20-Hz two-sided band is then 22 dB. Maximum permissible jamming will reduce S/N to a lower limit of 10 dB, or an equivalent C/N_0 = 21.3 dB. Table 4-15 gives C/N_0, S/N, P_j/C, and maximum initial acquisition time. P_j/C is the ratio of jamming power-to-signal power at the receiver input. It may be seen that the acquisition time depends on the jammer received power. If KRP acquisition has been achieved before coming into a jamming environment, KRP synchronization will be maintained up to values of P_j/C = 46.6 dB.

Table 4-15
Characteristics for the 1600-MHz Link

C/N (dB)	S/N (dB)	P_j/C (dB)	Maximum Acquisition Time (sec)
33.3	22		8.85
31.3	20	32.3	28
26.3	15	40.9	88.5
21.3	10	46.6	140

4.4.2.4 400-MHz KRP Receiver — The 400-MHz KRP navigational channel is used to compute the ionospheric corrections that are made to the 1600-MHz KRP range measurement. A 400-MHz KRP receiver has only one ranging channel, as compared to three or four for the 1600-MHz KRP receiver. Otherwise, the 400-MHz KRP user receiver is identical to the 1600-MHz receiver described in the previous section.

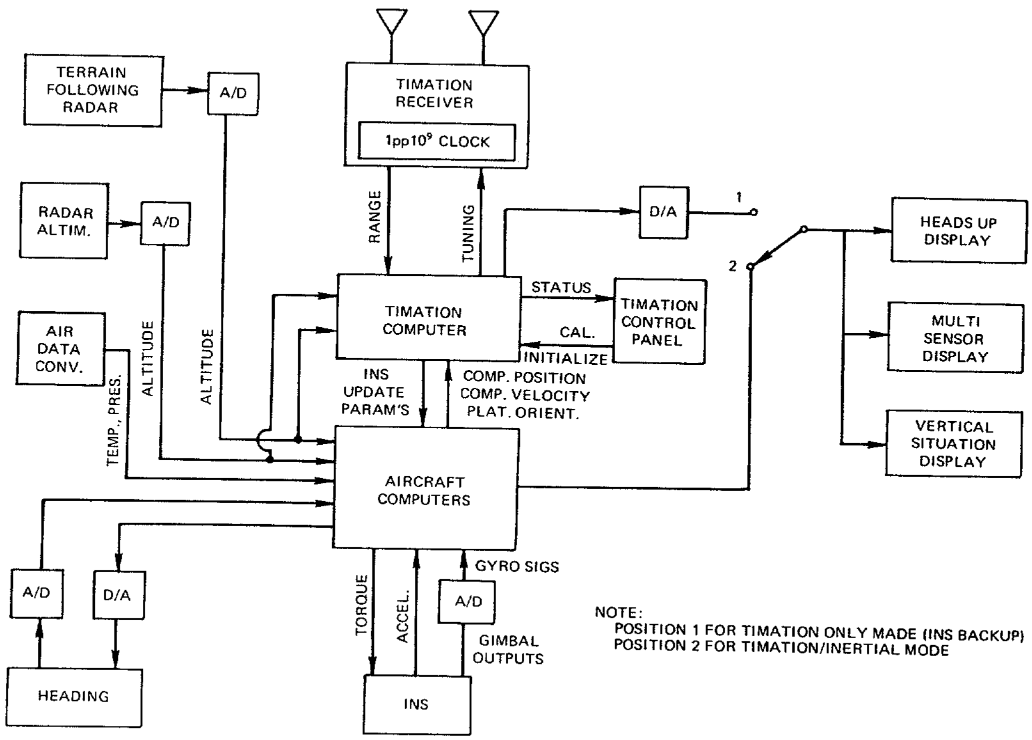


Fig. 4-34 - Signal flow diagram of the category 1 user equipment

4.4.3 The Category 1 System

4.4.3.1 Requirements — Category 1 users require continuous three-dimensional position fixes to within 50 feet rms; users are typically high-performance close-support aircraft and fighter-bombers. Preflight alignment time for a high-accuracy navigation system aboard such aircraft should be short to meet potential scramble situations. Operation of the system should be effective through 5G maneuvers. The category 1 user equipment should be designed for easy installation in a variety of aircraft and must be maintainable in a tactical field environment.

4.4.3.2 Equipment Configuration — A four-channel, KRP receiver with ionospheric refraction correction is employed in meeting category 1 navigation requirements. This receiver permits simultaneous acquisition and processing of the ranging signals from four satellites and a user-clock oscillator with a stability of 1 pp 10^9.

If the Timation satellite system navigation requires additional computer facilities beyond those available in the existing user computer, the existing computer will be supplemented by a separate computer performing Timation functions alone. The functional relationships between the computer facilities appear in the signal flow diagram of Fig. 4-34. No special displays are required by the category 1 user. Existing displays reflect Timation system navigation information which, in the normal operational mode, is used to update the user's inertial navigation system. In addition, an appropriate digital-to-analog signal converter is included so that the user displays can be directly driven from Timation system data.

4.4.3.3. System Performance — In adding the Timation navigation system to the category 1 facilities, three basic navigational modes become available – Timation/inertial, Timation and inertial.

The Timation/inertial mode will normally be used. In this mode the existing inertial navigation system is updated by the Timation data. The required 50-foot position-determination accuracy is achieved in this mode. The Timation computer will have the capacity to assume all navigation calculations when operating in the Timation-only mode. This mode provides a backup for inertial platform failure. The same 50-foot accuracy is achievable in the Timation-only mode. The inertial mode has inherent cumulative errors and would be operated along only in the event of Timation or both Timation and inertial failures, respectively. Operating mode performance for category 1 is summarized in Table 4-16. The major Timation and Timation/inertial performance specifications are given in Table 4-17.

Table 4-16
Category 1 Major Performance Specifications

KRP Performance Parameter	Value
Jamming margin $\left(\frac{\text{average jammer power}}{\text{average signal power}} \text{ for satisfactory performance}\right)$	56 dB
Maximum user acceleration	160 ft/sec^2
Acquisition time	15 sec

Table 4-17
Category 1 Operating Mode Performance

Mode	General Performance	Remarks
Timation/inertial	Highest-accuracy, continuous-readout position fix	Normal operational
Timation	Highest-accuracy, continuous-readout position fix	Inertial platform backup
Inertial	Cumulative error on order of 1 mph	Timation failure backup

4.4.4 Category 2 System

Category 2 encompasses military marine operations with typical users being submarines, aircraft carriers, and mine layers.

4.4.4.1 Requirements — Category 2 users require two-dimensional position fixes to within 50 feet rms. Because of the slower operating speeds and different mission characteristics, fixes are not required so often as for the category 1 user. As with the category 1 user, satisfactory system operation must be possible in a jamming environment.

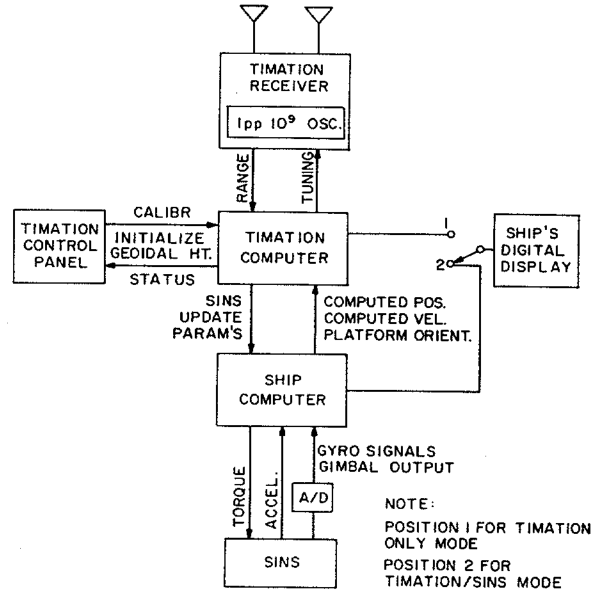


Fig. 4-35 - Signal flow diagram of the category 2 user equipment

4.4.4.2 Equipment Configuration — In meeting category 2 navigation requirements, a three-channel, KRP receiver with ionospheric refraction correction is employed. Independent altitude determination (with respect to a global geoid), accurate to within 50 feet, is readily available aboard ship. Range-difference processing is employed, thus alleviating the need for a precision clock. A crystal oscillator with a stability of 1 pp 10^9 provides a suitable reference.

