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DISASTER WARNING SATELLITE STUDY

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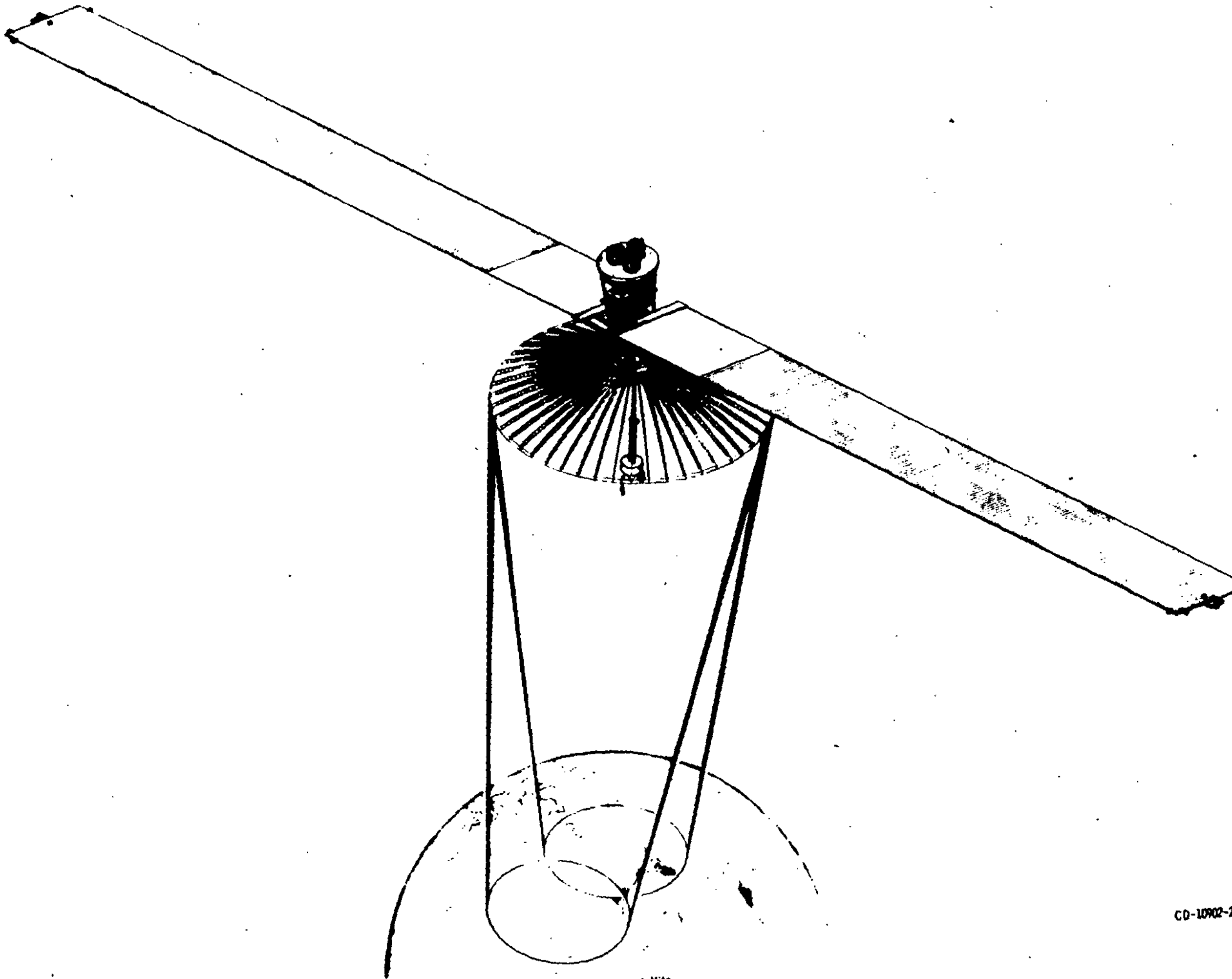

Disaster warning satellite

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DISASTER WARNING SATELLITE STUDY

I. PROGRAM SUMMARY

The Disaster Warning Satellite represents a practical use of space technology to alert the nation's citizens to threat of disaster from tornado, hurricane, surge flood, forest fire, tidal waves and to coordinate disaster relief efforts. The proposed operational system consists of two satellites in geostationary orbit, approximately 30 degrees apart in longitude. A single satellite can be used if one percent annual downtime due to eclipse can be tolerated. The use of two satellites of equal capability supplies redundancy and enhances flexibility to serve diverse areas of the country simultaneously with different priority level messages. The satellites are used to communicate between ground stations as well as to transmit disaster data from the ground control stations to home receivers. The term "home receivers" used throughout this document includes those placed in schools, offices, hospitals or anywhere else the service is desired. The spacecrafts, when providing ground control station to home receiver messages, are repeaters and message format is prepared on the ground. The home receivers are designed for continuous and/or emergency operation. In the continuous mode, all messages beamed to a specific climatological area will turn the home receiver on; whereas, in the emergency mode only messages coded "high priority" will actuate that home receiver. By employing time sharing, 192 discrete areas of the 48 adjacent states can be served with four one-minute messages every hour. Full time transmission to 16 areas is possible.

The main features of a spacecraft to perform this mission are shown in figure 1. The 10 foot high center body supports power distribution and control, telemetry, command, attitude control systems and other electronic equipment. Fixed to one end of the center body is the 19 foot diameter "wrap rib parabolic reflector" which is the main downlink antenna. After deployment, the transmitting antenna is accurately pointed at the earth by controlling the attitude of the center body.

Power for the spacecraft, propulsion system, and the transmitters is supplied by a flexible roll-up type of solar array, which, when extended, measures 130 feet from tip to tip. The array is sized to provide 5.0 kilowatts of power at the end of five years. Synchronous orbit can be achieved, for the 1562 lb. spacecraft, by means of several ascent techniques. Direct ascent to synchronous altitude using a Titan 3C vehicle is described in this report as well as a "spirial out" mission technique using a Thor/Delta/electric propulsion system. Cost savings can be effected by using the thrust augmented Thor/Delta to place the spacecraft in an initial low altitude earth orbit and an electric propulsion system to raise the spacecraft altitude to the geostationary orbit.

The proposed spacecraft is compatible with both the Titan 3C direct ascent mission and the spiral-out Thor/Delta mission. The spacecraft has been studied to the depth required to assure feasibility. However, the design is preliminary and subject to re-evaluation and revision. The spiral-out mission is a lower cost approach than the direct ascent mission.

Details of the uplink and downlink operational system, home receivers, spacecraft, launch vehicles and method of orbit attainment are presented in the technical plan.

It is concluded from this study that the high power spacecraft technology being developed by the Lewis Research Center (LeRC) has application to the

Disaster Warning Satellite. Development of the high power transmitting tubes, high voltage solar arrays, ion thruster systems, and the solar array slip ring assembly proposed for this spacecraft application have been underway at Lewis Research Center for a number of years.

II. INTRODUCTION

The President's Space Task Group, in their study on the use of space technology, emphasized the use of space technology in such areas as earth resources, communication, navigation, national security, science and technology, and international participation. The Disaster Warning Satellite System is a program that satisfies the recommendations of the Space Task Group.

The Disaster Warning Satellite System will provide NOAA* with an independent, mass communication system for the purpose of warning the public of impending disaster and issuing bulletins for corrective action to protect lives and property. The system consists of three major segments. The first segment is the network of state or regional offices that communicate with the central ground station; the second segment is the satellite that relays information from ground stations to home receivers; the third segment is composed of the home receivers that receive information from the satellite and provide an audio output to the public. The ground stations required in this system are linked together by two, separate, voice bandwidth communication channels on the Disaster Warning Satellites so that a communications link would be available in the event of disruption of land line service.

The home receivers are very similar to current home FM receivers (except that higher frequencies are used). The satellites systems would use to advantage the technology that the Lewis Research Center has been actively pursuing for the past four years in support of high power, direct broadcast satellite development efforts.

* National Oceanic Atmospheric Agency (formerly ESSA)

The satellite system combines the technologies of High Power Communication, Electric Propulsion, and Space Power into an application for the benefit of mankind. In the area of High Power Communication, LeRC has performed detailed studies of microwave amplifiers and considerable tube development work. The results of these studies and development efforts are the basis for the recommendation that the Cross Field Amplifier (CFA) be the transmitter output amplifier for this communication satellite. Its weight and size compared to other possible microwave amplifiers are one third and one fifth, respectively. Amplifier efficiencies of 75% are anticipated. The CFA has the characteristic of being able to operate at levels as much as 20 dB below saturation, which permits operation over the design range of 300 watts per channel to 2.5 kilowatts per channel by simply changing the drive signal level.

Lewis Research Center studies of communications system concepts and home receiver requirements have resulted in a selection of 900 MHz as the transmission frequency from satellite to home receivers. This frequency is chosen on the basis of trade-offs of building attenuation of transmitted signals for home receivers, versus the size and complexity of spacecraft antenna systems.

The proposed use of electric propulsion for orbit raising in order to use a low cost launch vehicle system has resulted from considerable technological effort by LeRC in the area of electric propuslion. The SERT II mission has demonstrated that electric propulsion has been brought to a mission-use status, such as that required for the proposed spacecraft. In addition, high specific impulse electric propulsion systems for attitude control have been under development at LeRC for some time. These systems permit five year life times to be achieved with systems much lighter in weight than existing attitude control systems.

In the field of power systems, studies by LeRC show that high voltage solar array systems offer several advantages over a low voltage array with conventional power conditioning. Considerable weight reduction for power conditioning and heat rejection systems can be realized using the high voltage array. Also, higher power utilization efficiencies and increased system reliability are achieved. Concomitant development has been pursued on liquid metal slip rings which conduct electrical power from the solar array to the spacecraft efficiently and provide a very low friction joint thus minimizing perturbations of the attitude control system.

The technical plan which follows describes the results of preliminary studies in the major areas of: the total system concept of the Disaster Warning System; consideration and trade-offs in launch vehicle cost and time required to achieve geostationary orbit; the design and development of the spacecraft; and the modular design of a high voltage solar array.

Only one spacecraft configuration and two methods of injecting the spacecraft into the desired synchronous orbit are discussed in the body of this report. The system described provides disaster warning coverage for the contiguous 48 states. A description of the spacecraft modifications required to provide disaster warning coverage to Alaska and Hawaii are described in Appendix A. Other Appendices briefly describe other approaches to configuring the spacecraft or mission. These approaches are only briefly discussed since they do not offer realistic alternatives. Launch vehicle configurations other than those discussed in the body of the report also are discussed in an Appendix. These other launch vehicle configurations represent variations of the two concepts discussed in the body of the report. Some of the vehicle configurations, however, may not be available for this mission.

III. TECHNICAL PLAN

1.0 Disaster Warning System Description

1.1 Implementation.

- The basic disaster warning system consists of a central ground station for encoding and transmitting routine messages to the spacecraft at 6.0 GHz, 48 regional ground stations for transmitting emergency messages to the spacecraft, a satellite system which can receive the signal, amplify it and retransmit at 900 MHz and small UHF-FM receivers placed in homes and other locations around the country. Except for eclipse conditions, continuous coverage of continental United States can be achieved with a single spacecraft transmitting two beams, using separate feeds to one 19-foot diameter parabolic antenna. Each beam will have four channels in a 4^{o} half power beam width (HPBW). Requirements for redundancy and the effect of eclipses on the operation of a single spacecraft necessitates the use of two spacecraft.

Eclipses result from the spacecraft passing through the shadows of either the earth or the moon. In synchronous orbit, the satellites pass through the earth's shadow for a portion of each day for two 52 day periods each year. These eclipse periods are centered at the spring and fall equinoxes. Maximum eclipse duration at the center of the periods is 72 minutes. Total eclipse time per year is less than 1 percent. Placement of two satellites at stations at least 20^{o} apart in longitude prevents both satellites from being in earth eclipse at the same time.

At times, both spacecraft can be simultaneously affected by the lunar shadow. This problem area needs further study; however, preliminary indications are that in a five year time span the two satellites are both in the lunar shadow on 1 occasion for 1 hour. It should be noted also that no study was undertaken of the effect of a partial lunar eclipse. Figure 2 illustrates the two-satellite system and the coverage of the continental United States by the two beams of a spacecraft.

Each spacecraft has two transmitting tubes and each tube carries four channels for a total of 16 continuous channels available under normal conditions. Capability exists for shifting 8 channels from one spacecraft to the other. Capability also exists for the full power of a tube to be directed into a single channel and increase the radiated power in that channel by 9 dB. This increase in radiated power permits more intensive coverage of a small regional area when required.

During routine operations, in order to supply information of very localized interest, each of the sixteen continuous channels available on the two spacecrafts can be time shared by several sections of the country. If twelve areas time share each of the channels, 192 discrete areas can be reached with a one-minute message each quarter hour. Table 1 summarizes the implementation to achieve localized coverage.

TABLE 1

Localized Coverage Capability

Item	Unit Quantity	Total
Spacecraft	Each	2
RF Beams	2/Spacecraft	4
Channels (Full Time)	4/Beam	16
Sub Channels (Time Shared)	12/Channel	192

The messages can be received by small UHF-FM audio home receivers as depicted in figure 3. Each home receiver has a built-in digital channel address decoder. The home receiver normally operates in a standby mode, where it monitors one of the sixteen channel frequencies. Upon receipt of an address on the monitored channel that matches the address set into its decoder, the home receiver is automatically switched to operate mode. This means that the audio output is switched on and the ensuing emergency message is heard. At the end of the message, another transmitted code automatically switches the home receiver back to standby. Operation of the unit is on commercial, 115 volt, 60 cycle power. In case of power failure, batteries would automatically supply power to operate the home receiver.

For localized coverage of the 48 adjacent states, the country can be divided into 192 climatological areas. Each area is assigned a three digit address number identifying it, similar to the telephone area code system. At the beginning of each message transmitted from the satellite, the numerical addresses of the areas that are to receive the message and the priority level of the message are transmitted. When a home receiver in the standby mode receives its own address, and the message is high priority, the unit switches to the operate mode, and the message being transmitted is heard.

Messages are broadcast at three levels of priority. At priority level 1, warning of a disaster in progress, the single channel covering the affected area receives the full saturated power output of one of the two satellite transmitter tubes, 2.5 KW. All home receivers in the affected area are turned on by the transmitted code. At priority level 2, warning of potential disaster, the full power capability of a satellite is split between its two transmitter tubes, with each tube having a single channel output of 1.5 KW. Level 2 transmissions will also be coded to turn on all home receivers in the affected area since the potential for injury to persons or property damage is quite high. At priority level 3, transmitting routine weather information and local interest data, the two transmitter output tubes on each spacecraft are operated in their linear range, 1.2 KW total rf output and four output channels from each tube (8 channels per spacecraft). During level 3 transmissions, because they are of only limited interest, the message is heard only on home receivers switched to the "local" or "continuous" modes. The output spectrums transmitted from each spacecraft for priority level 3 transmission are shown in figure 4. The number of different messages sent during level 3 transmission is expanded for each spacecraft from 8 to 96 by time sharing each continuous channel into 12 subchannels (expansion from 16 channels to 192 subchannels for both spacecraft transmitting). The 12 messages are broadcast consecutively, each message occupying nominally one minute. Thus, each subhcannel, or each area, receives a message about once every quarter hour during priority level 3 operation. The priority system and its impact on operations is summarized in Table 2. The reduced power channels shown at the edges of the frequency bands in figure 4 are used for coordination between the central and regional NOAA offices.

TABLE 2

Summary of Priority Characteristics

Priority 1	
Conditions	Imminent Disaster
Operating Mode	1 Channel
RF Power	2500 Watts per Channel
Coverage	Selected Regions within 1 Beam
Time	As Required
Priority 2	
Conditions	Severe Weather Watch
Operating Mode	2 Channels
RF Power	1500 Watts per Channel
Coverage	Selected Regions within 2 Beams
Time	As Required
Priority 3	
Conditions	Routine Weather Information
Operating Mode	16 Channels, Time Shared
RF Power	300 Watts per Channel
Coverage	192 Regions within 2 Beams
Time	4 Minutes per Hour - Each Region

The numerical addresses transmitted at the beginning of each message can be used in many ways to form different message transmission networks for level 1 and level 2 warnings. A special address could be used to turn on all the home receivers in the area that might be affected by a tidal wave. Another code could be used to switch on all the units in the country in the case of a civil defense emergency. A large number of special purpose networks could be set up to handle a wide range of emergencies covering more than one area. Home receivers in a given area would use an address recognition card that would program the decoder logic to turn the receiver on for all addresses that include that particular area. The address recognition card would be an inexpensive printed circuit card that would complete the logic interconnections required for address coding.

The number of subchannels assigned to each state is a function of the number and types of different weather or potential disaster areas present within the state. Each state will have a minimum of one subchannel assigned to it.

1.2 Uplink, Ground Station to Spacecraft. - One central ground station is proposed. In addition, 48 regional ground stations, one for each state, would also be provided. The central ground station would transmit routine, priority level 3, messages to the spacecrafts, and in addition, would originate the codes sent to the two satellites to change the operating conditions of the tubes as required by the changing priority levels of the messages, and as required by one satellite or the other being in shadow.

The priority level 3 messages transmitted from the central ground station (central NOAA office) would be assembled using weather information and local interest data obtained from the regional ground stations (regional NOAA offices) and/or originating at the central ground station. The regional office data would be sent to the central NOAA office either through the coordination channels available on the Disaster Warning Satellites, or through existing communication satellites and/or land lines. The method of communication between the regional NOAA offices and the central ground station would be worked out with NOAA to provide a system that is both reliable and cost effective.

The major equipment required at the central ground station is shown in figure 5. The assembled input messages are automatically recorded on tape cassettes and then played back at quarter-hour intervals until updated by a new recording. The numerical address codes and the priority level codes are added to the audio signals before modulation of the eight different disaster warning carrier frequencies sent to each satellite. A coordination channel is added to the transmitter for each satellite and uses the same uplink antenna as the message transmission. There are two 900 MHz receiving systems at the central ground station to receive the coordination channels re-transmitted from the two satellites.

Priority level 1 and 2 transmissions, because of their urgency, would be transmitted to the satellites directly by the regional ground station for the area involved. The equipment required at each regional ground station is shown in figure 6. The regional ground stations are similar to the central station but are much simpler because of the lesser number of subchannels transmitted. The transmitting antennas are fixed pointing to reduce the cost. The coordination channel antenna is aimed to receive the signals from both satellites, thus coordination communications are provided when the regional office's primary satellite is in shadow.

The uses of the coordination channels are given in Table 3.

TABLE 3

Coordination Channels Between Central and Regional NOAA Ground Stations

Uplink Frequency	Downlink Frequency	Uses
6000.0^{-} MHz	900.0^{-} MHz	Communicate from central ground station to regional ground stations in eastern U. S. mainland. Communicate from regional ground stations anywhere in U. S. mainland to central ground station, located in eastern U. S.
6000.4^{+} MHz	900.4^{+}	Communicate from central ground station to regional ground stations in western U. S. mainland.

With the frequencies used as shown in Table 3 the two coordination channels are maintained when either satellite is in shadow. This is explained further in the section on the Spacecraft Communications System.

The coordination channels would be used in the following manner, for a regional ground station to initiate a priority level 1 or 2 transmission.

1. The regional office contacts the central ground station over the 6000.0 MHz channel.
2. After verification of the need for the emergency transmission, the central ground station stops its routine transmissions to the satellite and configures the satellite for the emergency transmission.
3. The central ground station relays a "go-ahead" to the regional office over the appropriate coordination channel. Any time limits or other restrictions are also sent to the regional ground station.
4. During the emergency transmissions, the coordination channels are available for the communication of other emergencies to the central ground station.

The link calculation for the uplink transmission from the central ground station is given in Table 4. The frequency is 6.0 GHz, using a modulation index of 1. The power amplifier has 20 watts output for each of the disaster warning channels. This results in a calculated carrier to noise (C/N) ratio of 30 dB in in each channel with 6 dB of margin. The 30 dB C/N in the satellite receiver, will result in negligible degradation of the planned C/N of 20 dB for the receivers on earth. The coordination channel uplink power is 10 watts, resulting in 27 dB received C/N at the satellite.

The uplink system described uses the satellites only as repeaters. The function of controlling and combining the various

TABLE 4

Uplink Calculation for Central Ground Station

Transmitted Power per Channel	P_T	+13 dBW
Transmitting Antenna Gain, (20 ft. D., at 6 GHz)	G_T	49 dB
Feed Loss	L_F	- 1 dB
Distance Loss	D.L.	-162.5 dB
Transmitting Antenna Pointing Error	$A_{T_{PT}}$	- 0.3 dB
Receiving Antenna Pointing Error	$A_{R_{PT}}$	- 0.0 dB
Receiving Antenna Effective Area (17^o h.p.b.w.)	A_{EFF_R}	-15.3 dB
Atmospheric Losses	$-L_A$	- 0 dB
Polarization Loss	-P.L.	- 0 dB
Spacecraft Feed Losses	$-L_{FS}$	- 1 dB
Noise Power Density	-KTo	+204 dBW/Hz
Channel Bandwidth at rf (16 kHz)	$-B_{RF}$	-42 dB(Hz)
Noise Figure of Spacecraft Receiver	-N.F.	-7 dB
Margin	-M	-6 dB
Carrier to Noise Ratio in each Channel	C/N	30.9 dB

messages is performed on the ground at the central ground station. The satellites, then, are of minimal complexity, thus increasing total system reliability.

A detailed study of the optimum (cost, system security, etc) ground network system including ground control stations was not carried out. Such a study should be carried out in conjunction with NOAA since only they have knowledge of their long range program plans, funding capabilities and disaster warning needs. The results of the study reported on herein can serve to define a satellite capability framework around which a ground network system could be developed.

1.3 Downlink, Spacecraft to Ground Station. - For the transmission of voice messages with high sentence intelligibility, a receiver output signal to noise ratio of 15 dB or greater is required. The largest variable in the transmission path loss from the satellite to the receiver is the attenuation through the building in which the receiver is located. At 900 MHz, building attenuation varies from 10 dB for wooden structures, to 27 dB for steel or reinforced concrete structures. The downlink transmission loss calculation without building loss considered is summarized in Table 5 for priority 3 transmission.

