PROJECT STRATUM

A Student Design Project
Department of Aerospace Engineering
University of Michigan
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SUMMARY

STRATUM* is a satellite system which provides support for an experiment designed to measure the atomic and ionic composition, electron density, and intensity of airglow phenomena. Six scientific instruments (a retarding potential mass analyzer, a mass spectrometer, a langmuir probe, a resonance relaxation probe, an impedance ion mass spectrometer, and an airglow photometer) will measure the desired quantities.

The nature of the experiments presents a unique set of problems:

a) prevention of experiment contamination by outgassing
b) solution of the difficult thermal problem introduced by high power dissipation, polar orbit, and stabilization
c) attitude sensing ± 2 degrees throughout entire mission
d) necessary three-axis stabilization ± 20 degrees.

The experiment requires STRATUM to have a one-year lifetime at the lowest possible, near-polar, circular orbit. It also requires the orbit plane to turn 180 degrees with respect to the earth sun line every three months. The optimum orbit satisfying these requirements will have an altitude of 295 nautical miles and an inclination of 82.4 degrees. The Scout vehicle will inject the 235 pound satellite into this orbit.

STRATUM's octagonal body is 32.5" long and 23.8" from corner to corner. A 30" diameter, front-mounted, circular shield protects the experimental instruments from contamination by outgassing. Eight solar paddles, the size of the octagonal faces, deploy in orbit to an angle of 150 degrees with the sides of the spacecraft. This configuration yields large and constant amounts of power regardless of orientation. Annular heat pipes provide the necessary thermal control and eliminate the need for thermal coating, so that the spacecraft body can be used for solar cell mounting. Two swept back gravity gradient booms afford the desired three-axis stabilization. One magnetometer, four sun sensors, and four infra-red earth sensors measure orientation within the accuracy required.

Minitrack is used for commanding and tracking the satellite. Data is transmitted on S-band. The frequencies will be compatible with the ground stations at Fairbanks, Alaska, and St. Johns, Newfoundland.

STRATUM is designed to be launched early in 1970.

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*STRATUM (Stratified Thermosphere Research AT the University of Michigan).

SOMMAIRE

STRATUM,* un petit satellite aéronomique de recherche est projeté pour la mesure de la composition atomique et ionique, la densité électronique et l'intensité du phénomène d'aurore permanente. Six appareils scientifiques sont prévus pour la mesure des valeurs desirées (un analyseur de masse à potentiel retardateur, un spectromètre de masse, une sonde de Langmuir, une sonde à resonance de relaxation, un spectromètre de masse à impédance ionique et un photometre d'aurore permanente).

La nature elle-même des essais offre une classe singulaire des problèmes:

a) L'inhibition de contamination de l'expérience par dégazage;
b) La résolution d'un problème difficile thermique provenant de la dissipation d'haute énergie, de l'orbite polaire et de la stabilisation;
c) Le décèlement de position aux 2 degrés près pendant toute la mission;
d) L'exigence de stabilisation en trois axes aux 20 degrés près.

L'expérience demandera une vie d'un an pour STRATUM en orbite circulaire et quasi-polaire à l'altitude minimum. De plus, le plan orbital faudra tourner tous les trimestres de 180 degrés par rapport à la ligne solaire de terre. L'orbite optimum pour remplir ces conditions aura une altitude de 295 milles marins et une inclinaison de 82,4 degrés. Le satellite pesant 235 livres sera inseré dans l'orbite par le véhicule Scout.

Le corps octogonal de STRATUM est 32.5 pouces de longueur et d'un pas de 23.8 pouces. Un écran circulaire, monté en face, de 30 pouces de diamètre sert pour la protection des appareils de mesure contre la contamination par dégazage. Les huit aubes solaires de grandeur des faces octogonales se déployent en orbite formant un angle de 150 degrés avec les côtés du corps. La stabilisation desiree en trois axes est obtenue par deux longerons à flèche positive de gradient à gravité. Un magnétometre, quatre sondes solaires et quatre sondes infrarouges de terre mesurent l'orientation à la précision près demandée.

Le satellite sera commandé et suivi par un système minitrack. Des bandes de fréquence radar serviront pour la transmission d'informations. Les fréquences seront compatibles avec des stations de terre à Fairbanks, Alaska et à St. Johns en Terre-Neuve.

Le lancement de STRATUM est prévu au debut de 1970.

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*STRATUM (Stratified Thermosphere Research AT the University of Michigan [Recherche de la Thermosphère Stratifiée à l'Université de Michigan]).

1
THE SYSTEM

1.1 PROBLEM DEFINITION

The problem posed to the Project STRATUM group was to create a system to support the aeronomy experiment devised by scientists at The University of Michigan High Altitude Engineering Laboratory. (See Appendix A) Specifically this system must place the experimental apparatus in the proper region of the atmosphere and provide the necessary operating environment. The system must also supply power to the instruments, monitor their operation, store data over most of the orbit, and communicate the desired information to the ground.

Certain restrictions were placed on the system by the nature of the experiment. The experiment is to determine the densities of the various species of ions and atoms, as well as the density of free electrons. A secondary experiment measures the intensity of air glow phenomena. A circular polar orbit, of lowest possible altitude with a one year lifetime is best from the experiment's viewpoint. The experimenter also desires complete coverage of the earth every quarter year (i. e. the plane of the orbit rotates 180^o with respect to the earth-sun line every quarter year). Since part of the experiment measures small densities of atoms and since relatively large amounts of vapor outgas from solar cells, contamination of the experiment by outgassing must be prevented. Three instruments require specific orientation, making three-axis stabilization necessary. These instruments can function with relatively large ($\pm 20^o$) fluctuation in the three axes, but under these circumstances attitude information $\pm$ 2 degrees in all axes must be available for corrections.

These constraints create some difficult problems for the space systems engineer. Because the satellite is stabilized and the polar orbit precesses at a slow rate, the spacecraft has one side which is exposed to the sun while the other is exposed to free space for one and one-half full weeks. Large quantities of power are required resulting in a large amount of internal heat dissipation from the instruments, which further complicates the thermal control problem. The large power requirement will also account for considerable amounts of weight and space. The requirement of low altitude limits the communication time with any particular ground station. Since the experiments produce large amounts of data, storage, processing, and communication are difficult. Altitude information is needed on the dark side of the earth as well as on the light which implies the need for a sophisticated sensing system.

1.2 DESIGN PHILOSOPHY

The underlying principle of the entire design effort was optimization of all the resources. Every possible attempt was made to make the system realistic, reliable, efficient, and inexpensive while still meeting or exceeding the experimenter's requirements.

Each component in the system is state of the art for a program leading to an early 1970 launch date. Simplicity was a governing principle. Virtually all the equipment is not only available, but also flight tested. In cases where this is not true, the equipment is very simple and reliable. This should not be construed to mean that the devices are in any way outdated. Great effort was expended to see that full advantage was taken of the most recent technical advances, while still remaining within the region of reliability.

Reliability is naturally a goal in itself. Certain failures must be anticipated and prepared for if success is to be made certain. Therefore, we made judicious use of redundancy when possible without excessive cost in weight or space. Reliability was also a prime criteria in selecting each particular component.

Efficiency is a must in any space system. Weight and volume must be conserved. Methods, too, must be simple and effective. Considerable effort was extended to make best possible use of weight, space, and power.

Finally, cost was a major concern to the design group. As a general rule we designed for existent hardware to avoid high development costs. Cost was also balanced against effectiveness in many cases.

Clearly the above factors cannot all be maximized in each situation. The prime objective is optimization of the system. This takes precedence over optimization of any subsystem. Similarly optimization of a subsystem takes precedence over optimization of any one of its components.

1.3 PROPOSED SOLUTION TO THE DESIGN PROBLEM

STRATUM is the satellite system design with the above philosophy to meet the above requirements. The Scout vehicle launches STRATUM with an inclination of 82.4 degrees so that the necessary precession is obtained. A predicted altitude of 295 nautical miles will give the satellite a nominal lifetime in excess of one year. The Scout is the least expensive vehicle capable of orbiting the payload. The octagonal cylinder configuration was chosen for the satellite because the octagon's sides are easily removed for servicing, and it almost completely fills the shroud of the Scout vehicle, thus utilizing available space. Eight solar paddles the size of the cylinder's panels deploy in orbit to form an angle of 150^{o} with the spacecraft's sides. These factors were judged to outweigh the price paid in increased frontal area.

Because power was such a concern, we chose annular heat pipes as a solution to the thermal problem, thus freeing the satellite body area which would have been used for thermal coatings for solar cell attachment. Heat pipes are state of the art. Though this is not a common solution, heat pipes have been flown and have performed well.

Two sweptback gravity gradient booms afford the needed three-axis stabilization. No other system was as efficient and reliable as this method. Three types of sensors are necessary to give the desired attitude information in both lightness and dark. These sensors are all passive and relatively light-weight.

Considerable redundancy was built into the communication system because of the high value to cost (in terms of weight and space) ratio. Three tape recorders are incorporated into the system, because the tape recorder is known to be a short lifetime unit. Recorders usually fail in less than a year (attributable to belt wear), and when the first recorder fails, the second replaces it, etc.

Considerable effort was made to find the most highly developed state of the art equipment upon which to base weight and space estimates. Also, conservative weight allowances were made for cables and fasteners. We recognize, however, that not all eventualities can be accounted for in this type of study, and have built a twelve per cent growth allowance into the system. (See Appendix M.)

2

ORBITAL DETERMINATION

2.0 INTRODUCTION

The nature of the experiment and the experimental apparatus constrains the choice of orbits. STRATUM must have a circular orbit of lowest possible altitude yet maintain a one year lifetime. The orbit plane must also turn 180 degrees with respect to the earth-sun line every quarter year. This orbit plane must be positioned to give the spacecraft its maximum amount of time in sunlight at the beginning of the experiment. An early 1970 launch date is desired. Launch vehicle capabilities must also be considered in relationship to the orbit desired.

The Scout was selected as the launch vehicle most compatible with the mission. An altitude of 295 nautical miles satisfies the lifetime constraint and is within the Scout's capabilities. An inclination of 82.4 degrees, yields the proper turning of the orbital plane. A launch window of 2.5 hours will give a minimum of five days fully sunlit orbit.

2.1 LIFETIME AND ALTITUDE

The lifetime calculation (see Appendix C) specifies a minimum altitude of 295 nautical miles for a one-year lifetime. This calculation takes into account variations in atmospheric density due to the time of launch, the effects of inclination of the orbit, and the effect of diurnal bulge.

2.2 INCLINATION

The inclination chosen depends on the precession of the nodes of the orbit plane desired. Precession of the nodes is the mechanism by which the satellite's orbit plane will turn 180 degrees with respect to the earth-sun line every quarter year. The precession rate is the number of degrees the orbit rotates divided by the time expressed in days. Figure 1 shows that the precession rate of STRATUM is opposite to the earth's rotation and is 0.986 degrees per day. The orbit inclination needed to achieve this precession rate is 82.4 degrees. (See Appendix C)

2.3 LAUNCH WINDOW

The power available for instrument checkout and boom deployment is maximum in a fully sunlit orbit. Figure 36 represents the position of the earth with respect to the sun for January 1, 1970. The angle between the earth axis and the earth-sun line decreases from 23 degrees on January 1, to zero degrees during March. Figure 37 , shows, therefore, that STRATUM can be injected into a fully sunlit orbit for an early 1970 launch.

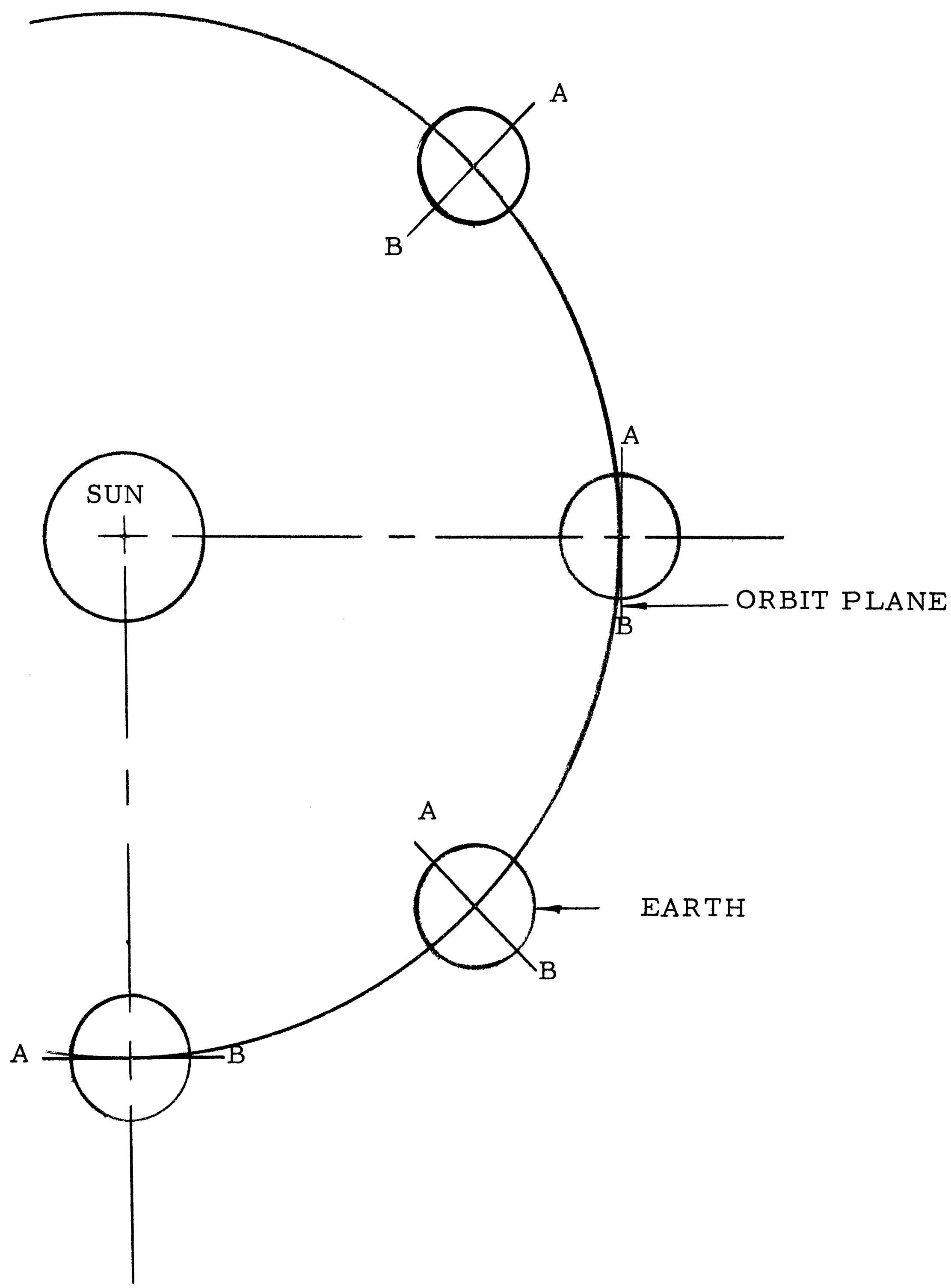


FIGURE 1 . ORBIT PRECESSION

The launch window is constrained by the injection error in inclination and the five days needed for fully sunlit orbit. The three sigma (sigma is the standard deviation) in inclination for the Scout is 0.95 degrees. This corresponds to a longitude change of 0.67 degrees at the earth's equator. (See Appendix C)

From geometrical considerations (see Figure 36), the orbit plane can be inclined 23 degrees from the perpendicular to the earth-sun line and still allow a fully sunlit orbit. This angle is reduced by the injection error of the rocket by approximately 1.4 degrees on each side of the angle. Further, allowing five days in sunlit orbit implies five degrees arc in launch window since the orbit precesses 0.986 deg/day westward. These restrictions reduce the angle of the launch window to 38.2 degrees. The launch window begins 1.1 hours before local sunrise and extends until 1.44 hours after local sunrise. Due to symmetry, the satellite can also be launched one-half day later. The launch window, then, extends from 1.1 hours before sunset to 1.44 hours after sunset. The best launch time is late in the morning window or late in the evening window.

2.4 ERRORS

The accuracy of the orbital analysis depends on the capability of the launch vehicle to inject the spacecraft into the prescribed orbit. The following values are three standard deviations in injection parameters for the Scout

Altitude (circular orbit)	24 miles
Flight Path Angle	0.95
Inclination	0.95
Velocity	200 ft/sec

A possible error of 24 miles in altitude could place the spacecraft in a 274 nautical mile orbit, thereby reducing the lifetime to 215 days. The lifetime required for STRATUM's mission may be insured by aiming for a higher altitude than minimal or by injecting at minimal altitude with a higher velocity. If STRATUM is injected into orbit at an altitude of 295 nm with a velocity greater than circular velocity, the orbit will be elliptical and the injection point will be the perigee. Such an orbit configuration would increase the spacecraft's lifetime. Therefore, if the minimal orbit is attained, STRATUM's lifetime will be at least one year.

3
SPACECRAFT CONFIGURATION AND STRUCTURAL DESIGN

3.0 INTRODUCTION

The satellite configuration and structure are designed, within the constraints of the mission, to be compatible with the other satellite subsystems. The most important considerations are good thermal conductivity to maintain the proper capsule temperatures, and high strength to withstand the loads and vibrations which are sustained during launch as well as during deployment of the paddles and booms. Therefore, the structure incorporates good thermal and strength characteristics at minimum weights.

The experimental package places several environmental constraints upon the configuration: the sensors of the mass spectrometer and the retarding potential mass analyzer must be mounted on a hermetically sealed, equipotential front surface; the hermetical seal overhangs on the front surface to prevent excessive contamination of these two experiments by outgasing from the body of the spacecraft. The front faceplate and several of the side panels can be removed to facilitate prelaunch servicing of the instrument package.

Because of its light weight, high strength, and excellent dynamic damping characteristics, honeycomb sandwich construction is used wherever possible, as in accordance with NASA recommendations. The honeycomb side panels combined with heat pipes provide sufficient conductivity to maintain an acceptable temperature distribution across the satellite.

3.1 DESCRIPTION OF COMPONENT PARTS

3.1.1 Front Faceplate (A)

The honeycomb sandwich of the faceplate extends outward only to the side panels, although the outer skin continues several inches further to form a 30 in diameter environmental shield. The overhang of the outer skin decreases the amount of outgasing contamination which can affect the sensors of the mass spectrometer and the retarding potential mass analyzer. The cell walls and inner skin are perforated to allow outgasing into the spacecraft. By welding the skin to the honeycomb, excellent hermetical sealing is attained. The outer skin is plated with a very thin layer of gold to provide an equipotential front surface for the experiments.

3.1.2 Side Panels (B)

The honeycomb side panels are designed primarily as mounting surfaces for solar cells and as thermal conduction paths for temperature control. The edges of the panels are sealed with a welded U-channel of the same thickness as the skin. The cell walls and inner skin are perforated to allow outgasing into the capsule. The high structural rigidity and excellent damping characteristics of honeycomb minimize the danger of solar cell damage from loads and vibrations. The approximate vibrational modes have been calculated and found to be within the acceptable range.

3.1.3 Heat Pipes (C)

Discussed under thermal control.

3.1.4 Equipment Mounting Platform (D)

The octagonal platform is the principal mounting surface for the instrument package. Standard honeycomb inserts are used wherever necessary to facilitate the use of fasteners. The edges are closed with a welded U-channel of the same thickness as the skin to prevent delamination and crushing. The cell walls and both skins are perforated to prevent cell blow-out in the vacuum of space. For the anticipated test loading, the platform has a safety factor which is 1.94 times the yield strength.

3.1.5 Platform Brackets (E)

The equipment platform is bolted to 1 1/2 x 1 1/2 in right angle platform brackets which are welded to the stringers.

3.1.6 Support Tube Collar (F)

The support tube collar consists of two 1 1/2 x 1 1/2 in right angle rings. The rings are welded to the inside and outside of the upper support tube and are bolted to the equipment platform.

3.1.7 Main Support Tube (G)

This tube is 9 in in diameter and 21.50 in long. During launch, it is the primary means of stress transfer from the capsule to the "E" section of the Scout launch vehicle. All of the major stresses are carried by the equipment platform or the stringers, which are in turn connected directly to the support tube. The tube has a safety factor which is 2.12 times yield strength.

3.1.8 Base Panel Mount (H)

A channel mount is welded to the circumference of the lower support tube to provide a fixed joint between base panels and the tube. The fixed connection allows stress and moment transfer to the support tube.

3.1.9 Base Panels (J)

The trapezoidal base panels (and lower stringers) are fixed to the lower support tube and the peripheral rib. The stringers-base panel configuration carries stresses and moments from the side panels and the solar paddles to the support tube. There are small perforations in the inner and outer panel skins to allow outgasing from the capsule to the wake.

3.1.10 Peripheral Rib (K)

A special peripheral rib connects the side panels to the base panels and provides a solid mounting for the foldout solar paddle hinges and their stops.

3.1.11 Attachment Ring (L)

The attachment ring provides the interface between the support tube and the "E" section of the Scout launch vehicle.

3.1.12 Support Tube Closure (M)

The support tube closure is a 9 in diameter honeycomb sandwich platform with 8 attached 1.0 x 1.0 in right angle brackets. The platform is a mounting surface for batteries and several sensors. It is bolted to the inside of the support tube and can be removed to provide access to the equipment in the tube. Like the base panels, it is perforated to allow outgasing from the capsule to the wake.

3.1.13 Foldout Solar Paddles (N)

Considered under power requirements.

3.1.14 Stringers (P)

The stringers are designed to carry the axial loads from the side panels and the outer portion of the equipment platform. The side panels and base panels are fastened with 3/16 in diameter bolts in between the stringer flanges. Several of the side panels have items mounted to them which requires they be permanently fixed, but most of the panels may be removed by unbolting the faceplate and sliding the panels forward.

3.2 STRUCTURAL ANALYSIS

To optimize the proposed structure, detailed theoretical analysis is required for all components. This necessitates making appropriate simplifying assumptions to facilitate the development of a mathematical model By repeated iterations of the entire structure, the best component part sizes, weights, strengths, and thermal properties can be determined.

A structural model is required for dynamic testing at qualification levels, which are 1.5 times the anticipated flight levels. For the STRATUM payload, the Scout launch vehicle will subject the satellite to the following flight loads:

1) 17 G's axial acceleration
2) 3 G's lateral acceleration
3) 30 G's shock acceleration for 10-15 milliseconds
4) 8 G's vibrational acceleration.

Without accurate physical or mathematical models to work with, it is only possible to analyze with respect to the axial test level. Hence, the structure is designed using the axial test requirement of 25.5 G's as the primary load criterion.

	NAME	REQ'D	WEIGHT	MATERIAL
A	Front faceplate	1	8.4	0.25 in thick welded stainless steel honeycomb sandwich. Skin: 0.02 in thick 17.7 stainless steel. Core: 1/4 in hexcells with 0.001P in thick walls at 6.73 lbs/ft^3; 17.7 PH stainless steel.
B	Side panels	8	6.6	0.25 in thick honeycomb sandwich. Skin: 0.016 in thick 2014-T6 aluminum. Core: 3/16 in hexcells with 0.0015P in thick walls at 4.4 lbs/ft^3; 5052-H39 aluminum. Bond: thermally conductive adhesive film.
C	Heat pipes	2		Considered under thermal control.
D	Equipment mounting platform	1	2.5	1.00 in thick honeycomb sandwich. Skin: 0.020 in thick 2014-T6 aluminum. Core: 1/8 in hexcells with 0.0010P in thick walls at 4.5 lbs/ft^3; 5052-H39 aluminum. Bond: thermally conductive adhesive film.
E	Platform brackets	8	0.3	0.10 in thick 6061-T6 aluminum.
F	Support tube collar	1	1.6	0.10 in thick 6061-T6 aluminum.
G	Main support tube	1	3.0	0.05 in thick 2024-T3 aluminum.
H	Base panel mount	1	0.7	6061-T6 aluminum.
J	Base panels	8	1.3	Same as side panels.
K	Peripheral rib	1	4.9	6061-T6 aluminum.
L	Attachment ring	1	1.0	6061-T6 aluminum.
M	Support tube closure	1	0.3	Same as side panels
N	Fold out solar paddles	8		Considered under power requirements
P	Stringers	8	4.8	6061-T6 aluminum.
R	Miscellaneous		1.0	
	TOTAL WEIGHT		36.4 lbs.	