Existing displays are used with all standard category 2 Timation-aided navigation system installations. Figure 4-35 shows the category 2 signal flow diagram.

4.4.5 Category 3 System

Category 3 users are military land vehicles requiring position fixing to within 50 feet rms in a jamming environment.

4.4.5.1 Requirements — Equipment is designed for easy installation in a variety of vehicles including jeeps and tanks. Also, it meets the requirements of field operation and maintenance. Information displays are supplied as part of the Timation equipment for the category 3 user.

4.4.5.2 Equipment Configuration — A KRP Timation receiver is required by the category 3 user for jamming environment protection. Three ionospheric refraction-corrected channels are sufficient, assuming that an independent measurement of altitude is made by the user. Barometric altitude determination can be used with conversion to geoidal height made by the computer. The required digital display will be an integral part of the control panel. Power is obtained from the vehicle's battery/generator supply, and voltage conversion and regulation are performed within the receiver.

4.4.6 The Category 4 System

Category 4 is designed for servicing the position location needs of military land operations performed on foot.

4.4.6.1 Requirements — The equipment must be small, lightweight, self-contained, and easily transportable. Category 4 user missions typically involve reconnaissance, advanced area operations, search and destroy, and operations in hostile territory.

4.4.6.2 Equipment Configuration — The category 4 equipment is built into a back-mounted manpack easily removed for operation by the carrier or operable by a second man while backmounted. The manpack contains a four-channel receiver, a data processor, and a battery supply. The unit will be limited to 15 lb in weight and 20 watts in power consumption. Ionospheric refraction correction and its associated hardware can be omitted if other field units are navigating via the Timation system in the same theater of operations. Relative position fixing results, wherein errors due to atmospheric refraction and satellite clock offset are correlated (and hence canceled) for the cooperating navigators.

4.4.7 The Category 5 System

Category 5 consists of transport aircraft with a position-fix accuracy requirement of 5000 feet.

4.4.7.1 Requirements — These users will not be subject to a jamming environment. Category 5 users are equipped with displays adequate for use with a Timation-aided navigation system.

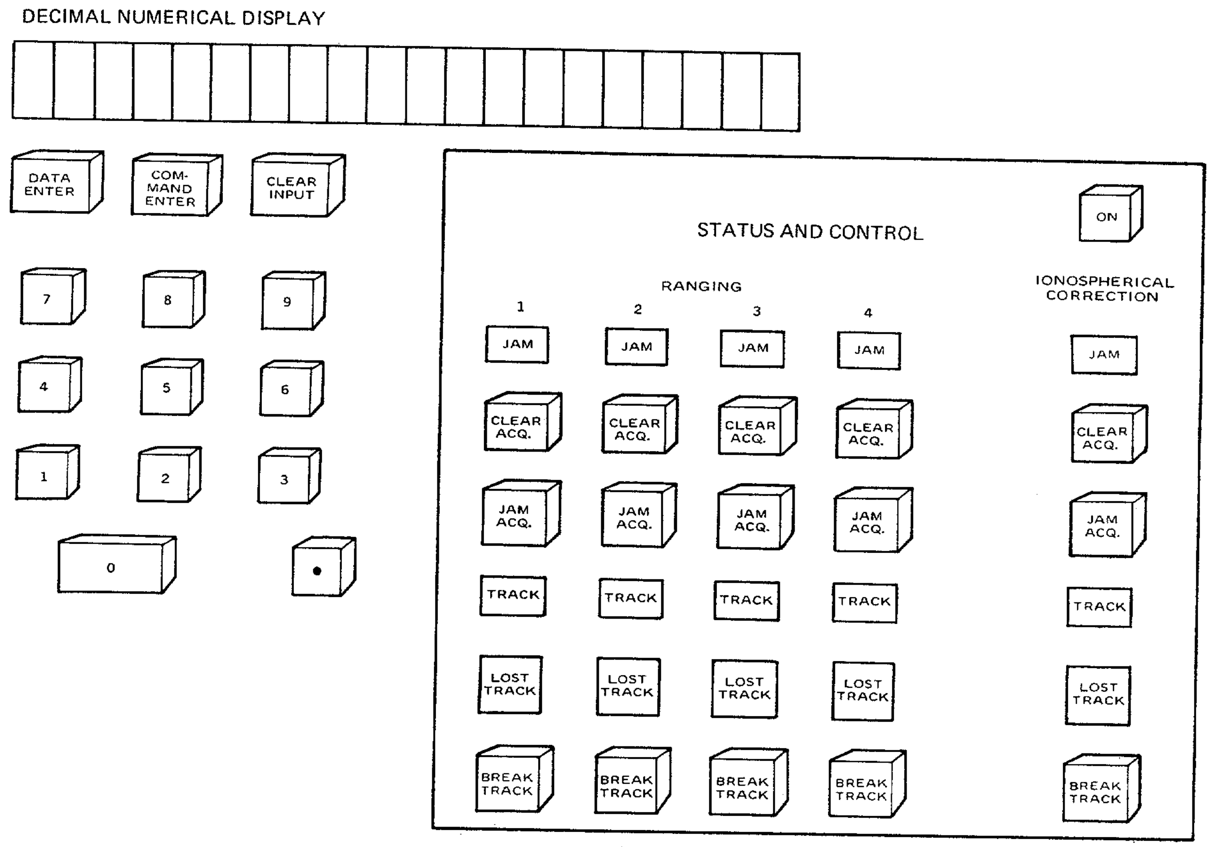
4.4.7.2 Equipment Configuration — Depending on the size of the aircraft involved, the category 5 user may possess an inertial (INS) platform. In such a case, the INS will be updated by the Timation navigation data with a Timation-only mode available as backup for the INS. All category 5 modes of Timation-aided navigation will employ a three-channel range-tone receiver. The computer thus requires an independent user altitude input readily available aboard the aircraft. The category 5 computer is not required to optimize the satellite constellation. This is done by a preflight programming of satellite constellations to be used during the flight.

4.4.8 Category 6 System

Category 6 is defined for general marine navigation.

4.4.8.1 Requirements — It is important to achieve satisfactory position-fixing performance at the lowest possible cost. Satisfactory performance does not require continuous position updates. A fix accuracy of 5000 feet is acceptable with 1500 feet easily available.

4.4.8.2 Equipment Configuration — The category 6 receiver consists of a single-channel range-tone receiver time-shared among three satellites. Fixes are available approximately every 6 minutes. The category 6 receiver is manually pretuned. An operator will select the satellites to be used by consecutively pretuning the phase-lock receiver to a minimum of three satellites. The navigation data obtained will be processed in a range-difference mode.

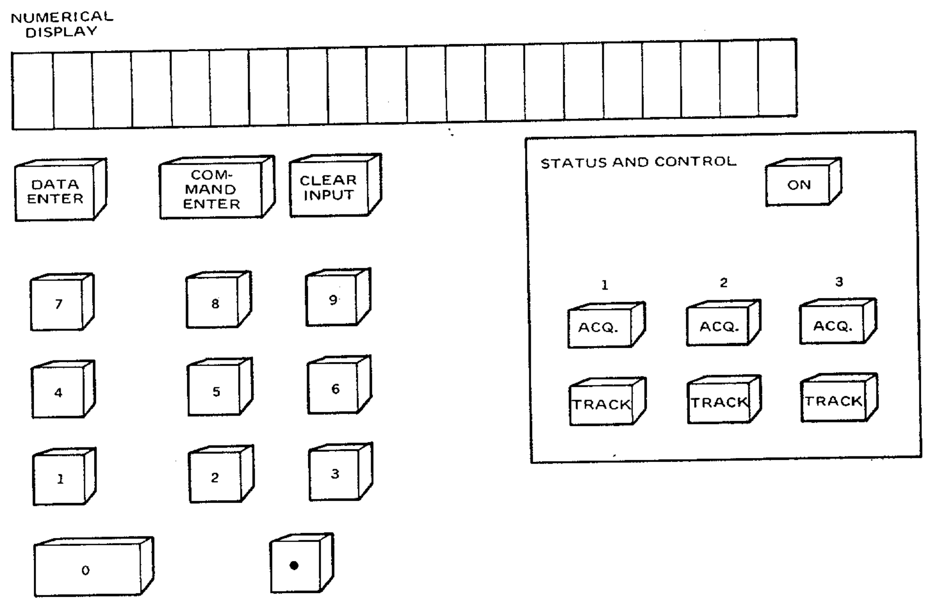


Notes: (1) Squares shown in relief denote lighted pushbutton switches.
(2) The panel shown accommodates 5 channels—as for a Category 1 user. For users requiring fewer channels, delete the appropriate columns of lights and pushbuttons from the status and control group.
(3) The status and control group must be part of the user equipment. The keyboard and numerical display group may be in a plug-in unit which remains at the user's home base. The appropriate location remains to be determined.