Figure 7 shows the home receiver output signal to noise (S/N) ratio as a function of building attenuation for the three priority levels. Recall that the spacecraft configuration is altered for each priority to put more or less transmitted power into specific channels as required. Priority level 1 messages could be received with high intelligibility within buildings having an attenuation of 21 dB or less. Level 2 and 3 messages could be received well within buildings having attenuations less than 19 and 12 dB, respectively. The owner of an emergency broadcast receiver would be instructed to locate his receiver at a point where the

TABLE 5

Downlink Calculation

Transmitted Power per Channel, (300 W)	P_{RF}	24.8 dBW
Transmitting Antenna Gain, (19 ft. diameter)	G_T	32.3 dB
Distance Loss, (48° Lat., 19° Long. Offset)	-D.L.	-162.7 dB/m^2
Antenna Feed Loss	$-L_F$	-0.5 dB
Beam Edge Loss	-B.E.	-3.0 dB
Transmitting Antenna Pointing Error Loss, (0.17°)	$-A_{T_{PT}}$	-0.6 dB
Receiving Antenna Pointing Error Loss	$-A_{R_{PT}}$	-0.0 dB
Effective Area of Receiving Antenna, (λ/2 dipole)	A_{EFF_R}	-18.4 dB(m^2)
Atmospheric Losses	$-L_A$	-0.0 dB
Polarization Loss, (circular to linear)	-P.L.	-3.0 dB
Margin	-M	-0.0 dB
Building Attenuation	$-Att_{Bldg}$	-0.0 dB
Noise Power Density	$-KT_o$	+204.0 dBW/Hz
Receiving System Noise	$-T_S/T_o$	-10.4 dB

TABLE 5 (cont'd)

Signal Bandwidth, (4 kHz)	$-B_L$	-36.0 dB
RF to Signal Bandwidth Ratio, (m=1)	$-2(m+1)$	-6.0 dB
Receiver Carrier to Noise Ratio	C/N	20.5 dB
FM Improvement, (m=1)	$\frac{2m^2 (m+1)}{3}$	1.3 dB
Pre-emphasis Improvement	I_{PRE}	8.0 dB
Output Signal to Noise Ratio	$(S/N)_o$	29.8 dB

NOTES:

1. Transmit frequency = 900 MHz
2. Half power beam width of 19 foot antenna at 900 MHz is 4.0^o
3. V_{PK}/V_{rms} for voice = 3
4. Single axis satellite pointing error - 0.12^o
5. Receiver noise figure = 10 dB
6. Indigenous noise = 8 dB above KT_o, in 10^o band at horizon
7. Pre-emphasis improvement assumes 150 sec. time constant and 4 kHz highest frequency

best signal is obtained. In a steel or a reinforced concrete building, locating the receiver near a window may be required.

1.4 Spacecraft Communication System. - The spacecraft communication system consists of the two parallel chains shown in Figure 8. The uplink signal is transmitted to the spacecraft using an uplink carrier frequency of 6.0 GHz. There are 2 information bands, nominally 100 KHz wide, and each containing 4, 16 kHz, frequency modulated disaster warning channels, spaced as shown in Figure 4. This uplink signal is amplified in each chain using tunnel diode amplifiers. The amplified signals are translated in the first mixer to an IF frequency of 4 MHz, with an IF bandwidth of 220 KHz. In spacecraft A bands 1A and 2A are separated and amplified by IF amplifiers 1 and 2, respectively. The adjacent band signals, 1B and 2B are 25 dB below bands 1A and 2A at this point, as received at satellite A. This is the result of the 30° spacing of spacecrafts A and B in synchronous orbit, and the narrow half-power beamwidths of the uplink antennas.

The IF signals are translated to 900 MHz by the second mixers and amplified by solid state driver amplifiers, followed by controlled attenuators. The attenuation is controlled by ground command to establish the required drive powers for the different priority levels.

Crossfield amplifiers (CFA) with multistage collectors are used as the UHF high power output tubes. The CFA is a microwave tube which converts DC electrical power into RF power through the medium of an electron beam. The electron beam drifts in crossed electric and magnetic fields past a slow wave structure which propagates an RF traveling wave approximately synchronous in velocity with the beam. The beam interacts with the RF wave and the DC beam potential energy is converted to RF energy. Past the RF output of the tube, residual potential and kinetic energy of the beam is recovered in a multistage collector and returned to the system, resulting in increased efficiency. A schematic diagram of the proposed tube is given in Figure 9 .

Experimental results obtained from test of a 2.0 GHz CFA, with a multistage depressed collector have shown stable operation from 30W to 3 KW power output. Intermodulation distortion produced was low for tube operation at 3 dB below saturated power output. The design characteristics for the crossfield amplifier are given in Table 6. The low level of intermodulation distortion, produced at the 1.2 KW output power level, permits operation without the use of narrow band output filters. Filters would normally be required to prevent interference with adjacent bands of spacecraft B. This characteristic enables the driver and crossfield amplifier to be operated with bandwidths sufficiently wide to amplify the combined spectrum required for bands 1A, 1B, 2A and 2B.

TABLE 6

Crossfield Amplifier Design Characteristics

Output Power kW	DC Input Pwr. kW	Eff. %	Gain dB	Input Drive Pwr. W
0.3	1.5	20	28	0.13
1.2	1.9	63	28	0.5
1.5	2.2	68	27	4
2.5	3.7	68	25	10

During eclipse of spacecraft A, that satellite's function is taken over by satellite B. The disaster warning channels normally directed to satellite A, are directed to satellite B. Satellite B receives these signals, amplifies them, and transmits them to the appropriate areas of the United States. The 220 KHz bandwidths of the spacecraft IF's allow spacecraft B to take over the function of A with no changes in the spacecraft. The coordination channel normally transmitted through spacecraft A also is handled by B when A is in shadow. When spacecraft B is in shadow, its functions are taken over by A in a similar manner.

The output of the cross field amplifiers are used to drive separate feeds of the downlink antenna. The feeds are offset to produce two 4^{o} HPBW patterns using a 19 ft. diameter parabolic reflector.

1.5 Home Receivers. - A sketch of the emergency braodcast receiver designed for indoor use is shown in Figure 3. The listener would be expected to locate the receiver and to position the antenna so that maximum signal is received. In normal operation, power is supplied by the 115 volt, 60 Hz line. In the case of a power failure, the receiver automatically switches to an internal battery supply which has the capacity for 72 hours of continuous operation. Using the three position switch shown on the receiver front panel, the listener may select the operating mode of the receiver. The usual mode would be "emergency", where the receiver has audio output only during emergency broadcasts that contain the address code of the home receiver. In the "local" mode, all broadcasts to the receiver's address code would be heard over the speaker. This mode would be used for monitoring routine weather information and other data broadcast on a non-emergency basis. In the "continuous" mode, the address decoding logic is bypassed and all subchannels being broadcast over the receiver's frequency are heard continuously.

Figure 10 shows the emergency broadcast receiver to be a conventional, FM voice receiver with address decoding logic added. The receiver is fixed-tuned to a single 16 kHz rf channel and, therefore, can receive 12 subchannels by address card change only. Receivers in different parts of the country would be factory-tuned to different rf channels. The receiver normally operates on a standby basis with no dc power supplied to the audio power amplifier, to reduce power consumption. When the transmitted address and priority level codes cause the final audio stages to be turned on, the succeeding message is heard. At the end of the transmission, the higher power stages are turned off.

The block diagram of the home receiver decoding logic is shown in Figure 11. The codes transmitted from the satellite are baseband pulses of greater amplitude than the peak voice signal. The code pulses are detected by the level detector in the receiver. The received code word is shifted into a 14 bit shift register for storage. After the entire code word is received, the received address is compared to the local address stored in the decoder.

The address comparator output, the emergency bit, and the ON bit are used together with the manual switch position to control the receiver's audio output. At the end of the message transmission, a transmitted pulse shifts the ON bit out of the storage register, which resets the decoder logic to its initial state.

The emergency broadcast receiver, exclusive of the decoding logic, would be comparable in size, quality, and cost to existing, portable, transistorized, FM broadcast receivers. The decoding logic could be built using several integrated circuits. Including the address recognition card, the decoding logic would be smaller than a cigarette package. The address recognition card, which contains the receiver's local address code, would be an inexpensive printed circuit card. A two-sided address card could give the listener the flexibility of listening to either his local area broadcasts or to those of an adjacent area.

2.0 Ascent Propulsion Selection

2.1 General. - Various combinations of chemical boosters and electric propulsion systems were considered to perform the launch and synchronous orbit injection phase of this mission. Only two systems are discussed here. Discussion of the other systems considered is contained in Appendix B. The key parameters which govern the launch vehicle selection are the cost of the launch and injection system, the time to achieve the synchronous orbit, and the size and cost of the solar array required by the mission.

The RF broadcast requirements of the disaster warning satellite requires that the solar array provide approximately 5 kilowatts of power at the end of the mission. The initial output must be greater, of course, to provide for radiation degradation during the life of the mission. In this regard, the initial output required (and hence the size array required) is dictated primarily by radiation from the Van Allen belt and solar flares. Using a Thor/Delta launch vehicle with electric thrusters exposes the array to both major sources of degradation for long periods of time and dictates an initial array output of 9 kilowatts. Direct ascent with a Titan 3 C reduces initial array output requirement (and hence required array size) to 7 kilowatts because essentially no time is spent in the Van Allen belts. Mission studies indicate, however, that total mission cost can be reduced with a Thor/Delta/electric thruster system even though a larger solar array is required.

The Thrust Augmented Thor/Delta launch vehicle can place the spacecraft in an initial 1475 nautical mile circular orbit. The spacecraft can then be spiraled out to synchronous altitude in 165 days propelled by three, 30 centimeter diameter, mercury bombardment ion thruster systems. Cost of this launch system(Thor/Delta/electric thruster and the two additional kilowatts of solar array not required by a direct ascent system) is approximately 9.5 million dollars. One kilowatt of solar array was estimated to cost one million dollars. In comparison, a Titan 3 C at a cost of 21 million dollars will place the satellite in synchronous orbit in one day.

The results of the mission study are contained in Figure 12. The solid lines depict the performance of Thor/Delta with electric thrusters over a wide range of initial array powers. The dashed portions of the curves represent the region where array degradation reduces the solar array power at the end of mission below the required 5 kilowatts. In a practical sense, array output for all Thor/Delta missions is reduced approximately 45 percent during the spiral out portion of ascent. Subsequent degradation is minimal by comparison.

The Titan 3 C direct ascent mission power requirements are indicated on the abscissa of Figure 12 since its ascent time is measured in hours. Its solar array also degrades (from 7 to 5 kilowatts) during the life of the mission but its degradation is not as severe as for the Thor/Delta missions which expose the solar array to the earth radiation belts.

The selected design point for this mission is shown on the Thor/Delta curve in Figure 12. The propulsion time of 160 days can be accomplished with either two or three ion thrusters. Three units are proposed because of the redundancy provided and also because they form part of the attitude control system described later.

Mission profiles are described in the following sections. Table 7 summarizes the major mission parameters for the Thor/Delta/ electric stage and the Titan 3 C mission. Appendix B contains information on the less promising vehicle systems for this mission, namely, the Atlas/Agena/electric stage and the SLV3C/Centaur/electric stage. Also considered is placement of multiple spacecraft in orbit with a single Titan 3 C booster system.

TABLE 7

Major Mission Parameters for Disaster Warning Satellite Study

Booster	Thor/Delta	Titan 3C
Shroud	Delta	Martin (UPLF)
Shroud Weight	543 lb.	2310 lb.
P/L Adapter Weight	70 lb.	230 lb.
Payload Weight	1565 lb.	1442 lb.
Mercury Weight	440 lb.	N/A
Power-initial	9.07 kw	7.00 kw
Ascent Degradation	45%	0%
After Ascent	4.99 kw	7.00 kw
After 5 Years	4.99 kw	4.99 kw
No. of Elec. Eng.	3	N/A
Type	Mercury Bombardment Ion Thruster	N/A
Size	30 cm dia.	N/A
Beam Current	2 amps	N/A
Net Accel. Volts	550v	N/A
Input Power	1.66 kw	N/A
Thrust	22×10^{-3} lb.	N/A
Spec. Imp.	2000 sec.	N/A
Prop. Util.	85%	N/A

TABLE 7 (cont'd)

Power Eff.	67%	N/A
Initial Alt., km	2730 km	38,500 km
Initial Alt., nm	1475 nm	19,300 km
Propulsion Time	160 days	1 day
Estimated Vehicle Cost	$6 million	$21 million

2.2 Thor/Delta/Electric Stage. - The booster for this mission is a modified version of the current Thor/Delta launch vehicle system. The current system consists of a long tank Thor first stage with six Castor II solids strapped on for thrust augmentation, an improved Delta second stage, and a spin stabilized solid third stage. The modified version requires an additional three Castors for a total of nine, substitution of the Titan Transtage engine for the present Delta engine and elimination of the third stage. The mission sequence is as follows:

1. Launch is in an easterly direction from Cape Kennedy. The launch date is selected to control the timing and duration of the shadowed period. The optimum daily launch windows occur during summer near local solar midnight and near noon in winter.
2. The Delta and payload are injected into a circular parking orbit at an altitude of 100 nautical miles and 28.4 degrees inclination.
3. Transfer is initiated 22 minutes later at the first decending node.
4. Transfer is completed at the first ascending node, leaving the spacecraft in a circular orbit at an altitude of 1475 nautical miles. Transfer time is 55 minutes. At each burn of the transfer ellipse, the inclination will be reduced slightly to minimize the time required for the electric propulsion system to complete the mission.

5. The spacecraft is separated from the Delta, the solar array is deployed, and the electric thrusters are started in order to raise the spacecraft to synchronous orbit and remove orbit inclination. Near optimal thrust vector steering is initiated resulting in simultaneous change in altitude and inclination. This will minimize time required to achieve synchronous orbit station. The initial thrust vector yaw offset angle to effect inclination change is approximately 25 degrees.
6. As the mission proceeds, the magnitude of the orbital velocity vector decreases and the thrust vector yaw angle increases to 50 or 60 degrees at synchronous equatorial conditions.
7. The magnitude of the thrust vector will be adjusted as the mission proceeds in order to control the final longitude reached.
8. Synchronous equatorial location is reached in 165 days allowing about 5 days for shadowing.

2.3 Titan 3 C. - A Titan 3 C booster could be used for a direct ascent mission. The Titan 3 C consists of the two-stage Titan "core" vehicle with two 120 inch solid propellant motors strapped onto the first stage, and the Transtage third stage. The mission sequence follows:

1. Launch eastward from Kennedy. There are no mission constraints on launch date or time of day.
2. Transtage and payload are injected into a circular parking orbit at 100 nautical mile altitude and 28.4 degrees inclination.
3. Transfer is initiated about 66 minutes later at the first ascending node. The transfer ellipse has 25.2 degrees inclination.

4. After 5.25 hours transfer time, apogee burn places the vehicle in synchronous equatorial orbit at approximately 95 degrees west longitude.

5. A velocity error of $\pm$30 feet per second can exist after final Transtage burn and must be removed. The impulse required for this correction is 1300 pound seconds and a cold gas system weighing about 50 pounds has been included in the payload weight for such purpose. After the payload is separated from the Transtage, its longitude drift will be monitored and the cold gas system used to correct it. One or two days have been estimated for this process.

3.0 Mechanical Configuration

3.1 General Description. - The spacecraft concept presented is packaged to fit inside the standard 5 foot diameter by 17 foot high Delta shroud. It will also fit in the shrouds of the other vehicles considered. The general mechanical configuration is shown in figures 13 and 14. When configured for on-orbit operation, a 19 foot diameter antenna deploys on one end of the spacecraft center body and each wing of the solar array extends 65 feet from the centerline of the center body.

For the spiral-out mission, three ion thrusters are mounted on the end of the center body opposite from the antenna structure. During the spiral-out flight sequence, the solar array must be continuously sun oriented while the center body is oriented to provide proper thrust direction. After achieving synchronous altitude, the center body is oriented to point the antenna continuously toward the United States. Slip rings are required to conduct solar array power to the center body in all flight modes.

The center body is a 10 foot high structure which supports the power, telemetry, command and thermal systems, attitude control components and the various electronic equipment of the systems. It is attached to the Delta second stage through the standard 54 inch diameter by 14 inch high Delta adapter and separation assembly.

3.1.1 Antenna. - The main downlink antenna is a "wrap rib parabolic reflector" that is stowed in a 8" D x 48" OD ring at launch. It is released by a pyrotechnic pin puller device. Deployment forces are provided by the stored energy in the ribs. The antenna feed for both down link beams is located near the focal point of the antenna. It is deployed into this position using the spring force from the rotating joint. The feed assembly is supported during launch by a structure mounted on top of the tower. This antenna is of a design which has been space-proven several times. The flexible mesh connecting the ribs of the antenna is radially pleated between the ribs while stowed. After deployment, at synchronous altitude, the antenna is located at the earth-facing end of the satellite tipped at a 7^{o} angle so it points at the United States.

3.1.2 Solar Arrays. - The solar arrays consist of conventional silicon cells mounted on flexible substrates that can be rolled up into two cylindrical packages for launch (see Figure 13). An alternative method of array storage by folding is also compatible with the spacecraft concept. The arrays are mounted on the sides of the tower on a common motor-driven shaft, which rotates the arrays to follow the sun. The slip rings for carrying current and signals from the array are also located in this shaft assembly. The arrays are deployed by rolling them off drums using battery power and extend from each side of the satellite longitudinal axis.

3.1.3 Communication System Crossfield Amplifiers. - Two CFA's and their associated filters and power monitors are located in the upper portion of the center body structure where transmission losses to the antenna and thermal interference with other components can be minimized. They are connected to the antenna feed by two coaxial cables along the supporting structure for the feed system. Cooling the amplifiers is accomplished by installing heat pipes from the amplifier body to the radiating surface of the spacecraft body.

3.1.4 Attitude Control Thrusters. - Eight cesium ion thrusters are required for attitude control and station keeping. Two cesium thrusters are located on the main body and a set of three thrusters are located at the tip of each solar array (65' from the satellite longitudinal axis) on an end cross member.

Two earth horizon sensor heads are located on the antenna feed structure. The heads are mounted on a motor-driven mechanism that provides two degrees of rotational freedom. This allows tracking during both the spiral-out and synchronous orbit phases of the mission.

3.1.5 Mercury Ion Thrusters. - The three mercury ion thrusters, used for primary propulsion during the spiral-out mode, are mounted on the base of the tower with their thrust vectors parallel to the spacecraft longitudinal axis. Thruster gimbal capability of approximately 10° is provided so that, if any thruster malfunctions, the thrust vector of the remaining thrusters can be directed through the satellite's CG. The propellant (mercury) for all three thrusters is contained in a single toroidal tank. This tank is mounted to the tower base but electrically isolated from it.

3.1.6 Electronics. - The CFA drivers, power systems, telemetry and command systems and portions of the attitude control systems' electronics are located in the lower portion of the tower. These components are mounted on a vibration damping structure and in a temperature controlled enclosure.

3.1.7 Spacecraft Structure. - The basic structure shown in Figure 13 is a tower fabricated from square, extruded, thin-wall magnesium tubing. A rectangular cross-section magnesium ring is used to attach the tower base to the launch vehicle through the Delta conical adaptor. Brackets welded to the tower or the base ring are used to support the antenna, slip ring assembly, toroidal mercury tank, crossfield amplifiers, power monitors and filters. The housekeeping electronics are mounted on viscoelastic tubes which are bracketed to the tower cross members at their ends. The ion thrusters are mounted to tower cross members that are welded to the inside of the base ring. All of the thermal radiators are mounted to the sides of the tower and all but the CFA radiators are part of the tower structure.

A weight breakdown for the spacecraft is presented in Table 8.

3.2 Thermal Design. - The spacecraft has an active thermal control system coupled with isolation from the environment. The solar array and antenna have a passive temperature control. Both techniques are similar to those used by other operational satellites. This design is unique only in the quantity of heat rejected and the sizes of the solar array and antenna.

The satellite starts operation in low earth orbit, and spirals out to synchronous orbit. During the major portion of the spiral-out mission, the spacecraft is in sunlight although two sides and both ends of the tower are alternately sunlit and shadowed as the satellite rotates around the earth. Once in synchronous orbit, the

TABLE 8

Spacecraft Weight Estimates

	Weight Summary	Totals
Attitude Control System	176 lbs.	
Power Distribution and Control System	179 lbs.	
Telemetry and Command System	54 lbs.	
Mechanical and Thermal Systems	240 lbs.	
Communication System	370 lbs.	
Solar Array Assembly	443 lbs.	
Spiral-Out Thruster Assemblies	100 lbs.	
Weight at Synchronous Altitude		1562 lbs.
Spiral-Out Propellant	440 lbs.	
Weight in Parking Orbit		2002 lbs.
S/C Adapter and Sep. Mech.	70 lbs.	
Payload at lift-off		2072 lbs.

entire satellite encounters periods of eclipse when it encounters the earth and moon shadows. During the shadow periods, the satellite can be in the earth's shadow for as long as one hour and in the moon's penumbra for as long as 4 hours. During the spiral-out, shadowing occurs more frequently, but the duration per incident is shorter. These wide variations in solar thermal inputs dictate a spacecraft design using near thermal isolation from the environment.

Thermal power dissipation in the spacecraft varies from approximately 100 watts with thrusters and the communication system off, to 2663 watts with the communication system on. The thermal power dissipation during spiral-out with the ion thrusters operational is 2000 watts. Thermal power dissipation also varies with the mission phase. The large range in thermal loads, together with a requirement to maintain moderate temperature ranges, require the use of an active thermal control system. Active control devices used are temperature controlled louvers, heaters, and heat pipes. The heat pipes are capable of transferring large quantities of heat from high heat dissipating components to exterior radiator plates with very small temperature drops.

The thermal requirements and designs of specific components are summarized in Table 9 and discussed in the following paragraphs.

