PROJECT OBSERVER

A Student Design Project
Department of Aerospace Engineering
University of Michigan
Fall Term 1968

FOREWORD

Project OBSERVER is the eighth in a series of preliminary design feasibility studies conducted in Aerospace Engineering 483. Previous studies have included a Mars probe, solar probe, earth orbiters and an earth synchronous communications satellite.

In this senior elective course, over a period of 15 weeks, the students are given a problem statement with specific boundaries. In the current problem an earth resources satellite, Scout launched, was specified for exploratory research and development. The interplay between the sensors, payload size, desired orbit, communications, power, etc., is then investigated and the design configuration determined.

The students set up their project team organization and meet two afternoons a week as a team and during weekends as sub-groups to coordinate the critical interface between all groups. The emphasis is on team effort and team results with a high degree of mutual interdependence.

Guest lectures from the faculty, university research specialists, industry and government are given together with sub-group conferences. The student is given the opportunity to apply his previous theoretical and analytical course work to the highly redundant field of preliminary design. He is also in active contact with professional experts in the field. Despite the short time available, it has been found possible to select current problems and make a significant contribution--as this report demonstrates.

Wilbur C. Nelson
Professor

PROJECT: OBSERVER

OBSERVER is a state of the art satellite system designed to pioneer the field of remote sensing of the earth's surface for natural resource management and further development of remote sensing technology. The sensing system is an optical mechanical scanner which provides sensing in 8 discrete wavebands from 0.4 to 1.0 microns with a resolution of 1000 feet. Scanning is accomplished by a four-sided continuously spinning mirror which acts as a mechanical multiplexer allowing four channels to be transmitted back at one time. Data is transmitted direct link on S-band using a F.M. system with a bandwidth of 3.001 MHz.

The scanning system requires a circular non-repeating orbit so that after 24 hours the satellite will pass 1.5 degrees west of the point it was over the day before. Since a constant sun angle is desired in all pictures, a sun-synchronous orbit is also required. A lifetime of one year is desired so that all four seasons may be observed. The orbit of 313.4 nautical miles and an inclination angle of 97.6 degrees satisfies these requirements. The Scout launch vehicle will inject OBSERVER into this orbit.

OBSERVER is provided with a small rocket motor for orbit corrections so that the orbit through injection errors does not fall into a repeating orbit. A repeating orbit would cover the same part of the earth every twenty-four hours.

The 196 pound OBSERVER is an octagonal prism 32 inches long, 12.9 inches from center to side. Two solar cell paddles perpendicular to the satellite provide power for the satellite along with batteries. The necessary 3-axis stabilization is accomplished with a three boom inverted "Y" configuration, eddy damped gravity gradient system. Thermal control is accomplished through suitable emmisive coatings and thermal insulation.

OBSERVER is designed to be launched early in 1971.

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SECTION I
SYSTEM SUMMARY

1.0 OBJECTIVES AND PHILOSOPHY

The need for a comprehensive earth resources survey is great now and will certainly increase as the population increases and our use of present resources increases at an even greater rate. Inputs are needed so that a complete survey of present and future resources can be made available. Today's space technology can be ideally fitted to supplying these inputs through a remote sensing orbiting platform. A satellite of this description is technically feasible today and would fill an urgent need.

Project OBSERVER is designed to provide inexpensive, accurate, useful data to further remote sensing technology and provide preliminary maps of various sections of the earth suitable for resource management. The satellite is a state of the art system that could be built today.

Project OBSERVER will provide a broad range of meaningful data so that all prospective users may benefit from the program. Some of its major goals are to provide surface configuration, distribution of water, distribution and quality of vegetation, and land use. The satellite will have a minimum life time of one year so that all four seasons may be observed. It will observe the land at local noon which facilitates the sensing of vegetation and reduces the chance of cloud cover slightly. A look angle of 20 degrees is used to provide very little geometric distortion in all the data. A constant sun angle, of great use in interpreting the data, is provided by a sun-synchronous orbit.

Data from the satellite will be made available quite quickly so that seasonal variations will be useful. All major variables do not change on the pictures so that a minimum of training and data handling techniques will be needed to interpret the pictures. A complete land map of the United States in several discrete wavelengths should be available approximately every three weeks or less.

The benefits from an earth satellite that will permit total mapping of the United States in a short time and provide a basis for effective management of the nations resources will far outdistance the project's initial cost. Project OBSERVER is an inexpensive program that will initiate the field of remote sensing from orbit.

2.0 APPLICATIONS

The vast amount of raw data that will be transmitted to the ground by OBSERVER will have to go through a well managed, efficient application program to be useful. This program should store the data, correlate it properly, and make it available to users in needed formats. The final output may be in the form of single waveband pictures or in the form of false color pictures using several wavebands. The very usefulness of the earth resources survey relies upon effective data management. The data will be made available to all countries through NASA. Any country may also receive direct data from the satellite to process and interpret provided it has adequate receiving facilities. Since the system is designed to use common receiving stations for readout it would be relatively inexpensive for a nation to build a receiving capability. Hence the OBSERVER system can be used by all countires that are within its transmitting range.

Once the data is stored in an easily attainable useful form, there are many profitable applications of Project OBSERVER. One of the first is to provide an accurate map of the observed areas within three weeks. This map will have a constant sun angle and little geometric distortion which greatly aids in its interpretation. Aerial maps have the problem of geometric distortion and varying sun angles from picture to picture since cameras are usually recording from both sides of the plane.

Every material; mineral, plant or animal, has a spectral signature. This signature characterizes the material from any other material, similar to a fingerprint distinguishing one person from another. The spectral signature is composed of several wavelengths of light which the material either emits or reflects strongly. By taking pictures in discrete wave bands, certain objects can be made to stand out from others. Healthy vegetation can be distinguished from diseased vegetation, dry soil from wet soil, wheat fields from oat fields, and many others. For instance, if a photoimage was made of the earth in a waveband from $.55\mu$ to $.60\mu$, wheat would appear much brighter than oats, see Figure 2.1 . This concept of spectral signatures is the heart of remote sensing of earth resources. The technology of signature interpretation is still in the early stages, OBSERVER will provide needed maps for the further development of signature analysis as another one of its major objectives.

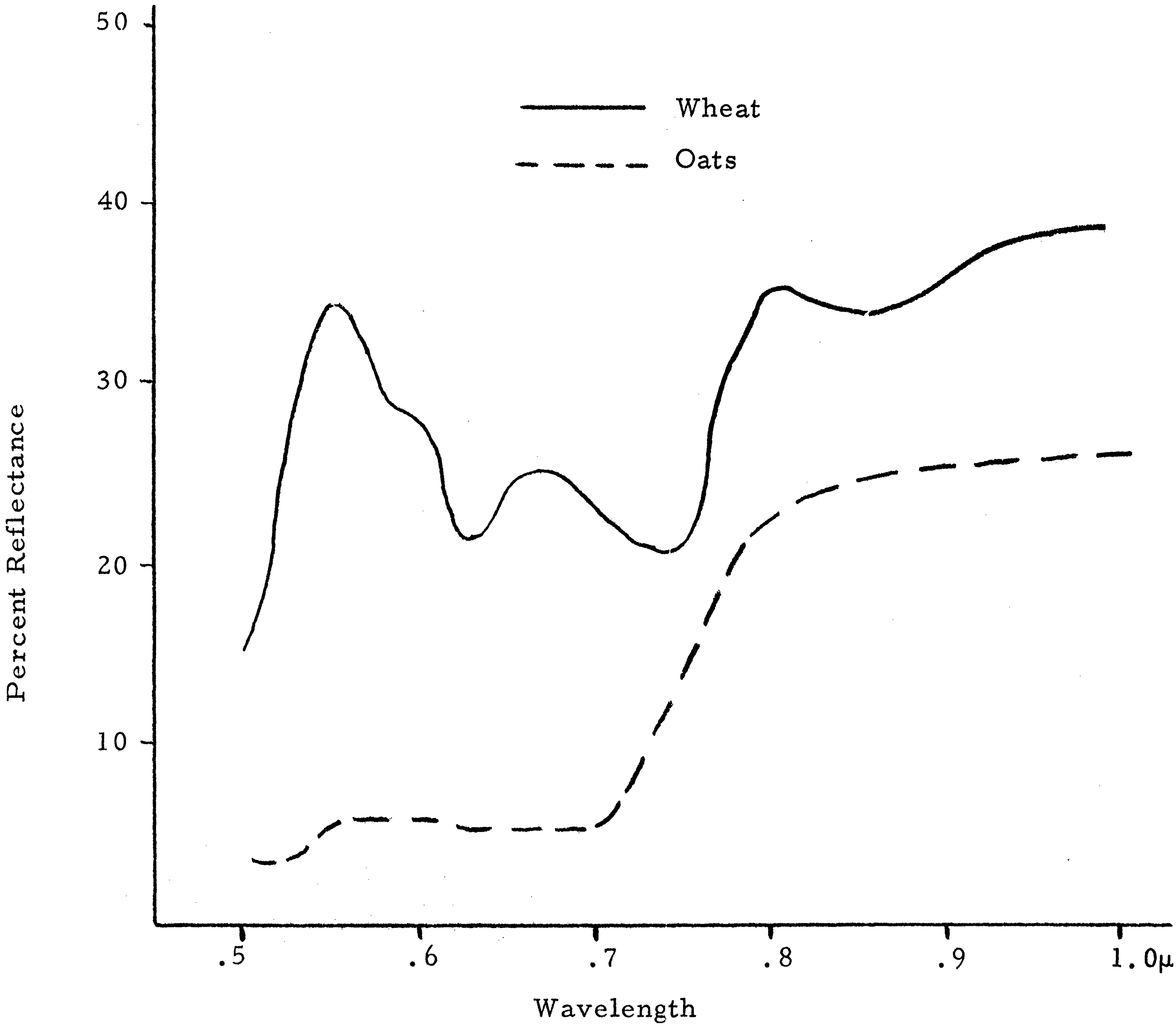


Figure 2.1 Wheat and Oats Reflectance

There are many other possible applications which OBSERVER can initiate, the following are a few of the more important. OBSERVER will help geologists to improve their knowledge of the nature and distribution of geometric features which may provide insight into energy exchanges associated with earthquakes, volcanic eruptions, and other crustal disturbances. Soil scientists may begin to catalog the important physical and chemical characteristics of soils, area by area, and relate these to associated geologic, climatic and vegetational factors with the broad data provided by OBSERVER. Hydrologists can begin to determine factors effecting the hydrologic cycle such as precipitation, evaporation, snow cover, movement of ice fields, and surface flow of water. This will be very usefult to the overall water managment in an area as it applies to dam control of rivers and sites for new dam construction. Foresters and agriculturists can begin to determine, along with signature analysis, the species and composition of vegetation in most areas in a very short time. This information can be applied to land management, estimate of crop vigor and eventual crop yield. Geographers will find OBSERVER useful in making fast, accurate regional and sub-regional analysis of land use patterns and in studying the interplay of climate, topography, plant life, animal life and human inhabitants in specific areas.

To be able to perform these tasks accurately, a remote sensing system must have available a system of ground-truth sites to aid in the calibration of the data returned. The 1000 foot ground resolution of the OBSERVER system requires a field 23 acres in size of the same crop in order to be able to identify crop types and vigor. At present, there is no network of test sites for satellite use. However, several of the NASA test sites now used for aircraft remote sensing studies could be adapted for satellite use. These test sites are used for agricultural, geological, and geographic studies. Work is in progress to develop test sites in Mexico and Brazil and is expected to be completed by mid-1969. These sites are also useful for agricultural, geological, and geographic research.

The ground test sites should provide data concerning:

1. radiation--incoming and outgoing
2. air and soil temperature
3. humidity and atmospheric pressure
4. soil moisture
5. wind velocity
6. photographic data from low flying aircraft

An aircraft system utilizing the same multispectral sensor system being used in the satellite should also be used to provide data for comparison, Figure 2.2 shows a map of the test sites now being used in the United States for aircraft studies.

Even though the earth resources program has great potential, practical value will only result when the user has a clear understanding of his information requirements. OBSERVER, being a first generation satellite, will help these users to further define their needs which will help develop more expensive, much more profitable and specific programs in the future. The earth resources program must be considered as a whole system and analyzed as such. Project OBSERVER has begun development of the data acquisition portion of the system, much work still needs to be done on the data analysis and distribution systems that will support OBSERVER and make it profitable to the user.

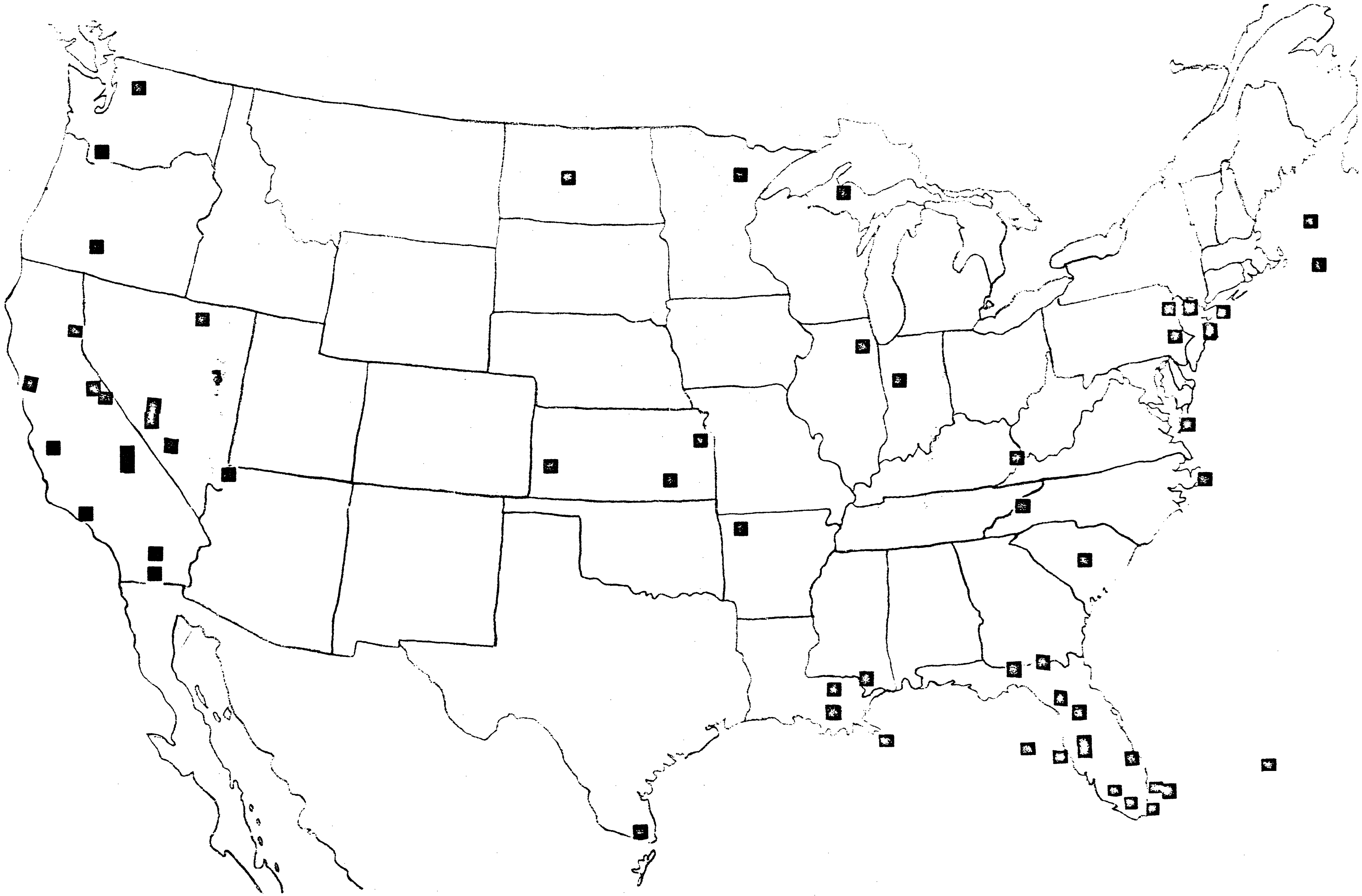


Figure 2.2 NASA Ground Test Sites

3.0 SATELLITE DESCRIPTION (Figures 3.1 and 3.2)

3.1 SENSOR DESCRIPTION

The primary component of an Earth Resources Satellite is its sensor system. This system must be capable of providing information that meets the requirements of the users and that can be easily and simply evaluated by the users.

The OBSERVER sensor system utilizes a multispectral, mechanical scanner to provide information in several spectral bands lying in the .4 - 1.0 micron region of the electromagnetic spectrum. The system covers a ground swath of 126 miles with a ground resolution of 1000 feet. Scanning is accomplished by utilizing a four-sided, motor driven mirror rotating at 22.65 rps. The mirror acts as a mechanical multiplexer that allows the system to transmit information from four detectors at once. A Cassegranian collector is used to focus the incident light into the spectrometer. The spectrometer uses a collimating lens, a fused quartz prism, and a re-imaging lens to disperse the light into its spectrum and to focus the spectrum on the fiber optics. The fiber optic bundles are placed in the focal plane of the re-imaging lens in the spectral channels of interest. OBSERVER will use spectral channels of interest to agriculture, geology, geography, and oceanography. The fiber optics relay the dispersed light to the detectors. Photomultiplier tubes of type S-10 and S-1 are used to convert the incident light energy to a voltage signal that can be amplified and transmitted to the ground.

3.2 COMMUNICATIONS

Data from the optical sensors is transmitted to the ground in real time and is processed and distributed to the proper areas for investigation. For transmission of the sensor data, OBSERVER uses a slot-turnstile antenna; the data will be direct frequency modulated to the ground on a carrier of 2.202 GHz. S-band transmission is being used because of the wide data bandwidth which we have to consider for this type of data. Since the data is direct link transmitted, a world wide receiving network is needed so that as much of the earth's surface can be mapped as possible. Currently there is not a global communications network using S-band. An extensive S-band network is presently under consideration since the Apollo program utilizes this frequency. Since STADAN is an established world wide network, it could fit our data receiving needs if the S-band network is not built. Adaptation of the STADAN network to S-band could be fairly easily accomplished. For housekeeping, tracking and command, OBSERVER will use the STADAN network.

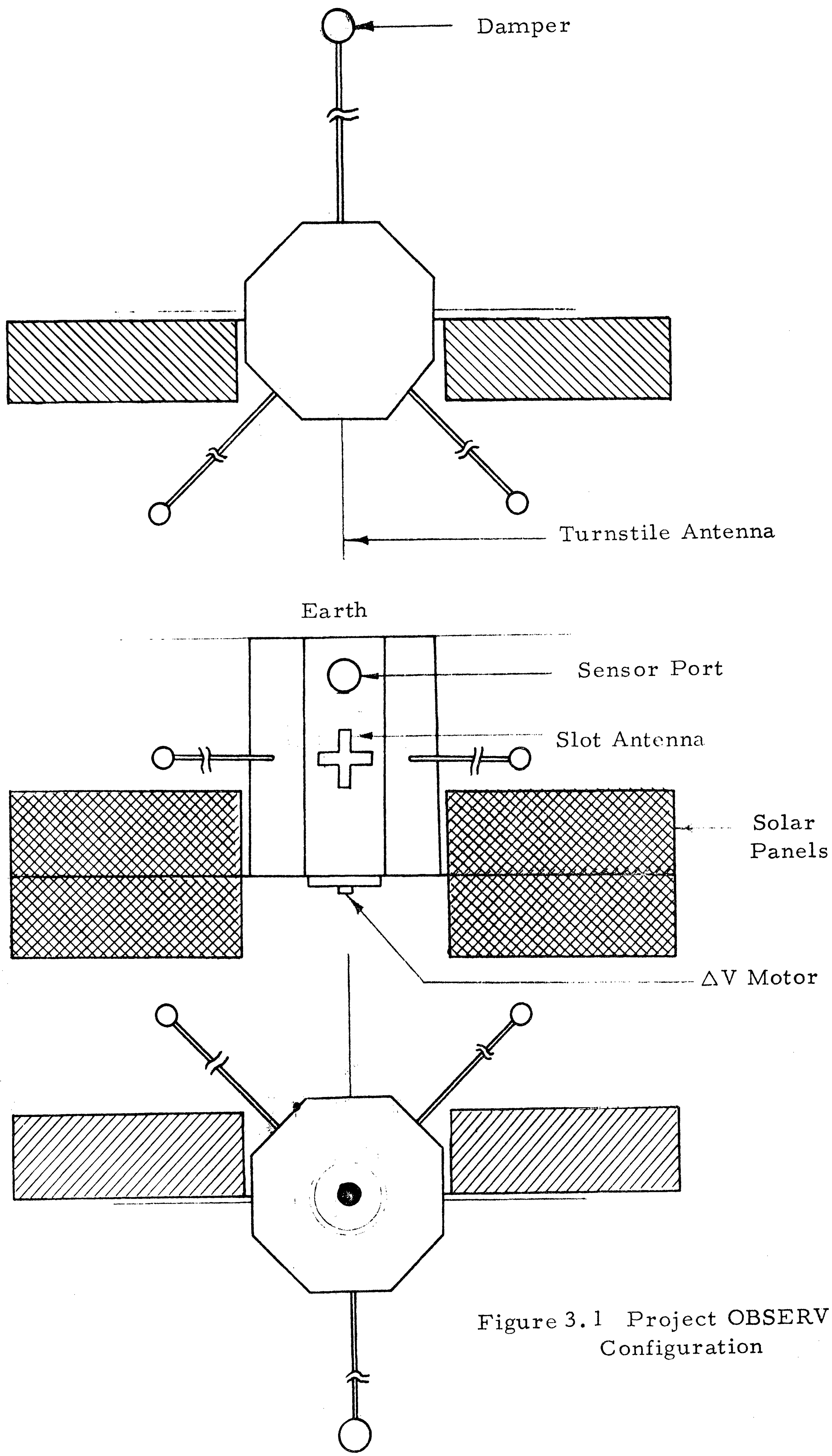


Figure 3.1 Project OBSERVER
Configuration

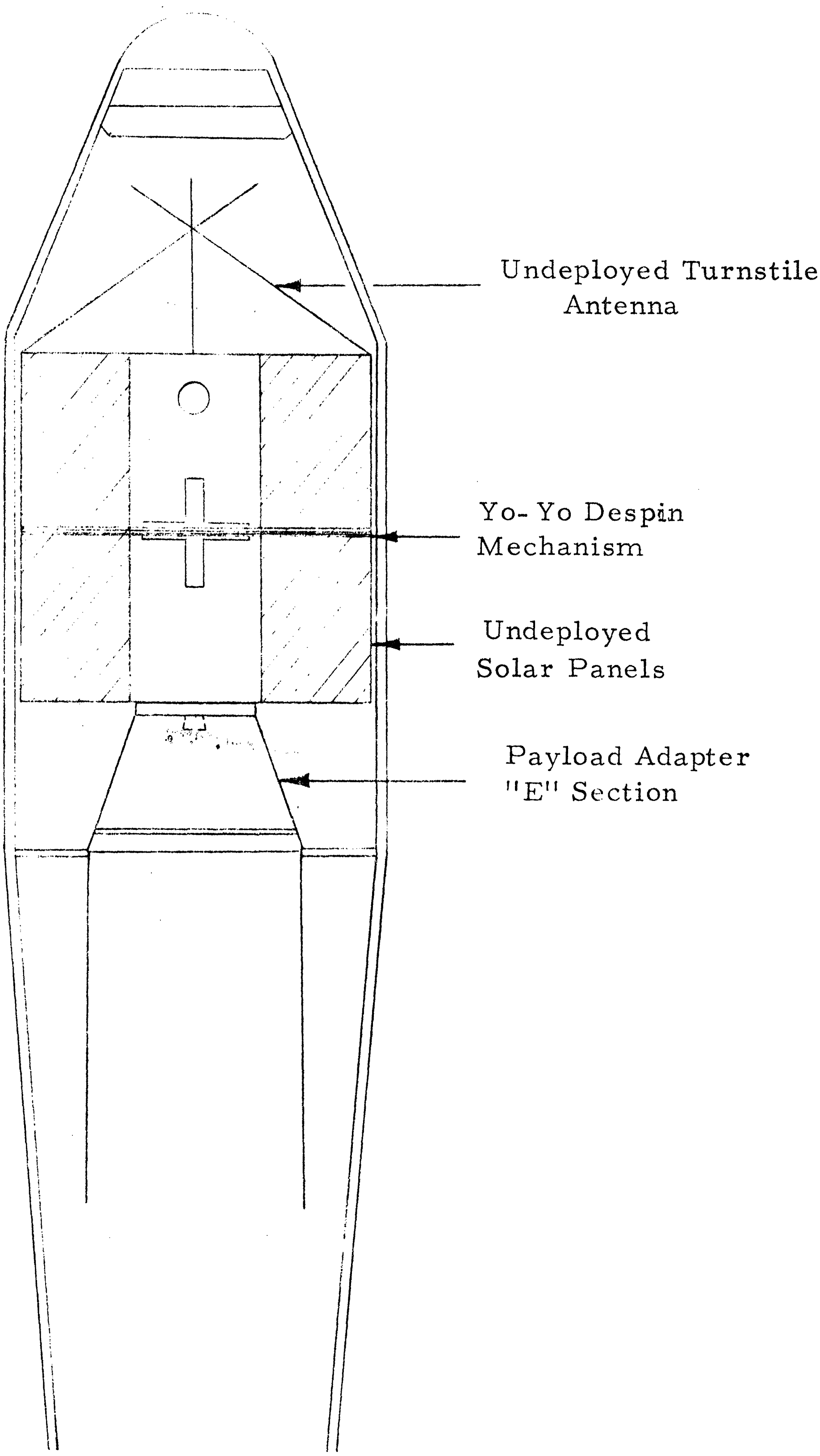


Figure 3.2 OBSERVER and Scout Vehicle Integration

To broadcast the condition of the satellite and the data, OBSERVER is equipped with a down-link housekeeping transmitter. This transmitter also doubles as the beacon tracking transmitter when not in use at its full capacity. The down-link transmits the status of the satellite to the ground at various times when it is considered necessary. This is accomplished through use of a pulse-code-modulated/frequency modulated transmitting system broadcasting on a VHF carrier of 230.9 MHz.

Since there are various functions which the ground desires the satellite to perform, OBSERVER is provided with an up-link on a VHF carrier of 250.7 MHz. Commands, sent up using a PCM/FM system can be accomplished at the time of transmission or they may be stored for later operation by use of the memory computer. All commands will be monitored by the down-link to insure the ground that only valid commands are received and interpreted. One whip-turnstile antenna will be used to receive and transmit this data.

3.3 ORBIT

OBSERVER will be in a near circular orbit with an altitude of 313.4 nautical miles and an orbital period of 96.4 minutes. The plane of the orbit will be coincident with the earth-sun line and nearly passing through both poles; this orbit is a sun-synchronous polar orbit. It is a non-repeating orbit, that is it will not exactly fly over the same point on the earth each day. Instead, it moves 1.5 degrees west on the ground every twenty-four hours from the point over which it passed the day before.

The satellite will be launched into the polar sun-synchronous orbit from Point Arguello, California. The booster to be used is the Scount launch vehicle. The inclination angle of the satellite's orbital plane is 97.6 degrees as measured from the plane of the earth's rotation around the sun to the satellite's ascending node.

3.4 STABILIZATION

The stabilization for OBSERVER is provided by a tri-boom gravity gradient system with damping which provides three-axis pointing accuracy of 2-3 degrees as required. The booms are in a "Y" configuration with each boom swept back 15 degrees and the lower two booms separated by 90 degrees.

The lower booms consist of Beryllium-Copper 2 mil x 2 inches x 60 feet tape which rolls up to form a cylinder with a 1/2 inch radius. The tip weights are spherical and weight about five pounds each.

The upper boom points away from the earth's center and its tape is four mils thick. It is capped by a passive Eddy current damper. The booms are silver plated on the outside to reflect thermal radiation which reduces solar bending.

A motor-guide mechanism is used with each boom for erection and retraction. The entire stabilization system is passive once extended.

3.5 STRUCTURE

The structure of the satellite was selected to accommodate the requirements of the sensor system while remaining light weight. The satellite is a regular octagonal prism, 32 inches long, 12.9 inches center to side. The structure weighs 20 pounds. The supporting structure consists of four aluminum channel frames fastened in a cruciform shape. Eight 1/4 inch thick aluminum side panels of sandwich honeycomb construction are fastened to the basic structure. The ends are made of 1/4 in thick aluminum honeycomb. The basic structure has been rated to carry loads of 30 g's longitudinally and 10 g's radially.

The primary power for this system will be provided by solar cells mounted on panels that extend perpendicular to the long axis of the satellite. They are oriented such that peak power can be provided from 80 degrees north to 80 degrees south of the equator. During launch, they will be folded along the side of the satellite. To insure reliability deployment will be by a passive system consisting of a spring-loaded mechanism.

During the initial orbits before deployment of panels, power will be provided by a Ni-Cd battery. This battery will also provide the needed power during eclipse.

The satellite is thermally controlled passively using surface coating and thermal installations. A surface coating with an effective absorptivity of .1 and emissivity of .12 is used to control the satellite skin temperature so that it varies from 52-56°F. The interior components with similar temperature restrictions are packaged together and are held at appropriate temperatures using fiberous thermal insulation. Temperature gradients within the satellite are minimized by heat conduction through the satellite structure.

SECTION II - SYSTEM DESCRIPTION

1
SENSOR SYSTEM

1.1 EARTH SENSING SYSTEM

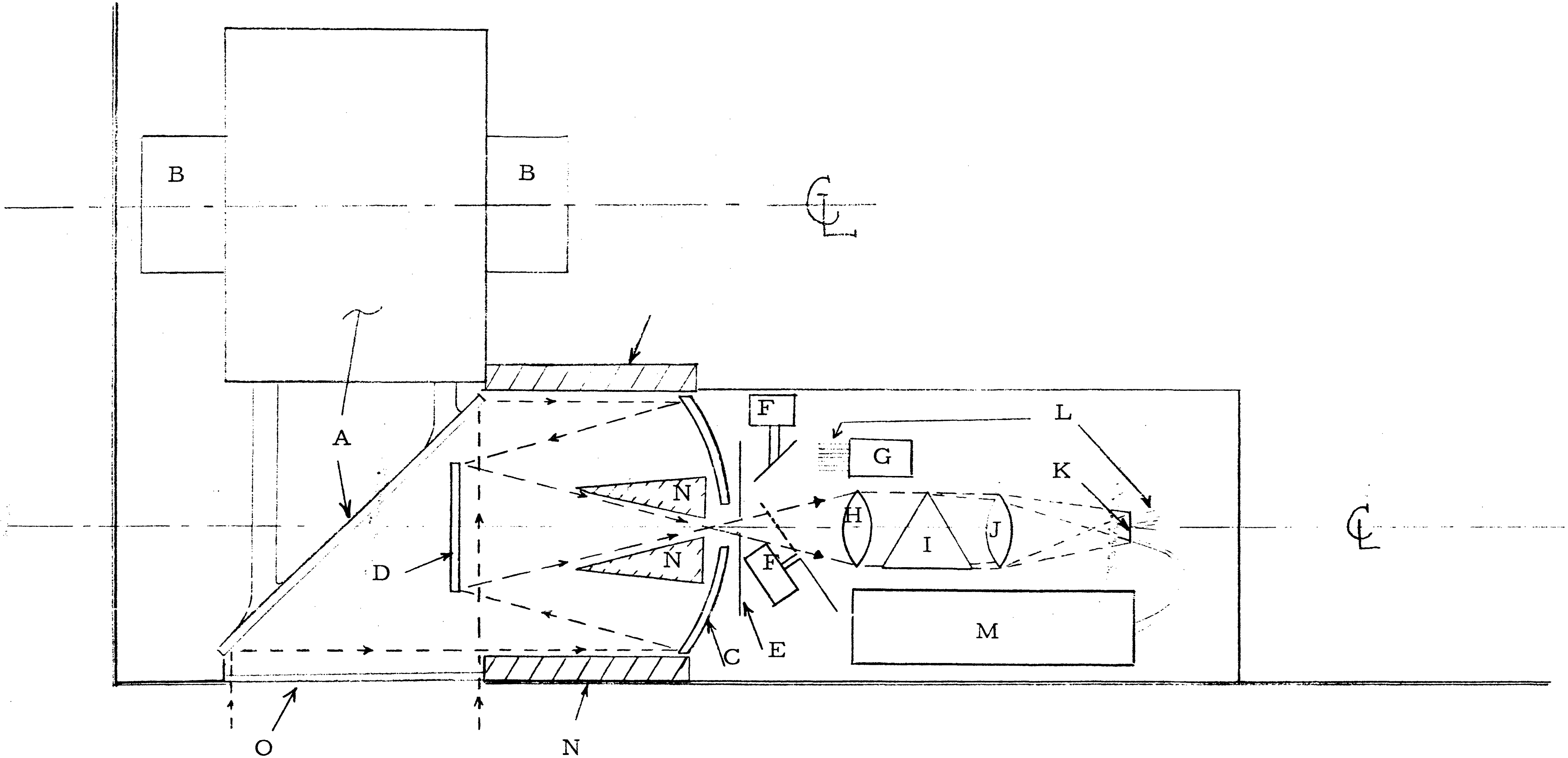
In the development of an earth resources satellite, the selection of a sensor best capable of generating data that satisfies the requirements of several users is of primary importance. The urgent need for such a system indicates that a small, inexpensive, state-of-the-art payload would be most useful in a first-generation satellite. Since data of the type needed from space is essentially non-existent, a system such as this could not only provide data for individual users, but would also provide information that would indicate what changes should be made in future remote sensing systems. A system such as that proposed by Project OBSERVER could prove more cost-effective in this area than a first generation system composed of highly sophisticated sensor systems or multiple satellite systems.

1.2 SENSOR SYSTEM

The OBSERVER satellite will employ an optical-mechanical, multispectral scanner as its remote sensing system. The system will measure reflected energy from the ground in eight channels lying in the visible and near infrared regions of the spectrum (0.4-1.0μ). In order to avoid excessively large communication bandwidth problems, the scanning mirror has four faces and acts as a mechanical multiplexer. This allows the system to monitor any four channels at one time without increasing the bandwidth.

The system records data during that part of the scan that is ± 10 degrees from the vertical. This gives a ground swath of 126 statute miles with a minimum of distortion from the curvature of the earth. The collected light is focused through a spectrometer that contains a fused quartz prism to disperse the light into its spectrum. Fiber optic bundles placed in the desired regions of the focused spectrum transmit the light to the detectors. The detectors are photomultiplier tubes which are sensitive to the desired spectral intervals. A schematic of the sensor system is shown in Figure I.1.

Although a ground resolution of 100-300 feet is desirable, the communication bandwidth required for such a resolution is larger than that which is currently available. Due to this restriction, OBSERVER will provide data with 1000 feet resolution. This resolution can be tolerated because, as a first generation earth resources satellite, OBSERVER'S primary goal is to gather large amounts of data over a period of one year. A finer resolution can be employed by future systems which will be capable of on board data processing to reduce the bandwidth problems. In the event that a wider communication bandwidth is alloted for earth resources in the future, OBSERVER can be converted to produce a ground resolution as much as an order of magnitude smaller than presently planned.



A. Scanning Mirror
B. Drive Motor
C. Parabolic Primary Mirror (Collector)
D. Plane Secondary Mirror
E. Slit
F. Chopper Motor and Mirrors
G. Calibration Source
H. Collimating Lens
I. Fused-Quartz Prism
J. Re-imaging Lens
K. Focal Plane
L. Fiber Optic Bundles
M. Photomultiplier Tubes
N. Stray-Light Baffles
O. Window

Figure 1.1 Scanner-Spectrometer Assembly

1.3 SENSOR ASSEMBLY

1.3.1 Scanning Mirror

The OBSERVER sensor system uses a four-sided scanning mirror in order to allow more than one channel to be monitored at a time without increasing the bandwidth. If each mirror face is used to scan for one specific detector, then the output of four detectors may be read for one mirror revolution. This method acts as a mechanical multiplexer to place four signals in the smallest bandwidth. The mirror is oriented at 45 degrees with respect to the center line of the sensor package and spins at 22.65 rps.

The scan of the mirror on the ground produces an overlapping pattern of successive mirror faces. Each face produces an overlap of the previous face by three-quarters. Figure 1.2 shows how the successive faces overlap while still maintaining contiguous ground coverage with all four wavelength intervals.

A suggested design for the scanning mirror is shown in Figure 1.3. The mirror faces are rectangular, polished beryllium, approximately 10" x 9" x 1/4". The mirror faces are connected by a skeleton structure of beryllium. The objective of the skeleton structural framework is to keep the weight to a minimum and to prevent distortion of the mirror faces during the scanning operation. The total weight of the mirror subsystem with a skeleton framework is 10-12 pounds.

1.3.2 Collecting Optics

The scanning mirror directs the beam to a Cassegrainian collector with f/2 and a diameter of 6.56 inches. The Cassegrainian primary reflects back to a plane secondary mirror located near the scanning mirror. The secondary mirror is half the diameter of the primary mirror, 3.28 inches. This secondary mirror reflects the beam through the center hole of the Cassegrainian to the slit. The slit which defines the field of view is 0.175 mm square and acts as the entrance into the spectrometer. Baffles are assembled around the collector mirrors in order to prevent the possibility of inaccurate detector signals caused by stray light.

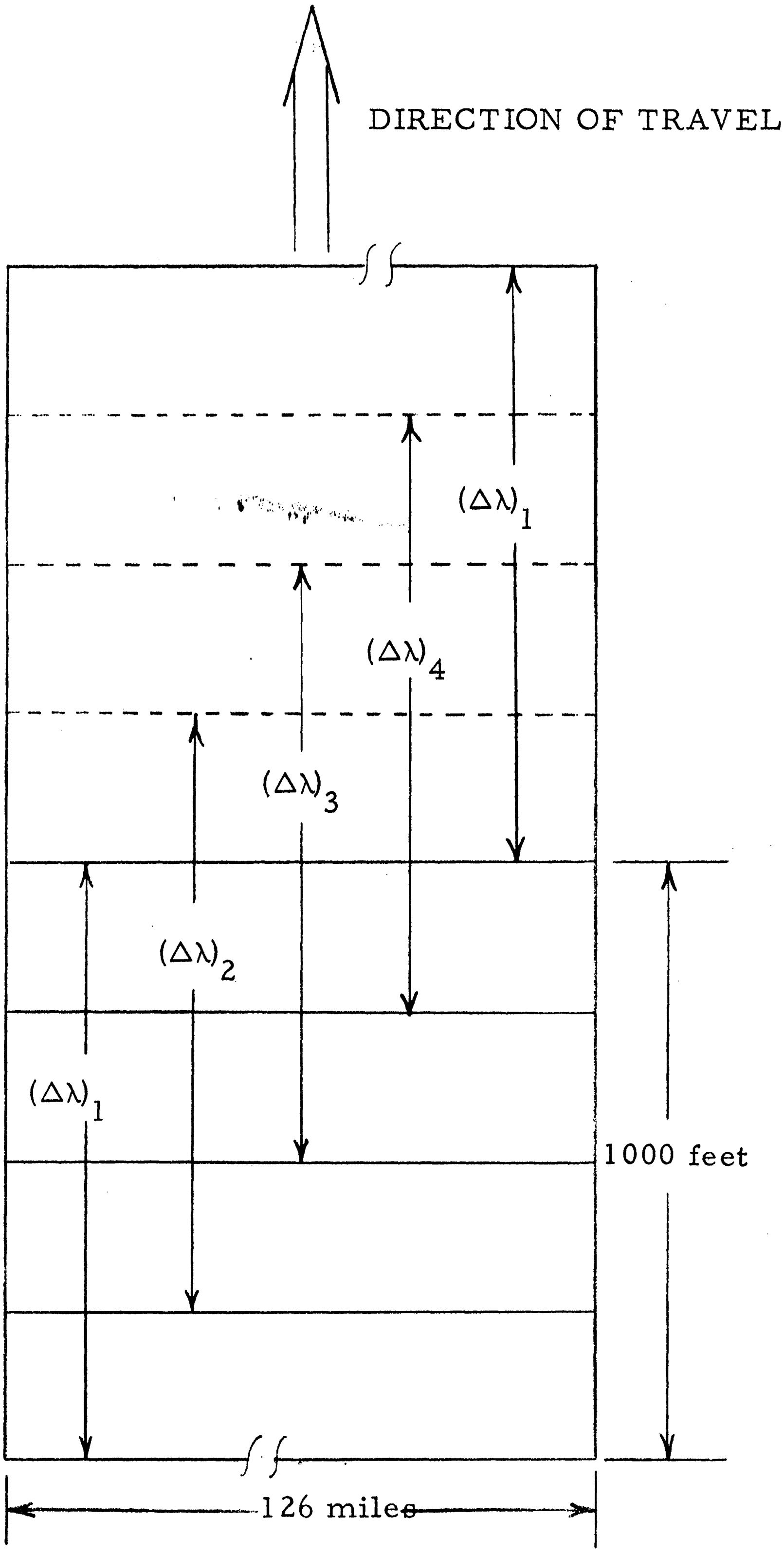


Figure 1.2 Overlap of Ground Coverage (Schematic)

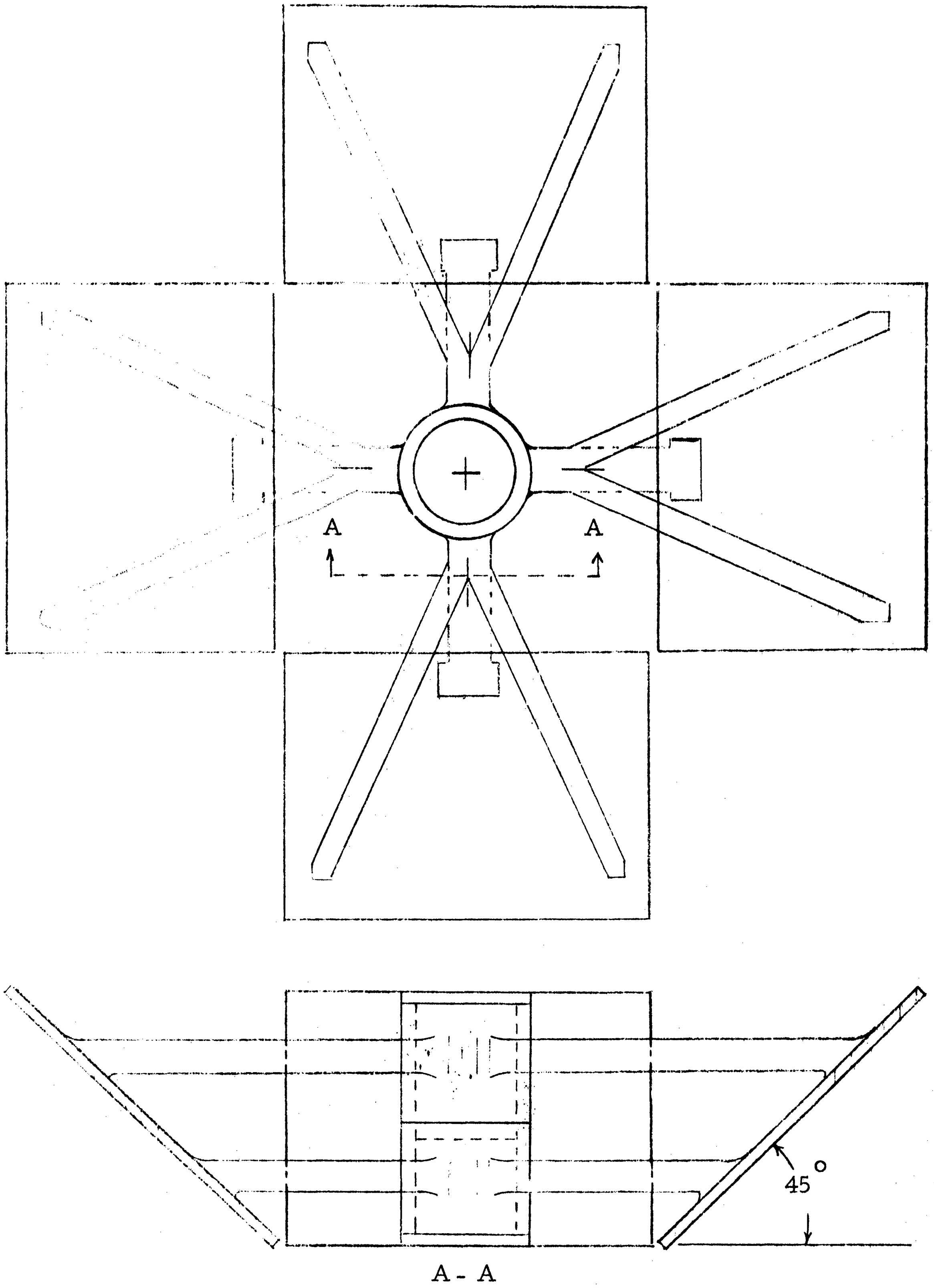


Figure 1.3 Scanning Mirror Assembly

1.3.3 Dispersing Optics

The light beam that enters the spectrometer is directed into a fused-quartz prism by a collimating lens. The collimating lens has a diamter of 1.89 inches. It forms parallel beams of light so that the light entering the prism is uniform. The prism has an apex angle of 60 degrees and a base of 1.89 inches (see Appendix A for prism calculations). The prism disperses the beam into its spectrum. A re-imaging lens brings the dispersed spectrum to a focus. The re-imaging lens is the same size as the collimating lens. At the focal plane, the spectrum is divided into the specific spectral intervals by fiber optic bundles. The bundles then carry the collected energy of their respective wavelength bands to the appropriate detectors. Figure 1.4 shows a schematic representation of the fiber optics-detector combination.