Fig. 4-36 - Timation antijamming system control panel for categories 1, 2, and 3 users

4.4.9 Control Panels

Only categories 4 and 6 possess control panels which are integral parts of the receiver. Other user category installations will, in general, best be served by an outboard panel appropriately mounted. Figure 4-36 shows the panel to be associated with the military users of categories 1, 2, and 3. Table 4-18 outlines its use. Figure 4-37 gives the category 5 control panel whose use is outlined in Table 4-19.



Notes: (1) Squares shown in relief denote lighted pushbutton switches.
(2) The status and control group must be part of the user equipment. The keyboard and numerical display group may be in a plug-in unit which remains at the user's home base. The appropriate location remains to be determined.

Fig. 4-37 - Timation system control panel for category 5 users

Table 4-18
KRP System Control and Display for Categories 1, 2, and 3 Users

Operation	Method	User Control			User Display			
		Event	Keyboard	Dedicated Pushbutton	Event	Nixie Tubes	Lighted Pushbuttons	Lights
On-off	Manual	On		x			x	
Set clock as 10 ms error	Alternates: (a) Radio reset from base transmitter, (b) Plug-in standard	Set clock command	x		Repeat command success or failure	x x		
Lead KPR Generator settings for each satellite to be used. Two per satellite.	Manual entry of 3 words per satellite PRF generator setting Pulse compression code generator setting Time at which the settings apply	Load command Enter data	x x		Repeat command Report data Complete report	x x x		
Checkout and individual tracker bias adjustment	Plug-in test set and manual entry	Command range measurement Enter correction Command individual checkout steps	x x x		Repeat command Report measurement Repeat correction Repeat commands	x x x x		
Obtain ephemerides	Alternates (a) load manually (b) Acquire satellites in NO JAM environment	Command enter data No-jam acquire command	x	x	Repeat command Repeat data Complete report No-jam acquisition Mode report Track mode report	x x x	x	x
Calibrate common mode bias	During track, enter known position and velocity. System will self-calibrate.	Calibrate command Enter data	x x		Track report Repeat command Repeat data Complete report	x x x		x
	Premission ↑							
	Mission ↓							
Acquire track and obtain fix	Manual command	Jam acquire command		x	Jam acquisition mode Track mode		x	x
Handover to another satellite	Automatic - programmed on basis of time schedule	None	-		Lost track report Acquisition status Track		x	x x
Reacquisition	Automatic	None	-		Lost track report Acquisition status Track		x	x x

Table 4-19
Tone System Control and Display for Category 5 Users

Operation	Method	User Control			User Display			
		Event	Keyboard	Dedicated Pushbutton	Event	Nixie Tubes	Lighted Pushbuttons	Lights
On-Off	Manual	On		x			x	
Checkout and peaking tests	Plug-in test set and manual data entry	Individual checkout commands	x		Repeat commands	x		
Calibrate common mode biases	During track enter known position and velocity. System will self-calibrate	Acquire		x	Acquisition mode		x	
					Track mode			x
		Command calibrate	x		Repeat command	x		
		Enter data	x		Repeat data	x		
					Complete report	x		
	Premission ↑							
	Mission ↓							
Acquire track and obtain fix	Manual command	Acquire command		x	Acquisition mode		x	
					Track mode			x

Section 5

SYSTEM IMPLEMENTATION PROGRAM

SUMMARY

The system implementation program is divided into the three main phases of concept formulation (CF), contract definition (CD), and engineering development (ED). The total program duration of the program is approximately 10 years beginning in the advanced development stage of the Timation concept and culminating in a completely installed, tested, and evaluated system.

The concept formulation phase lasts approximately 5 years, during which a demonstration system is developed, tested, and evaluated. The principal products of the CF phase are a revised technical development plan (TDP) and a data package, both of which form the basis for a decision to proceed with engineering development and the performance specifications for contract definitions.

The contract definition phase is estimated at 18 months during which an RFP (request for proposal) is issued, the proposals are evaluated, the contract decision is reviewed, and the source selection is approved.

The engineering development phase is a 3-1/2 year phase beginning with the awarding of ED contracts. The principal steps call for the installation and operation of the ground segment and the installation of an initial space segment for technical evaluation. This phase culminates with the completion of the space segment. A program schedule and milestone chart is shown in Table 5-1. The estimated cost schedule for concept formulation is shown in Table 5-2, and the contract definition and engineering development cost schedule is shown in Table 5-3.

5.0 SYSTEM IMPLEMENTATION PROGRAM

5.1 Concept Formulation

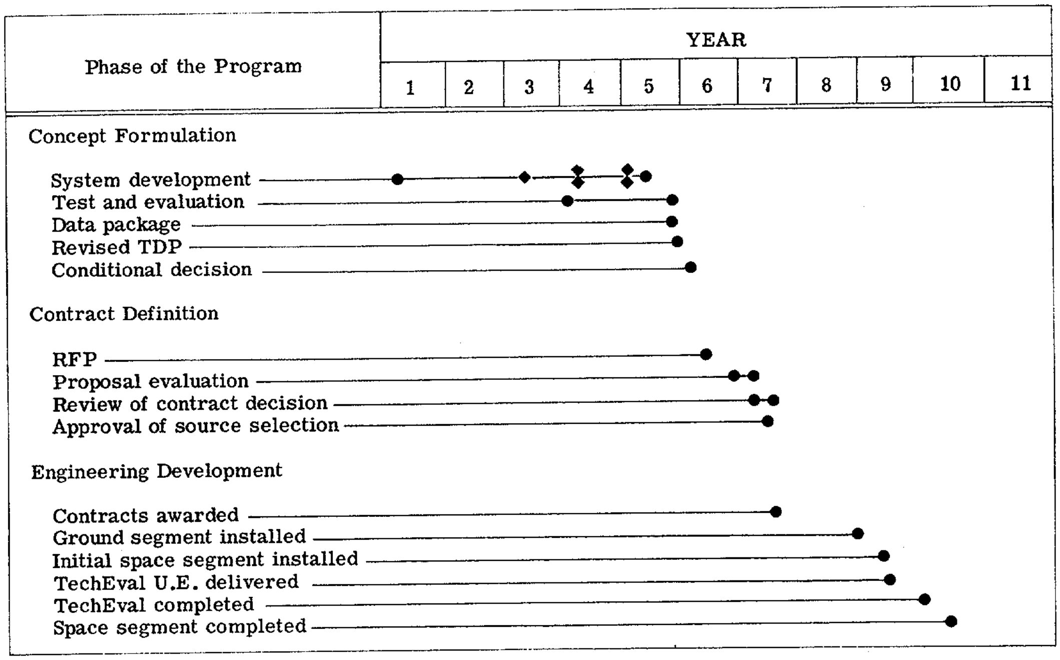
The concept formulation phase of the program is concerned with the implementation of a demonstration navigation system which will exercise the principal components of space segment, ground segment, and user equipments. This demonstration system is based on experiments, tests, and studies performed as exploratory and advanced development tasks. The demonstration system is designed to prove that (a) the technology needed is in hand, (b) the best technical approaches have been selected, and (c) cost and schedule estimates are credible. The present estimated cost schedule is contained in Table 5-2.