3.2.1 Crossfield Amplifiers. - The two CFA's are very concentrated heat sources. Each tube rejects 1000 watts at a maximum temperature of 200°C (390°F). As this heat cannot be radiated or conducted away without unacceptable weight

TABLE 9 - THERMAL DESIGN PARAMETERS

Component or System	Location	Max. Therm. Pwr. Watts	Temp. Range °F	Thermal Control
CFA No. 1	Upper Tower	1000	40 to 390	Heat Pipes, Insulation & Htrs.
CFA No. 2	Upper Tower	1000	40 to 390	Heat Pipes, Insulation & Htrs.
CFA Driver No. 1	Center Tower	200	40 to 120	Heat Pipes and Insulation
CFA Driver No. 2	Center Tower	200	40 to 120	Heat Pipes and Insulation
Receiver	Antenna Feed	100	-30 to 120	Louvers and Insulation
Power System	Center Tower	30	40 to 120	Louvers and Insulation
Telemetry & Command Sys.	Center Tower	73	40 to 120	Louvers and Insulation
Attitude Control System	Center Tower	60	40 to 120	Louvers and Insulation
Solar Array	Appendage	-	-200 to +160	Passive - Coatings
23 Ft. Antenna	Top of Tower	-	-200 to +160	Passive - Coatings
Thrusters Prop. Tanks	Tower Base		-20 to +160	Louvers and Insulation

penalties, a convective system is required. Heat pipes were selected for this system because of their efficiency and simplicity. One or more heat pipes are provided for each of the CFA collectors with the heat carried to two 10 square feet radiators surfaces. When a CFA is not being operated, the cathode heater remains on to prevent cathode damage due to thermal shock.

3.2.2 <u>CFA Drivers</u>. - Each CFA driver must reject 200 watts at 50°C (120°F). The low temperature and the large heat load from a small package also necessitates heat pipes for heat transfer. Several heat pipes will be used to transfer the heat to two 5 sq. ft. radiators.

3.2.3 <u>Ion Thrusters</u>. - Each ion thruster rejects about 250 watts of heat. Normally, a single thruster can reject this heat directly to space. Grouping the thrusters decreases the radiation from each thruster to space, so additional radiating surface is added to maintain acceptable temperatures. The propellant tank is protected from freezing during thruster off periods with insulation and thermally controlled louvers on the sun sides of the structure.

3.2.4 <u>Spacecraft Electronics.</u> - The spacecraft electronics are located in the lower structure where they normally reject 120 watts at a temperature less than 50°C (120°F). Their temperature is maintained with thermally-controlled louvered radiator surfaces to 27 $\pm$22°C (80° $\pm$40°F).

3.2.5 <u>Solar Array and Antenna</u>. - The temperatures of the array and antenna are determined by the surface properties, applied coatings and the environment as well as the power being supplied. Special coatings are selected to prevent overheating of the array and thermal distortion of the array and antenna structures.

4.0 Power System

4.1 General Description. - The power system consists of a large deployable high voltage solar array blanket, liquid metal slip rings which transfer power across the rotating joint between the sun tracking solar array and the earth oriented spacecraft center body. The power switching and control system is located in the spacecraft.

The solar array structure has a total span of 130 feet, and 900 square feet of active solar cell area. The blankets are constructed of thin kapton substrate, on which the cells are mounted. The blankets are stored on a drum during launch and deployed simultaneously be means of extendable booms. Although a high voltage array was selected as the prime power source, the mission can also be performed with a more conventional low voltage array with power conditioning but with an increase in weight.

4.2 Solar Array. - In the past, specific power requirements on spacecraft have been met by converting and conditioning a fixed block of solar array power nominally at 100 volts or less. This approach has resulted in the use of heavy electronic packages consisting of regulators inverters, transformers and rectifiers. Conversion efficiencies of approximately 85% are achieved by such systems. Also, for complex conditioning requirements, the large number of parts required by such systems makes long term reliability a problem.

A logical first step toward simplification is to generate the power on the array at the voltages required by the load. This is the technique proposed for this mission. It is possible to assemble a block of solar cells with a specific series/parallel configuration to match a load requirement. Series and parallel arrangements can be exchanged by switching as loads change and cells can be shorted by switches to provide regulation to a high degree of precision. Because of their lightweight and small size, the required switches can be integrated on the array and thus eliminate most of the conventional and high voltage power conditioning equipment. Such an array system could have a power utilization efficiency approaching 98%

During this mission, equipment on the spacecraft requiring conditioned power changes as the mission progresses. Specifically, during the orbit raising portion, the array provides the majority of its power directly to ion thrusters. During synchronous orbit operation, the power is used to drive the communications system. Housekeeping power and attitude control thruster power will also be provided by the array throughout the mission.

4.2.1 Regulation Concept. - Regulation for particular loads will be done on the array by shorting (or short removal) of blocks of solar cells to provide voltage and/or current regulation. A conceptual example of such a regulating system is shown in

Figure 15. The switches will be solid state devices integrated with their driving circuitry on a substrate and attached to the array in close proximity to the blocks of solar cells to be regulated. The switch states will be controlled by signals from the spacecraft computer.

The spacecraft computer which operates at low voltage is electrically isolated from the switches on the array which may be at high voltage.

The switches will be electrically connected across blocks of solar cells which are sized according to a binary progression. Regulation to whatever precision is required will be accomplished by shunting blocks of the appropriate binary size.

4.2.2 Reconfiguration Concept. - In general, reconfiguration will consist of switching large and small blocks of cells into various series and parallel combinations as required by changing mission loads. The switches used will consist of solid-state and electromechanical devices connected to the array blocks to adapt the array as the mission dictates.

A conceptual example of reconfiguration is shown in Figure 16. Signals to control the setting of the reconfiguration switches will be provided in the same manner as described above for the regulation switches.

4.2.3 Mercury Thruster Load. - The three thrusters receive their power, conditioned as required, directly from the solar array. The array power block requirements for one of these thrusters is shown in Figure 17.

The building blocks as shown are intended only to convey the general concept of modular configuration. Figure 18 which will be discussed in detail, shows how the Figure 17 blocks of power are reconfigured to power the CFAs. Further development of mission details will make possible a more optimized specific solar array design. Such a design study was not carried out for this proposal and, therefore, the descriptions provided herein should be considered only as preliminary concepts for this mission.

Table 10 shows more specifically the nominal power breakdown for an orbit raising thruster. The constraints assumed in developing the power systems for the orbit raising thrusters were as follows:

a. Initial power available from the solar array for the thrusters is 9 KW.

b. The minimum voltage permitted on V5 for thruster operation is 800V.

c. The thrusters screen supply (V5) current should be 2 amps at start of mission and maintained as high as possible within constraint (b).

d. Solar array power degradation expected during the orbit raising transit time is 45 percent.

e. Nominal values only are considered (i.e., no error or redundancy provisions are incorporated).

TABLE 10

30 CM Mercury Thruster Power Breakdown

Supply	Voltage (volts)	Current (amps)	Nominal Power Init. (watts)	Nominal Power Final* (watts)	Number of Solar Cells Series	Par.	Total
V1	3	3	16.4	9	10	35	350
V2	3	3	16.4	9	10	35	350
V3	20	3	109	60	63	35	2205
V4	50 to 40	15-10	730	400	122	125	15250
V5	995 to 800	2-1.37	1990	1095	2440	15	36600
V6	-688 to -500	0.1-0.075	61.8	38.5	1525	1	2100
V7	10	3	54.5	30	31	35	1085
V8	30	0.2	10.9	6	92	3	276
V10	12	0.4	8.75	4.8	37	5	205
			2997.75	1652.3			58421

*assuming 45% degradation

The V1, 2, 3, 7, 8 and 10 power blocks must be capable of producing the nominal voltage and current shown in Table 10 throughout their period of operation. The V4 and V5 power blocks supply power within constraints (b) and (c) above but voltage and current will vary such that the resulting engine thrust will minimize the orbit raising time. The values in Table 10 for V4 and V5 current and voltage, therefore, represent the range of operation for these loads resulting from array degradation and thrust optimization. The same comments apply to power block V6. Solar cell arrangements within the power blocks are also shown in Table 10. The number of cells shown can produce slightly higher than nominal power levels. No attempt was made during this preliminary look at configurations to optimize cell usage. All regulation required to satisfy the Table 10 conditions will be done using regulation switches on the array. The regulation switches will be set by signals from the spacecraft computer as it receives signals by either ground command or automatic load sensing.

4.2.4 Crossfield Amplifier (CFA) Load. - The two identical CFAs are both powered directly from the array. CFA power is obtained from the same solar array sections that were used to power the orbit raising thrusters. Upon reaching sync-orbit the thrusters are shut down and the solar array reconfigured to power the CFAs. At this point in the mission, the portion of the array supplying the mercury thrusters has degraded from approximately 9 KW to approximately 5 KW. Additional degradation will be experienced in synch-orbit. However, the amount of degradation experienced after reaching synch-orbit to the end of mission is only a small fraction of that experienced during orbit raising. The solar cell block configuration for powering the CFAs is shown in Figure 18. Each block is further identified as to its original location when it was powering the mercury thrusters. The precise regulation required by the CFAs will be accomplished by the regulation concept described previously. The power available in excess of the CFA requirements will provide degradation reserves and a margin for redundancy over the five-year mission.

The array utilization on a relative area basis is depicted in Figure 19. The figure shows the area required to operate the orbit raising thruster loads, the area required to operate the 8 cesium engines used for attitude control and the area required for housekeeping loads. The figure also shows the thruster areas to be converted over for operation of the two CFA.

4.2.5 Cesium Attitude Control Engines. - Direct conditioned solar array power will be provided for the 8 cesium engines used for attitude control. In general, the individual engine loads will be electrically connected in parallel to a common array segment. Each engine consumes 100 watts of power. An array section capable of providing 800 watts of power at the end of the mission is required for these engines. Such an array section will be built up and controlled in a manner similar to that shown for the mercury thrusters.

4.2.6 Housekeeping Power. - Low voltage power, regulated where necessary, will be provided in the anticipated amount of 250 W. Since this will be a relatively constant load throughout the mission, a separate section of the array will be devoted to this purpose.

4.2.7 Development Status. - High voltage solar arrays with integral power conditioning have been under study and development for some time. These studies and development efforts have been general in nature and not directed at any particular spacecraft missions. This effort has been carried on both in-house and by contract.

The studies and development efforts to date indicate that a high voltage solar array with integral power conditioning is practical and attractive. Such a power system compared to the conventional approach should result in lower weight, higher reliability, lower cost, smaller size, and greater flexibility both in operation and spacecraft design. However, considerable development effort remains in order to bring such a system to operational status.

4.2.8 Low Voltage Array. - As an option to the integrally conditioned and configured high voltage solar array proposed above a low voltage array, nominally 100 volts or less, can be provided along with conventional electronic power conditioning. The technology necessary to build the low voltage array exists. The array would, of necessity, have to be of higher power and greater size to make up for the power conditioning losses and resistive power losses in spacecraft to array cabling. An array approximately 15 percent greater in size would be required with the conventional approach, due primarily to losses in the power conditioner. In addition, a discrete conventional power conditioner for the mercury thrusters, CFA's and cesium engines would weigh approximately 267 pounds. It would also be necessary to incorporate in the spacecraft design provisions for rejecting the waste heat generated by the conventional power conditioners.

4.3 Slip Rings. - Since the cells of the solar array at all times must face the sun while the antenna beam at all times must face the earth, there is a 1 revolution per day relative rotation between the two components. This rotation can be accommodated by means of an rf joint between the spacecraft center body and antenna or by means of an electrical joint between the solar array and the body. Studies contracted by LeRC and experiments have indicated that, for high power satellites, the most efficient joint in terms of power loss is an electrical joint using slip rings for the power transfer. The antenna then can be rigidly mounted to the spacecraft center body.

The liquid metal slip ring/solar array orientation mechanism (LMSR/SAOM) which is currently under development by the LeRC is a major subsystem of the spacecraft. The LMSR represents a considerable advance over conventional slip rings in terms of transferring electrical power and signals across a rotating joint with minimum power losses or electrical disturbances. It has virtually no wearout mode and it creates minimum mechanical disturbances (no stick slip).

In this design, a liquid metal such as gallium is retained by capillary action between fixed and rotating metal electrodes, and serves as the conducting medium. The complete system LMSR/SOAM also includes bearings, shaft position sensors and torquers.

4.4 Housekeeping Power System. - The housekeeping power system uses a small portion of the total solar array producing a peak power of 300 watts at 27.5 to 50 VDC. The normal housekeeping power consumption is presently estimated at 151 to 250 watts.

A functional block diagram is shown in Figure 20. The power control unit is the central switching unit for the system. It accepts all inputs from the power sources and transfers them to the individual system loads. This unit handles approximately 25 discrete ground commands for functions such as battery charging, AC power on/off and switching the various loads on and off. Approximately 15 instrumentation channels are provided to monitor all phases of power system performance.

The two secondary batteries are initially used to release and deploy the solar array. They are then kept on a low level trickle charge using solar array power to keep them at full capacity. During the spiral-out phase of the mission, a series of partially shadowed orbits will be encountered. The batteries will be used during the shade period to power the essential spacecraft loads, including thruster heaters.

Table 11 shows the power requirement for the housekeeping system.

5.0 Command, Tracking and Data System

5.1 General. - The command and telemetry system provides radio links for receiving ground command signals and transmitting experiment and housekeeping data from the spacecraft. During that portion of the operation when the spacecraft is being raised to synchronous orbit, the NASA Satellite Tracking and Data Acquisition Network (STADAN) will command the spacecraft and receive telemetry in near real time. Block diagrams of the telemetry and command system are shown in Figures 21 and 22.

TABLE 11. - Housekeeping Power Budget

Normal Housekeeping Loads - Spiral Out and Synchronous

Communications System	73 w.
Horizon Sensor	10
Computer	30
Torquer	3
Power Control Unit	5
Basic Loads	121 w. regulated DC
Losses	10
Trickle Charge (2)	20 w.
Solar Array Power	151 w. unregulated DC

Maximum Housekeeping Loads - Spiral Out and Synchronous

Basic Loads	121 w.
Inverter and Gyro	17
	138 w. regulated DC
Losses	12 w.
Battery Charging (2)	100
	250 w. unregulated DC

TABLE 11 (cont'd)

Battery Loads - Spiral Out

Communications System	73	
Horizon Sensor	10	
Computer	30	
PCU	5	
Gyro and Inverter	17	
Regulated Loads	135	w. regulated DC
Losses	12	
Cesium Thruster Htrs (6)	30	
Hg Thruster Htrs (3)	180	
Total - Battery Power	357	w. DC for 5/6 hr. max.

Maximum Battery Discharge - 297 watt-hours

Battery Loads - Synchronous Orbit

Regulated Loads	135	w. regulated DC
Losses	12	
Cesium Thruster Htrs (6)	30	
Tube Heaters (2)	30	
Receiver and Driver	300	
Total Battery Power	507	w. DC for 6/5 hr. max.

Maximum Battery Discharge - 607 watt-hours

5.2 Tracking. - One of the spacecraft transmitters will be on continuously for tracking and data acquisition when the spacecraft is in view of the ground support network during the orbit raising period. Sufficient residual carrier power exists in the RF signal to permit tracking to occur simultaneously with data acquisition. This technique is used for the SERT II and other spacecraft missions.

5.3 Telemetry. - The telemetry system is a PCM/FM configuration. The system will operate at 600 bits per second with a 9-bit data word plus one bit parity. System accuracy excluding sensor and signal condition error will be better than 0.2% of full scale.

The PCM multiplexing equipment consists of four subcommutators with a total capacity of 540 analog measurements, 90 discrete digital measurements, and five PCM serial measurements. The PCM multiplexer samples through its 60 channels once each second which constitutes a minor frame. The complete PCM sampling format or a major frame is composed of 15 minor frames. The PCM multiplexer output phase modulates either one of two 5 watt transmitters which are selectable by command. The subcommutators and PCM multiplexers are redundant units for added reliability, either of which may be activated by ground command. The transmitters operate in the 136-137 MHz range with each transmitter having its own unique frequency.

5.4 Command System. - The command system used is the STADAN standard PCM command system operating in the 148 MHz range. The system will provide for the reception of commands at the spacecraft with verification via telemetry, and separate command execution. The capability of the system will be 510 discrete commands with an output of nominally 50 milliseconds pulse duration. In addition, two command outputs will provide a digital message which is used to program the spacecraft computer. Command rate for discrete commands will be at 2 commands per second without ground verification, and slightly longer when ground verification is required.

An omnidirectional antenna system is used jointly by the telemetry and command systems. The assembly provides circular polarization in a near omnidirection pattern with coverage of approximately 95% of the radiation sphere.

6.0 Attitude Control, Station Keeping and Orbit Raising

6.1 General. - Attitude control during the spiral-out portion of the mission is accomplished by cesium ion thrusters, together with selective throttling of the three mercury ion thrusters used for the orbit raising. After reaching synchronous orbit, the cesium ion thrusters are used to maintain attitude control and to provide both north-south and east-west station keeping.

Figure 23 shows the location of the eight cesium ion thrusters, the three mercury thrusters and the earth horizon sensors, sun sensors and gyro used in the attitude control system of the spacecraft. An on-board computer is provided to analyze sensor inputs and provide thrusting signals to the proper thrusters to maintain the desired spacecraft orientation.

6.2 System Operation. - This section describes the attitude control system operation throughout a sample mission in which the orbit changes are effected sequentially (rather than simultaneously as in the near optimal thrust vector steering used to compute the mission times in Section 2.2). This is done to facilitate a clearer understanding of the control modes and mission plan. For purposes of discussion, the launch is assumed to occur at about 3:45 AM (EDT) on June 15, using a Thor/Delta booster.

6.2.1 Initial Acquisition. - The Delta attitude control system is used to orient the spacecraft to the correct attitude after second burn is complete. With the spacecraft oriented and prior to separation of the spacecraft from the Delta, the solar array is extended and the spacecraft attitude control system activated.

6.2.2 Orbit Raising Normal Operation (without eclipsing). - Orbit raising is accomplished by aiming the thrust vectors of the ion engines in the same direction as the orbit velocity, thereby adding energy to the system. The result is an increasing altitude and a decreasing velocity. The orbit remains essentially circular and at 28-1/2 degrees inclination throughout.

During ascent, the only attitude requirement is due to thrust vector aiming. An accuracy of 10^{-1} or 10^{-2} radians is adequate.

During both ascent and plane changing, six cesium thrusters will be operating continuously except during periods of shadowing. Control torques will be produced by thruster beam vectoring up to $\pm 20^{\circ}$ as illustrated in Figure 23.

The spacecraft is oriented during the initial spiral-out period with the mercury thrusters thrusting continuously tangential to the orbit. Tangential thrusting is accomplished by pitching the center body through 360°, once each orbit on a continuous basis. In order to maintain the solar array active surface directed toward the sun, the spacecraft center body is oscillated about the roll axis at the orbit frequency and with an amplitude of $\pm 45^{\circ}$. At the same time, the solar array is oscillated around its shaft axis relative to the center body. This technique provides solar array alignment in both axes and maximizes the solar power available to operate the ion thrusters.

This portion of the spiral-out operation is carried out under control of the onboard computer. Knowing the angle between the orbit normal and sun, the orbit period, and with a suitable timing pulse, the requisite shaft angles are generated to rotate the solar array relative to the center body. At any instant of time then the overall control system operates on the entire configuration (center body and solar array) as a rigid body using the horizon sensor signals to indicate pitch and roll errors and the array sun sensors to indicate yaw error.

The sun angle is fed into the computer on a daily basis from the ground. The orbital period is measured on board as the interval between timing pulses which are produced by the sun sensors on the horizon sensor platform. This mode of operation continues for 47 days at which time the sun angle has increased to the point at which shadowing starts.

6.2.3 Orbit Raising Eclipse Operation. - Before orbit shadowing commences, the mode of operation of the spacecraft is altered. This change in operating procedures is required to stabilize attitude control rates during periods when the spacecraft enters shadow. Since the attitude control system is shut down during shadow, this operation minimizes orientation offsets when coming back into sunlight. The center body is oriented in the roll axis such that the array axis is normal to the orbit plane and this roll orientation is held fixed. The array itself rotates continuously relative to the center body i.e. around its own shaft axis so as to face the sun. The elevation of the sun above the orbit plane is neglected and will produce power degradation as a cosine function. The initial degradation is 10%, but this decreases rapidly.

In this mode of operation, the main body rotates (in pitch) continuously in inertial space, the array rotates continuously backwards about the shaft supported in the center body so that the array is inertially stationary. The long axis of the array is at the gravity gradient null so that it is properly aligned perpendicular to the orbit plane, resulting in zero gravity gradient torques. When the shadow is entered, the cesium thrusters are shut off but the horizon sensor, computer, and array shaft motor remain active. When the spacecraft emerges from the shadow, the errors should be small and the cesium thrusters are activated and resume control. If the Earth has drifted out of the horizon sensor field of view, the computer will have the last valid readings in storage and these can be used to control a reacquisition mode.

6.2.4 Orbit Raising Post Eclipse Operation. - After approximately 50 days of eclipse operation, the orbit leaves the shadow. The sun is now at an angle of 105 degrees to the orbit normal and the power loss is 3.5%. In order to prevent further loss of power, and the accompanying extended mission time, the attitude control operation reverts to the pre-eclipse mode. At this time, the mercury thrusters will be shut down for as long as 20 hours and restarted at a specified longitude in order to be at the correct longitude when final synchronous orbit is achieved.

6.2.5 Plane Change. - When synchronous altitude and the correct longitude is reached, the thrust vector is reoriented so that it is normal to the orbit plane. Thrusting south at the ascending node and north at the decending node will cause a gradual decrease of the orbit plane inclination toward zero. The thrust vector orientation will be held for half the orbit period and reversed halfway between the nodes.