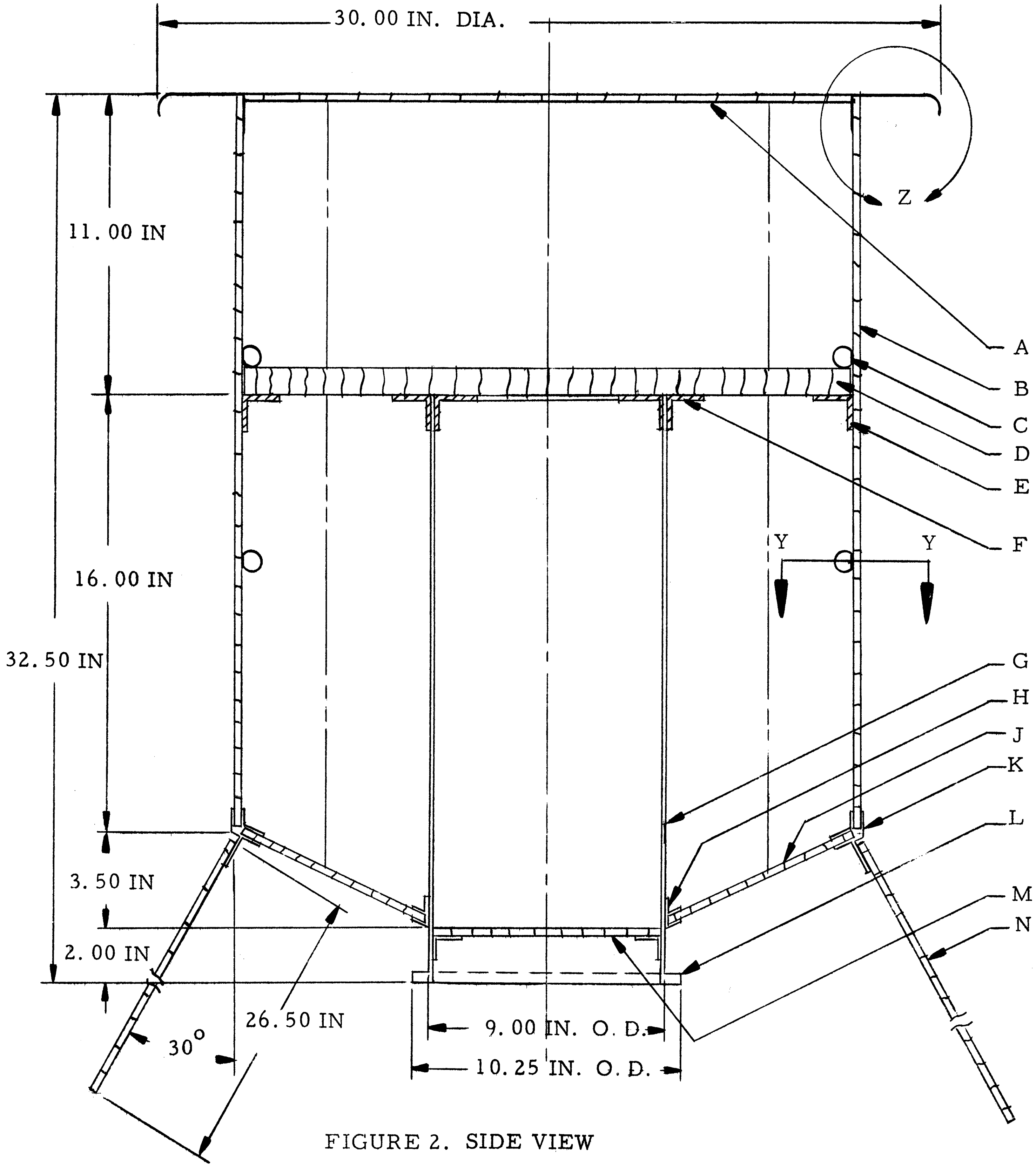


FIGURE 2. SIDE VIEW

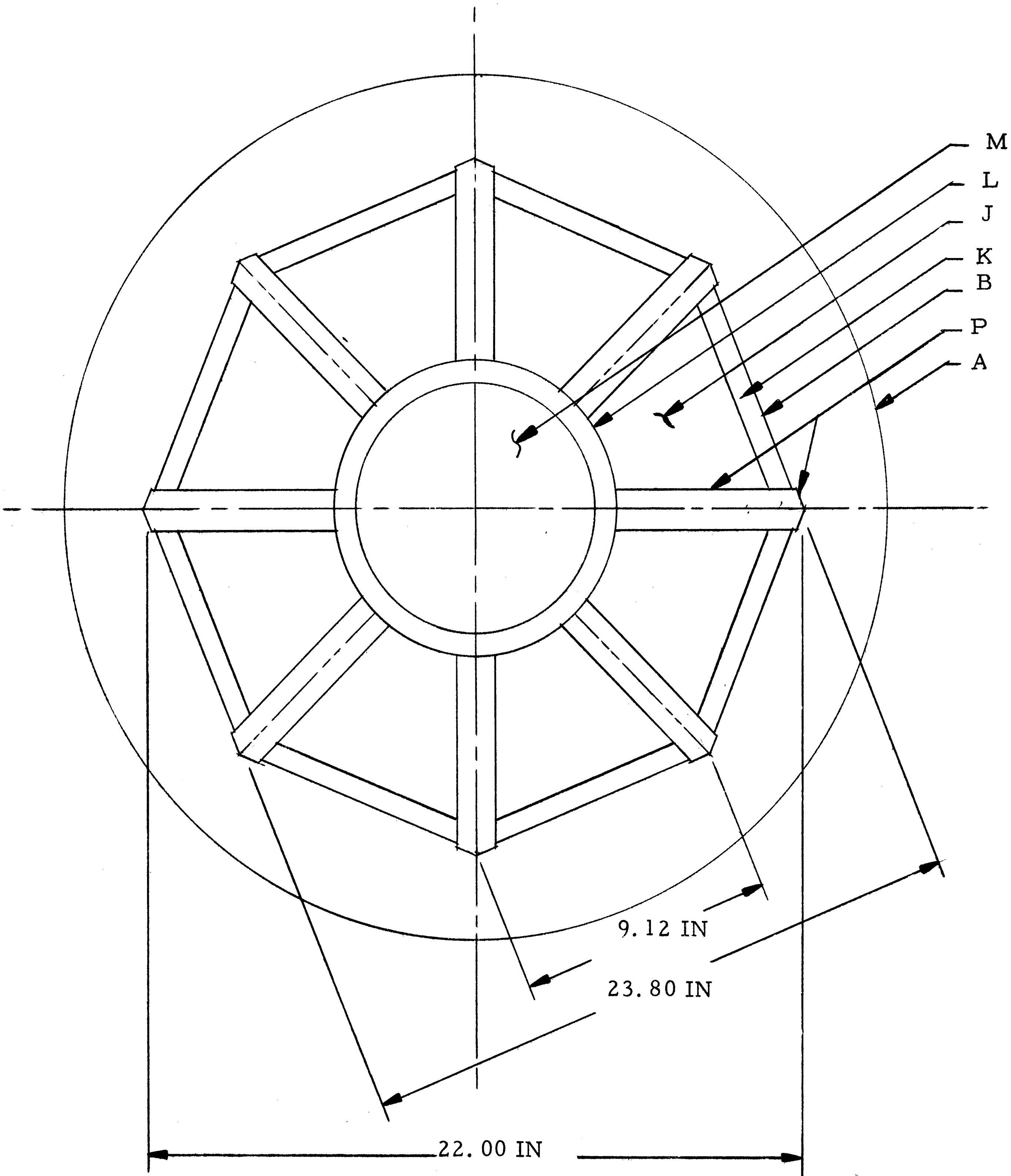
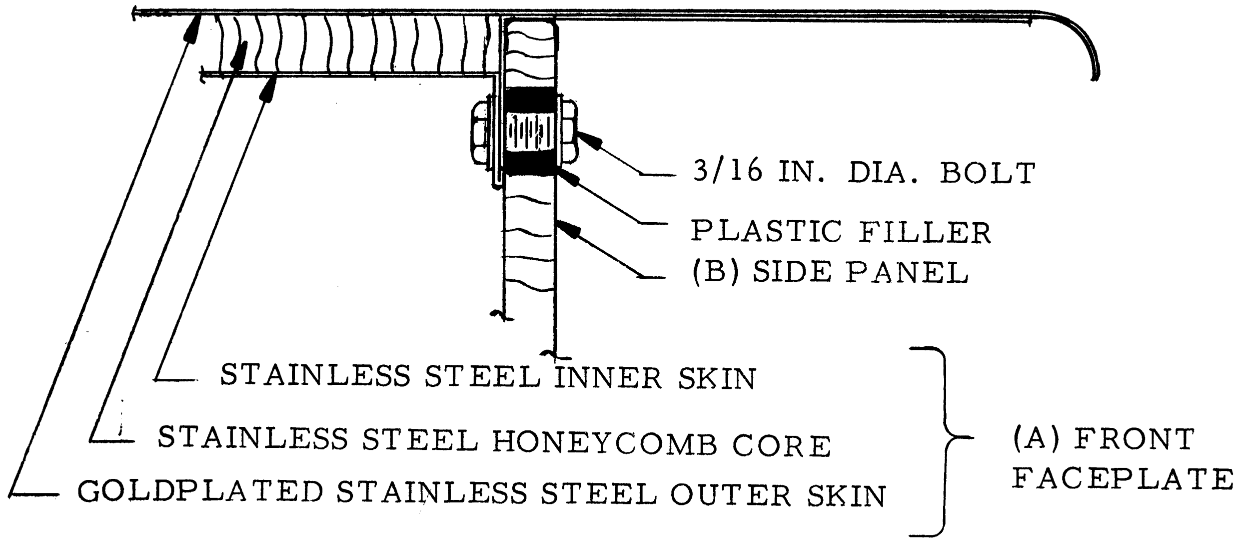
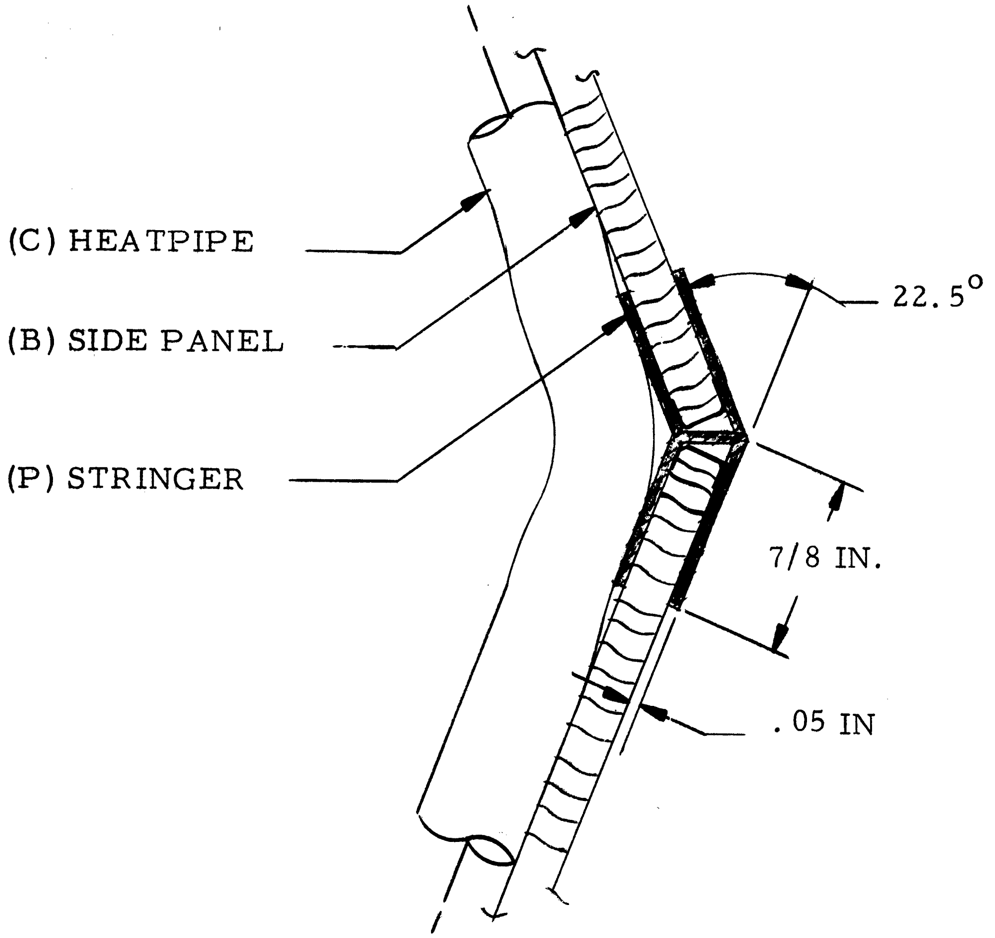


FIGURE 3. REAR VIEW



VIEW IN CIRCLE Z
FIGURE 4



THE SAME FILLER-BOLT COMBINATION AS ILLUSTRATED IN VIEW Z IS USED TO FASTEN THE STRINGERS TO THE SIDE PANELS.

SECTION Y-Y
FIGURE 5

4
STABILIZATION

4.0 INTRODUCTION AND MISSION

There are several known methods of space vehicle attitude control. The choice of a stabilization system depends on the desired orientation of the spacecraft. STRATUM must have its roll axis pointing into its velocity vector, and its yaw axis pointing toward the earth. The vehicle cannot tolerate oscillation greater than $\pm 20^\circ$ in its pitch, roll, or yaw axes.

4.1 GRAVITY GRADIENT SYSTEM

Two types of systems were considered, an active system (requires power), and a passive system (requires no sustained power). A passive system has several advantages:

1) It is usually less expensive
2) It requires no sustained power
3) It has few moving parts, and therefore high reliability
4) It is light in weight.

Its main disadvantage is that it is not as accurate as an active system. However, since the system on STRATUM is required merely to stabilize within $\pm 20^\circ$ in all axes, a passive system will be used. The magnetic stabilization technique was discarded because of its effect on the instruments and experimental package. Also it restricts orientation along the earth's magnetic flux lines and limits the data gathering view fields while subjecting the vehicle to possible tumbling at high inclination orbits as the lines of flux dip to the poles. The gravity gradient stabilization system is probably the most widely used passive system. It was selected because of its simplicity and its negligible magnetic fluctuations.

There are two influences which govern the gravity gradient system: 1) gravitational forces pointing toward the center of the earth at all times, and 2) centrifugal forces which face outward and are parallel to the orbit plane (always perpendicular to the satellite's axis of revolution, but not necessarily going through the earth's center). The stabilizing torques (T) about the spacecraft's three axes are:

$$T_{roll} = 4/2 \ \omega_o^2 \ (I_{pitch} - I_{yaw}) \sin 2\theta_{roll}$$

$$T_{pitch} = 3/2 \ \omega_o^2 \ (I_{roll} - I_{yaw}) \sin 2\theta_{pitch}$$

$$T_{yaw} = 1/2 \ \omega_o^2 \ (I_{pitch} - I_{roll}) \sin 2\theta_{yaw}$$

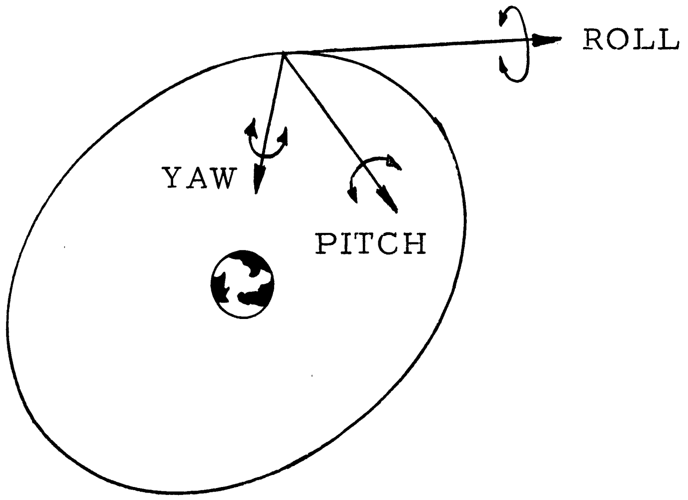


FIGURE 6 . ORIENTATION AXES

where ω_o is the orbital rate (in this case 1.11×10^{-3} rad/sec), I is the mass moment of inertia about its respective axis, and θ is the angular displacement from the preferred orientation. These equations ignore non-linear cross-coupling effects since all mass is contained in one plane. For a detailed report on the principles involved in gravity gradient stabilization see Appendix

4 .2 CONFIGURATION

The configuration chosen for STRATUM is a sweptback horizontal "V" with the spacecraft at the apex of the "V". The booms are each 50 ft long and are sweptback at an angle of 25° from the vertical. Eddy current dampers are placed on each boom tip, and two extenible gravity gradient rods form the legs of the "V".

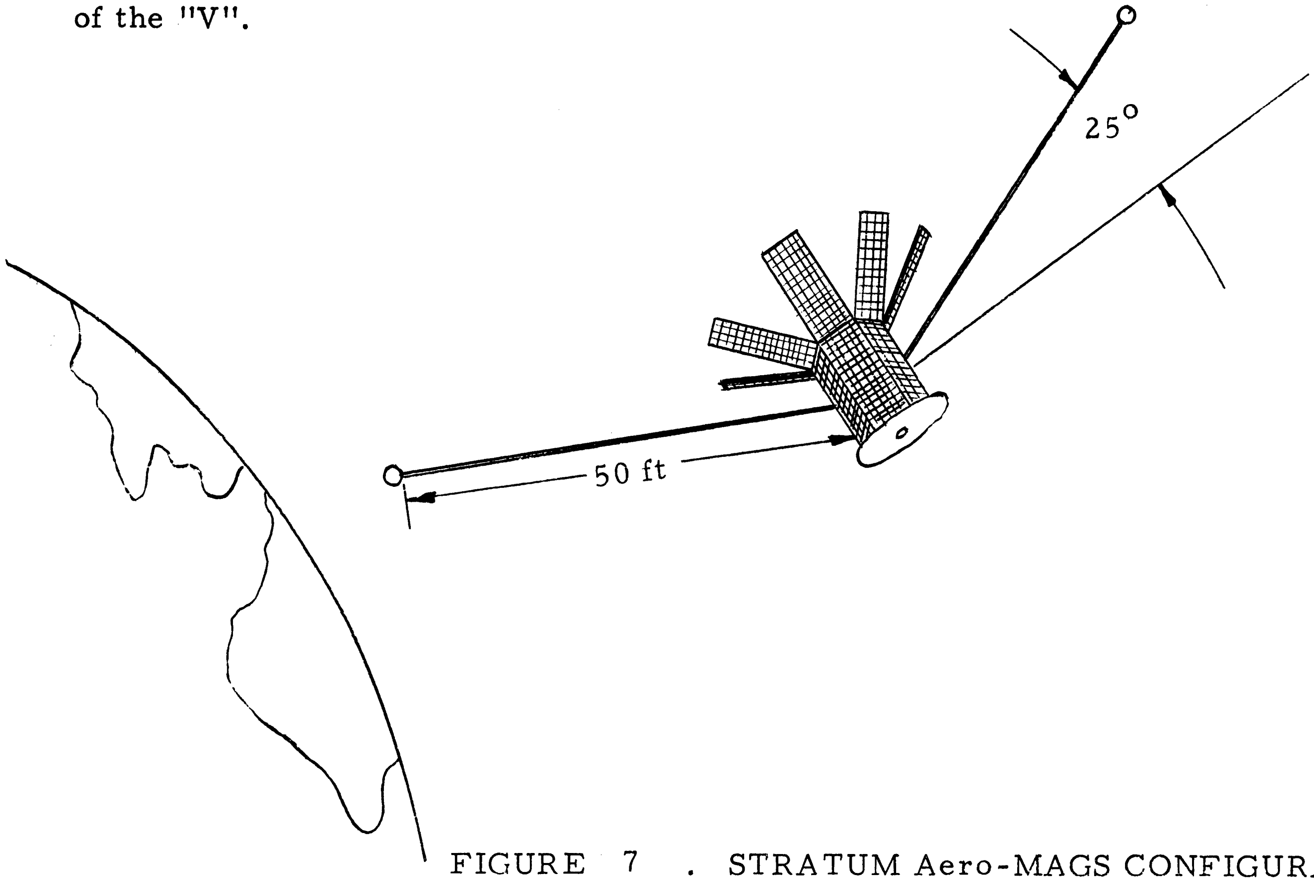


FIGURE 7 . STRATUM Aero-MAGS CONFIGURATION

The "V" configuration provides aerodynamic restoring torques in the pitch and yaw planes. In other words, if the vehicle is disturbed in the yaw plane, tere is enough aerodynamic drag present to return the spacecraft to its preferred orientation. This torque is in phase with the centrifugal torques of the boom. The two torques are additive.

4.3 AERODYNAMIC STABILIZATION

To determine the aerodynamic effects at 295 nautical miles, the Jacchia mathematical equations were consulted. Deriving results from satellite acceleration phenomena at various altitudes and with a constant 20-cm solar flux, the Jacchia Model indicates that the density values during peak solar activity at 295 nautical miles is $\rho = 4 \times 10^{-14}$ slug/ft^3. Accordingly, a dynamic pressure of 1.0×10^{-5} lbf/ft^2 is realized. Aerodynamic torques are, therefore, not negligible and do create another mode of stabilization in both the yaw and pitch planes. These aerodynamic torques do not affect roll phenomena.

4.4 DAMPING

Damping is necessary to provide the decay of initial errors and to prevent unstable build-up of oscillations due to disturbing torques. A gravity gradient vehicle will remain unstable indefinitely if damping is not provided. A totally passive system is achieved when passive damping is utilized in conjunction with the orientation obtained by gravity gradient. In order for the damper to function properly there must be an energy dissipation mechanism (through relative motion between two parts), and a damper reference or anchor point. There are several fields which can be used as an anchor or referencing torque. Magnetic damping is the mode chosen for STRATUM. It is achieved by using a freely moving magnet as the heart of the damper reference mechanism.

4.5 DISTURBING TORQUES

In the swept "V", or Aero-MAGS (Aerodynamic and Magnetically Anchored Gravity System) configuration, the vehicle is subjected to several disturbing torques, tending to divert the spacecraft from the desired orientation. As a result of the orbit chosen, disturbance torques may fall into any of the following categories: solar, thermal bending, magnetic, micrometeoroids, orbit eccentricity, and aerodynamic (including cross-winds at the equator since the earth's atmosphere tends to follow the earth's rotation). For magnetically anchored gravity systems there is also a damper induced error. This results from the variations in direction of the earth's magnetic field relative to the gravitational field as the satellite traverses the orbit.

4.6 GOLDENROD

All the disturbing torques mentioned in the Appendix have been incorporated into a computer program by the General Electric Company. The program, called Goldenrod, is written for magnetically anchored gravity gradient systems. It is capable of accommodating aerodynamic orientation or disturbance torques and hysteresis damping. This program computes the time history of the spacecraft's orientation in yaw, pitch and roll as a result of all torques encountered in orbit. A Goldenrod run for an orbit similar to STRATUM's assured the validity of using an Aero-MAGS system to stabilize the spacecraft. Maximum amplitudes in roll never exceeded ± 7 degrees. Pitch and yaw amplitudes never exceeded ± 9 - 10 degrees. The program run above assumed an orbit with apogee of 800 km and perigee of 300 km, thus providing an eccentricity of 0. 036. For a circular orbit with nominal attitude of 500 km, maximum amplitudes of less than ± 5 degrees in roll and ± 8 degrees in pitch and yaw are expected. These are well within the design constraints imposed on STRATUM (± 20 degrees in all axes).

4.7 BOOM MECHANISM

The choice of extensible booms and erection mechanism is extremely difficult. Examples of this type of equipment include the Westinghouse ATS Boom Deployer, Fairchild-Hiller Model 50GG10M, the General Electric Model GG-C, and the Dehavilland A-18. The DeHavilland A-18 STEM (Storable Tubular Extendible Member) is chosen for two reasons; light weight, and compactness. This "A-18" is a device with the capability for extending, retracting and storing up to 60 feet of 0. 5 inch diameter STEM element, weighing 0. 0124 lb/ft. The tube thickness is 0. 0020 in. This element is stored in flattened form on a drum and is made of specially heat-treated beryllium copper such as will form a tube when extended. (See Figure 9
The erection mechanism weighs approximately 1. 5 lb without the tubular element fitted and its size is approximately 5. 6 inches x 3. 0 inches x 2. 8 inches. The STEM element is stored on a drum which is geared to a motor located in the unit to provide a variety of extension rates. A guidance assembly supports the STEM element during the initial part of the transition from the stored-flattened condition to the fully-formed tubular shape. To keep the erection mechanism size as small as possible the guidance length is kept short. Therefore the root bending strength of the emerging boom is less than that of the fully formed element. The root bending strengths are shown below

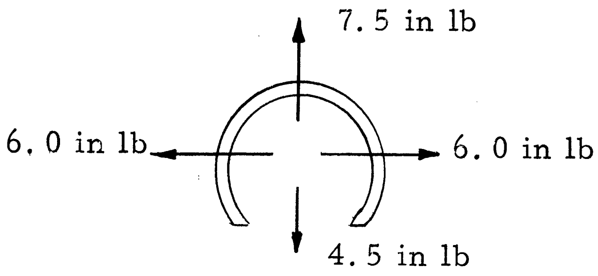


FIGURE 8. STEM Cross-section at Entrance From A-18 Guidance Assembly.

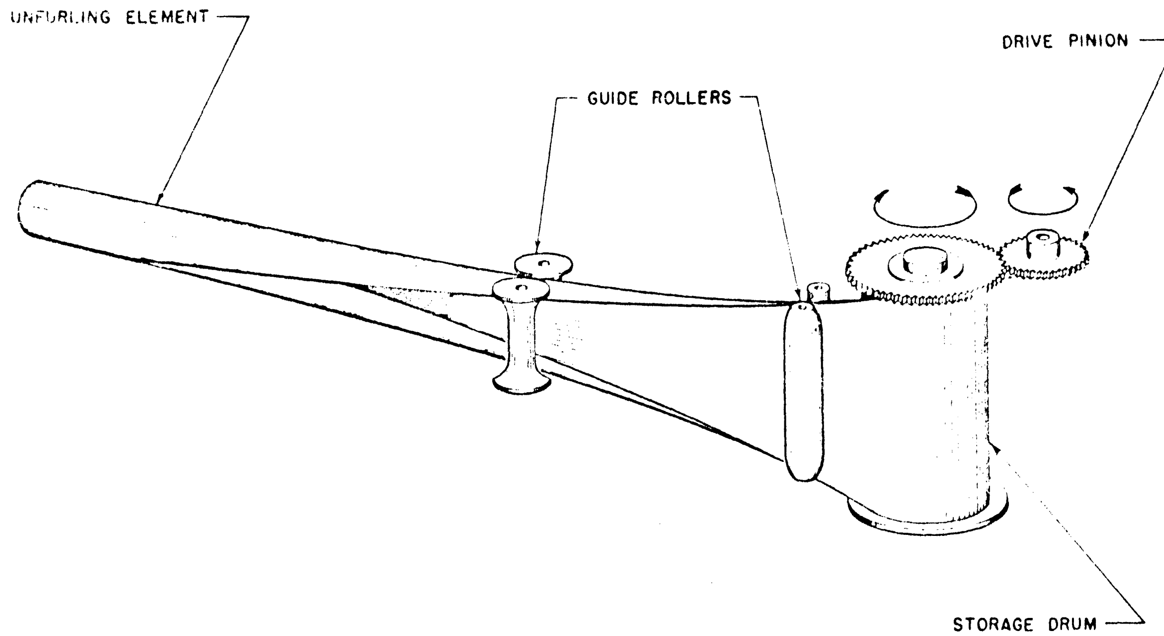


FIGURE 9. THE STEM PRINCIPLE

The edges of the flattened tube are provided with teeth so that the two edges interlock when the rod is extended. The STEM element is also perforated, coated on the outside with silver plating, and coated on the inside with an absorptive material. This decreases the temperature difference across the tube from $\Delta T = 6.6^{o}F$ (for an unplated, unperforated element) to a $\Delta T = 0.53^{o}F$ and decreases the deformation of a 100 ft length of tubing from $\Delta L = 8.25$ ft to $\Delta L = 0.66$ ft. This in turn tends to stabilize the spacecraft in the preferred orientation. Since STRATUM is in a region of significant aerodynamic effects, an extension rate on the order of 1-2 inches/sec is advised so as to alleviate large torque rates.