5.1.1 System Development

The principal elements of system development are engineering, spacecraft, ground stations, user equipment, control center, radiation hardening, and launch vehicles.

The engineering costs provide for studies, design, development, implementation, and evaluation of the demonstration system.

Table 5-1
System Implementation Program
Program Schedule and Milestone Chart



◆Denotes concept formulation satellite (launch).

Table 5-2
Concept Formulation Phase Cost Schedule
(millions of dollars)

Stage of the Concept Formulation Phase	YEAR 1	2	3	4	5		
System Development							
Engineering	1.2	1.2	1.2	1.2	1.2		
Spacecraft	0.2	2.0	8.0	5.0	—		
Ground stations	0.4	1.4	3.0	2.0	0.6		
User equipment	0.4	0.4	0.5	0.5	0.5		
Control center	0.1	0.1	0.5	1.5	1.5		
Radiation hardening	—	0.5	0.5	1.6	—		
Launch vehicle	—	—	—	6.0	—		
Test and Evaluation							
User equipment	—	—	1.0	1.0	—		
Facilities and ranges	—	—	—	0.5	0.5		
Data package	—	—	0.5	0.5	0.5		
Total	2.3	5.6	15.2	19.8	4.8	Grand Total	47.7

Table 5-3
Engineering Development Phase Cost Schedule
(millions of dollars)

Stages of Engineering Development Phase	YEAR						
	6	7	8	9	10	11	
Contract Definition	0.5	—	—	—	—	—	
Engineering	1.2	1.2	1.2	1.2	1.2	—	
Contracts	—	—	—	—	—	—	
Ground segment	—	—	—	—	—	—	
Ground stations	—	—	15.0	—	—	—	
Control center	—	—	5.0	—	—	—	
Space Segment	—	—	—	—	—	—	
Spacecraft	—	—	57.6	28.8	—	—	
Launch vehicles	—	—	38.0	19.0	—	—	
Operation and Maintenance	—	—	—	—	—	—	
Ground stations	—	—	3.0	3.0	3.0	3.0	
Control center	—	—	4.0	4.0	4.0	4.0	
TechEval	—	—	—	—	—	—	
User equipment	—	—	5.0	1.0	—	—	
Facilities and ranges	—	—	—	—	1.0	—	
Total	1.7	1.2	128.8	57.0	9.2	7.0	Grand Total 204.9

The spacecraft, ground stations, and user equipment are designed to simulate the proposed system to provide a firm basis for reliability analysis, system performance, and cost projections.

The control center will be simulated using existing computational facilities modified to perform in a semioperational mode.

The radiation hardening program is conducted concurrently with the system development program designed to meet the objectives of the advanced development objective (ADO) in determining the vulnerability of the systems and to form a basis for a firm tradeoff analysis compared to the initial vulnerability and hardening assessment of this plan.

The launch of Timation III (the launch of the prototype satellite for geodetic measurements) is based on the provisions of a SESP (Satellite Experiment Support Program) booster. The launch of Timations IV and V also assumes a SESP provided vehicle. A launch vehicle cost item has been included to provide for the launch of Timations VI and VII.

The availability of SESP launches is being investigated to determine the compatibility of scheduled vehicles with the requirements for the demonstration system. The Timation III launch has been approved by SESP.

5.1.2 Test and Evaluation

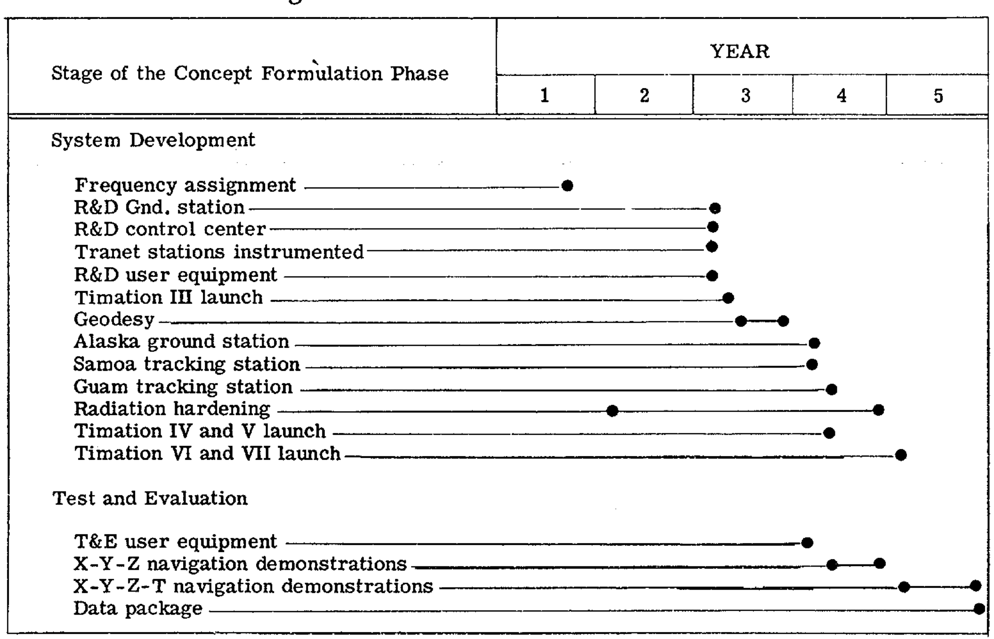
The test and evaluation phase is concerned with the operation and evaluation of the demonstration system. This phase involves user equipment and facilities and ranges for test and evaluation of system and user equipment. It includes a data package containing a total assessment of concept formulation. This evaluation forms the basis for the conditional approval to proceed with engineering development. This data package contains the elements for a revised TDP and provides the basis for contract definition performance specifications.

5.1.3 Concept Formulation Program Schedule

A program schedule and milestone chart for the concept formulation phase showing major milestones, development phases, and test and evaluation phases is shown in Table 5-4.

An experimental satellite (Timation III) will be launched in the first half of FY73 for geodetic studies, synchronization measurements, and user equipment development. This particular satellite is needed to confirm the calculated error budget and will be controlled by the R&D ground station and control center. Geodetic measurements and synchronization data will be obtained using suitably instrumented Tranet stations and Timation II stations equipped with developmental receiver systems.

Table 5-4
Concept Formulation
Program Schedule and Milestone Chart



The Alaska ground station and tracking stations on Samoa and Guam will be installed prior to the simultaneous launch of Timations IV and V. Test and evaluation user equipments will be available at this time for two-dimensional navigation demonstrations. The final launch in this phase of Timations VI and VII will provide a full four satellite constellation in view periodically for navigation demonstrations in three dimensions with clock update.

A data package contract will be initiated in the third year of the program to prepare an assessment of concept formulation and the performance specifications for contract definition.

5.2 Contract Definition

The data package will provide the basis for contract definition performance specifications. The time schedule provides 7 months from the completion of the data package to the issuance of RFP's for contract definition. This time schedule provides for data review, request for approval memorandum, program change proposal as required, a revised TDP, and the preparation of RFP's. A RFP will solicit planning proposals for engineering development and a firm proposal covering contractor's effort during contract definition.

Six months is scheduled for contract definition and 4 months for the evaluation of the proposal, the review of the contract decision, and the approval of the source selection.

5.3 Engineering Development

The engineering development phase is estimated at 3-1/2 years after the ratification and the awarding of contracts. The ground station segment consisting of four ground stations is scheduled for completion 18 months after contract award. The initial space segment consisting of two planes of nine satellites each will be installed 24 months after the awarding of the contract. The first plane of satellites will be launched at month 18 in the engineering development phase for initial test and synchronization of the ground stations and the second plane at month 24. Technical evaluation (TechEval) user equipments are scheduled for delivery at month 24. TechEval is scheduled for completion, utilizing the ground segment and initial space segment, at month 30. The total space segment schedule of completion is estimated for month 36. The engineering development phase cost schedule is contained in Table 5-3.