To accomplish the plane change, the spacecraft is oriented with the thrust vector normal to the orbit plane and with the long (shaft) axis of the solar array in the orbit plane. The array axis is aligned normal to the earth/sun line and the array is rotated around its axis until it faces squarely at the sun. The horizon sensor rotates continuously relative to the center body to track the earth. The thrust reversal is accomplished by rotating the center body around the array axis. It is initiated by ground command when the spacecraft is at its northernmost and southernmost latitudes. It is estimated that the reversal maneuver will require approximately 80 minutes at each anti-node giving a thrusting duty cycle of 88%. The mercury thrusters remain on during the reversal. This operation is maintained until the inclination is below 0.6 degrees.

6.2.6 Eccentricity Reduction. - If the eccentricity is greater than 0.005, it must be reduced. This is accomplished thrusting against the velocity vector at perigee and thrusting with the velocity vector at apogee. A reversal occurs as the spacecraft crosses the minor axes of the ellipse. During one-half of the orbit, the orientation is identical to that used for altitude raising, including the shaft angle oscillations. The reversal occurs by rotating the center body around the array shaft angle and must be initiated by ground command as before.

6.2.7 Synchronous Equatorial Operation Near Solstices. - The above maneuver places the spacecraft at its final altitude, longitude, inclination, and eccentricity. The horizon sensor is rotated through 90 degree relative to its mounting shaft and locked in place so that the horizon sensor line of sight is directly along the shaft. The shaft and horizon sensor are fixed in place for the balance of the mission, reducing the spacecraft to a single internal degree of freedom.

The solar array shaft axis is normal to the orbit plane and the longitude of the center body lies along the earth radius vector. The transmitting dish, fixed to the center body, is aimed about 6 1/2 degrees north of the equatorial plane in order to illuminate the U. S. mainland with the RF signals.

In this configuration, the array perpendicular is off the earth/sun line by the amount of the sun's right ascension. This displacement results in a power loss of 8 1/2 percent maximum. This maximum value of power loss occurs only twice a year (at the solstices).

In the above attitude configuration, the horizon sensors are used for pitch and roll control references. The sun sensors mounted on the array provide a yaw control reference. Also, in this attitude configuration, the solar array rotates once per day relative to the center body so that the array can be sun oriented and the center body with the antenna can be earth pointing.

On station in synchronous equatorial orbit, the cesium thrusters will be used to maintain attitude control and provide station keeping. The attitude control accuracy requirement is dictated by the antenna pointing error. A range of two to six milli-radians is assumed to be adequate and is compatible with horizon sensor errors. The error applies directly to the pitch and roll axis. A yaw error ten times greater would be required to produce a comparable pointing error.

A station keeping accuracy of 10^{-2} radians is assumed based on uplink transmitter power constraints. This translates into a maximum allowable eccentricity of 0.005 and an inclination of 0.6 degrees.

6.2.8 Synchronous Equatorial Operation Near Equinoxes. - For approximately one month before and after the equinoxes shadowing occurs, reaching a maximum duration of 72 minutes right at the equinox. As before, the cesium engines are shut off and the broadcast operation is interrupted. After exit, the Earth is reacquired.

During this same time period, the yaw error signal detected by the solar array sun sensors is very weak, reaching zero at noontime on the equinox. To control the yaw error at these times, the yaw gyro is activated and provides a backup reference capability. The solar array sun sensors can correct for gyro drift in the morning and evening when the sunline is normal to the earth radius vector along which the center body of the spacecraft is pointing.

6.3 Orbit Raising Thrusters. - The orbit raising mercury thrusters are scaled up from the flight-tested 6.25×10^{-3} pounds thrust SERT II configuration to the required thrust level of 22×10^{-3} pounds.

Some individual thruster load requirements are supplied from a common array supply; while sensing, protection, and control of these inputs is on an individual thruster basis. Available power will decrease as the orbit is raised, due to initial solar array degradation.

The thruster will be throttled to match the available power.

Two aspects of the thruster system differ from the flight-tested configuration. The propellant tank is electrically isolated from the vapor sections of the propellant feed system which permits the use of a common propellant tankage. Insulated optics (glass coated accelerator grid) enable thruster operation over the required ranges of thrust, specific impulse, and input power.

The thruster configuration is shown schematically in Figure 24. Its operation is as follows: Liquid propellant is fed by positive displacement to the porous metal phase separators. The phase separators are heated to pass a controlled flow of mercury vapor through the isolators to the thruster hollow cathode and the thruster discharge chamber simultaneously. A discharge from the hollow cathode to the anode is maintained. An interaction between the discharge and the vapor in the discharge chamber creates a mercury plasma. The magnetic field is provided to improve the efficiency of the ionization process. The baffle and magnetic shaping pole pieces produce desirable characteristics of the discharge. The plasma ions diffuse to the accelerator and are extracted and focused to provide thrust. The ion beam is neutralized by an equal current of electrons provided by the hollow cathode neutralizer. A ground screen encloses the thruster to prevent ambient electrons from streaming to thruster components at positive high voltage. Control loops maintain desired thruster operation.

The three mercury bombardment ion engines used for ascent can be positioned with their thrust axes parallel to the longitudinal axis of the main body. Varying the magnitude of their thrust vectors, say ± 10 percent about their nominal rating, would then produce attitude control torques which could be used as a redundant backup for four of the cesium strip engines.

6.4 Cesium Attitude Control Thrusters. - The eight cesium ion thrusters are capable of being thrust vectored. Six are mounted on the outboard ends of the solar array and two on the main body, as shown in Figure 23. The center body engines produce torque about the array shaft axis for attitude control purposes and also are used for east-west station keeping. The central engines on the array tips produce torques about an axis normal to the array for attitude control purposes. The outboard engines are redundant and produce torques about the longitudinal axis of the center body for attitude control purposes. All six array-mounted engines also are used for north-south station keeping.

Each engine produces 3×10^{-4} pounds thrust at a specific impulse of 5000 seconds and can be electrically thrust vectored through $\pm$ 20 degrees in one plane only. Power consumption is 100 watts per thruster. The array mounted engines can produce torques of 128×10^{-4} foot-pound, the body mounted engines 5×10^{-4} foot-pound. These figures are for two engines, one vectored +20 degrees, the other -20 degrees. Six engines can be operated continuously for the 160-day ascent period, consuming only 5 pounds of propellant. Once on orbit, a limit cycle operation can be initiated to conserve propellant.

For north-south station keeping, any two of the three engines on the ends of the array would be used without vectoring, giving 6×10^{-4} pounds thrust, or 3.74×10^{-7} g acceleration. The duty cycle would be such that the north thrusters were on for 5.25 hours each day, and the south thrusters for 5.25 hours per day, with 6.75 hours off between firing periods.

For east-west station keeping, the main body engines furnish 3×10^{-4} pounds thrust or 1.87×10^{-7} g acceleration. With this acceleration level, a monthly correction would suffice.

6.5 Sensors. - The advanced O.G.O. horizon scanner is propesed for this mission because of the following characteristics: it is capable of adjusting to the wide variation in altitude during ascent with the electric thrusters; it has a null accuracy compatible with the beam pointing requirements; it contains some measure of internal redundancy; and most of the components of the sensor assembly have a flight background from the OGO program.

The system consists of four scanners, contained in two packages, and a central processing unit. Each scanner has a 1.2 degree field of view and searches over an 80 degree arc for the space-to-earth transition. Having located the horizon, it locks on and a dither motion is introduced in order to provide a 2400 Hz output signal whose DC component is a measure of the horizon location.

The system would be constructed so that at synchronous altitude the horizon is on the pick-off null, giving a system accuracy of 0.11

degrees (3 sigma). At lower altitudes, the accuracy would be degraded somewhat, since pick-off errors are greater off-null.

The four sun sensors mounted at the ends of the array as shown in Figure 23 are coarse sun locators having a field of view of 2π steradians. When all four are nulled, the sun must lie on the array normal. A fifth sensor, not shown in Figure 23, would be necessary to reject the backward null when the sun lies on the inactive side of the array.

The two sun sensors mounted on the horizon sensor platform cross zero when the projection of the longitudinal axis of the center body is parallel to the earth/sun line on the orbit plane. This occurs twice per orbit and the negative-going transition will be rejected producing a single timing pulse per orbit.

In synchronous equatorial orbit, the array will be erected normal to the orbit plane facing the sun while the center body rotates about the solar array axis once per day as the antenna tracks the earth. Twice per day, the yaw axis is in-line with the array normal. At these times the yaw error signal from the sun sensors is extremely weak, and at the spring and fall equinoxes, it vanishes completely. A single axis integrating gyro is proposed to provide yaw signals during these intervals.

The perferred unit for this application is the Honeywell GG334A gas bearing gyro. This unit shows the greatest promise for long lifetimes but, even so, multiple redundancy would be required to give adequate confidence of completing a five-year mission. Further analysis may show that year-round operation is not necessary. In this case, the gyros will be operated only in the spring and fall and the lifetime reliability problem will be alleviated.

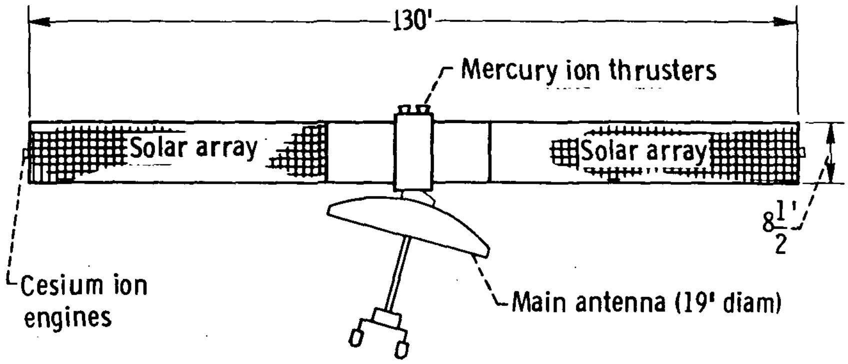


Figure 1. - Disaster warning satellite - orbital configuration.

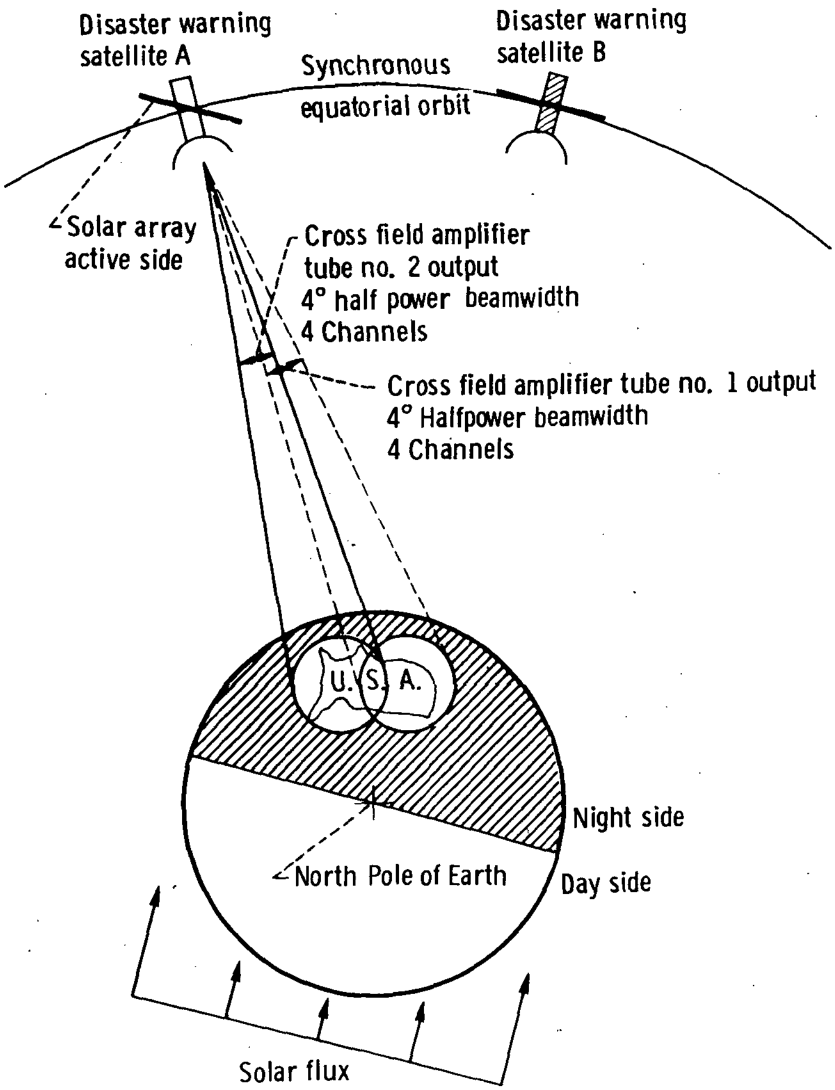


Figure 2. - Satellites in orbit. Satellite B is in Earth shadow while satellite A transmits to mainland.

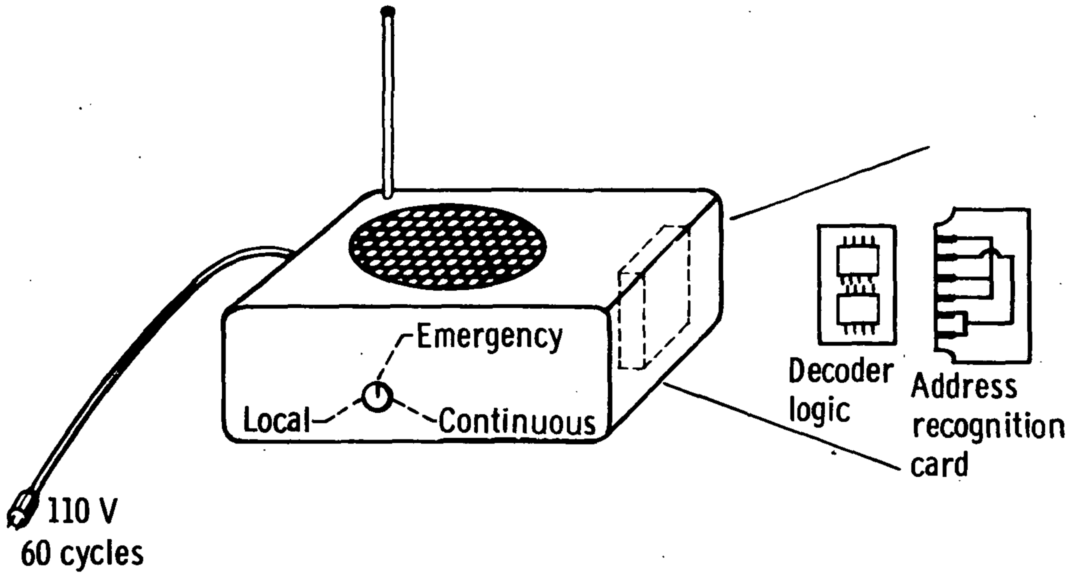


Figure 3. - UHF-FM voice channel receiver showing IC decoder logic and address recognition card plug-ins.

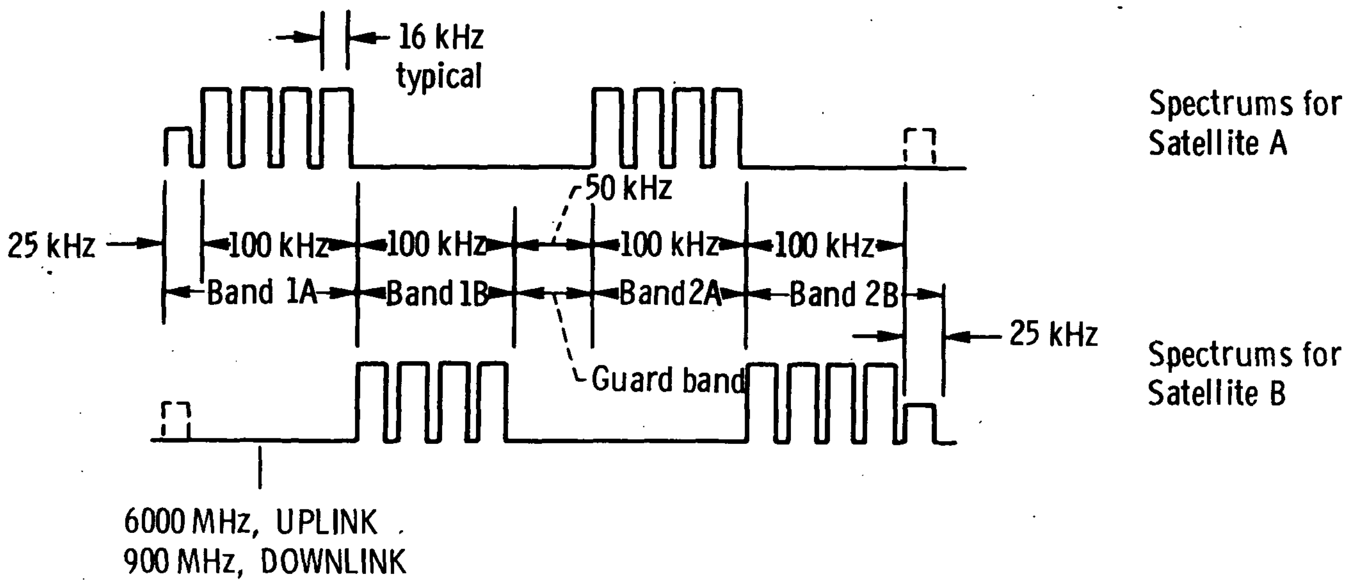


Figure 4. - Spectrums of the signals transmitted to and from each satellite.

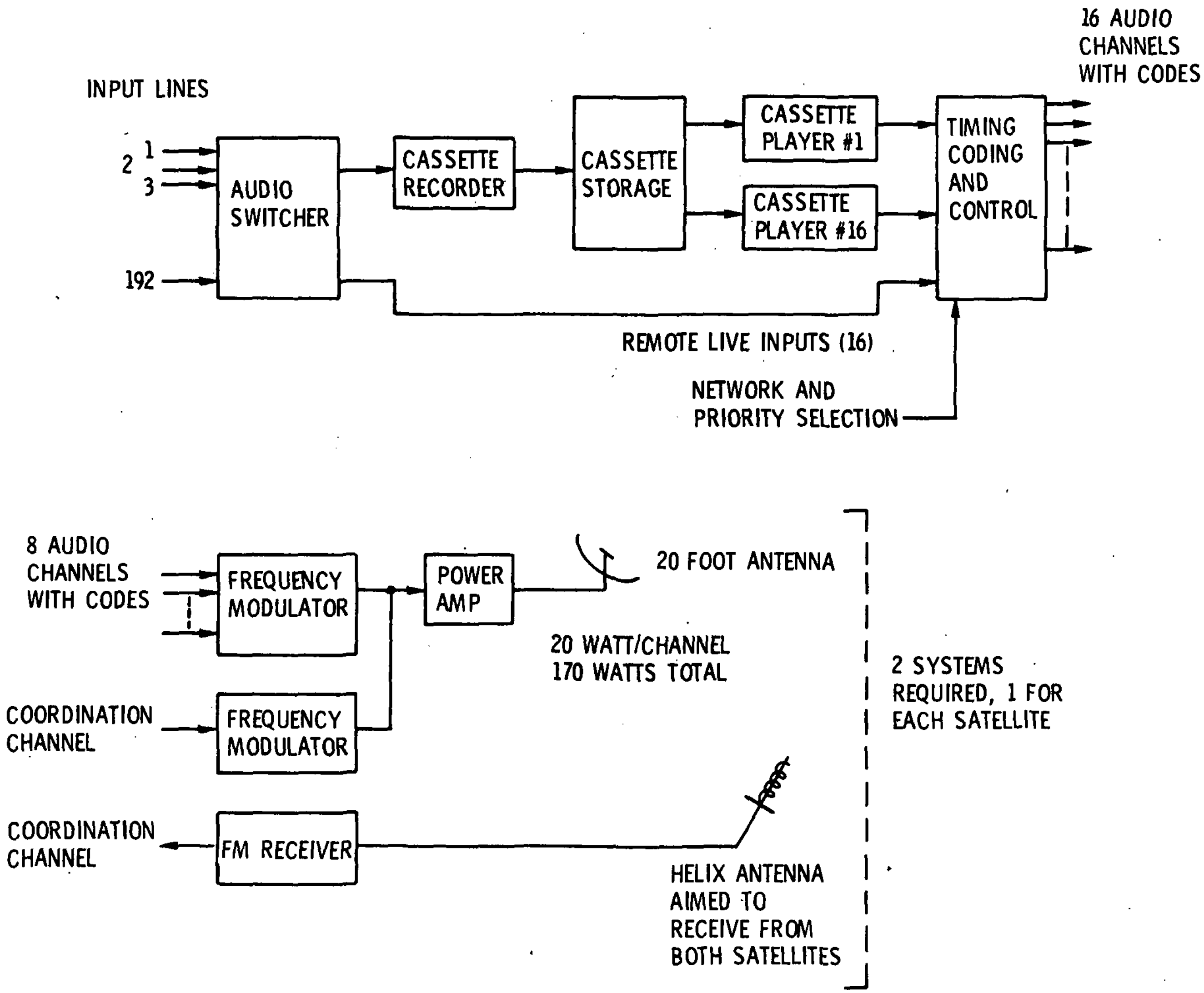


Figure 5. - Central uplink ground station for 16 audio channels (two satellites) and coordination channel.