1.3.4 Detectors

The detectors used in the sensor are photomultiplier tubes that respond to the energy carried to them from the focal plane by the fiber optic bundles. The five detectors in the visible region are S-10 type photomultipliers. The near infrared wavelength bands use S-1 type photomultipliers. All tubes are cooled to -40 degrees centigrade by radiating to space. This cooling minimizes the effect of the dark current on the signal power. The response characteristic of the tubes versus wavelength is shown in Figure 1.5.

1.3.5 Spectral Channels

With a ground swath of 126 statute miles and an orbital altitude of 313 nautical miles, it will take OBSERVER approximately two weeks to map the earth once. If three mappings are made to reduce cloud obscuration of the ground, it will take one and one-half months for a single map of the earth in four spectral bands. Even though one and one-half months is needed to generate one complete set of data, much of the data will be repeated every two weeks which is useful in studying agricultural cycles. If each four-channel map is to be repeated every season, Project OBSERVER can provide two combinations of four-channel maps. By making use of the data on spectral signature that has been gathered by airborne scanners, Project OBSERVER has chosen two combinations in such a way that ground element identification can be optimized with respect to certain tasks, particularly those concerning agriculture. Listed below is a chart of the eight spectral intervals and the two mapping combinations suggested for use in a first-generation satellite.

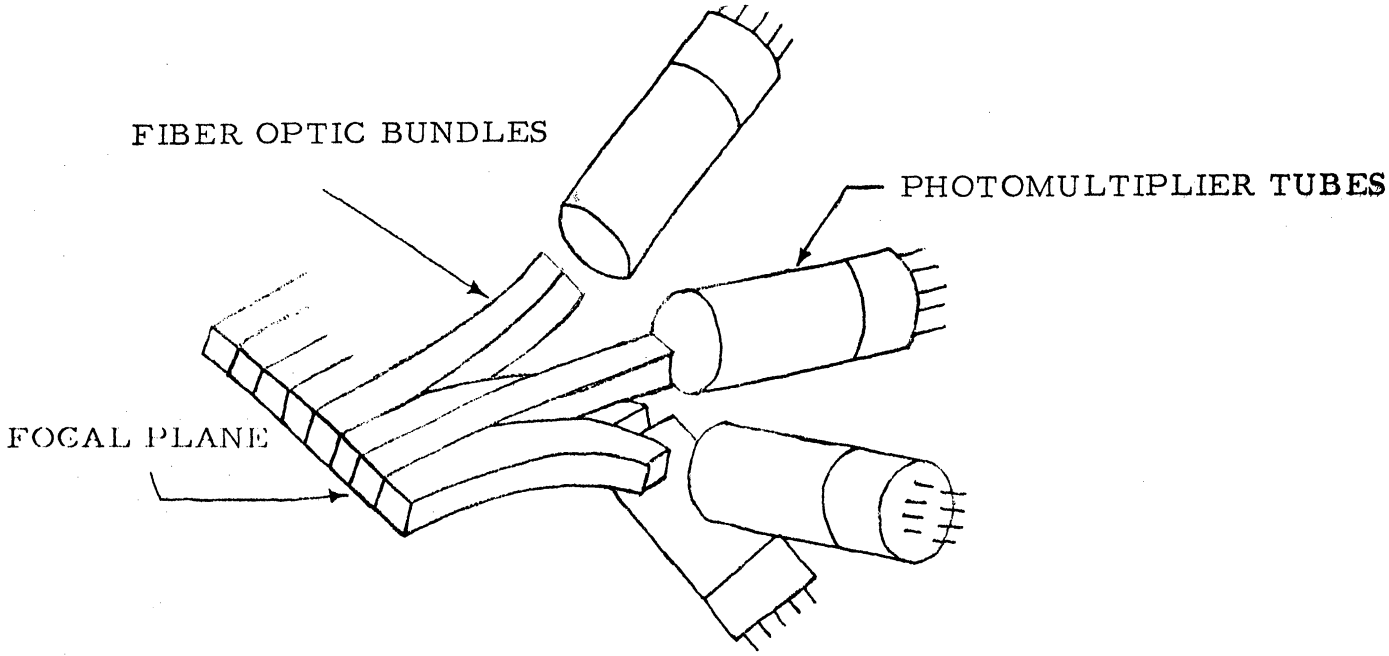


Figure 1.4 Fiber Optics--Detectors Schematic

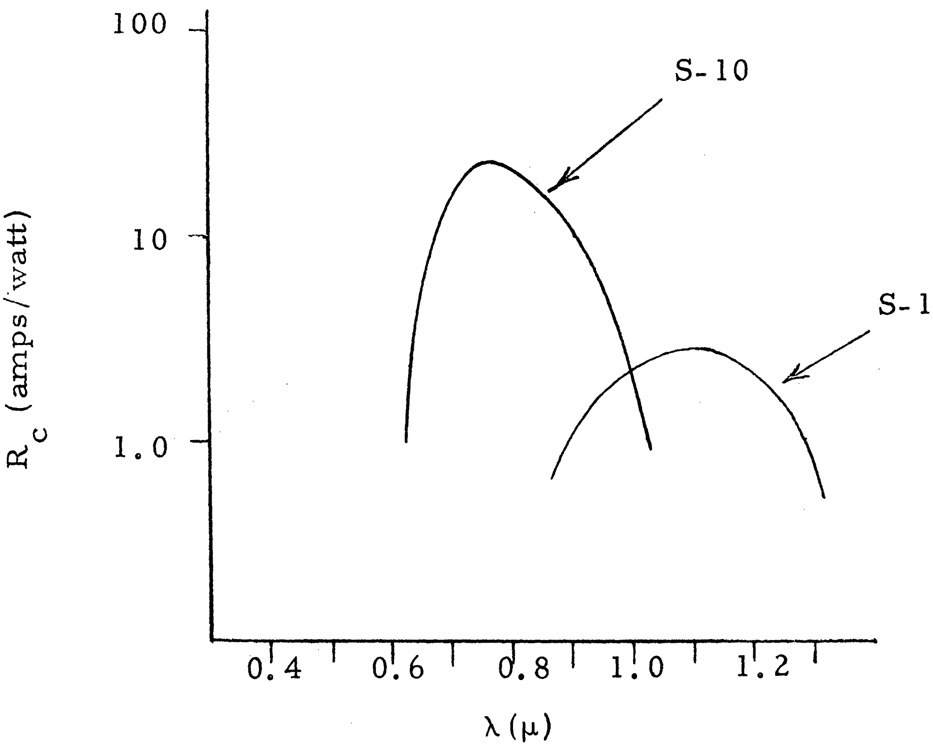


Figure 1.5 Cathode Sensitivity vs λ

Channel Number		Wavelength Band (microns)	
1	S-10 phototube	.40-.50	visible
2		.50-.58	
3		.58-.62	
4		.62-.66	
5		.66-.72	
6	S-1 phototube	.72-.80	
7		.80-.88	near infrared
8		.88-1.0	

Combination	Channels
1	2, 6, 7, 8
2	1, 3, 4, 5

1.3.6 Calibration

Calibration of the signals is accomplished by using a light emitting diode as a calibration source. Fiber optic bundles receive the light from the diode and direct it to a chopping mirror ahead of the collimating lens. When the chopping mirror obstructs the beam from the scanner assembly, the calibration source is observed by the detectors. The current to the source is monitored so that any variation in the calibration signal strength can be measured. Because each detector is returning data to the ground for 20 degrees of the mirror scan, the next 70 degrees of scan is used for calibration.

1.3.7 Operating Mode

Because the sensor measures reflected sunlight, it can operate only on the sunlit side of the earth. The earliest and latest points in its orbit at which operation is possible is when the sun vertical angle is 80 degrees. This point is referred to as the "operational horizon". Due to the large sun-vertical angle at this point, the reflected energy will be a minimum and the signal-to-noise ratio will be at its lowest value. At this point the detector bias current is turned off or on. It is off on the dark side of the earth in order to conserve battery power. The motors, however, are not turned off in order to avoid the power and time problems associated with mirror spin-up. This also helps in extending the lifetime of the motors.

1.3.8 Weight and Power Budget

Component	Weight (lbs.)	Power (watts)
Optical components	15	
Scan motor	4	6
Electronics	3	4 (detectors) 2 (amplifiers) 1 (calibration source)
Chopper motor	2	1
Structure	10	
Redundant motor, electronics	5	
	39 lbs	14 watts (operating) 7 watts (not operating)

1.4 SYSTEM PERFORMANCE

1.4.1 Scanning Geometry

The OBSERVER sensor system has a scanning geometry as defined in Figure 1.6. The rotational velocity of the scanning mirror required to keep the scan line continguous is given by:

$$\dot{\alpha} = \frac{2\pi v}{r} = \frac{2\pi\left(\frac{v}{h}\right)}{\beta} \quad \text{radian/sec}$$

where
v = ground track velocity (feet/second)
r = side dimension of ground resolution square (feet)
h = satellite altitude (feet)
β = instantaneous field of view (radians)

A point on the ground directly below the scanner will be observed for a dwell time:

$$t = \frac{\beta}{\dot{\alpha}} = \frac{\beta^2}{2\pi\left(\frac{v}{h}\right)} \quad \text{seconds}$$

The electronic noise bandwidth is then given by

$$\Delta f = \frac{1}{2t} \ \text{Hz.}$$

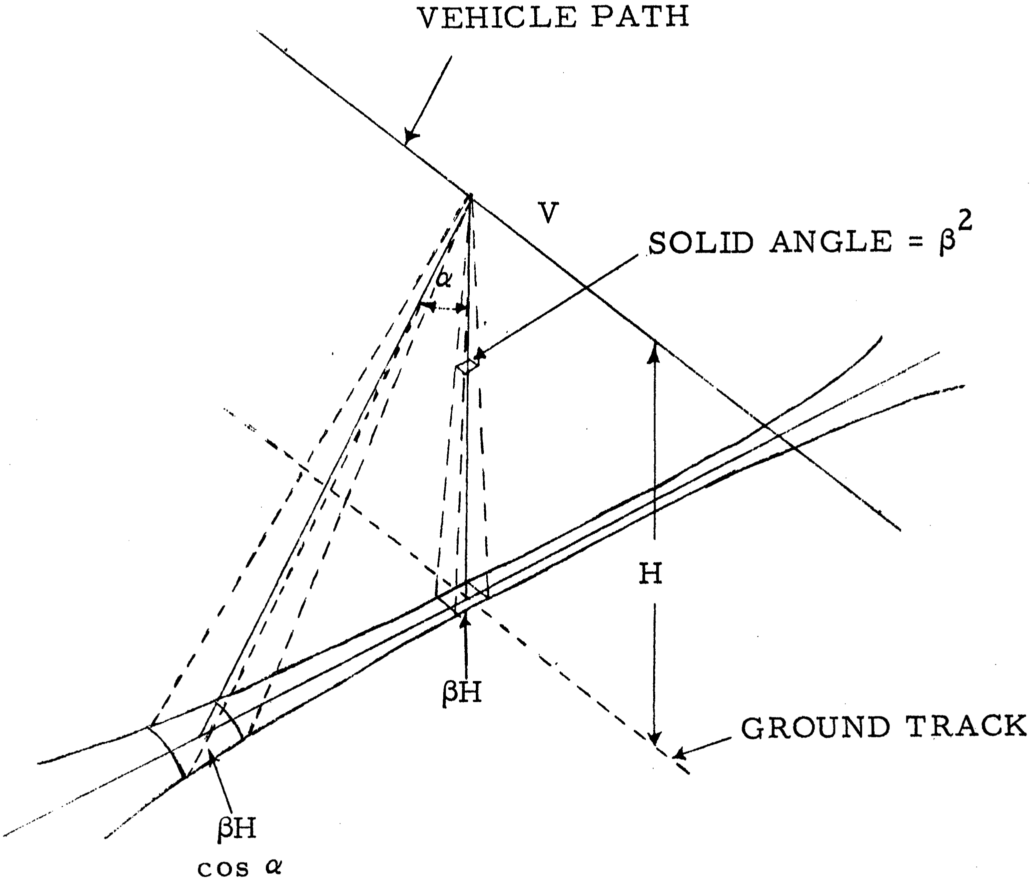


Figure 1.6 Sensor Scanning Geometry

For OBSERVER, with the parameters

$$r = 1,000 \text{ feet}$$
$$v = 22,650 \text{ fps}$$
$$\beta = .524 \times 10^{-3} \text{ rad}$$
$$h = 313.4 \text{ n.m.}$$

we find that

$$\dot{\alpha} = 22.65 \text{ rps}$$
$$\Delta f = 136 \text{ kHz}$$
$$t = 3.74\ \mu\text{sec.}$$

The bandwidth, Δf, of the system is a very important parameter for two reasons. First, it directly affects the noise signal of the detectors. Because the values of the signal-to-noise ratios involved in this system are not always very large, any significantly large value of bandwidth, and hence, noise, could reduce the fidelity of the data received. Secondly, the electronic noise bandwidth directly affects the size of the communication bandwidth, and should be kept as low as possible in order to keep the communication bandwidth within practical limits.

1.4.2 Signal-to-Noise Ratios

The performance of the sensor system is best demonstrated by the signal-to-noise ratios of the detectors. For detectors of the photomultiplier type, the signal-to-noise ratio may be represented as

$$S/N = \frac{i_s}{\left[2 e \Delta f \ i_s \left(1 + \frac{i_d}{i_s}\right)\right]^{1/2}}$$

where

e = electronic charge, 1.6×10^{-19} coulomb
Δf = electronic noise bandwidth
i_s = cathode signal current
i_d = dark current

Because the term i_d/i_s is negligible when the photomultipliers are cooled to -40 degrees centigrade, the expression for S/N becomes

$$S/N = \frac{i_s}{\left[2e \Delta f \ i_s\right]^{1/2}}$$

The cathode signal current is

$$i_s = R_c P_s$$

where

R_c = cathode sensitivity (amps/watt)
P_s = detector signal power

The signal power of the detectors is just the power incoming to the system degraded by the optical and electrical efficiencies of the system. If we call the optical/electrical efficiency σ,

$$i_s = \sigma R_c R_i$$

To determine the incoming power, consider the flux, F, of sunlight reflected from a square ground element of side length r.

$$F = \frac{H_\lambda \tau^2 \rho \cos^2 \psi \; r^2}{\pi} \qquad (W - \mu^{-1} - \text{steradian}^{-1})$$

where

H_λ = solar spectral irradiance above the atmosphere ($W\text{-}m^{-2}\text{-}\mu^{-1}$)
τ = transmission of the atmosphere
ρ = reflectivity of the ground scene
ψ = sun-vertical angle

The solid angle subtended by the collecting aperture is

$$\Omega = (4\pi)\left(\frac{\pi D^2}{4}\right)\left(\frac{1}{4\pi h^2}\right) \quad (3/4)$$

$$\Omega = \frac{3\pi D^2}{16\, h^2} \qquad \text{steradians}$$

where

D = collector diameter (6.5 in)
h = sensor altitude (313 n.m.)

The factor (3/4) takes into account the area blocked by the secondary plane mirror in the collector assembly.

The incident power is expressed by:

$$P_i = (F\Omega)(\delta\lambda)$$

where $\delta\lambda$ = spectral interval of interest

Therefore

$$P_i = \left(\frac{3}{16}\right)\left(\frac{D^2}{h^2}\right) \quad H_\lambda \tau^2 \rho \, r^2 \cos^2 \psi \, \delta\lambda$$

The expression for S/N becomes

$$S/N = \left[\frac{i_s}{2\,e\,\Delta\,t}\right]^{1/2}$$

$$S/N = \left[\frac{3\,D^2\,H_\lambda \tau^2\,r^2\,\rho\,\cos^2\psi\,\delta\lambda\,\sigma\,R_c}{32\,h^2\,e\,\Delta\,f}\right]^{1/2}$$

Since D, h, e, f, and r are all constant for the OBSERVER sensor, S/N may be expressed as

$$S/N = (1.82 \times 10^2) \quad \left[H_\lambda\,\tau^2\,\rho\,\cos^2\psi\,\delta\lambda\,\sigma\,R_c\right]^{1/2}$$

Figure 1.7 and 1.8 show H_λ and τ as functions of wavelength. The values of R_c are determined from Figure 1.5 . The reflectivity, however, is determined by the ground scene. It can vary from as low as .03 for bare soil to as high as 0.9 for snow. However, a reasonable maximum to expect in non-snow covered scenes is 0.5. For our calculations of representative S/N values, we will use ρ_{min} = .03 and ρ_{max} = 0.5. $\delta\lambda$ and σ are determined by the physical system and ψ is determined by the orbit. Using these parameters and the expression for S/N, we have generated S/N values for all wavelength bands at maximum and minimum reflectivity and maximum and minimum sun-vertical angles. Since the sun-vertical angle is zero at the "operational equator", we can expect the system to respond best between 23.5° N latitude and 23.5° S latitude. The resulting S/N values are listed in Figures 1.9 and 1.10.

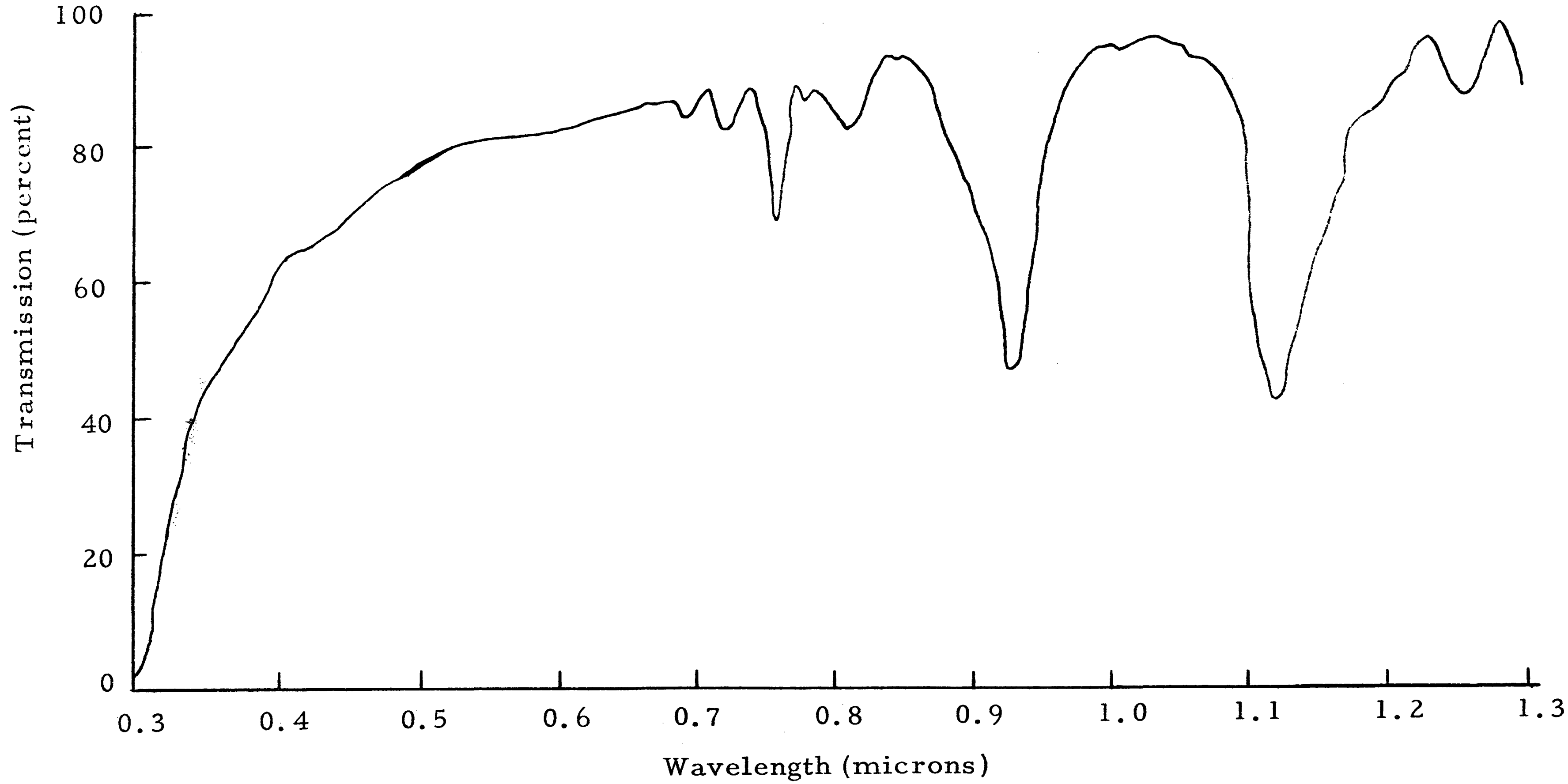


Figure 1.7 Optical Transmission of the Atmosphere as a Function of Wavelength

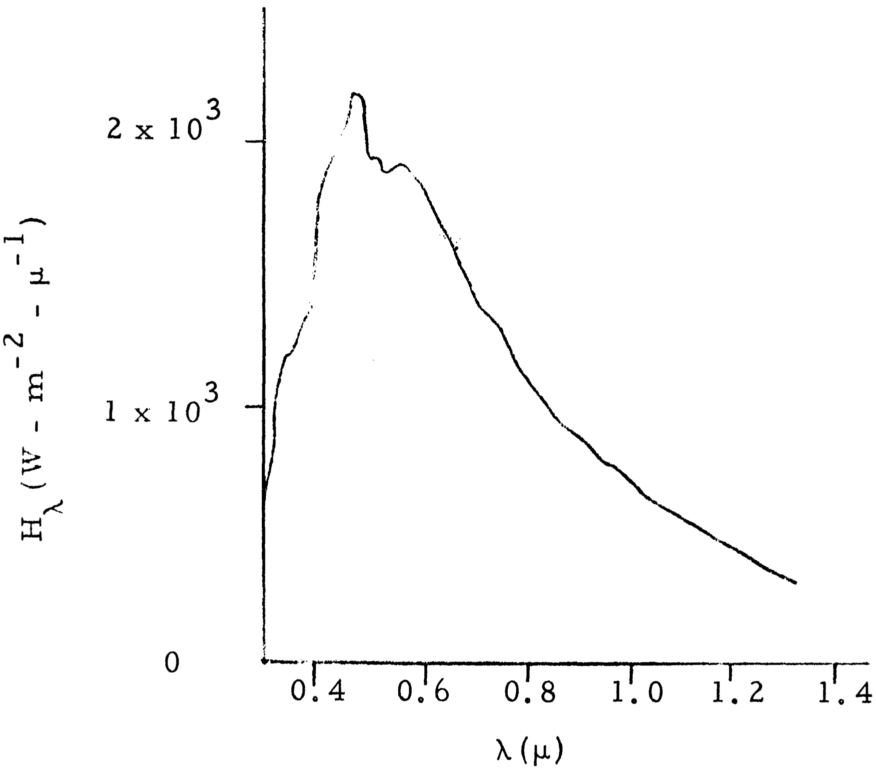


Figure 1.8 Spectral Irradiance vs. λ

Wavelength Interval	.40 to .50μ	.5 to .58μ	.58 to .62μ	.62 to .66μ	.66 to .72μ	.72 to .80μ	.80 to .88μ	.88 to 1.0μ
$(S/N)_{min}$ for $\psi = 80^o$	3.4	2.9	1.5	1.1	.8	.7	.9	.6
$(S/N)_{max}$ for $\psi = 80^o$	14.0	11.9	6.2	4.4	3.1	2.9	3.7	2.1
$(S/N)_{min}$ for $\psi = 0^o$	21.1	18	9.3	6.7	4.6	4.4	5.5	3.3
$(S/N)_{max}$ for $\psi = 0^o$	86	73	38	27	19	18	22.5	13.3
Phototube Type	S-10	S-10	S-10	S-10	S-10	S-1	S-1	S-1

Figure 1.9 Representative S/N Values

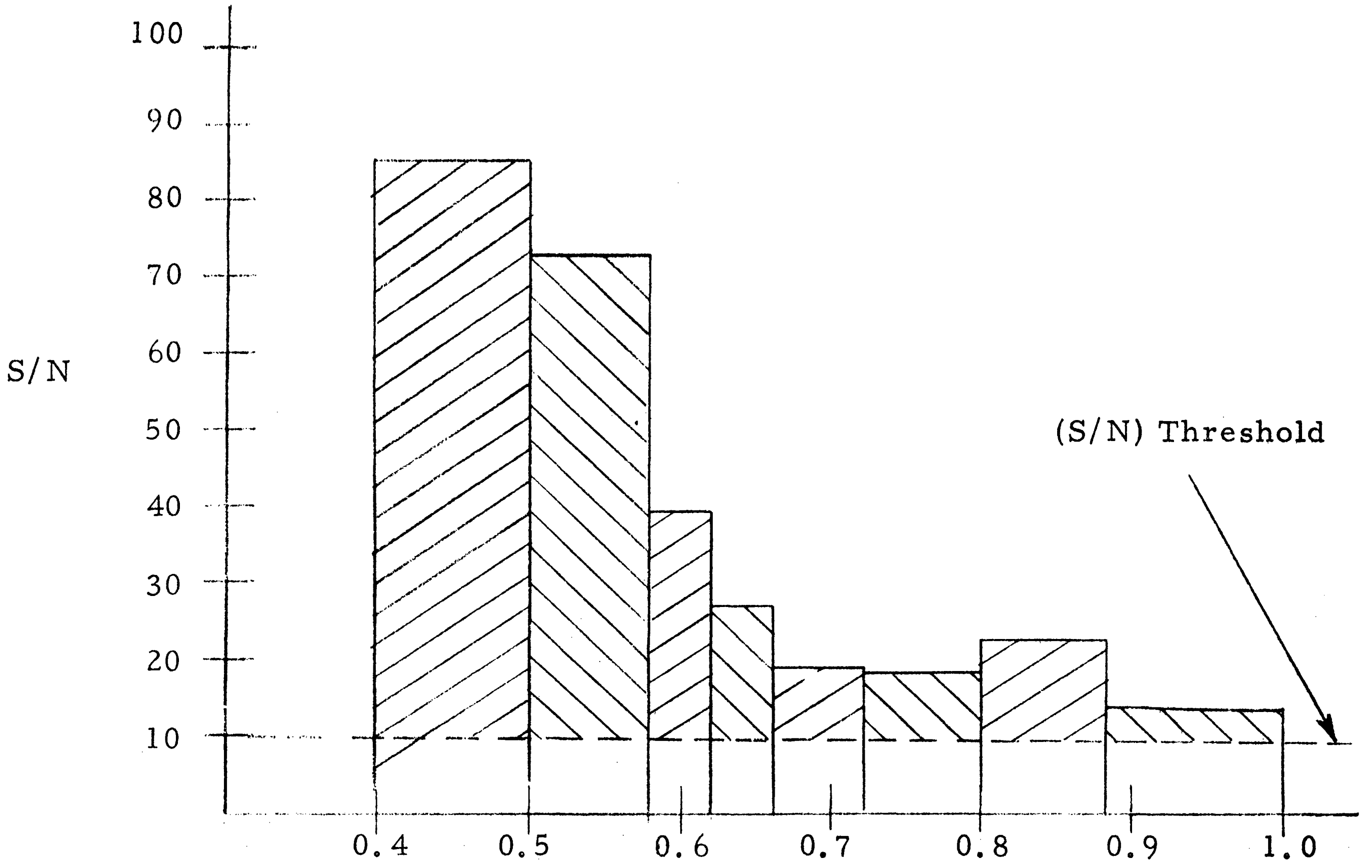


Figure 1.10 S/N vs. Wav(ength Interval (μ)

1.5 AREAS OF CONSIDERATION

1.5.1 Cloud Cover

The ability of the sensor system to detect the reflected light of the earth's surface is directly related to the amount of cloud cover. The greater the percentage of the total sky that is covered by clouds, the greater the number of passes by the satellite to insure adequate probability of detection.

The OBSERVER satellite is to provide information of global scope and interest. However, the agriculture and surface features of North and South America are of major interest and the passes needed to map these regions are used as the criteria for the satellite mission. Several other areas of interest such as Africa, the Middle East, and Australia show cloud cover near or below the North and South American average and could be scanned by the OBSERVER system if political considerations can be resolved.

The minimum cloud cover periods over North and South America coincide with the agricultural growing season of approximately April to September. Because of the importance of the agricultural data these cloud cover figures were used as a basis for the calculation of the number of satellite passes needed for the mission and the number of detectors that could be used. Cloud cover maps show that from April to September the average cloud cover is about 60% of the sky. Using this figure of 60%, a probability of detection of about 80% is found for three passes by the same set of detectors. Orbital data shows that one complete pass of the earth will be made every two weeks. The passes required will take about one and one-half months to complete. Although a high probability of detection is important, shorter cycles in the mapping process are more desirable for data interpretation. Because of this, the three pass cycle was adopted for OBSERVER.

1.5.2 Redundancy

Successful sensor operation is dependent on the reliability of the scanning motor, the detector array, and the calibration element.

The scanning motor is the heart of the scanning mirror subsystem. A failure of this motor will make the operation of the sensor dependent on the final orientation of the scanning mirror as it slows to a stop. Two possibilities can arise because of this orientation, either a small amount of light may be collected or none may be collected. Neither of these possibilities is advantageous to the operation of the system. The importance of the motor, along with its small size and light weight, make redundancy essential and easy to implement. The OBSERVER sensor will therefore be provided with two scanning motors.

The detector array of photomultiplier tubes is also of critical importance. A failure of one of the tubes will mean the loss of one of the spectral channels, but will not seriously effect the success of the mission. The tubes show good reliability over extended periods of time and use. This consideration along with the difficulty of providing a parallel set of fiber optic bundles for a second set of detector tubes led to the decision not to provide a redundant set of photomultiplier tubes.

The calibration element of the sensor system is important if the data is to be interpreted correctly. The light emitting diode and fiber optic bundle that make up the calibration unit can easily be duplicated and such a redundant unit is included in the OBSERVER system.

1.5.3 Data Interpretation

Because of the varied uses for the data produced by a mission such as OBSERVER, it is important that the raw data be highly accurate. The data can be distorted in several ways, including variation in detector performance, variation in calibration source output, and errors in the orientation of the system caused by stabilization limitations. The shortcomings of the electronics and the scanner can be greatly reduced by sufficient housekeeping data. The housekeeping data should include the current drawn by the calibration source and the detectors, and the angular speed of the scanning mirror. The pointing errors of the satellite cannot be overcome by the system itself. Errors in the capability of the gravity-gradient system and the tracking radar cause an uncertainty in knowing exactly what the sensor is looking at at any one time. This can be overcome to some extent by use of an attitude sensing system. The data from this system along with the errors in radar tracking give an error of less than 1,600 feet on the ground.

1.5.4 Bearings and Lubricants

The mission objective of one year operation and the continuous operation of the scanning mirror produce problems with the bearings and lubrication of the mirror drive shaft. It is impractical to use conventional bearings and lubrication because of the high vacuum environment of space. The oxidation of the metallic surfaces and the evaporation of the lubricants effects the performance of the bearings. Progress has been made in the field of solid-film lubricants, but the reliability for extended missions is low. At present, the best solution is to use a hydrocarbon or silicon lubricant sealed to prevent excessive evaporation loss and to prevent contamination of the optical surfaces. To further extend the lifetime of the lubricant, small reservoirs of lubricant can be contained in the system. The scanning radiometers of the Tiros meteorological satellites have employed such a reservoir system with good success.

1.5.5 Political Considerations

Because of the scope of OBSERVER'S ground coverage, there will arise the problem of the satellite's capability to produce data violating a nation's sovereignty and security. Several characteristics and possibilities of the OBSERVER system, however, would seem to minimize this problem. For example, a spy satellite would require a much finer resolution than the 1,000 feet resolution of OBSERVER. Secondly, the fact the OBSERVER'S data is broadcasted in real time to all receivers in range seems to preclude the fact that any one group is using the satellite for espionage purposes.

1.5.6 Alternative Systems

In determining what type of sensor system should be used for the OBSERVER satellite, several restrictions and considerations were encountered. Among these were problems such as:

1. Size and weight restrictions imposed by the Scout Launch Vehicle
2. Ground resolution
3. Data bandwidth
4. Cost and availability of the sensor system
5. Usefulness of the data produced by the sensor
6. Political considerations of remote sensing
7. Tradeoffs within the design of the chosen system

Several different sensors were considered for Project OBSERVER and were found to be incompatible with the restrictions mentioned above. These sensors are presented below with a brief description of each.

Television. The weight and power requirements of television systems is a major drawback to their use in the OBSERVER satellite. A typical RCA vidicon system would weight 15-30 pounds and would require 50-75 watts of power. Because each camera can look at only one spectral interval, a package of several cameras or a system of filters would have to be employed. The time required to process a picture and prepare the screen for a second photograph would limit the number of filters that could be used to two. Such a limitation on the number of spectral intervals available for a monitoring makes systems of television cameras unfeasible for the OBSERVER satellite.

Thermal Mappers. The data received from thermal mappers is very limited in its applicability to a wide range of users. A useful application of thermal mappers has already been demonstrated in the meteorological satellites and future possibilities exist in oceanographic studies, but in each case only specific experiments are possible. A second factor against the use of thermal mappers is that the time constants of the detectors is too long to permit OBSERVER to monitor a ground resolution element small enough to produce useful data.

Infrared Detectors. Sensors that would look at wavelengths greater than 1.0μ would provide much useful information that could not be obtained in the visible wavelengths. However, the problems associated with generating a large signal-to-noise ratio with the present semi-conducting detectors require a system too large to be compatible with the OBSERVER satellite structure. In a larger satellite or in a more sophisticated program, an infrared capability should be of prime importance.

1.6 REFERENCES

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2

COMMUNICATIONS SYSTEM

2.1 DATA TRANSMISSION SYSTEM

2.1.1 Sensor Output

The data output of the sensor must be transmitted to the ground station and analyzed. This is accomplished by the data link transmitter. The system provides for eight detectors on board the satellite; of these, the communications system will handle four at any one time. Using a four-faced mirror as the mechanical scanner, each active detector will receive four equally spaced flashes of light during one revolution of the mirror. These correspond to four scanned strips of data on the ground. Between each of these data segments there is a time when no information is sent by the detector; this matches the 70^{o} between two faces of the mirror where the geometry prevents any scan data. This space will be used for in-flight calibration and synchronization signals. The multifaced mirror provides OBSERVER with the means of physically multiplexing the data from the detectors without increasing the bandwidth needed to transmit this information. During each quarter revolution of the mirror, (of the four detectors which "see" the earth,) only one of the detectors will be read. During the second quarter, the next active detector will be read, and so on until all four have been read. At this time the cycle would repeat. Figure 2.1 shows the output of each active detector and its relationship to the combined output. The actual time values of the combined output can be seen in Figure 2.2. The switching trigger, which takes one active detector each quarter revolution and puts it on the combined detector output should operate in precise relation to the spin rate of the mirror. For this reason it is necessary to operate the switch from a mask attached to the mirror shaft. This can be accomplished by using a photo-cell which is activated by a small light source that shines through the mask at the proper times.

2.1.2 Transmission Details

The bandlimited output of the sensor has a frequency response of 136 KHz. Since it is important to keep the final bandwidth of the transmitted data to a minimum while maintaining an adequate signal to noise ratio on the ground, OBSERVER has chosen to directly FM the RF carrier. Other communications systems were considered but

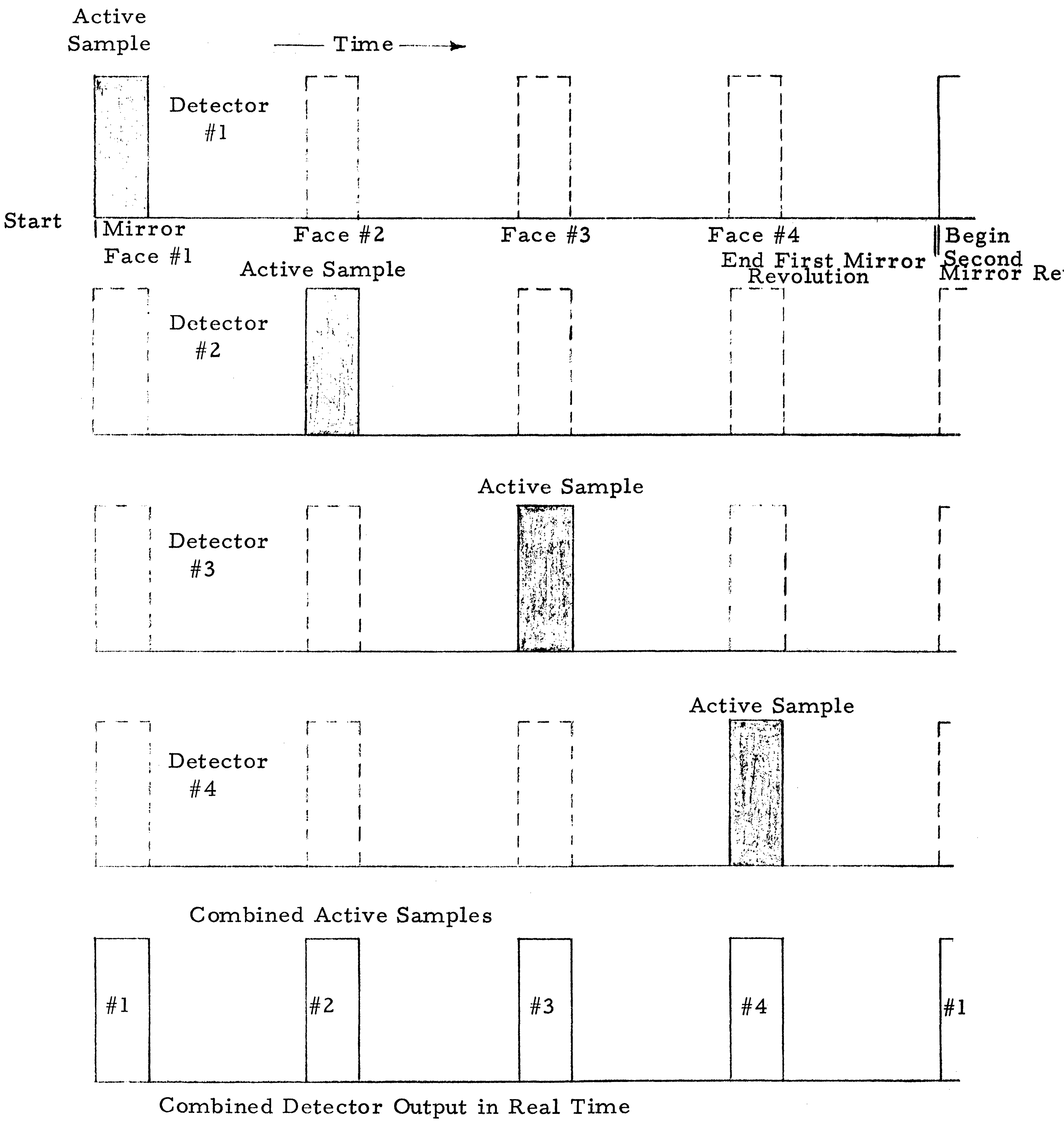


Figure 2.1 Detector Data Sampling Scheme

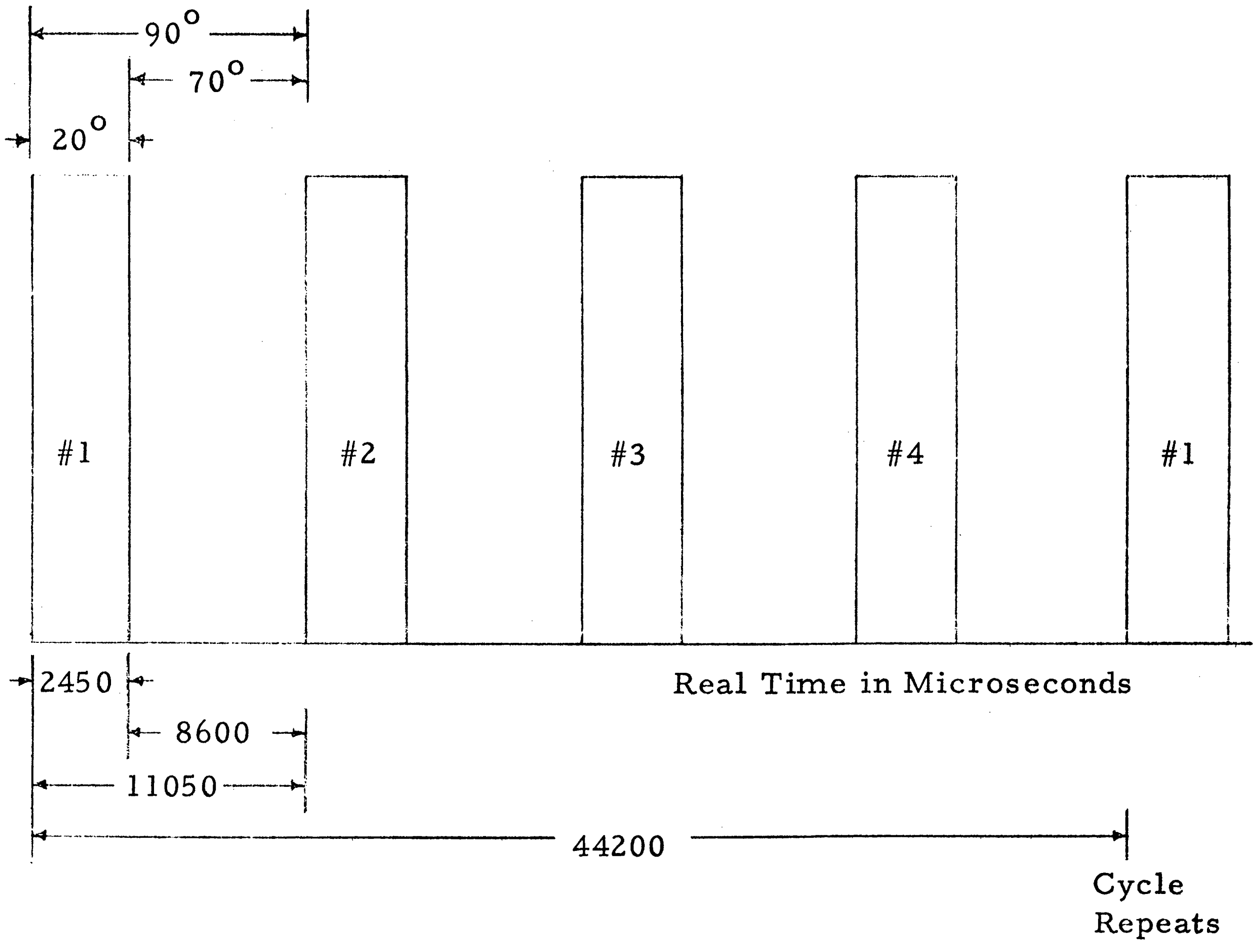


Figure 2.2 Sensor Time Scale vs. Data Output

the large increases in bandwidth over FM eliminated them for that reason. For an RMS error of approximately 1%, the bandwidth is 2.38 MHz. See Appendix B for the sample calculations. Effects of doppler shift add .111 MHz, and system frequency instabilities also increase it by .04 MHz. Considering 5% guard bands, OBSERVER'S final bandwidth is 2.77 MHz. On a carrier of 2.202 GHz, the frequency deviation is $\pm$ 1.385 MHz. A block diagram of this system can be seen in Figure 2.3. The entire system will be controlled by the command link, either directly or indirectly through the computer's stored command capability.