4 .8 DAMPING MECHANISM

Both types of damping mechanisms considered for STRATUM are manufactured by General Electric. The viscous fluid damper consists of two concentric spheres separated by a viscous fluid. The internal sphere contains a bar magnet which causes the inner sphere to lock onto the lines of force associated with the earth's magnetic field. The viscous fluid dissipates energy when relative motion takes place between the inner and outer sphere. The eddy current damper is similar to the viscous fluid type, having magnetic anchoring and diamagnetic suspension characteristics. In the eddy current damper, instead of an inner sphere assembly, there is simply a magnet assembly; instead of the viscous fluid, there is an electrically conductive spherical shell (made of a special graphite compound) which is part of the outer sphere assembly. The relative motion created by satellite oscillation between the magnet assembly and the electrical conductor generates currents in the conductor which produce forces tending to oppose the motion. As in the viscous fluid type, the damping produced in the eddy current damper is proportional to the relative angular velocity of the inner and outer elements.

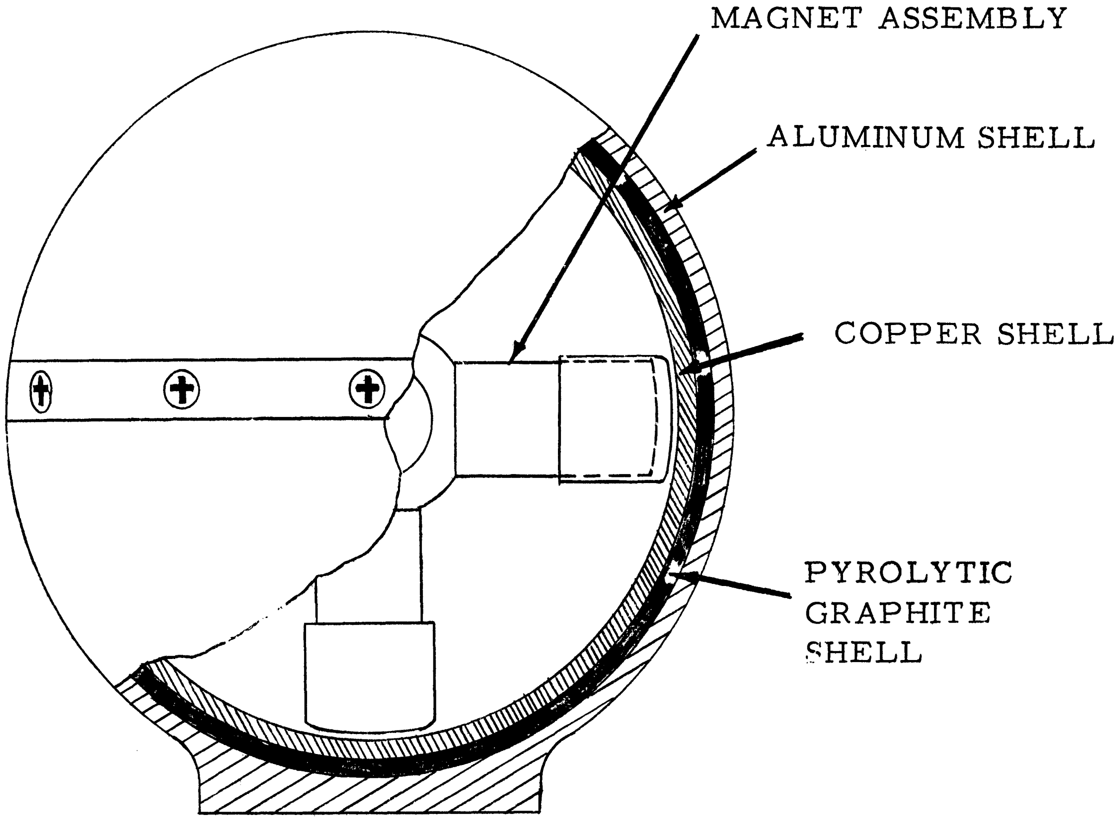


FIGURE 10. MAGNETICALLY ANCHORED EDDY CURRENT DAMPER

Two eddy current dampers are selected for STRATUM because they have smaller magnetic moments, are usually lighter in weight, and are more highly developed than the viscous fluid type. The damping coefficient selected is enough to decay the transient state to steady state in less than 20 orbits and is small enough to cause few inherent errors in crossing the equator. This damper weighs 6.5 lbs and has an outer spherical diameter of 5.3 inches. This specific damper was flown on APL-GEOS/EXPLORER 29 and the NRL-GGSE-4.

5

ATTITUDE SENSING

For the experimental data to be interpreted, attitude information $\pm 2^{o}$ must be available for all three axes in any part of the orbit.

The spacecraft is gravity gradient stabilized with no provision made for attitude correction. Therefore, the sensing devices are passive in nature for simplicity, observing only the attitude of the spacecraft and not giving orientation correction information.

The orbit is polar, and precesses 180^{o} every three months from the earth-sun line. The spacecraft has an ever-changing position with respect to the sun, the earth, and inertial space. This orbit complicates yaw orientation detection but does not affect pitch or roll sensing to a great degree.

Because the orbit is polar and orientation information is needed in the dark as well as in the light, three types of sensors are necessary. Pitch and roll information is obtained primarily by the use of IR earth sensors. Since yaw information is a little harder to detect and evaluate, a magnetometer in conjunction with sun sensors will be used. Combinations of any of these two types would give three axis orientation data in this orbit, but sun sensors will not always provide information due to dark time, and the magnetometer cannot give useable information when going over the poles. Therefore, full time coverage of yaw information in this orbit requires the extra type of sensor. The IR earth sensors for pitch and roll information are always functional.

The magnetometer arrangement consists of three axis magnetic sensing to give a concise magnetic vector. Three single axis magnetic aspect sensors situated on mutually orthogonal axes give the same information as a tri-axial magnetometer unit with less weight and reduced cost. The sensors measure the angle between the sensor axis and the magnetic vector component along that axis.

The IR radiation balance earth sensor system furnishes both pitch and roll information. Two sensors view the horizon on opposite sides of the spacecraft. The sensors detect the horizon using a look-down angle of 20 degrees. The spectral range of this sensor is 14-16 u which was chosen for the purpose of eliminating "cold cloud" inaccuracies, by only being sensitive to H_2O vapor and not CO. Radiations in this spectrum come from very high in the atmosphere, above the high clouds that absorb CO. (Figure 12)

The sun sensor system provides yaw information while the spacecraft is over the polar regions, the regions in which the magnetometers will not give useable information. For the system to have complete coverage for any orbit condition and meet the necessary requirements, it is necessary to utilize four sensors. Two are mounted front and back and two on the right and left sides. The maximum acquisition lies in the horizontal or yaw plane.

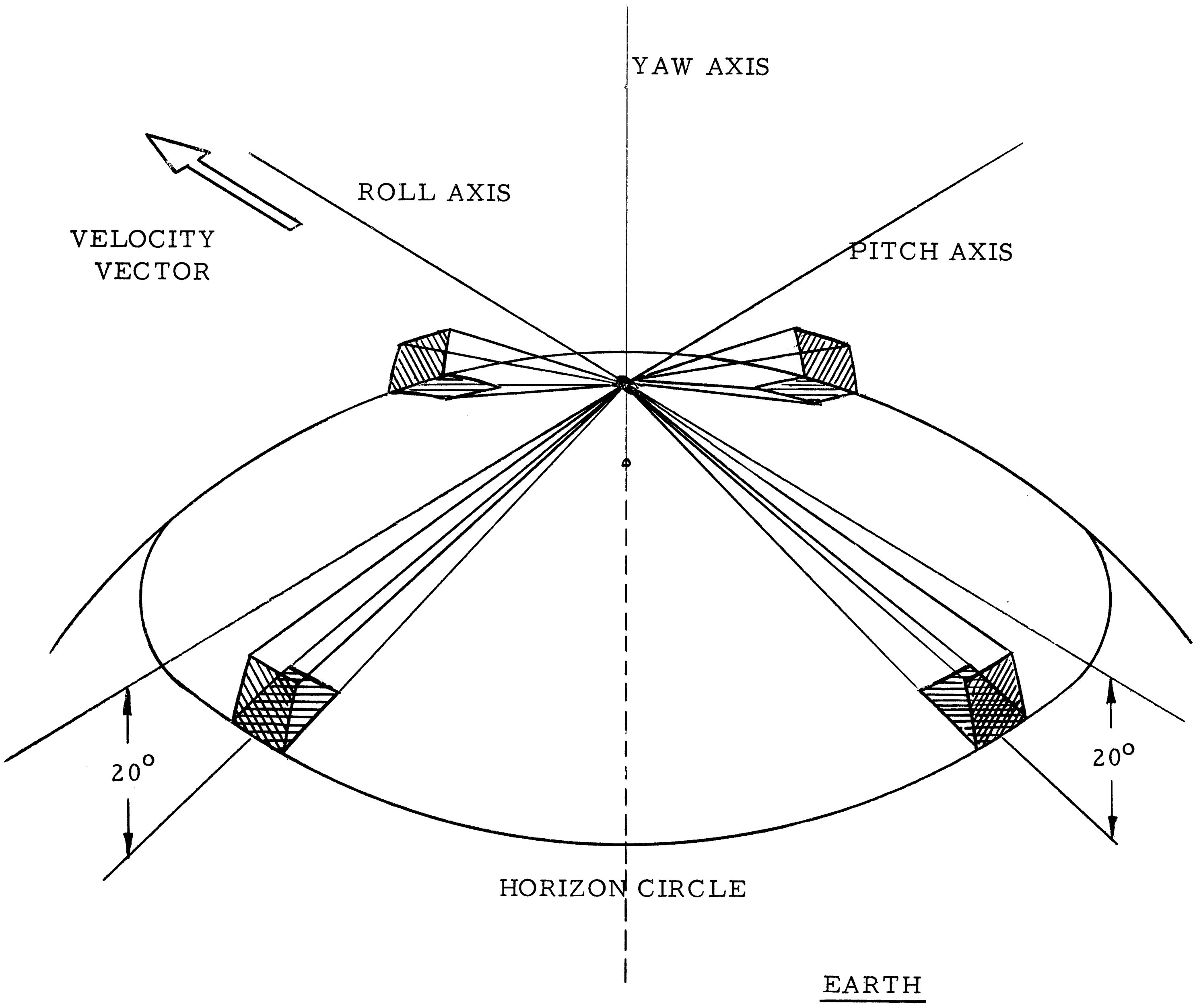


FIGURE 11. IR RADIATION BALANCE EARTH SENSOR
LEVEL ATTITUDE SCAN CONFIGURATION
NOT TO SCALE

This sensor is an original design that was conceived to meet the particular requirements of the orbit and experiments. The sensor was designed to have a 100 degree horizontal and 40 degree vertical acquisition angle. The horizontal acquisition will allow for complete 360 degree coverage around the yaw axis of the spacecraft. The 40 degree vertical acquire allows for sensitivity while in orbits that tend to be contained in the sun-earth plane. This acquisition allows for overlap with the useful magnetometer region.

The sensor geometry is such that a thin beam of light is admitted through a small reticle and is projected onto a curved surface on the interior of the sensor. The surface is photo sensitive and divided into 50 discrete elements. The elements are mono-lithic integrated circuits, or micro circuits. With a view angle of 100 degrees, each discrete element will be sensitive to 2 degrees, within the accuracy that is required. The geometry is such that the incident light is horizontally normal to the curved surface and, therefore, to each element. (Figure 12)

The reticle will have a quartz window which will eliminate degradation and possible outgassing. Each element when struck by a beam of light activates a silicone rectifier (SCR) trigger which then sends a unique and discrete voltage to the telementer system. The activated voltage will follow a step approximation to a ramp function which relates voltage to acquisition angle. Therefore, each discrete voltage implies a unique angle of acquisition. The four sensors utilize a single electronics package since only one sensor will be operating at a given time. Sensor size is 1" x 1/2" x 2 1/2". One watt powers the system and the weight is 2.5 pounds.

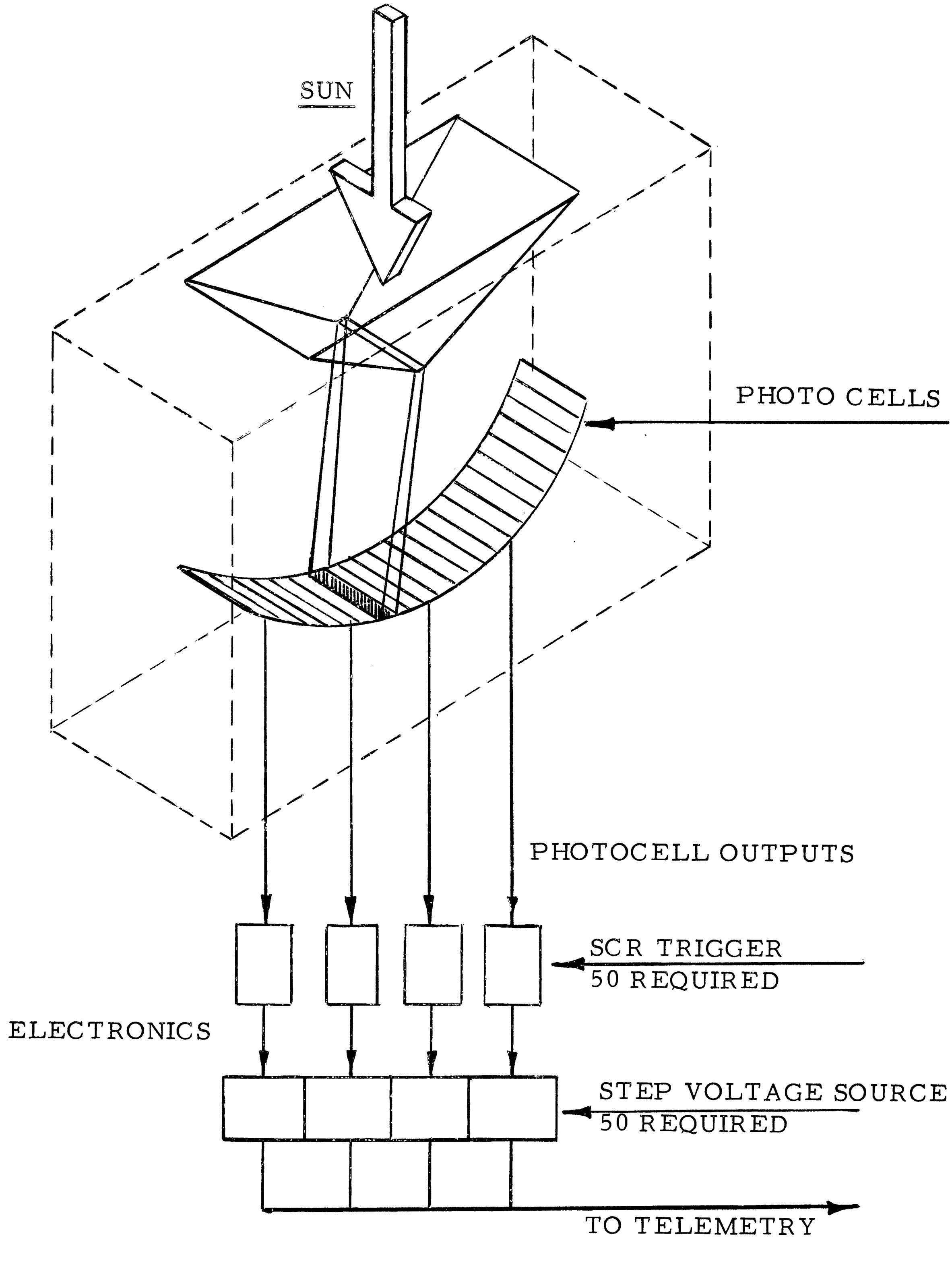


FIGURE 12 . ANGULAR DETERMINATION SUN SENSOR

6
POWER

6.0 INTRODUCTION

The three-axis stabilization requirement in conjunction with the specified polar orbit of Project STRATUM impose unusual problems upon the power system. During the entire twilight orbit, the same panels on the octagonal body are flooded with solar energy, whereas in the high-noon orbit the satellite fluctuates between full sunlight and full shadow once per orbit necessitating energy storage. As the orbit plane precesses, the satellite will be in the dark between zero and thirty-eight percent of the orbital time. In order to use a simplified power system control it is necessary to decrease the dependency of the power system upon the vehicle orientation with respect to the sun.

The simple octagonal shaped satellite with solar cells mounted on its eight sides is inadequate, since the power requirements for the experimental package and equipment are in excess of the power which this configuration can produce. There is also a major drop in the power profile over the poles of the earth. The power profile for such a configuration is very similar in shape to a rectified sine wave with the zero output occurring at the poles. This is in direct conflict with the need to acquire data over the polar regions.

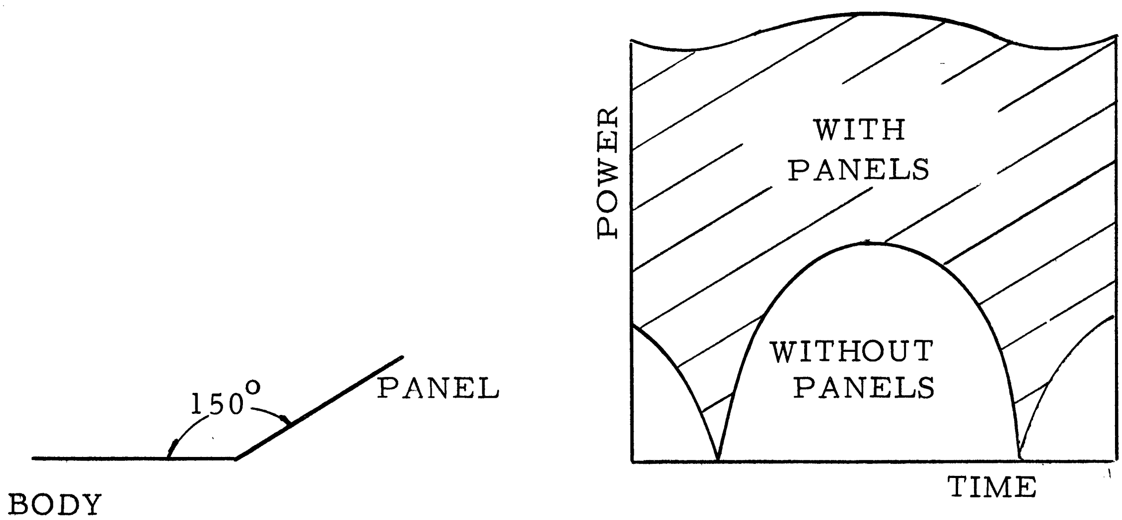


FIGURE 13. PANEL EFFECT ON POWER

In order to decrease the satellite's dependency upon this phenomenon, it is proposed that panels, covered with solar cells on both sides, be deployed in orbit such that each of the eight panels will rotate 150 degrees about its respective hinge and then remain fixed in that position. This configuration almost doubles the peak output of a simple octagon, while smoothing out the power profile curve since the front and rear facing effective solar cells areas are able to generate over the poles 90% of the power available to the satellite when it crosses the equator. (Figure 13)

6.1 SOLAR CELL ARRAY

Silicon N/P solar cells are selected as the primary source of power based upon power requirements, one year lifetime, compatibility with experimental package constraints, weight, and cost. The array consists of 6, 200 2cm x 2cm N/P silicon solar cells (10.5% efficiency at 85^{o}F) which will be assembled for simplified attachment in series-strings on modules to each of the sides of the octogonal main body and on both the inner and outer facing sides of the eight trailing panels. No cells are mounted on the front and rear ends of the main body. Interchangeable rectangular girth panels, each containing a single string of cells, cover the eight sides of the main body as well as both sides of the trailing panels. The girth configuration of the series strings allows minimum power loss due to shadows cast by the front circular skirt and booms, since each module is protected by a bypass diode to prevent major voltage drops during such periods. The diodes will be located in the power control box. This facilitates replacement and improves reliability, since a short circuit in a connecting cable will result in the loss of a single module rather than an entire panel which might occur if the diodes were located at the cell modules.

The weight allocation for the trailing panels may be charged to either the structure or power system. The panels must be sufficiently rigid to prevent cracking of cells and cover glass sheets during launch, yet provide an adequate thermal conduction path to maintain the cells at favorable temperature. The temperature sensitivity of the cells makes the thermal design important. A longitudinally corrugated aluminum honeycomb was selected as able to provide an optimum combination of support and thermal transfer with a minimal weight penalty.

The ratio of active cell areas to the support face is .80. This takes into account the losses associated with module mounting, boom holes, and other surface defects. The ratio of active cell area to paddle area is .85 which includes provisions for mounting and support structure.

N/P solar cells display superior radiation resistance over P/N cells. The radiation degradation over one year in the 295 nautical mile orbit of STRATUM is estimated to be 15%

Ten mil fused silica covers were selected for the cells mounted on the main body and inner face of the trailing panels to reduce the sensitivity of the array to radiation degradation. The insignificant weight penalty imposed by the thick covers is overshadowed by the reduction of manufacturing, assembly, and handling problems. The outer face of each of the trailing panels is covered by a single sheet of glass, rather than the normal 2 x 2 cm cover glass. This unusual design is needed to decrease the outgassing contamination of the solar cells upon the experimental sensors located on the front face of the body. The single sheet design, unlike the individual cover glass arrangement, does not allow the outgassed particles to travel in a direction which might interfere with the sensors. This sheet configuration must be able to survive the launch and deployment environment, as well as handle the problem of varying thermal expansion due to the different materials used. The glass is mounted on rubber grommets, allowing damping of motion as well as sufficient flexibility in the system to cope with the thermal expansion.

Blue-red solar cell filters with a blue cut-on wavelength of .43 microns are used. The filters are placed on the cover glass to reject the solar energy which cannot be efficiently transformed into power, thus minimizing the effects of solar heating and increasing efficiency. The bonding material mounting the modules to the structure must have low outgassing characteristics. A metallic bond on the twenty-four sides would prove too heavy. A thin layer of epoxy-impregnated fiberglass cloth is used as a compromise between weight and outgassing restrictions.

6.2 BATTERIES

Energy storage is required to provide power for the satellite during launch, initial orbit acquisition, periods of darkness (38% in the worst orbit), and any peak power requirement periods. Nickel-cadmium batteries are chosen because they are capable of both cyclic recharge and continuous recharge, have a long life, and have been proven reliable in flight application.

There are two problems inherent in nickel cadmium batteries. The first is weight. The second is the requirement that the average charge voltage must be appreciably higher than the average discharge voltage (1.43 to 1.2 volts) resulting in an 84% efficiency. This is further complicated by the fact that the cells do not exhibit a state-of-charge-signature, which makes regulation difficult. Recently a third electrode battery has been developed by General Electric. This battery indicates when a specified charge has been reached by measuring the internal oxygen buildup during overcharge and terminates or reduces the charge to a rate which the cell can accept. By interspersing four of these cells among the twenty-six required and connecting them such that any two of the four are required to develope the signal to terminate the charge, a possible weight reduction is allowed due to increased efficiency. This also conserves power which might be wasted in sustaining a continuous overcharge, and allows an efficient use of a relatively simple charge control. The batteries will be charged by a current which initially may be as high as one-third the charge capacity and then allowed to decrease to roughly one tenth the charge capacity. The charging voltage will be limited to 38 volts by a Zener diode located in the control box.

6.3 POWER CONTROL BOX

As indicated earlier, the power control box will house the solar panel blocking diodes, a battery charge current regulator with a bypass diode for battery discharging, and a Zener diode to limit the solar panel output voltage to 38 volts (1.5 volts per cell). When the spacecraft is in the earth's shadow, battery power is supplied to the load through a diode that bypasses the current regulator. Pre-launch power is provided through an umbical cord connected to the satellite by a twist-pull, fly away plug. This lead will allow the experiments and equipment to be checked and the battery charged while the vehicle is on the launch pad. Instrumentation for measuring the output bus voltage, total solar panel current, and battery current are included in the power box.

When the battery is discharging, its voltage differs from the bus voltage by the drop across the bypass diode. Thus both of these voltages can normally be inferred. The solar panel voltage, which cannot be measured if it is less than the battery discharge voltage, is also measured since in this case there is no solar panel current. The current in or out of the battery is measured, and will indicate the state-of-health of the batteries when related to the battery discharge voltage and battery temperature histories. The total current to the load is equal to the total solar panel current less the battery charge current. Thus the sequence of instrument operation can be used as a means of enhancing battery operational characteristics.

6.4 SYSTEM OVERVIEW

The power output from the solar panels will not be heavily dependent upon the vehicle orientation due to the trailing panel configuration. The orientation will be continuously changing as the vehicle precesses about the earth, yielding various percentages of light to darkness. The most pronounced effect upon the power system occurs when the orbital plane lies parallel to the solar vector, immersing the vehicle in darkness for 38% of the orbit.

Figure 14 shows typical power profile curves at various orbital orientations with respect to the sun. The crisscrossed area indicates energy supplies to the satellite by the batteries. The unidirectional cross-hatched area represents the energy supplied directly by the solar array, while the remaining clear area represents the energy from the solar array used to recharge the batteries.

It should be noted that the modes shown in the figure are only indicative of the possible utilization of the power generated. The array energy allocated to battery recharge is nominally 50% greater than the energy supplied by the batteries in order to account for the battery charge inefficiency. The sequence of experiments and their time duration may be altered as long as the ratio of clear to crisscrossed area remains 150%.

If failures occur which necessitate real time data transmission, the satellite will be primarily limited to sunlight operation, because the batteries will be unable to supply enough continuous power to the experiments and equipment to allow operation of extended duration. If major data acquisition and transmission is done in the dark segment of the orbit, the batteries will be severely drained, leading to longer recharge time and a rapid decrease in lifetime. In the sunlit portion of the orbit, a majority of the instruments and all necessary equipment will be able to function simultaneously.