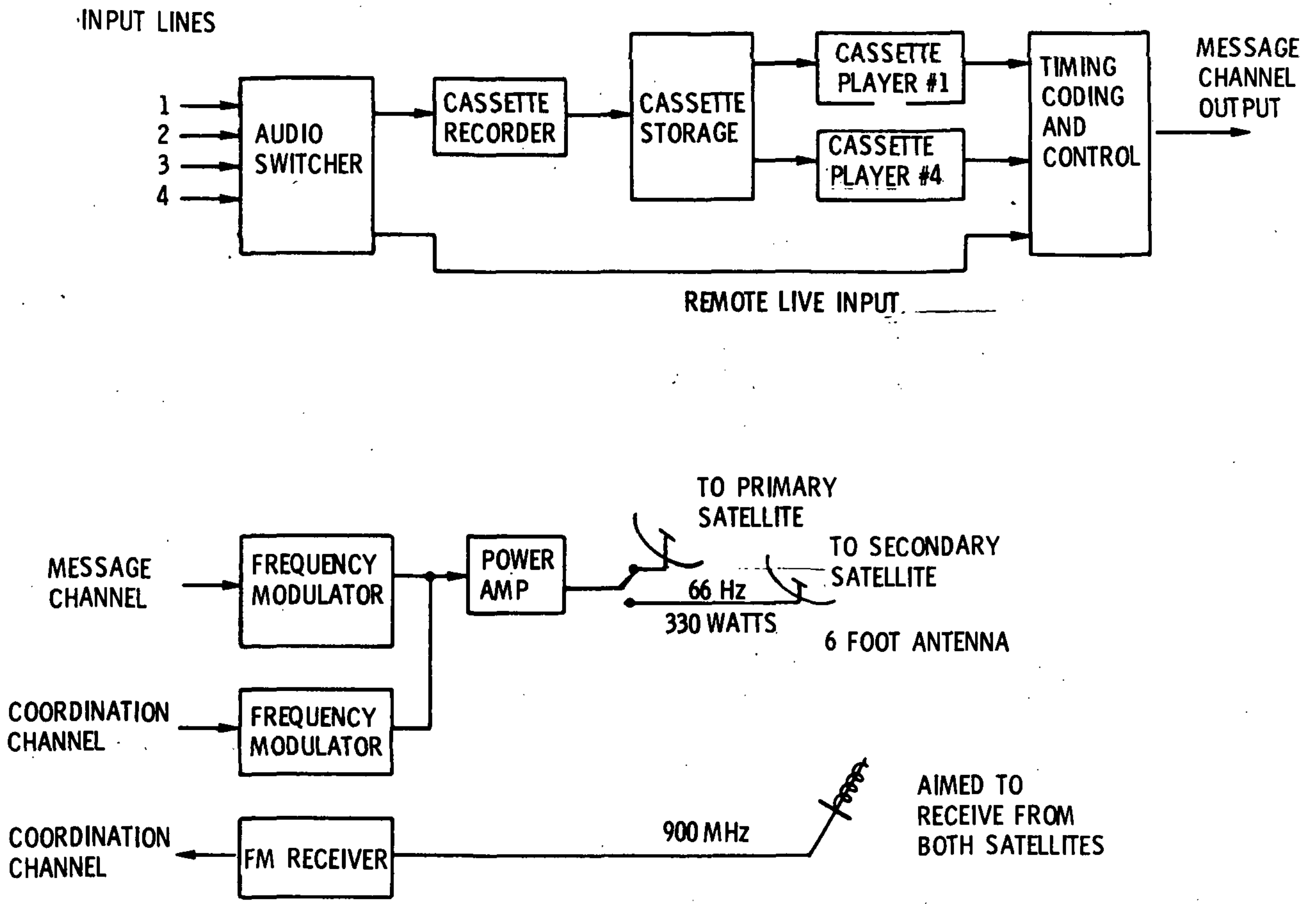


Figure 6. - Regional ground station for four subchannels and coordination channel.

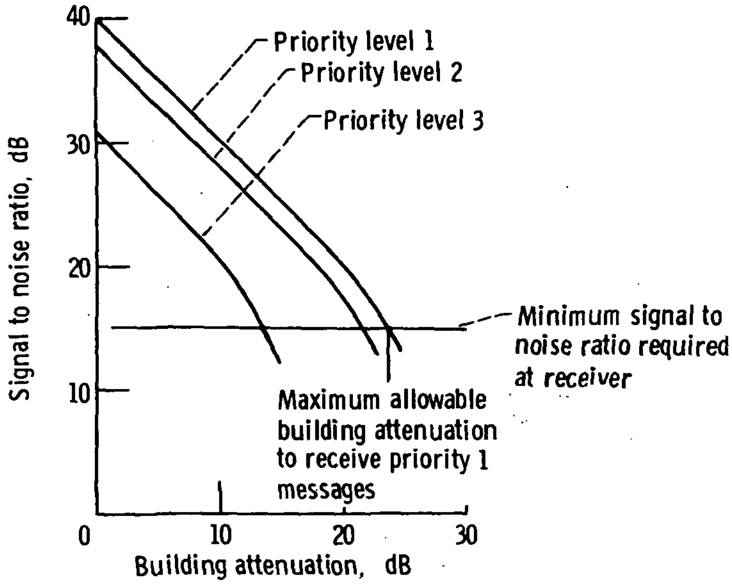


Figure 7. - Output signal to noise ratio versus building attenuation. Minimum required receiver output signal to noise ratio.

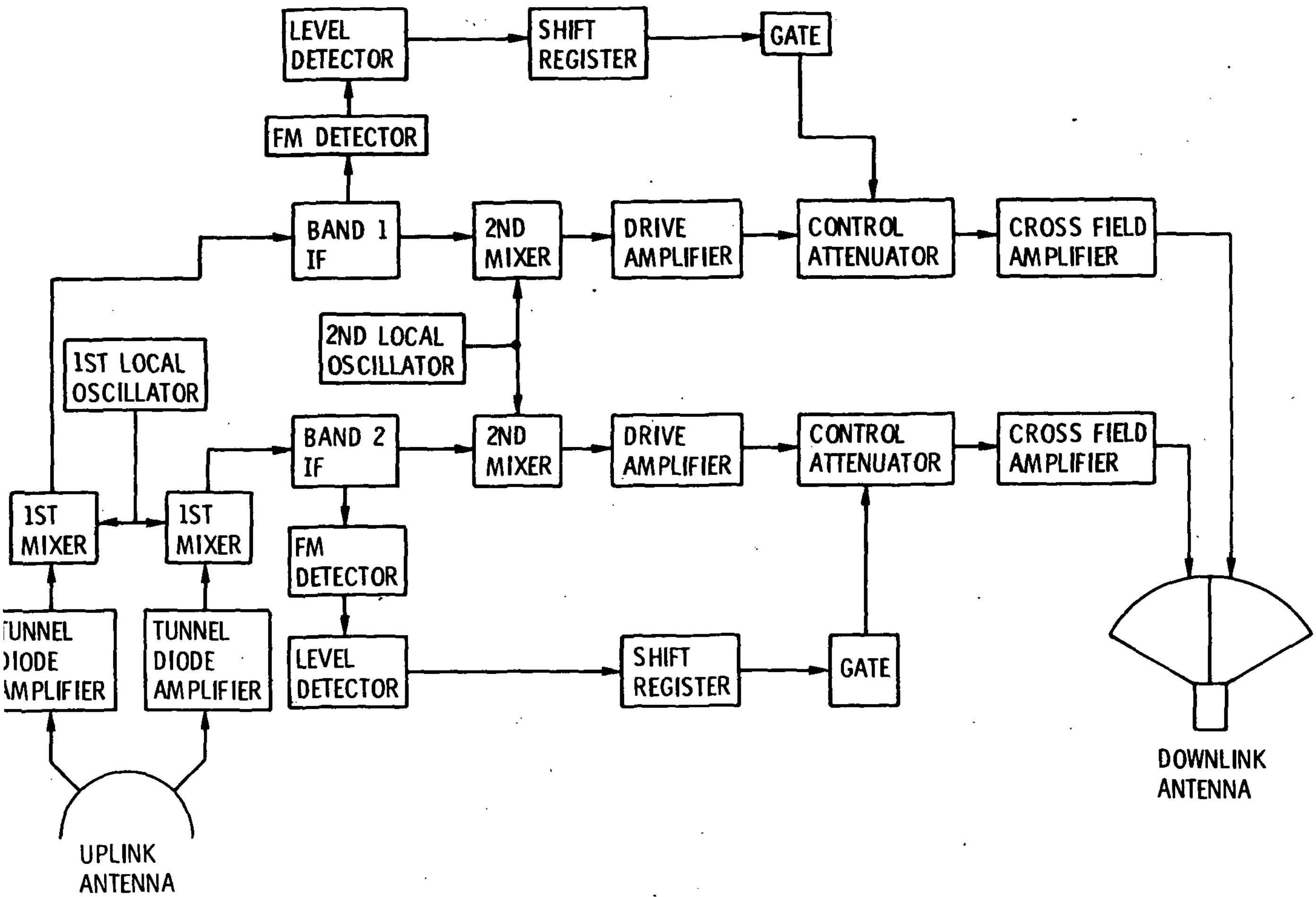


Figure 8. - Spacecraft communication system.

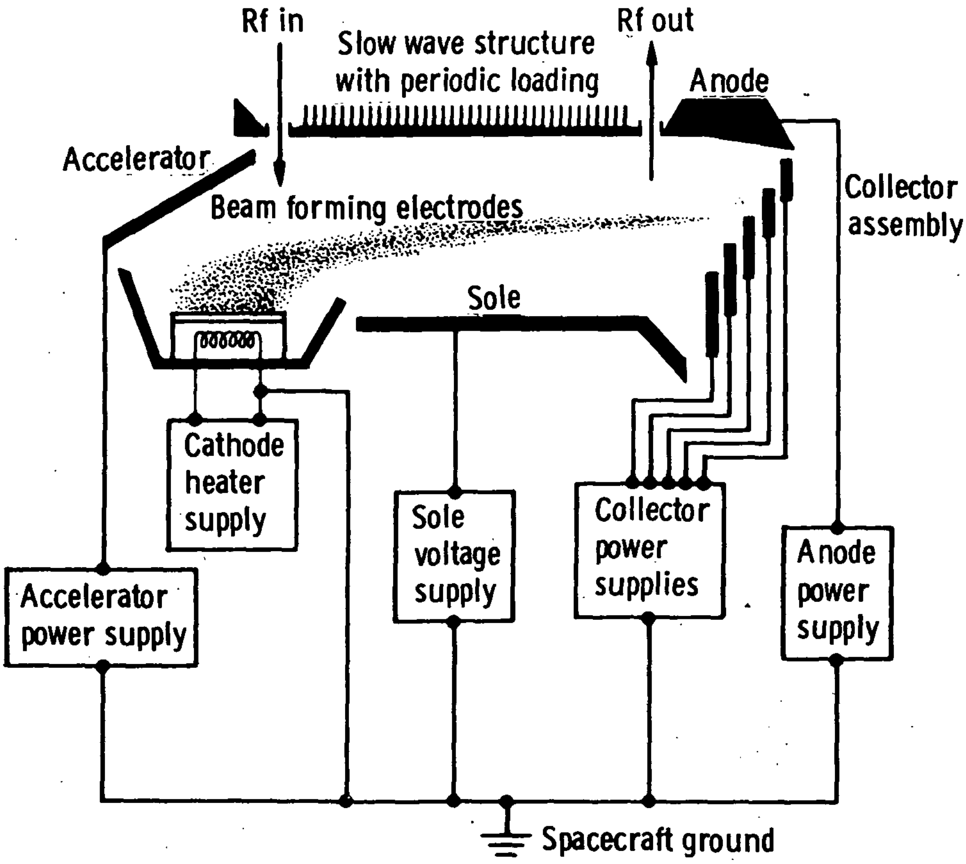


Figure 9. - Narrow beam cross field amplifier schematic.

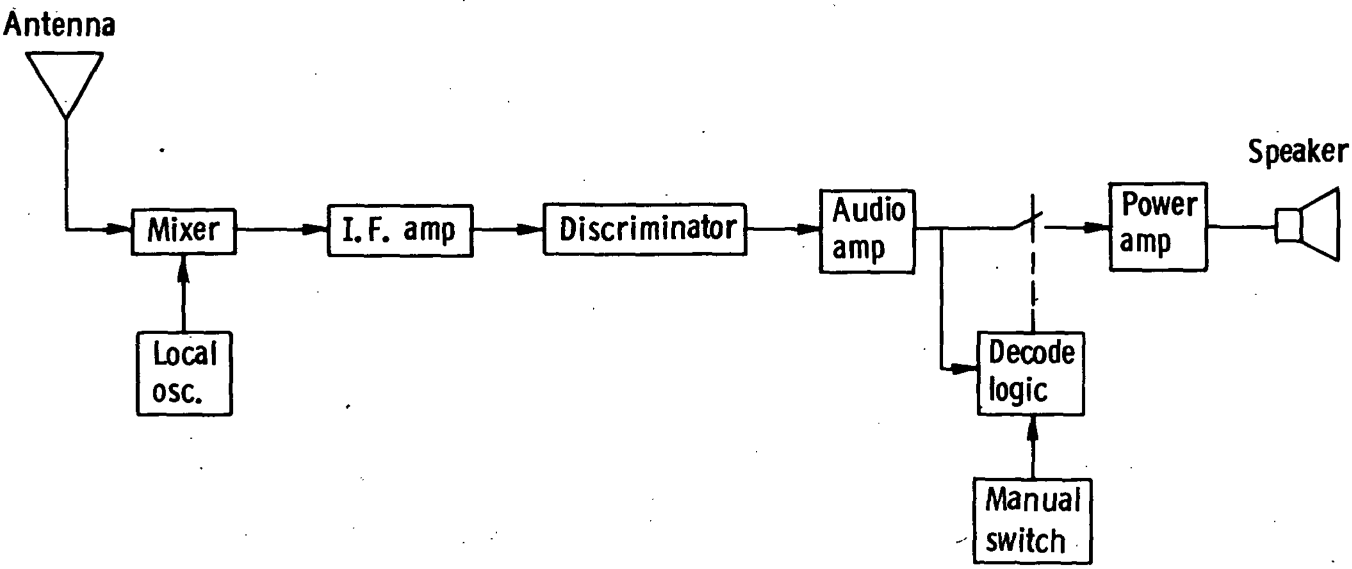


Figure 10. - Block diagram of emergency broadcast receiver.

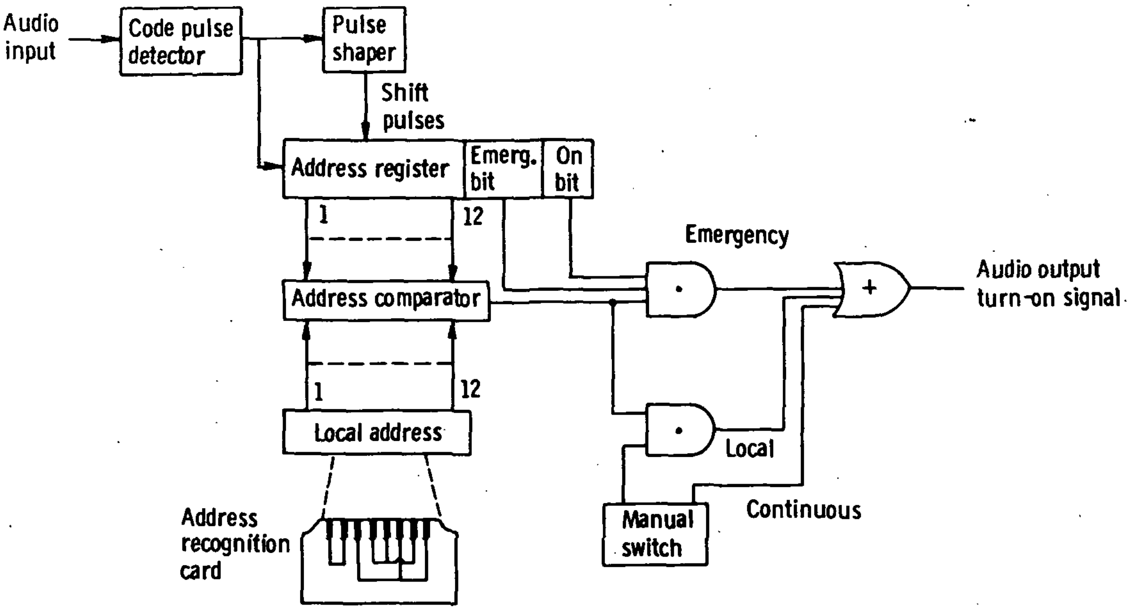


Figure 11. - Block diagram of receiver decoding logic.

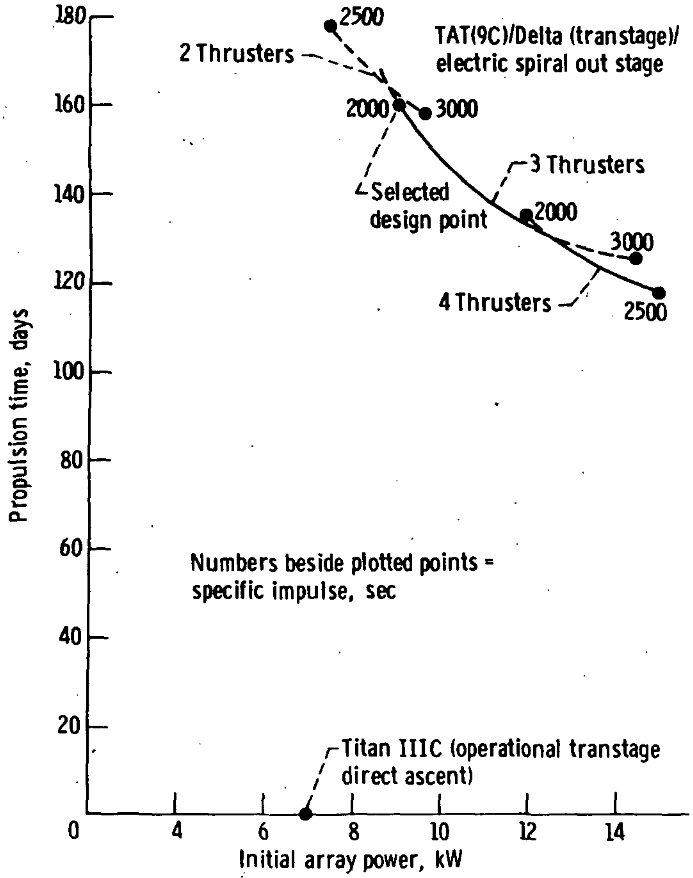


Figure 12. - Propulsion time to achieve synchronous altitude as a function of booster system and initial (undegraded) solar array size.

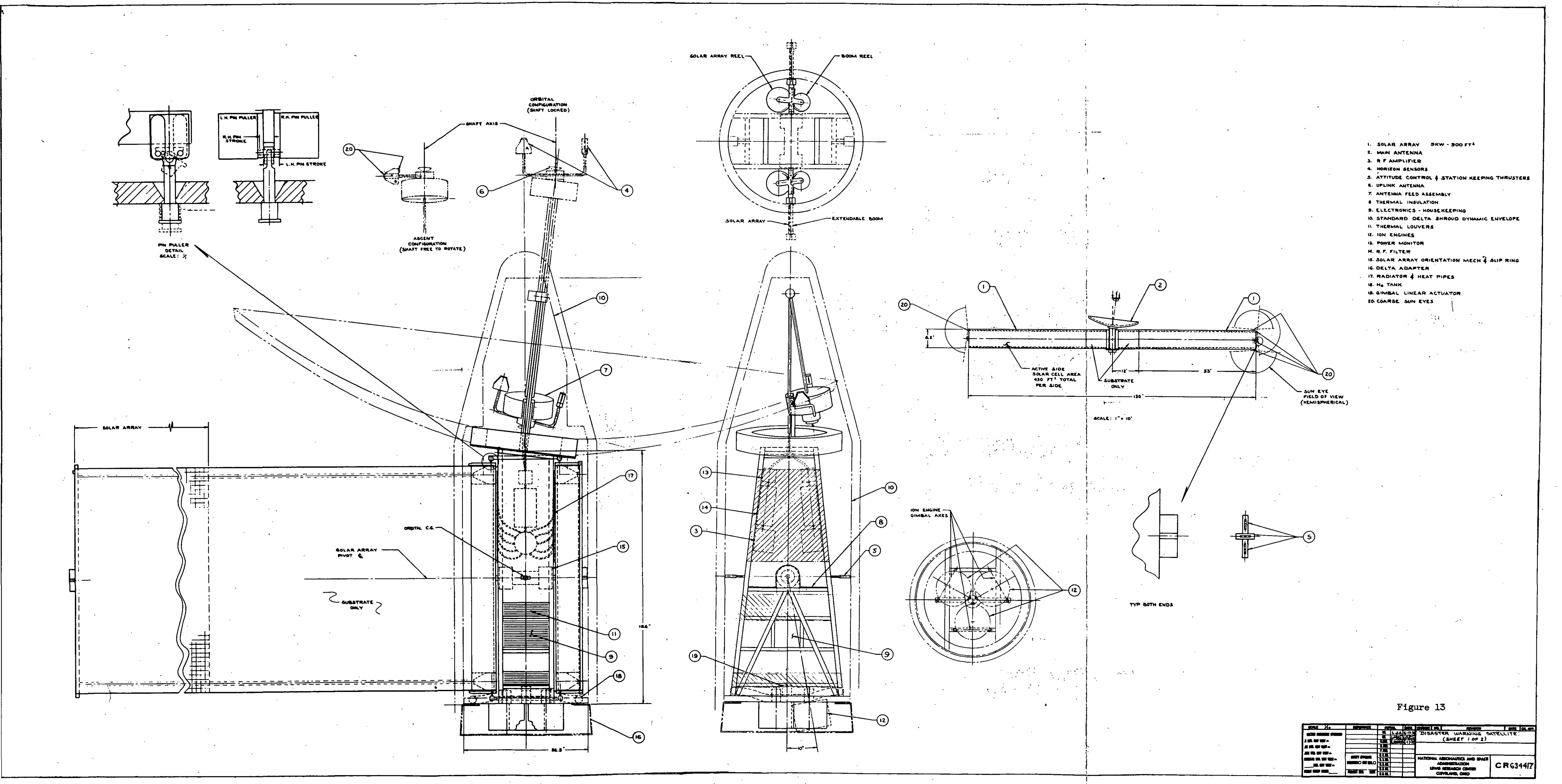
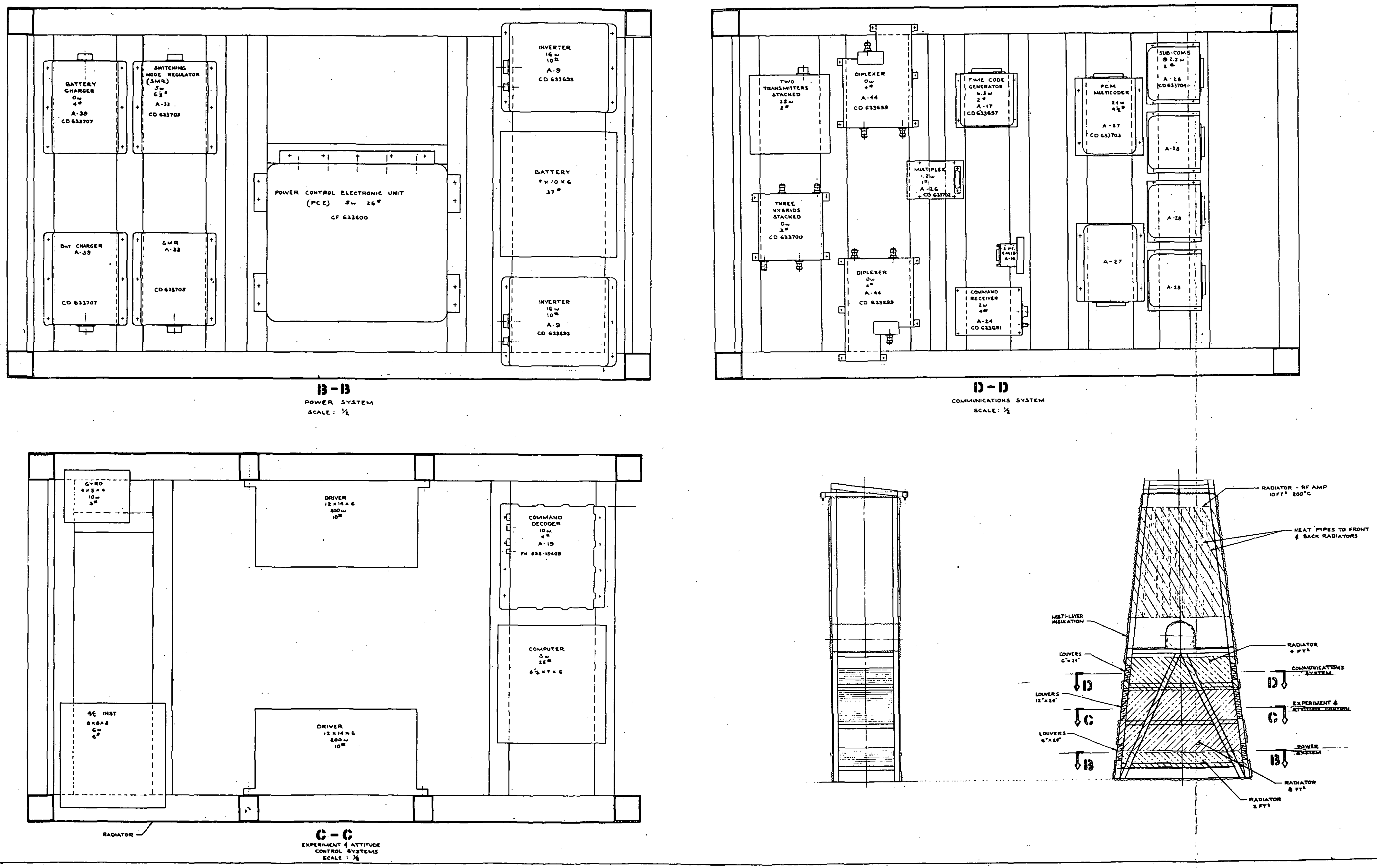