2.1.3 Power Constraints

Power transmitted is an important constraint on the system. Using a 40' parabolic receiving antenna on the ground, the power needed to transmit the data is 1.11 watts. If we consider a 14' receiving dish, often portable, the necessary power transmitted is 9.05 watts. See Appendix B for sample calculations. To handle these two power regimes, OBSERVER is utilizing two low-power transmitters to reach the vast majority of the ground stations with one high-power transmitter for the smaller antennas. Two low-power transmitters are used to provide the needed redundancy to promote a one year life-time.

2.1.4 Trade-Offs

2.1.4.1 Digital Magnetic Tape Storage. OBSERVER has considered using a digital magnetic tape storage device on board the satellite. This would provide the system with the capability of obtaining data from beyond the range of the ground station receivers. For the sensor output, the minimum bit rate would be 1.36 megabits per second, assuming a five bit code and a sampling rate of twice the highest frequency. Considering the limited ability of tape storage density to be 8000 bits per inch, for a 10^{5} binary error probability, the recording rate would be 170 ips. If the data were recorded for 160^{o} of one orbit and dumped at one station, it would require a tape speed of 757 ips to dump the 35,700 feet of tape obtained during the recording period. No satellite tape system, up to this time, has come near this capability and therefore this approach was discarded.

2.1.4.2 Analog Tape Storage. Analog tape storage was also considered. This requires the storing of a frequency corresponding to the varying voltage levels of the sensor output. With a frequency response of 136 KHz, the system considered a modulation frequency of from D.C. to 200 KHz. Using the Inter-Range Instrumentation

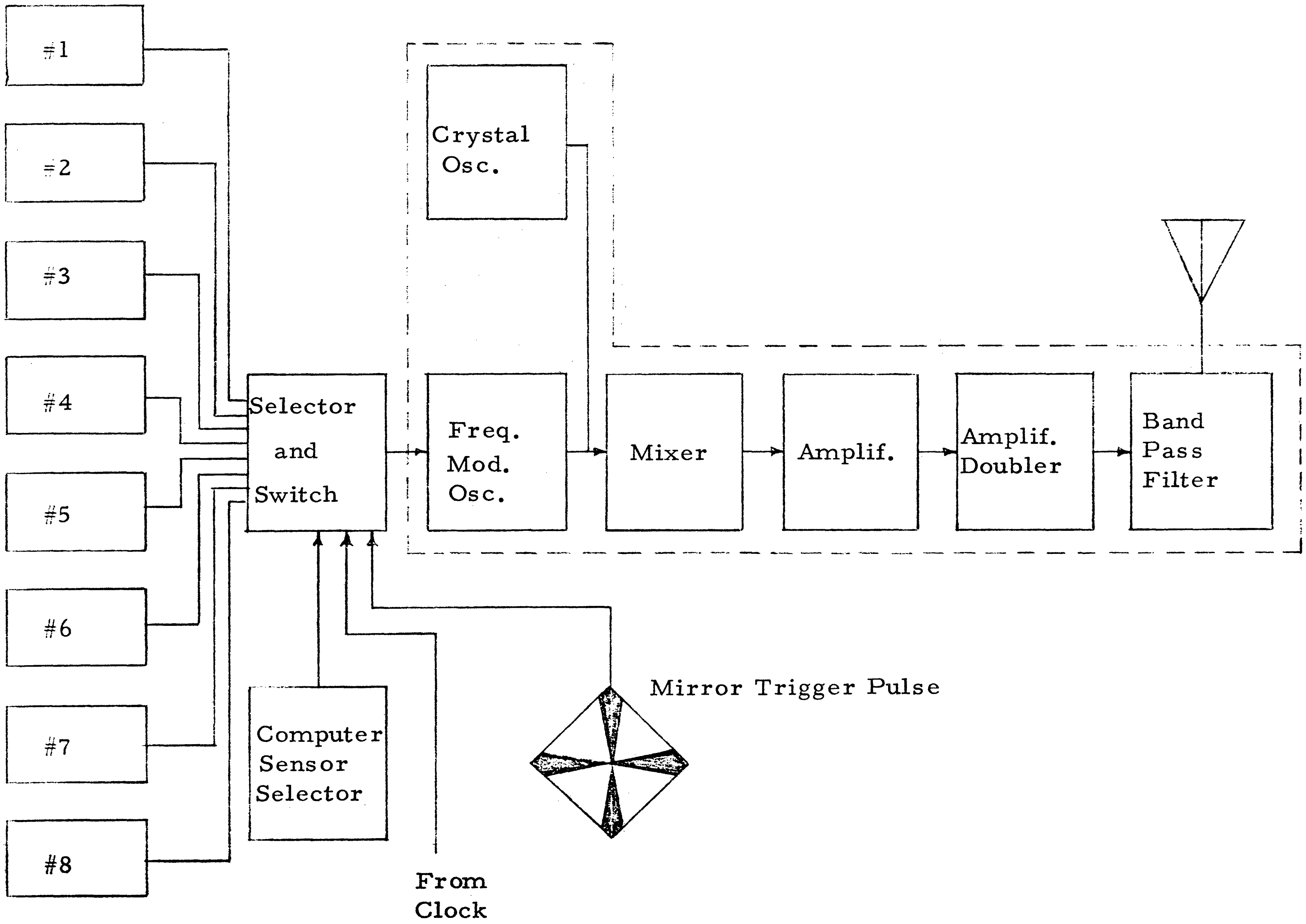


Figure 2.3 Data Transmission System

Group (IRIG) standards, this yields a tape recording rate of 60 ips; of course, dump times would be considerably higher. At these tape speeds, the tape and tape heads have a short life-time and would require a number of redundant on-board elements. For this reason, the scheme was dropped.

2.1.4.3 Buffer Storage. In considering buffer storage, with the same bit rate as that of the digital magnetic tape storage, the memory required to hold 160° of one orbit's data would be 3.43×10^{9} bits. This would require an operating input rate of 735 nanoseconds per bit and a readout rate of 165 nanoseconds. Though these response times are feasible for present buffers, the size of this unit would be excessive.

2.1.4.4 Analog Delays. The above limitations indicate the reason for using the live-time mode of transmission. OBSERVER believed that it might be possible to reduce the transmitted bandwidth of the live transmission by using analog delays. They would spread the 20° of data received from one detector over the hole between the next detector output pulse. Current analog delays do not have the ability to handle delays on the order of 10,000 microseconds but are capable of only a few microseconds. For this reason, analog delays were also dropped.

2.1.5 Computer Applications

OBSERVER has chosen an on board memory computer to handle the functions of command distribution. The details of this unit, the SPD-3, will be explained later. Using this device, it is possible to experiment with the compaction of the data during flight. If it could remove any adjacent bits of redundant data and code the level, it is possible to obtain significant reductions in the bandwidth. One assumption which has to be proven is if there is a significant amount of adjacent identical data which can be removed. Since the system considers visible and near infrared reflections, the cloud cover is opaque. If the computer knew the signature of cloud data, it would be able to remove this prior to transmission. This would eliminate about half of the required tape storage on the ground. It must be remembered that in order to obtain a reduction in the transmitted bandwidth, the data would have to be reduced significantly since the relative bandwidths of a digital system are much higher than FM systems.

2.2 GROUND SYSTEMS

2.2.1 System Availability

OBSERVER has chosen the STADAN (Space Tracking and Data Acquisition Network) and its sites to handle the functions of tracking, command, and telemetry readout. For tracking, it will use the beacon transmitter on board OBSERVER in conjunction with the Minitrack Interferometer System, the Minitrack Optical Tracking System, and the Goddard Range and Range Rate System. These three systems are included in STADAN. Tracking accuracies for orbit parameters are ± 20 seconds of arc (± 184 feet) at 313 n.m. The commands to OBSERVER will be handled on the SATAN (Satellite Automatic Tracking Antenna Network) Command System. All these systems have a proven record of performance. All sites which have either a 14', 40', or 85' parabolic antenna will receive the data from OBSERVER on S-band. Table 2.1 lists the sites OBSERVER will use and the equipment available at each site.

2.2.2 Conversion to S-Band

The Unified S-Band Network was considered and actually preferred for the transmission of the sensor data. However, due to congressional cutbacks, this network is not expected to be operational until the early 1970's. OBSERVER can not count on the USB system being ready for its use, therefore, STADAN, which is only equipped to handle VHF and UHF transmissions, will have to be updated to handle the S-band.

> "The conversion of VHF telemetering ground stations to S-band offers some serious problems. However, none of these seems to be insurmountable, nor beyond that which a knowledgable application of existing technology cannot nicely overcome. Much new equipment will be required.... The one serious danger is that the new S-band RF system will be treated as the old with respect to alignment, operational verification, calibration, maintenance and inspection, and the compilation of a running history of performance... The generation of spurious intermodulation products is a problem area that will require careful investigation."[5]

STADAN Network

Site	Telemetry	Tracking	Command
Barstow, California	40'		X
Carnarvon, Australia		3	
Fairbanks, Alaska	85'	1 2 3	X
Fort Meyers, Florida		1 2	
Johannesburg, S. Africa	40'	1 2	X
Lima, Peru		1 2	
Orroral, Australia	85'	1 2	X
Quito, Ecquador	40'	1 2	X
Rosman, S. Carolina	85'	1 3	X
Santiago, Chile	40'	1 2 3	X
St. Johns, Newfoundland		1 2	
Tananarive, Malagasy Rep.	40'	1 2 3	X
Toomba, Australia	40'		X
Winkfield, England	14'	1 2	

TRACKING

1. Minitrack Interferometer System
2. Minitrack Optical Tracking System
3. Goddard Range and Range Rate System

COMMAND

Commands to the spacecraft will be given over the SATAN Command system.

Table 2.1

2.2.3 Data Compilation

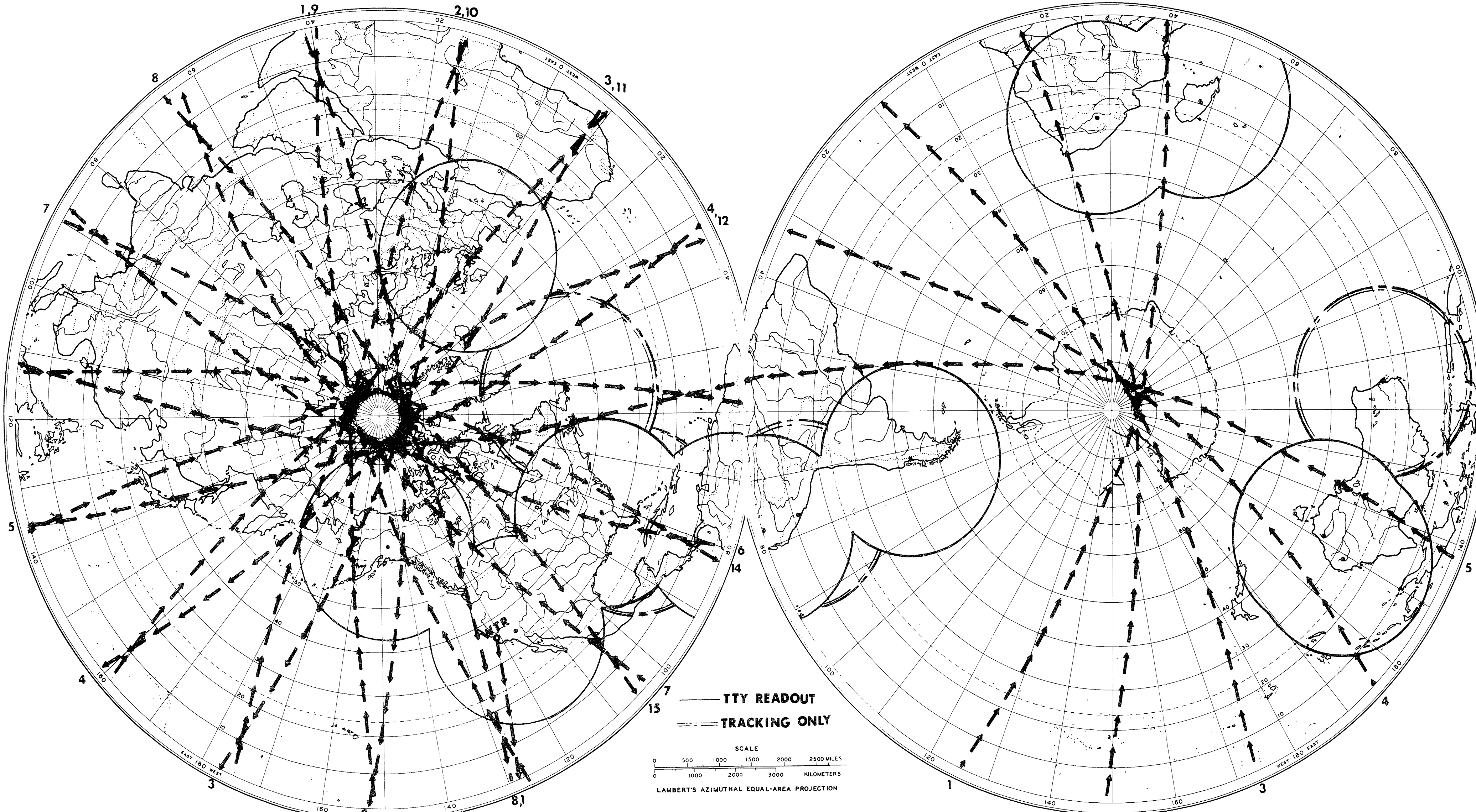
Since OBSERVER will be used for direct-link communications, all STADAN sites, with the exception of Darwin, Austrailia, that have parabolic antennas will be used for handling the data. Information processing support is a central activity for the entire STADAN and is located at the Goddard Space Flight Center (GSFC). This facility will be involved with the processing of scientific data from the satellite sensor. The bits of data obtained from the spacecraft are recorded on magnetic tape at high speed by STADAN sites and forwarded to the information processing center. These tapes are reviewed, edited, converted to computer format, and then processed to a functional readout for use by the analyzer. In the event that only certain wavelengths of the information are of use to a particular user agency, each wavelength will be compiled as a separate global map in that discrete wavelength band. Figure 2.4 shows the ground station's tracking and data separation.

2.2.4 Coverage Capabilities

The coverage capabilities of any ground station will be taken as approximately equal. The first step is to determine the look distance for each station. For an altitude of 313 n.m., the station can "hear" approximately 1950 n.m. This uses the refraction of the atmosphere to bend the transmission beyond the direct line between the satellite and the horizon. It is admitted that a number of errors occur in the signal at this extreme distance. Figure 2.5 shows the tracking geometry. To consider handling the data which will be sent to the ground, we have taken a line 5° above the horizon as the nominal maximum range which will eliminate most errors arising from horizon interferences. On this assumption, we find that a pass through the diameter of a ground station range circle yields a dump time of 9.44 minutes. This range circle, which also shows the maximum area of coverage of the sensor has a radius of 1209 n.m. This corresponds to a ground radius of 1150 n.m. The distance between the two limits of the range circles (1950-1209 = 741 n.m.) will give the ground stations nearly three minutes to obtain a good fix for the satellite pass. Then as soon as OBSERVER passes into the inner range circle, the sensor data transmitter can be turned on. (See following page.)

2.2.5 European Advantages

OBSERVER felt that data obtained over central and eastern Europe was of importance. For this reason, OBSERVER chose to use the Winkfield, England site as a dump station also. This does



GROUND TRACK ON POLAR MAP

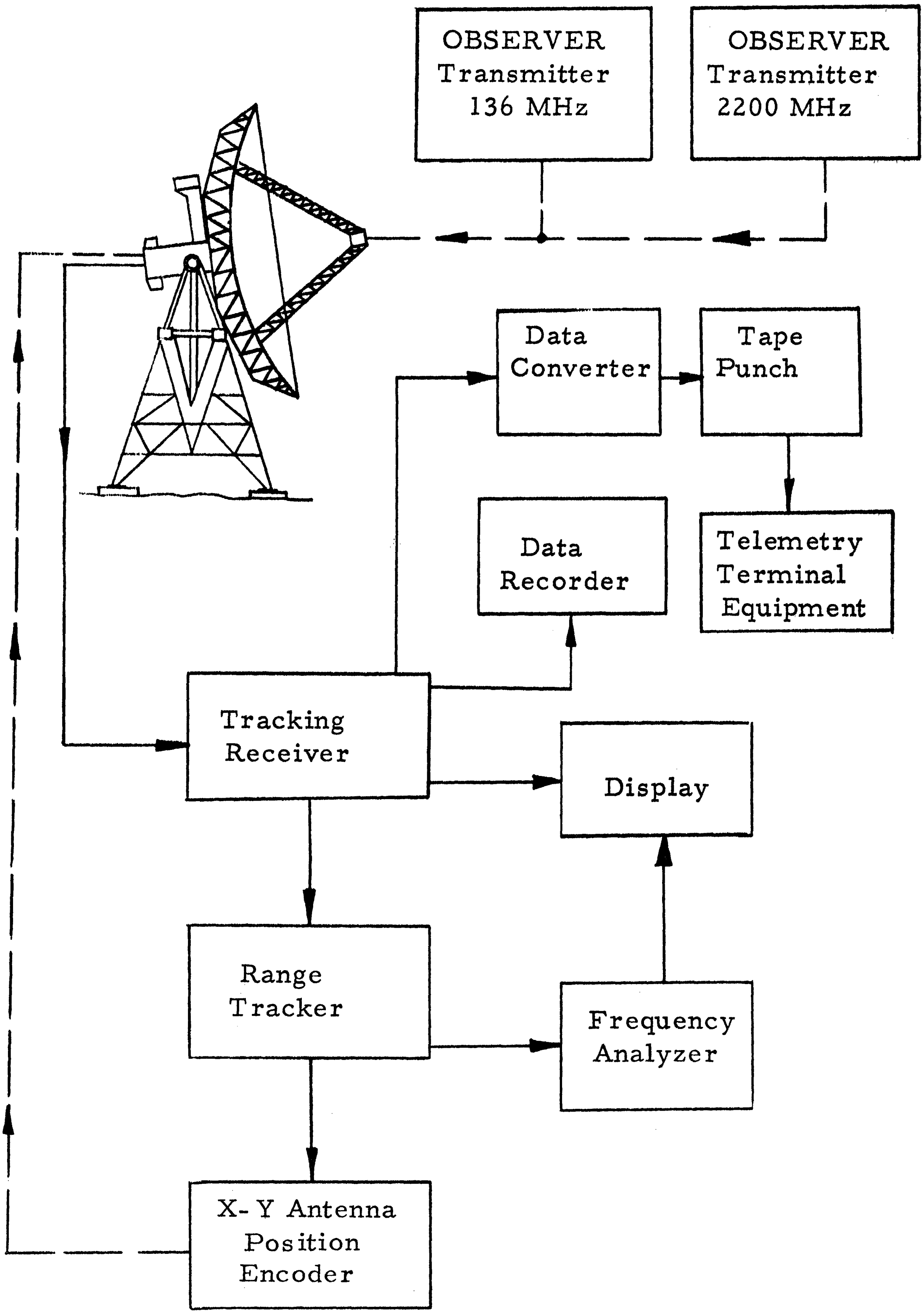


Figure 2.4 Ground System

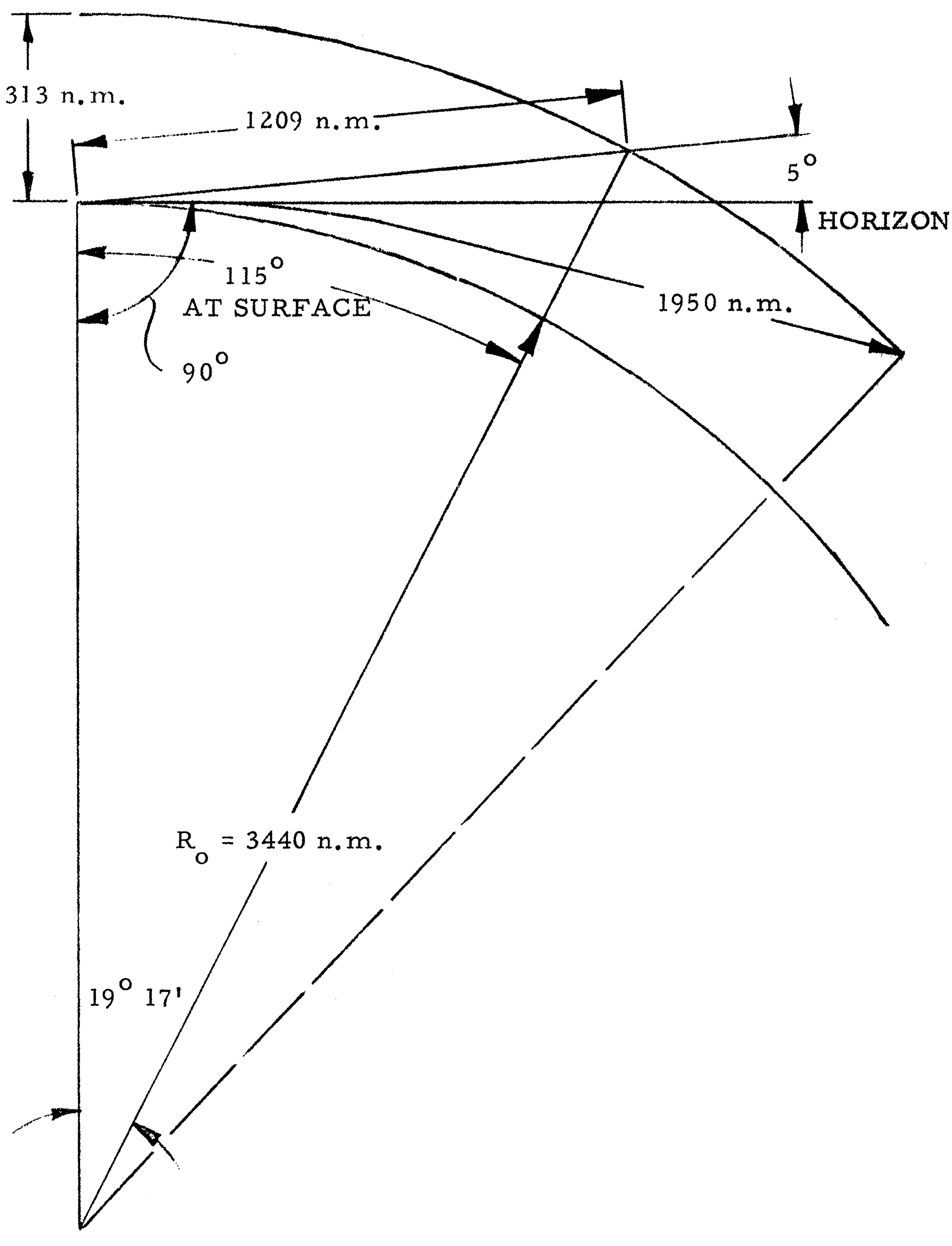


Figure 2.5 Satellite Tracking and Transmitting Geometry

impose a problem because the station has only a fourteen foot parabolic dish. To transmit to this station requires higher power output. For this reason we have added a high power S-band transmitter to the satellite to be switched on in place of the normal transmitter in this area. Figure 2.6 shows a block diagram of the system and its application to data interpretation.

2.3 HOUSEKEEPING AND COMMAND SYSTEMS

2.3.1 Housekeeping System

Figure 2.7 shows the housekeeping transmitter system. This system monitors the condition of the satellite. For this transmission we have chosen a PCM/FM communications link. The PCM unit can handle 45 analog inputs which are sampled, multiplexed, and digitized. This small, integrated circuit unit, includes its own clock and A/D converter along with the commutator and other associated electronics. It can handle inputs of up to 50 cps or supercommutate those which require higher frequency responses. The sampling rate is fixed at 100 samples per second per channel. From the PCM unit, the signal is sent to the VHF-FM transmitter. This frequency modulates a carrier of 137.5 MHz. Table 2.2 is a list of the functions which will be monitored by the housekeeping system. The total bandwidth of the down link system will be kept to the standard of $\pm$ .25 MHz of the carrier. This includes the guard bands. It was felt that a redundant PCM unit was not necessary for the housekeeping system because it is not required to be active for long periods of time or at close intervals.

2.3.2 Command System

The command system uses a PCM/FM transmission up link on a carrier of 136.5 MHz., see Figure 2.8. The bandwidth requirements which will be maintained are similar to those of the housekeeping system. At the entrance to the receiving unit, it will have a tuned circuit which can be activated from the ground to turn on the receiver. In this manner, it will not have this unit on continuously. This system will receive the commands from the ground, put them into temporary storage and readout the commands back to the ground from this temporary storage through the housekeeping system for purposes of verification. Upon receiving an enable signal, the command previously stored in the memory will either be activated directly or held for activation at a later time. This capability will allow the system to complete various functions which should occur beyond the range of any ground stations and which can not be pre-programmed before lift-off. Since this series of functions is critical

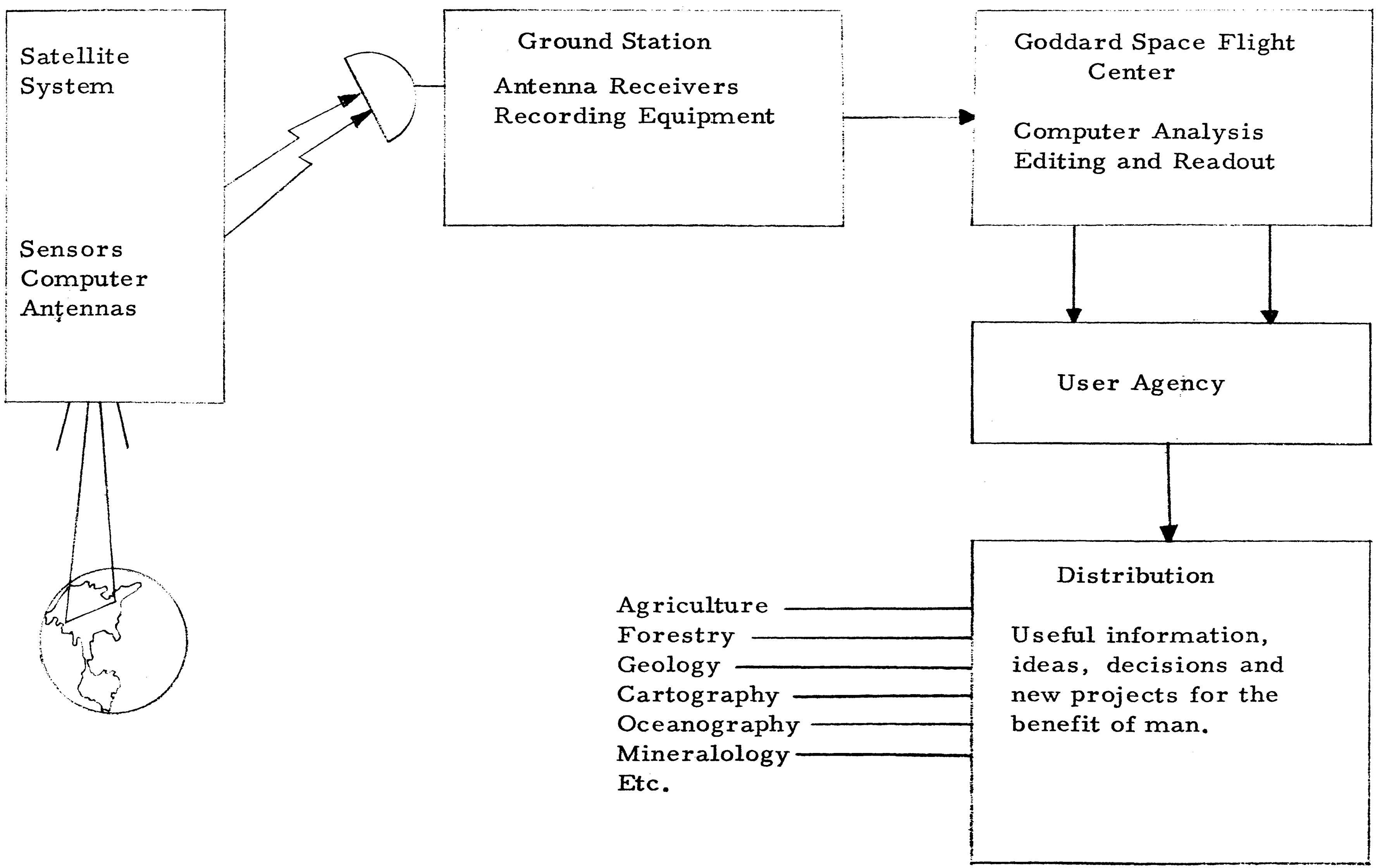


Figure 2.6 Data Distribution Block Diagram

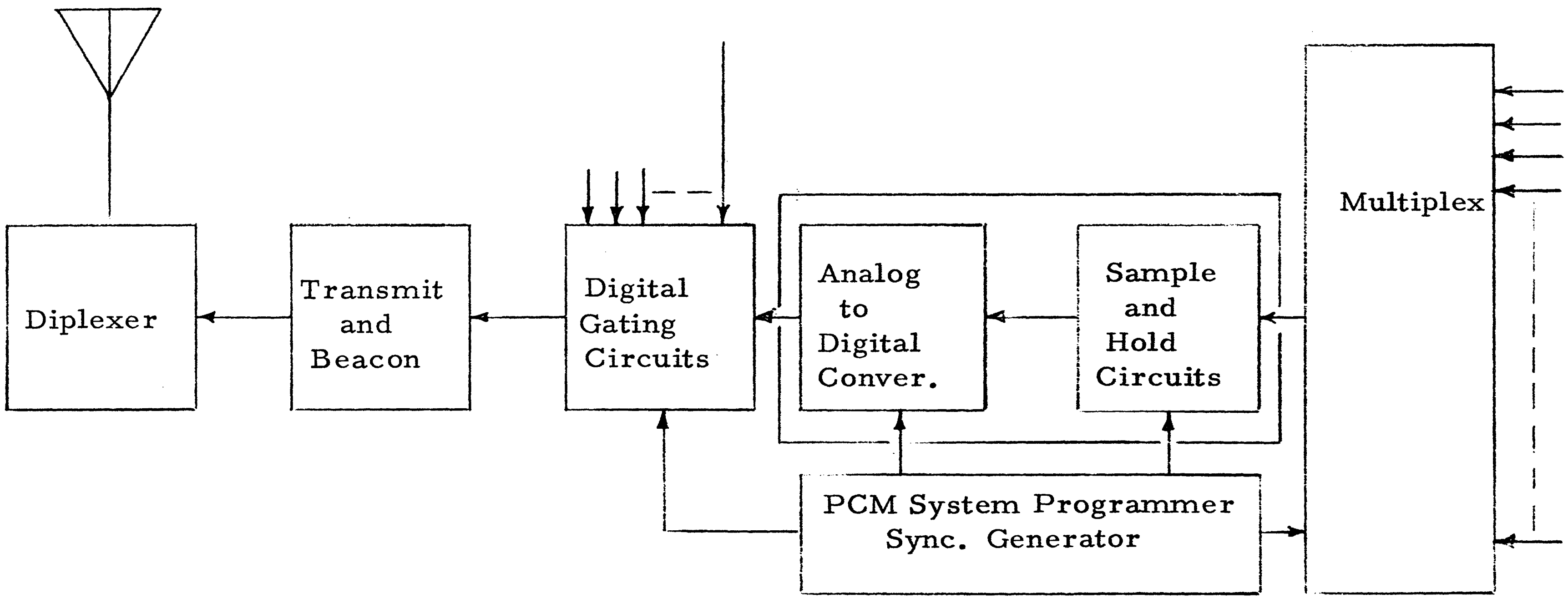


Figure 2.7 Housekeeping Transmitter System

Table 2.2

Housekeeping Functions

1. Motor spin rate (mirrors)
2. Temperature of the photomultipliers
3. Bias current on the amplifiers
4. Communications compartment temperature
5. Battery temperature
6. Delta "V" compartment temperature
7. Solar panel #1 temperature
8. Solar panel #2 temperature
9. Skin temperature probe #1
10. Skin temperature probe #2
11. Skin temperature proble #3
12. Solar cell array current
13. Solar cell array voltage
14. Battery charge current
15. Battery voltage
16. Shunt regulator current
17. Switching regulator's output voltage (communications)
18. Switching regulator's input voltage
19. Switching regulator output voltage (sensor)
20. Magnetic sensors
21. Side Sun sensor
22. Center Sun sensor
23. Command receiver monitor (digital, from temporary storage)
24. Data transmitter situation
25. Command receiver situation
26. Clock time
27. Position of gravity gradient boom #1
28. Position of gravity gradient boom #2
29. Position of gravity gradient boom #3

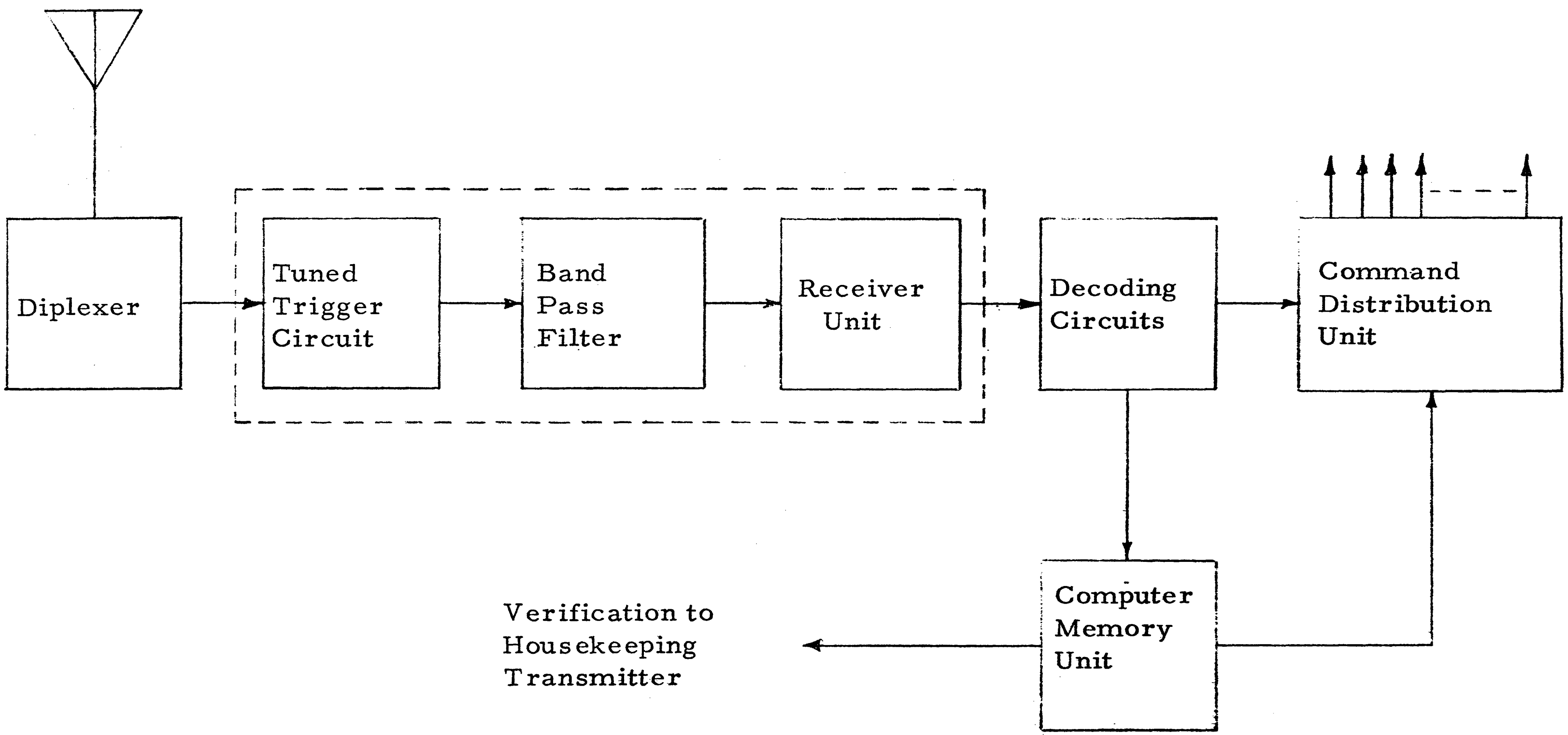


Figure 2.8 Command Receiver System

to the operation of the satellite, we have provided for a complete redundant command system with the exception of the memory. The SPD-3 memory computer will handle the vast majority of these functions including the distribution of the commands; see Figure 2.9. As a backup system, in case the computer fails, we have included a command distribution unit. It is felt that a second memory unit was not needed because of the reliability of current memory systems; instead, it will be disconnected from the computer if that unit should fail. The command system also has dual receivers, one as a back-up in case one fails. Table 2.3 is a possible list of the commands which this system will consider. In Appendix B is a list of the equipment we have considered and the power, weight, and volume of the communications system.

2.4 ANTENNAS

2.4.1 Data System Antenna

The data system will use a boxed-slot turnstile antenna to transmit the "S" band data down to the ground. The "look angle" of the satellite is 133 degrees as shown in Figure 2.10. The half-wave turnstile antenna is right hand circular polarized and is fed in phase quadrature. It realizes a polarization loss of - 1 db for the worst case of the station at the horizon + 5 degrees. Figure 2.11 shows the dimensions of this antenna which does not require any mechanical operation for activation or operation.

2.4.2 Housekeeping-Command System Antenna

The housekeeping and command system will share a half-wave turnstile antenna which is located across the front plane of the spacecraft. During the ascent, while the fiberglass shroud is covering the satellite, the whips will be folded into the nose of the shroud and as the shroud is parted, the spring loaded whips will pop outward. It is assumed that, if necessary, we will be able to receive and transmit housekeeping and command through the shroud. Further tests will have to be made with a mock-up satellite on a ground antenna range to determine the effects of the structure in blocking or reflection of the signal from any of the protruding obstructions such as the solar panels, gravity gradient booms, etc. At this stage, it appears that this will prove to have no major correctional problems. The antenna will be fed in phase quadrature and each whip will be 90 degrees to the next and in a plane perpendicular to the direction of travel. Each whip will be approximately 1.8 ft long to yield a half-wave turnstile. Figure 2.12 is a block diagram of the entire communications system showing the redundant units considered.