Section 6

TEST AND EVALUATION PLAN

SUMMARY

The test and evaluation plan outlined in this chapter addresses the test and evaluation (T&E) program for the concept formulation (CF) phase. Only technical evaluation (Tech-Eval) and operational evaluation (OpEval) phases of the engineering development phase are scheduled for definition in the data package and for specification in the contract definition (CD) phase.

The objectives of the T&E plan are to:

1. Demonstrate system performance
2. Establish performance baselines for subsequent TechEval and OpEval
3. Confirm equipment design characteristics and provide reliability and maintainability statistics
4. Develop operating procedures
5. Identify the elements of a revised TDP

The elements of this plan are:

1. A constellation consisting of two planes of two satellites, oriented such that four satellites are in view periodically to demonstrate real-time three-dimensional navigation and clock up-date.

2. A ground segment consisting of a ground station, two tracking stations, and a control center. These ground stations and tracking stations provide the necessary data for orbit determination. In addition the ground stations provide command and control functions for the satellites and monitor telemetry and calibration from the satellites. The control center is the focus for the collection of data, orbit determinations, ephemeral predictions, and system performance evaluation.

3. Demonstration User Equipments. A small number of user equipments are scheduled for demonstration and evaluation. Since this equipment must meet the most stringent requirements in terms of accuracy and dynamic performance, the aircraft (a/c) user equipment has been selected for the primary demonstrations. User equipments for other classes of users such as surface ships can logically be derived from the a/c user equipment. The foot soldier or field user equipment is a second category of demonstration devices because it is not a logical derivative of the high-accuracy a/c user equipment.

4. Facilities and Ranges. Calibrated facilities and ranges are required for navigation demonstrations. The specifications for a range such as the one of the Naval Air Test Center, Patuxent River, Maryland, indicate that this theodolite-radar range will be adequate for a/c navigation demonstrations.

6.0 TEST AND EVALUATION PLAN

6.1 Test and Evaluation Objectives

The objectives of the T&E program are to:

1. Demonstrate system performance
2. Establish performance baselines
3. Confirm design characteristics
4. Develop operating procedures
5. Identify problem areas

6.2 Test and Evaluation Elements

To meet the objectives of this T&E plan it is necessary that all of the elements of a system be included in a demonstration system.

6.2.1 Space Segment

The launch of the experimental satellite (Timation III) will provide a test bed for satellite components, an evaluation of launch performance in terms of the accuracy of orbital parameters, a study of geodetic characteristics of medium-altitude orbits, an evaluation of system synchronization techniques, and a tool for user equipment development. The Timation IV and V satellites will provide a capability for the initial navigation demonstrations. The use of two fully instrumented satellites will allow two-dimensional navigation tests and an evaluation of all of the elements of a complete system. The subsequent launch of Timation VI and VII will provide a total system evaluation where navigation in three dimensions with clock update is possible. The launches of Timations III, IV, and V are predicated on vehicles provided by SESP. A figure of 6.0 million dollars has been included for the launch of Timations VI and VII.

6.2.2 Ground Segment

The initial ground segment for the Timation III phase will utilize an R&D ground station for the operation and control of the satellite. Suitably instrumented Tranet stations and refitted Timation II stations will provide the data for orbit analysis and related geodetic studies. A simulated control center will consist of a central analysis facility at NRL. This center will use those computational facilities now available for the Timation II work. The Alaska ground station and the tracking stations at Guam and Samoa will complete the ground segment and be available for the subsequent operations with Timations IV, V, VI, and VII. An enhanced control center with near real-time data links will be implemented for the navigation demonstration phase of the T&E program.

6.2.3 Demonstration User Equipments

The studies of the Timation III satellite will use equipments derived from the user equipment development program. Procurements of a suitable number of developmental user equipments are scheduled for the navigation demonstration phases of the T&E program.

Because this equipment must meet high accuracy and severe dynamic performance specifications, the first phases of the user equipment development program are aimed at the high-performance a/c user. A test a/c, suitable for antenna system modification, will be needed for a/c user demonstrations. Several other classes of user equipment may be derived from the a/c unit.

A second category of user equipment, which is not a derivative of the a/c user equipment, is the foot soldier or field user. This special class of user equipment has been selected as the second type of developmental user equipment for navigation demonstrations.

6.2.4 Facilities and Ranges

For a/c user demonstrations and instrumented range, such as the one at Naval Air Test Center at Patuxent River, Maryland, will be used. This theodolite-radar range can be calibrated with the navigation system by colocating with fixed receiver systems. Successive a/c flights through the range will provide data for absolute and relative navigation demonstrations. Four satellites will be in view simultaneously for two 1-hour periods each day at the Patuxent River latitude.

Section 7

INTEGRATED LOGISTIC SUPPORT CONSIDERATIONS

7.0 INTRODUCTION

The initiation of integrated logistic support (ILS) considerations into the Timation navigation program will consist of forming an ILS planning group. This group will formulate a "plan for use" for the system, which will set goals and the philosophy for addressing the elements of ILS throughout the system life cycle. The major elements to be addressed by this planning group are reliability, maintainability, and human factors. The methodology of approaching these major elements is discussed below.

7.1 RELIABILITY PLAN

The reliability program will be formally initiated at the outset of the preliminary system design phase. The ILS planning group will determine the initial reliability goals. These will be combined with experimental results and analyses to produce reliability estimates. The estimates and technical data will be used in the system engineering design phase. The refinement of subordinate reliability goals will be coordinated by the planning group between the system's major component development, i.e., spacecraft, ground station, and user equipment, to establish the total system reliability.

The contractual system data package to be used for contract definition will have the reliability inputs that are based on mathematical models and experimental results, amenable to reliability analysis, formulated during the engineering design. These derived estimates and predictions will be used by management to formulate and, subsequently, refine the reliability program plan. The technical development activity will implement this plan by use of a management policy and practices guide, which will be approved by the program manager. The project, during the phase of engineering development, will construct a demonstration system for test and evaluation that will demonstrate the capability of the system to meet requirements. The results of the demonstration system tests will be used to update the allocated reliability goals and to provide the first empirical assessment of system reliability. Upon entering into the test and evaluation phase a failure and technical problem program will be instituted to report and analyze all failures and technical problems during the system demonstration, to determine and implement corrective action, and to verify the correction. The end result of the test and evaluation phase will be a data package that will be used to contract for the total system production.

System implementation will require the formulation of detailed production design specifications, including equipment and system performance parameters. The final reliability prediction will be made in these specifications and the reliability goals set. The system will then be produced, concurrent with operational evaluations, which will demonstrate the actual initial system reliability, prior to installation of the user equipment throughout the services.

7.2 MAINTAINABILITY PLAN

When the ILS planning group is formed, maintainability philosophy shall be established to provide emphasis on this ILS aspect throughout the system life cycle. Included in the technical data for system engineering design will be the results of the maintainability analysis, in the form of a maintainability prediction. The prediction will be used to assign maintainability goals for engineering design.

Initiation of engineering design will require the formulation of a maintainability program plan. Mechanization of the plan will be in the same manner as the reliability program plan, which is governed by the management policy and practices guide. During this development phase, an interim maintainability prediction will be made as system data are generated and, if necessary, adjusted with feedback from prediction analysis. The first empirical data on system operation for increased credibility in maintainability prediction will be provided by the demonstration system. The completion of test and evaluation of the demonstration system would be the milestone marking a system data package for systems implementation, which will contain the most credible maintainability prediction.

System implementation will commence with detail design for production on which a final maintainability prediction will be based. Specifications for support and test equipment, supply support, transportation and handling, technical data necessary for maintenance, personnel and training requirements, and management data for system operation will be generated at the end of this phase, which would include operational evaluation results. These specifications and system technical data will permit production of the defense navigation satellite system (DNSS), and subsequent installation and use throughout the services.

7.3 HUMAN FACTORS

Consideration of the human factor element of ILS planning will also be initiated during the preliminary design phase. An analysis of man's role in the system will be performed, and a sound program will be formulated for the integration of organizations with human factors capabilities, into the program in an effective manner.