PIN PULLER DETAIL SCALE: ½
L.H. PIN PULLER
R.H. PIN PULLER
R.H. PIN STROKE
L.H. PIN STROKE
ORBITAL CONFIGURATION (SHAFT LOCKED)
SHAFT AXIS
ASCENT CONFIGURATION (SHAFT FREE TO ROTATE)
SOLAR ARRAY REEL
BOOM REEL
SOLAR ARRAY
EXTENDABLE BOOM
SOLAR ARRAY
ORBITAL C.G.
SOLAR ARRAY PIVOT ℄
SUBSTRATE ONLY
ION ENGINE GIMBAL AXES
ACTIVE SIDE SOLAR CELL AREA 450 FT² TOTAL PER SIDE
SUBSTRATE ONLY
SUN EYE FIELD OF VIEW (HEMISPHERICAL)
SCALE: 1" = 10'
TYP BOTH ENDS
1. SOLAR ARRAY 9KW - 900 FT²
2. MAIN ANTENNA
3. R F AMPLIFIER
4. HORIZON SENSORS
5. ATTITUDE CONTROL & STATION KEEPING THRUSTERS
6. UPLINK ANTENNA
7. ANTENNA FEED ASSEMBLY
8 THERMAL INSULATION
9. ELECTRONICS - HOUSEKEEPING
10. STANDARD DELTA SHROUD DYNAMIC ENVELOPE
11. THERMAL LOUVERS
12. ION ENGINES
13. POWER MONITOR
14. R.F. FILTER
15. SOLAR ARRAY ORIENTATION MECH & SLIP RING
16. DELTA ADAPTER
17. RADIATOR & HEAT PIPES
18. H₂ TANK
19. GIMBAL LINEAR ACTUATOR
20. COARSE SUN EYES
DISASTER WARNING SATELLITE (SHEET 1 OF 2)
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
LEWIS RESEARCH CENTER
CLEVELAND, OHIO
CR634417

Figure 13



CR634418

Figure 14

SCALE: ½	DISASTER WARNING SATELLITE (SHEET 2 OF 2)
	NATIONAL AERONAUTICS AND SPACE ADMINISTRATION LEWIS RESEARCH CENTER CLEVELAND, OHIO
	CR634418

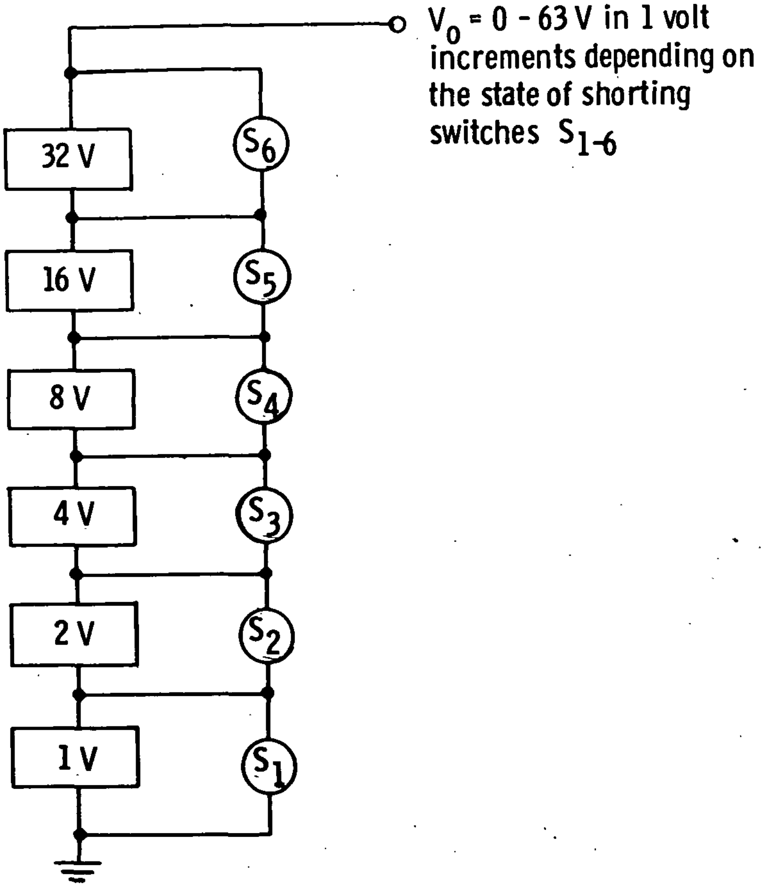
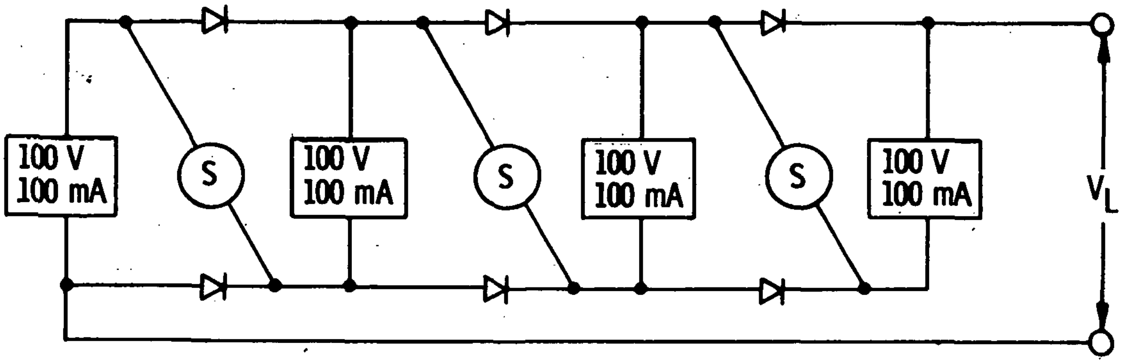


Figure 15. - Conceptual regulation scheme.



1. With all switches, S , open the parallel configuration exists and V_L is 100 V at 400 mA.
2. With all switches, S , closed the series configuration exists and V_L is 400 V at 100 mA.

Figure 16. - Conceptual reconfiguration switching design.

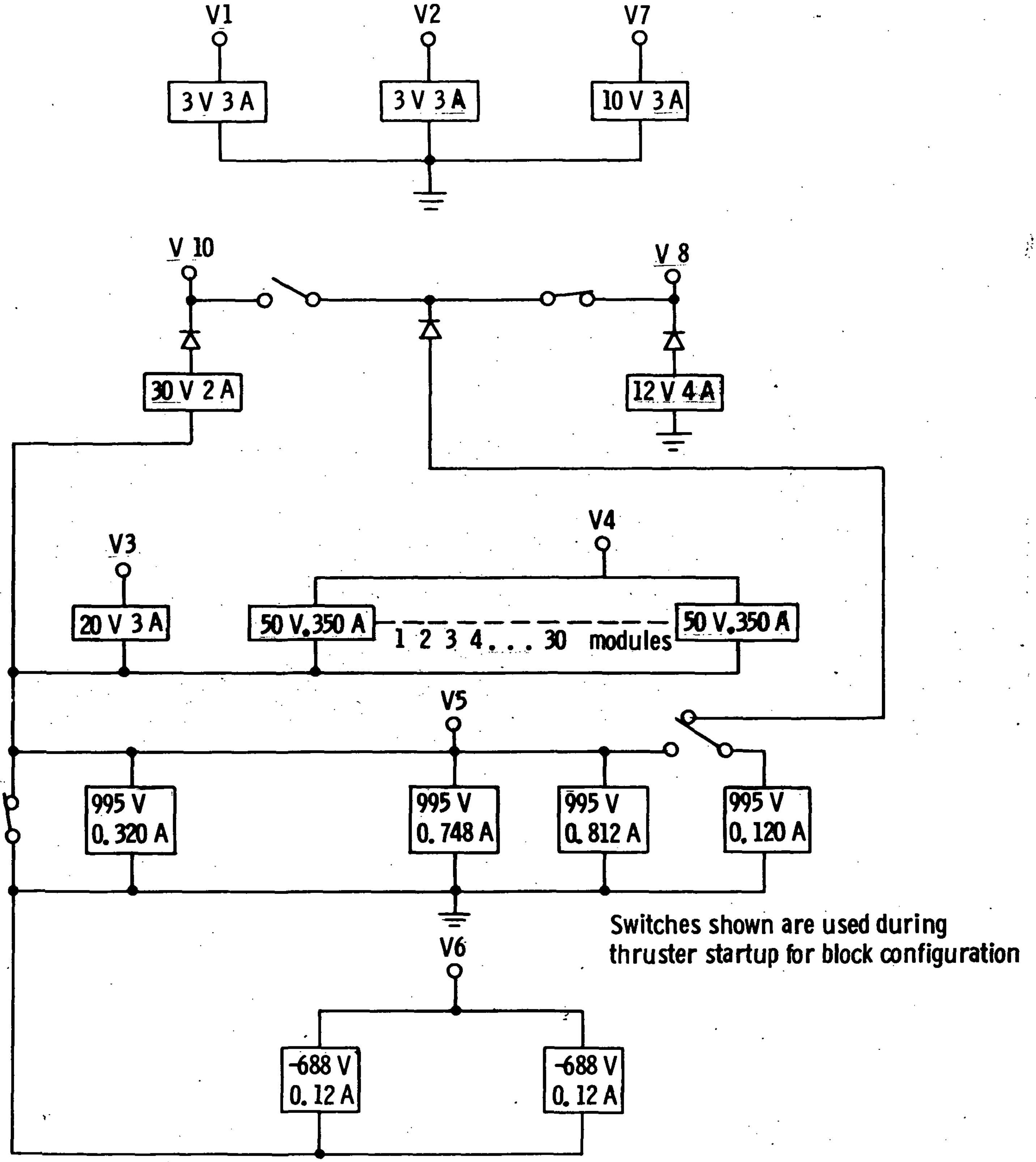
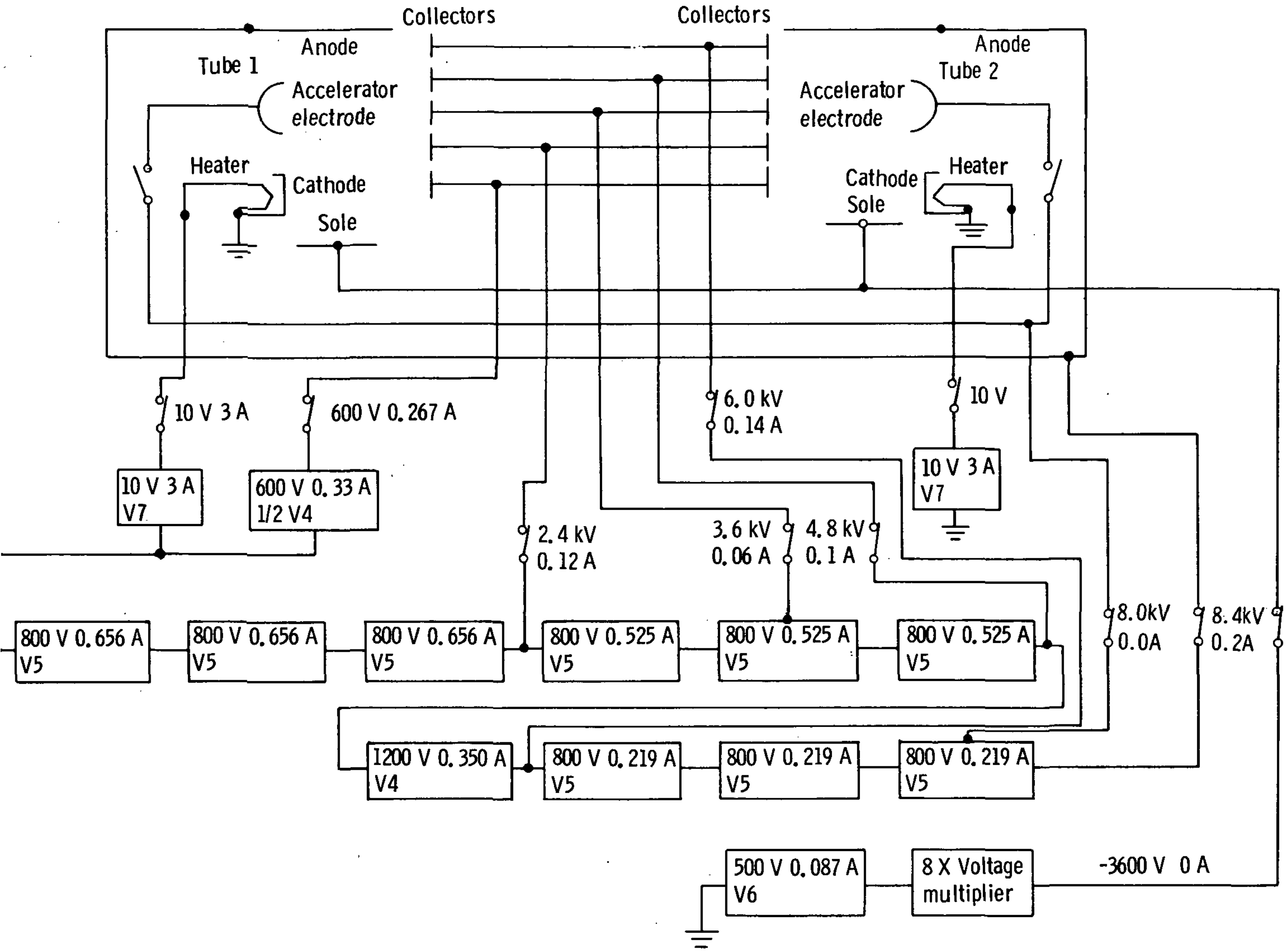


Figure 17. - Array configuration at beginning of mission for a mercury thruster.



Array used: 3 - V5 sections 1 - V6 section
$1\frac{1}{2}$ - V4 sections 2 - V7 sections

Figure 18. - Hi-voltage array configuration for 2-CFA-UHF tubes.

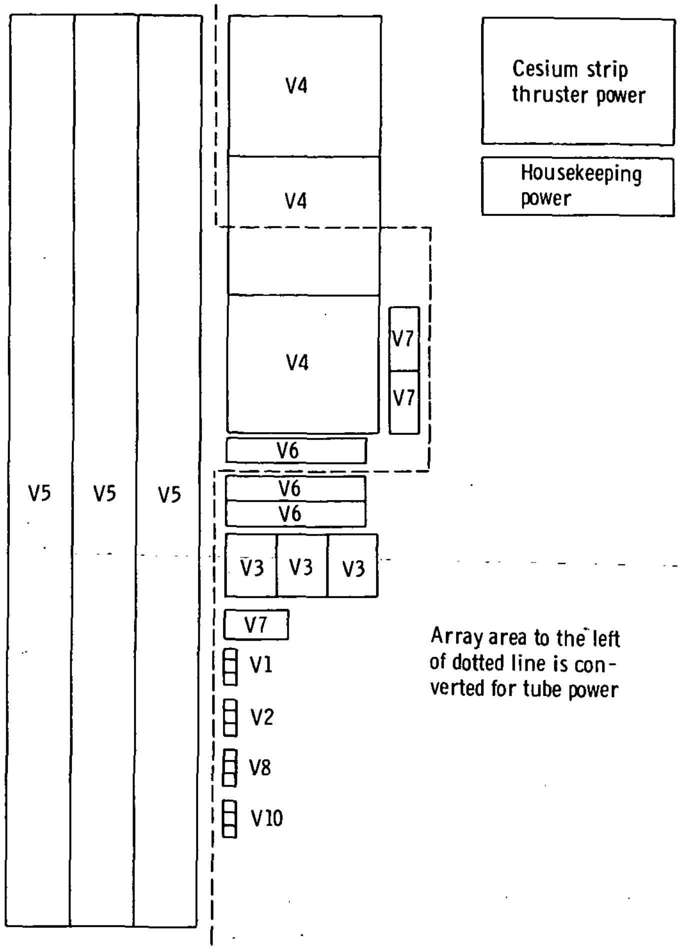


Figure 19. - Relative array area.

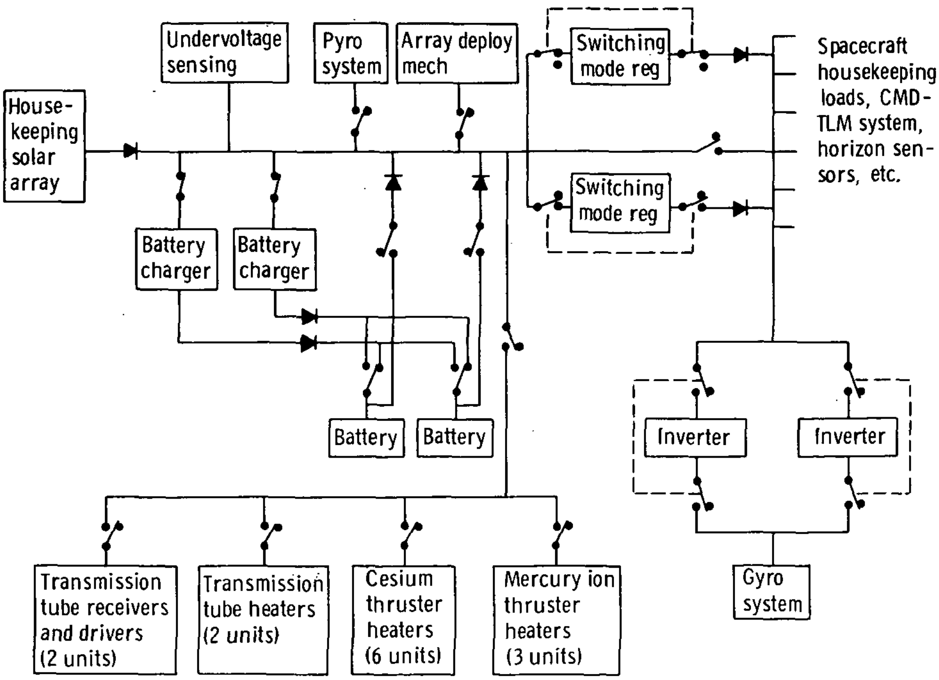


Figure 20. - Housekeeping power system block diagram.