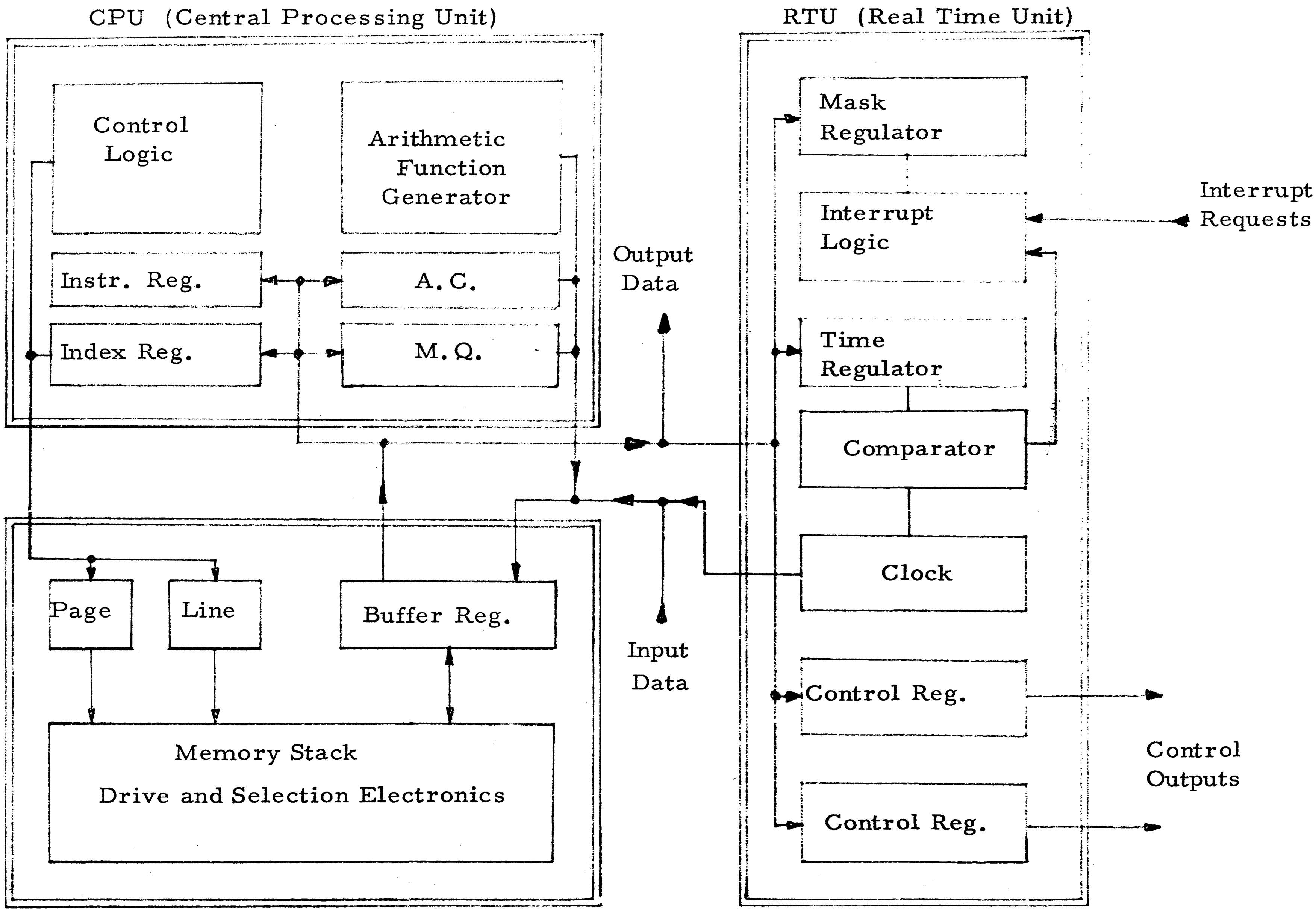


Figure 2.9 SDD-3 Memory Computer

Table 2.3

Command Functions

1. Start mirror motor
2. Switch on photomultiplier #1; switch off
3. Switch on photomultiplier #2; switch off
4. Switch on photomultiplier #3; switch off
5. Switch on photomultiplier #4; switch off
6. Switch on photomultiplier #5; switch off
7. Switch on photomultiplier #6; switch off
8. Switch on photomultiplier #7; switch off
9. Switch on photomultiplier #8; switch off
10. Command power on to valve (delta "V")
11. Command power off to valve
12. Pulse power to valve
13. Release panels from body (solar panels)
14. Switch to 90^{o} tent angle
15. Deploy panels
16. Command despin
17. Jettison despin ring
18. Extend boom #1 (gravity gradient)
19. Extend boom #2
20. Extend boom #3
21. Stop boom #1
22. Stop boom #2
23. Stop boom #3
24. Retract boom #1
25. Retract boom #2
26. Retract boom #3
27. Power on to "S" band transmitter #1 (low power)
28. Power on to "S" band transmitter #2 (low power)
29. Power on to "S" band transmitter #3 (high power)
30. Power off to "S" band transmitter #1
31. Power off to "S" band transmitter #2
32. Power off to "S" band transmitter #3
33. Turn on receiver #1
34. Turn on receiver #2
35. Turn off receiver #1
36. Turn off receiver #2
37. Turn on housekeeping transmitter
38. Turn off housekeeping transmitter
39. Command lock-out (protection)
40. Sensor slot #1
41. Sensor slot #2
42. Sensor slot #3
43. Sensor slot #4
44. Switch decoders

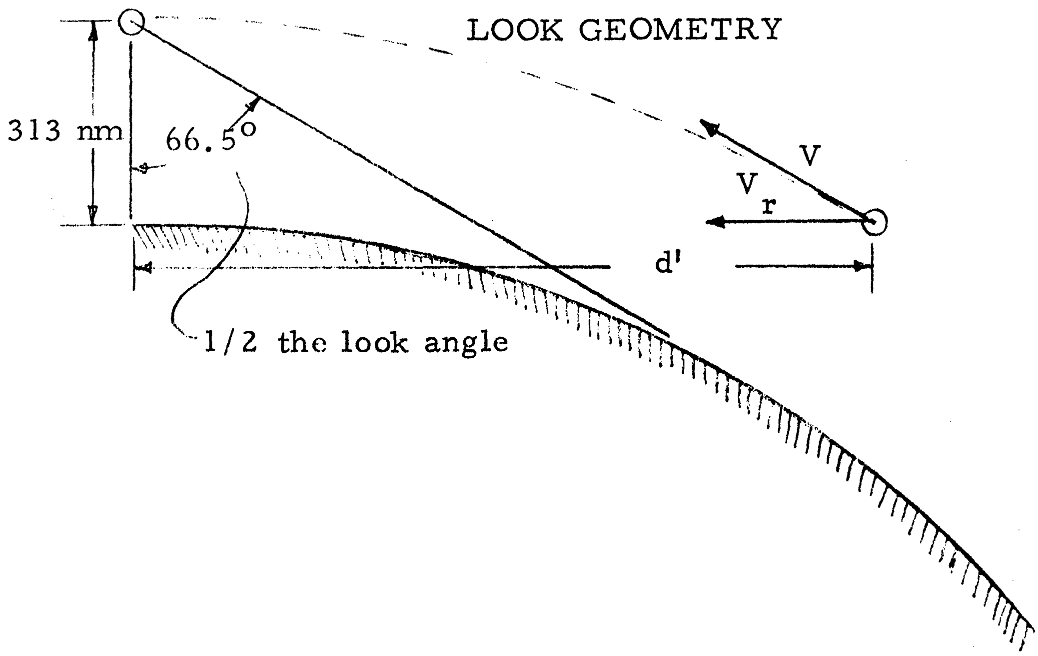


Figure 2.10

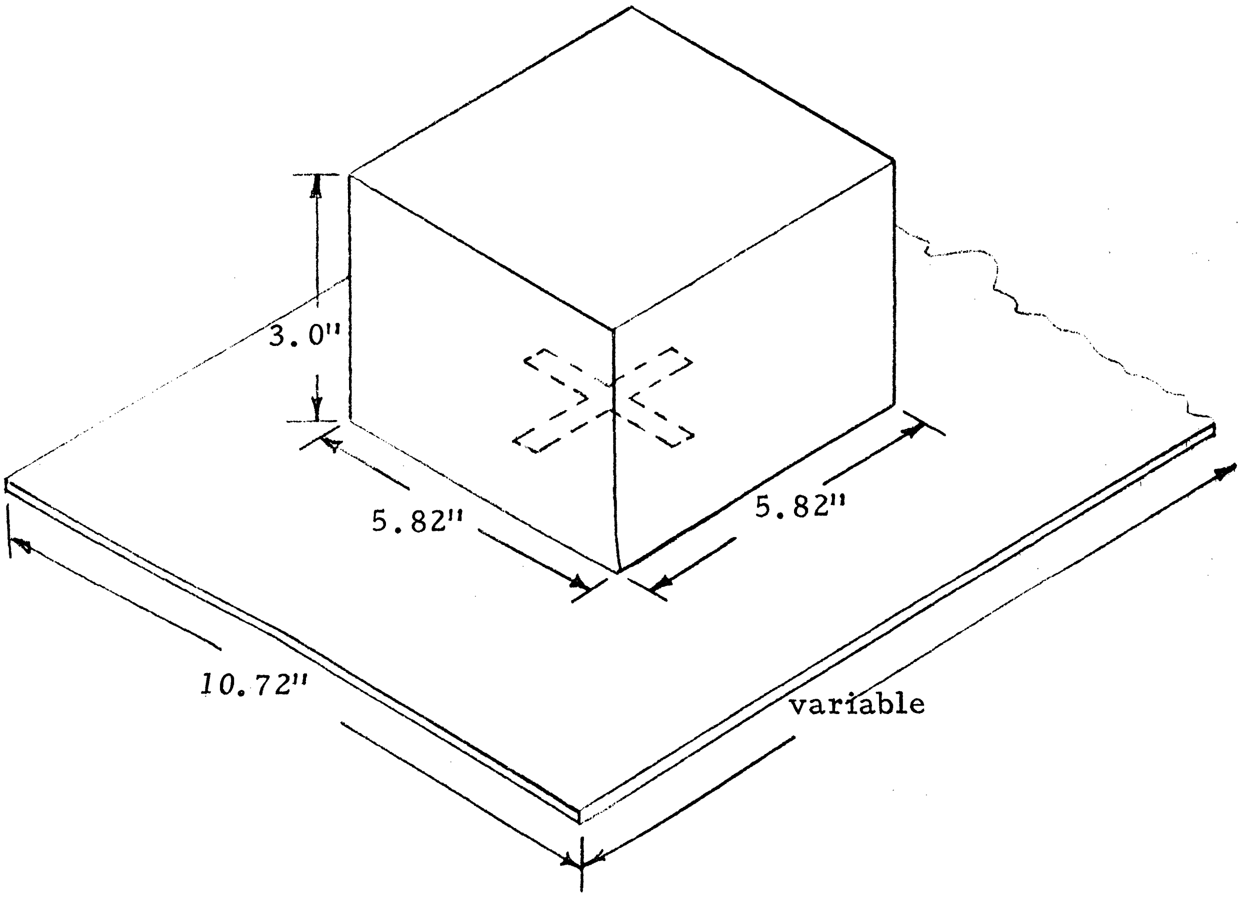


Figure 2.11 Data Transmitting Antenna

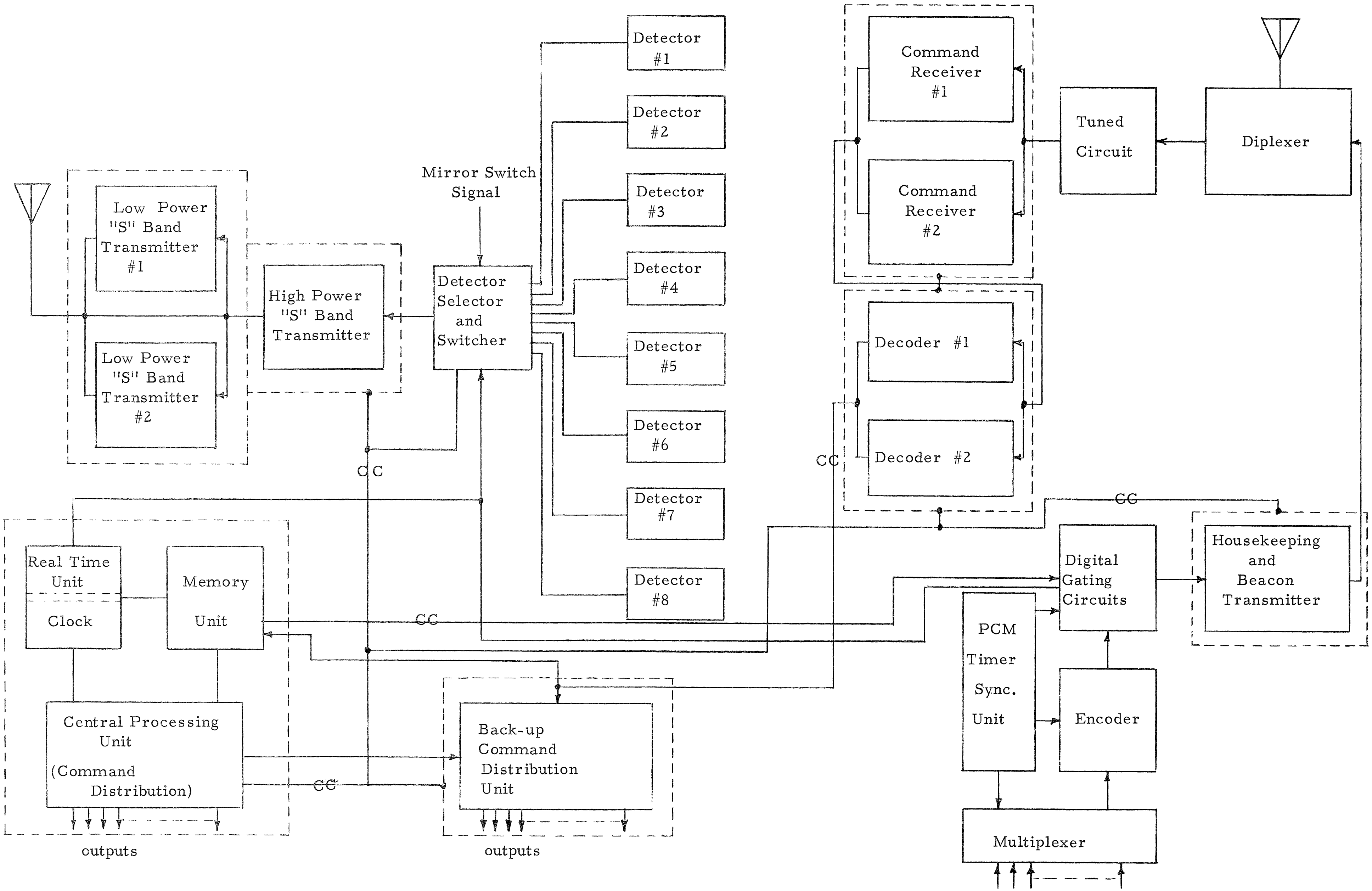


Figure 2.12 Communication Schematic

2.5 POWER MODES

Since power is a critical consideration to the operation of a satellite system, we have found that to transmit to the ground station in England, we will require that all other functions be shut off during this transmission. During part of the pass over England's dump station, in the lower latitudes, there is a sufficient amount of power available to allow the housekeeping system to be turned on. In this way, the data from the "S" band transmitter can be spotted by the attitude sensors to indicate where the data sensor is looking. Combination of the attitude sensors and the data link system will be synchronized on the ground to give the reference latitudes and longitudes. It is possible to program the computer with a priority shut-down which can turn off less critical instruments in areas where the power drain becomes critical. Table 2.4 shows the various operating modes which we are considering. The standard mode is the usual power requirement for all communication systems turned on. The shut down mode is the passive state of the satellite. During the launch phase with the command and housekeeping system turned on, there is a power drain of 19.65 watts.

2.6 BOOSTER TELEMETRY SYSTEMS

The Scout vehicle is equipped with a telemetry system capable of monitoring subsystems before launch, the flight performance of all four stages, tracking via a radar beacon, and commanding the destruction of the missile. The performance data of the missile is telemetered to earth by a standard IRIG PAM/FM/FM system handling 18 IRIG sub-carrier channels and broadcast on the 225 to 260 mc band.

The Scout vehicle has a radar beacon with a minimum peak R-F power output of 400 watts, single pulse. The beacon's antenna is an H-plane sectoral horn mounted externally on the lower "D" section of the missile. Radar tracking data are generally available up to ignition of the fourth stage at the WTR launch site.

2.7 REFERENCES

1. NASA TN-D 3411, "Telemetry Processing for NASA Scientific Satellites".
2. NASA TN-D 4507, "Space Trajectories and Errors in Time, Frequency, and Tracking Station Location".
3. NASA TN-D 3470, "Atmospheric Tracking Errors at S-Band Frequencies".
4. NASA TN-D 2311, "MRIR-PCM Telemetry System-microelectronics".

Table 2.4

Power Modes

Item	Power Required
Standard Mode	
Low power S-band transmitter	15.00 watts
VHF receiver	5.40
Memory and computer	2.00
Diplexer	.25
VHF transmitter	9.00
PCM network	1.00
Miscellaneous	2.00
TOTAL	34.65 watts
Shut-down Mode	
Memory and computer	2.00 watts
Decoder	.56
Diplexer	.25
Miscellaneous	2.00
TOTAL	4.81 watts
Hi Transmission Power Mode	
Hi power transmitter "S" band	40.00 watts
Memory and computer	2.00
Miscellaneous	2.00
TOTAL	44.00 watts
Launch Mode	
(Same equipment as in the standard mode except the S-band transmitter is off)	19.65 watts

5. STADAN Specifications, Published by NASA, 1967.
6. NASA SP 87, "Unified "S" Band".
7. NASA SP 5038, "Magnetic Tape Recording".
8. NASA TN-D 3988, "Application of the Stored-Program Computer to Small Scientific Spacecraft."
9. The SDP-3. "A Computer for Use on Board Small Scientific Spacecraft." by R. A. Cliff. Flight Data Systems Branch, Spacecraft Technology Division. NASA/GSFC. Reprinted from the EASCON '68 Conference.
10. NASA SP 154, "Aerospace Electronics System Technology".
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12. IRIG, "Telemetry Standards", (Revised March 1966).
13. Unpublished notes of Professor L. Bauer, Aero 490; currently Aero 485.
14. The Handbook of Telemetry and Remote Control, McGraw-Hill, 1967.
15. Telemetry Journal, Feb/March, 1968, pp 23-25.
16. Radio Telemetry, M. H. Nichols, and L. L. Rauch, John Wiley and Sons, N.Y. 1956.

3

ORBITAL ANALYSIS

3.1 ORBITAL DETERMINATION

3.1.1 Orbital Restrictions

The nature of the Project OBSERVER placed four major restrictions on our choice of orbits.

1. The orbit must be circular to avoid distortion of sensor data caused by a variation in V/h.
2. The altitude must be high enough for a lifetime of one year, yet not so high as to unnecessarily sacrifice sensor resolution or vehicle payload.
3. The orbital period must not be critically undesirable, i.e., critically undesirable orbital periods are those for which the ground tracks are the same every day. This causes repetitious and incomplete Earth coverage. Project OBSERVER'S orbital period should be compatible with an ideal ground track procession rate of 1.5°/day.
4. The orbital plane must be sun-synchronous.

3.1.2 Resultant Orbital Parameters

Calculations show that for the OBSERVER satellite the altitude required for a one year lifetime was any which was greater than 300 nm. For a sun-synchronous type orbit, the orbital period, upon which the ground track procession rate is solely dependent, is a function of the Keplerian radius only. Therefore, for a circular sun-synchronous orbit, the ground track procession rate is solely dependent upon the altitude. Corresponding to the prescribed ground track procession rate of 1.5°/day, the lowest altitude above 300 n.m. is 313.4 n.m. The period, τ, is 96.4 min. The V_{1c} is 24,770 ft/sec which corresponds to a $\dot{\phi}_e$ of 3.75°/min ($\dot{\phi}_e$ is the sweep rate = the angular rate of the s/c in degrees of latitude).

3.1.3 Orbital Inclination (Figure 3.1)

To achieve a sun-synchronous type orbit, there must be a secular nodal precession rate of $+ 0.986^\circ$/day, equal to the angular rotation rate of the Earth around the sun. The angle of inclination, i, measured from the ascending node, determines the precession rate. The desired $i = 97.6^\circ$.

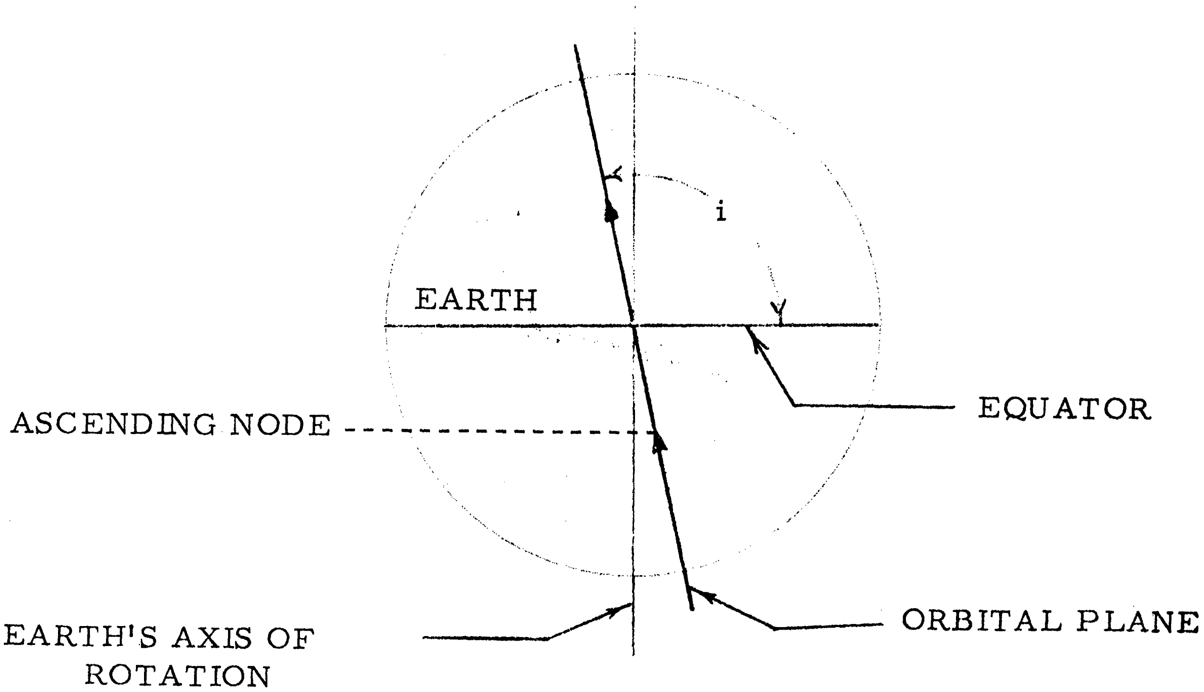

Figure 3.1 Inclination of Orbital Plane

3.1.4 Launch Window (Figure 3.2)

Sun-synchronism is determined by the angle measured between the satellite's orbital plane and the Earth-Sun line. When this angle is zero, the orbit is perfectly sun-synchronous. Usually an orbit is considered sun-synchronous if that angle is less than 7°. This gives rise to a 56 minute (14°) launch window. For perfect sun-synchronization injection into orbit should take place at noon. Since the time lapse between launch and injection into orbit is 585 seconds, the center of the launch window will be at noon -585 seconds. The resultant launch window lasts from 11:22 a.m. until 12:18 p.m.

3.1.5 Orbital Attainment

The circularity of the orbit and the exact altitude are very critical problems to the project. Since the Scout Launch Vehicle has injection errors which are intolerable to the mission, Project OBSERVER utilizes an apogee-perigee kick motor to circularize the orbit at the desired altitude.

Due to problems of stability, all impulses from the ΔV (kick) motor will be applied before despin. In the spun-up mode, the satellite is fixed in its orientation with respect to the Earth. This leads to the following method or orbital correction (Figure 3.3).

1. The satellite is injected at an altitude greater than 313.4 n.m. with an under-velocity small enough so that $h_p \leq 313.4$ n.m.
2. Apply a $+\Delta V$ at apogee to bring $h_p = 313.4$ n.m.
3. Apply $-\Delta V$ at perigee to bring $h_a = 313.4$ n.m.

An analysis of all probable errors of injection (Appendix C) shows that $\Delta V)_{max} = 383$ ft/sec.

If W = 200 lbs, M = 6.2 slugs, and $I_t)_{max} = 2380$ lbf-sec.

3.2 LAUNCH VEHICLE

3.2.1 Scout Capability

In keeping with the basic theme of Project OBSERVER, the or of a low cost, highly reliable satellite, the launch vehicle chosen is the Scout. This vehicle has been used to launch satellites of the same weight

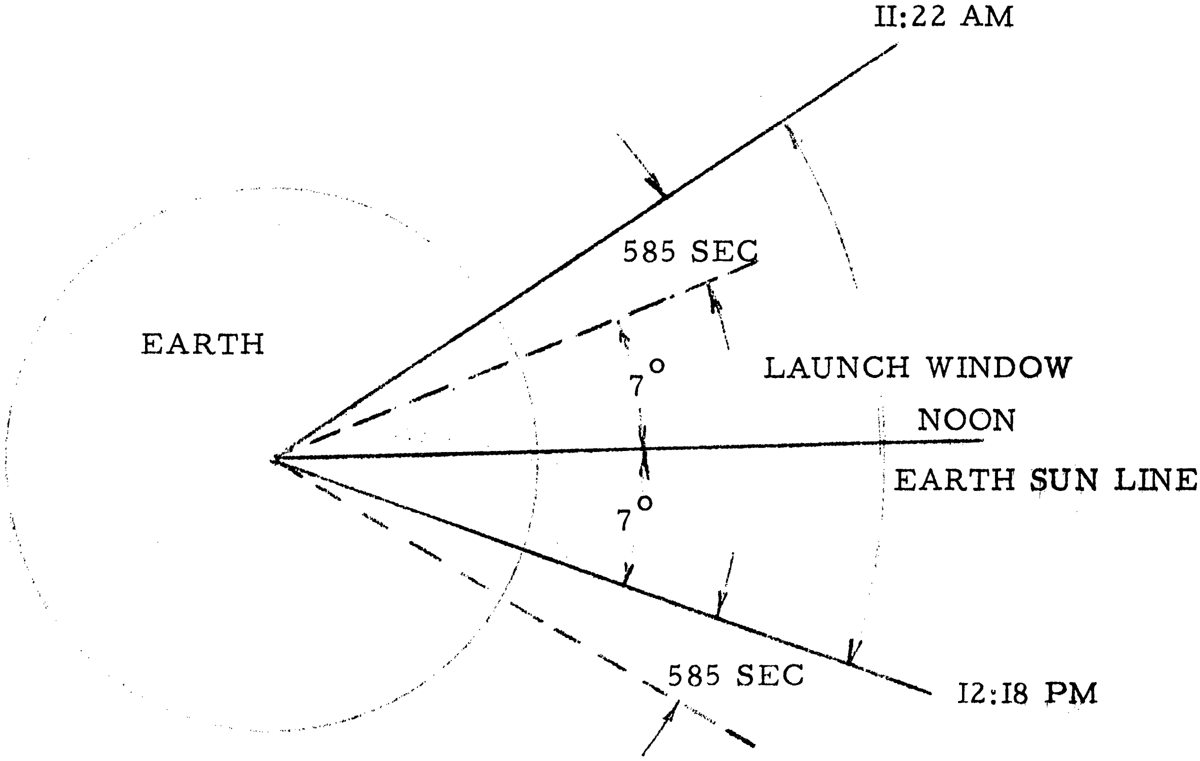

Figure 3.2 Launch Window for Sun-Synchronous Polar Orbit

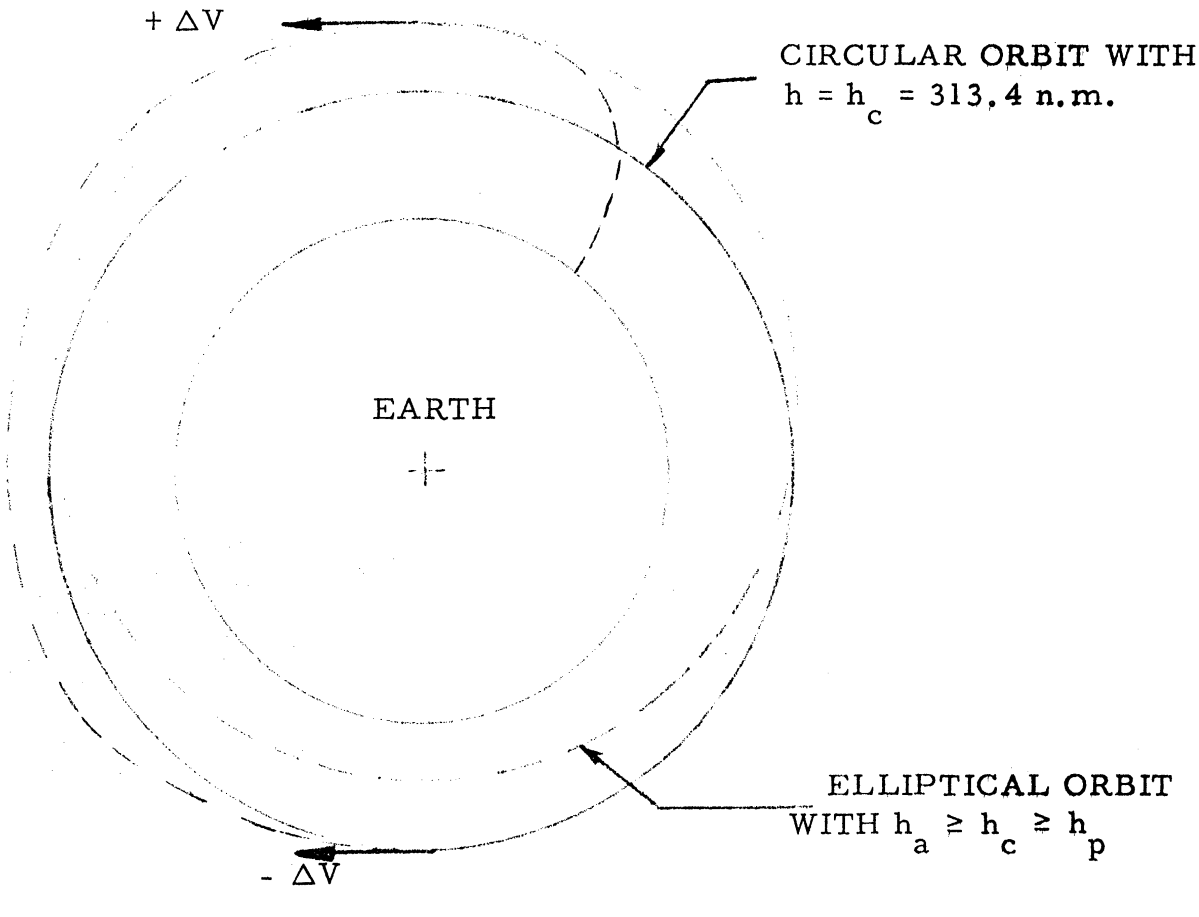

Figure 3.3 Orbital Correction

and size as OBSERVER, and it has injected them into orbits with altitudes close to that determined for Project OBSERVER. Significant details for the vehicle-payload interface may be found in the structural portion of Project OBSERVER.

The Scout vehicle is a four stage rocket with all four stages being powered by solid fuel rocket motors. The Scout vehicle is capable of injecting a 245-pound satellite into an orbit with an altitude of 313.4 nautical miles. It is a relatively inexpensive launching system with a basic system cost of 1.5 million dollars. Included in this basic cost are the launch facilities and the payload interfacing service. The success record of Scout vehicle launch system is over 90%

3.2.2 Guidance

The Scout vehicle uses a guidance system which is programmed on the ground. Once in flight the guidance system compares its programmed trajectory to the inputs it receives from gyros and accelerometers located in the guidance section. These comparisons give the guidance system the deviations of the vehicle from its intended trajectory, and controlled forces are applied via altitude control rockets in the yaw, pitch and roll axis. During the burning of the first stage motor, the control is by jet vanes inserted into the exhaust gases and aerodynamic fins at the base of the first stage. This system can give an accuracy of less than one degree in insertion of our satellite into the correct orbital plane. The guidance system controls the vehicle through the firing of the first three stages. It remains attached to the fourth stage until it is aligned to the proper trajectory and has been stabilized by spinning it up. Separation of the third and fourth stages occur, and the fourth stage flies the last portion of the trajectory unguided.

3.2.3 Space Craft Separation

Once orbital height and speed have been attained, the payload is separated from the vehicle by the "E" section and adaptor ring that the Scout vehicle has incorporated into its fourth stage. The section contains batteries, timers, separation springs, and spring retainer ring. The adaptor ring remains with the payload after separation. A marmon clamp firmly attaches the adaptor ring to the fourth stage structure. The separation springs detach the satellite from the fourth stage and impart a small velocity to it so that the spacecraft and the burned-out fourth stage do not collide.

The layout of the Scout vehicle can be seen in Figure (3.4).

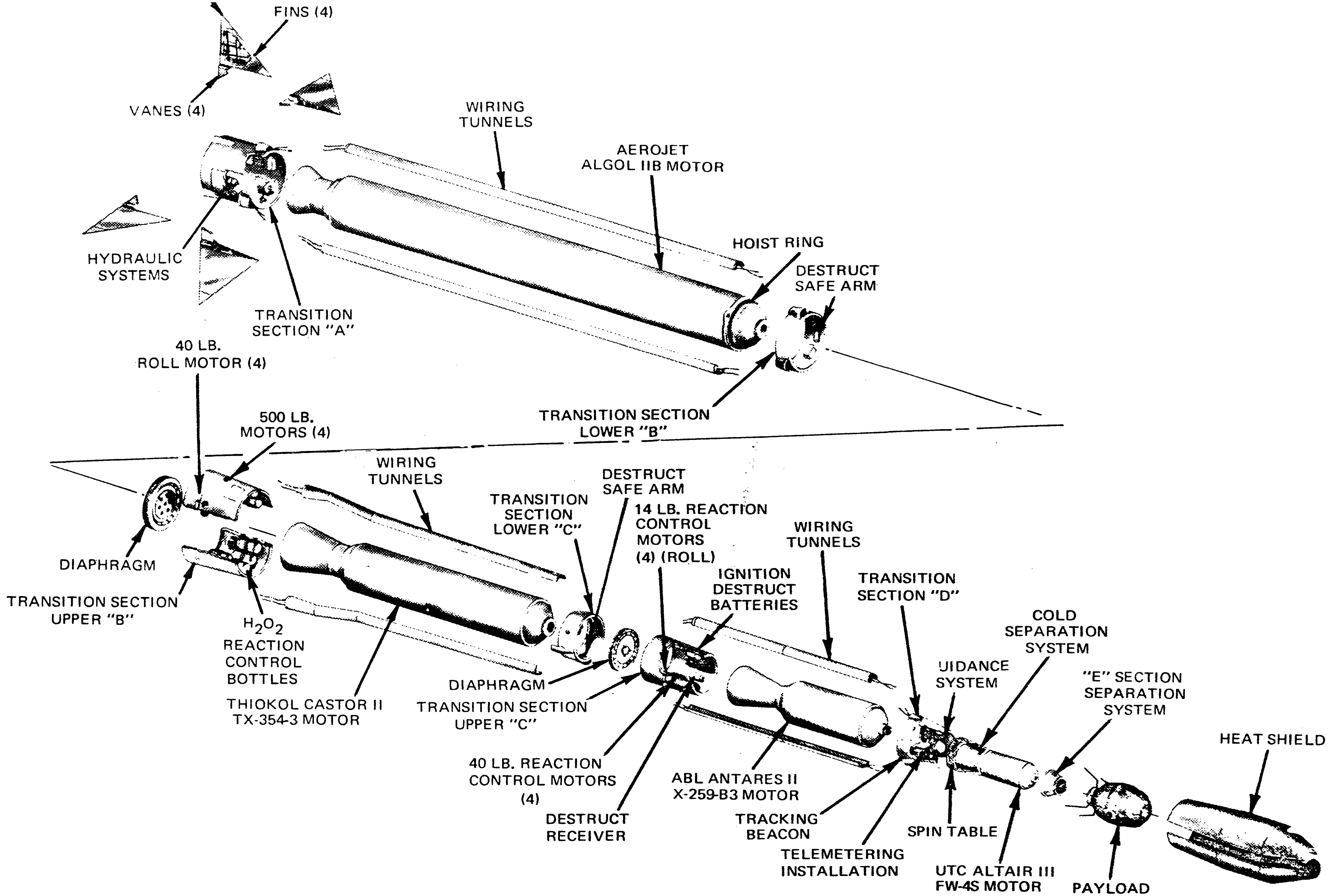

Figure 3.4 Exploded View of LTV's Scout Launch Vehicle

3.3 ΔV MOTOR

The type of sensor that will be used by Project OBSERVER requires that the satellite's orbit should be very nearly circular. The Scout launch vehicle's probable errors in launching will not insert the payload into an orbit close enough to the orbital parameters needed to be tolerable, so some capability for orbital correction must be carried by Project OBSERVER. The small rocket motor on board will have the job of insuring the correct orbit within the tolerances allowed.

The motor needed had many design requirements to fulfill. It had to be light, small enough to be easily integrated into the structural layout of the satellite, and be highly reliable. Among the first considerations was the type of rocket best suited for our application. Solid fuel rockets, it was discovered, are very hard to control in the duration time of thrust. This led to the consideration of liquid propellant motors. This type of motor can be easily controlled, but there are problems with fuel tanks, piping, and valves. The controllability of the motor was the important parameter that had to be considered. This was the factor that determined the choice of a liquid rocket motor.

The motor selected is an attitude control rocket that is used on the Scout launch vehicle. This unit has been flown many times on several different types of missions with a high degree of reliability.

The motor specifications are as follows:

Motor weight	2.0 lb
Propellant weight	16.15 lb
I_{sp}	130.0 (lbm/lb) sec
Total impulse	2020 lb-sec
Burn time	146 sec
Thrust	14 lb
Fuel	H_2O_2 (hydrogen peroxide)
Tank Size	9.4 in (diameter)
Power to valves	19-28 watts
System total weight	30 lb

This motor is a monopropellant unit that uses a catalyst to increase performance. It uses an electrically operated valve to control the flow of hydrogen peroxide into the combustion chamber. The fuel system consists of a fuel tank and a pressure tank that contains nitrogen gas at 5000 psi. The nitrogen gas is used to force the propellant hydrogen peroxide into the combustion chamber. The propellant tank has a Teflon bladder inside so that the hydrogen peroxide will be kept at the drain end of the tank. This insures the motor of the fuel flow it needs once the fuel flow valve is opened. This valve can be operated either by direct uplink signal from the ground or by a signal from the memory of the on board computer.

The propellant tank and the pressure tank are filled shortly before launch and the system is sealed by valves at the aft end of the satellite. The propellant tank is to be constructed of S-2 aluminum. This type of aluminum has proved to be resistant to the corrosive action of H_2O_2, and it is fairly light. It is a very good possibility that all the fuel in the system will not be used, so the tank has to be fabricated from a material that is resistant to hydrogen peroxide for long periods of time. S-2 aluminum more than fills this requirement.

Figure 3.5 shows a schematic diagram of the ΔV motor.

The specifications were arrived at in the following way. From the orbital correction, the total I = 2380 lb-sec for a 2σ deviation from the intended orbit;

now,

$$I_t = 2380 \text{ lb-sec}$$

$$\text{Motor } I_{sp} = 130 \text{ (lbm/lb) sec}$$

$$\text{Wt.} = \frac{I_t}{I_{sp}} = \frac{2380}{130} = 18.30 \text{ lbm}$$

$$\rho\, H_2O_2 = 90 \text{ lbm/ft}^3$$

$$\text{Volume} = \frac{18.30 \text{ lbm}}{90 \text{ lbm/ft}^3} = .203 \text{ ft}^3$$

$$V = .203 \text{ ft}^3 \times 1728 \text{ in}^3/\text{ft}^3 = 315.5 \text{ in}^3$$

Area Sphere $= 4/3\pi r^3$

solving for r, we have

$$r = 4.225 \text{ in}$$

$$\rho N_2 = .0704 \text{ lbm/ft}^3$$

$$\text{Weight } N_2 = .0704 \text{ lbm/ft}^3 \times .1795 \text{ ft}^3$$

$$= .127 \text{ lbm}$$

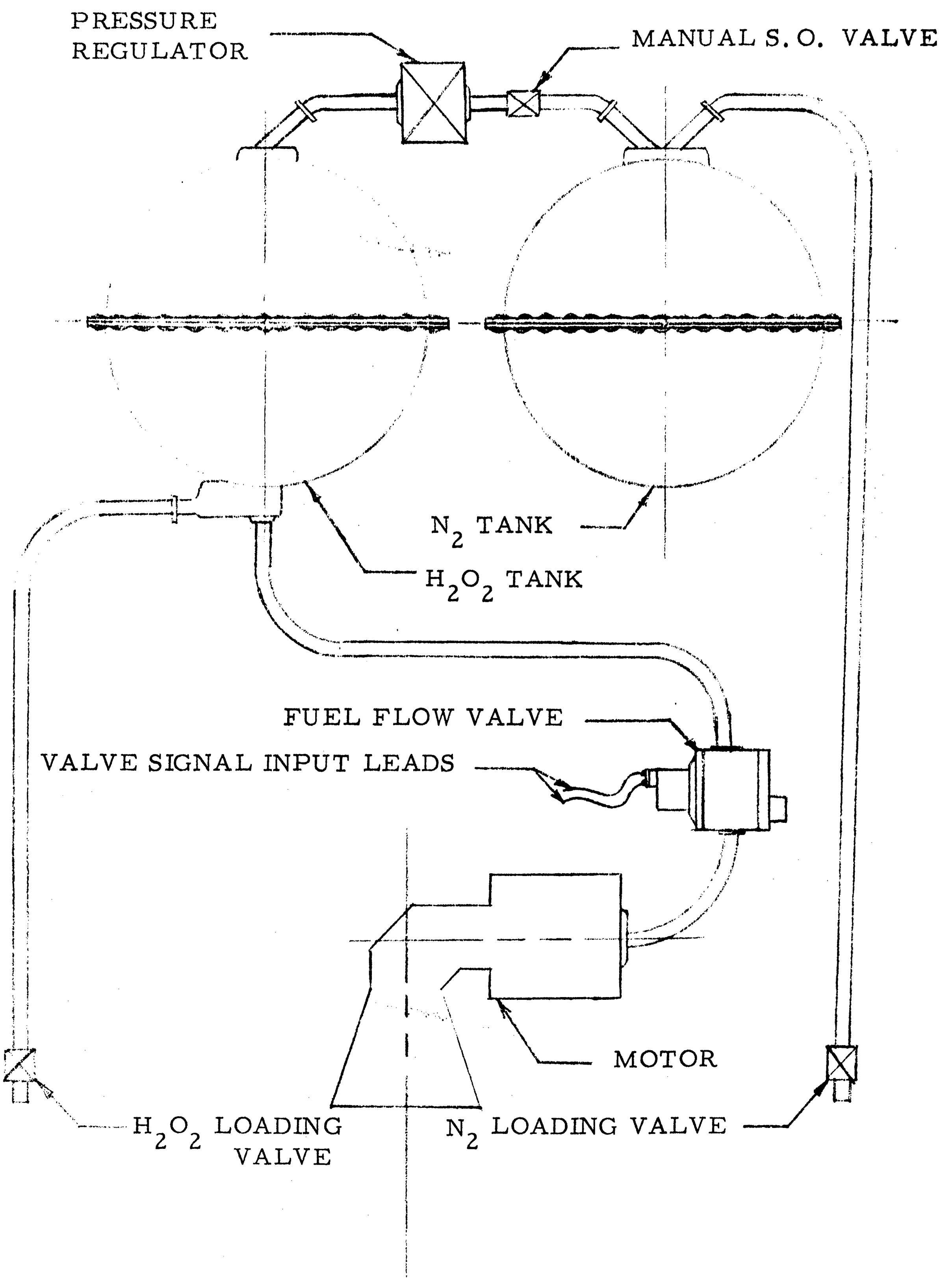

Figure 3.5 Schematic of the ΔV Motor

Total system weight:

Motor	2.0 lb
H_2O_2	18.30 lb
N_2	.130 lb
Valves and Piping	5.0 lb
Tanks	6.0 lb
Total	31.43 lb

3.4 LAUNCH SITE

San Marco, Wallops Island and the Western Test Range are three choices of launch sites for Project OBSERVER; the San Marco offshore launch site, located just off the coast of Kenya is primarily intended for satellite launches into an equatorial orbit. The Wallops Island site, located on the east coast of the United States, would not be acceptable to our mission because a dogleg tragectory must be flown to attain a polar orbit. This essentially is a right turn that must be performed by the launch vehicle. This turn in the ascent trajectory limits the weight of the satellite that can be injected into a polar orbit with an altitude of 313.4 nautical miles. This weight limit is not acceptable to Project OBSERVER.

The Western Test Range, located at Point Arguello, California, is designed primarily for polar launches. Trajectories flown from this site have no dogleg requirement for insertion into a polar orbit. For this reason, the lifting capacity of the Scout is at a maximum for a polar launch and this weight limit is comfortably above the maximum weight of Project OBSERVER. For these reasons, the Western Test Range has been selected as the launch site for this project. Figure 3.6 details the expended impact areas for the WTR.

3.5 REFERENCES

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2. Bollinger, Loren E., Martin Goldsmith, and Alexis W. Lemmon, Jr., Liquid Rockets and Propellants, New York, Academic Press, 1960.
3. Kit, Boris and Douglas S. Evered. Rocket Propellant Handbook, New York, MacMillan Company, 1960.
4. Scout Users Manual, Dallas Ling Temco Vought, 1968.