The analysis of man's interface with the system will provide a foundation for system engineering design. At the outset of engineering design the following will be assembled: a complete file of human factors documents or a source reference of directives, standards, specifications, and guides under which participating activities and the program manager will operate. Smooth and timely communications between the program manager and supporting activities will be established to define assignments of organizational responsibility. Engineering tasks for supporting activities and the technical development activity to meet human factors objectives will be formulated. The objectives will be verified during the test and evaluation of the demonstration system, and will then be used to generate the equipment specifications for the contractual data package. The detailed design performed before system production, will be utilized to solidify the human factors specifications, used for the operational evaluation equipment, and subsequently for the mass production of equipment for installation throughout the services.

Appendix A

ALTERNATIVE GROUND STATION AND SATELLITE CONSTELLATION

A.1 INTRODUCTION

The ground station locations used in the basic plan were constrained to be located only on U.S. territory. However, if different criteria are used, such as, "Four ground stations may be used, one of which may be located on foreign soil for which the U.S. has base rights," a different resultant system is obtained. Figure A-1 shows the coverage from the ground stations if one such station is located in the Chagos Islands, in addition to the four U.S. sites previously mentioned (Alaska, St. Croix, Guam, and Samoa). The noncoverage area is greatly reduced, and the Guam station is redundant to Chagos and Samoa; therefore, it may be eliminated.

If the satellite orbital altitudes are raised to provide 12-hr periods, the coverage from each station is also increased. As shown in Fig. A-2 the coverage from each station combines to leave, only a small area uncovered.

The 3-by-9 constellation with 8- and 12-hr orbital periods are compared by their position dilution of precision (PDOP) in Fig. A-3. It can be seen that both constellations are nearly the same for percentages less than 90%, but above that figure the 12-hr period becomes appreciably better. Figures A-4 and A-5 show the PDOP as a function of latitude and time, and again the 12-hr period has considerably smaller values of PDOP with noticably smaller peaks.

The maximum time a 12-hr, 53-degree inclination satellite can be out of view of the ground stations used is approximately 140 minutes. During this time the satellite clock having an assumed stability of 2pp10^{12} will incur an RMS error of 17 nanoseconds. Figures A-6 and A-7 were plotted using a satellite in this worst position; the other satellites were in their corresponding positions and the error budget was 10 nanoseconds. The case depicted in Fig. A-6 involves a user having a stable clock—one which is updated at the times of good GDOP's and is stable enough to maintain the stability through the periods of high GDOP's. The case in Fig. A-6 corresponds to the use of the values of GDOP at the bottom of the GDOP (or PDOP) envelope, Fig. A-5.

For the worst case of a continuously corrected clock the maximum PDOP at each latitude was used. Figure A-7 is the result of using these PDOP values, the 10-foot error budget, and the RMS clock errors where they apply.

Figures A-8 and A-9 are the combined charts showing the error contours for both the 8-hr 55-degree inclination constellation with the four U.S. stations and the 12-hr, 53-degree inclination constellation with the Guam station moved to Chagos.

From these Figs. A-8 and A-9 it is seen that the Chagos-12-hr satellite provides an improvement factor in maximum error contours of approximately 1.3.

The RMS clock errors were used in the following manner: The earth was divided into 30-by-30-degree segments. (At low and high latitudes the longitude interval was 60 degrees.) At each segment corner the number of satellites was found that could be viewed above a 10-degree mask. If some of these satellites had been out of view from a ground correction station, this information was noted. An estimate of each clock error was obtained by multiplying the stability factor by the ellapsed time since the satellite was last visible. This additional clock error was orthogonally added to the basic 10-nanosecond error to arrive at a total error for each satellite. As a simple means of estimating the navigational error for an observer, the average of the total errors of all satellites visible to the observer was computed and used in connection with the dilution of precision factor. This method of approximation works well when only a small percentage of the visible satellites has increased clock errors and the increased clock errors are not large compared to the basic error size.

A.2 SUMMARY

In summary, one can improve the system accuracy by a factor of 1.3 by moving the Guam station to Chagos and by increasing the constellation to a 12-hr period.

While the accuracy contours show errors which are considerably below the requirement, it should be recognized that other factors exist that will increase the errors shown. The shown errors assume that the navigator observes all satellites above the minimum 10-degree elevation mask. If fewer satellites are observed, the fix errors will increase.

Experience with error budgets on other systems (Minitrack, Baker-Nunn cameras, Hewitt cameras) shows that the predicted error is approximately one-half the observed error. This appendix has therefore aimed at finding a predicted error twice as good as stated in the requirements.

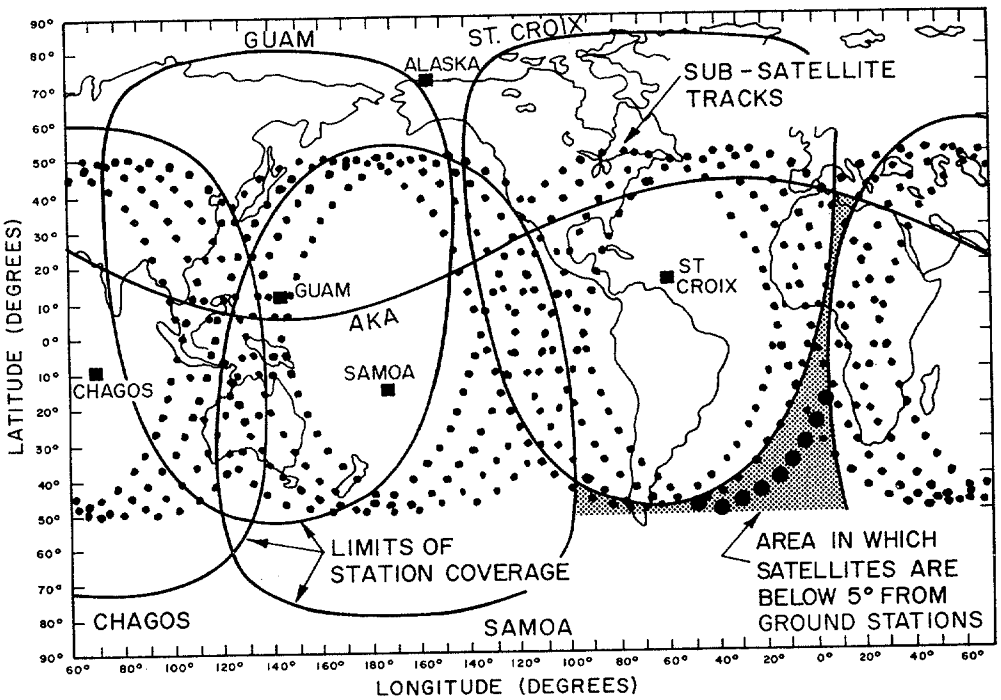


Fig. A-1 - Chart showing the station coverage and subsatellite track for 8-hr, 50-degree orbits

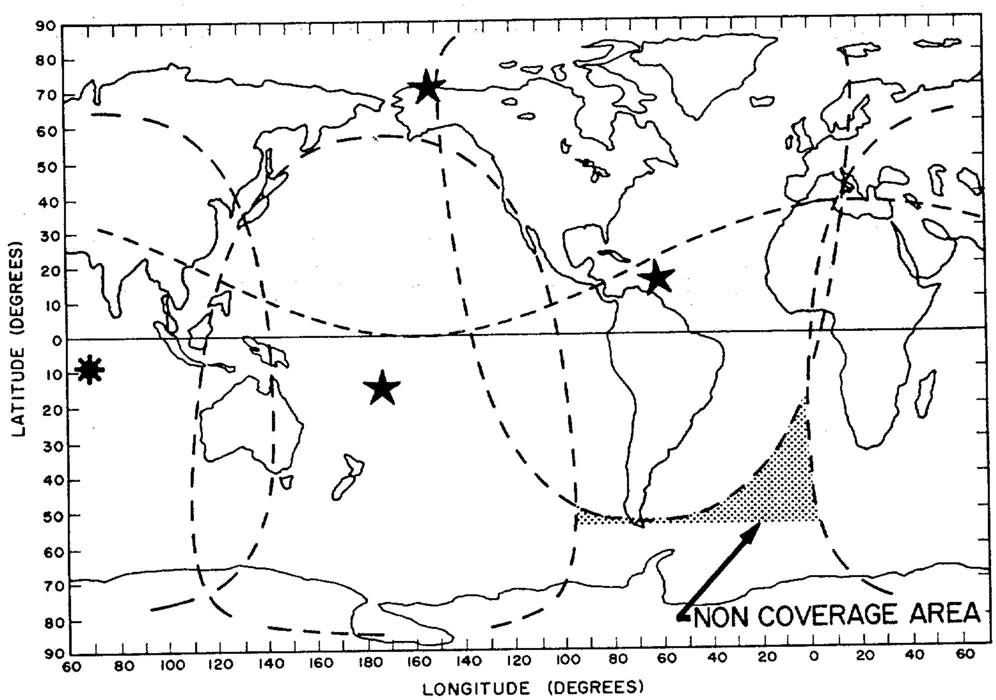


Fig. A-2 - Station coverage and noncoverage for locations shown and for 12-hr, 53-degree inclination orbits