6.5 POWER SYSTEM WEIGHT SUMMARY

ITEMS	WEIGHT IN POUNDS	
Solar Cells		
Body Mounted	5.0	
Panel Mounted	10.0	
Honeycomb Panels	4.0	
(including frame, fasteners, etc.)		
		19
Batteries		
26 cells at 6 amp/hr	15.0	
		15
Power Control Box	2.0	
		2
TOTAL		36

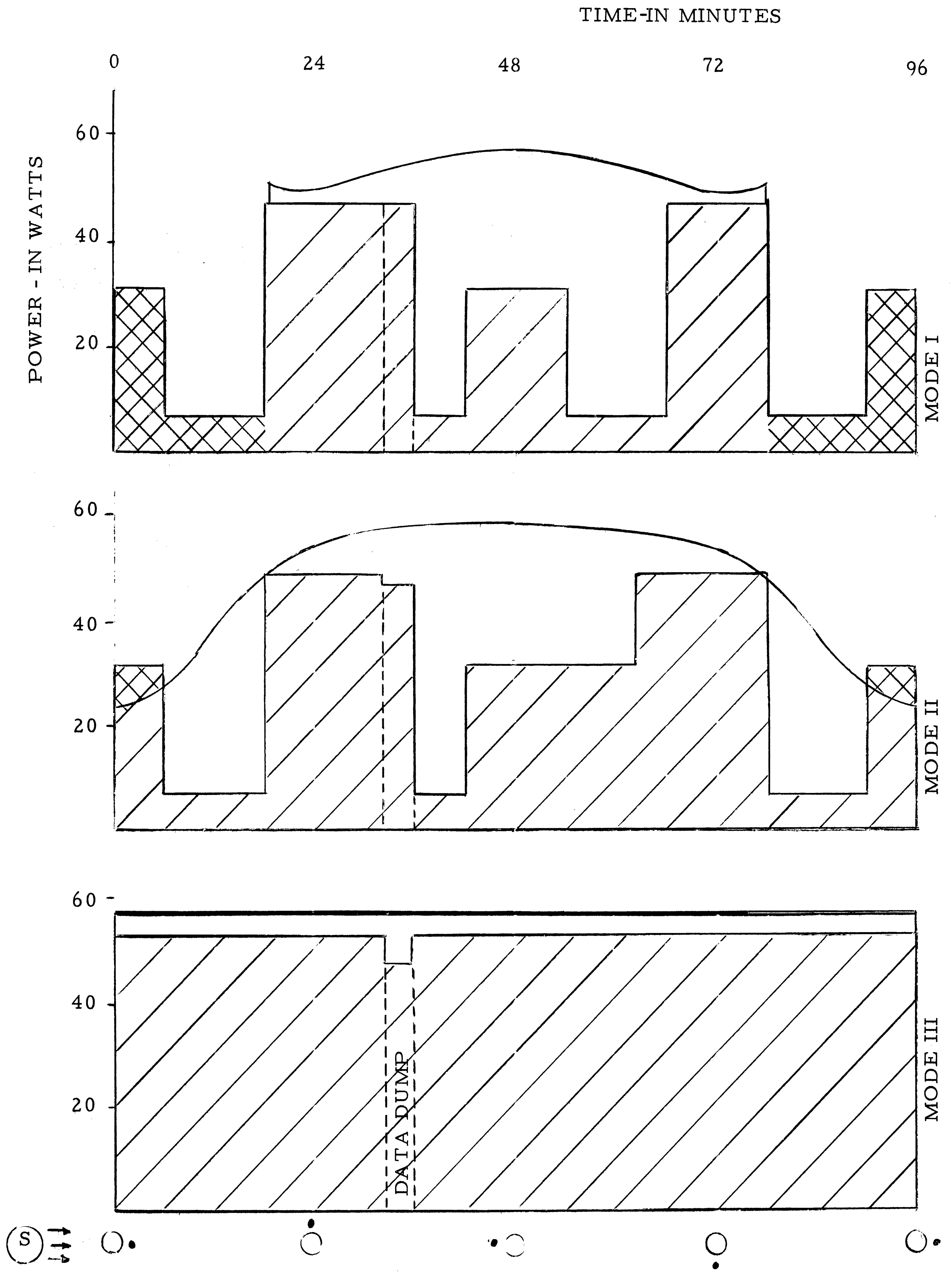


FIGURE 14 . POWER HISTOGRAM

6.6 FOLDOUT SOLAR PANELS

The solar panels which employ a simple hinge-lock are deployed after initial orbit for additional power and simultaneously uncover the folded turnstile command antenna. The hing mechanism includes small coaxial springs to open the panels upon their release and a pin-lock device to keep them fully opened.

The hinge is shown schematically in Figure 15. The detail drawings show the type of spring and the pin housing. Motion of the hinge tab into the slot of the pin housing displaces the spacing pin and permits the spring-loaded pin to enter the tab hole. A soft rubber grommet surrounds the hole to damp oscillations of the panel at the moment of locking. A soft rubber pad also prevents excessive shock in the event of paddle tab overtravel before the pin can lock in the hole securely. It is felt that a mechanism of this simple type should suffice without presenting a serious threat of damaging the foldout panels. Failure of the panels to lock is unlikely but would not be harmful since the hinge springs continue to exert torque. Deployment is automatic upon release, which is provided by a pyrotechnique device.

6.7 SOLAR PANELS WEIGHT ESTIMATE

Hinge tabs	.1 each	.8
Pin housings	.1 each	.8
Cables		None
Fasteners		.6
	Total	2.2 lbs

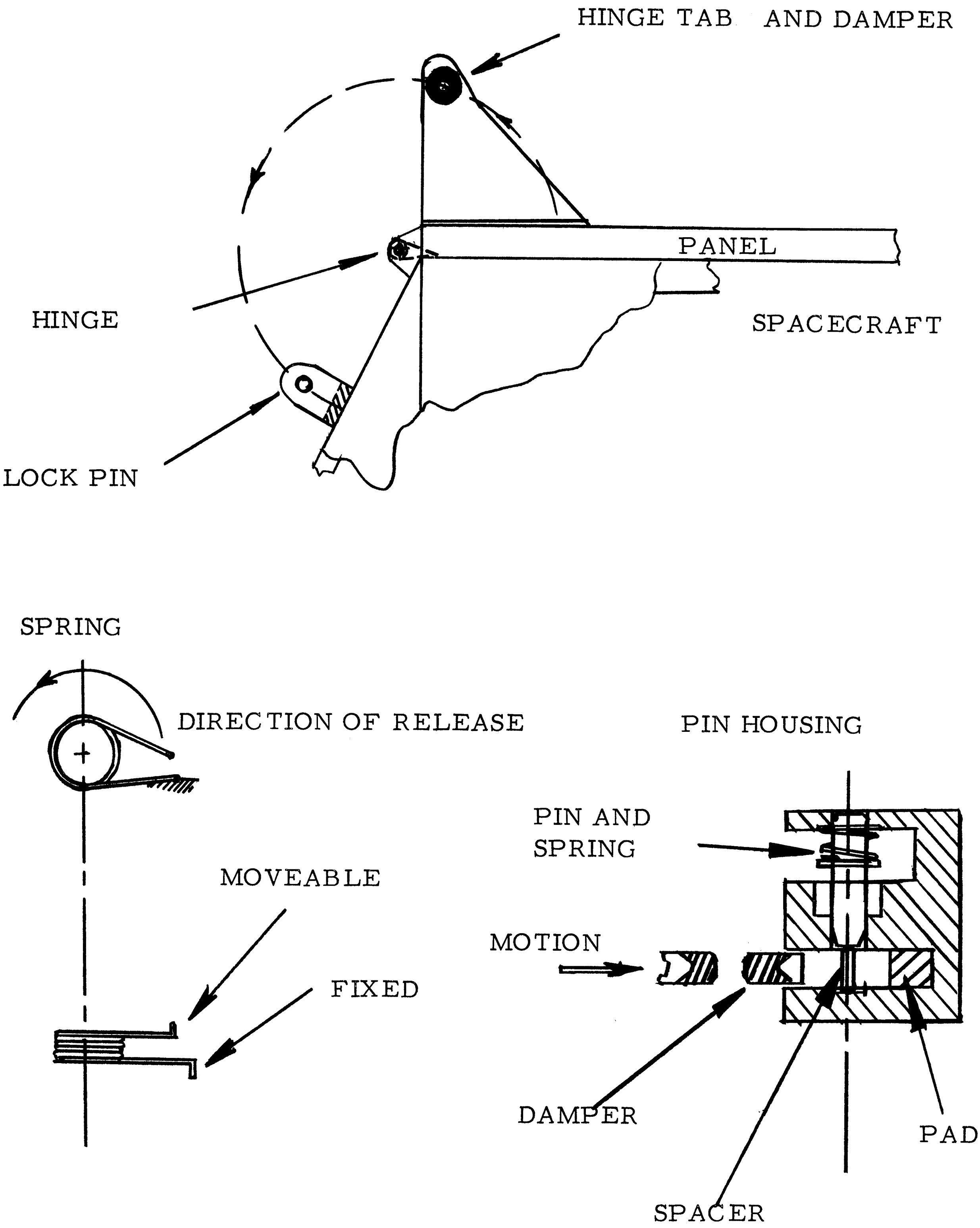


FIGURE 15 . PANEL HINGE MECHANISM

7
THERMAL CONTROL

The principal objective of thermal control is to provide an internal environment for the satisfactory operation of all the instruments on board. A secondary objective is to keep the surface temperature low enough to allow efficient solar cell operation.

The combination of polar orbit, three-axis stabilization, and slow precession rate complicates the thermal problem. A consequence of these factors is that one side of the satellite is exposed to the sun continuously for about 1 1/2 weeks every quarter year. This thermal control system must maintain a temperature range of -4°F to 140°F for effective operation of the instruments. The satellite orbits for one year so the mechanism used for thermal control must be highly reliable as well as light weight. The space problem also limits the types of thermal controls possible. All of these limitations create a difficult problem.

In conjunction with the payload group, and taking into account our power shortage, it was decided to use the surface of the satellite for a power source rather than for thermal control. This decision eliminates coatings as an alternative method.

The solution we chose to Project STRATUM's difficult thermal control problem is a combination of two annular heat pipes and low absorptivity, highly emissive coatings. The thermal coatings have an a/e of .26 and cover 15% of the external satellite surface.

Heat pipes are devices which transfer heat much more effecitvely than the best metal conductors. The governing mechanism of a heat pipe is the evaporation of a fluid from a hot surface and its resulting condensation on a cold surface. A wick inside the heat pipe returns the condensate to the hot end by capillary action. Through this process large quantities of heat can be transferred rapidly. The heat pipes used in the STRATUM satellite contain methyl alcohol as their working fluid. The heat pipes are conductively coupled to the inside walls of the satellite and are positioned such that one encircles the instrument platform and the other is 10 in from the base of the satellite. These positions were chosen to insure both low solar cell operating temperature and an acceptable internal temperature.

For a twilight orbit (100% sunlight), these heat pipes operate at a temperature of 100°F (560°R). In effect, the heat pipes act as heat sinks. The heat in the hot panels flows toward the pipes and is transferred to the cold side of the satellite, (Figure 18) lowering the average temperature of the hot panel. During a twilight orbit, for example, the surface temperature is reduced from 676°R to an average temperature of 590°R. This is shown graphically in Figure 17.

In the upper portion of the satellite, the instruments block internal heat transfer by radiation. Therefore, the interior walls of the satellite are painted with a very low emissivity coating. This helps to prevent overheating of the instruments near the satellite walls. In addition, the top inside plate of the satellite is painted with a high a/e coating to facilitate the transfer of the heat given off by the instruments. Wherever possible, the instruments will be highly polished to aid in heat transfer. The instrument platform is built of honeycomb aluminum with aluminum sheets on each side. Heat conduction through these sheets will help to decrease any temperature gradients caused by the instruments. With these thermal controls, the internal temperature range of the satellite should remain well within the mission constraints.

The solar cell paddles projecting from the rear of the satellite do not pose any thermal control difficulties. The back of each paddle radiates its heat into the natural heat-sink of space, and an efficient solar cell operating temperature is maintained.

The STRATUM satellite is thermally designed for the hottest possible case--the twilight orbit, during which the satellite has one surface facing the sun for the entire orbit. This design criterion is possible because the thermal time constant $\left(\frac{\Delta T}{\Delta t} = \frac{q}{wc_p}\right)$ for the satellite is, at most, $10^{\circ}F/hr$. During the coldest orbit, the satellite is in darkness for 36 minutes. Thus the internal temperature cannot decrease more than $6^{\circ}F$.

While the internal temperature could be controlled without the aid of heat pipes, mission power requirements dictate their use. Due to limited space, average solar cell efficiency must be 10.5% if the complete mission is to be realized. This efficiency and the resulting power output of approximately 50 watts can only be accomplished by decreasing the solar cell temperature through the use of heat pipes. Without heat pipes and assuming 20% of the satellite surface has an a/e = .26, the power output decreases nearly one-third to 35 watts, well below mission requirements. Heat pipes have been orbited previously and have functioned well in the space environment.

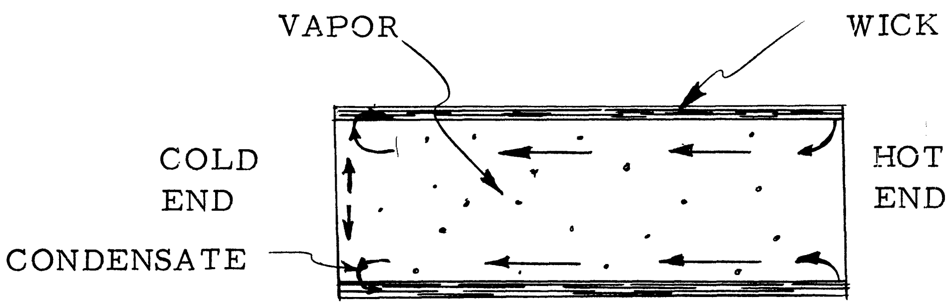


FIGURE 16. HEAT PIPE

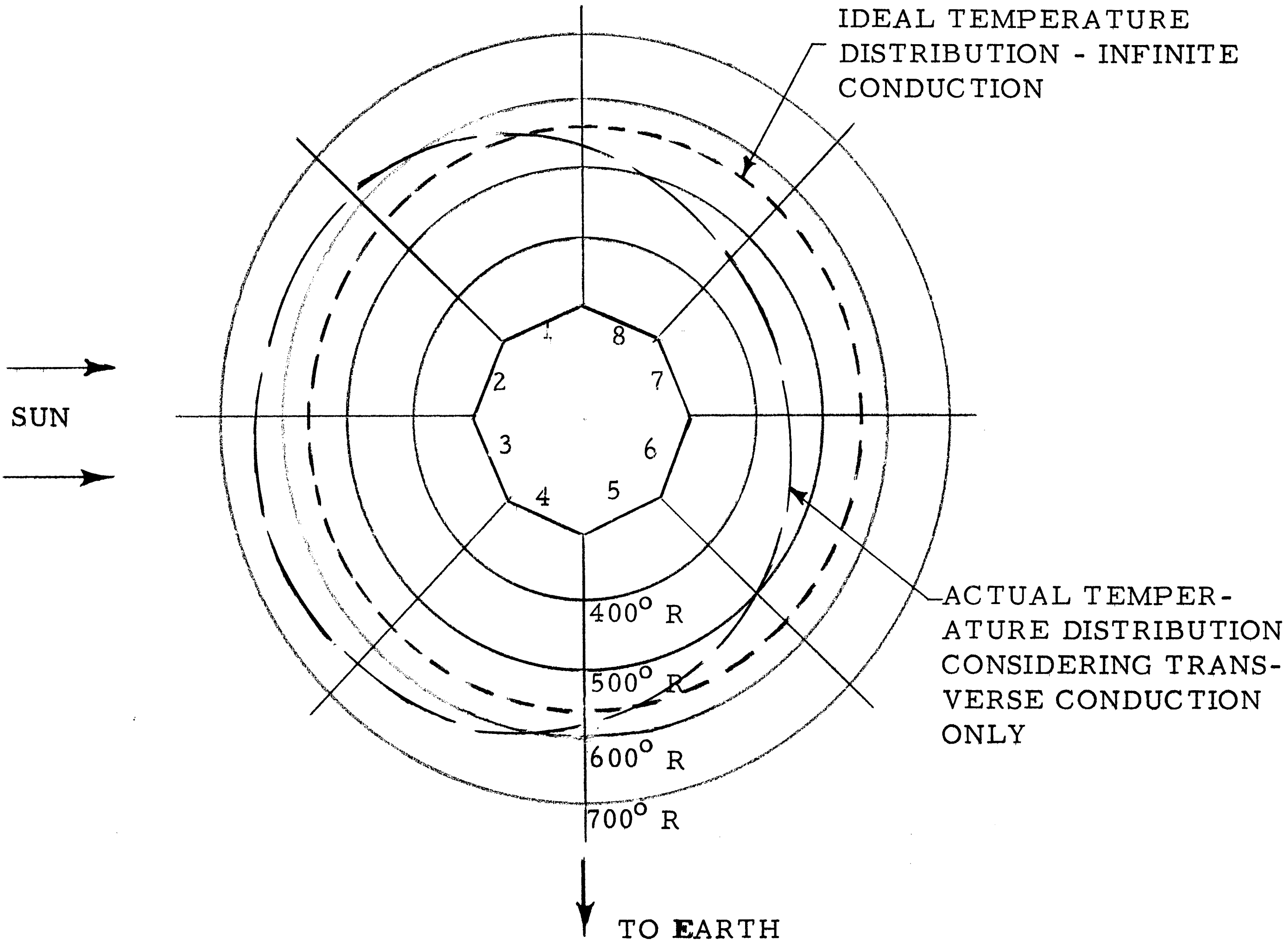


FIGURE 17. TEMPERATURE DISTRIBUTION AROUND THE SATELLITE

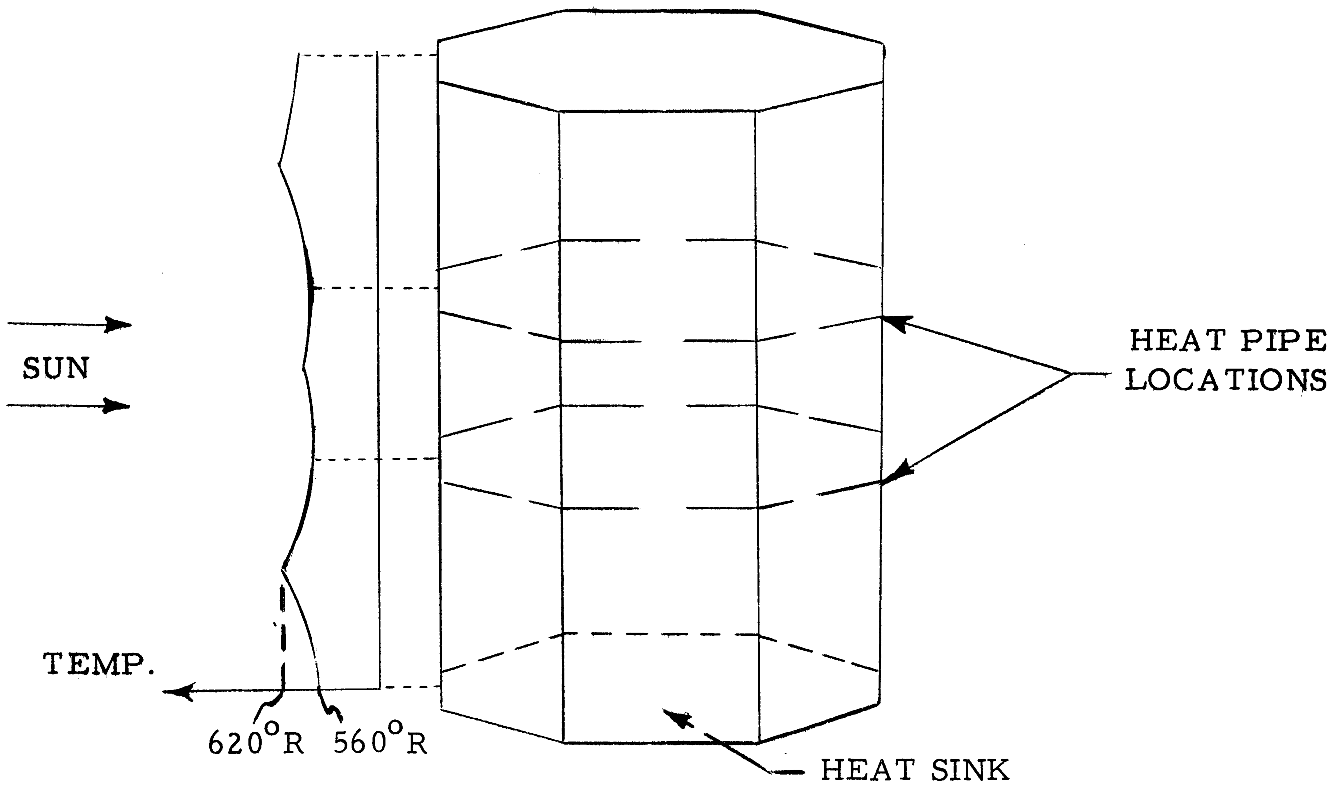


FIGURE 18 . AXIAL TEMPERATURE DISTRIBUTION WITH HEAT PIPES

8

COMMAND CAPABILITY

A command subsystem is required aboard the spacecraft so that the ground facilities can communicate with the satellite. In designing the system, several considerations were taken into account:

1) The spacecraft must receive commands from the ground stations guarding against the receipt of spurious commands.
2) The decoders must decode commands from the ground stations and either process them for "real time" operation or store them for "delayed" commands.
3) The system must re-transmit commands received from the ground for the purpose of command verification.
4) The system must provide the master timing signals required for synchronization of spacecraft operation.
5) The system must route commands and internally sequenced signals to the various electronic systems of the vehicle.

Commands are received from the ground via the spacecraft antenna. The uplink signals pass through the diplexer filter and then to the two receivers which act in parallel. The receivers receive the command and convey it to the decoders. If the command is not in error, the analog "enable" signal is received and the communications channel is opened. Direct commands are carried out immediately, and stored commands are sent to the memory unit. At the time the delayed command is to be executed, a pluse is sent from the quartz crystal (clock) oscillator to the memory, and the command is released. The command is immediately transmitted to the command distribution unit (CDU). From there it is relayed to the proper instrument.

A desirable feature of the command subsystem is that each command word or the significant part of the work can be conveniently retransmitted to earth by the telemetry link after reception in the spacecraft. A small monitoring device is attached to the output of the receiver, such that as the command passes this point, a verification in the form of a voltage signal is conveyed to the beacon transmitter. Once this signal is received on the ground station, verification of command reception in the spacecraft is confirmed.

The diplexer aboard the vehicle serves to make the antenna simultaneously available for a transmitter and a receiver. The lumped constant diplexer uses both high and low pass filters. The lower cutoff frequency of the diplexer reduces the spurious noise generation. The specific characteristics of this diplexer are: 1) an insertion loss of less than 1 db at the transmitter frequency, and 2) less than 2 db at the receiver frequency. The diplexer is designed by ESCO and has physical dimensions of 3 in by 1 in by 2 in. The diplexer weighs 2.5 lbs and has a power consumption of 0.25 watts.

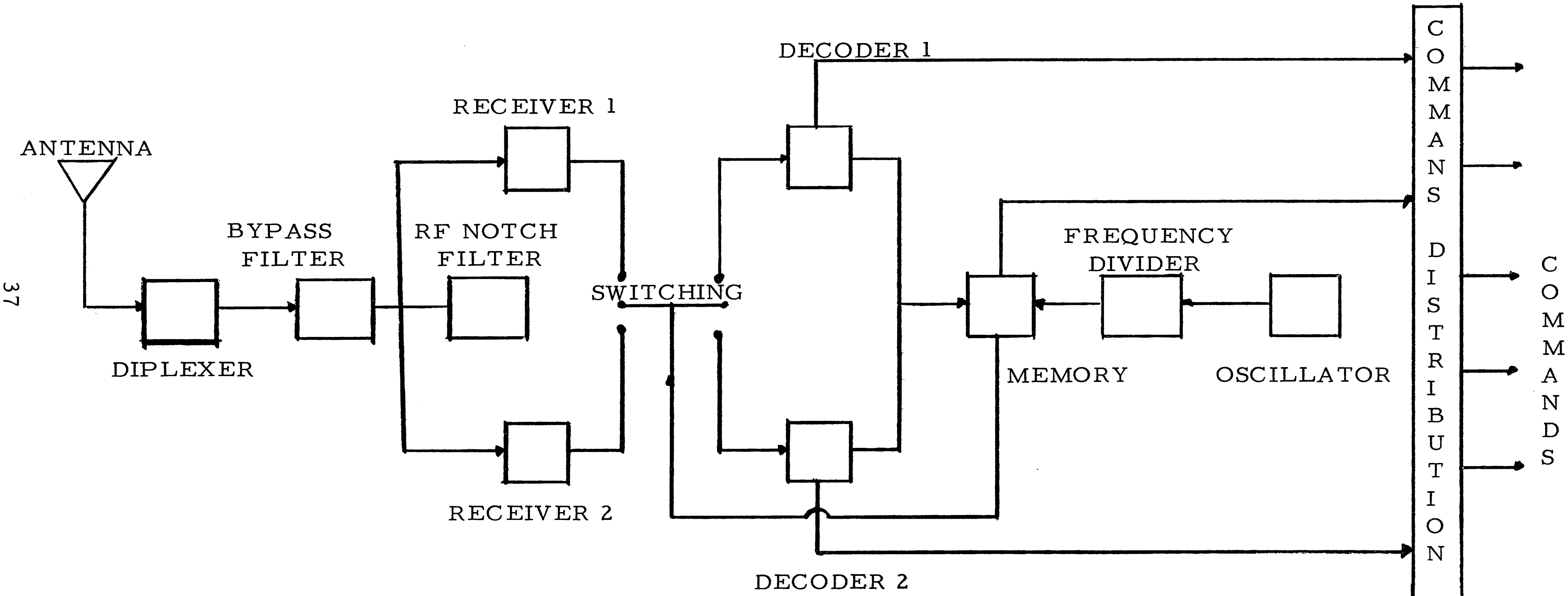


FIGURE 19 . COMMAND SUBSYSTEM

The two receivers aboard the spacecraft act in parallel. A compression packaging method enables the unit to withstand high vibration and shock levels. The receivers being used on the satellite are similar to those produced by the Conic Corporation of San Diego, California; model CCR-FM. The specific model chosen is designed to meet the stringent environmental reliability requirements of STRATUM. The unit also has high adjacent subcarrier channel rejection and protection against false triggering by noise or power supply voltage transients.

The decoders and receivers used on STRATUM are built in one unit. Each decoder consists of circuits for decoding the commands. An "enable" signal must be received by the command and control equipment before the communications channel is opened. Once "enabled" the decoder accepts the frequency shift keyed command signals from the ground station and, provided the correct spacecraft address is included, supplies the commands to the CDU or to the memory.