WESTERN TEST RANGE LAUNCH

ENCOMPASSES EXPENDED STAGE IMPACT AREAS FOR CIRCULAR ORBITS BETWEEN 300 AND 700 N. M. ALTITUDE

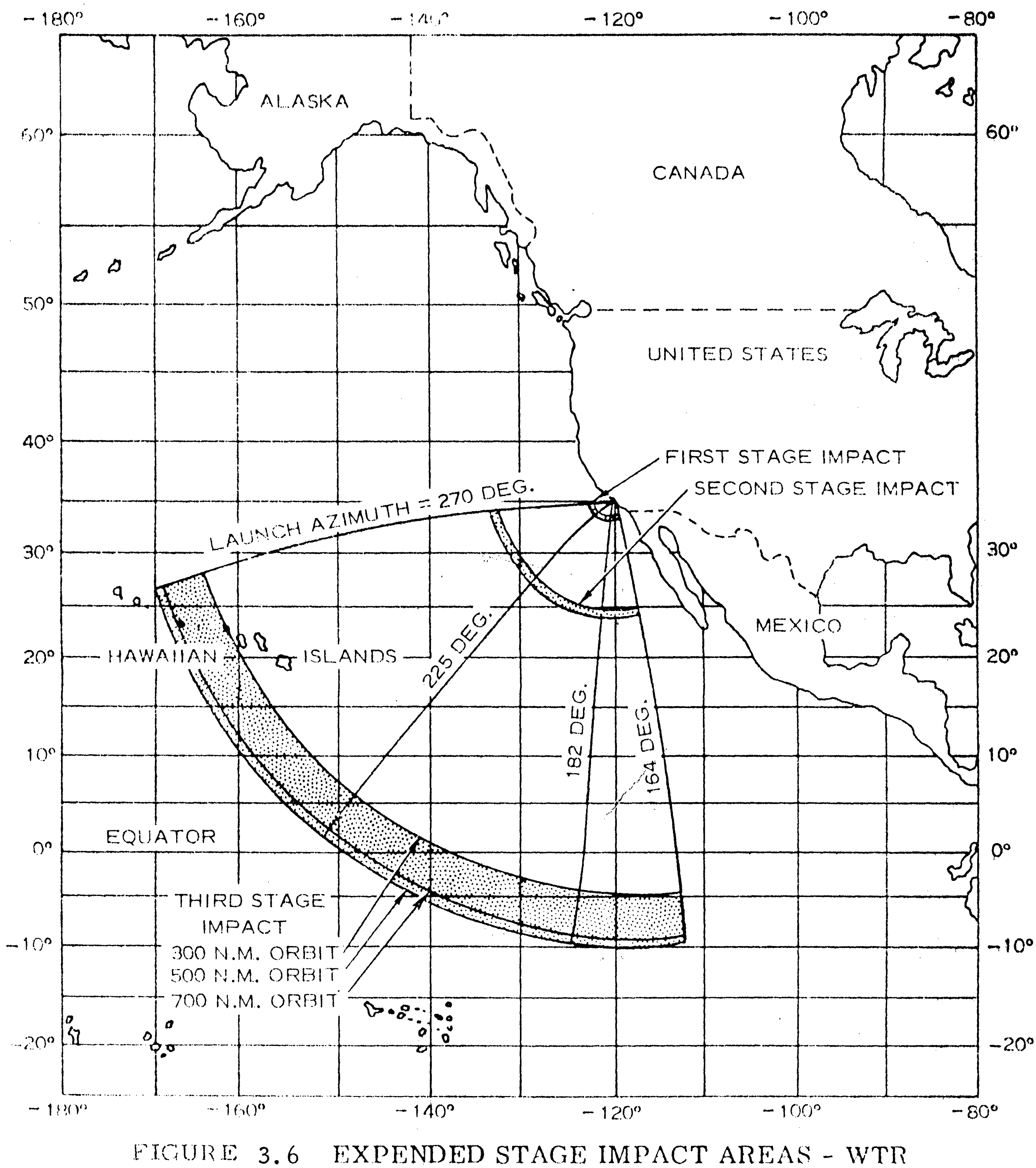

FIGURE 3.6 EXPENDED STAGE IMPACT AREAS - WTR

4

STABILITY AND REFERENCE SENSORS

4.1 GRAVITY GRADIENT STABILITY SYSTEM

4.1.1 System Considerations and Selection

In deciding on a stability system for OBSERVER, several criteria had to be considered and met: (1) It must point the yaw axis of the satellite toward the earth, (2) It must be accurate within ± 2-3 degrees variation for all three orthogonal axes, (3) It should be relatively inexpensive.

Two basic stability systems are available at the present time: active systems, such as gas jets and momentum wheels, which require monitoring and periodic operation; and passive systems, like the gravity gradient system, which once employed, do not require power from the satellite for operation.

The gravity gradient system was chosen over either the active gas jets, or momentum wheels, or magnetic stability systems, because it is inexpensive, light in weight, highly reliable, independent of power once deployed, and it satisfies the accuracy requirement of ± 2-3 degrees set up by the sensor system. The only important question arising concerning this system is the accuracy in the yaw direction. With research being carried out at present and new features being incorporated in the system at present, it is expected that the yaw accuracy will be within the required limits.

The gravity gradient system is neutrally stable, so some form of 3-axis damping must be provided for use with it. The eddy current damper was chosen over other systems, like the spring damper or the viscous fluid damper, because it is passive, inexpensive, highly reliable, and it has been used exclusively with the gravity gradient system. The damper will be designed with a time constant such that the steady state will be reached with 6-8 orbits.

4.1.2 System Configuration

OBSERVER'S gravity gradient system uses a tri-boom, "Y" configuration as shown in Figure 4.1 . The booms are swept back 15 degrees which, along with the "Y" configuration, optimizes the coupled restoring torque in the yaw direction. A tip weight at the end of each of the two lower booms provides the mass needed for the gravitational and centrifugal forces, and these forces along with the

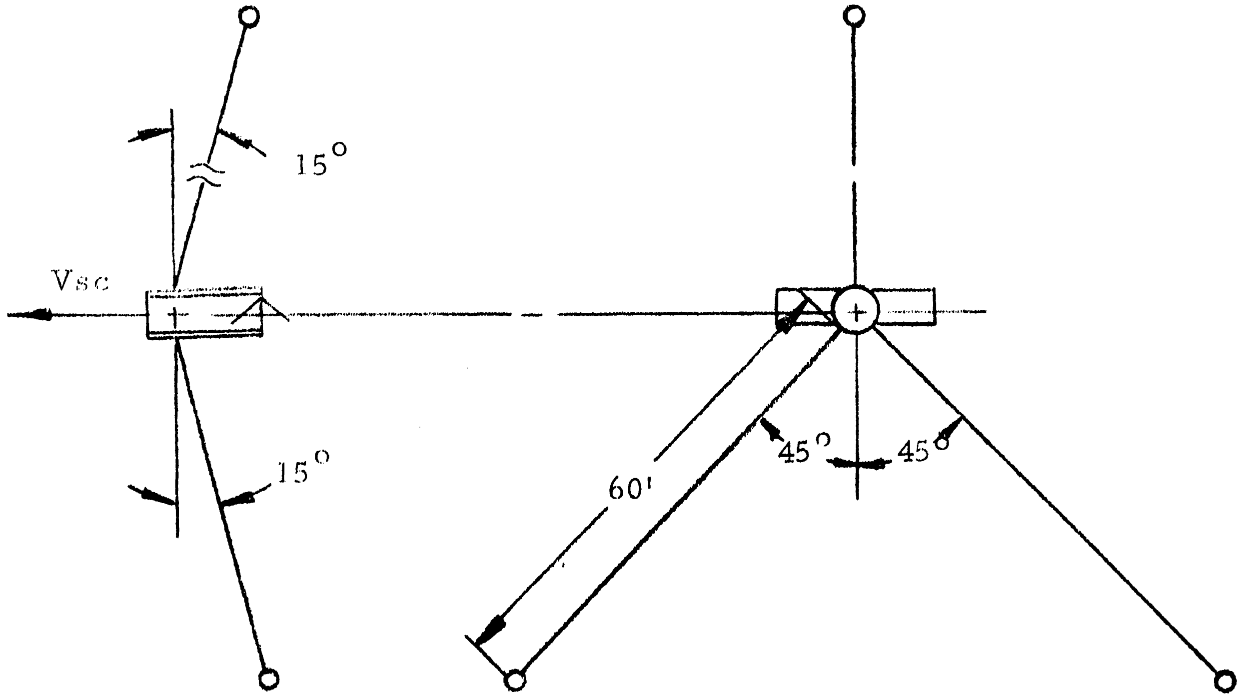

Figure 4.1 Boom Configuration

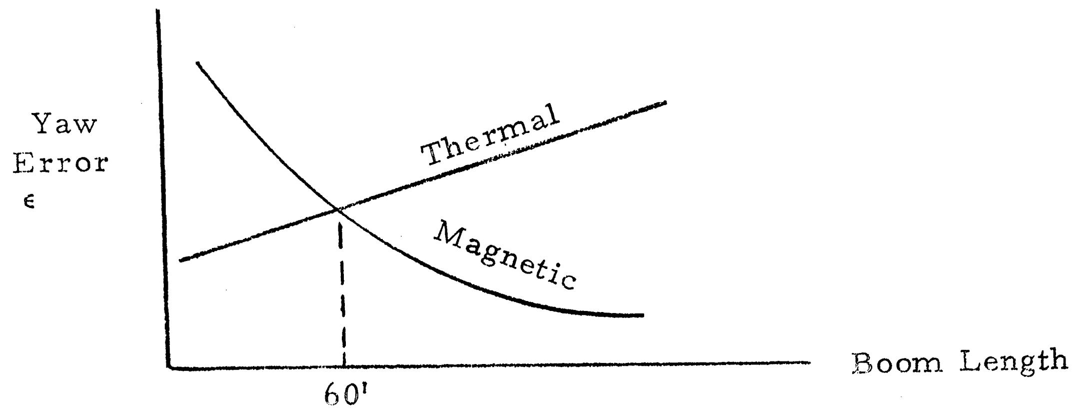

Figure 4.2 Optimum Boom Length

boom length provide the restoring torques. The upper boom is topped by a General Electric eddy current damper, which also acts as the tip mass. The mechanism for the damper may be found in Appendix D.

The booms, which are manufactured by three companies (General Electric, Westinghouse, and deHavilland of Canada), are beryllium-copper strips 2" wide which are rolled on a spool for storage. When the strip is deployed by a motor, it rolls into a 1/2" diameter tube due to its previous heat treating. The Westinghouse boom system was chosen for OBSERVER because of its improvements in structure, such as the interlocing edges, which greatly improve its accuracy.

At OBSERVER'S altitude of 313 n.m., there are several natural phenomenon which cause disturbing torques, and these must be eliminated or compensated for. The atmospheric pressure at this altitude is about $5/10^{-6}$ lb/ft^2. This gives a corresponding aerodynamic drag which is negligible. Solar radiation and pressure are significant, however, and compensation is made by plating the exterior surface of the booms with reflective silver, the interior with an absorptive material, and punching a spiral helix perforation pattern along the booms. This effectively cuts the temperature gradient across the tape thickness from 6 to 0.6 degrees F, which reduces greatly the deflection of the booms due to thermal bending. This deflection is known as "Thermal Twang" and occurs when the satellite enters the sunlight from the earth's shadow. Another problem has arisen in the past due to twisting of the booms. This torsional problem has been eliminated by Westinghouse by a new series of V-notches along both edges of the tape which interlock as it is deployed. This approximates a closed tube, and it virtually eliminates any twisting due to relative movement between the edges. Each boom is deployed at a rate of 1.5 to 2 inches per second by the 1.5 pound motor-guide mechanism, and this minimizes errors due to the dynamic bending of the booms upon erection. The apogee kick motor has virtually eliminated any torques due to orbit eccentricity, since the orbit is very nearly circular.

The pointing error due to thermal and bending torques is dependent on the boom length, with a linear relationship for thermal and an inverse, second order relationship for the magnetic. An optimum point, a boom length of 60 feet, has been found in various tests including a General Electric Applications Technology Satellite (see Figure 4.2).

Complete optimization of the stability and damping systems requires employment of a 3 axis computer program, like that of General Electric, which is prohibited by cost, time, and lack of exact inputs on moments of inertia and other satellite parameters for all but the final iteration.

4.1.3 Inverted Satellite at Injection

In the case of an inverted satellite at orbital injection, which can be sensed by the attitude sensors, one gravity gradient boom will be retracted to change the inertia enough to rotate the satellite 180^{o} where it can be recaptured in the correct orientation.

4.1.4 References

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2. Wheeler, P. C., et al, Evaluation of a Semi-active Gravity Gradient System, Vols. I and II, TRW Systems, Redondo Beach, Calif.
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4. Rushing, F. C., et al, "'Unbendable' Booms for Gravity Gradient Systems", Space/Aeronautics, August 1968, pp. 76-7.
5. Keller, Jack L., "Satellite Disturbance Torques", SER, Fall 1965. pp. 11-18.

4.2 YO-YO DESPIN

4.2.1 Theory

To remove the spin from the OBSERVER satellite, the yo-yo despin technique is used. The despin is accomplished by the extension of two diametrically opposed weights connected by wire cables to the main body of the spacecraft which transfers the initial angular momentum from the payload to the end weights. The despin system reduces the initial spin rate of 150 rpm to 0 rpm by releasing two half-pound weights that spin out from the satellite. The cable is 132 inches long, allowing the weights to wrap around the spacecraft one and one-half times. Protection for the solar cells is provided by the despin ring, an aluminum U channel ring, that prevents the cable from damaging the solar panels prior to their deployment which is then jettisoned. Initial spin-up of the satellite is required to maintain orbit injection attitude for the subsequent circularization of the orbit by the Delta-V motor. Since the OBSERVER satellite requires a non-spinning, earth-oriented platform to carry out its Earth Resources Study, a despin system must be chosen.

4.2.1 Types of Despin Systems

Many systems are available for despin, some passive, some active. Examples of the latter are the cold gas and hot gas retro devices, which will provide accurate despin to any desired level provided that the initial spin rate is known and the spin moment of inertia can be determined accurately. The despin thrusters are complex, requiring sophisticated control systems and incurring a large weight penalty due to the on board fuel carried.

An example of the passive system is the yo-yo despin. There are two types, one, the stretch yo-yo, used for accurate despins to a specified final spin rate. This system incorporates a spring with the cable to compensate for initial spin errors. But if the despin system is designed to remove all the angular velocity, the cable length and end weight will be independent of initial spin rate. This would allow the second and less complicated type, the rigid yo-yo system, to be used. The rigid yo-yo consists of a cable and end weight.

4.2.3 Design of the System

During despin, the satellite is shrouded by solar panels, leaving a very limited amount of space on the satellite body for attachment and control of the despin mechanism. Consequently the design starting point was chosen as a cable length that would allow optimal use of the space available on the satellite body.

If the cable wraps around the satellite one and one-half times, the yo-yo release mechanism and yo-yo anchor system will coincide. Once the cable length has been specified, the corresponding end weights may be determined. The shorter the length of cable chosen the larger the end weights will be and the faster the system will despin. Care must be taken to prevent the angular de-acceleration from exceeding the structural g-limit for the satellite and to prevent the cable from parting.

A 1/8" thick cable is used in the design, the weight and strength being well within the design limits for the yo-yo despin system. The entire system is to be placed at the center of gravity of the satellite. A pyrotechnic thruster releases the tip weight by kicking the hold down clamp free of the anchor point. Prior to solar panel deployment the despin ring is jettisoned by two guillotiene actuators that sever the thin double wrap of wire that retains the spring loaded ring. Figure 4.3 details the release mechanisms for the end weights and ring.

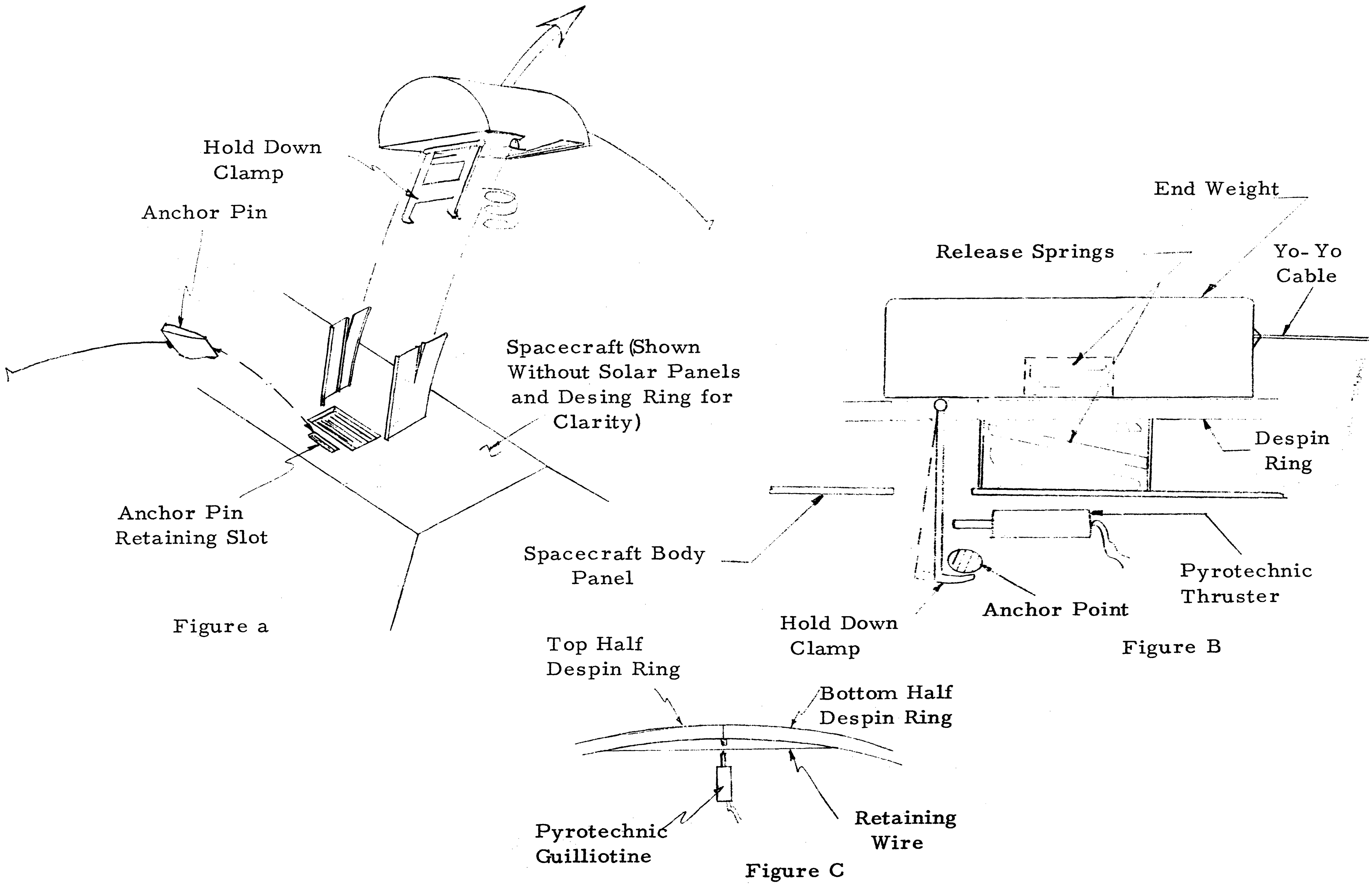

Figure 4.3 Yo-Yo Despin Release Mechanisms

4.2.4 System Details

The total weight per yo-yo is .6099 pounds, which includes 1/3 of the cable weight. The cable is 1/8 inch thick steel cable that wraps one and one-half times around the circumference of the satellite. To protect the solar panels, which cover the entire satellite prior to deployment, a U-channel ring is provided. This requires a section of the solar panels to be cut away at their apex, allowing the ring and yo-yos to wrap around. The U-channel is of aluminum, and is actually two C-ring devices, spring-loaded and held together by a double wrap of thin wire. Refer to Figure 4.4.

The end, or tip weights, are solid steel half cylinder with hollow center in which the release spring will fit. They are released by a pyro-technic thruster actuator that kicks the release hinge free of the retaining slot. Each tip weight is spring-loaded to bias the release 1/2 degree off vertical to insure that the cable or weights will not foul each other during deployment.

The cables are attached to the satellite with a slot device that will allow the yo-yos to release when they are fully deployed. The thin wire that retains the despin ring is severed by a pyro-technic guillotine, and the ring is released. To guide the tip weight in the initial deployment, a short curved guide rail is provided, the curve matching the initial trajectory of the weight and preventing the weight from fouling in the panel area. A vertical guide rail forces the despin ring to separate smoothly, without twist or tumble. Refer to Appendix D for sample calculations on the despin system and the model for the moment of inertia of the satellite.

Design parameters:

I = spin moment of inertia 4.0×10^2 slug-in^2
L = each cable length 131.95 inches
W_e = end weight .5690 pounds
W_c = cable weight .0411 pounds
r_i = initial spin 150 rpm or 15.7 rad/sec
r_f = final spin 0 rpm

end weight dimensions half cylinder; length 2 in.
radius .7995 in. with a hollow for release spring
total volume: 2.01 cu. in. steel density; .283 lb/in^3
cable: 1/8 inch steel cable, .00031 lbs/in
131.95 in. length, total weight per cable is
.0411 pounds.

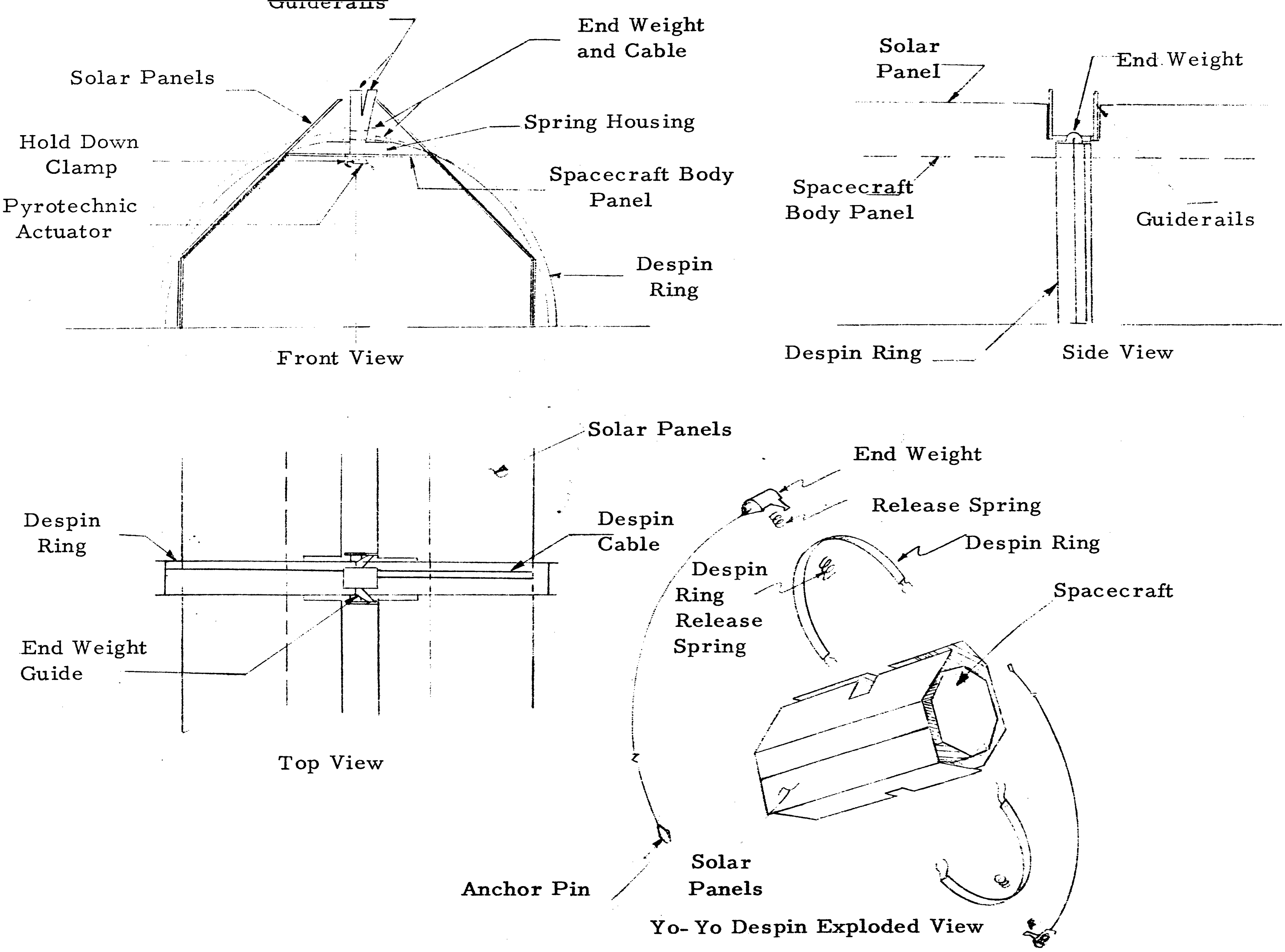

Figure 4.4 Yo-Yo Despin Details

Weight budget:

Yo-yo weight	1.140 lbs
Cable	.082 lbs
Springs	.50 lbs
Despin ring	3.0 lbs
Thruster actuators	.10 lbs
Total	4.822 lbs

4.2.5 References

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2. Eide, Donald G., and Vaughn, Chester A., "Equation of Motion and Design Criteria for the Despin of a Vehicle by the Radial Release of the Weights and Cables of Finite Mass," Langley Research Center, NASA TN D-1012, Jan. 1962.
3. Fedor, Joseph, "Analytical Theory of the Stretch Yo-Yo for Despin of Satellites," Goddard Spaceflight Center, NASA TN D-1676, April, 1963.

4.3 ATTITUDE SENSING

4.3.1 Sensor Selection

Because of the need to know the exact direction the earth resources sensor is pointing at any one time, an attitude sensing system was developed. In order to accomplish this attitude sensing, a combination of sun sensors and magnetometers is used. The system which finally evolved was chosen from several alternatives.

Horizon sensors were considered in the early study. However, due to drawbacks of horizon scanners, they were discounted. Horizon sensors are still in the research and development stage, and are not sensitive enough. In order to gain the accuracy needed for attitude sensing, a thermister balometer would have to be used. It must be cooled to 77^{o}K, and the colling aspect is critical since it is paid for with added weight. If the balometer is used at higher temperatures, the accuracy decreases exponentially. The horizon scanners work on the principle of sensing the difference between the earth's radiation and the radiation of outer space. Also, for horizon sensors it is necessary for the vertical axis to be fixed. The sensors are also altitude dependent for their accuracies and they are sensitive to changes in earth-space radiation. They are sensitive to changes in the shape of the area of the earth under them due to the equatorial bulge. Cold cloud inaccuracies can cause errors on the order of 20 degrees. For these and many more reasons sun sensors were adopted.

4.3.2 Sun Sensors

The orientation of the five sun sensors is shown in Figure 4.5 The solar sensors consist of silicon solar cells.

The center sensor consists of four "rough" sensors which are placed on the outsides of the cube and a more sensitive sensor in its interior. Each of these "rough" sensors consist of a silicon cell covered by an artificial sapphire glass cover. Between the cell and the glass is a liquid to prevent cell damage in the event of cover fracture by micrometeoroids. The cell will still be 80% efficient if fractured. Each cell operates on the principle that changing angles of the sun's rays produce varied degrees of light intensity. Figure 4.6 indicates the geometry involved.

$$\Delta I = I_{max} \left[\cos(\alpha - \theta) - \cos(\alpha + \theta) \right]$$

where

I = the maximum light intensity

ΔI = the difference in intensity on the two photovoltaic cells

and the angles are as given in Figure 4.6 . The more sensitive sensor is activated whenever the sun comes into its field of view. This is accomplished by the trigger mechanism, which is a photocell that saturates when the light intensity reaches a certain level. When the trigger saturates it acts as a switch which turns on the more sensitive sensor.

The side sensors are similar to the center one. However, they don't have rough sensors on their sides. They can be completely buried inside the satellite structure. The trigger mechanism is similar to the center sensor trigger mechanism and is used as an on-off switch for the more sensitive sensor in the side sensors.

The sun sensors have two operating ranges. The "rough" range occurs when the sun is relatively low on the local horizon and as the satellite passes between the earth and sun the more sensitive sun sensors are activated. Depending on the sun angle during the orbit, different sensors are in operation. A nadir angle is the angle between the satellite position measured from the center of the earth and the zenith direction. At high nadir angles near the poles the side sensors

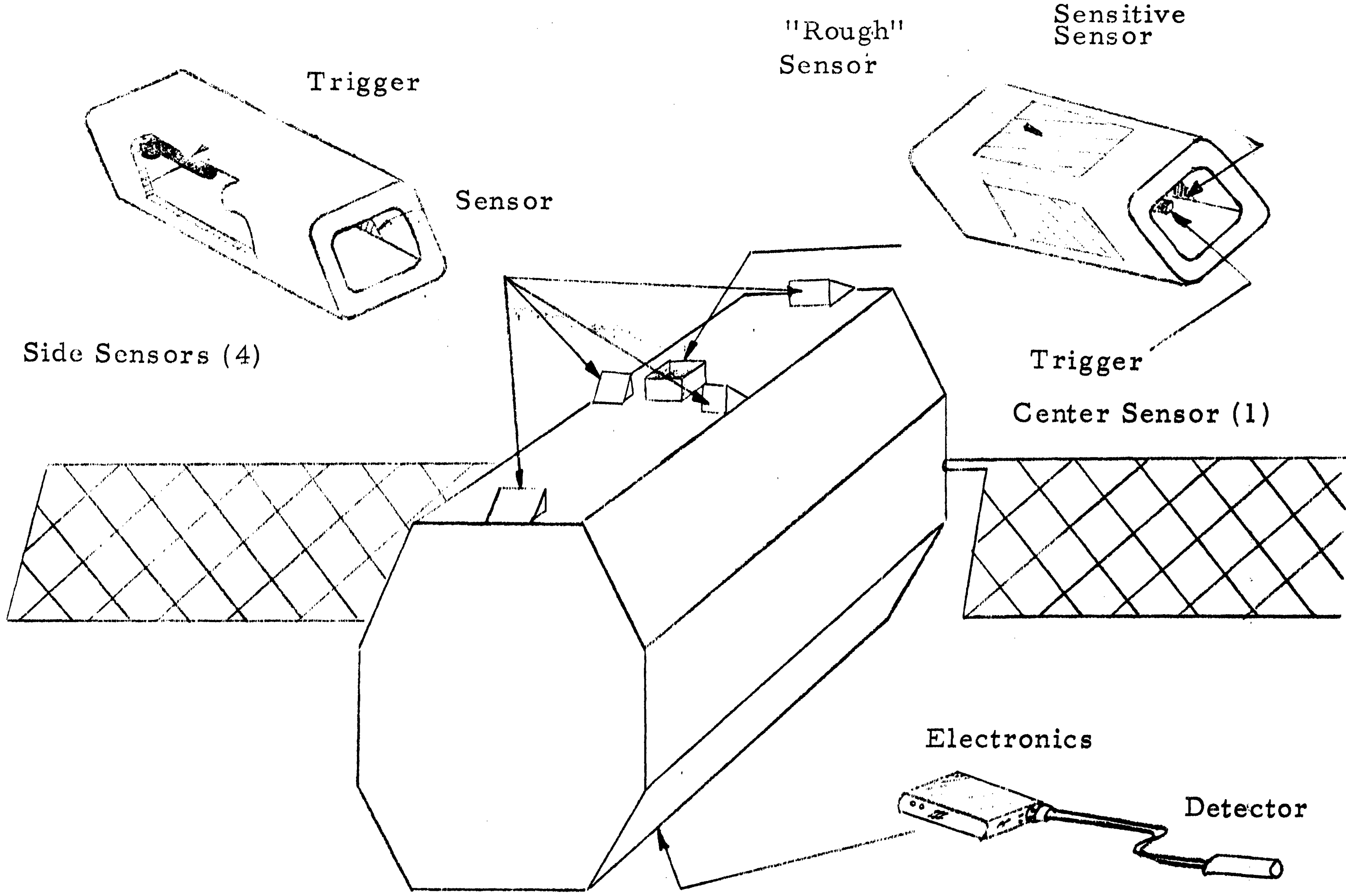

Figure 4.5 Sensor Orientation

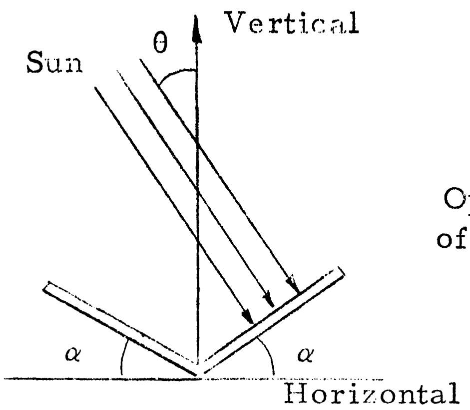

Figure 4.6 Incident Ray Geometry on Photovoltaic Cell

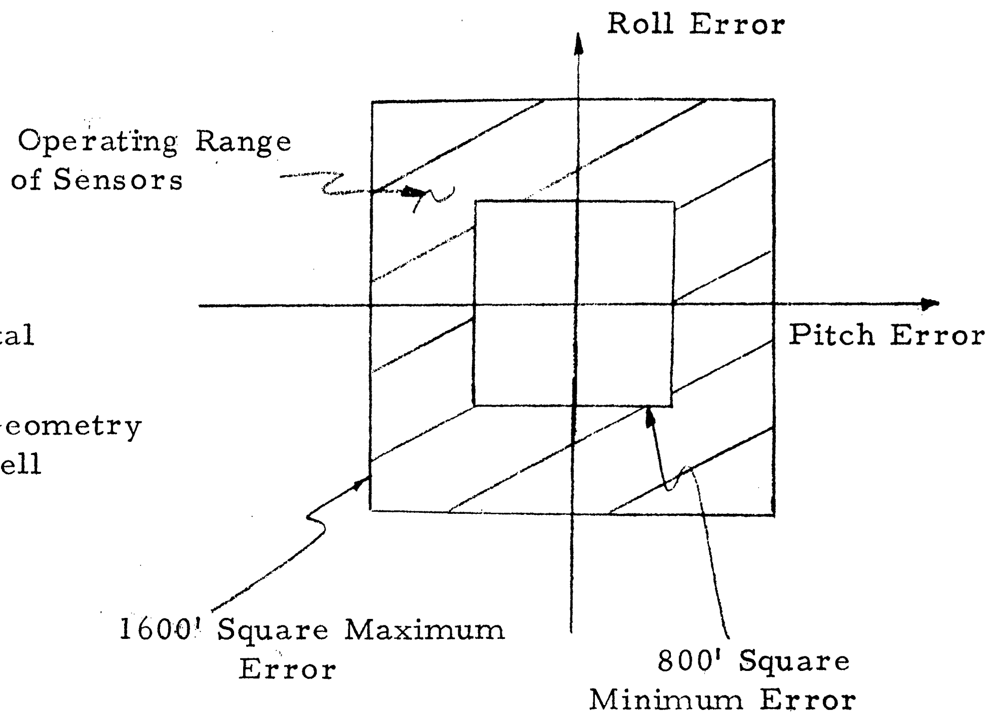

Figure 4.7 Attitude Error Limits for Roll and Pitch

are operative. Near the equator the side sensors are switched to the center sensor. This occurs at a nadir angle of 25^{o}, although there can be some system overlap. At any one time one or two sun sensors and three magnetic sensors can be read out.

4.3.3 Magnetometers

The magnetometers will be used only for yaw orientation determination. The pitch and roll components will be taken by the sun sensors. The Ram-3 Heliflux Magnetic Aspect Sensors, made by the Schonstedt Instrument Company, are placed along the three primary axes of our spacecraft. The magnetic vector can be determined by the three magnetometers and compared with the calculated magnetic vector at that location in space. The difference is the orientation error of the spacecraft with respect to the magnetic vector. For this reason there must be constant tracking of the probe and access to a computer program similar to the one by the Jet Propulsion Laboratory. The computer system compares the magnetic vector at the actual position of the satellite to the magnetic vector detected by the magnetometers. The difference in the two values is the attitude error. The yaw error can be determined to less than 1 degree by this method (see Appendix D for specifications).

4.3.4 Reference Error

The total attitude sensing system gives a yaw error of less than 1 degree and an uncertainty on the ground in pitch and roll of $\pm$ 800 ft. This error is still small enough to give a good reference point to the picture data. The reference ground error is shown in Figure 4.7.

4.3.5 References

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3. Fontana, Anthony, "Flight Investigation of a Solar Orientation Control System for Spacecraft", NASA TMX-944.
4. Fontana, Anthony, "A Photovoltaic Solar Sensor for Use in Spacecraft Orientation Control Systems", NASA TND-3279.
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5

POWER

5.1 INTRODUCTION

Three forms of primary power sources were considered for OBSERVER:

1. Radioisotope thermoelectric generator (RTG)
2. Fuel cells
3. Solar cells

The basic intent of low cost, weight, one year lifetime and high reliability determined the power system. RTG's are expensive and have a low power to weight ratio. Fuel cells, at the present time, provide too low a power to weight ratio and create a storage problem for space applications. Thus, these two ideas were rejected. Solar cells were chosen since they are relatively inexpensive, provide a high power to weight ratio and have been proven reliable. Solar paddles were utilized rather than body-mounted cells. Due to three axis stabilization and the form of the satellite body, body mounted cell arrays resulted in insufficient power output and inefficient use of solar cells. For these reasons, they were also rejected as a supplementary source. A battery will supply the secondary power.

A nickel-cadmium battery will supply the power needed until the satellite is in its operating mode and during eclipse. It will also provide supplemental power at the very beginning and very end of each scanning period while in orbit. The battery size is determined by the initial orbits prior to the operating mode, which is more than adequate for the eclipse requirements.

In the design of the power system, oriented paddles were considered. To achieve a constant normal incidence angle of sunlight to the paddles (thus utilizing a minimum of area), a drive motor would have had to have been incorporated. This motor could fail or the bearings could freeze thus causing a mission abort. The probability of this type of failure occuring within the one year lifetime is extremely high. Therefore, oriented paddles were rejected.

5.2 SOLAR PANEL ARRAY

The final configuration was designed about the major requirement to produce the maximum power of 61 watts continuously between the regions of 80 degrees North and 80 degrees South of the equator during the sunlight portion of the orbit. The operation limits were chosen with consideration of area of interest to scan, variation in the inclination of the earth's axis over one year and required paddle area. Benefits in terms of data received by increasing the scanning region are not justified in relation to the increase in cost and increase in complexity of the design for storage and paddle deployment. These both would result from the consequent expansion in required paddle area. The fixed paddles are in a tent-like configuration (see Figure 5.1). This array for the solar cells provides a simple design for producing a symmetric power output with respect to the earth-sun plane. Tent angles from 0 degrees to 180 degrees were studied in terms of efficient use of solar cells and percentage of output power available to charge the battery. An average of 25% power excess during the scan period was desired. For the operational limits of 80 degrees North and 80 degrees South of the equator, and included tent angle of 90 degrees was found to best meet the design criteria. This was verified through consultation with Bendix Aerospace of Ann Arbor. The paddles are folded and stored during launch and are deployed when the satellite is in the final orbit after despin. The cells used are 2 by 2 cm by 12 mil thick, 2 ohm-cm N on P silicon cells with a 6 mil coverglass. N on P cells are used since they offer more radiation resistance than P on N cells. A blue-red filter is used to reject the solar energy which cannot be efficiently transformed into power, thus minimizing the effects of solar heating and increasing efficiency. In choosing a filter, the ratio of solar absorptivity to black body emissivity (α / ϵ) is made as low as possible. A blue filter has a higher ratio than a blue-red filter and therefore was not used. By using a blue-red filter, a 5% loss in power is acquired. Using these coverings, a 15% degradation of the solar cells is estimated at the end of one year. The solar cell array consists of 5 strings. Each string contains 5 parallel rows of 140 cells connected in series.

The solar cells are bonded to an aluminum honeycomb panel. This panel consists of a honeycomb core covered by an aluminum plate secured with an epoxy adhesive. A non-conductive material (G-10) is placed between the aluminum plate and the solar cells to insure isolation of the cells. The backs of the paddles are painted to produce a low absorptivity and a high emissivity in order to keep the operating temperature as low as possible for maximum efficiency.