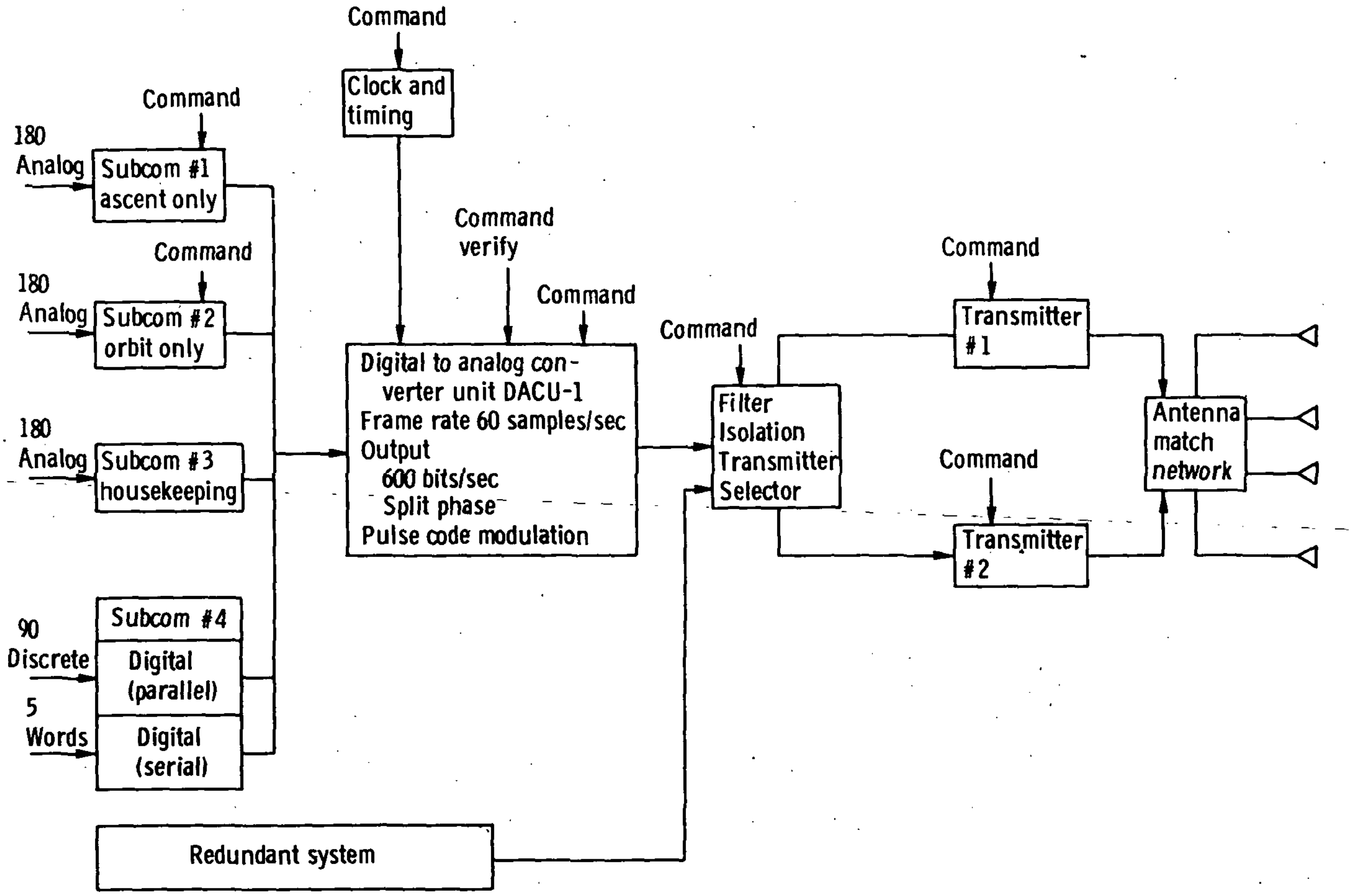


Figure 21. - Telemetry data system block diagram.

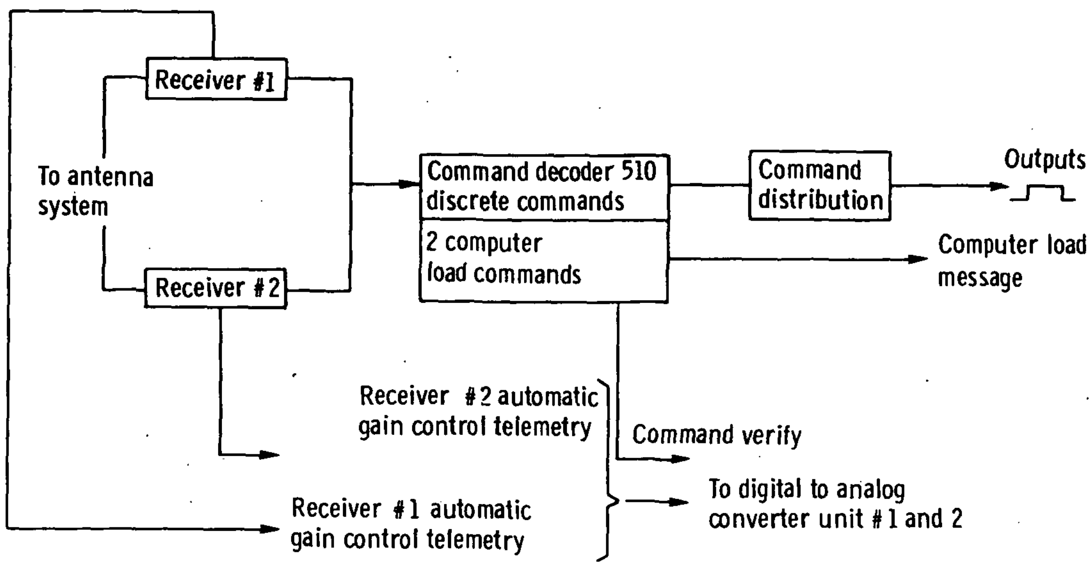


Figure 22. - Command system block diagram.

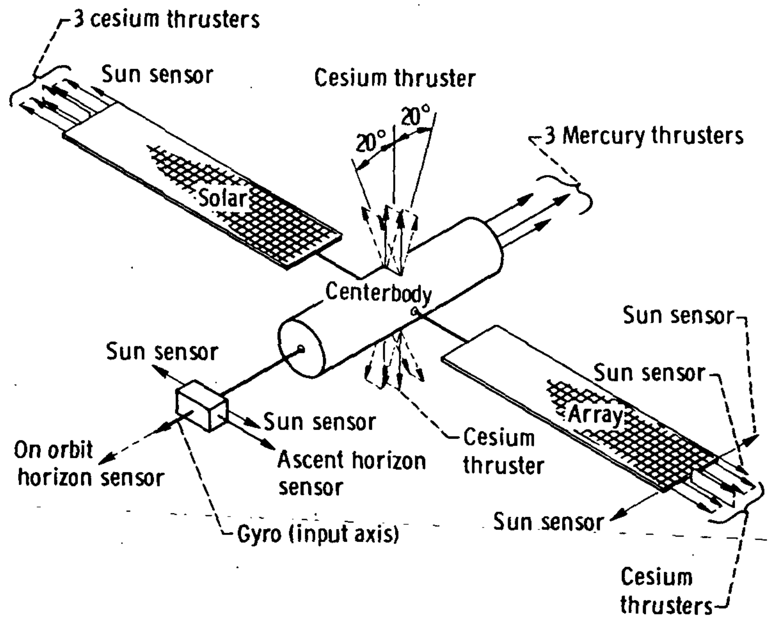


Figure 23. - Attitude control component location.

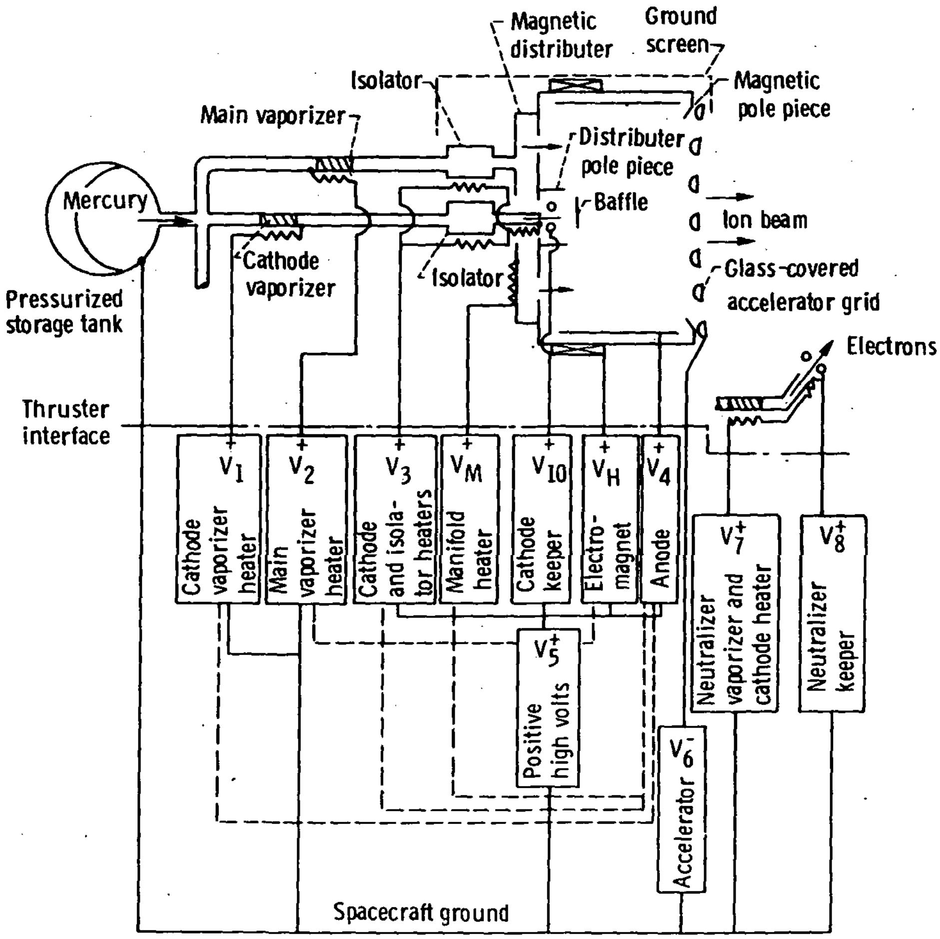


Figure 24. - Orbit raising mercury thruster diagram.

APPENDIX A

Considerations of Expanding Coverage to Alaska and Hawaii

This appendix contains information on changes that are required in the Disaster Warning Satellite concept in order to increase capability from coverage of the contiguous forty-eight states to include coverage of Alaska and Hawaii. Inclusion of these widely separated areas results in a change in system implementation which requires additional regional ground stations and additional area addresses to be handled. Because these areas are so widely separated from the mainland, their inclusion requires two additional rf beams (for a total of four) along with revised antenna beam width specifications and the use of different satellite station locations. Addition of the spacecraft communication system equipment to accommodate this capability results in a spacecraft weight increase of 145 pounds. This weight increase results in a 20 day increase (from 160 to 180 days) in the propulsion time required to spiral to synchronous altitude using electric thrusters.

A-1 Implementation

The Disaster Warning System, with expanded capability, to cover Hawaii and Alaska, consists of modifications to the basic approach as described in the main document. The following changes have been made in the system. Regional ground control stations are added in Alaska and Hawaii for uplink transmission to the spacecraft. Home receivers remain unchanged, except for the addition of address codes for Alaska and Hawaii. The spacecraft transponders are changed to permit the addition of two information channels to either spacecraft bringing the total for both spacecraft to 18 full time channels. The spacecraft downlink antennas must provide up to four, 4-1/2° half power beamwidth (HPBW) patterns from each spacecraft. Satellite spacing in geostationary orbit is now 20° instead of 30° with spacecraft A at 120°W and spacecraft B at 100°W longitude. The additional two channels in the system are available for time sharing as previously, thus bringing the total number of time shared subchannels for routine operation from 192 to 216. Capability exists for shifting channels from one spacecraft to the other in case of eclipse to preserve overall capability of the system to transmit priority 1 and 2 level messages. Also, as in the basic configuration, capability exists to communicate between the regional ground stations and the central ground station using a link through a Disaster Warning Satellite. This is explained further in the spacecraft communications system section.

A-2 Antenna Coverage

Because of the wide geographical separation between Alaska, Hawaii and the U.S. mainland, it was decided to add two additional beams (for a total of four) to the satellite capability and increase the HPBW from 4 to 4-1/2°. This results in an antenna diameter of 17 feet rather than 19 feet. Figures A-1 and A-2 shows the coverage resulting from spacecraft located at 120°W and 100°W longitude respectively. These specific station locations were selected as having enough separation so that the earth could only eclipse one at a time and yet being close enough together so that the extremities of the total U.S. can be covered. Figure A-3 contains the locus of 5° look angles in Alaska and 10° look angles in northeast Maine for satellites at various longitude locations. Five degrees in Alaska and 10 degrees in Maine were chosen as Alaska has few tall buildings to limit reception in inhabited areas.

A-3 Spacecraft Communications System

The spacecraft communication system is composed of two identical transponder units, one on spacecraft A and one on spacecraft B. They differ, in operation, only in the selected downlink signal which they process and transmit as weather information. This difference is determined by ground command and in this manner permits spacecraft A or spacecraft B to provide continuing capability during satellite eclipse periods.

The spacecraft transponders are used for weather information broadcast as priority level 1, 2, or 3, and also are used for communication between regional ground stations and the central ground station.

The uplink signals transmitted to the spacecraft use a total frequency band of 440 kHz. Four information bands are used, and are centered at 6000.1, 6000.2, 6000.3 and 6000.4 MHz. The signal content of these bands as received by spacecraft A and spacecraft B is shown in figures A-4 and A-5 respectively. It should be noted that the "A" bands can be received by the "B" spacecraft and the "B" bands processed by the "A" spacecraft when properly configured via ground command. In fact, this operation will be routinely applied to cover during eclipses. Bands 1A and 4B each contain 5 frequency modulated carriers with 9 kHz spacing between bands as shown. Bands 2B and 3A contain four FM carriers with 9 kHz spacing betwern bands. Bands 2A and 4A contain only 1 each with 46 MHz spacing. One channel in band 1B is used during eclipse of spacecraft A. The pertinent characteristics of the information bands are summarized in Table A-1 below.

TABLE A-1

Information Band Characteristics

Information Band	Coverage Area	Uplink Center Frequency MHz	Downlink Center Frequency MHz	Bandwidth MHz	Number of Channels
1A	U.S. Main East	6000.088	900.088	120	5
2A	Alaska	6000.2	910.0	100	1
3A	U.S. Main West	6000.3	900.3	100	4
4A	Hawaii	6000.412	909.788	120	1
*1B	U.S. Main East	6000.088	900.088	120	1
2B	U.S. Main East	6000.2	900.2	100	4
3B	-	-	-	-	-
4B	U.S. Main West	6000.412	900.412	120	5

*used during eclipse of spacecraft A for coordination only

The signals of 1A through 4A of spacecraft B are 25 dB below the signals shown in figure A-5. In like manner, the signals of 2B and 4B at spacecraft A are 25 dB below the signals shown in figure A-4. This is the result of the 20° spacing of spacecrafts A and B in synchronous orbit, and the narrow, $1/2^{\circ}$ HPBW of the ground antennas used for uplink transmission to the spacecraft.

The spacecraft transponders are shown in figure A-6. The uplink signals received are amplified using a tunnel diode amplifier, and translated in the first mixer to IF frequencies of 4.1, 4.2, 4.3, and 4.4 MHz. The IF bandwidths are 120 kHz each. There are four parallel signal paths consisting of IF amplifier, second mixer, filter, and control attenuator. In the following explanation these signal paths will be referred to as 1, 2, 3, and 4 from upper most to lowest path in figure A-6.

In spacecraft A, the signals in information bands 1A and 3A after translation to IF frequencies of 4.1 and 4.3 MHz follow paths 1 and 3. Here they are amplified, and translated in second mixers to downlink frequencies of 900.1 and 900.3 MHz. A local oscillator frequency of 896 MHz is used for path 1 and 3 second mixers. The signals in bands 2A and 4A after translation to IF frequencies of 4.2 and 4.4 MHz in the first mixer follow paths 2 and 4, are amplified and translated in second mixers to downlink frequencies of 910.0 and 909.8 MHz. A local oscillator frequency of 914 MHz is used for mixers in paths 2 and 4.

The driver signal levels for the CFA are set by ground command signals, which bias control attenuators. The level of the driver signals are determined by the message priority level as previously described.

The output of paths 1 and 2 are combined and amplified and drive the upper CFA. This composite signal, composed of the 900.1 MHz and 910 MHz downlink information bands, is amplified in the CFA. The bands are separated in an output diplexer and used to drive antenna feeds F1 and F2. Similarly the output of paths 3 and 4 are combined, amplified, and separated. These information bands, 900.3 and 909.8 MHz drive antenna feeds 3 and 4 respectively. The coverage of antenna beam patterns F1 through F4 are U.S. Mainland East, Alaska, U.S. Mainland West, and Hawaii respectively.

In spacecraft B the signals in information bands 2B and 4B, after translation to IF frequencies of 4.2 and 4.4 MHz, follow paths 2 and 4. They are amplified and translated in second mixers to downlink frequencies of 900.2 and 900.4 MHz. A local oscillator frequency of 896 MHz is used for the second mixer in paths 2 and 4 on spacecraft B. The drive signal levels for the CFA are set by ground command.

The output of paths 2 and 4 are amplified and drive the CFA's. The output of the CFA's drive feeds F2 and F4. Paths 1 and 3 are used during eclipse of spacecraft A.

The transponder operational configurations and corresponding descriptions may be interchanged by (1) switching the local oscillator frequency for the second mixer, and (2) transmitting bands 1A through 4A to spacecraft B while spacecraft A is eclipsed; or transmitting to spacecraft A bands 1B through 4B while spacecraft B is eclipsed. By ground control of uplink signal transmissions and selection of spacecraft local oscillator frequency, complete coverage capability for priority 1 and 2 messages is maintained during eclipse periods.

The channels within the information bands used for coordination communications are shown in Table A-2. The power level used for these channels can be 3 dB below the adjacent weather channels since fixed higher gain antennas are used by regional ground stations and no allowance for building attenuation is required:

TABLE A-2

Communications Channels for Coordination

Information Band	From Control Station to:	To Control Station from:	Time
1A	U.S. Main East	All Areas	Full
2A	Alaska	-	Shared
3A	U.S. Main West	-	During Eclipse only Shared
4A	Hawaii	-	Shared
1B	U.S. Main East	All Areas	Eclipse Only
4B	U.S. Main West	-	Full

A-4 Impact on Spacecraft Design

In order to accomplish the extended coverage of the system to include Alaska and Hawaii some revisions are required to spacecraft systems other than those of the communications system previously described. The addition of two feeds to the antenna together with the other components required to complete the 4 IF chains instead of 2 will require the addition of 145 pounds to the spacecraft. If the Thor/Delta mission with electric thrusters used for spiral out is used for ascent, 20 additional days for propulsion time must be allowed (for a total of 180 days). A single spacecraft on a Titan 3C type mission would not be affected since a performance margin of approximately 1258 pounds existed for this launch vehicle.

Also the spacecraft power system would require an additional 50 watts from the solar array to power the additional hardware. This is not prohibitive when added to the 5000 watt array.

Although it has not been studied in detail, the attitude control system "boresight" accuracy requirement would be more stringent (on the order of 1^{o}) if Alaska and Hawaii coverage are required. This results from the large dimensions of the combined radiation pattern where small vehicle angular offsets result in comparatively large misalignment of the beam pattern on the earth.

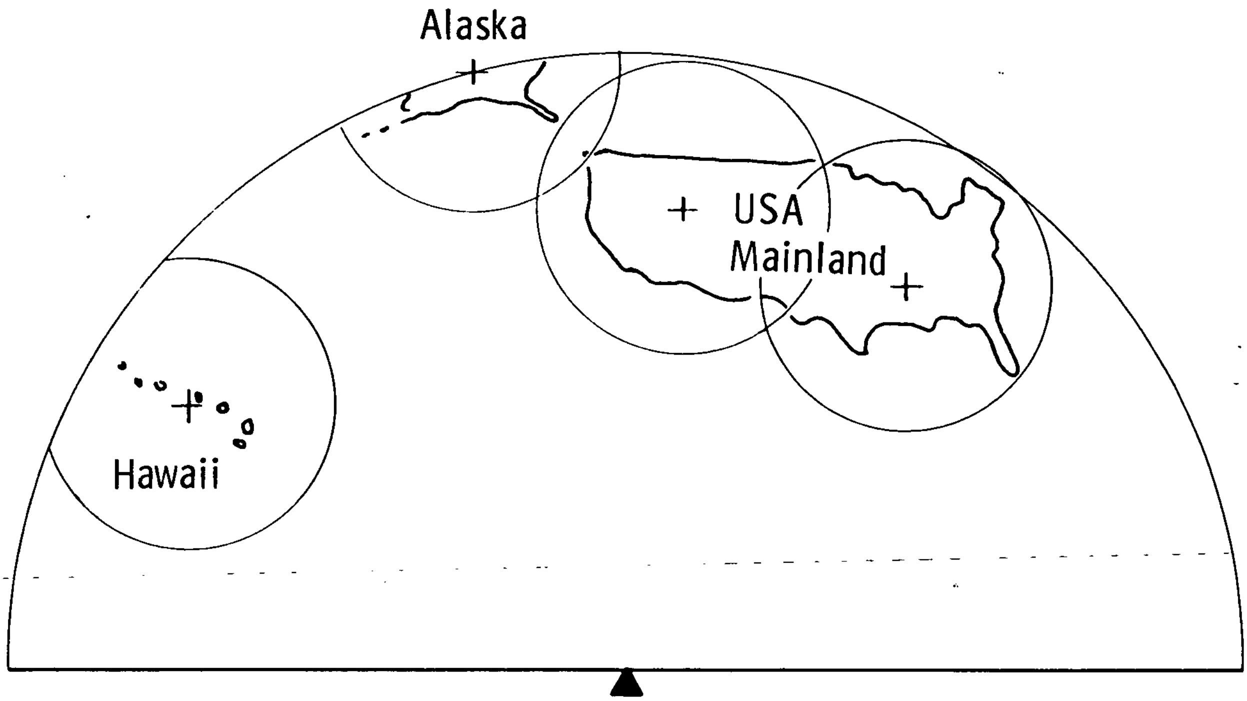


Figure A-1. - 3 dB beam coverage contours from Spacecraft A orbit station 120° W longitude.