Only one receiver and one decoder will be operative at a time. However, any possible combination of receiver and decoder can be used upon command from the ground. The same process will also be used in determining which decoder will be operative and how many channels will be in use.

The modules are encapsulated with clear silicon rubber, stacked under compression using foam pads, and wrapped with vibration damping tape. These visco-elastic materials subdue individual part resonances so that the unit vibrates as a homogeneous mass, thus eliminating the damaging effects of relative motion.

Selection of the command receiver will depend on a command stored in the core memory. A switching system connects the two receivers with the two decoders. After every other orbit, the receivers will be switched, i. e. the receivers will alternate operating modes. Also the decoders will be switched every orbit. By this switching it is possible to use any combination of decoder and receiver for the system. Also if any one receiver or decoder should fail, the system will be inoperative for a maximum of three orbits.

A clock command system is required in order to provide time signals and control over the satellite when it is out of range of the ground stations. All of the delayed pulses are derived from a single clock oscillator which is very stable. The timer used is unique in that it combines versatility, high-reliability, low power consumption, and ease of installation, programming, and checkout in a package of minimum size and weight. The Time Reference Generator (TRG) is an accurate 24 hours clock which counts up to 8 days and then recycles. A thermistor is bonded to the crystal casing and its output is used to monitor the temperature of the crystal. The thermistor is needed because a high temperature variation brings less accurate timer results. The timer on board STRATUM will be similar to that manufactured by Adcole, Model 227.

TABLE 2

ITEM	NUMBER REQUIRED	TOTAL WEIGHT	TOTAL VOLUME	POWER REQUIREMENTS	MANUFACTURER
Telemetry commutator	2	2.0	65.7	-24.5 VDC 100 ma	Dynaplex
Subcarrier oscillator	42	5.3	72.3	28 -4 VDC at 3 ma	Conic Corp.
Mixer amplifier	42	5.3	16.0	28 V at 20 ma	Conic Corp.
S-band transmitter	1	0.8	9.8	28 ± 4 VDC at 600 ma	Conic Corp.
Command receiver	2	2.0	34.0	24 - 36 VDC, 15 ma at 28 VDC	Conic Corp.
Command decoder	2	2.0	28.0	24 - 36 VDC, 15 ma quiescent 22 ma for each channel at 28 VDC	Conic Corp.
Command distribution	1	2.5	30.0	0.25 watts	--
Memory	1	3.5	80.0	3.0 watts operational 0.3 standby	DI/AN Contr
Clock	1	1.4	168.0	28 -4 VDC, 0.7 amp/event	Adcole
Frequency divider	1	0.5	9.0	0.25 watts	--
Diplexer	1	2.5	6.0	0.25 watts	ESCO
Beacon transmitter	1	0.8	9.7	28 + 4 VDC at 250 ma	Conic Corp.
Antenna/Filter	1	6.0	2.0	--	--
Cables	--	10.0	--	--	--
		44.6	530.5		

*The instruments used in STRATUM will be similar to models made by these companies.

A magnetic core memory is installed on the spacecraft in order that commands sent from the ground can be stored and later executed in the orbit. The memory will be manufactured by DI/AN controls and is to have a storage capability of 230 words, 7 tones per word. The magnetic core memory has a high reliability and has been flown several times. The search and find time for one particular bit of information is about 9 nanoseconds. Even with this high speed, it has a high degree of accuracy.

The frequency output of the oscillator is broken down by a chain of multivibrators to give various frequencies to the experiments. Widely known as a "flip-flop", the bistable multivibrator is a circuit containing two active elements so arranged so that only one active element can be conducting at a given time. The application of a trigger pulse of the correct polarity causes the active elements to exchange roles.

The command distribution unit is analogous to a large switchboard. This unit distributes commands to the designated unit or subsystem and provides impedance matching between subsystems.

It should be noted that within the command subsystem, some of the units are redundant. These units may be selected independently by ground command to form an operating system. As a result, only failure of identical redundant units could prevent the command subsystem from operating. The units that are not redundant have been found to be highly reliable, and therefore only one unit is required aboard the vehicle.

Both the command distribution unit and the frequency divider are devices that have to be manufactured to our specific specifications. Contract bids must be submitted before the cost and availability of these items can be determined.

8.1 SEQUENCING

Before the launch of STRATUM, a sequence of commands will be programmed into the magnetic core memory of the command subsystem. This command system will insure that the several operations which have to be performed before the satellite can function correctly after the vehicle is in its initial orbit will occur at the proper times. The times of execution of these commands will be determined in conjunction with the manufacturer of the launch vehicle and fed into the clock memory unit before launch. During the flight, when a particular time is reached, the clock will send a pulse to the memory and the command will immediately be released and executed. As soon as the command is released, the memory is capable of storing another command in that exact location. It will be necessary for one of the tape recorders to be running and all verification devices or sensors (approximately 80) to be operative during the initial stages of the flight. Once the satellite is in orbit, all the experiments should be turned on to insure that they are working properly. As soon as the vehicle is over a ground station, all the data will be dumped. From this information it can be determined if everything is functioning correctly. Following is a list of some typical commands for the initial sequencing. It is not complete, but the commands are typical of the system.

1. Select tape record 1, Enable recorder 1
2. Yo-Yo despin
3. Separation
4. Deploy panels
5. Activate marmon clamp, activitate potentiometer, deploy booms
6. Instruments on
7. Activate IR sensor 1 thru 4
8. Activate sun sensors 1 thru 4
9. Select telemetry commutator, enable commutator 1
10. Select transmitter 1, enable transmitter 1
11. Request telemetry data
12. Playback recorder 1

NOTE ON 4: The paddle deployment will simultaneously deploy the VHF turnstile antenna by uncovering it.

9
DATA STORAGE AND READOUT

Project STRATUM proposes to place six scientific instruments into a near-polar orbit. In addition, it must accurately determine the critical variables, such as temperature, current drain, voltage, on-off state, etc., which affect the operation of the on-board equipment. The operating staff base their day to day command schedules on this information. Data which aids in the reduction of the experimental returns, such as attitude information, satellite position, etc., also must be telemetered to ground stations. The need for a versatile and flexible data storage system is apparent.

Currently the only two techniques of data storage which apply to the field are magnetic core storage and magnetic tape storage. The core system, while rugged, extremely fast, and accurate, suffers from a relatively large weight and volume to storage capacity ratio. It is also restricted to information of a digital nature. Since STRATUM uses a communications system based on the frequency division multiplex process, the data stored is in an analog form, which eliminatescore storage from consideration.

Tape recorders can be made very compact for a large data capacity, but are not as inherently reliable as core systems. The spacecraft uses three small recorders to attain the one year project requirement. The units can be switched on and off at will in any combination, or can be bypassed completely to permit real-time transmissions in case of total storage failure. Should an orbit pass without a data transmission, the overflow can be absorbed without loss by activating a second recorder. Figure 20 is a system block diagram.

The entire communications system determines the basic operating requirements of the tape recorder. Tape speed is 1.875 inches per second in the record mode, and the units have a frequency response at this speed of 6 KHZ. The orbital mechanics permit ground contacts of less than ten minutes; the actual data transmissions should occupy as small a part of this time as practical. Input signals to the recorder are at IRIG standards from the SCO units; hence the readout will be some multiple of these standards. In order to compress the 90 minutes of data into the small transmission time, a speed change of 32 was selected, with recorder playback mode running at 60 ips. At this speed total data readout requires 3 minutes maximum. Furthermore, in the recording mode the data bandwidth is 6 KHZ, so on readout it is also multiplied by 32, to give a bandwidth of 192 KHZ. The tape speed while recording is 1.875 ips, thus the orbital requirement is 845 feet of tape. For a tape recorder similar to the Kinelogic Corp. model LG, the one used for the spacecraft layout, the tape lengths are 900 feet. Without going to longer tapes and larger recorders, or multitrack systems, this implies a data readout once

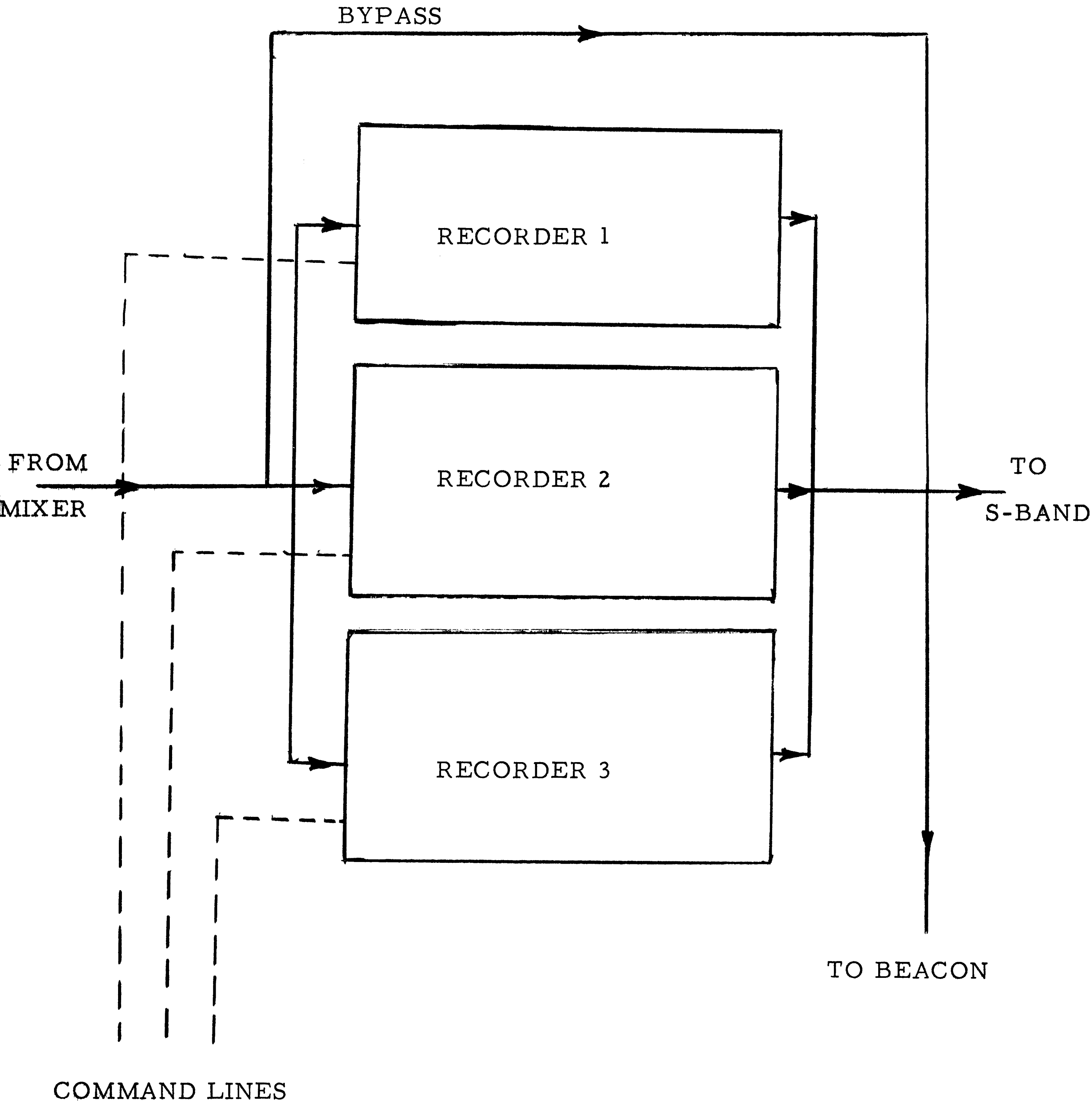


FIGURE 20 . STORAGE SYSTEM DIAGRAM

every orbit. Multitrack systems need less tape but have additional complexity in track switching and playback logic to permit access to all the stored data. Since every orbit will be in contact with a ground station, there is no need for the added multitrack capability.

Since the recorders are the reel-to-reel type, they have a preferred mounting orientation. Their acceleration limits are less in directions parallel to the reel axes. (Figure 21)

Operational life of each tape recorder is limited primarily by the effects of wear on the thin mylar belts which drive the reels. These are subjected to severe stretching stresses during each start-stop cycle, and the abrasive effects of the tape oxide particles in the zero-gravity environment weaken them after a few months. Typical lifetimes are 4400 hours of continuous operation, or about 180 days. This decreases under the wear of start-stop and speed changes. Three recorders should provide enough margin to insure accurate service during the entire year. Data loss from a recorder failure is limited to one orbit since the backup recorders are operated by ground command. The real-time mode somewhat limits the usefulness of the satellite, but the wide range of ground stations available for use under emergency conditions does not make this a strict limitation.

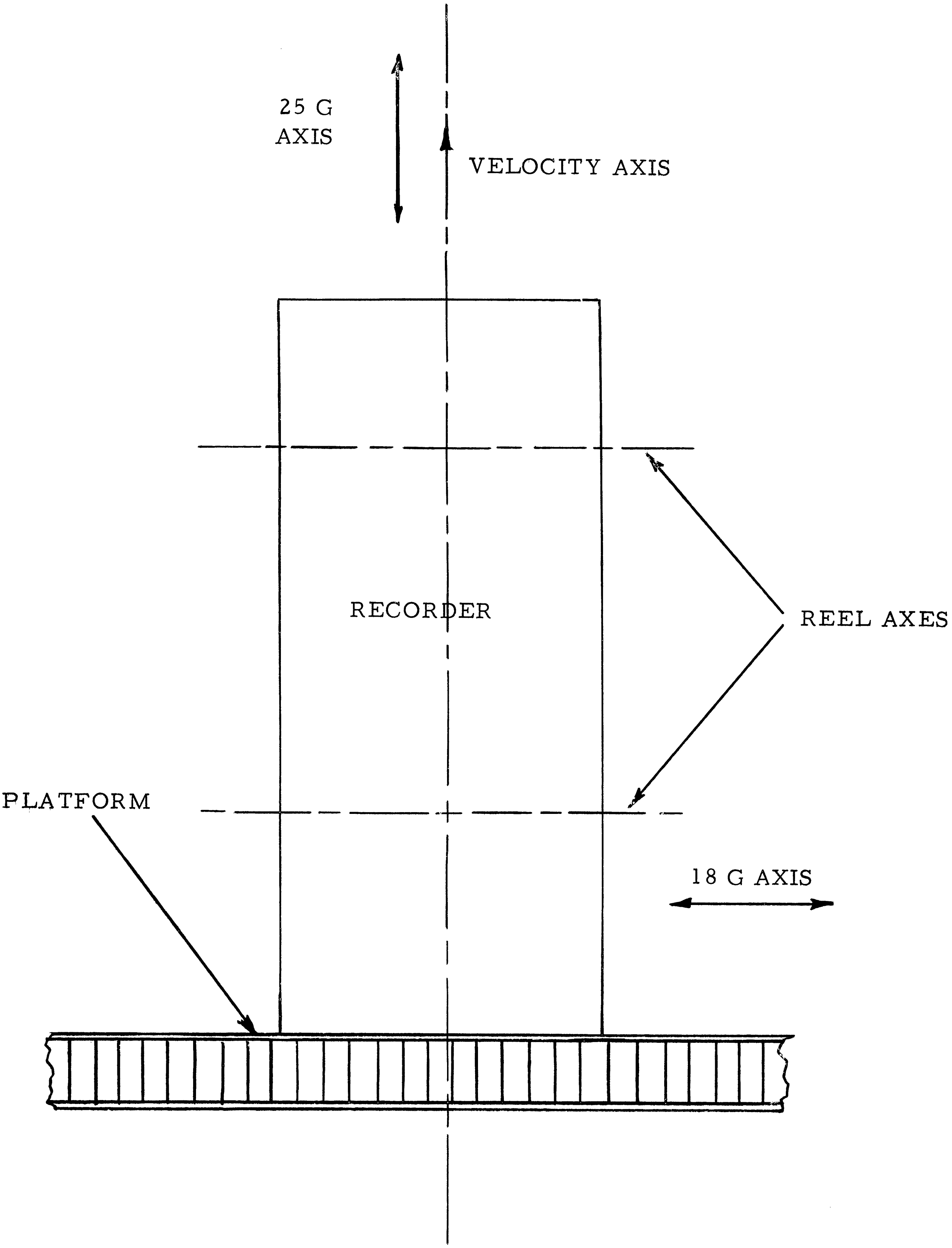


FIGURE 21 . TAPE RECORDER MOUNT DETAIL

10
COMMUNICATIONS AND DATA PROCESSING

10.0 INTRODUCTION

The communications and data processing system includes the functions of command, satellite tracking to determine orbit parameters, and the transmission of sensor and housekeeping data to ground terminals. To fulfill these requirements, it was decided to use existing ground facilities and systems since performance characteristics of the overall system do not exceed the capability of the ground system. This approach is the most economical because little, if any, additional ground equipment and personnel are needed to collect the data. After investigation, it was decided to use the Minitrack system for tracking and command, and to use S-band for transmission of the sensor data and housekeeping telemetry. Important aspects of the communication systems are discussed in greater detail in the following paragraphs.

10.1 GROUND SYSTEMS

The choice of ground support systems has been based on systems which will be available in the near future. For command and orbit parameters, the Minitrack system has been specified. This system is available at all ground sites and has a proven record of performance. Tracking accuracy for orbit parameters is $\pm$ 20 seconds of arc. To transfer data from the spacecraft, the S-band system will be employed. This sytem is preferred by NASA and is available at a sufficient number of ground stations. The systems, along with important characteristics, are shown in Figure 22.

For the down link a signal level of -150 dbw is required. This requirement is met with 1 watt transmitted when the following conservative assumptions are made: 1) antenna gain at the satellite of zero db, 2) range of 1800 statute miles ("worst case"), and 3) 4 db loss due to the antenna diplexer and cable losses. The signal will increase by about 10 db for those orbits which pass directly over the ground terminal.

The command (up) link is the most critical of the three communication links. Loss of the command link will abort the mission since no sensor data will be transmitted. Fortunately, adequate signal-to-noise ratio will be obtained at all orbit locations. For the "worst case" (i. e., a range of 1800 statute miles), a signal-to-noise ratio in excess of 20 db will be obtained. This will increase to about 30 db for orbits which pass over the ground station.

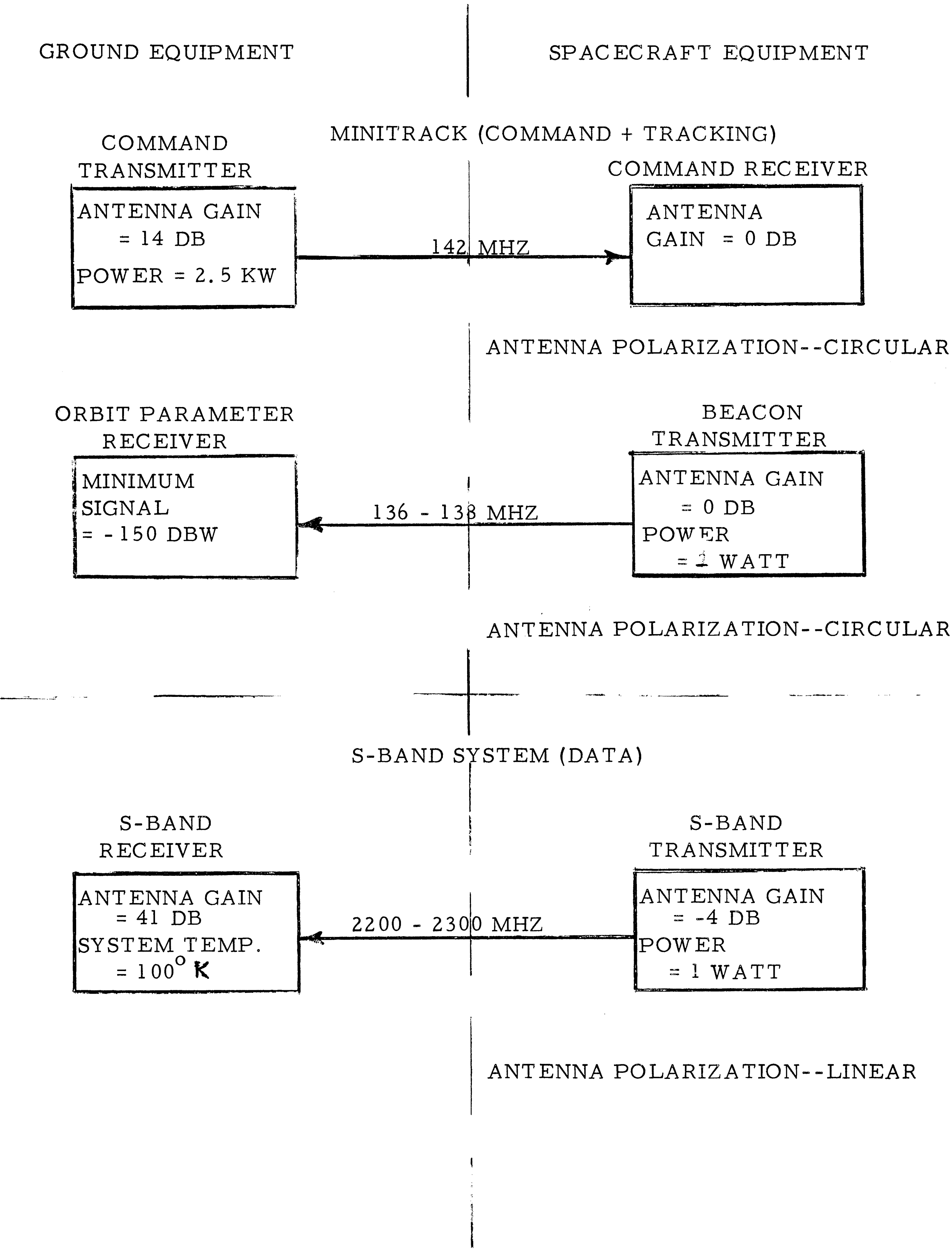


FIGURE 22. PROPOSED COMMUNICATION LINKS

The S-band system has the following specifications:

Power transmitted	=	1 watt
Spacecraft antenna gain	=	-4 db
Ground antenna gain	=	41 db
Spacecraft loss (cable, filter)	=	2 db
System noise temperature (ground)	=	100^{o} Kelvin
R.F. bandwidth	=	400 KHZ
Free space loss (1800 statute miles)	=	169 db

These specifications result in a predicted signal-to-noise ratio of 20 db for the "worst case" operation. The signal-to-noise ratio for the "best case" will be 35 db. Thus, adequate signal levels are obtained for all links for any orbit.

10.1.1 Selection of Ground Stations

Two stations are to be used regularly, Fairbanks, Alaska, and St. Johns, Newfoundland. To attain the desired data drop every orbit and with a five minute drop time, this is the most practical network available. A drop is missed occasionally due to a 2^{o} coverage gape at 90^{o}W longitude. The earth turns 1.2^{o} during minimum drop time so the chance of the satellite being in this zone is small. Depending on the data storage capability, a delayed command may be given to rewind the recorders, thus minimizing loss. Wallops Island, Virginia may be used if the particular data drop is critical.

10.1.2 Station Coverage

The coverage capabilities of any station, for a fixed altitude in near polar orbit, may be examined conveniently in earth longitude and compared to the satellite's earth-path. The earth-path is the locus of points on the earth's surface that are nearest the satellite. After using the three-quarter earth approximation to account for the atmospheric bending of radio signals, the remaining problems are geometrical. The earth's rotation results in a slanted earth-path with respect to the earth's longitudinal lines. The first step in the calculation is the determination of the look distance, the region in which the station can "hear" the satellite.

With 3/4 earth, and an altitude of 300 statute miles we have:

$$D = \sqrt{2h} = \sqrt{2(1.64 \times 10^6)} = 1810 \text{ miles}$$

where, D = look distance in statute miles
h = altitude in feet

For any station, five minute dump time corresponds to 5.3% of the orbit or d = 1425 miles as shown in Figure 23.

$$\frac{5 \text{ min.}}{94 \text{ min.}} = 5.3\%$$

$$(5.3\%) \text{ (orbit circumference)} = (.053)\ (26,800) = 1425 \text{ miles}$$

A top view shows the station coverage circle at a radius of 1810 miles and the minimum drop locus at ξ^{o} from the 0^{o} or direct overhead locus. (Figure 23)

d = 1425 miles, corresponding to 5 minute drop time at ξ^{o} longitude

We now find p, the distance from station to earth-path:

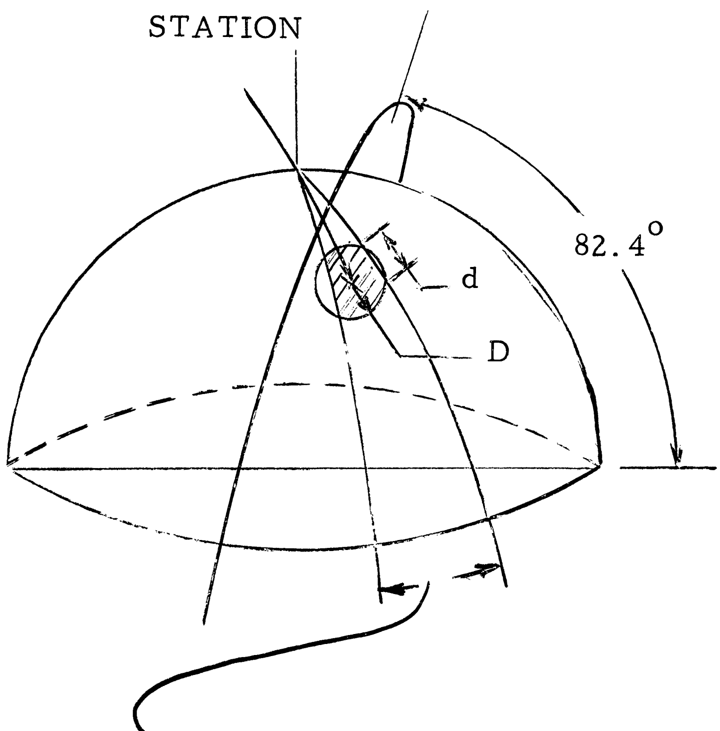
$$\cos(a) = \frac{d/2}{1810} = .393$$

$$a = 66.6^{o}$$

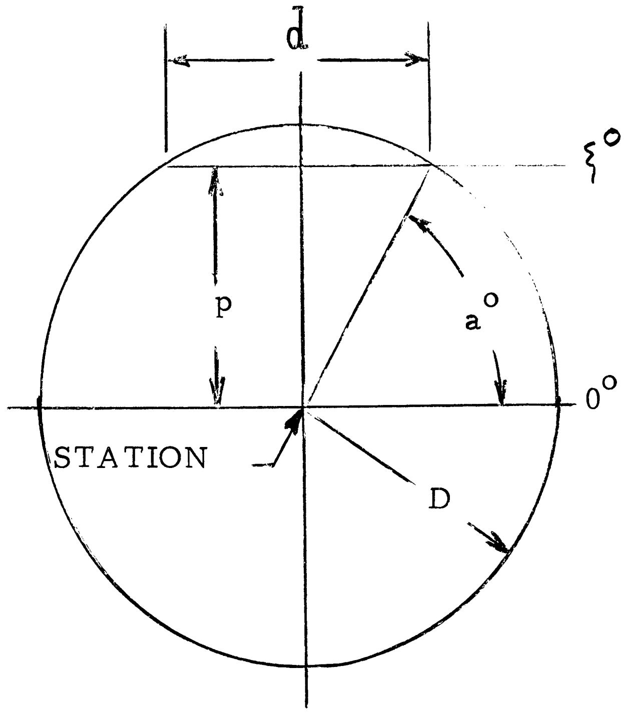
$$p = (\sin a)\ 1810 = 1665 \text{ miles}$$

Now we examine this coverage for a specific station, for example Fairbanks at 65^{o} N latitude, 147^{o} W longitude. The 65^{o} N line is 10,500 miles long, so p is 15.8% of longitude or 57^{o}. The coverage is then 147^{o} W longitude $\pm$ 57^{o} longitude, or 90^{o} W to 156^{o} E longitude.