5.3 BATTERY

Nickel-cadmium cells were chosen because they have a long recycle life and are capable of being overcharged. Silver-cadmium and silver-zinc cells were considered, but both were rejected. The life of silver-zinc is too short for our mission and silver-cadmium

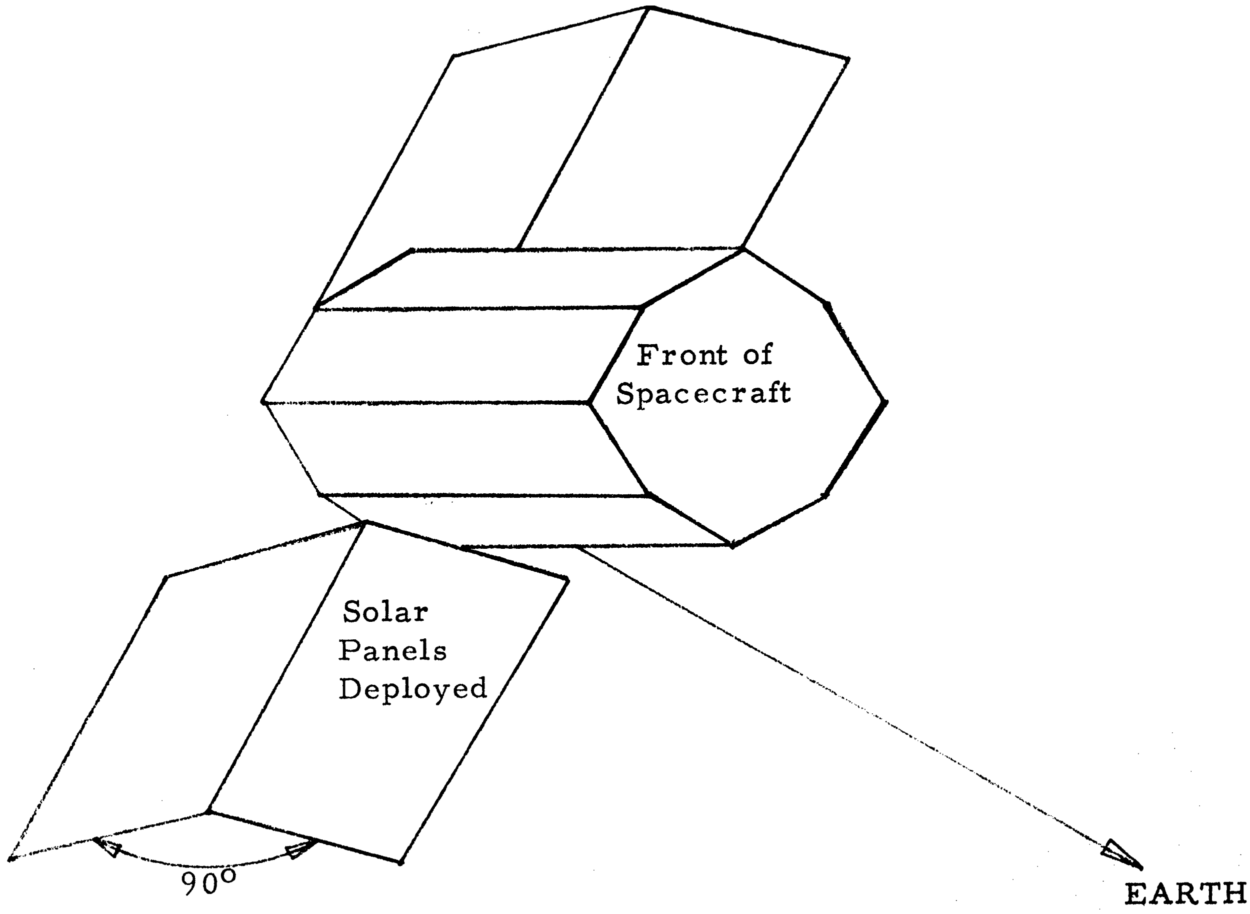

Figure 5.1

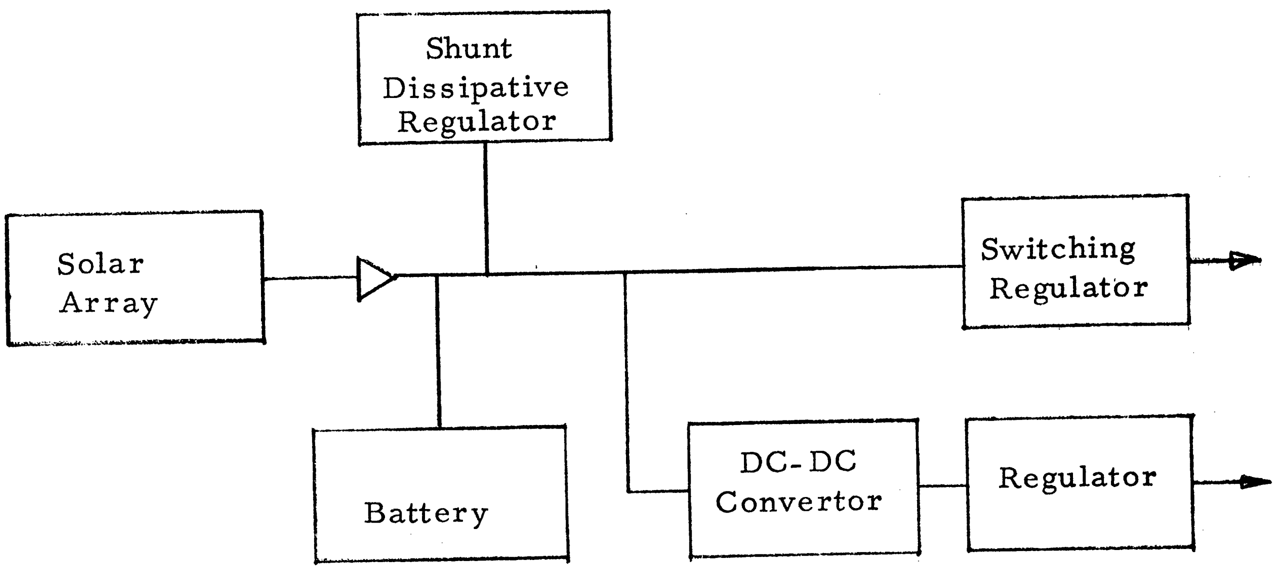

Figure 5.2 Power Block Diagram

cells have a shorter recycle life than nickel-cadmium. Based on a 96 minute orbit, our mission requires 5475 cycles. Since it is allowable to overcharge nickel-cadmium cells, the sophistication of the charging circuit is lessened; regulation of the charging circuit does not have to be as precise as it would if no overcharge was allowed.

The size of the battery was determined by the number of initial orbits and initial power required before the paddles are deployed. Figuring 80% discharge initially, 28 8 amp-hr cells formed the battery which will supply initial power required for 4 orbits. After the satellite is in its operating mode, the first two orbits will be used solely to charge the batteries. During the operating mode, the battery will supply the 16 watts on the dark side of the orbit and also provide additional power at the beginning and end of each scanning period. During periods of full sunlight, the battery will be charged from the 25% (average) excess power supplied by the solar array. At end of life, the solar cells provide a charge capacity of 25.8 watt-hr.

5.4 POWER CONTROL UNIT

The Power Control Unit will provide the different voltage inputs for the communication section and the sensor section, see Fig. 5.2. The unit also contains the charging circuit regulator for the battery and the shunt regulator to dump excess power. Communications (including housekeeping) circuits operate at 28 volts dc. To provide a constant 28 volts form the solar cells, a switching regulator is used. The sensor section requires a bias voltage of 1 kv for the photoelectric detectors. To provide this bias voltage a dc-dc converter is used to increase the voltage. This again is followed by a switching regulator. Excess power is delivered to the battery. Once the battery has achieved its charge, the shunt regulator dissipates the power as heat. Shunt regulation is achieved through a Zener diode.

Based on .4 volt line loss, .7 volt diode loss and 15% regulator loss, the solar panels must provide a minimum of 34 volts. Diodes are placed in the line from each solar array section to prevent discharge into those cells not illuminated. Both the switching regulator and the dc-dc converter are connected in parallel to the solar array output. Due to the efficiency of the regulator following the dc-dc converter, the dc-dc converter must deliver a minimum of 1180 volts.

5.5 PANEL DEPLOYMENT (Figure 5.3)

Panel deployment involves three steps:

1. Upon release of the yo-yo despin ring, the two outside sections of each paddle will swing out and lock to form two flat "plates". The "plates" are still held to the sides of the satellite by a retaining cord.
2. Electric current is passed through the cord on command which will melt the cable and allow the flats "plates" to fold away from the satellite side to a final position perpendicular to the long axis.
3. The central hinge is unlocked with a similar device to allow acquisition of the 90 degree tent angle.

All motions in the deployment mode are effected by torsion springs located about the hings and pivots. The device that unlocks the central hinge in step 3 is similar to the retaining cord in step 1. That is, the locking device consists of a taut cord that inhibits motion of the spring loaded system. To release the hinge, the cord is melted with an electric current.

Table 5.1

Power Breakdown - Watts

	Launch & Initial Orbits	Operating Mode	Eclipse
Communications	21.21	44.74	8
Sensors	0	14	8
Delta V Motor	19-28	0	0
Gravity Gradient	4.5-7.5	0	0
Attitude Sensors	0	2	0
		60.74	16

5.6 POWER PROFILE

As seen in the power profile (Figure 5.4), maximum power is delivered between 80 degrees North and 80 degrees South of the equator. Investigation of the profile indicates that all power required is provided and there is an excess available to charge the battery. If for any

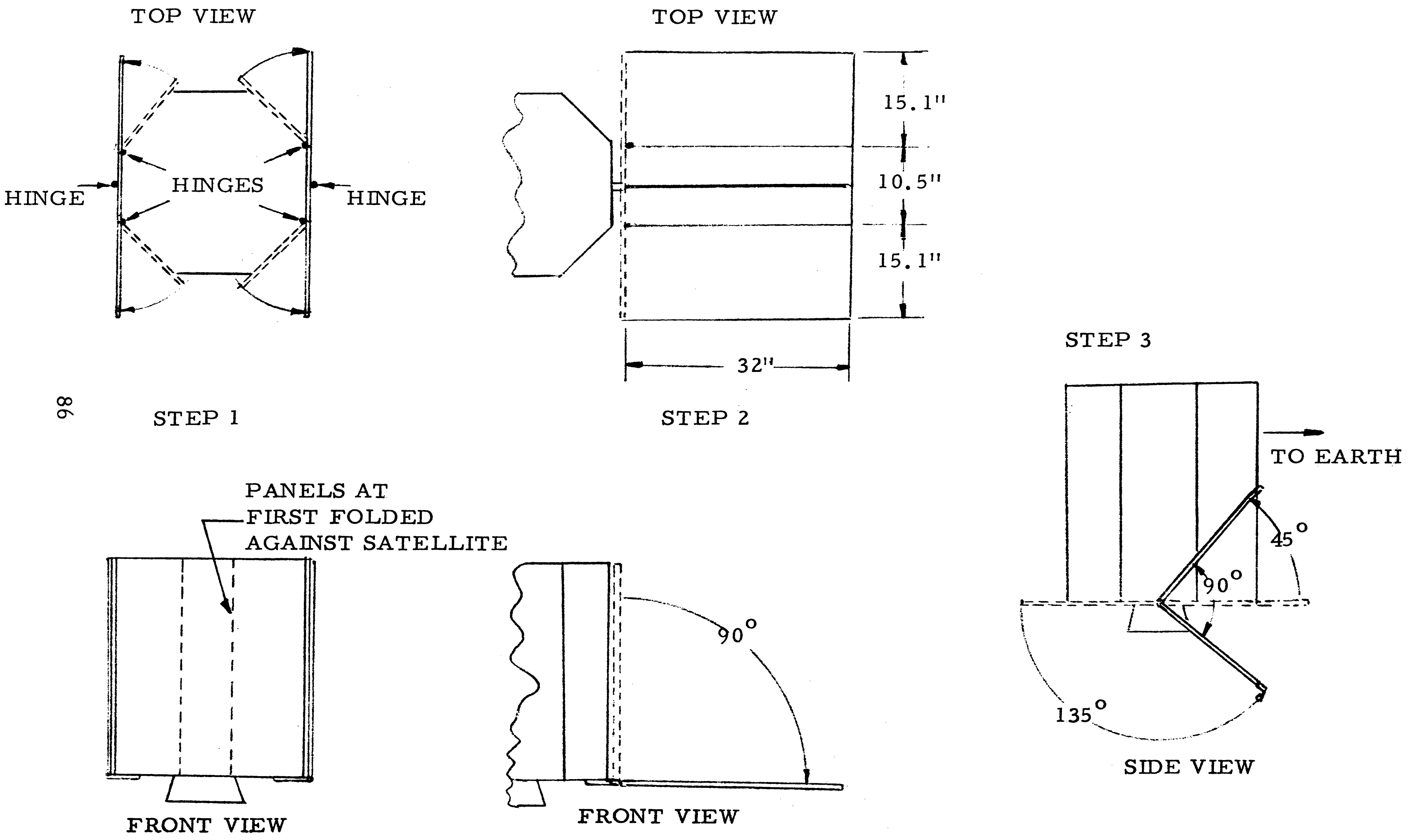

Figure 5.3 Paddle Deployment Schematic

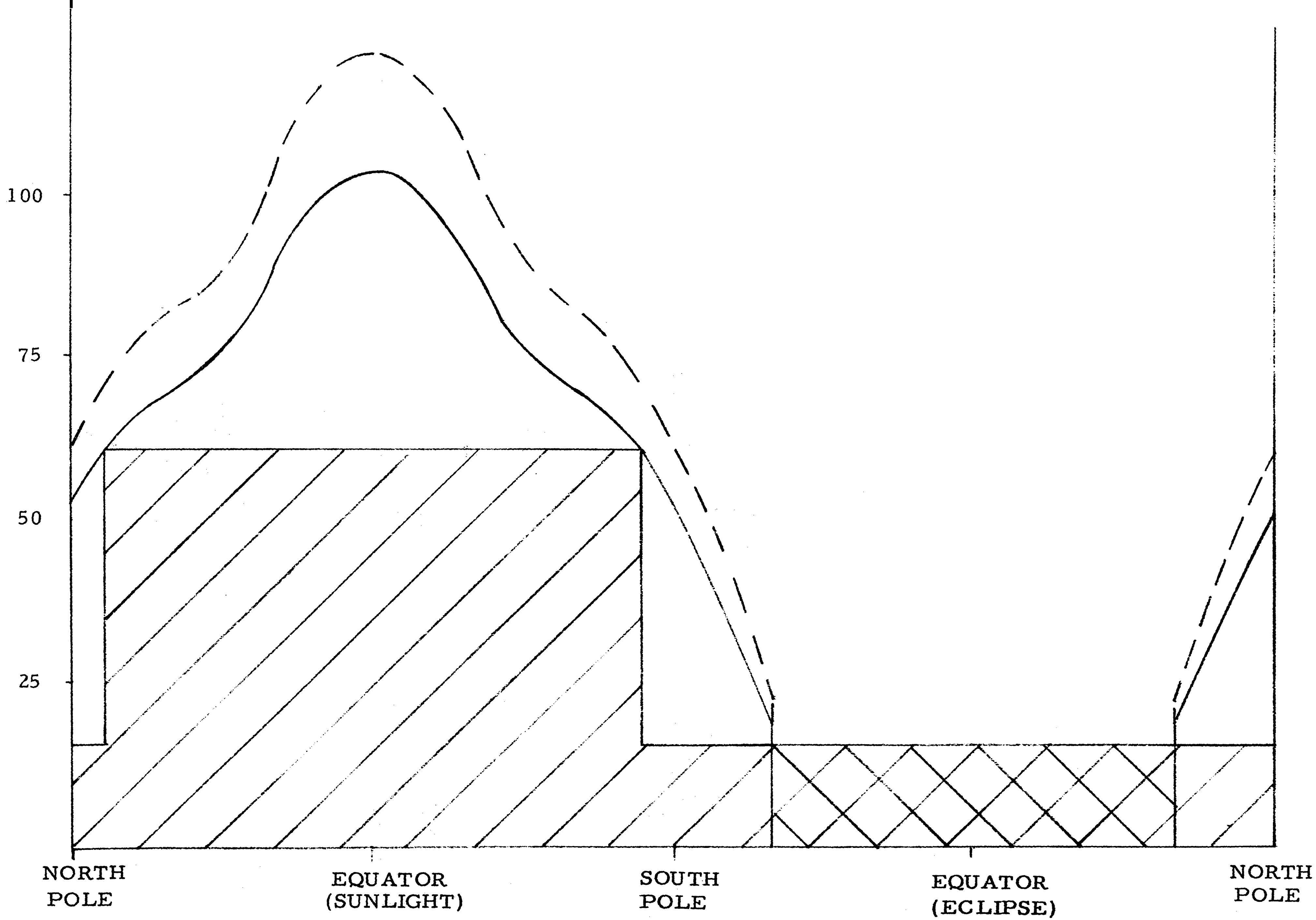

Figure 5.4 Power Profile

reason an increase in power is required above the 61 watts, there is still an excess of power left after the battery has been charged. This is likely to happen over England where a high power transmitter is required. However, all functions will not be performed (see Section II, 2.0). On the profile, the dotted line represents initial power available and the full line, the end of life power available. The uni-directional area is the power required and the cross-hatched area is the power provided by the battery. The open space is the excess power used to charge the battery.

Table 5.2

Weight Breakdown - lbs.

A.	Solar cell		
	1. Silicon cell	2.9	
	2. Solar cell solder	1.45	
	3. Coverglass .006 quartz	1.085	
	4. Adhesive (XR63489) (coverglass to cell)	.904	
	5. Adhesive (RTV577) (cell to substrate)	.724	
	6. Connector tabs	.543	7.606
B.	Solar panel		
	1. Aluminum 6061 rectangular (4)	3.016	
	2. Adhesive - HT 424 film (4)	2.672	
	3. Honeycomb core	.695	
	4. G-10 non-conductive material	.495	6.878
C.	Battery		
	1. 28 Ni-Cd cells at .88 lbs/cell	24.7	24.7
D.	Power control unit		
	1. Dc-Dc converter	5.0	
	2. Shunt dissapative regulator & switching regulator	2.0	7.0
			46.184

5.7 REFERENCES

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6

STRUCTURE AND THERMAL ANALYSIS

6.1 STRUCTURAL DESIGN

6.1.1 Structural Considerations

The structure of OBSERVER resulted from the consideration of many design areas and mission objectives. All satellite systems and components must be supported and enclosed in the best possible configuration. It was determined that no more than 15% of total payload weight would be allocated for the basic structure and the enclosing panels. The structure must be able to withstand 25 g's axial acceleration during launch and also have good radial strength characteristics because of payload spinup prior to orbital injection. It must have good damping characteristics and be low in cost. The structure should also permit a wide range of equipment mounting positions, each of whic provides pathways for thermal conduction. A maximum latitude of equipment mounting positions will insure that the center of gravity can be placed on the spin axis, and that the principal moment of inertia will be parallel to the spin axis. Another important design consideration was the support of a large sensor system to be mounted off-axis on the earth side. The thermal properties of the structure are covered in the Thermal Control section of this report.

6.1.2 Structural Configuration

The structure chosen for Project OBSERVER was a cruciform shape constructed of four quadrant subassemblies. Each of the quarter-frame assemblies is made of sheet aluminum connected by aluminum open channel. The quarter-frame assemblies are connected at two points along the axis of the satellite by center fittings of machined aluminum. Two spacing rings of tubular aluminum maintain the proper orientation of the quadrant assemblies and also serve for attachment of the outer panels. Spacing rings are connected to 1/4" aluminum honeycomb outer panels by clamp fittings bolted to a clip angle bonded to the panel inner surface. The satellite is octagonal and each of the eight outer panels is 10.72 inches in width and 32 inches in length. The eight-sided configuration is more fully discussed under Thermal Control. An attachment ring is connected to four points at the bottom of the structure and provides the interface between the structure and the "E" section of the Scout launch vehicle.

Riveted construction was used in the quadrant assemblies. While only slightly heavier, this type of construction provides much better damping characteristics and is much cheaper than welded construction. Quadrant assemblies are similar to those used in the relay communications satellite. Such quarter-frame assemblies have been extensively tested and have proved capable in actual practice of supporting 1500 lbs per assembly. The total spacecraft, therefore is capable of 30 g's of acceleration if the total weight is 200 lbs. Detailed structural analysis may be found in Appendix F . The completed spacecraft should be able to withstand an acceleration of 10 g in the radial direction.

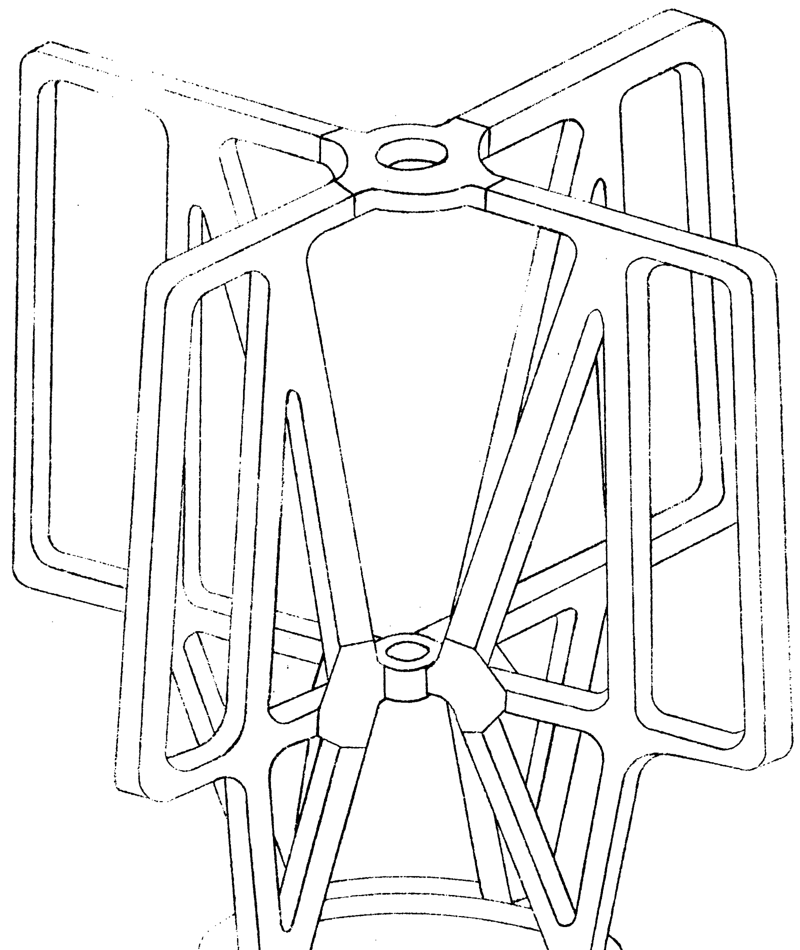
6.1.3 Structural Components

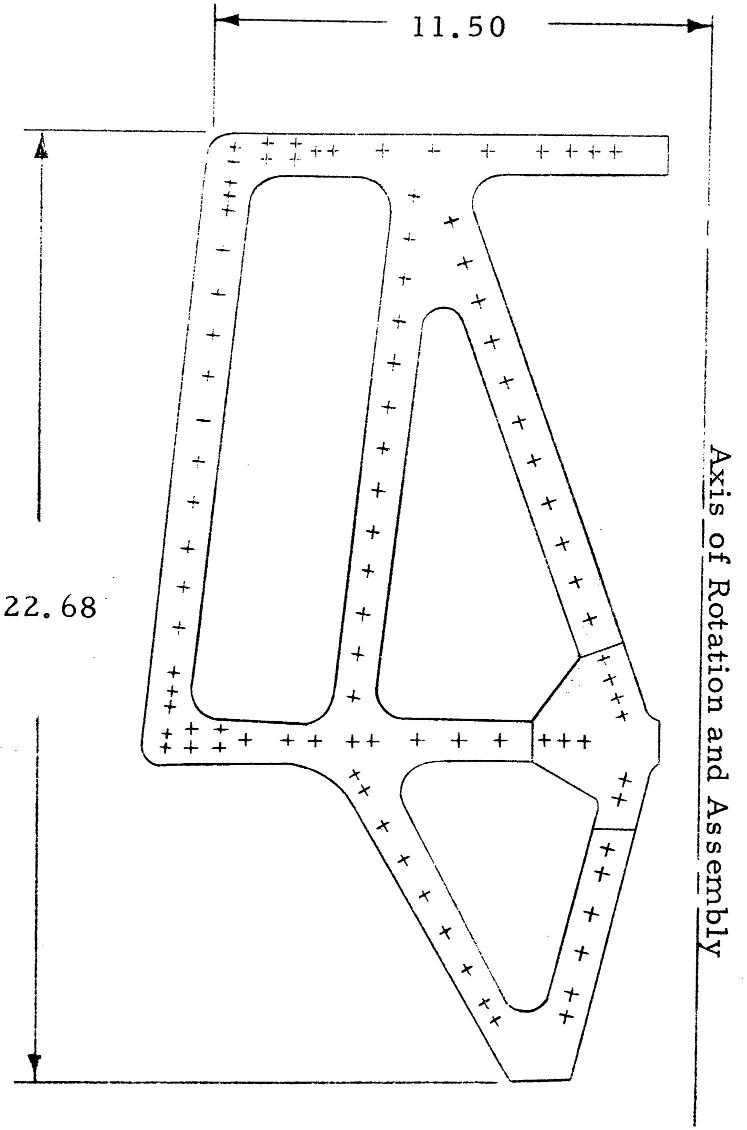
No.	Component	Description	Total Wt.
4	Quarter-frame assy.	2024-T3 Alum.	5.4 lb
2	Center fittings	6061-T6 Alum.	0.7 lb
2	Spacing rings	1" O.D. Tubular Alum.	2.4 lb
1	Separation ring	6061-T6 Alum.	1.0 lb
16 sets	Clamp and angle fittings		0.5 lb
8	Outer panels	0.25" Alum.	10.0 lb
1	Top panel	honeycomb,	
1	Bottom panel	thermally cond. bonding adhesive, double skin, perforated cell walls and inner skin.	
			20.0 lb

6.1.4 Choice of Cruciform Structure (See Figure 6.1, 6.2)

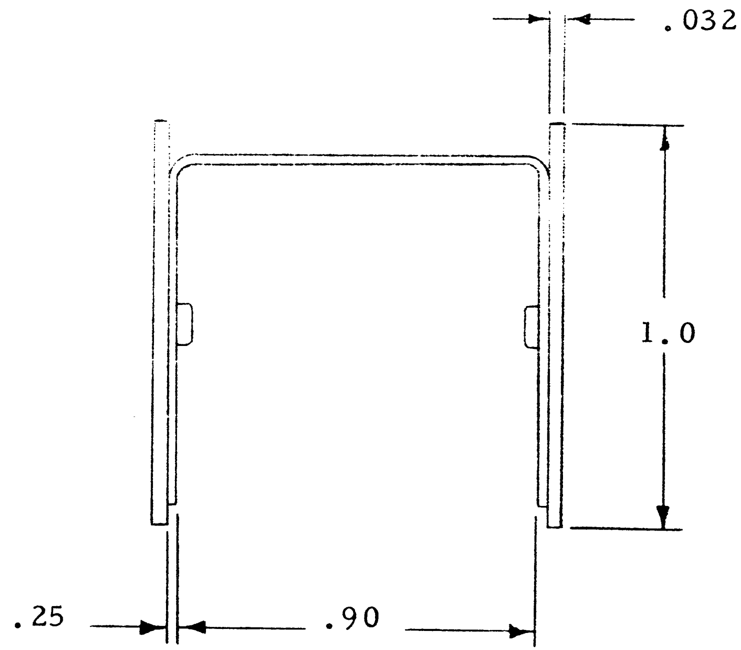
The final choice of structural configuration was based on a combination of factors. A baseplate configuration was considered, but the higher weight was unacceptable, since internal structural elements were still necessary for support of a rotating mirror and a relatively large sensor system. The requirement for a wide latitude of equipment mounting locations was met by the cruciform configuration as was the requirement for the support of a sensor system which extended along the axial direction of the spacecraft. Equipment is mounted directly to the structural members which eliminates the need for mounting platforms.

Figure 6.1 Basic Structure (Spacing Rings Not Shown)





QUARTER-FRAME ASSEMBLY



CROSS SECTION OF STRUCTURAL MEMBER

Figure 6.2

Structural weight of the cruciform structure comprises less than 10% of the total payload weight. The cruciform structure will be able to withstand the required accelerations in both axial and radial directions.

6.1.5 Equipment Mounting (See Figures 6.3, 6.4)

Equipment is mounted using bolts connecting directly to main structural members. In the case that direct bolting cannot be accomplished, angle mountings are used to transfer stresses and heat to the spacecraft structure. Spacing rings are also mounted to the basic cruciform structure so as to provide good thermal paths between rings and structure. Rivet heads are flush to provide maximum contact area between components and the structure in order to satisfy thermal requirements.

The Delta V motor is mounted between the quarter-frame assemblies in the lower portion of the spacecraft, just beneath the lower center fitting. It is attached so that the thrust axis passes through the center of gravity. The fuel tanks for the Delta V motor are also located in the lower portion of the satellite at the positions shown in Figure 6. The rotating mirror assembly is located above the basic structure of the spacecraft. One of it's driving motors is located inside the mirror assembly and the other is located beneath it. The sensor system is located on the side of the spacecraft which faces the earth. Other spacecraft components are positioned to insure proper center of gravity location, moment of inertia considerations, thermal restrictions, and mounting compatibility.

6.1.6 References

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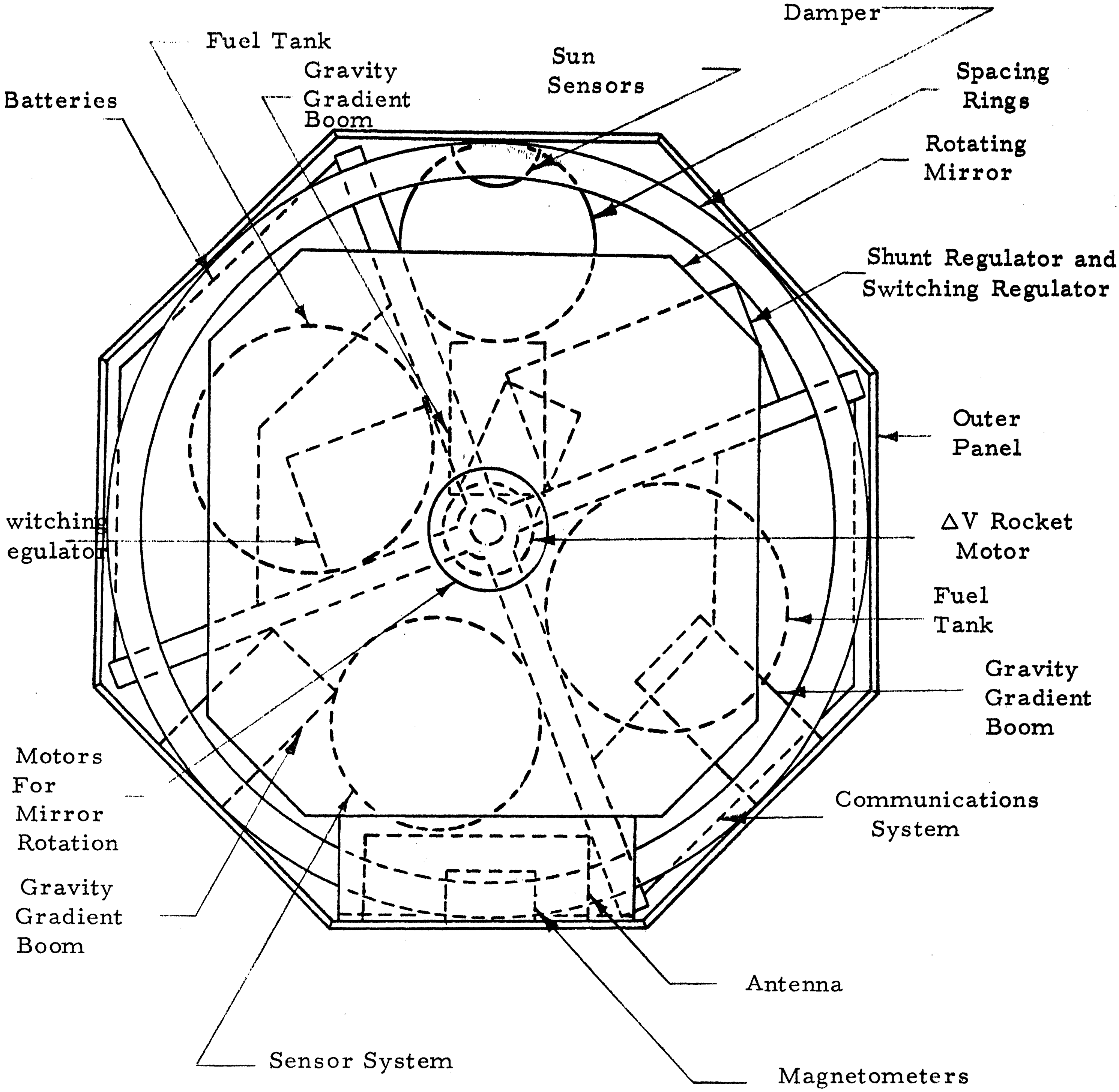
6.2 THERMAL CONTROL

6.2.1 Criteria

To insure effective and continuing operation of the previously described satellite systems, it is necessary to control the thermal environment of the satellite. Most all satellite components are temperature sensitive, particularly the electronic parts. Therefore the hottest and coldest temperature at which these components operate must be controlled. For a one year lifetime, these temperatures can be simply

Figure 6.3 Equipment Placement



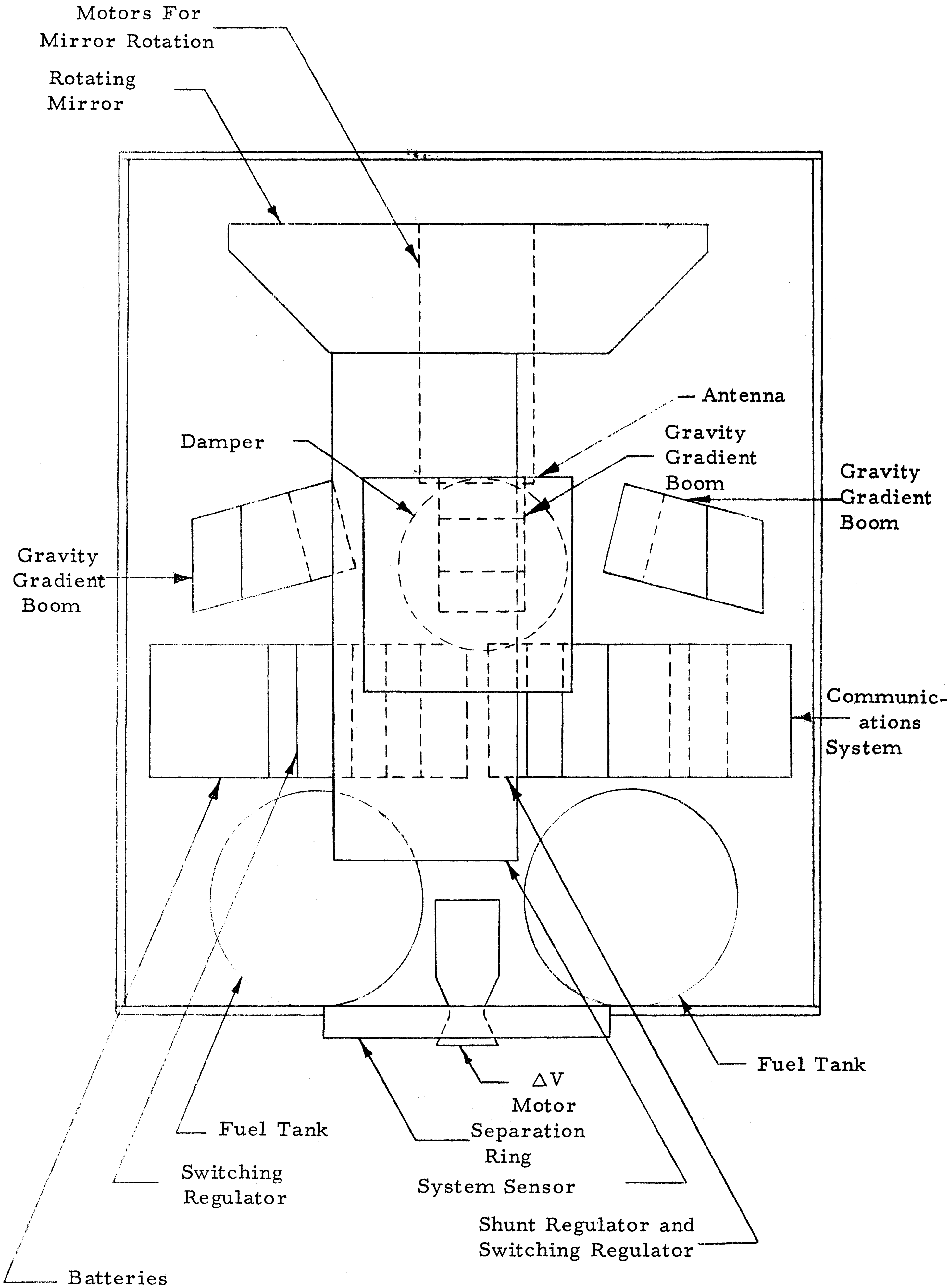


Figure 6.4 Equipment Placement Side View (Basic Structure not Shown)

and reliably managed through passive means. A passive thermal system makes use of surface paints and coatings, heat conduction through the structure and thermal insulating materials. In this manner, all provisions for thermal control are provided proceeding the launch of the craft so that during the satellite lifetime, no thermal dependence on mechanical systems is necessary. Thus very reliable yet economical thermal control is provided.

6.2.2 Thermal Control of OBSERVER

During insertion into orbit OBSERVER is protected from aerodynamic heating by the Scout vehicle's heat shield. The temperature inside the shroud increases to approximately 100°F. Then as the satellite is put through orbit correcting maneuvers the temperature inside OBSERVER decreases, but not enough to impair operation. Thus no extra provisions need be made for this portion of the mission.

Once the satellite attains operating status, however, more stringent thermal limitations are imposed on the system due to the temperature sensitive components being used. Surface coatings are primarily used to control the temperature within the satellite. By covering OBSERVER with a coating which provides an effective absorptivity of .1 and an effective emissivity of .12 the satellite skin temperature is held to an average of 54°F, with fluctuations of ± 3° during the orbit. With this very stable skin temperature, the temperature of internal components can be controlled with insulating material and selective positioning. Equipment with similar permissible temperature ranges are grouped together in packages. For instance communications are grouped together in one package, covered with an insulation such as fiberglass, and the temperature range is held between 32°F and 180°F at all times. During sun side of the orbit communications dissipate about 40 watts of their input power as heat, thus this package heats itself staying above the 32°F limit. The batteries are also grouped in a package which is likewise covered with a fiberous insulation. This package is held at 40°F while the batteries are recharging. The last thermal package contains the phototubes from the sensor system. This package is similarly covered with a fiberous insulation. In this case the insulation prevents heat conduction into the package which is held at -40°F. To maintain this temperature, the package is placed next to the satellite side panel which always faces outerspace. This panel acts as a heat sink; heat is conducted from the package to the panel. In the small area adjacent to the sensor package, the panel is coated with black paint to facilitate radiation of this heat.

In all cases, the thermal packages to be used must be tested in a simulation environment in the laboratory to determine specific insulating fibers and the thickness of these coverings. Also while OBSERVER is orbiting, temperatures of the satellite skin and of the individual packages are monitored.

6.2.3 Sources of Radiation (Figure 6.5)

While orbiting the earth, OBSERVER is subject to three distinct external energy radiations: solar, albedo, and planetary. The primary energy radiation incident on the satellite is from the sun. This energy flux is nearly constant in that it varies as the inverse square of the distance from the sun. The eccentricity of the earth's orbit causes a variation of $\pm$ 3% during the year. The second energy source is the earth's albedo, the sunlight reflected from the earth's surface and atmosphere. This energy flux occurs in the same wavelength as the solar radiation and is a function of altitude, which for a circular orbit is constant. This flux, however, is incident on the satellite only during 180° of the sun side of the orbit. The earth thermal radiation is the least of the energy fluxes. The energy from this source is in the infrared region and is incident on the satellite during the complete orbit. Again this flux is a function of altitude and thus constant. For OBSERVER'S orbit, a polar orbit which is in the plane of the earth and sun, these radiations are the same each revolution.

6.2.4 Thermal Energy Balance

If the satellite was to continuously absorb energy from the three described sources without radiating any energy from itself, the temperature of the satellite would increase exponentially. Of course the satellite does radiate heat to outer space, which has a temperature of about 4°R. It emits this energy in the infrared wave lengths and the amount it radiates is a function of the temperature, the area and the coating of the satellite surface. If the heat dissipated by the electronic and mechanical devices in the satellite is taken into account, the steady state condition can be described as:

$$q_a + q_i = q_e$$

where

q_a = energy absorbed from solar, albedo and earth radiation

q_i = internal heat dissipated by satellite equipment

q_e = energy emitted by the satellite to outerspace

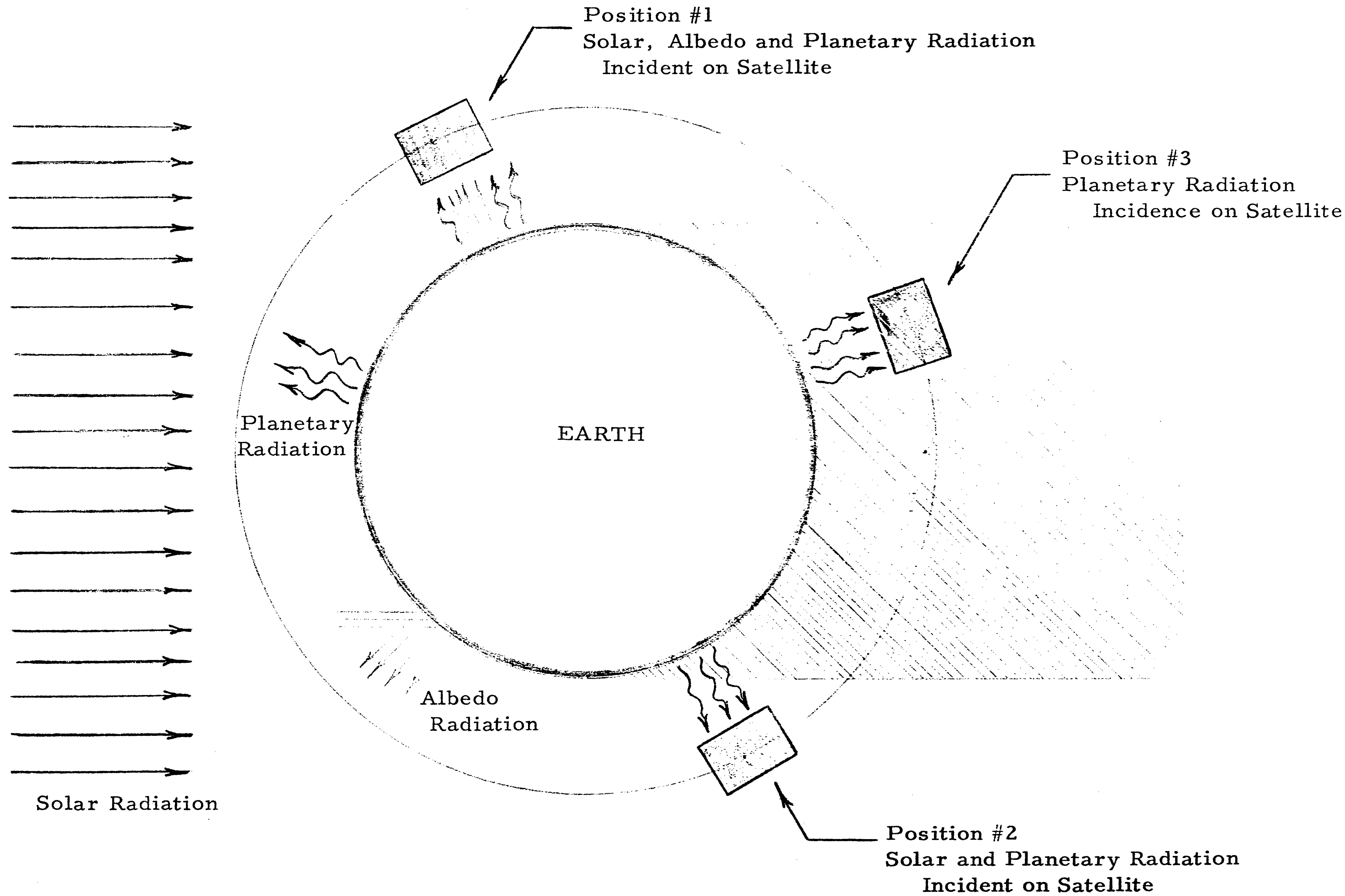


Figure 6.5 Diagram of Radiation Incident on OBSERVER at Different Positions in Orbit
(Three different combinations of radiation occur during the orbit)

This steady state condition would enable us to dictate the temperature at any position in the orbit if the satellite temperature were allowed to reach equilibrium. However, due to the dynamics of the satellite system, equilibrium is never reached. Therefore only a "temperature range" within which the satellite will remain can be calculated with the steady-state equation. Calculation of actual temperatures is an iterative process best left for a computer.

6.2.5 Interior of Satellite

The above calculation produces an average satellite temperature, that is an average temperature of all components, structure and skin of the satellite. Certainly temperature gradients appear within the satellite. The sun side of the satellite is a heat source during the day light portion of the orbit; the side facing outer space is a heat sink at all times. Also, internal components such as radio transmitters and battery units dissipate different amounts of heat at different times.

Due to the difference in temperature of points within the satellite, heat will be transferred in an attempt to reach thermal equilibrium. This heat will be transferred by thermal radiation and thermal conduction. The thermal radiation is analogous to the previous analysis; "heat flux in" equals "heat flux out" in the steady-state case. Thermal conduction may be described as follows:

$$q_t = \frac{T_1 - T_2}{R}$$

where

q_t = amount of heat transferred temperature

$T_1 - T_2$ = difference between two points

R = thermal resistivity

Thus the interior of OBSERVER can be thermally controlled by coating the exterior of all components and structure, by controlling heat paths, and by using insulating materials.