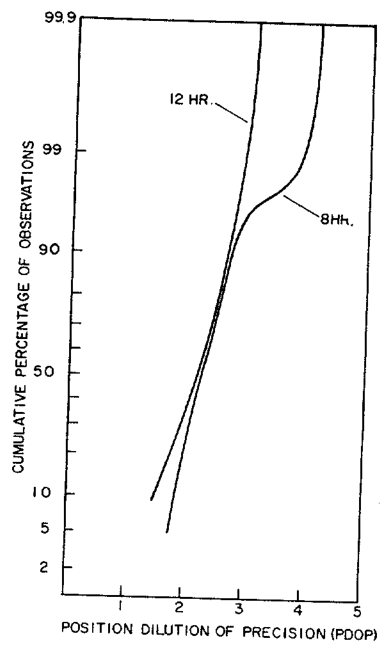


Fig. A-3 - PDOP comparisons of 8-hr and 12-hr orbits using 3-by-9 constellations, 10-degree masks, 53-degree inclination for the 12-hr orbit, and 55-degree inclination for the 8-hr orbit

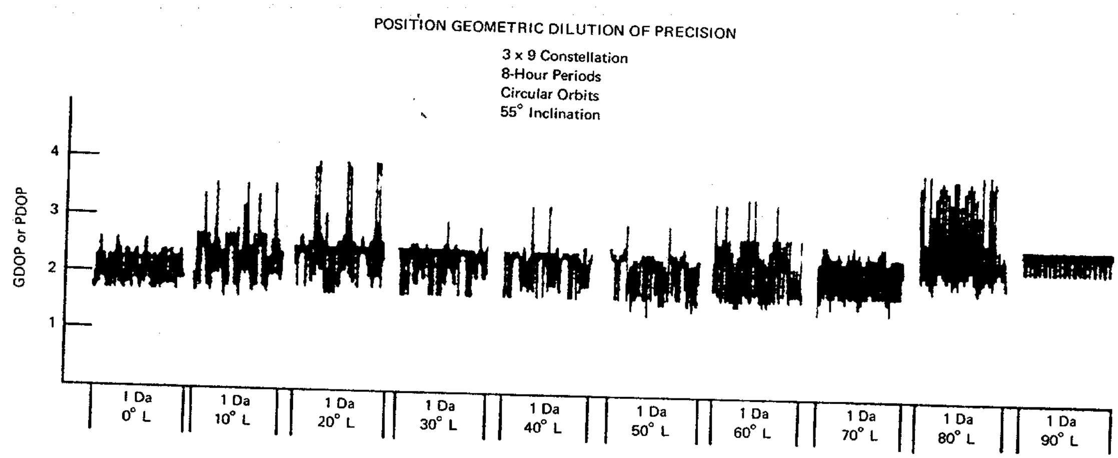


Fig. A-4 - Position geometric dilution of precision for the 3-by-9 constellation using 8-hr periods and circular orbits at a 55-degree inclination

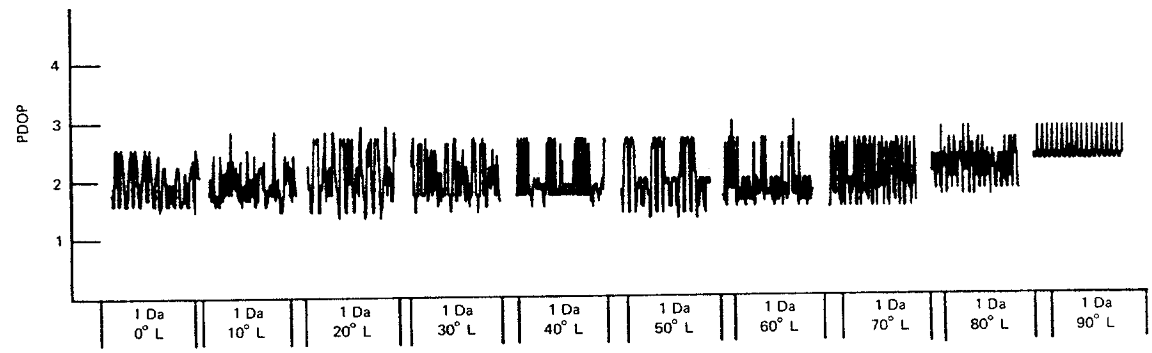


Fig. A-5 - Position geometric dilution of precision for the 3-by-9 constellation using 12-hr periods and circular orbits at a 53-degree inclination

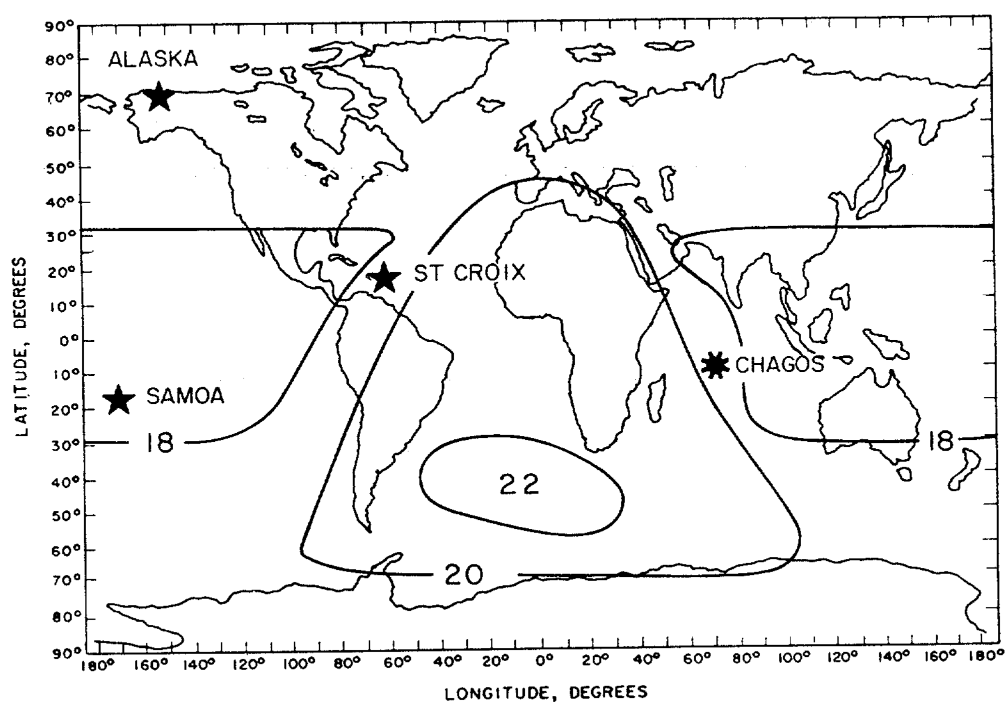


Fig. A-6 - Error contours in feet for 3-by-9, 53-degree inclination, 12-hr constellation. The ground stations are shown for good user clock.