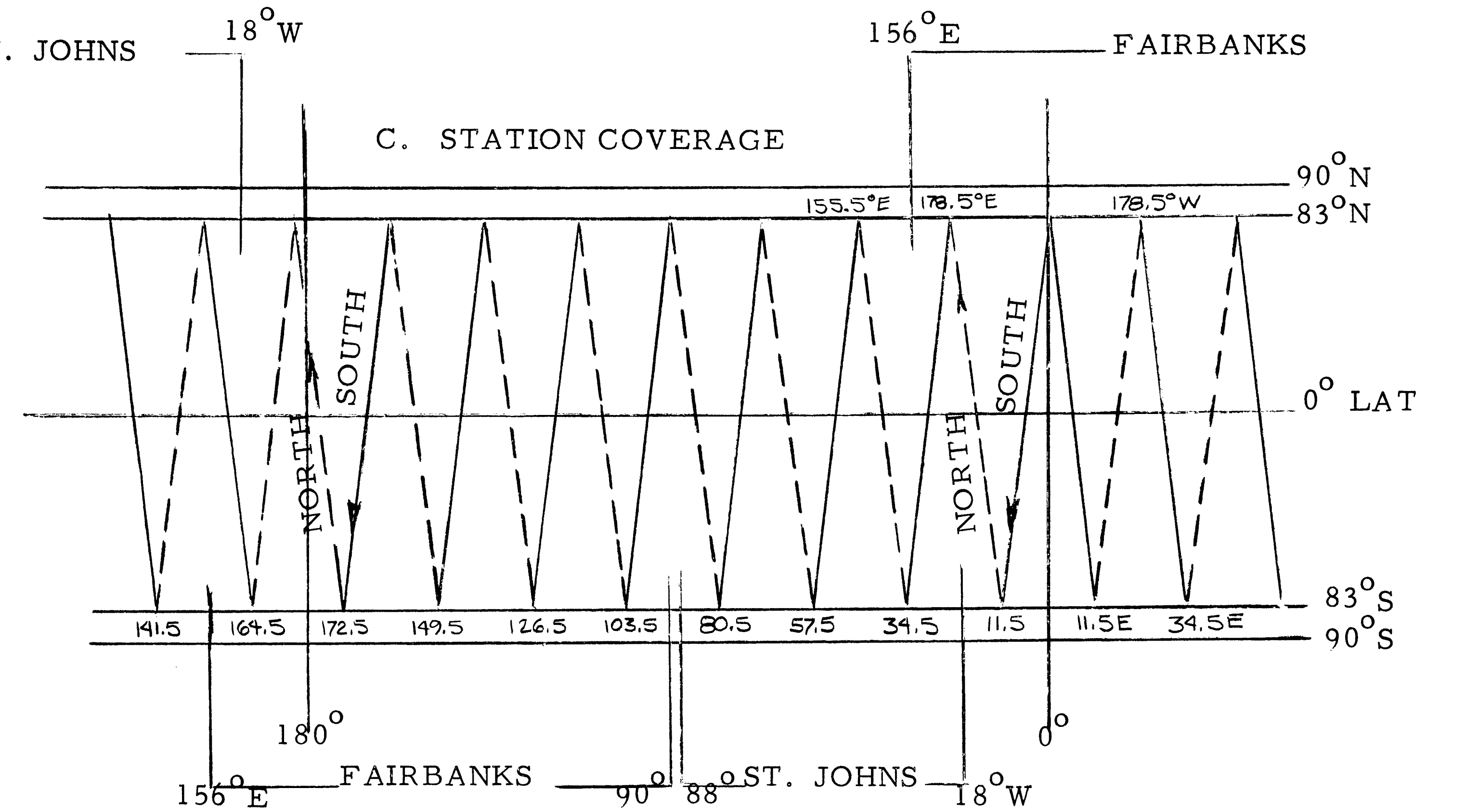
The ground station coverage for Fairbanks and St. Johns is shown in Figure 23 . Due to the earth spin, any two points at the same latitude in consecutive orbits are 23^{o} apart. Every 7.8 orbits the satellite will approximate a 0^{o} longitude orbit and therefore the direction of pass-over for any station will be reversed. The earth-path origin for the station coverage calculation and diagram is arbitrary because the pre-orbit tracking network is fixed.



A. STATION COVERAGE ZONE



B. TOP VIEW OF STATION



C. STATION COVERAGE

FAIRBANKS - 90°W to 156°E
ST. JOHNS - 18°W to 88°W

DARK LINES - PASSOVER WESTERN HEMISPHERE
DOTTED LINES - PASSOVER EASTERN HEMISPHERE

Real time data drops have been considered in the case of recorder failure. This can be done within the range of existing stations

1) Over the north pole using Fairbanks
2) Over the north-south Minitrack stations in U. S. -South America and over Europe-Africa, limited by available power for the S-band transmitter.

10.2 TELEMETRY SYSTEM

The telemetry system will be used to collect the housekeeping data and will include the attitude sensor data and calibration voltages. An available commutator has a commutating rate of 16 2/3 channels per second when driven by a 133 1/3 HZ signal and will accommodate 90 data channels. For this system the following characteristics are obtained.

a) Each channel is sampled at a rate of 0.185 samples/sec.
b) Information bandwidth for each channel is 0.0925 HZ (for bandlimited data).
c) The sample rate for the commutated data is 16 2/3 samples/sec.

10.3 DATA SYSTEM

The data to be stored and transmitted via the S-band system are shown in Figure 24 . Sampling rates are shown and are based on data roll-off of 24 db/octave and interpolation filters of the Butterworth type of fourth order. The values of sampling rates, f_s, are then obtained from the source by McRae*. Thus, if time division multiplexing or other sampling techniques are used, the sampling rates shown must be met or exceeded.

In designing the data system, the constraining item is the recorder used for storage. In order to store the data for one orbit, a low tape speed of 1 7/8 ips is required. The low tape speed yields relatively narrow recorder bandwidths. Bandwidths of DC to about 10 KHZ are typical. Thus, sampling and coding, say into a PCM signal, and then recording the sampled and coded signals requires excessive recorder bandwidth and will not be considered for this application. Instead, frequency division multiplexing will be used. The system is shown in Figure 25 . Each data channel is applied to the subcarrier oscillator and the outputs of the subcarrier oscillators are then added to form a baseband which is then recorded.

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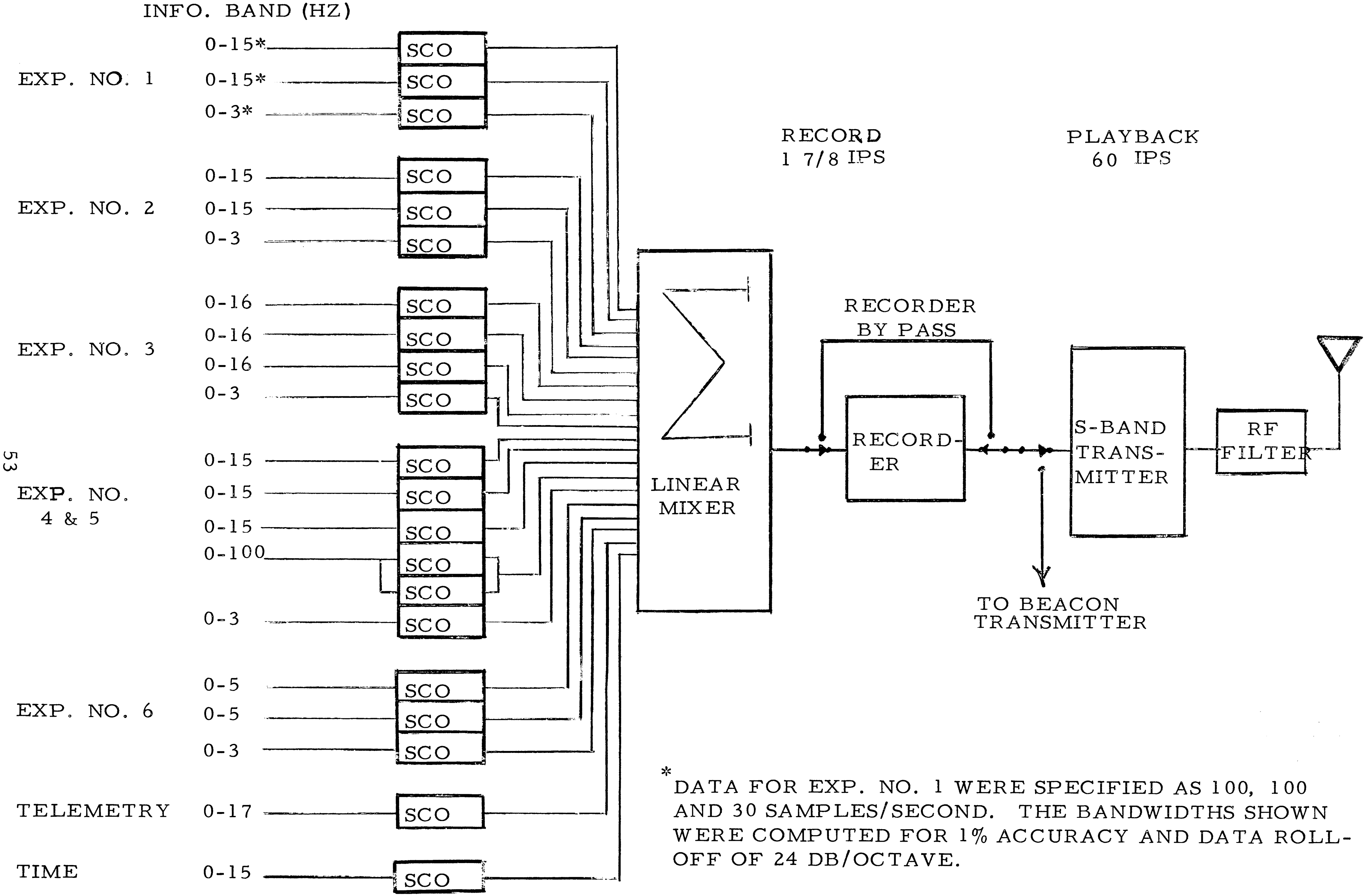
*McRae, D. D., "Interpolation Errors," Radiation Inc. Report, Technical Report # 1, Part 1, 15 February 1961.

	Data Bandwidth	ACC	(1) (2) $f_s = \alpha f_o$	f_s (bps)
	100S/S			100
Exp. No. 1	100S/S			100
	30S/S			30
Exp. No. 2	0-15HZ	1%	$12f_o$	180
	0-15HZ	1%	$12f_o$	180
	0-16HZ	1%	$12f_o$	192
Exp. No. 3	0-16HZ	1%	$12f_o$	192
	0-16HZ	1%	$12f_o$	192
	0-15HZ	1%	$12f_o$	180
Exp. No. 4	0-15HZ	1%	$12f_o$	180
& No. 5	0-15HZ	1%	$12f_o$	180
	0-100HZ	5%	$5.5f_o$	550
Exp. No. 6	0-5HZ	1%	$12f_o$	60
	0-5HZ	1%	$12f_o$	60
Monitors				
Exp. 2	0-3HZ	1%	$12f_o$	36
Exp. 3	0-3HZ	1%	$12f_o$	36
Exp. 4 & 5	0-3HZ	1%	$12f_o$	36
Exp. 6	0-3HZ	1%	$12f_o$	36
Telemetry				16 2/3
Time				15

(1) f_s = sampling rate

(2) f_o = corner frequency of the data

FIGURE 24. DATA BANDWIDTH



*DATA FOR EXP. NO. 1 WERE SPECIFIED AS 100, 100 AND 30 SAMPLES/SECOND. THE BANDWIDTHS SHOWN WERE COMPUTED FOR 1% ACCURACY AND DATA ROLL-OFF OF 24 DB/OCTAVE.

FIGURE 25. S-BAND DATA SYSTEM

The 21 subcarrier oscillator specifications are those of the IRIG Constant Bandwidth Subcarrier ((A) Channels) standard scaled down by the factor of 1/32. Thus, when the recorder is played back at 60 ips, the output of the recorder will be the IRIG standard. The (A) standard covers the frequency range of 14 KHZ to 178 KHZ with 21 channels. The center frequencies of the channels are separated by 8 KHZ and the deviation is $\pm$ 2 KHZ. A modulation index of 5 results in an information bandwidth of 0-400 HZ. When scaled down by the factor of 1/32, the information bandwidth for each channel is 12.5 HZ. Since a number of the sensor outputs have bandwidths of 0 to 15-16 HZ, a slightly lower modulation index can be used (assuming the frequency deviation remains fixed). For example, an index of 4 would provide a bandwidth of about 16 HZ. The 0 to 100 HZ bandwidth of experiments 4 and 5 can be accommodated by using two channel spaces and a still lower modulation index of about 1 to 2. Although the data for this sensor will be degraded, the specified accuracy of 5% is met and the degraded data is acceptable. To accommodate the data system, a recorder bandwidth of 6 KHZ at a tape speed of 1 7/8 ips is required.

The data system will contain switching which will allow the recorder to be by-passed to permit data to be transmitted in real time. This provision will permit continued operation of the mission (although at reduced capability) should the recorder fail in flight. A back-up system is also provided to relay data via the beacon transmitter should the S-band transmitter fail. Because of this redundant operation, single beacon and S-band transmitters are specified to conserve weight and power.

For the S-band system, the RF bandwidth required for the data is 178 KHZ. To this must be added additional bandwidth for Doppler shift and frequency instability. The Doppler shift can be computed by the following expression

$$\Delta f \approx \pm f_t \; v/c \cos \theta$$

where f_t is the transmitted frequency, v is the satellite circular velocity, c is the velocity of light and θ is the angle between the local velocity vector and the direction vector which lies on a line between the satellite and ground terminal. For this system the maximum Doppler shift is $\approx \pm$ 50 KHZ which requires a bandwidth of 100 KHZ. If a conservative value of frequency stability of $\pm$ 0.001% is assumed, a frequency variation of $\pm$ 22 KHZ results and additional bandwidth of 44 KHZ is required. The total RF bandwidth (maximum) due to the above three factors is approximately 310 KHZ.

10.4 SATELLITE ANTENNA SYSTEMS

The satellite antenna systems are critical elements of the STRATUM system. Adequate earth coverage must be provided and the antenna must be light weight and easily deployed. The coverage requirement is most demanding for the command link. The coverage for the command link should be spherical to insure that the satellite can be commanded in any orientation. Further, the antenna for the command link must be deployed before any other items of the satellite are deployed. The requirement for ground coverage of the S-band antenna is not as stringent as the command link. The satellite will be stabilized, and near hemispherical coverage about the local vertical of the spacecraft will be adequate. Because of the wide frequency separation of the VHF and S-band system, separate antennas are specified for each frequency.

A turnstile antenna has been selected to provide the necessary coverage for the VHF systems of command and tracking. (A diplexer will keep the command and tracking signals separated.) Turnstile antennas have been used successfully on a number of satellites*. The antenna configuration is shown in Figure 26.

The turnstile will be mounted at the rear of the spacecraft where the solar panels are hinged to the vehicle. The elements are quarterwave whips mounted on edges of the satellite body at angular distances of 90^o. The elements are fed in phase quadrature. Circular polarization is obtained to the front and rear of the satellite (designated as $\theta = -90^o$ and $+ 90^o$ in Figure 26. Elliptical polarization is obtained as the angle tends to $\theta = 0$ and 180^o. Finally, linear polarization is obtained at $\theta = 0$ and 180^o. Shown in the figure is a cone angle θ about the local vertical. The cone angle is the viewing angle between spacecraft local vertical and the spacecraft ground station line. Thus, a cone angle of $\theta = 0$ degrees is obtained when the spacecraft is directly over the ground station and tends to $\pm 90^o$ as the satellite approaches the station from the horizon (-90^o) and leaves the station heading for the horizon ($+90^o$). Maximum cone angles of approximately $+70^o$ were calculated for the STRATUM orbit.

The graph of Figure 26 shows the theoretical absolute gain for a turnstile antenna for a cosine pattern of short dipoles and for a range of cone angles of $\pm 90^o$ and left and right circular polarization. If right circular is obtained along the direction of flight, then the opposite sense is obtained at the rear of the spacecraft. (The sense of the polarization can be varied by the phasing of the elements.) The shaded area is the range of

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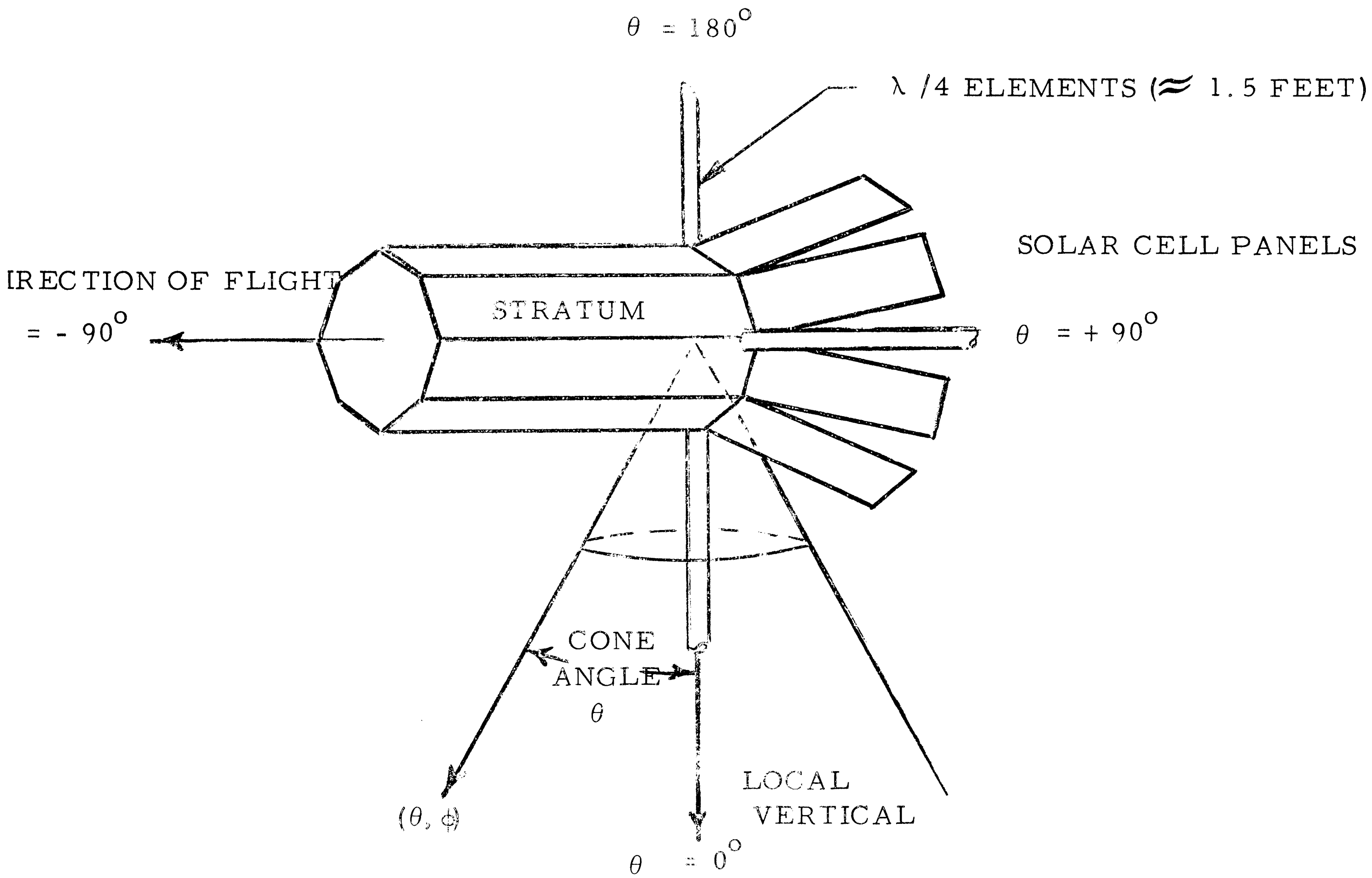
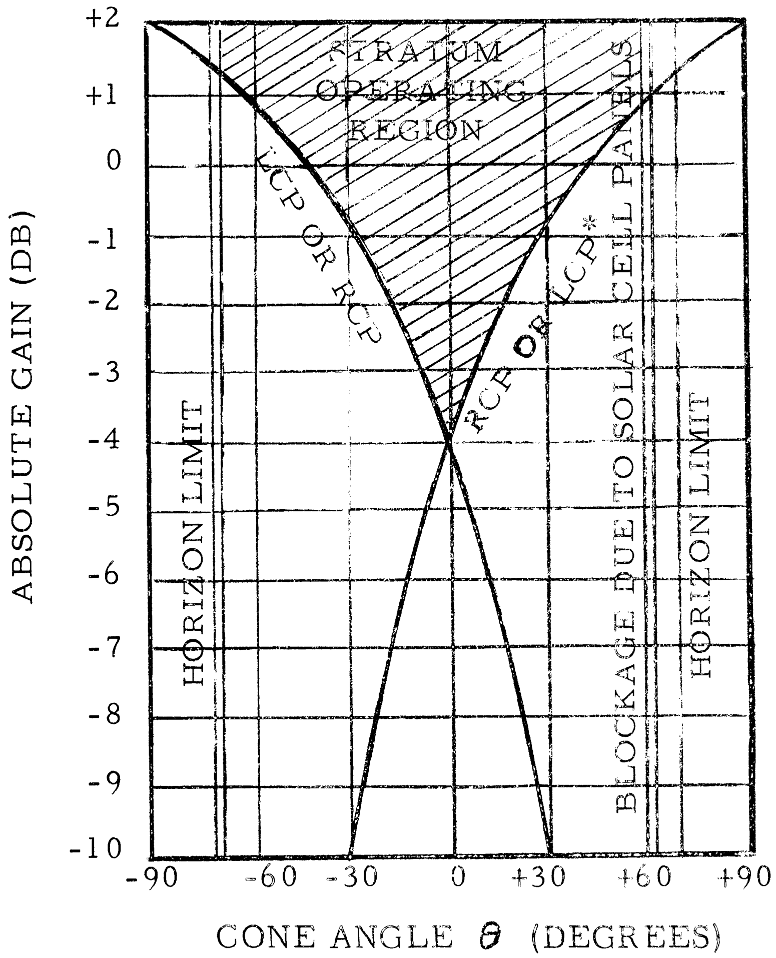
*The STRATUM staff is indebted to Mr. Ron Kern of General Electric Space Division for data on the use of turnstile antennas for space applications.





NOTE: A CONE ANGLE OF $\theta = 0°$ LOCATES THE SATELLITE DIRECTLY OVER THE GROUND TERMINAL

*LCP/RCP: LEFT (RIGHT) CIRCULAR POLARIZATION

FIGURE 26. VHF ANTENNA

operation of STRATUM. The coverage pattern about the direction of flight will be a figure of revolution of the shaded area about the direction of flight. One bound is shown as the horizon limit at $\theta = -70^{\circ}$ and a second bound on θ at $+60^{\circ}$. This latter bound is due to the 30° flair angle of the solar cell panels at the rear of the satellite. It is not certain that total blockage will result at this angle, but even if total blockage does occur it should not significantly impair the operation of the system since the minimum standard tracking for STADAN system is 10° above the horizon.

Although not indicated in Figure 26 several items deserve attention. Antenna efficiency is not shown, but efficiencies of 90% are easily obtained. This efficiency yields an additional loss of about 0.5 db. The use of half-wave dipoles produces a variation in gain of about $\pm$ 0.5 db for some viewing angles. The effect of interference of the antenna field with induced fields from the spacecraft and spacecraft protrusions cannot be determined by analysis. The measurement of model patterns will be necessary to completely determine these effects. However, since the critical command link has a calculated range of signal-to-noise ratio of 20 to 30 db, pattern nulls of 10 db can be tolerated.

It is proposed that the S-band antenna be a halfwave boxed-in slot* antenna. The long axis of the dipole will be oriented along the direction of flight. The slot will be located on the bottom of the spacecraft. Approximately hemispherical coverage will result but the pattern will be scalloped due to the finite extent of the mounting sheet. The depth of the scallops and the effects of the spacecraft configuration on the pattern can best be determined by model measurements. As with the command system, adequate signal-to-noise ratio exists to accommodate null depths of 10 db.

10.5 GROUND DATA HANDLING

The 20 channels of data will be received at the ground terminals and stored on magnetic tape in analogue form for analysis. Ground equipment will require filters and discriminators compatible with the IRIG Constant Bandwidth Subcarrier Channels (A Channels) to separate the data channels. A nonstandard filter discriminator must be procured for that data which utilizes two IRIG channels. A compatible telemetry decommutator will also be required to present readout of the housekeeping data. Experience has shown that immediate readout of the sensor data is important to the success of an experiment. Therefore, provision should be made for sensor readout at the ground stations in addition to the magnetic tape recordings.