The solution of the temperatures and heat transfers within OBSERVER is a very complex procedure. An accurate mathematical analysis would require a detailed knowledge of the geometric placement of all components and structural parts within the satellite as well as an extensive knowledge of the materials and properties of all components. Only through actual testing can the thermal characteristics of the interior be determined. Certainly such testing procedures will isolate any problems and appropriate changes can then be made.

6.2.6 Approach to Thermal Control of OBSERVER

The thermal control system for Project OBSERVER should be low cost and function for one year. It was initially decided that the thermal system should be passive so that it would be economically attractive yet very reliable. Paints and insulation are therefore used.

Originally the OBSERVER satellite was to have a hexagonal shape. Calculations indicated, however, that it would be thermally advantageous to use an eight sided structure. This provided two readily available heat sinks in the sides facing outer space and also improved the ratio of actual surface area to surface area projected towards radiation sources.

Next it was necessary to chose an absorptivity and emissivity suitable to the mission. The steady-state temperatures were calculated at the point in the orbit of the largest area projected towards the sun and at a point in the earth's shadow. In this manner, the two extreme temperatures possible in a steady-state condition were calculated. Different α's and ϵ's were tried until finally what seemed to be an optimum range of possible temperatures was calculated. This is 125°F at 30° from the equator and -95°F in the earth's shadow for $\alpha = .1$, $\epsilon = .12$. Then an average temperature was calculated for one complete steady-state orbit to be 54°F. The thermal time constant, which is an indication of how fast the satellite cools during the eclipse, was found to be 8.4°/hr. These parameters seemed acceptable, thus a computer program was written to determine the dynamic or actual temperature throughout the orbit. A temperature of 52.4°F was assumed as the satellite came into the sun and the temperature was observed to vary from 52.2°F at the North pole to 56.3°F at 85° South of the equator. This program and calculations are found in the thermal appendix.

One particular coating material with an absorptivity of .1 and emissivity of .12 may not be available. In this case two or more coatings can be used in some pattern of stripes or patches to yield an effective absorptivity of .1 and emissibility of .12. Or possibly some combination of surface coatings and insulation within the satellite might be used to produce the desired effect.

Due to OBSERVER'S three axis stability, the energy flux on each side panel is not equal. For instance, in the sunlight portion of the orbit, the side facing the sun receives more radiation than the side facing the earth and that receives more radiation than the side facing outer space. Thus heat must be transferred from panel to panel to achieve some sort of temperature stability across the satellite. The heat can be transferred through the aluminum honeycomb panels themselves. This transfer is facilitated by applying a thermal grease to the panel joints. Originally heat pipes were considered. They would very reliably transfer heat across the satellite isothermally. It was thought that the panel ring supports could be constructed as annular heat pipes with whatever wick and volatile liquid necessary, incorporated in them. However, for the amount of heat that would be transferred with such pipes, the cost proved prohibitive. Instead the ring supports walls have been thickened to 1/4 inch to transfer more heat.

Inside OBSERVER heat is conducted through the structure. A simplified model of this conduction might be solved with an analog computer. An actual solution is analytically unfeasible. Similarly, heat radiation within the satellite cannot be calculated because it is impossible to calculate the actual view factor needed for this calculation.

The internal components are grouped in packages and the temperature range of these is controlled by the choice of insulating material. Fiberous insulations are used for a number of reasons: 1) they have a low thermal conductibility, 2) they have a relatively low density, 3) they have structural integrity, 4) they are easily integrated into the thermal system, and 5) due to their flexibility they easily conform to the shapes an contours of the structural components. There are three requirements which critically limit the temperature of the three main packages: 1) keep the VHF receiver above 32°F, 2) keep the batteries at about 40°F while charging, and 3) keep the phototubes at -40°F while in operation. All these can be taken care of with proper selection of insulation. To help keep the phototubes at -40°F, they are placed next to the side panel facing outerspace, which is a heat sink, and this panel will be painted black in the immediate area of the phototubes.

One other precaution is taken so that the Delta V motor does not overheat the satellite system while in use. The motor itself is wrapped in a fiberous insulation. Also a thin (about 1/4 inch) high temperature multilayer insulation, such as aluminum-fiber-glass insulation, covers the bottom surface of the satellite to prevent the heat radiation from the nozzle from overheating the bottom aluminum honeycomb plate.

As implied above, mathematical models are dependent on extensive information about specific details of the satellite. The mathematical analysis becomes very complicated but on the other hand, oversimplification increases the variation between model and system and thus is not always particularly useful. Actual measurements to determine thermal conductivity within the satellite must be performed within an environmental test cell using the calculated temperature range and adjustments can then be incorporated as needed.

From a consideration of the orbital analysis and information regarding pasive thermal control systems, it seems feasible to obtain a satisfactory thermal system for OBSERVER under the prescribed limitations.

6.2.7 References

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7

TIME AND COST ANALYSIS

7.1 ANALYSIS SCHEME

The method used to generate the cost analysis and development schedule for the OBSERVER system is detailed within this section. As part of a feasibility study, the cost and development schedule must be approximated to determine if there are any factors prohibitive to the development of an active Earth Resources Satellite system. Refer to Figure 7.1 for the summary of the important satellite development parameters.

7.2 TIME DEVELOPMENT

It is desirable to have the OBSERVER satellite operational by March 1971 to coincide with the vernal equinox. At this time of year the vegetation in the northern hemisphere is beginning a new growth cycle. This period of agricultural development would be most interesting to observe, and it would be advantageous to acquire information while the satellite is at a high level of effectiveness.

Referring to Figure 7.2 , notice that engineering development testing does not require complete component development. Thus, many parts may be developed simultaneously to achieve operational status of the system by the February 1971 launch date. This type of development requires extremely good coordination among the various subcontractors to insure good system implementation at the qualification testing level. At the end of engineering and development testing, there is a five month delay allowed for engineering re-designs and manufacturing delays.

7.2.1 Ground Systems

The development of the vast network needed to receive and process the information from the satellite will be initiated at the same time as the design study. This will require a period of several year, for tracking stations must be coordinated and test sites established or updated for satellite use.

7.2.2 Scout Launch Vehicle Coordination

After the flight model has been acceptance testing, it will be delivered to LTV Scout Vehicle Group. They are responsible for orbiting the satellite, adapting the payload to the booster and providing the support system until the satellite is injected into orbit.

Figure 7.1 Development Chart for the OBSERVER Satellite

System	Weight	Power	Cost	Development Time
Sensor	38 lbs	14 watts $_{max}$	\$ 250,000	15 months
Communication	28 lbs	44.74 watts	\$ 300,000	2 months
Power Systems	46.2 lbs	---	\$ 120,000	2 months
Stability & Despin	19.7 lbs 6 lbs 25.7 lbs	7.5 watts $_{max}$	\$ 27,600 \$ 5,000 \$ 32,600	2 months
Attitude Sensors	7.1 lbs	2 watts	\$ 6,000	2 months
ΔV Motor	31.43	28 watts	\$ 20,000	2 months
Structure	20 lbs	---	Structure Mod. \$ 150,000 Thermal Mod. \$ 120,000 Qualification Model \$ 150,000	3 months develop 5 week Phase I test 5 week Phase II test 2 week Phase III test
Satellite thru Acceptance Testing	196.33 lbs	---	\$1,148.600	---
Launch Vehicle	---	---	\$1,500,000	15 months
Orbit Total	---	---	\$2,648,600	---
Ground Support	---	---	\$2,000,000	---
System Management + 10% fee	--	--	\$ 300,000 \$ 294,860	2 years
TOTAL for (1) Satellite Operational for one yr.	---	---	\$5,243,460	---

7.3 COST ANALYSIS

The cost model assumes that a central agency will coordinate subcontractors who will do the final design, fabrication, and testing of the satellite systems. This central agency is responsible for a system design study, coordination of the various subcontractors and the acceptance testing of the satellite as specified by a buyer. The buyer is assumed to be responsible for data interpretation as delivered by the ground support system. The cost of the systems management for the OBSERVER satellite is estimated at $300,000 for a two year period plus a 10% fee of $294,860. This fee is for initial cost estimate only and the type of fee assessment would be determined when the contract was awarded. Figure 7.2 details the development of the OBSERVER system.

7.3.1 Structure Cost

The satellite structure is defined as the component support structure and the covering panels. The structural contractor is responsible for final engineering design and testing. This phase will necessitate two engineering satellite models: a structures model for static and vibrational studies, and a thermal model that will become the qualifications model in later development stages. It is assumed that the structural contract will continue through qualification testing of the satellite. The cost of the entire satellite structure, for a structures model and a flight model, as delivered for acceptance testing, including development and testing, is $420,000.

7.3.2 Components

Like the structures, all other satellite systems will either be contracted out or purchased and delivered by the structures contractor for qualification testing. The cost of satellite subsystems, enclusive of structure, but inclusive of individual development and space testing is $1,148,600.

7.3.3 Launch Vehicle Group

After acceptance testing of the proposed flight model, the OBSERVER is delivered to the LTV Scout Launch Vehicle Group. They are responsible for the payload to booster integration, checkout, and launch to orbit. The cost is $1,500,000.

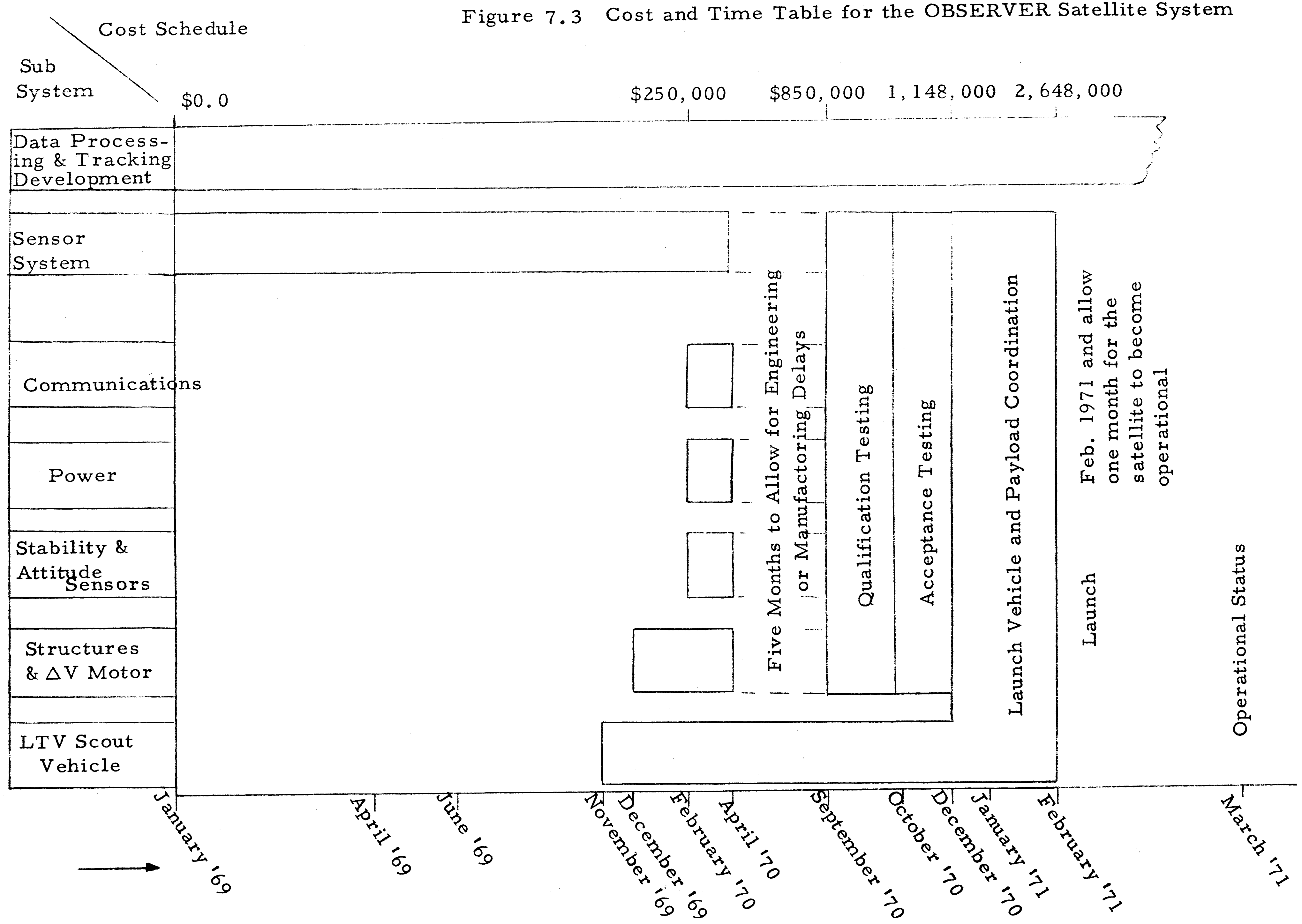
7.3.4 Ground Stations

The communications system requires that STADAN network be rented for one year at a cost of $2,000,000. It is assumed that test sites and data gathering and processing costs will be paid by the user of the OBSERVER system.

7.4 SUMMARY COST AND TIME

There appears to be no factor that could prohibit the development of the OBSERVER System. The allotted time is more than ample for the development of a system, including five months of delay time for possible design problems. The cost is reasonable, assuming the most economical system development utilizing one test model and one flight model with no redundant systems. Once the initial flight model has been completed other satellites may be produced at a relatively low cost for the expansion of the Earth Resources Study. For a detailed analysis, refer to Figure 7.3.

Figure 7.3 Cost and Time Table for the OBSERVER Satellite System



APPENDIX A
SENSOR SYSTEM

A.1 SENSOR SYSTEM DESIGN

In designing the sensor the following relationships were used.

1. $d_1 = \beta D_1 F_1$

2. $d' = \beta D_1 F_d$

3. $\delta\lambda = \dfrac{D_1 \beta}{D_2 (d\theta / d\lambda)}$

where

β	= instantaneous field of view
D_1	= diameter of collector
F_1	= focal number of collector
D_2	= minimum diameter of dispersing optics
F_d	= focal number of beam at the focal plane
d'	= width of focal plane
$\delta\lambda$	= resolved wavelength interval
$d\theta/d\lambda$	= angular dispersion of prism.

Figure A.1 shows a sensor system schematic with these parameters displayed.

A.2 PRISM CALCULATIONS

The best optical image in the case of a prism can be obtained if the prism is traversed by parallel light and if the light rays pass through the prism parallel to its base. These conditions can be realized for only one wavelength and cannot be true simultaneously for a range of wavelengths passing through a prism spectrometer. In the construction of a prism instrument, however, an approximation to these conditions is made and the prism is said to be set for minimum deviation. Figure shows this condition of minimum deviation.

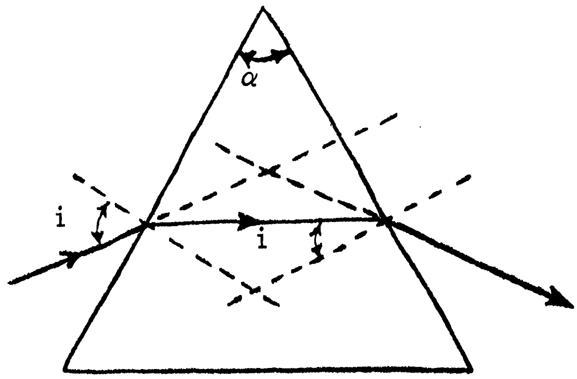


Figure A.1 . Ray of Light Passing Through a Prism at Minimum Deviation.

The dispersion by a prism is expressed as a measure of the angular separation of two light rays which differ in wavelength. This dispersion is the deviation $d\theta/d\lambda$.

$$d\theta/d\lambda = \left(\frac{d\theta}{dn}\right) \left(\frac{dn}{d\lambda}\right)$$

where

$\frac{dn}{d\theta}$ = change in refractive index with wavelength

$\frac{dn}{d\lambda}$ = characteristic dispersion of the prism material

$$\frac{dn}{d\theta} = \frac{\sqrt{1 - n^2 \sin^2 1/2\, \alpha}}{2 \sin 1/2\, \alpha}$$

where

n = refractive index

α = prism apex angle

A design point of 0.6μ was chosen. For fused quartz, $n = 1.5$ and $\frac{dn}{d\lambda} = .03\ \mu^{-1}$ at 0.6μ.

A prism apex angle of 60 degrees was chosen.

$$\frac{d\theta}{d\lambda} = 4.55 \times 10^{-2}\ \frac{\text{rad}}{\mu}$$

A.3 SCAN MIRROR CALCULATION

Any face of the scan mirror must be as large as the collector plus a factor to allow for the scanning motion.

$$L = D_1 \left[1 + 2 \sin \alpha\right]$$

where

L = length of face

D_1 = diameter of collector

α = scan angle

For D_1 = 6.56 inches

α = 10 degrees

L = 8.85 inches

APPENDIX B

COMMUNICATIONS

List of Variables

B_{IF}	Total bandwidth necessary for transmission
B_m	Bandwidth of input signal
c	Speed of light
D	Deviation ratio
d	Receiver dish diameter
d'	Transmission distance
f	Operating frequency
F.S.L.	Free Space Loss
G_R	Gain of receiving antenna
G_T	Gain of transmitting antenna
I_G	Ground receiver frequency instability
I_S	Satellite transmitter frequency instability
k	Boltman's constant
L_A	Antenna attenuation loss
L_L	Cabling and equipment loss
L_M	Loss margin
P_N	Receiver noise power
P_T	Transmitted power of S-band transmitter
$(S/N)_{in}$	Signal to noise ratio received
T_A	Atmosphere noise temperature
T_G	Galactic noise temperature
T_{top}	Total noise temperature
T_S	System noise temperature
v_r	Component of radial velocity
Δf	The increase in bandwidth due to frequency instability and doppler shift
Δf_c	The carrier deviation frequency
Δf_D	Bandwidth due to Doppler shift
Δf_S	Bandwidth due to frequency instabilities

β Frequency ratio

n Antenna efficiency

λ Wavelength of carrier

SAMPLE CALCULATIONS FOR THE COMMUNICATION SYSTEM

Initial Values

G_r	= 0.00 db
d'	= 1209 n.m. = 1392 s.m.
v_r	= 24,770 ft/sec
T_S	= 100'K
T_A	= 100'K
T_G	= 3.5'K
f	= 2.202×10^9 Hz
k	= 1.38×10^{-23} joules/'K
I_G	= 10^{-7}
I_S	= 10^{-5}
L_A	= 0.35 db
L_L	= 5.00 db
L_M	= 5.00 db
n	= 55% = .55
B_m	= 136 KC
$(S/N)_{in}$	= 11 db
c	= 1.866×10^5 mi/sec = 9.85×10^8 ft/sec

Worst Case Power Link Calculations (refer to Figure 2.10)

The equation for determing the transmitted power required by the satellite is:

$$P_T = \frac{(S/N)_{in} \, P_N \, (F.S.L.) \; L_L \, L_A \, L_M}{G_T \; G_R}$$

$$P_N = k \, T_{op} \, \Delta f_c$$

$$T_{op} = T_A + T_G + T_S = 203.5\text{'K}$$

$$\Delta f_c = 2\Delta f_D + \Delta f_S + B' + 2(.05) B'$$

$$\Delta f_D = f \frac{v_r}{c} = 5.537 \times 10^4 \text{ Hz}$$

$$2\Delta f_D = 11.074 \times 10^4 \text{ Hz}$$

$$\Delta f_S = 2f(I_S + I_G) = 4.448 \times 10^4 \text{ Hz}$$

or $\Delta f_c = 2.77 \times 10^6$ Hz

and $P_N = 7.73 \times 10^{-15} = -141.2$ db

$$F.S.L. = 37\text{ db} + 20 \log R + 20 \log F$$

$$R = \sqrt{2h} = 1390$$

$$F = f/10^6 = 2202$$

and $F.S.L. = 166.72$ db

$$G_R = \frac{n \pi^2 d_S^2}{\lambda^2}$$

$$\lambda = c/f = .4473 \text{ ft}$$

Calculation for a 14 foot receiving dish ($d_S = 14$ ft) gives the receiver gain:

$$G_{R_{14'}} = 5640 = 37.50$$

And for a 40 foot dish ($d_S = 40$ ft):

$$G_{R_{40'}} = 43,273.6 = 46.36 \text{ db}$$

And therefore the transmission power required for the 14 foot dish is:

$$P_{T_{14'}} = 9.05 \text{ watts}$$

and for the 40 foot dish:

$$P_{T_{40'}} = 1.11 \text{ watts}$$

FM Modulation Bandwidth Calculations

A deviation ratio, $D = 5.0$, and interpolation error, $\gamma = .01$, gives $\beta = 3.5$. The primary bandwidth is found as follows:

$$\Delta f_C = D\ B_m = 680 \text{ KC}$$

$$B' = \Delta f_C\ \beta = 2.38 \text{ MHz}$$

With 5% guard bands the total required bandwidth is:

$$B = B' + \Delta f + 2(.05)B' = 2.77 \text{ MHz}$$

Power, Weight, Volume Distribution

Item	Number Required	Total Weight (lbs)	Total Volume (inches)	Power Required (at 28 ± 4 vo) (for one unit)
Data System				
S-Band Transmitter				
Hi power	1	1.00	18.0	40.00
Low power	2	2.00	36.0	15.00
Slot Antenna	1	.50	64.0	-----
Command System				
VHF Receiver	2	2.00	34.0	5.40
Turnstile Antenna	1	5.00	4.0	------
Memory and Computer	1	4.00	200.0	2.00
Diplexer	1	2.50	6.0	.25
Command Distributor	1	2.50	30.0	.25
Decoder	2	2.00	28.0	.56
Housekeeping System				
VHF Transmitter	1	2.00	26.0	9.00
PCM Network (Commutator, A/D Converter, Clock	1	1.00	21.0	1.00
Miscellaneous		2.50		2.00

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Total Weights: 27.00 pounds

Total Volumes: 467.0 cu. in.
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SPECIFICATIONS ON INSTRUMENTS IN COMMUNICATION SYSTEMS

Hi and Low Power S-Band Transmitters	Source: Vector Inc.
Frequency Range	2200 to 2300 MHz
Frequency Stability	$\pm$ 0.002% of center frequency over temperature range
Modulation Frequency Range	20 Hz to 3.00 MHz
Frequency Response	$\pm$ 1.5 db
Distortion	Less than 1% at $\pm$ 3.00 MHz deviation
Load Impedance	50 ohms
Modulation Type	True FM
Temperature Range	-40 to + 85 $^{\circ}$C, with heat sink, temperature excess of + 85°C
Vibration	20 g peak sine or 20 g rms random, 20 to 2000 cps, 10 millisec along each of 3 mutually perpendicular axes.
Shock	100 g impact shock in each direction of 3 mutually perpendicular axes, each shock 11 milliseconds duration
Acceleration	100 g for 5 minutes in each direction of 3 mutually perpendicular axes.

VHF Receiver	Source: Conic Corp.
Frequency	136-138 MHz
Antenna Impedance	50 ohms
Tuning Accuracy	$\pm$ 0.005%
Noise Figure	6 db maximum
RF bandwidth	200 KC, 600 KC maximum
Maximum RF Input	2 V RMS
Threshold Sensitivity	1 microvolt nominal for correct relay closure 2 microvolt maximum over temperature range
Signal Level Indication	0 to 4 volts DC into 10 K ohm load

VHF Transmitter	Source: Teledyne Telemetry Co.
Modulation Frequency Range	50 Hz to 500 KHz
Frequency Deviation	$\pm$ 225 KHz
Frequency Response	$\pm$ 1.0 db from 100 cps to 100 KC
Frequency Stability	$\pm$ 0.01%
Temperature Limits	- 20 to + 85°C
Vibration	20 g peak, 20 to 2000 cps in any axis, and higher if required
Acceleration	120 g any axis and higher if required
Shock	120 g for 11 millisecond for any axis, higher if required

Diplexer	Source: ESCO
Dimensions	1" x 2" x 3"
Weight	2.5 lbs

PCM Network	Source: Dynatronics
Frame	54 9-bit words: 4 synchronization words 45 8-bit analog words plus parity 5 9-bit discrete words
Analog Inputs	45 0 to + 5.1 volt inputs
Sampling Rate	100 SPS
Encoding Accuracy	$\pm$ 0.15% ($\pm$ 9mv)
Overvoltage	$\pm$ 15 volts causes no crosstalk
Output	
Data	Serial NRZ-L PCM
Impedance	Greater than 100 ohms
Clock	Bit rate clock output at 48.6 KC
Clock Stability	$\pm$ 0.05% over temperature
Input Power	
Voltage	22 to 34 volts
Consumption	1.0 watts independent of input voltage
Overvoltage	To +45 VDC no damage
Temperature limits	-20 to +85'C
Shock	100 g's for 11 minutes
Vibrations	0-500 cy $\pm$ 10 g's peak 500-2000 cy $\pm$ 6 g's peak

Memory and Computer SDP-3 — Source: NASA/Goddard Space Flight Center

Memory size	4K consisting of 16 pages of 256 words each
Bit period	610 ns
Instruction cycle	64 bit times
Power Consumption	2 watts
Weight	4 lbs
Volume	200 cu. in. approx.

Decoder — Source: Conic Corp.

Number of channels	Up to 7 and IRIG channel 1 through 20
Simultaneous usable tones	3
RF deviation required for proper relay operation	30 KC peak (for each channel)
Channel bandwidth (-2db)	± 1% minimum
Adjacent channel rejection	40 db minimum
Noise immunity	Relays will not operate on noise or upon power "turn on" and/or "turn off"
Type of output	One DPDT relay per channel, contact rating is 2 amperes resistive at 28 V DC
Total number or relay contacts	42 for 7 channels
Total number of possible command functions	128 for 7 channels
Temperature limits	-40 to 70°C
Vibration	20 g's sine, 20-2000 cps, 3 axes
Shock	50 g's, 11 milliseconds, 3 axes
Acceleration	100 g's, 3 axes

APPENDIX C
ORBIT ANALYSIS

Terms and Subscripts

V	velocity increment which causes a change in altitude
n	nominal; denotes attempted injection condition (injection condition that is aimed for)
p	perigee condition
a	apogee condition
i	actual injection condition
c	condition corresponding to a circular orbit with h = 313.4 n.m.

<u>Injection Errors</u>

$2\sigma_h$	two standard deviation in altitude at injection
$2\sigma_v$	two standard deviation in velocity at injection
95.4%	probability that the errors will be equal to or less than 2σ
$2\sigma_h$	= 12.8 n.m.
$2\sigma_v$	= 128 ft/sec

Governing Equation: From the energy equation, $\Delta V = 1/4 \sqrt{\mu/r^3}\, \Delta r$

The method of orbital correction, as stated previously in this text, is the use of an apogee-perigee kick motor. All impulses are applied while the satellite is in the spun-up mode. This method is to inject the satellite into an orbit with $h_i = h_a \geq h_c \geq h_p$, and then circularize the orbit. There are only two conditions that <u>must</u> be met during injection for this method to work.

1. $h_i = h_a \geq h_c$
2. V_i small enough so that $h_p \leq h_c$.

For $h_i = h_a \geq h_c$:

$$h_n = h_c + 2\sigma_h = 326.2 \text{ n.m.}$$

$$313.4 \text{ n.m.} \quad h_i \quad 339.0 \text{ n.m.}$$

If h_i = 339.0 n.m., (the worst case)

$$(V_i)_{max} = 24{,}644 \text{ ft/sec to insure } h_p \leq h_c.$$

$$V_n = (V_i)_{max} - 2_v = 24,516 \text{ ft/sec}$$

$$24,644 \text{ ft/sec} \quad V_i \quad 24,388 \text{ ft/sec}$$

These values of h_i and V_i represent the range of possible injection values satisfying the given conditions.

The nominal values are:

$$h_n = 326.2 \text{ n.m.}$$
$$V_n = 24,516 \text{ ft/sec}$$

The greatest altitude deviations are:

$$(h_a)_{max} = 339.0 \text{ n.m.}$$
$$(h_p)_{min} = 85 \text{ n.m.}$$

The actual deviation from the circular orbit must be determined by a tracking station. After the actual orbit is determined, then ΔV_a and ΔV_p can be specified. These in turn, determine the I_a (impulse at apogee) and I_p. These values can easily be translated into rocket burn times. Here is an example of the maximum deviation.

Corresponding to $V_i = 24,388$ ft/sec and $h_i = 313.4$ n.m.,

$$h_a = h_c$$
$$V_i = V_{1c} - 383 \text{ ft/sec}$$
$$\Delta V_a = +\ 383 \text{ ft/sec}$$
$$\Delta V_p = 0$$
$$I_t = I_a + I_p = (383 \text{ ft/sec})\ M$$

For M = 6.2 slugs,

$$(I_t)_{max} = 2380 \text{ lbf-sec}$$

APPENDIX D

STABILITY AND REFERENCE SENSORS

D.1 GRAVITY GRADIENT MOUNTING CONFIGURATION

Figure D.1 shows the mechanism for the storing and deployment of the gravity gradient booms. The motor, drive gears, storage spool, guide rollers, and tip weight or damper are all contained in a canister which can be attached to the main frame of the satellite.

When the booms are to be deployed, the drive motor turns the storage spool, and the tape unwinds. As it proceeds through the guide rollers, it forms a 1/2 inch diameter tube, which is then extended to its full length of 60 feet.

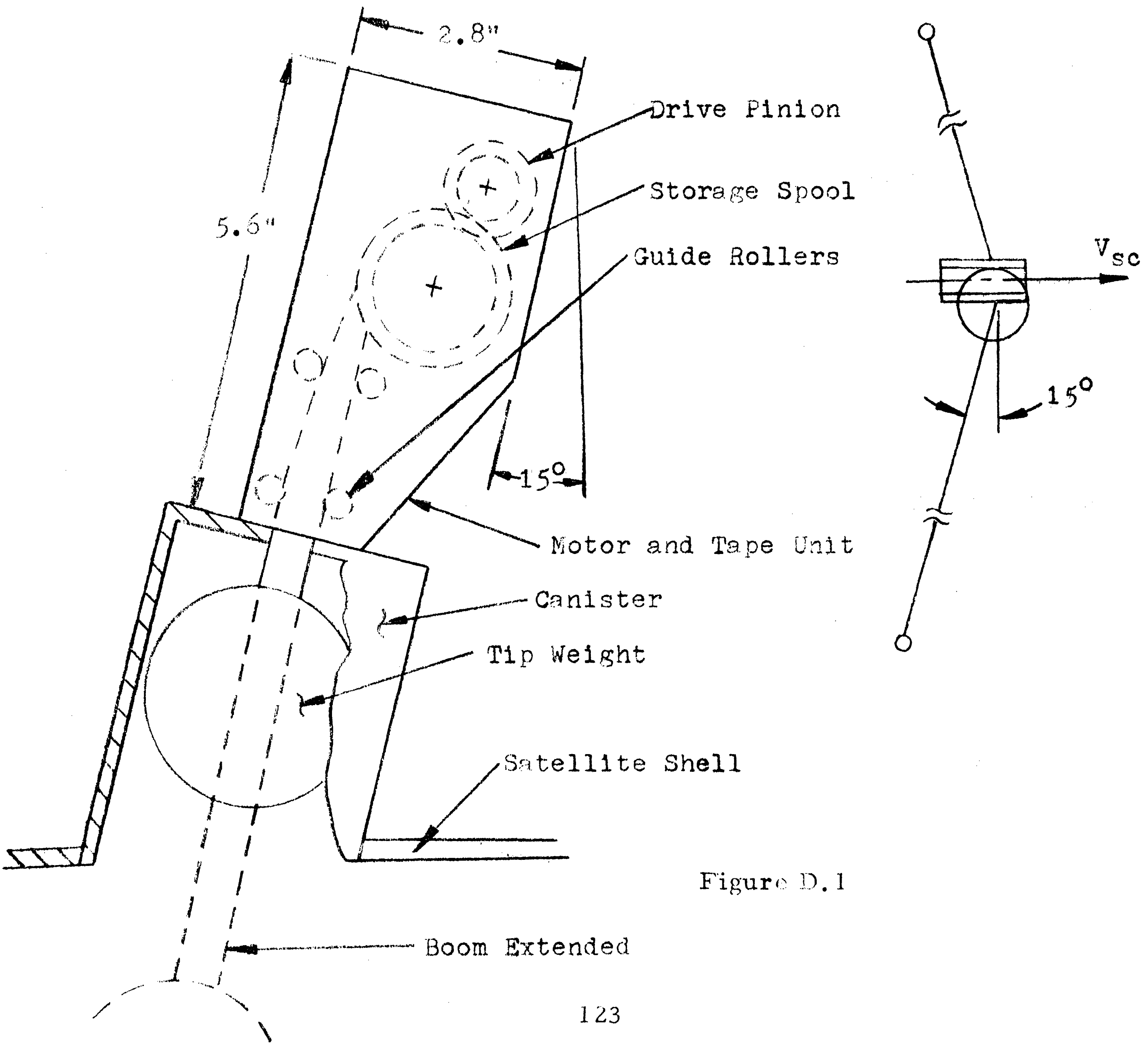
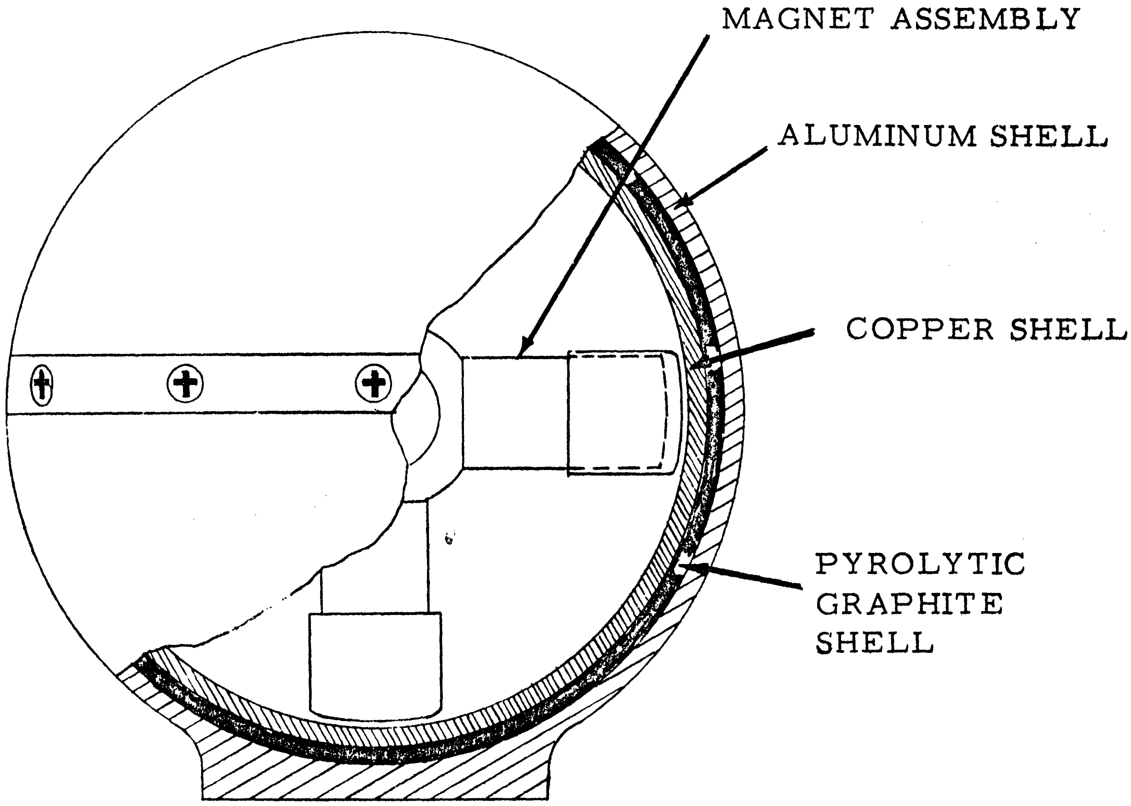


Figure D.1

D. 2 EDDY CURRENT DAMPER MECHANISM

The eddy current damper effectively damps out the unwanted oscillations of the satellite by using a permanent bar magnet which creates an eddy current thru relative motion with a conductive sphere as the magnet follows the earth's magnetic flux lines. This induced current can then be dissipated as heat. A cross section of the damper is shown below.



D. 2 PRINCIPLE OF THE GRAVITY GRADIENT SYSTEM

For a rigid body in a nearly circular earth orbit, the action of centrifugal and gravitational forces tend to align and hold the body such that the axis of the minimum moment of inertia will be normal to the plane of the orbit, the maximum will be along the local vertical, and the intermediate will be along the velocity vector. The gravity gradient system uses this principle by distributing the satellite's weight outward.

Consider a two dimensional "dumbbell" system, where the force of gravity on the tip mass at the end of the dumbbell is given by Newton's Law of Gravitational Attraction, $F = G\, Mm/R^2$, where G is the gravitational constant ($= 1.4/10^{16}$ ft^3/sec^2), M and m are the masses of the earth and tip weights, respectively, and R is the radius from the earth's center to the tip's center. Now, since one mass is further from the earth's center than the other, its R is greater, and therefore, its gravitational force smaller. The centrifugal force on each weight is related by $F_c = mw^2 R$, where w is the angular velocity of the mass about the earth's center. When the dumbbell is disturbed from its local equilibrium position, these forces tend to restore it to this position, although it will not damp out all oscillations present. By adding another boom and using the swept back "Y" configuration, the coupling torques of the roll and yaw directions give a restoring torque for the yaw direction, which otherwise would have none.

In the case of the simple dumbbell system, the general expression for the restoring torque present because of the gravity gradient is

$$T = \frac{1}{2}\left[\frac{k}{R^3}\,(ml^2)\sin 2\theta\right]$$

where k is the gravitational constant times the mass of the earth, m is the tip mass, l is the length of one boom, θ is the angle between the local vertical and the boom, and R is the distance from the center of the earth.

The restoring torques of the roll, pitch, and yaw directions are given by the following equations, see Figure D. 2.

$$T_{yaw} = I_y \theta + \theta_R w_o (I_p - I_R - I_p) + w_o (I_p - I_R) \theta_y$$

$$T_{roll} = I_R \theta_R + \theta_y w_o (I_R + I_y - I_p) + 4w_o^2 (I_p - I_y) \theta_R$$

$$T_{pitch} = I_p \theta_p + 3w_o^2 (I_R - I_y) \theta_p$$

Coupled Torque   Restoring Torque

Notice that there are coupled restoring torques only in the yaw and roll directions, while there are restoring torques in all three directions.

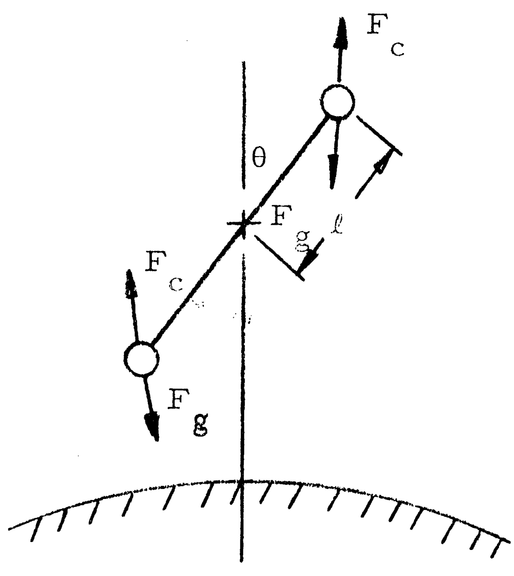


Figure D.2 Restoring Torques

D.4 SAMPLE CALCULATIONS DESPIN

The deviation of the equation used to determine the cable length is based upon equating the initial and final angular momentum and kinetic energy. This equation assumes the following:

1. The system is conservative
2. Effect of gravity is neglected
3. Weights are considered point masses
4. Cables are flexible, unstretchable, and of uniform mass per unit length
5. Motion is two-dimensional
6. Unwound portion of the cable is a straight line

for the length of cable that will wrap around 1 1/2 times the circumference. A radius of 14" is used and will yield a length of 132 inches.

The equation relating the cable length to the end weight is

$$\frac{I}{(L + a)^2 - a^2} = M$$

where

L is the length of the unwound cable in inches
a is the radius of the satellite in inches
I is the spin moment of inertia of the satellite in slug-in^2
M is the end mass of the yo-yo in slugs

For our satellite the L = 132", R = 14"

I has been estimated at 400 slug-in^2

$$M = \frac{I}{(132 + 14)^2 - 14^2}, \quad M = .01894 \text{ slugs}$$

since 1 slug $= \dfrac{1 \text{ lb}}{32.2 \text{ fps}}$ .01894 slug = .6099 pounds

time to despin is given by

$$t = \left[\frac{I + m R^2}{m R^2 R_i^2}\right]^{1/2}$$

for the initial spin rate (R_i) of 150 rpm, or 15.7 rad/sec

$$t = .517 \text{ sec}$$

the deacceleration is, assuming linear deacceleration

$$\frac{15.7 \text{ rad/sec}}{.517 \text{ sec}} = 30.38 \text{ rad/sec}^2.$$

Also assuming the satellite components are mounted at a radius of 10 in from the spin axis of the satellite, the g level encountered is:

$$(-30.38 \text{ rad/sec}^2) \left(\frac{10 \text{ in}}{\text{rad}}\right) \left(\frac{1 \text{ ft}}{12 \text{ in}}\right) \left(\frac{1 \text{ g}}{32.2 \text{ f/sec}^2}\right) = -.784 \text{ g's}$$

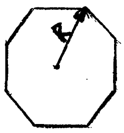
and the maximum tension in the cable is found by using the following:

$$T_{max} = \frac{3 I R_i^2}{4} \left[\frac{3 m}{2 (I + 2 m a^2)}\right]^{1/2}$$

$$T_{max} = 178.5 \text{ pounds.}$$

Moment of Inertia Calculations 200 pound satellite

Model 1



$$I = m \, \frac{1 + 2\sqrt{2}}{6} \, R^4, \quad m = \frac{\text{mass}}{\text{unit area}}$$

$R = 14''$ mass uniform over area equal to 200 lbs cross section area = 498.76 in.