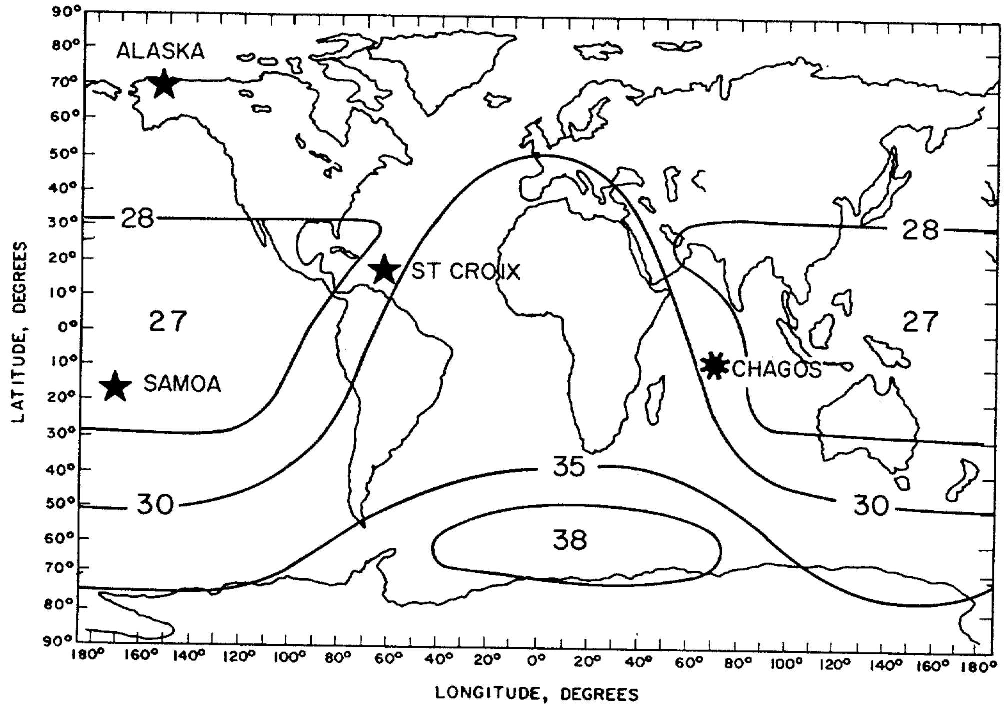


Fig. A-7 - Error contours in feet for a 3-by-9 constellation of a 53-degree inclination for 12-hr satellites. The ground stations are shown for a continuously updated user clock.

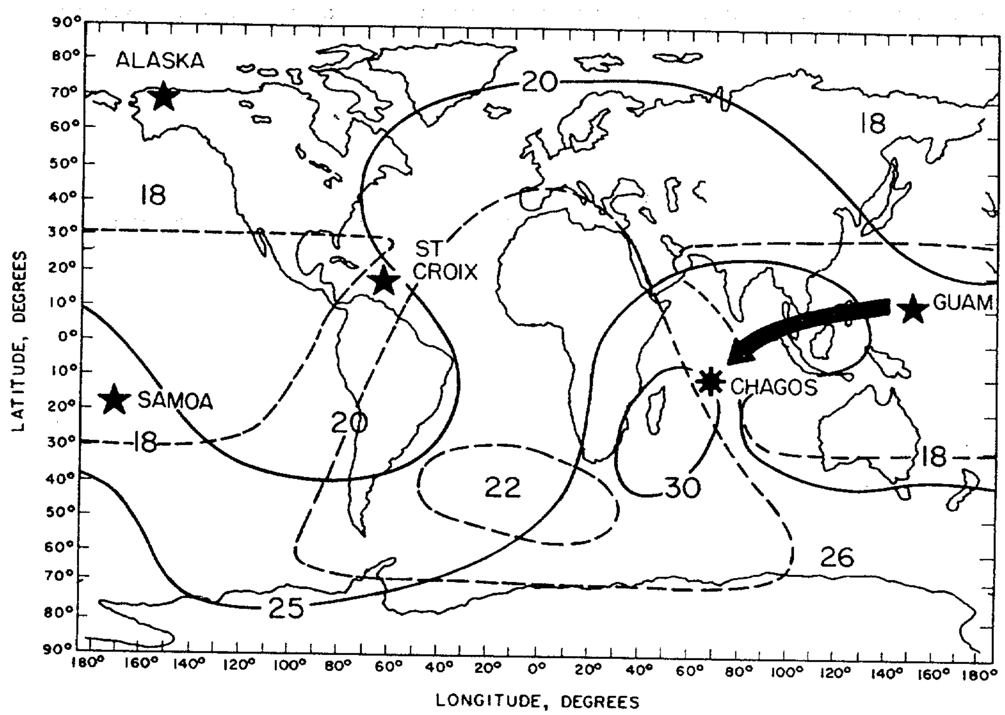


Fig. A-8 - Error contours in feet using Guam and constellation of 8-hr satellites (solid lines) using Chagos (vice Guam) for 12-hr orbits (dashed lines) for a good user clock $1_{pp}\,10^{11}$ stability

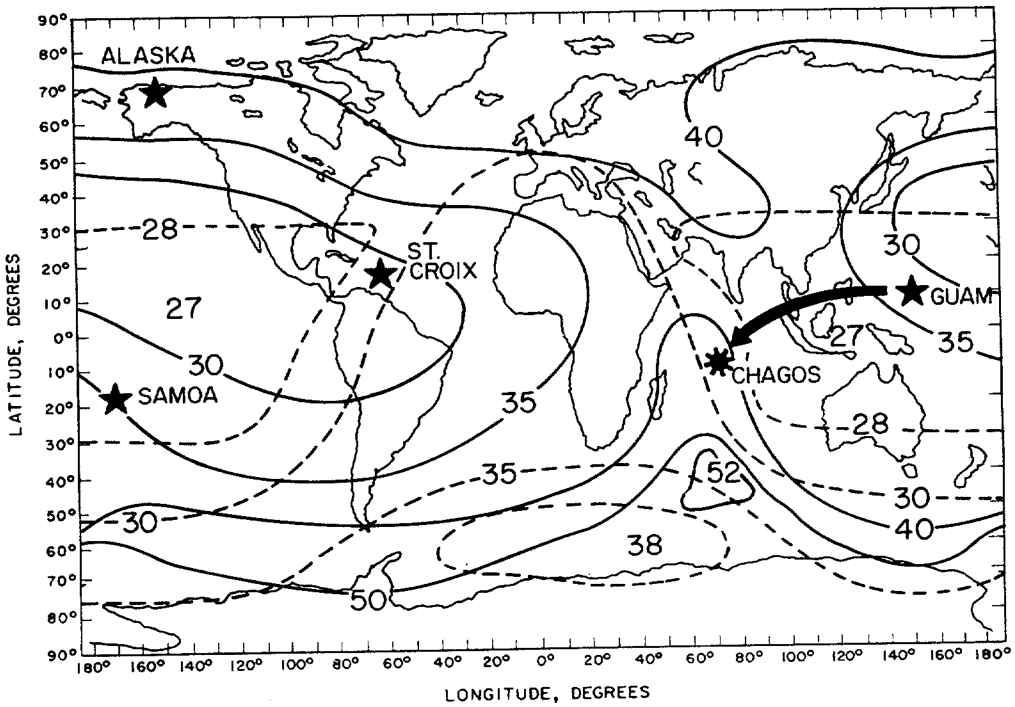


Fig. A-9 - Improvement in error contours resulting by moving Guam station (solid lines) to Chagos (dashed lines) and by increasing orbital periods from 8 hr to 12 hr continuously updated user clock (four parameter fix)

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13. ABSTRACT (Unclassified)

TIMATION (Time Navigation) is a technological development program designed to study, identify and develop the critical components and make measurements necessary to describe a navigation system which meets JCS requirements.

The purpose of the navigation system development plan is to describe the technical parameters, tradeoffs, experiments, and costs encompassing the implementation of a satellite position fixing and navigation system that meets the requirements promulgated by the Joint Chiefs of Staff (JCS) Navigation Study Panel in 1968. This navigation system provides continuous all weather instantaneous readout to an unlimited number and variety of users on a worldwide basis.

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Precise time						
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Stable crystal oscillators						

DD FORM 1 NOV 65 1473 (BACK)
(PAGE 2)