10.6 COMMUNICATION SYSTEM

The complete communications system is shown in Figure 27.

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*Kraus, J. D., Antennas, McGraw-Hill, pp. 353-373.

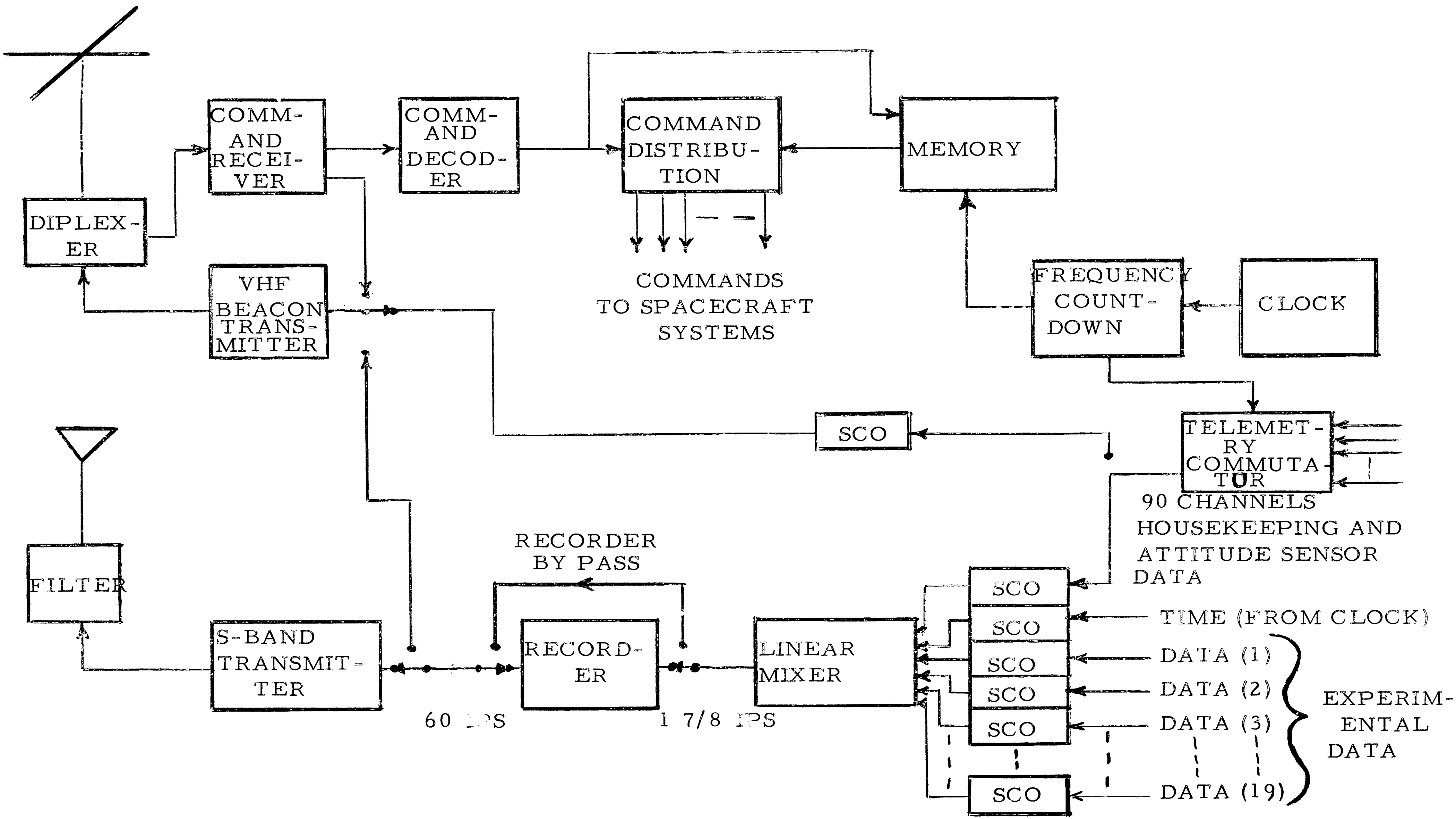


FIGURE 27. COMMUNICATION AND DATA SYSTEM

11
INSTRUMENT PLACEMENT

In placing the instruments in the satellite, dynamic and static restrictions which have to be adhered to due to the nature of the launch vehicle were taken into account. The static restrictions are 12 oz-in and the dynamic considerations are 200 oz-in-in. Also accessibility and position of related components are considered. Due to size restrictions, the batteries for the vehicle are placed inside the main support tube. All other instruments are either mounted on the front surface, side panels, or the main mounting platform. Three packages are mounted on the front surface of the vehicle: the quadrupole mass spectrometer, the retarding potential mass analyzer, and a sun sensor. The remaining sensors, antennae, and magnetometers are mounted on the side panels of the vehicle. All other instruments are directly mounted to either the top or bottom of the main mounting platform. It should be noted that the mass spectrometer is also mounted to the platform by means of three inch diameter tube. This gives the instrument more structural rigidity.

It is necessary for some equipment to be stacked because of the lack of mounting room. The hardest device to mount is the STEM canister and the canister for the 5" diameter mass to which it is attached. The STEMS and canisters must be both thermally and electrically insulated, because they act as heat leaks and also because the booms serve as sensors for two experiments. In order for the system to fit into the vehicle, the STEM canister is mounted upside down. The canisters main structural support is the main support tube to which it is welded. Additional angles, brackets, and fasteners are deemed necessary so that the system is able to sustain the experimental 25 G axial loading.

The canisters for the masses are both mounted to the STEM canister and to the side panels. On the following pages are sectional diagrams showing the placement of the instruments and the fastening of the booms into their positions.

11 .1 FASTENING TECHNIQUES

Two types of fasteners were considered for STRATUM's particular needs: those with internal threads and those with an external nut. A grommet-type fastener with external nut was chosen because it is structurally superior and easier to remove should the instrument malfunction. The fasteners are made of aluminum and can be manufactured for any thickness of honeycomb. Below is a schematic diagram of a fastener.

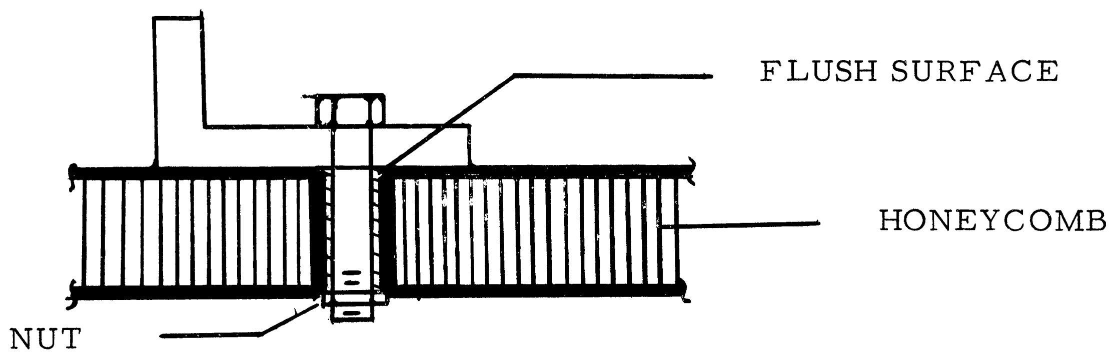


FIGURE 28. HONEYCOMB FASTENING TECHNIQUE

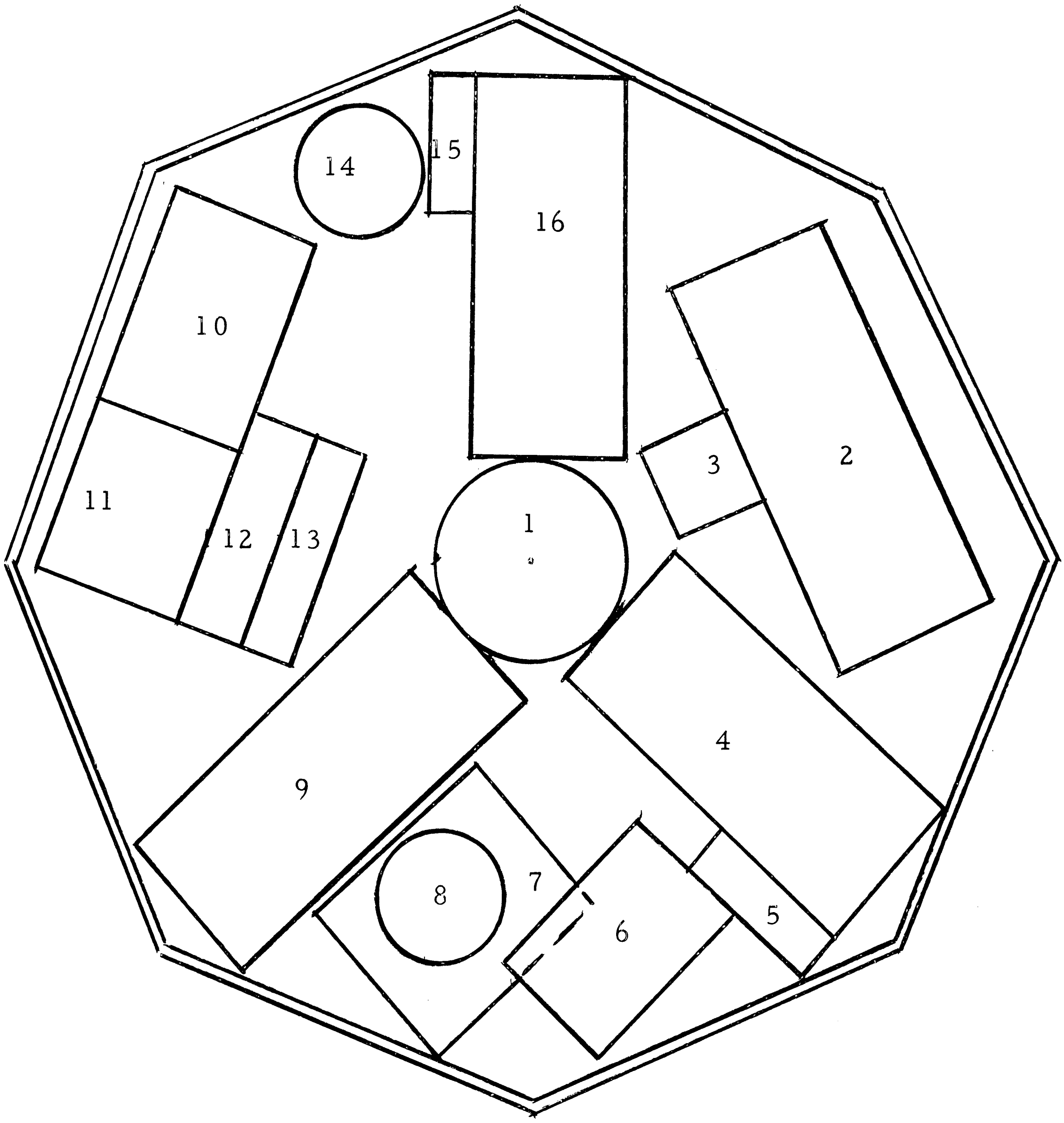


FIGURE 29 . MOUNTING PLATFORM TOP SURFACE

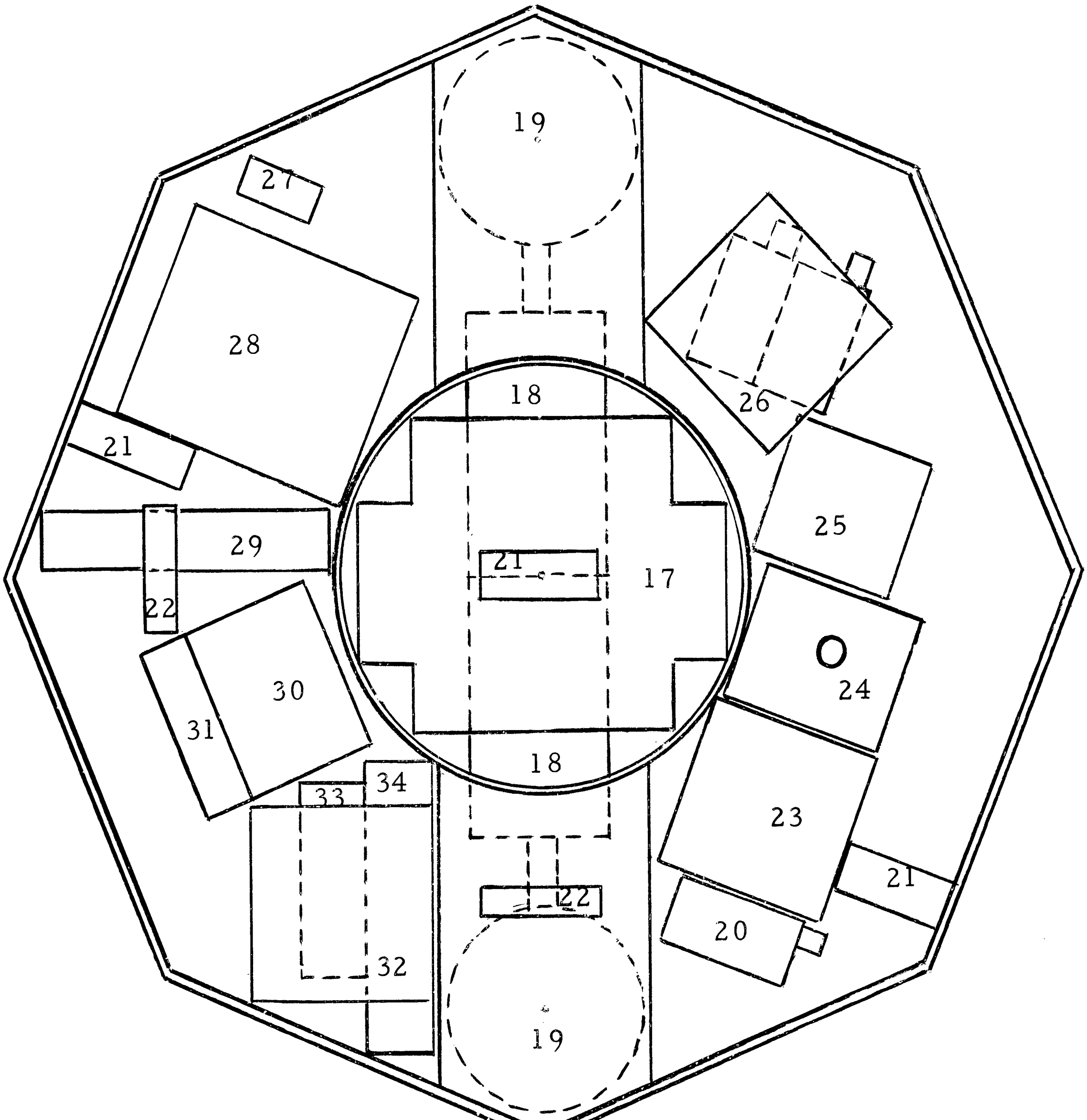


FIGURE 30 . MOUNTING PLATFORM BOTTOM VIEW

11.2 Parts key to Figures 29, 30.

1. Mass spectrometer
2. Experiment # 1 electronics
3. S-band transmitter
4. Tape recorder
5. Diplexer
6. IR sensor
7. Experiment #2 electronics
8. Experiment #2
9. Tape recorder
10. Command distribution unit
11. Experiment #1 electronics
12. Tape logic
13. Tape logic
14. Beacon transmitter
15. Tape logic
16. Tape recorder
17. Batteries
18. Boom deploy canisters
19. Tip mass and magnetic damper
20. Magnetometer electronics
21. Sun sensors
22. Magnetometer sensor
23. Telemetry commutator
24. Command memory
25. SCO and mixer amplifier
26. Power conditioning unit
27. Sun sensor electronics
28. Command receiver
29. Langmuir probe
30. Clock
31. Electronics for experiments 4 and 5
32. IR horizon sensor
33. Photometer electronics
34. Photometer sensor

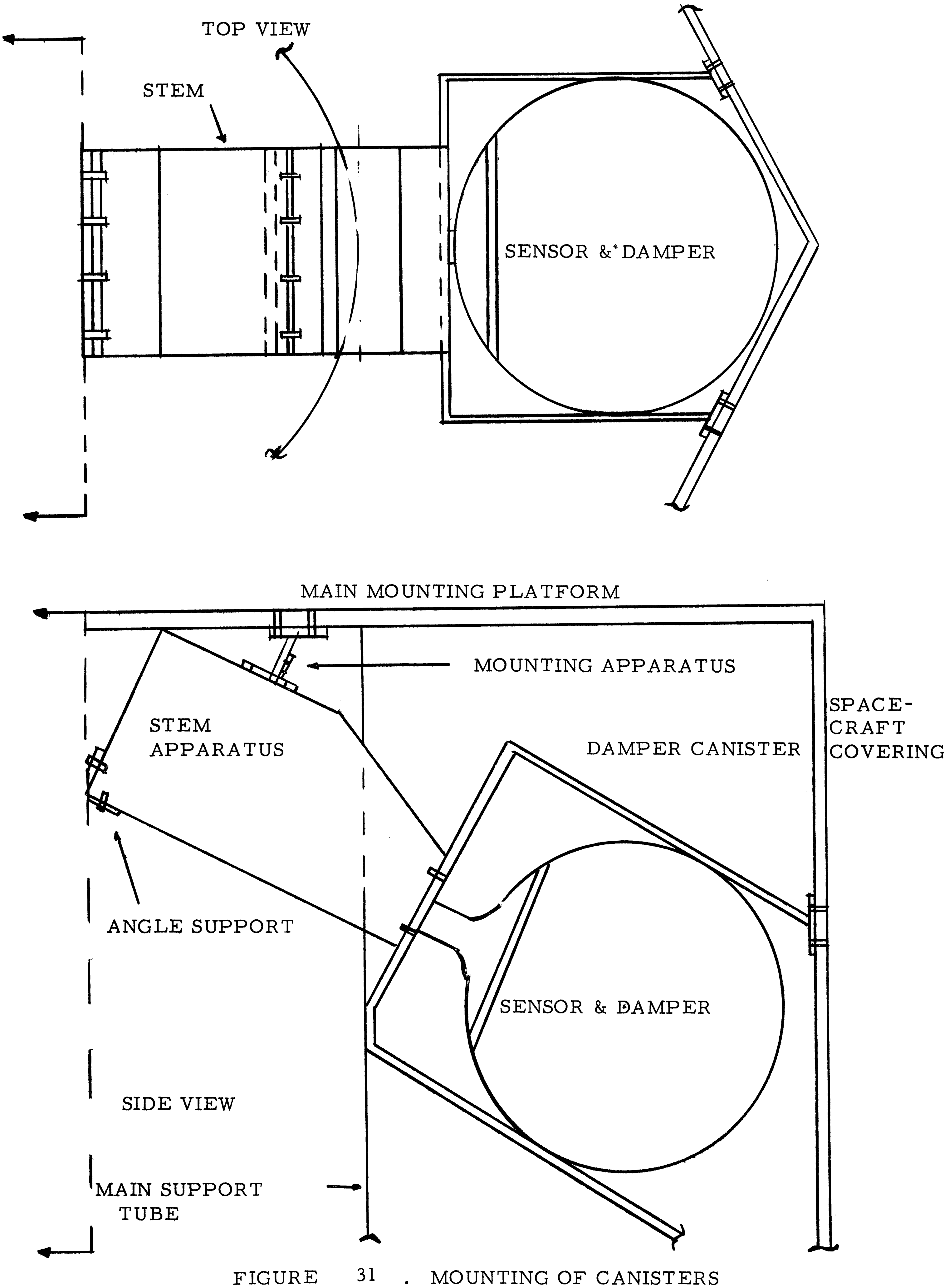


FIGURE 31 . MOUNTING OF CANISTERS

12
LAUNCH VEHICLE

12.0 INTRODUCTION

Project STRATUM is designed as a low-cost, reliable satellite system. For this reason, it is desirable to have an economical booster with the capability of injecting STRATUM into the prescribed orbit. On this basis, the Scout launch vehicle was chosen. Significant details concerning STRATUM deal with the vehicle-payload interfacing and the launch facilities. Other, more detailed specifications of the Scout are readily obtainable and present no restraints on STRATUM's mission.

The Scout is a four-stage solid propellant rocket booster. Although it is the smallest standard launch vehicle in the United States, the Scout is capable of injecting a 250 lb payload into a 295 nautical mile, near-polar orbit. It is a relatively inexpensive booster costing about 1.5 million dollars. The cost includes the launch facilities and the payload interfacing service. The present Scout configuration has been successful in thirty of thirty-three launches, proving it reliable.

12.1 VEHICLE-PAYLOAD INTERFACE

Although the Scout is adaptable to a wide variety of payloads, certain restraints are imposed on it. These include the size limitation of the spacecraft due to the heat shield and basic "E" section, the accessability to the payload through the shroud, the separation mechanism, and the launch environment and facilities. All of these areas influence the design of the satellite.

12.1.1 Heat Shield

The Scout heat shield imposes the size restriction. This is schematically shown in Figure 32.

Accessibility to the payload once it is located in the shroud is important for final checkout and/or replacement of instruments. For this reason there are doors on the heat shield. STRATUM plans five doors provided this does not interfere with the structural integrity of the shroud.

12.1.2 Separation System

The separation system is included in the basic "E" section shown in Figure 32 . The fundamental parts of this system are the separation springs, the spring retainer ring, and the batteries and timers. The payload

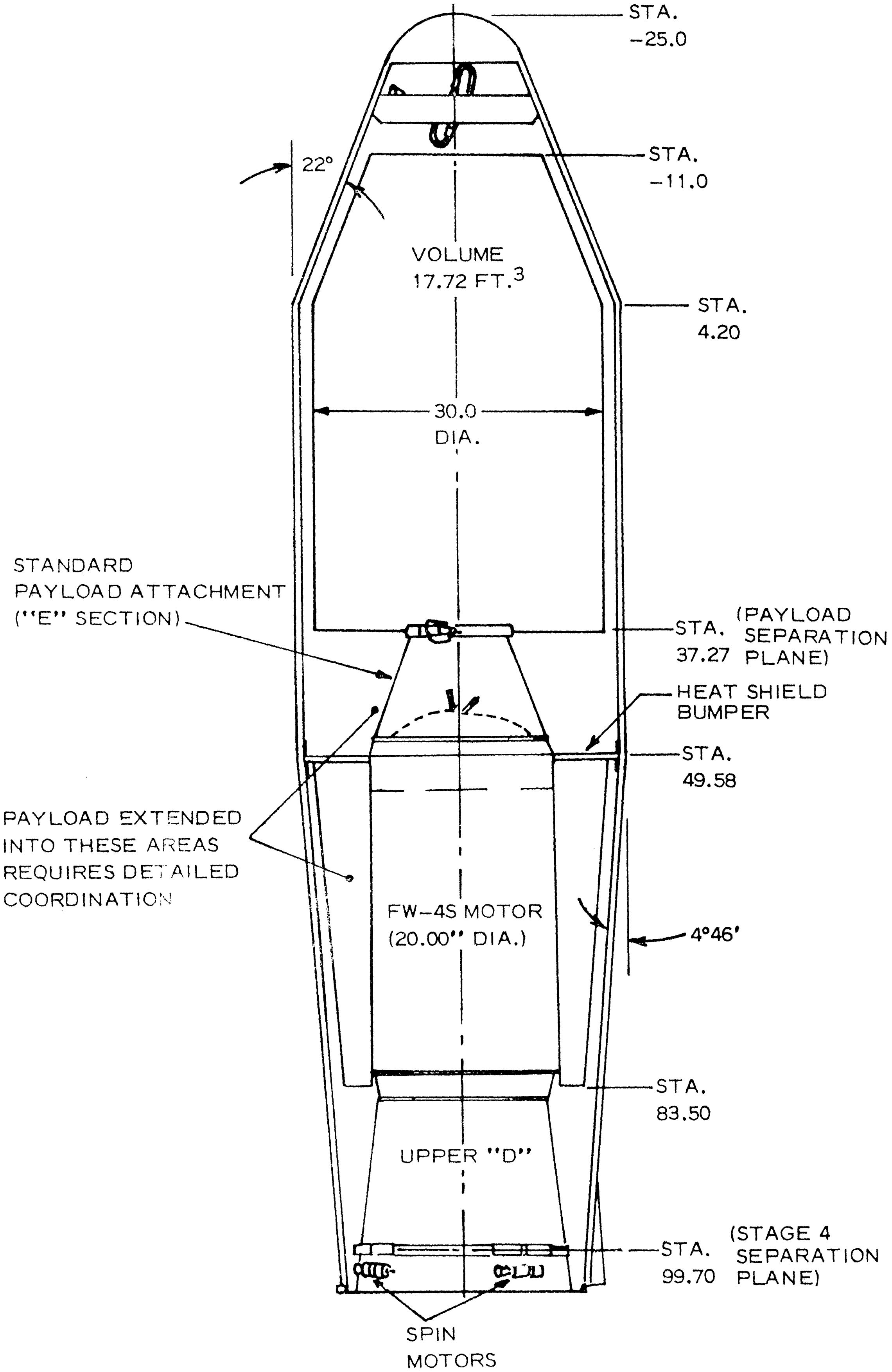


FIGURE 32. SCOUT PAYLOAD ENVELOPE

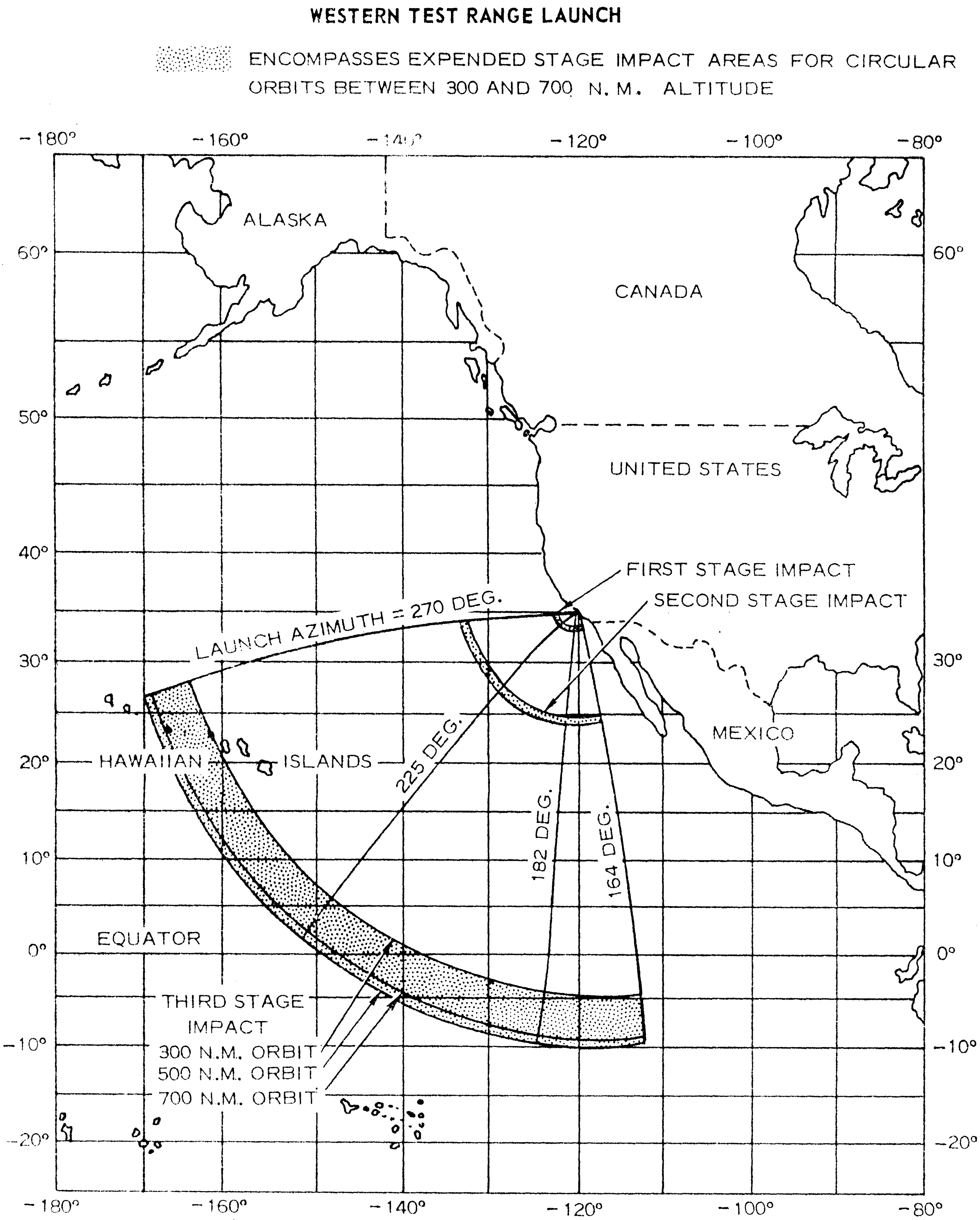


FIGURE 33. EXPENDED STAGE IMPACT AREAS - WTR

support ring which is attached with bolts remains with the satellite in orbit. Marmon clamps firmly hold the support ring to the fourth stage structure. The separation springs disengage the spacecraft at the prescribed time, giving it the necessary added velocity to prevent collision with the expended fourth stage booster.