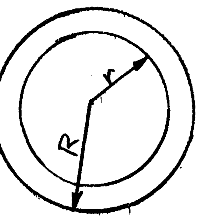
$$I = m\left(\frac{3.828}{6}\right) R^4 = m\,(.638)\,R^4$$

$$I = m \;\; .638\;(38416 \text{ in}^4) \qquad m = \frac{200}{32.2} = 6.211 \text{ slugs}$$

$$I = 152{,}227.933088 \text{ slug-in}^4$$

$$I = 1.5228 \times 10^5 \text{ slug-in}^4 / 498.76 \text{ in}^2 \Rightarrow I = .3053 \times 10^3 \text{ slug-in}^2$$

Model 2 Mass uniformly concentrated over area from 10" to 14" from spin axis.



$$I = \frac{m\pi}{4} \quad (R^4 - r^4) \qquad A = 301.594 \text{ in}^2$$

$R = 14 \qquad r = 10$

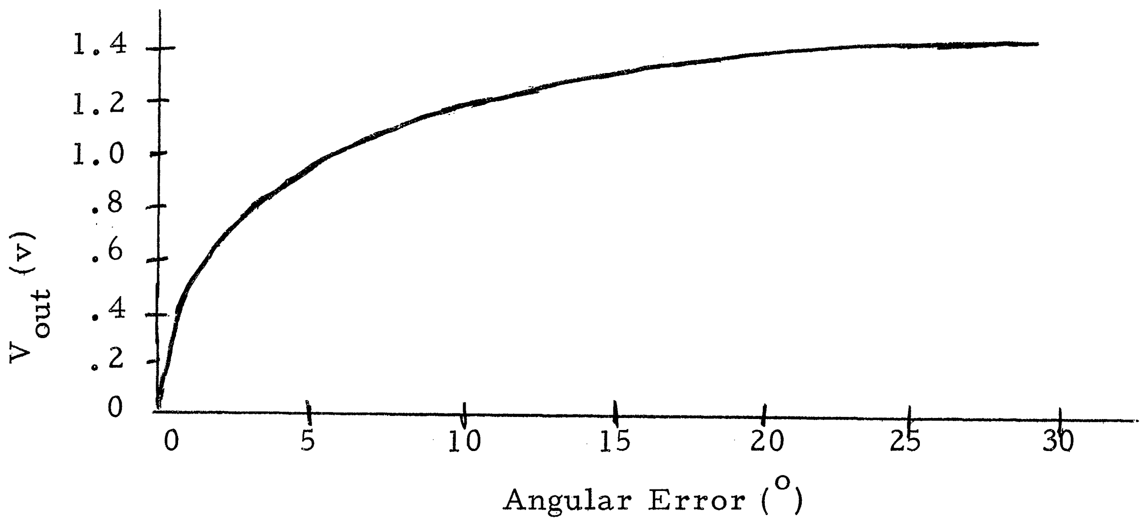
$$R^4 = 38416 \text{ in}^4, \quad r^4 = 10^4 \text{ in}^4 \Rightarrow (3.8416 - 1.0) \times 10^4$$

$$I = 6.211 \text{ slug} \left(.7854 \;(2.8416 \times 10^4)\right) = 13.86165 \times 10^4 \text{ slugs-in}^4$$

$$I = 1.3862 \times 10^5 \text{ slug-in}^4 / 301.594 \text{ in}^2 = 4.5963 \times 10^2 \text{ slug-in}^2$$

Use: 4.0×10^2 slug-in^2 combination of 1 and 2

D.5 ATTITUDE SENSOR SPECIFICATIONS



Null Type Magnetometer Angular Error vs. Output Voltage

Magnetometers

The RAM-3 Heliflux Magnetic Aspect Sensor

$$E_b = 2.4 + .004\,H\cos\phi$$

where E_b is the output voltage

H is the magnetic field strength

Load: 1000K-ohm
Voltage supplied: 6.3 volts
Field range: $\pm$ 600 millioersteds

Total weight: 7 lbs
Total volume: 26.9 in^3
Total power: 2 watts

Has a monaxial flux-gate

Solar Cell

Internal impedance	1000Ω
Pointing accuracy	$\pm$ 2.4 seconds per minute of arc
Degradation	less than 10% per ten years
Rough sensor pointing range	$\alpha \leq \theta \leq 180^{o}$
More sensitive sensor range	$0 \leq \theta \leq \alpha$ where α is $25^{o} \leq \alpha \leq 35^{o}$

APPENDIX E

POWER

E.1 OPERATING TEMPERATURE OF SOLAR PANELS

The number of volts per cell and the number of amps per cell are dependent upon the operating temperature of the paddles. This calculation is for normal sun incidence, the worst condition.

$$S \alpha_{sun} = \epsilon_1 \sigma T_p^4 + \epsilon_2 \sigma T_p^4$$

$$T = \left[\frac{S \alpha_{sun}}{\sigma(\epsilon_1 + \epsilon_2)} \right]^{1/4}$$

S = solar constant = 140 watts/sq. ft.
σ = Stefan-Boltzmann constant = 5.26×10^{-9} watts/sq. ft. K^{o4}
α = solar absorptivity of front surface = .700
ϵ_1 = thermal emissivity of front surface = .835
ϵ_2 = thermal emissivity of back surface = .90

$$T_p = 322^oK = 49^o C$$

E.2 AREA OF SOLAR PANELS

The area was calculated at 61 watts operating at 28 vdc. A packing factor of .85 was used and 15% degradation was allowed for 1 year.

$$\text{at } 49^oC, \ .45 \frac{\text{volts}}{\text{cell}}, \ .128 \frac{\text{amps}}{\text{cell}}$$

$$\frac{1 \text{ cell}}{.45 \text{ volts}} \times 34 \text{ volts} \times \frac{1}{.85} = 89 \text{ cells in series}$$

$$\frac{1 \text{ cell}}{.128 \text{ amps}} \times 1.79 \text{ amps} \times \frac{1}{.85} = 16 \text{ cells in parallel}$$

$$\frac{84 \times 16}{.975} = 1460 \text{ cells for normal incidence}$$

area of each cell = .62 sq. in.

$$\frac{(1460 \text{ cells}) (.62 \text{ sq. in.})}{.85 \left(144 \frac{\text{sq. in.}}{\text{sq. ft.}}\right) \text{ cells}} = 7.4 \text{ sq. ft. total area}$$

for apex angle = 90 degrees, correction factor = 2.44

$$A_t = (7.4)(2.44) \text{ sq. ft.}$$

$$A_t = 18.1 \text{ sq. ft.}$$

25 cells in parallel

140 cells in series

E.3 CALCULATION OF BATTERY SIZE

(based on 28 volts at 1.02 amp for 4 orbits = 6.43 hours)
efficiency of switching regulator = .85

$$\frac{28.0 \text{ volts} \times 1.02 \text{ amp}}{.85} = 33.6 \text{ watts}$$

$$33.6 \text{ watts} \times 6.43 \text{ hours} = 216 \text{ wt-hr}$$

$$\frac{216 \text{ wt-hr}}{28 \text{ cells} \times 1.2 \text{ volt/cell}} = 6.44 \text{ amp-hr}$$

This is the number of amp-hrs that OBSERVER will use initially. Figuring 80% discharge for the first run, the battery size will be 8 amp-hr.

E.4 CHARGING CAPABILITIES

$$\text{discharge time} = \frac{120^\circ}{3.75^\circ/\text{min} \times 60 \text{ min/hr}} = .535 \text{ hr}$$

discharge (eclipse) = 16 wt x .535 hr = 8.55 wt-hr = .306 amp-hr

The energy discharged from the battery equals 75% of the available energy to recharge.

$$\frac{\text{energy taken out}}{\text{energy put in}} = .75$$

excess energy (from solar array) = .922 amp-hr

available energy taken out = .75 x .922 amp-hr

= .69 amp-hr

APPENDIX F

ANALYSIS OF STRUCTURE

Analysis of Structural Member (see Figure F.1)

$I_{x-x} = \sum A_z^2 + \sum I_{cx} - A(z^2) = .01091 \text{ in}^4$

$I_{z-z} = .02485 \text{ in}^4$

$\rho_{x-x} = .2833 \text{ in}$

$\rho_{z-z} = .4424 \text{ in}$

where

$\sum A = .1270$

$\sum A_z = .07144$

$\sum A_z^2 = .04424$

$\sum I_{cx} = .006135$

$\sum A_x^{2\prime} = .0243017$

$\sum I_{cz} = .00055237$

The frame cross-section is divided into components for analysis and thus the $\sum$ is used.

Tension:

Static yield strength of 2024-T3 Alum. = 48.000 lb/in^2
Areas of members A-H = .1270 in^2
(48,000 lb/in^2) (.1270 in^2) = 6096 lb

Compression:

Allowable crippling stress in members A-H = 12,000 lb/in^2 and in member J it is 27,500 lb/in^2. This gives approximately 1530 lb allowable load in shorter members and just a slightly smaller allowable load in the longer members.

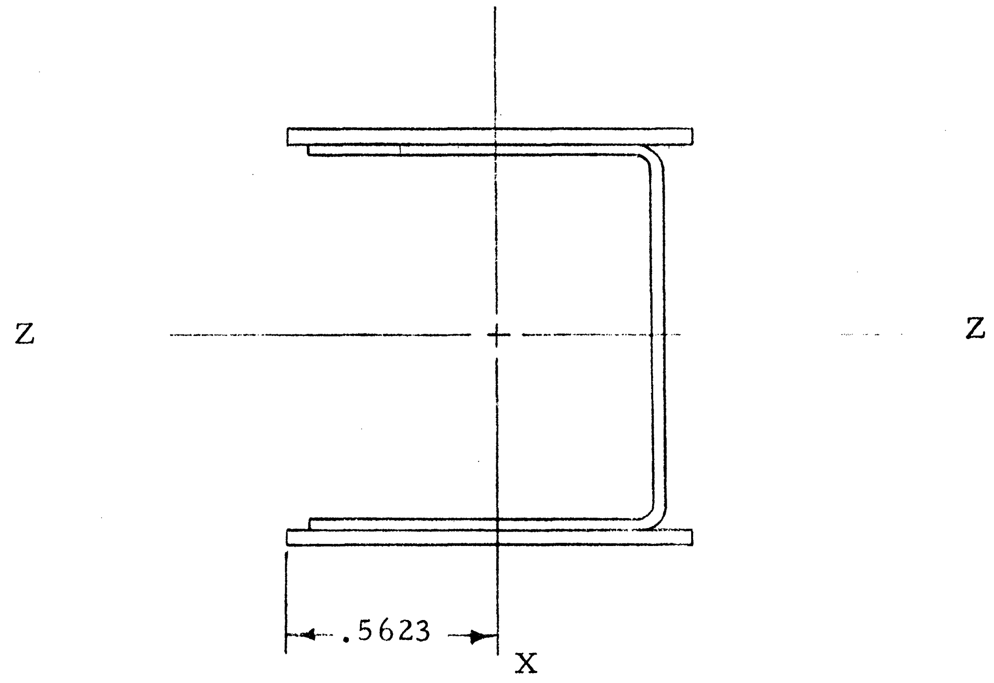
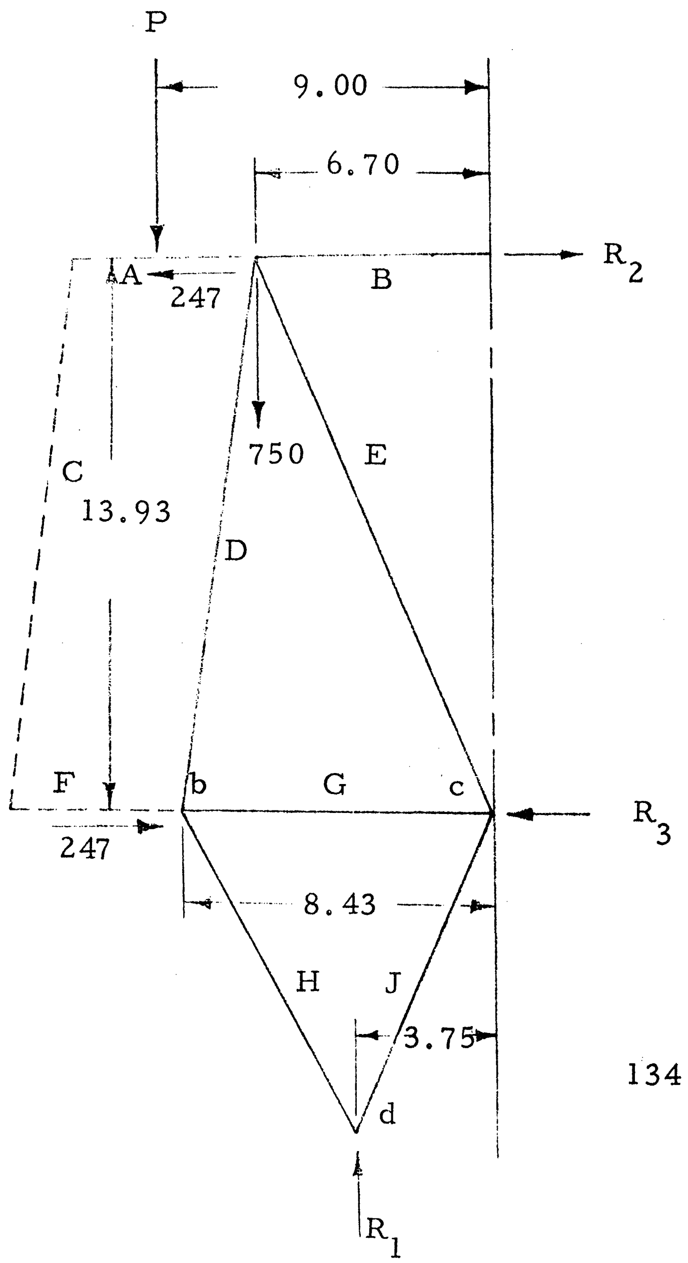


Figure F.1 Cross Section of Structural Member



A force P = 1500 lbs is assumed to be applied at a point 9 inches from the axis of rotation on the top of the structure. It is also assumed that the vertical reaction from the load carried by the shear beam is P/2 at both a and b.

Figure F.2 Quarter-Frame Assembly

Member	Force under 30 G Load	L/ρ	Allowable Load Tension	Allowable Load Compression
A	----	15.22	6096 lb	1530 lb
B	658 lb T	23.66	6096 lb	1460 lb
C	----	49.53	6096 lb	1429 lb
D	97 lb T	49.53	6096 lb	1429 lb
E	931 lb C	54.57	6096 lb	1403 lb
F	----	15.22	6096 lb	1530 lb
G	113 lb T	24.05	6096 lb	1505 lb
H	760 lb C	33.49	6096 lb	1486 lb
J	920 lb C	31.71	8757 lb	4652 lb

Analysis of Quarter-Frame Assembly (see Figure F.2)

Horizontal reactions at A and B found by equating moments:

$$13.93\ X = 1500\ (9-6.7) \text{ or } X \doteq 247 \text{ lb}$$

$R_1 = 1500$ lb

Taking moments about point c, we have:

$$R_2 = \frac{750\ (6.7) + 750\ (8.43) - 1500\ (3.75) + 247\ (13.93)}{13.93} = 658 \text{ lb}$$

$R_3 = 658$ lb also

$\sum F = 0$ at point d

$F_x = 0 \Rightarrow .49H = .41\ J$

$F_y = 0 \Rightarrow .86\ H + .91J = 1500$

J = 920 lb compression

H = 760 lb compression

Similarly, $\sum F = 0$ at point b gives:

D = 97 lb tension
G = 113 lb tension

And $\sum F = 0$ at point a gives:

$E = 931$ lb compression
$B = 663$ lb tension

Testing showed a need for an increase in thickness of channel section in member J to .050" thick. Further testing with components mounted, may indicate increasing the thickness of the channel section in member E.

APPENDIX G

THERMAL CONTROL

The method and details of the thermal control solution shall be discussed in this appendix. Recall the steady-state condition for:

1. thermal radiation
   (energy absorbed) + (internal energy released) = (energy radiated)
2. thermal conduction
   (energy transmitted) = (difference in temperature) / (thermal resistance)

G .1 Parameters for Project OBSERVER

$A_{panel} = 2.38 \text{ ft}^2$	area of one side panel
$A_e = 3.85 \text{ ft}^2$	area of one end panel
$A_t = 26.7 \text{ ft}^2$	total satellite surface area
$A_{p_s} = \lvert 5.74 \cos\theta \rvert + \lvert 3.85 \sin\theta \rvert \text{ ft}^2$	area projected to sun
$A_{p_e} = 5.74 \text{ ft}^2$	area projected to earth
$S = 130 \text{ watts/ft}^2$	solar radiation flux constant
$R = 41.4 \text{ watts/ft}^2$	albedo radiation flux constant
$E = 18.45 \text{ watts/ft}^2$	earth radiation flux constant
$C_p = .23 \text{ B.T.U./lb-}^\circ\text{F}$	specific heat of aluminum
$k = 99 \text{ B.T.U./hr-ft-}^\circ\text{F}$	thermal conductivity
$W = 200 \text{ lb}$	weight of OBSERVER
$\sigma = .1718 \times 10^{-8} \text{ B.T.U./hr-ft}^2\text{-}^\circ\text{R}^4$	Stefan-Boltzman constant
$Q = 16$ watts in shadow $60 \pm 5.8 \cos\theta$ in sunlight	power to be dissipated as heat

(The excess power produced by solar paddles which must be dissipated as heat is 11.6 watts $= \int_{270}^{360} 5.8 \cos\theta \, d\theta + \int_{0}^{90} 5.8 \cos\theta \, d\theta$)

$\theta = 0^{o}$ at equator

90^{o} at South Pole

270^{o} at North Pole

$\alpha = .1$ absorptivity

$\epsilon = .12$ emissivity

G .2.1 Determination of Extreme Temperature (see Figure G.1 and G.2)

The minimum and maximum possible satellite skin temperature can be determined by treating the satellite as if the satellite is fixed at one position in the orbit. Solar radiation is the primary source of absorbed energy, so the hottest possible steady state temperature occurs when the largest area of the satellite is projected toward the sun.

$$A_{p_{s\ max}} = |5.74 \cos \theta|\ ft^2 + |3.85 \sin \theta|\ ft^2 \qquad \theta = 30^{o}$$

$$A_{p_{s\ max}} = 6.88\ ft^2$$

Thus at $\theta = 30^{o}$

$$A_{p_s} S \alpha + A_{p_e} R \alpha + A_{p_e} E \epsilon + Q = A_t \epsilon \sigma T^4$$

$$(6.88\ ft^2)(130\ watts/ft^2)(.1) + (5.74\ ft^2)(41.4\ watts/ft^2)(.1) +$$

$$(5.74\ ft^2)(18.45\ watts/ft^2)(.12) + 63\ watts = (26.7\ ft^2)(.12)(.1718$$

$$x\ 10^{-8}\ BTU/ft^2\text{-hr-}^{o}R^4)\ T^4$$

$$T = 585^{o}R = 125^{o}F$$

Similarly the minimum possible temperature is derived by solving the stady-state equation in the earth's shadow.

$$A_{p_e} E \epsilon + Q = A_t \epsilon \sigma T^4$$

$$(5.74\ ft^2)(18.45\ watts/ft^2)(.12) + (16\ watts) = (26.7\ ft^2)(.12)$$

$$(.1817\ x\ 10^{-8}\ BTU/ft^2\text{-hr-}R)\ T^4$$

$$T = 365^{o}R = -95^{o}F$$

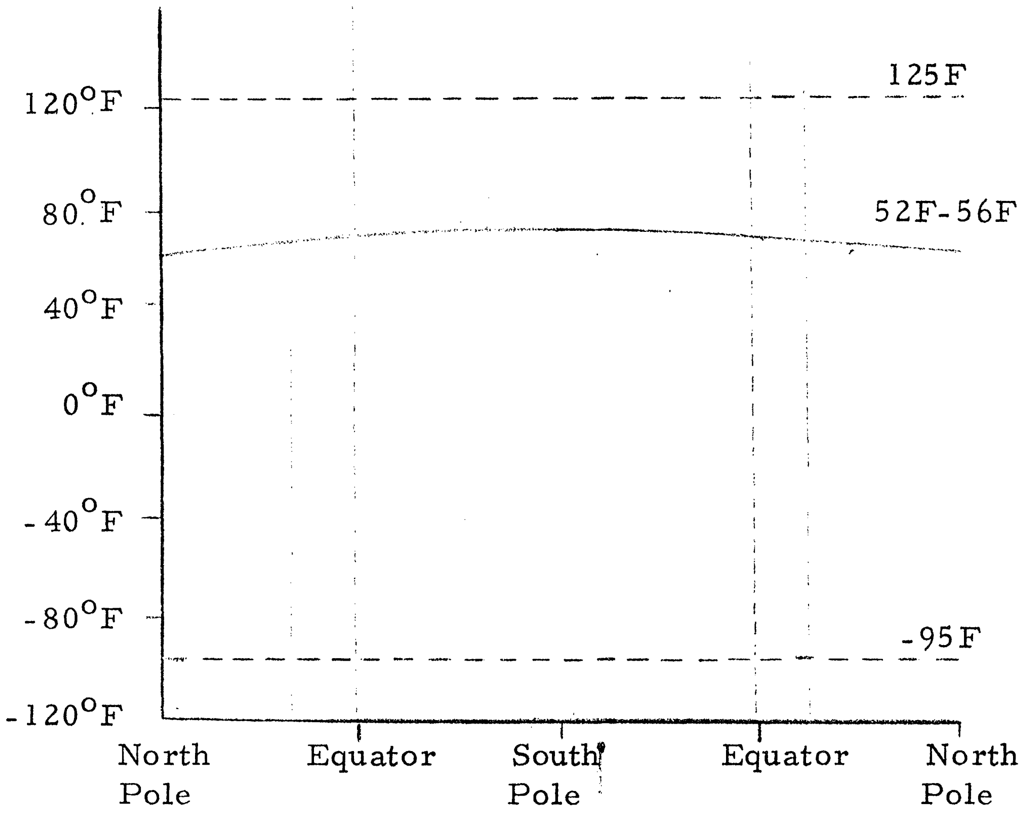


Figure G. 1 Fluctuation of Skin Temperature Relative to the Steady-State Minimum and Maximum Temperatures

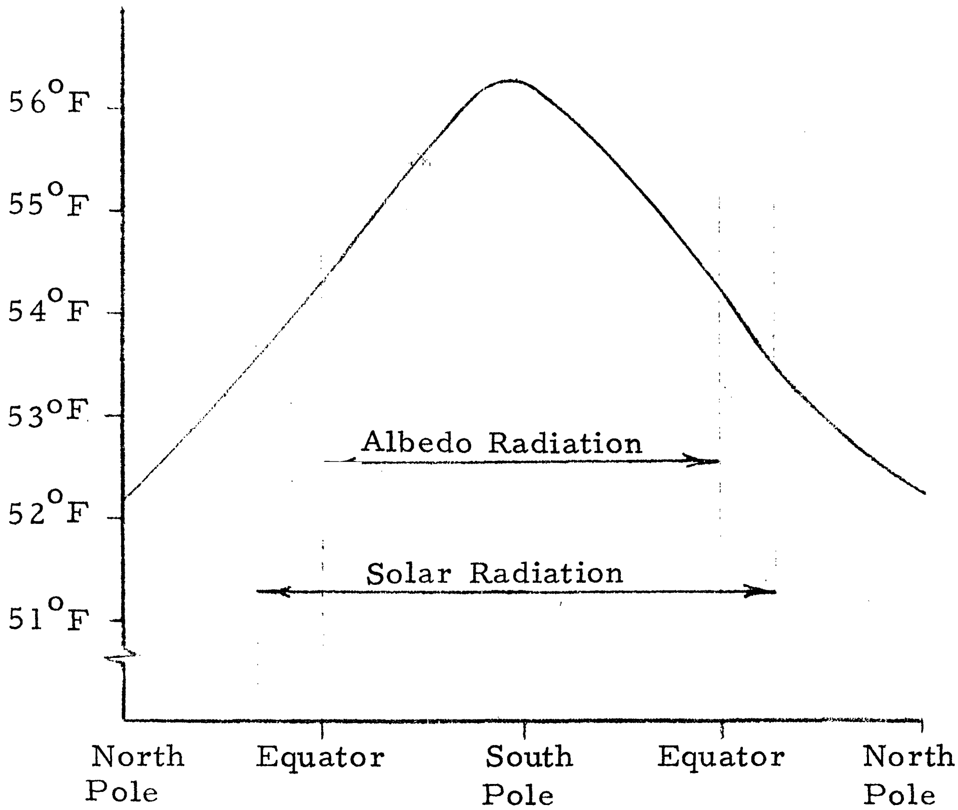


Figure G. 2 Actual Skin Temperature Fluctuation

G.2.2 Average Temperature

After several orbits the temperature of the satellite will stabilize and fluctuate about some average temperature. This average temperature can be calculated for the steady-state condition for one orbit.

$$.63\, PA_{p_{s\,ave}}\, S\,\alpha + .5\, PA_{p_e}\, R\,\alpha + PA_{p_e}\, E\,\epsilon + PQ_{ave} = PA_t\, \epsilon\, \sigma\, T^4_{ave}$$

$$T_{ave} = 514^{o}\ R = 54^{o}F$$

P = time of one orbit = 96.4 min.
Q_{ave} = average power for one orbit = 40 watts
A_{p_s} = average projected area for one orbit = 5.85 ft^2
T_{ave} = average temperature for one orbit

G.2.3 Thermal Time Constant

The thermal time constant indicates how fast the satellite will cool while it is in the earth's shadow. This gives some indication of how close OBSERVER approaches the minimum possible steady state temperature.

$$C_p\, W\, \frac{T_p - T_2}{t_1 - t_2} = A_t\, \epsilon\, \sigma\, T^4$$

where

$$\frac{T_1 - T_2}{t_1 - t_2} = 8.4^{o}/hr$$

$\frac{T_1 - T_2}{t_1 - t_2}$ is not actually a constant but is a function of the skin temperature which is continually fluctuating. However, the low value of the thermal time constant indicates the satellite temperature will decrease approximately 5^{o} while in the earth's shadow. Therefore, the skin temperature is not going to approach the minimum possible temperature, $-95^{o}F$, very closely.

G . 2. 4 Actual Temperature Fluctuation

To determine the actual temperature of the satellite a iterative process must be employed in which the temperature at each point in the orbit is determined. The following relationship must be employed:

$$C_p W \frac{T_1 - T_2}{t_1 - t_2} + A_{p_s} S \alpha + A_{p_e} R \alpha + A_{p_e} E \epsilon + Q - A_t \epsilon \sigma T_2^4 = 0$$

A program was written to calculate the satellite temperature every 5° of orbit (see page 145 of Appendix G). The average temperature which is calculated above was assumed as the temperature of the satellite as it came out of the earth's shadow. The program was used to calculate the temperature at this point after a number of successive orbits when a temperature equilibrium for the satellite was reached. When equilibrium was attained, this temperature as the satellite came out of the shadow was found to be 52.38° F. During the orbit the temperature of the satellite varies from a low of 52.2°F at the North Pole to a high of 56.3°F at 85° south of the equator.

This small temperature fluctuation can be explained by the large heat capacity of the satellite and the low absorptivity and emissivity of the coating.

G. 2. 5 Heat Conduction (see Figures G. 3 and G. 4)

All the above calculations are based on the assumption that the temperature of each side panel is the same. This is not true due to the differences in energy fluxes incident on the different sides. Therefore the incident heat is ideally transferred so that the amount of heat per unit area which must be radiated is equal for each side and end panel.

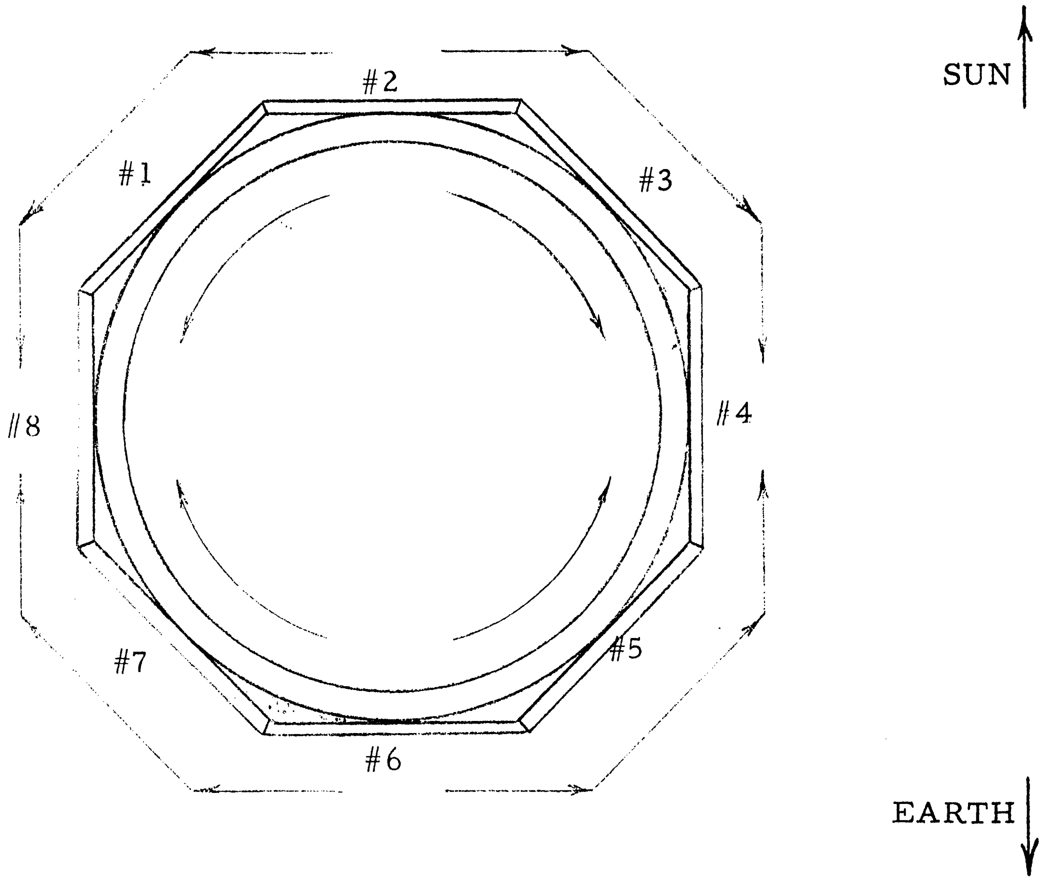


Figure G.3 Heat Conduction Through Outer Structure of Satellite on Sunside of Orbit

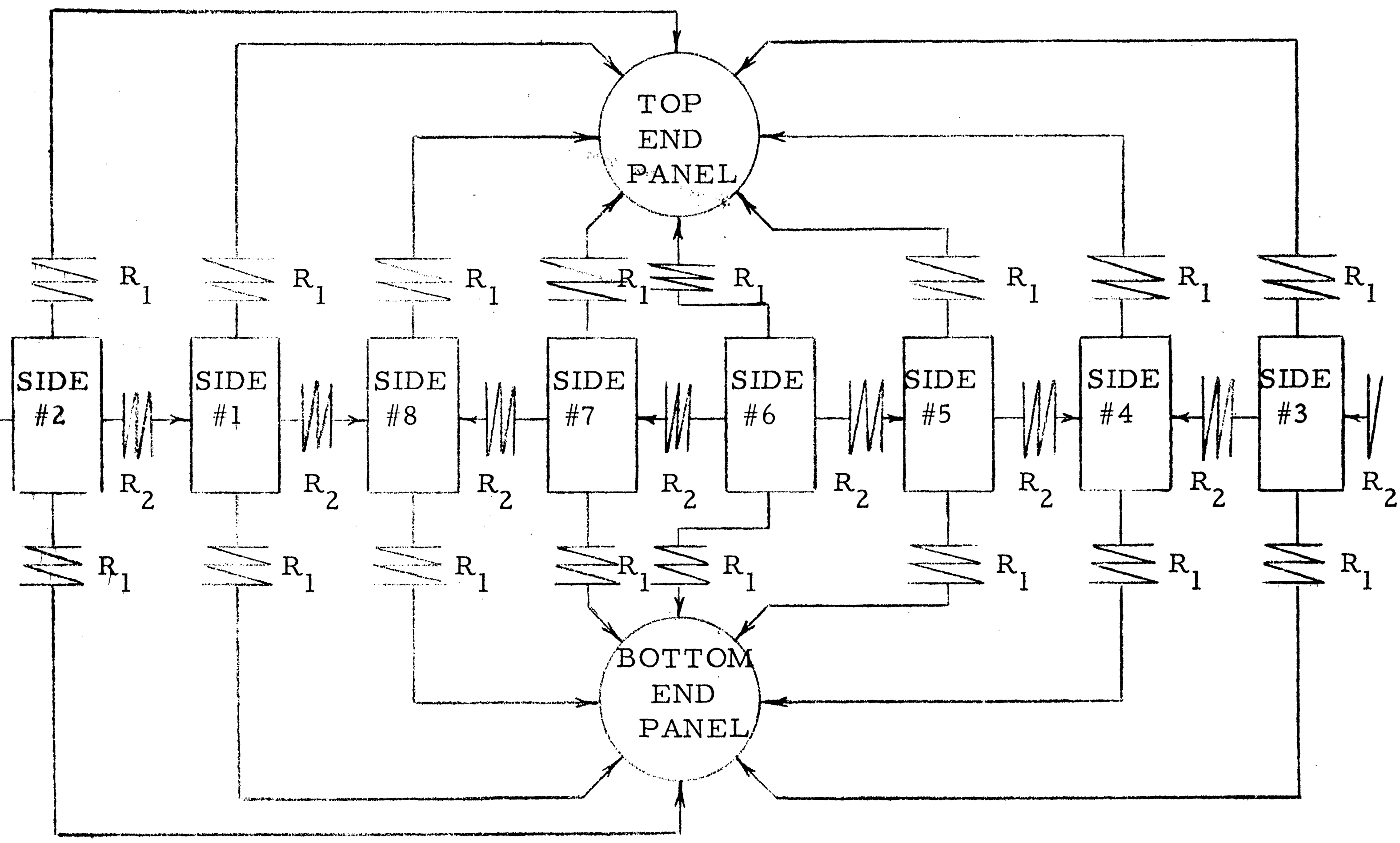


Figure G.4 Schematic of Heat Paths Through Outer Structure of Satellite (Direction of heat flow shown is for position at sunlit equator)

At Equator Side	1&3	2	4&8	5&7	6	Ends
Projected Area	1.61	2.38	---	1.61	2.38	3.85
Energy Flux						
Solar (watts)	20.9	30.9	---	----	----	----
Albedo (watts)	----	----	---	6.65	9.85	----
Earth (watts)	----	----	---	3.56	5.26	----
Internal heat	5.86	5.86	5.86	5.86	5.86	9.45
Total (watts)	20.76	36.8	5.86	16.07	20.97	9.45
steady state condition with no conduction						
Temperature	205°F	259°F	-6°F	125°F	165°F	-11°F
for even heat distribution						
Amount of heat transfer (watts)	-13	-23	7.8	-2.4	-7.3	12.8

The above table shows the incident energy and the amount of heat distribution needed for even heat distribution at the sunlit equator, the worst case. The greatest amount of heat that must be transferred is 23 watts; thus, use of heat pipes would be inefficient. Instead the heat can be conducted through the side panels and the side panel support ring. At this particular point in the orbit there are four effective heat sinks, the two sides facing outerspace (4&8) and the two end panels.

The amount of heat that can be transferred from panel to panel is approximately:

$$q_2 = \frac{T_1 - T_2}{R_2} \quad , \quad R_2 = \frac{Ak}{L}$$

$q_2 = .9\ \Delta T$ watts/°F in each direction

$A = (1/16'')\ (32'') = 2$ sq. in.

$L = 10.7$ inch

$k = 2.4$ watts/°F-inch

The amount of heat that can be transferred from panel to panel through the support ring is approximately:

$$q_3 = \frac{T_1 \quad T_2}{R_3} \qquad R_3 = \frac{(5.9 \text{ inch}^2)\,(2.4 \frac{\text{watts}}{^\circ\text{F-inch}}}{10 \text{ inches}}$$

$$q_3 = .15\ \Delta T \frac{\text{watts}}{^\circ\text{F}} \quad \text{in each direction}$$

And finally the amount of heat that can be transferred from side panel to end panel is approximately:

$$q_1 = \frac{T_1 - T_2}{R_1} \qquad R_1 = \frac{(1.25''^2)\,(2.4 \text{ watts}/^\circ\text{F-inch})}{8 \text{ inches}}$$

$$q_1 = .38\ \Delta T \text{ watts}/^\circ\text{F} \quad \text{in each direction}$$

The ΔT is not the difference of temperatures from the above table but is some smaller difference, for as heat is conducted between two points the two temperatures approach one another. However, as the above calculations indicate only a small temperature difference is needed to transfer enough heat to equalize the heat distribution. (Note in the above calculations, internal heat is assumed to be evenly distributed on the satellite surface.)

The heat conduction within the satellite is best obtained using an analog computer after a detailed schematic is drawn to show all possible heat paths and the thermal resistances of all these paths. Thus the amount of heat dissipated by the internal component is known. The thermal resistance of the structure is determined from the geometry of the structure. This thermal resistance is the sum of resistances of all materials through which the heat is transferred. The temperature of a side panel can be determined. Then the temperature at the energy source can be calculated.

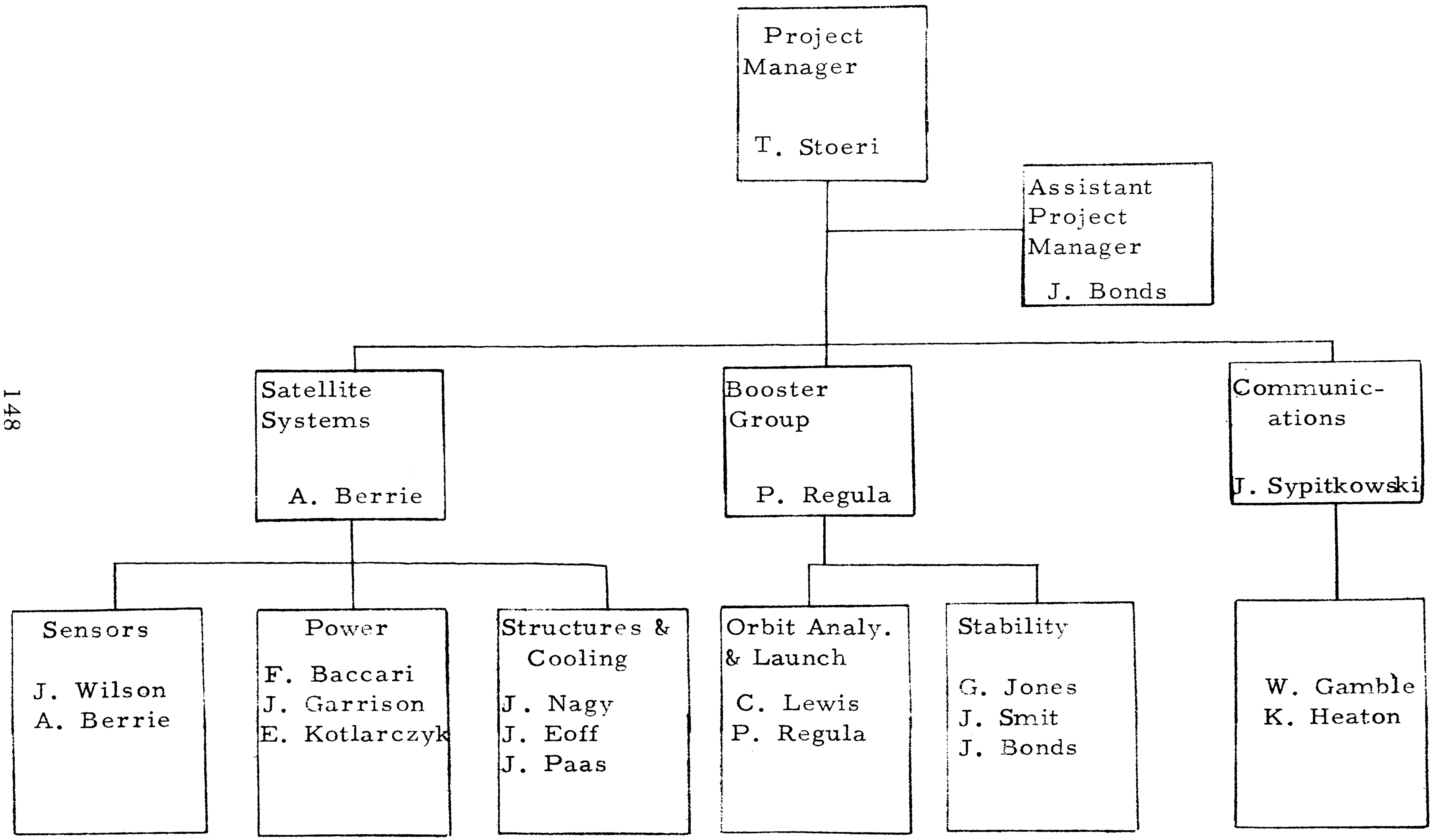
PROGRAM FOR DETERMINING THE ACTUAL TEMPERATURE OF OBSERVER*

```
Compile
C to find the area projected toward the sun
            Dimension C(360)
            C(360) = 5.74
            DO 10 I = 5, 110, 5
            Z = .0174*I
            APS = ABS(5.74*cos (Z)) + ABS(3.85* sin (Z))
            C(I) = APS
            J = 360 - I
            C(J) = APS
   10       Continue
C
C to find Temperature Every 5° of Orbit
            Read, Temp, A, E, Num, Num 2, Num 3, Num 4, Num 5
C 250°  ≤  θ ≤ 270°
            DO 59 K = 250, 270, 5
            Q = 16.0
            T1 = Temp
            T2 = Temp
            PAS = C(K)
            SUM = PAS*130.*A + 105.9*E + Q
52          D = (T1-T2)*602.1 + SUM - 26.7*E*5.0E-10*T2**4
            IF (ABS(D) .LE. Num) Go To 58
            IF (D) 53, 58, 54
53          T2 = T2 - .07
            Go to 52
54          T2 = T2 + .05
            Go to 52
58          Temp = T2
            Print, K, Temp
59          Continue
C
C 275 ≤ θ ≤ 360
            DO 69 KK = 275, 360, 5
            O = .0174*KK
            Q = 60. + 5.8* cos (O)
            T1 = Temp
            T2 = Temp
            PAS = C(KK)
            Sum = PAS*130.*A + 105.9*E + Q + 41.4*5.74*A
62          D = (T1-T2)*602.1 + Sum - 26.7*E*5.E-10*T2**4
            IF (ABS(D) .LE. Num 2) Go to 68
            IF (D) 63, 68, 64
```

```
 63          T2 = T2 - .07
             Go to 62
 64          T2 = T2 + .05
             Go to 62
 68          Temp = T2
             Print, KK, Temp
 69          Continue
C
C 0 ≤ θ ≤ 85
             DO 79 KKK = 5, 85, 5
             O = .0174*KKK
             Q = 60. + 5.8* cos (O)
             T1 = Temp
             T2 = Temp
             PAS = C(KKK)
             Sum = PAS*130.*A + 105.9*E + Q 41.4*5.74*A
 72          D = (t1-T2)*602.1 + Sum - 26.7*E*5.E-10*T2**4
             IF (ABS(D) .LE. Num 3) Go to 78
             IF (D) 73, 78, 74
 73          T2 = T2 - .07
             Go to 72
 74          T2 = T2 + .05
             Go to 72
 78          Temp = T2
             Print, KKK, Temp
 79          Continue
C
C 90 ≤ θ ≤ 110
             DO 89 K2 = 90, 110, 5
             Q = 16.
             T1 = Temp
             T2 = Temp
             PAS = C(K2)
             Sum = PAS*130.*A + 105.9*E + Q
 82          D = (T1-T2)*602.1 + Sum -26.7*E*5.E-10*T2**4
             IF (ABS(D) .LE. Num 4) Go to 88
             IF (D) 83, 88, 84
 83          T2 = T2 -.07
             Go to 82
 84          T2 = T2 + .05
             Go to 82
 88          Temp = T2
             Print, K2, Temp
 89          Continue
C
C 115 ≤ 0 ≤ 245
             DO 99 KDARK = 115, 245, 5
             T1 = Temp
             T2 = Temp
 92          D = (T1-T2)*602.1 + 105.9*E + 16. -26.7*E*5.E-10*T2**4
```

```
            IF (ABS(D) . LE,Num 5) Go to 98
            IF (D) 93, 98, 94
93          T2 = T2 = -.07
            Go to 92
94          T2 = T2 - .05
            Go to 92
98          Temp = T2
            Print, KDARK, Temp
99          Continue
            End
```

* Fortran IV used.



Project OBSERVER Organizational Chart

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