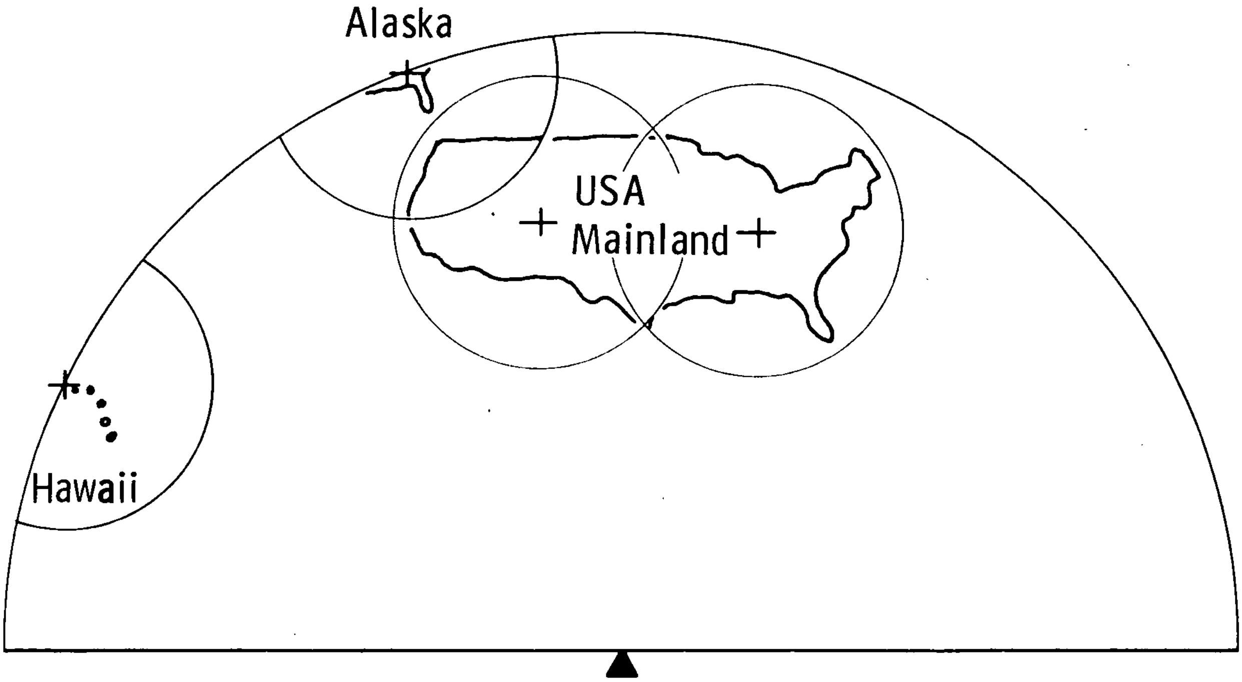


Figure A-2. - 3 dB beam coverage contours from Spacecraft B orbit station 100° W longitude.

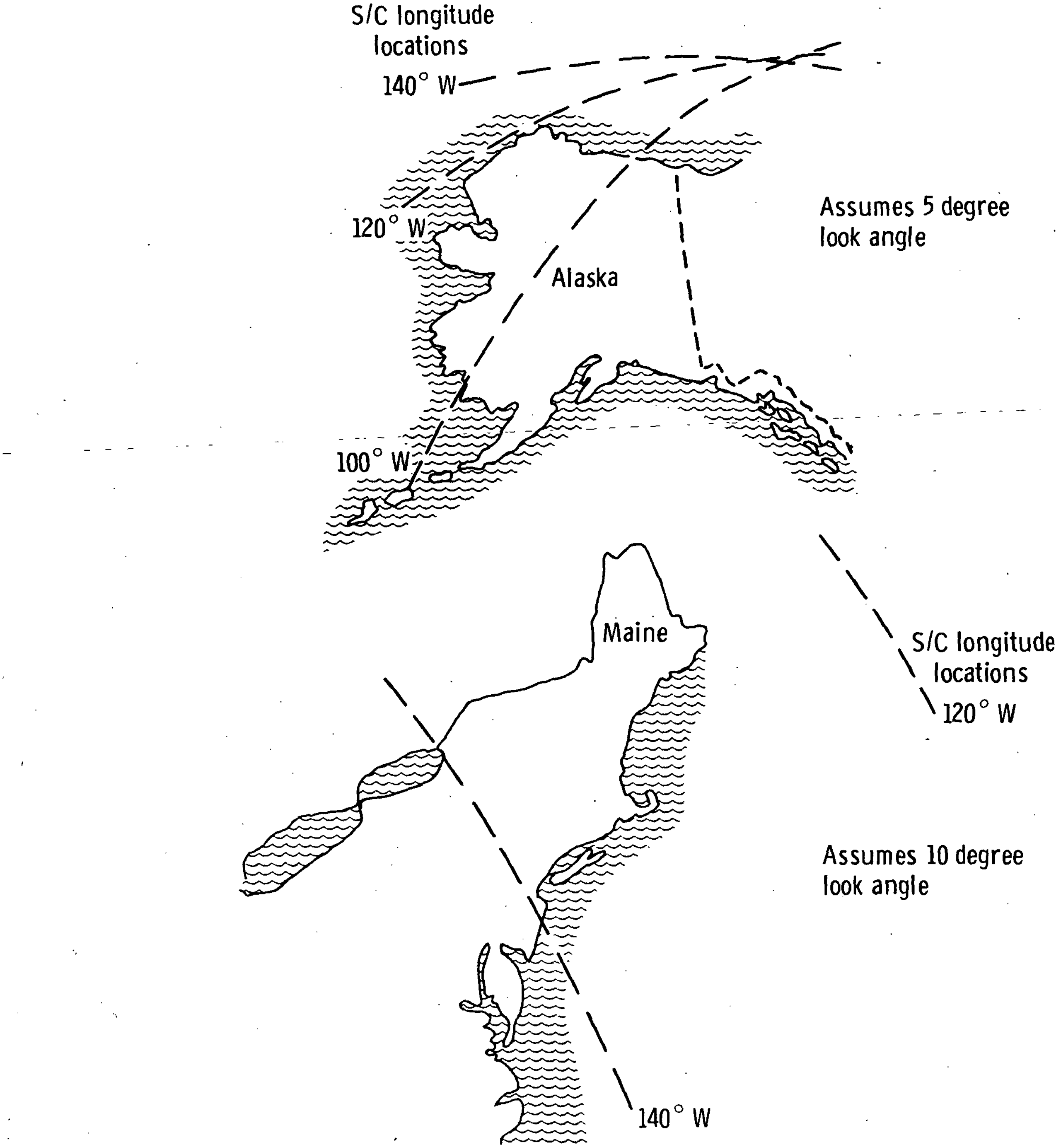


Figure A-3. - Locus of home receiver look angles for Alaska and Northeast U. S. Mainland.

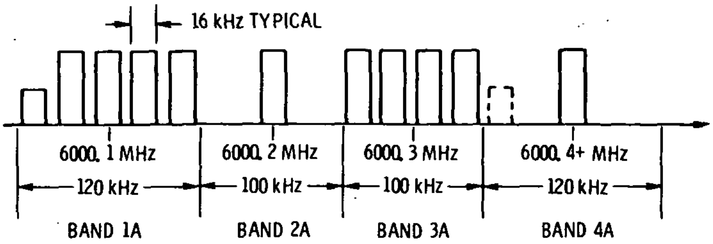


Figure A-4. - Frequency spectrum of signals normally received by Spacecraft A.

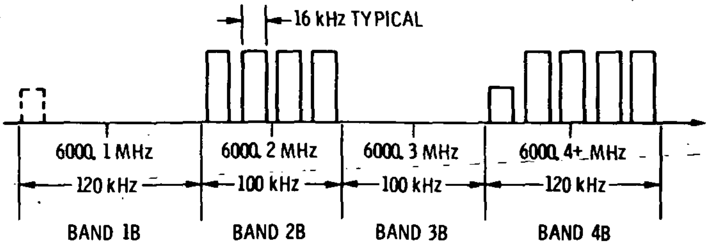


Figure A-5. - Frequency spectrum of signals normally received by Spacecraft B.

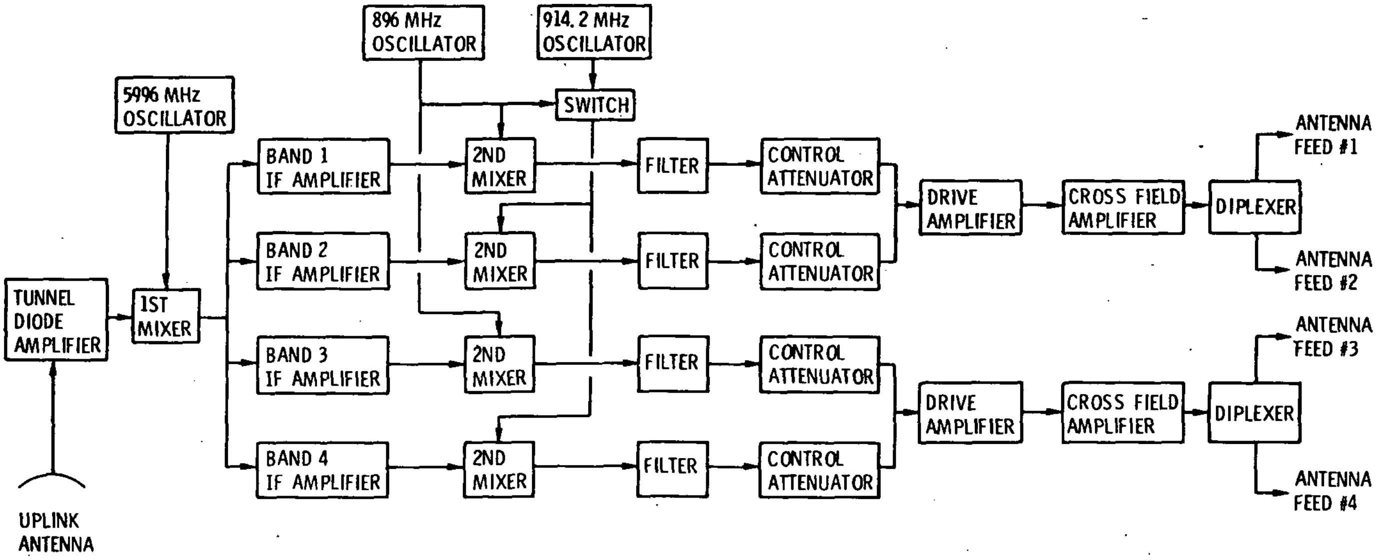


Figure A-6. - Spacecraft communications system to provide coverage of U. S. mainland, Alaska, and Hawaii.

Alternative Orbit Attainment Methods

The most promising orbit attainment techniques are described in the main body of this report. They included the least expensive method using a Thor/Delta/Electric stage or the more expensive method using a Titan 3C. Also considered were the Atlas/Agena/Electric stage which is no longer an operational system and the Atlas/Centaur/Electric stage which is relatively expensive and still requires approximately 75 days to achieve synchronous orbit station. Consideration was also given to launching two satellites simultaneously with a Titan 3C and also using an Atlas/Centaur/Burner II. These systems are discussed in detail below.

B-1 Atlas/Agena/Electric Stage

The booster for this mission is the standard Atlas/Agena launch vehicle system. It consists of the Atlas with an Agena upper stage. The mission sequence is similar to the Thor/Delta sequence of paragraph 2.2. Starting altitude for the electric thruster is 4150 nautical miles, array degradation has been reduced (estimated at 30% based on computer simulations), and consequently synchronous equatorial conditions would be achieved in about 103 days (allowing 3 days for shadowing).

B-2 Atlas/Centaur/Electric Propulsion

The booster for this mission is the standard Atlas/Centaur launch vehicle, consisting of the Atlas with a Centaur upper stage. Coast time for the Centaur is limited by the thermal constraints placed on its hydrogen fuel storage system. With the thermal constraint, the first transfer burn cannot be delayed beyond the first descending node. The mission sequence would be as follows:

1. Launch eastward from Cape Kennedy. The launch period is chosen to minimize shadowing and is best near the solstices. Summer launch windows are near local solar midnight, winter windows near local solar noon.
2. The Centaur and payload are injected into a 200 nautical mile, 28.4 degree inclination, circular parking orbit.
3. Final Centaur burn occurs at the first descending node and places the vehicle in a large elliptical orbit at a reduced inclination. Apogee is 38,450 nautical miles; perigee 200 nautical miles; inclination 15.5 degrees. This orbit is selected to have a semi-major axis equal to the synchronous radius, and, therefore, a synchronous period.
4. The payload is separated from the Centaur, the array deployed, and the electric thrusters started. The orbit is then circularized by firing against the velocity vector on the perigee half of the ellipse, and with it on the apogee half, switching as the minor axis is crossed.
5. When the orbital eccentricity has been reduced to zero, the vehicle will not be at the desired longitude and must be "walked" into place by firing in line with the velocity vector. Firing eastward causes a westward drift and vice versa. Seven days have been estimated for this process.

The major mission parameters for the above two approaches are shown in table B-1. The table also repeats the values cited in Section 2.0 on the Thor/Delta/Electric stage and the Titan 3C for comparison. Figure B-1 shows the mission time as a function of initial array power-again compared to the two systems described in Section 2.0.

TABLE B-1

Major Mission Parameters for Disaster Warning Satellite Study

Booster	Thor/Delta	Atlas/Agena	Atlas/Centaur	Titan 3C
Shroud	Delta	Lunar Orbiter	Surveyor	Martin (UPLF)
Shroud Weight	543 lb.	358 lb.	2067 lb.	2310 lb.
P/L Adapter Weight	70 lb.	100 lb.	116 lb.	230 lb.
Payload Weight	1562 lb.	1550 lb.	1506 lb.	1442 lb.
Mercury Weight	440 lb.	270 lb.	300 lb.	N/A
Power-initial	9.07 kW	8.44 kW	7.35 kW	7.00 kW
Ascent Degradation	45%	30%	7%	0%
After Ascent	4.99 kW	5.91 kW	6.84 kW	7.00 kW
After 5 Years	4.99 kW	4.99 kW	4.99 kW	4.99 kW
No. of Elec. Eng.	3	3	3	N/A
Type	Mercury Bombardment Ion Thruster			N/A
Size	30 cm dia	30 cm dia	30 cm dia	N/A
Beam Current	2 amps	2 amps	2 amps	N/A
Net Accel. Volts	550v	750v	920v	N/A
Input Power	1.66 kW	1.97 kW	2.28 kW	N/A

(Continued Next Page)

TABLE B-1 (Cont'd)

Thrust	22×10^{-3} lb	25×10^{-3} lb	28×10^{-3} lb	N/A
Spect. Imp.	2000 sec.	2400 sec.	2700 sec.	N/A
Prop. Util.	85%	88%	89%	N/A
Power Eff.	67%	76%	81%	N/A
Initial Alt., km	2730 km	7690 km	71,200 km (apogee)	38,500 km
Initial Alt., nm	1475 nm	4150 nm	38,450 nm (apogee)	19,300 km
Propulsion Time	160 days	100 days	75 days	1 day
Estimated Vehicle Cost	$6 million	$8 million	$14 million	$21 million

B-3 Atlas/Centaur/Burner II (1440)

The Burner II is a solid propellant upper stage which is attitude controlled and therefore does not require spinning for stability. It has been previously flown but not on the Atlas/Centaur vehicle. It appears to be feasible to integrate it with an Atlas Centaur and obtain a vehicle capable of performing this mission in a direct ascent approach. The following sequence of events could be used.

1. Launch east from Cape Kennedy.
2. Centaur and Burner II are injected into a circular parking orbit at 100 n.m. altitude. Orbit plane inclination will be 28.4 deg.
3. At approximately 22 minutes after launch the second Centaur burn injects the Burner II and spacecraft into a synchronous transfer orbit and reduces the inclination about 10 degrees, leaving the transfer orbit inclined at 18 degrees.
4. Transfer is completed approximately 16.5 hours later at the second apogee of the transfer orbit by firing the Burner II to circularize the orbit and remove the remaining 18 degrees of inclination.
5. The spacecraft is now at 50 degrees west longitude and at an altitude slightly higher than true synchronous causing it to drift westward to the desired longitude.
6. Vehicle capability using this sequence is 1480 lb. The spacecraft configuration for this mission is the same as for the Titan 3C.

which weighs 1442 pounds. No despin is required since the Burner II is attitude controlled and does not require spin stabilization.

B-4 Two Satellites on a Titan IIIC

In consideration of this approach the data from Table B-1 is helpful. The Titan capability to the desired orbit is 2700 pounds and the weight of a spacecraft configured for a Titan 3C launch is 1442 pounds per table B-1. Two satellites then exceed the vehicle capability. Additionally, capability would be required to separate the satellites in synchronous orbit as it is necessary that they be at different stations.

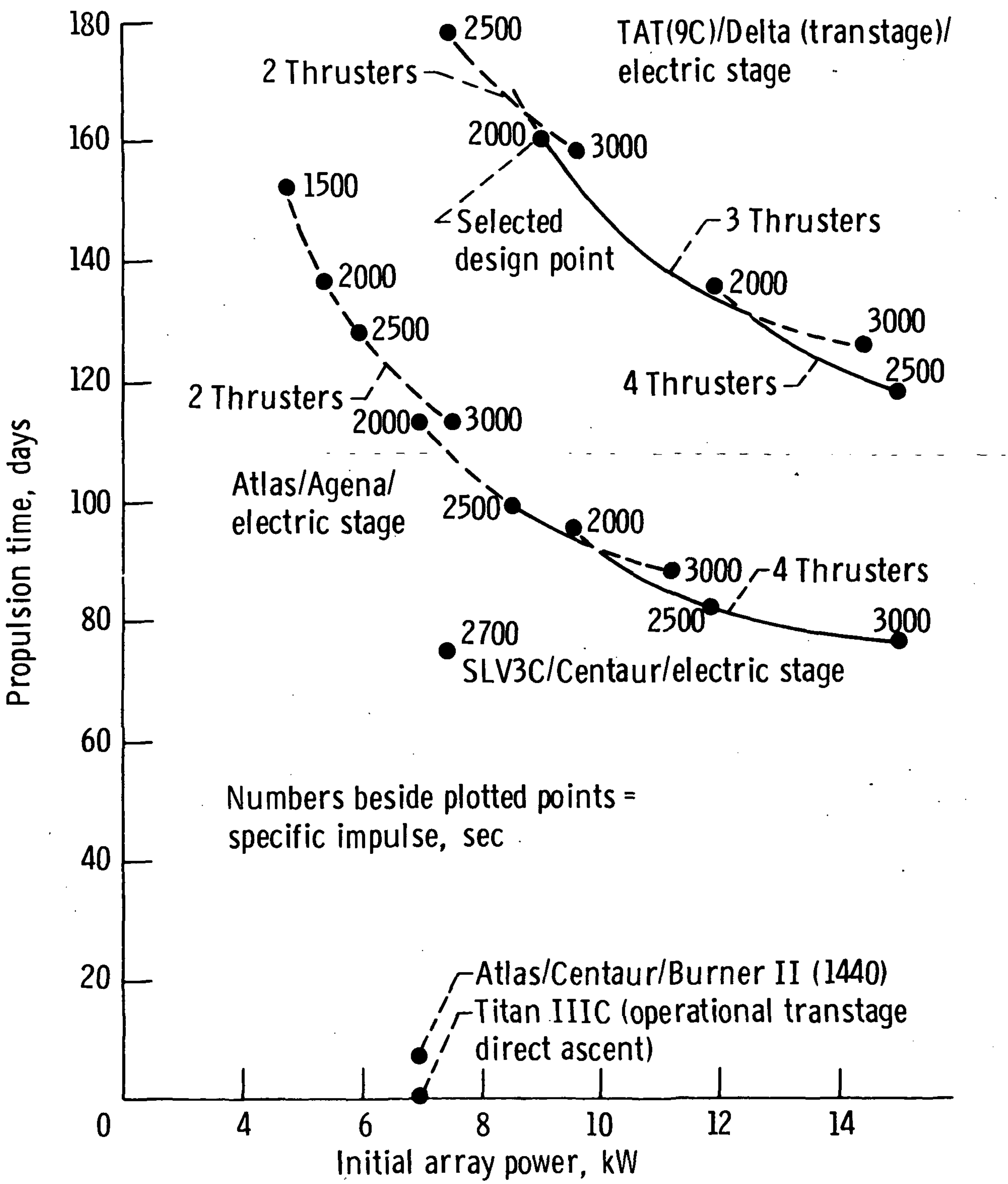


Figure B-1. - Propulsion time to achieve synchronous altitude as a function of booster system and initial (undegraded) solar array size.

APPENDIX C

Alternative Power System Considerations

This appendix is included to present information on three power system proposals about which specific questions have been raised. They involve using battery power to maintain broadcast capability during the shadow period, thus eliminating the need for a second satellite; also considered was the use of an Isotope system or a Solar Brayton Cycle system in place of the solar array as a prime power source.

C-1 Use of Batteries During Eclipse Periods

The technical plan of this report proposed two satellites to provide redundancy and to prevent loss of coverage during eclipse. One approach to solving the eclipse problem is to carry sufficient batteries aboard a spacecraft to provide power to the communication system during the eclipse. This approach does not appear to be a very attractive alternative to the two satellite approach. On a weight basis alone some 600 pounds of batteries and power conditioning would be required to support a priority level 1 message capability. With one spacecraft the number of available channels is reduced, the redundancy available with two spacecraft is lost and the complexity of the spacecraft is increased. For the above reasons, a detailed study was not carried out for a spacecraft employing batteries for eclipse operation.

C-2 Use of an Isotope Power Source

Isotopes were briefly investigated as a power source but quickly dropped from further consideration for the following reasons. A

PU^{238} system to produce 5 kW of power would weigh in the neighborhood of 3500 pounds and the PU^{238} alone would cost approximately 13 million dollars. Sr^{90} is considerably cheaper than PU^{238} but is much more hazardous and presently has no technology base to support development.

C-3 Use of Solar Brayton Cycle Power Source

A solar Brayton system is the only system that might someday compete with a solar array system at the 5 kW power level. It is extremely doubtful, however, as a 5 kW system would weigh approximately 4000 pounds.