Prior to launch, the entire system is armed through remote control from the blockhouse using a fly-a-way umbilical attached to the "E" section.

12.1.3 Launch Facilities

STRATUM will use the Western Test Range for the pre-launch specification tests and for the actual launch, since it is the only facility capable of injecting satellites into polar orbits. The range and the flight path of the Scout is presented in Figure 33.

12.1.4 Launch Environment

Environmental control differing from the $70^{o}F$ of the Scout shelter can be maintained by keeping the heat shield around the payload. The heat shield is purged and cooled or heated with filtered, oil-free air provided by an environmental system. Within the heat shield, temperatures may be maintained from $25^{o}F$ below ambient to $150^{o}F$ maximum, at approximately $-40^{o}F$ dew point.

13
YO-YO DESPIN

13.0 INTRODUCTION

Since the Scout fourth stage spins the payload up for stabilization purposes, a method of decreasing the spin rate must be considered in order for the solar cell paddles and the gravity gradient booms to deploy. The only entirely passive system that will do this is the yo-yo despin system. It is economical, lightweight, and requires no sustained power. The principle involved is an increase of the vehicle's spin moment of inertia over a short period of time.

13.1 YO-YO MECHANISM

A system of two weights attached to the spacecraft by two cables which wrap around the circumference of the vehicle provdes this increase. After the fourth stage motor has burned out, the two weights are released by firing squibs contained in small mechanical motors. The weights "unwind" in a tangential direction so that when all cable has payed out, the moment of inertia is large enough to slow the satellite's spin rate to zero RPM. The other end of the cable is released by a hook and ring arrangement when the moment of inertia reaches its maximum. See diagram in Appendix G for details of release mechanism. The fourth stage of the launch vehicle will be separated before despin. There are no large tumbling phenomena expected after separation. The yo-yo despin mechanism cannot correct for coning if it should be present. Therefore, the gravity gradient booms will be deployed slowly enough (1 - 2 inch/sec) so as to reduce coning motion. The total weight of the despin cable, tip masses, and mechanical motors has not been determined, because the vehicle's spin moment of inertia is not known at this time. The total spin moment of inertia has been graphed versus the cable and tip mass weight so that when the spacecraft's spin moment of inertia is established, the weights can be determined (see Appendix G).

13.2 RECAPTURE

There is a possibility that the spacecraft could be captured upside down. The process of re-capture, or inverting the satellite is planned as follows. The action of the spacecraft revolving about its pitch axis once every orbit makes it possible to recapture in the desired orientation. This is done by retracting one boom and leaving the other fully extended. The speed of rotation of the satellite is increased by reducing the moment of inertia about the pitch axis. When the satellite has traversed one half an orbit, and has made one half a revolution about its pitch axis, the boom is extended again recapturing the satellite in its new orientation.

14
RELIABILITY ANALYSIS

A complete reliability assessment can be made for a system that has all of its components specified. For a feasibility design study, however, several instruments may be suggested for a single subsystem component, so that Failure Modes and Effects Analysis (FMEA) must be used to investigate system redundancy and flexibility. In this section, FMEA is limited to the communications, data processing, and command system. The reliability logic diagram is presented for that system along with a listing of the most serious failure modes.

The reliability logic diagram in Figure 34 outlines the functional interdependency of the components, and shows where redundancy is and is not provided. The two antennas, the diplexer, and the filters are passive elements, and are not redundant because of their high reliability. The beacon transmitter can be used if the S-band transmitter fails, and the S-band signal can be used for tracking if the beacon transmitter fails, and so neither of these components have back up units. However, as the system is further developed, a second beacon transmitter should be considered because of its low weight, volume, and power requirements. The clock-frequency divider combination, while critical for mission success, can be made with high reliability. The memory and the command distribution unit are the remaining components for which redundancy has not been provided. The command distribution unit is the more critical of the two because the failure of one relay to operate causes complete loss of a particular function. For a memory circuit failure, only delayed command capability is lost. Therefore, the failure modes of the command distribution unit require careful analysis as the explicit commandable functions of the system are chosen.

The most informative partial and complete failures of the system are listed in Table 3 with detection methods and final effects considered. Some explanation is required for the failure modes of the SCO-linear mixer combinations. There are two redundant series of 21 SCO units, and two redundant series of 21 linear mixer units. Single SCO (linear mixer) unit failures in each SCO (linear mixer) series may or may not effect the same channel of data. Therefore, the decision of which defective series to use is based on the channels needed for a particular experiment.

The failure of the heat pipes has also been considered. Normally functioning heat pipes permit approximately 10% solar cell efficiency. On the other hand, the result of heat pipe failure is a 1/3 reduction of power available because of the increased temperature of body mounted solar cells. Thus, in addition to effects on payload environment, the duty cycle would be significantly reduced.

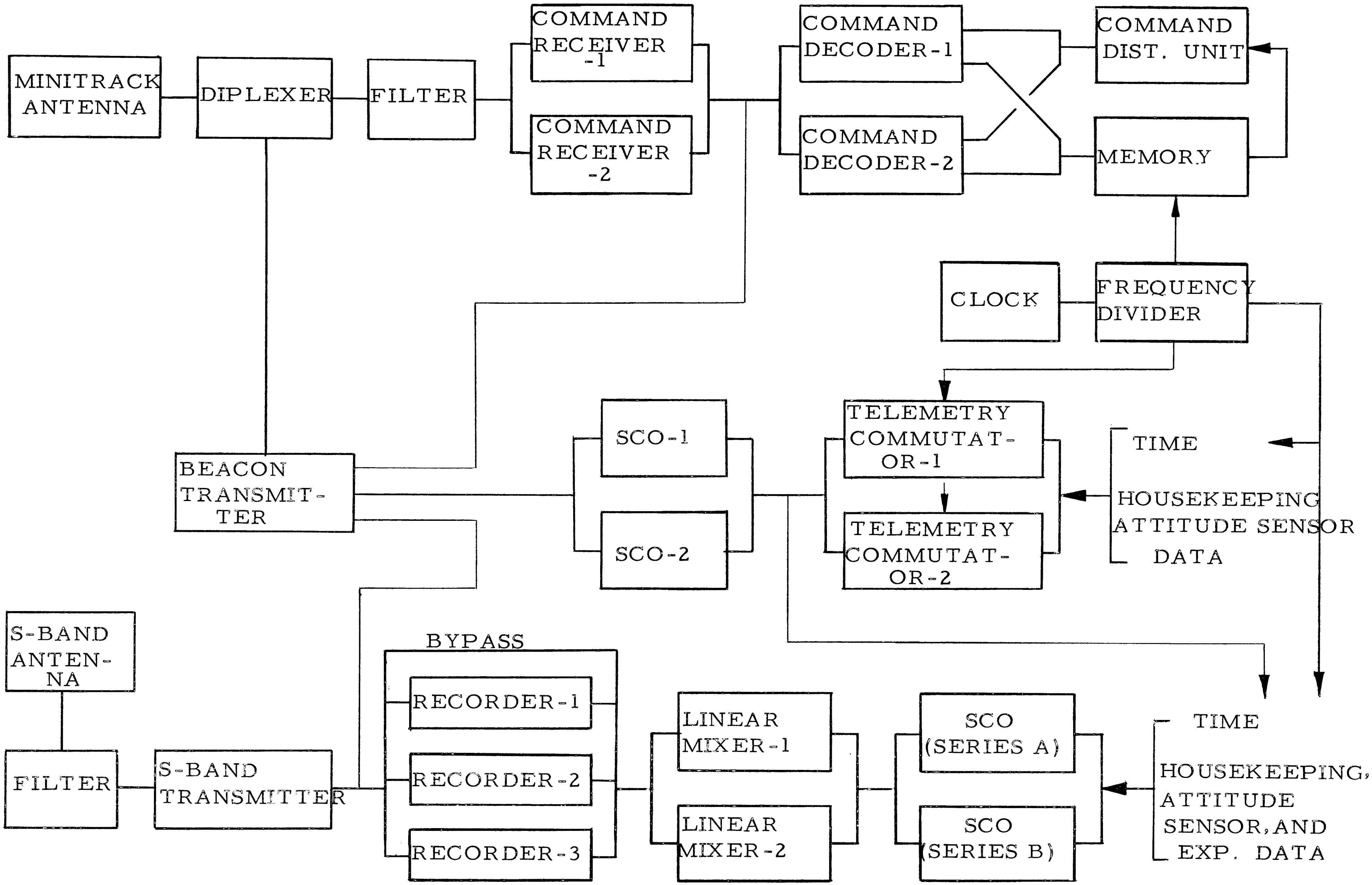


FIGURE 34. RELIABILITY LOGIC DIAGRAM - COMMUNICATIONS, COMMAND, AND DATA PROCESSING SYSTEM

Failure of	Detection by	Corrective action and final effect
S-band Transmitter	lack of S-band signal at ground stations	command change to beacon transmitter for data transmission; forced to use Minitrack ground receiver for data
Beacon Transmitter	lack of tracking signal at ground stations	forced to use S-band signal to track spacecraft; no return of commands received for verification
Recorders 1, 2, and 3	failure of recorder to play back correctly, or at all	command change to recorder bypass; limited to real time data recording
Command Receivers 1 & 2	no return of commands through beacon transmitter (if operating)	mission abort
Command Decoders 1 & 2	return of commands through transmitter (if operating) but no response to any command	mission abort
Memory (one word position)	failure of a delayed command to execute; possibly detected on recorder playback record	dependent on command in particular word location
Command Distribution Unit	lack of response to a command; possibly detected on recorder playback record	dependent on function controlled by particular relay
SCO series A and B (one or more channels of each series)	lack of one (or more) data channels on recorder playback	command "on" SCO series with best remaining choice of channels; effect dependent on combination lost
Linear Mixer 1 & 2 (one or more channels of each series)	lack of one (or more) data channels on recorder playback	command "on" Linear Mixer with best remaining choice of channels; effect dependent on combination lost
Clock	lack of time, attitude, and housekeeping data, no execution of delayed commands	loss of time, attitude, and housekeeping data; loss of delayed command capability
Frequency Divider (one or more frequencies)	improper delay time of delayed commands; lack of time, attitude, and housekeeping data	dependent on frequency affected

15
PROGRAM AND COST ANALYSIS

15.0 INTRODUCTION

The success of Project STRATUM depends on the coordination by NASA of the experimenter, the spacecraft contractor, and the launch vehicle contractor. The program plan developed for the project must consider the technical requirements of the spacecraft, the resources of possible contractors, and the launch date desired by the experimenter. Reliability considerations further dictate that design, fabrication, and testing periods be of sufficient length to insure high probability of success. Finally, it is economic factors that partially determine the existence of the program, the contract dates, and the progress rates.

15.1 PROGRAM DEVELOPMENT PLAN

Figure 35 summarizes the STRATUM development schedule beginning in mid-1967 and leading up to a late March 1970 launch. This schedule is intended for planning purposes only, with the major variable being the contract award date. The development phases of the project are outlined below, and are compatible with the needs listed above.

15.1.1 Phase A: Conceptual/Feasibility Study

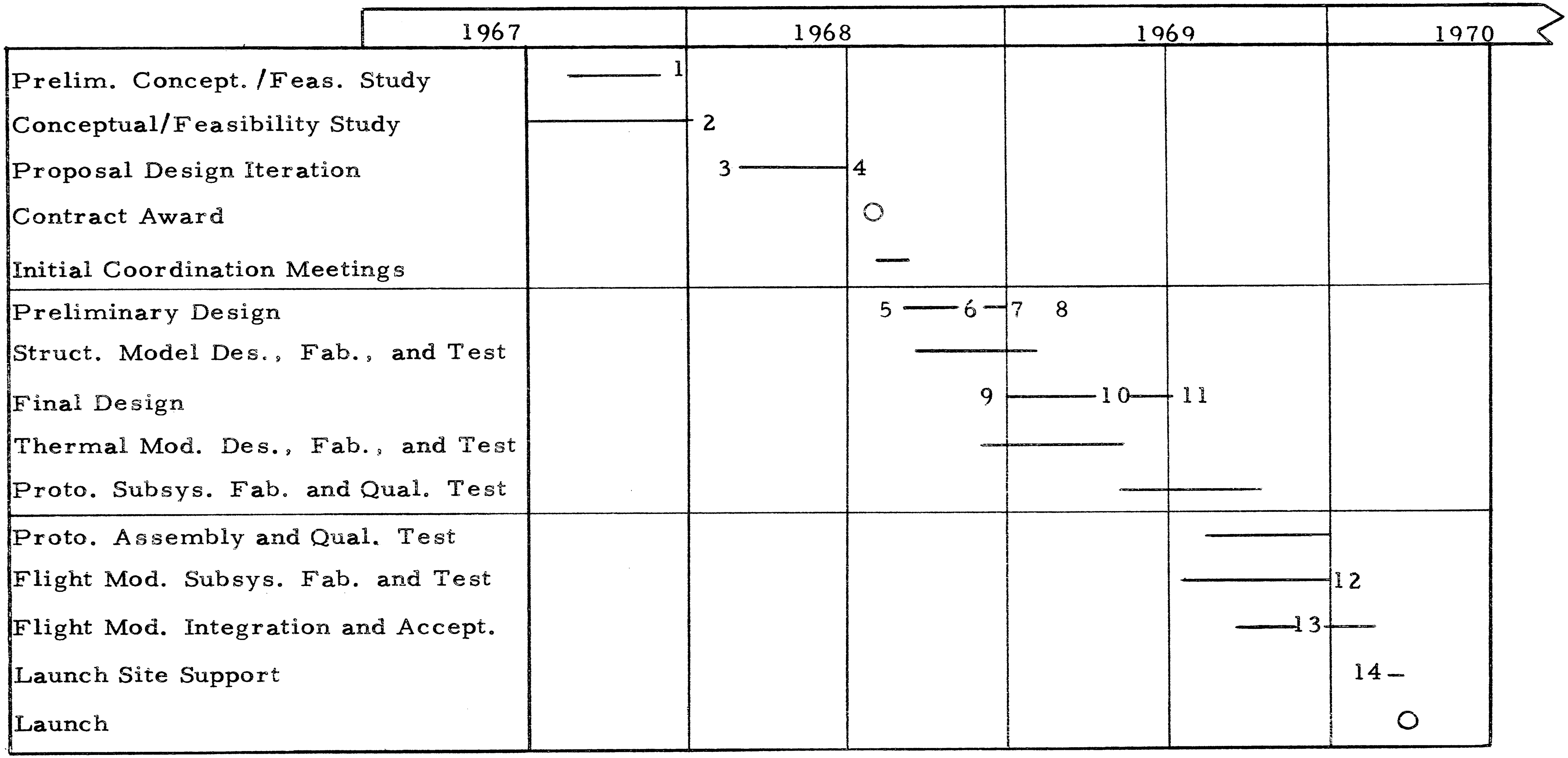
This report is the result of Phase A. Engineering assessments have been made to identify technical requirements, recommendations have been made, and gross scheduling and cost estimates have been outlined.

15.1.2 Phase B: Preliminary Definition

Preliminary definition studies are conducted by prospective spacecraft contractors following the Request for Proposal. These studies include:

1) Assessment of total mission requirements.
2) Preliminary design data (including preliminary system specifications).
3) Preliminary assessment of manufacturing and test facilities and techniques.
4) Estimation of resource requirements.
5) Identification of advanced research and technology required.
6) Assessment of experiment/spacecraft interface problems.

The identification of a single approach for Phase C is then made and is summarized in the final proposal.



1. Submit Prelim. Conceptual/Feasibility Study
2. Submit Conceptual/Feasibility Study
3. Request for Proposal
4. Submit Proposal
5. Prelim. Experiment Requirements
6. Design Review Meeting
7. Submit Preliminary Design Report
8. NASA Approval
9. Freeze Experimenter-Contractor Interface
10. Contractor Receipt of 1^{st} set of Experiments
11. Submit Final Drawings
12. Ground Support Equipment Complete
13. Contractor Receipt of Flight Experiments
14. Deliver Spacecraft

FIGURE 35. STRATUM DEVELOPMENT SCHEDULE

15.1.3 Phase C: Final Definition and Development

After Contract Award, requirements of the experiments are updated and the results of a short preliminary design iteration are submitted to NASA for approval. Structural and thermal models (and all recommended breadboards) are designed, built, and tested as the final designs of the prototype and flight models are produced. Subsystem fabrication and test is followed by assembly and test for the prototype and flight models. The flight model is launched approximately 21 months after Contract Award.

15.1.4 Phase D: Operation

This phase requires the support of the experimenter and spacecraft contractor, and will continue until spacecraft re-entry or until complete spacecraft failure. Experimental data is collected and the condition of the supporting subsystems is closely monitored.

15.2 COST ANALYSIS

The cost of Project STRATUM can be broken into the following four categories:

1) The cost of experiments.
2) The spacecraft contract
3) The launch vehicle contract
4) Orbital support.

15.2.1 Experiments

The University of Michigan High Altitude Engineering Laboratory is the experimenter, and will be responsible for the development, fabrication and testing of the experiments. Some of the experiments listed in this report have been flown. Others are presently in the development stage so that much of the development cost has already been funded. The cost to build each set of experiments has been estimated and is given below.

Experiment	Estimated Dollar Cost *
Quadrupole mass spectrometer	30, 000
Retarding potential mass analyzer	15, 000
Langmuir probe	10, 000
Resonance probe and ion spectrometer	15, 000
Photometer	5, 000

* Does not include costs of calibration or environmental testing

15.2.2 The Spacecraft Contract

The cost of the spacecraft has been estimated based on the development schedule outlined in this section. The contractor is assumed to be a private corporation with adequate experience and test facilities. Other assumptions are listed below.

1) Cost plus fixed fee contract.
2) Burden rate: 125%
3) (General and administrative rate: 15%) plus (independent research and development rate: 5%) = G&A/IR&D = 20%
4) Cost of one engineering man month: $3,200

Table 4 lists the cost breakdown. Each subdivision includes the costs of direct labor, burden, direct material, G&A/IR&D, and other direct costs such as travel.

15.2.3 The Launch Vehicle Contract

LTV is the launch vehicle contractor, and will provide the Scout launch vehicle and launch site support (including the mating of the spacecraft to the Scout) for 1.5 million dollars.

	Estimated Dollar Cost
Structure	180, 000
Thermal Control	120, 000
Attitude Control	60, 000
Power	240, 000
Telemetry and Command	1, 150, 000
Ground Support Equipment	210, 000
Test Plans and Procedure	65, 000
Integration and Testing	280, 000
Reliability	130, 000
Quality Control	80, 000
Systems Engineering and Support	310, 000
Management and Program Control	220, 000
Technical Reports and Manuals	55, 000
	3, 100, 000
Fee (10 percent)	310, 000
Total Cost (Spacecraft Contract)	3, 410, 000
Total Cost (Launch Vehicle Contract)	1, 500, 000
	4, 910, 000

TABLE 4 . COST ANALYSIS

APPENDIX A

JUSTIFICATION OF CONSTRAINTS AND SELECTION OF INSTRUMENTS

The project STRATUM group had responsibility for the development of a system to support an aeronomy experiment. The selection of the instruments was made by members of the University faculty involved in aeronomy research. Since their decisions introduced constraints upon the system, it is appropriate to give some insight into their rationale. This is not meant to be a study in depth, but a survey of the highlights.

The major constaints on orbital parameters follows:

1) Orbit must be near polar
2) Orbit must be circular
3) Orbit plane must rotate 180 degrees with respect to earth-sun line each quarter year.
4) Satellite must have one year lifetime.
5) Satellite must have lowest possible altitude (within the limits of 1-4).

A near polar orbit is the only means of obtaining complete coverage of the earth. A circular orbit will yield information which is more meaningful and easier to interpret than an eccentric orbit which would introduce the new variable of altitude. Have the orbit plane rotate 180 degrees every quarter year with a satellite lifetime of one year provides full earth coverage for each season. Finally, it is desirable to have a low altitude because density will be higher and the measurement of desired quantities will be simplified.

Following is a general argument for the selection of the specific instruments. The absorption of extreme ultra violet radiation from the sun by the earth's atmosphere occurs mainly in the altitude region 100 to 300 km. The radiation below 2000 Å, which is only a minute fraction of the sun's total, accounts for nearly all of the heating and photochemical effects which occur in the lower thermosphere.

There are several important problems concerning the thermal profiles and heat balance of the thermosphere. Temperatures at the lower bound, at least on an average basis, are known from rocket experiments. The fact that at high altitudes, say above 300 kilometers, the temperature must be isothermal has been established by consideration of the time-dependent heat conduction equation and the very long mean free paths, which shows it to be impossible to maintain thermal gradients for more than a few hours. From satellite drag, the magnitude of the isotherm and the thermal gradient between the bounds are known generally, as is the fact that the atmosphere bulges upward in the daytime with a maximum at 2 PM local time. Satellite drag has even shown us that there are two bulges of different composition and phase.

The fine detail of these structural changes and the specific identification of casual phenomena, however, require measurements with time resolution of the order of an hour. This type of measurement has not been accomplished with drag methods. The approach, in the absence of an instantaneous neutral temperature method, is to measure composition with good time resolution using a spectrometer. Measurements in the thermosphere show that the density/thermal bulge occurs 3 hours earlier than current models predict. It has been calculated that horizontal convection and molecular conduction can account for the phase lead and that horizontal convection, with subsidence, of helium can account for the helium bulge. It is thought also that atomic oxygen is involved in similar patterns which affect energy distribution. Thus, mass spectrometric observations on a global basis can confirm these theories by establishing the postulated composition distributions.

The measurement of horizontal distributions of composition is of particularly timely consequence with respect to analytic models of the neutral atmosphere. Ten years ago the advent of satellites and the observation of drag brought a rush of data which was finally treated by comparison with estimated profiles constructed step-wise from time-invarient solutions of the conduction and diffusion equations. Four years ago these equations were successfully integrated numerically by computer, thus permitting a systematic introduction of the source functions covering solar and atmospheric radiation. The computers available permitted handling only vertical variations, however. Recently, with a new computer, a three-dimensional model has been achieved resulting in the analysis of horizontal distributions of species now exceeding the availability of experimental data, and new measurements are urgently required. In order to exploit the improved modeling technique, a circular orbit is suggested. A circular orbit is thought to be most useful, because experience with satellite data shows that the so-called vertical scans obtained with eccentric orbits, being made up of points separated in time are misleading and prevent the achievement of good horizontal scans.

The structural parameters of the atmosphere (pressure, temperature, density and composition) are related to photochemical phenomena. For example, measured concentrations of electron density in the noon F_2 maximum are possible only in an atmosphere with a low concentration of molecular gases (highly dissociated). Further, the high altitude of the night-time F_2 maximum in the presence of the contraction of the neutral atmosphere can be accounted for only by including an additional source of electrons. It is believed that this source is primarily a downward flux of the electrons and hence ions stored in the lower magnetosphere. However, this question is intimately related to the general composition of the neutral atmosphere and cannot be discussed without knowledge of its behavior. Similarily, during the day the electron gas takes on its own characteristic temperature due to the production of energetic photo-electrons created by solar extreme ultra-violet ionizing of the neutral atmospheric gases. Here the neutral atmosphere plays an important role being both the source and the ultimate sink for the energy stored in the hot electron gas. The various ion species also enter into this process, since the electrons tend to transfer their energy to ions due to the large coulomb cross section. As a consequence the various ions tend to have their own characteristic temperatures lying between the neutral gas temperature and the electron temperature.

These processes are illustrative of the intimate relationships which unite all the physical phenomena occuring in the thermosphere. One cannot consider the ionosphere or the neutral atmosphere as separate entities. They are both manifestations of a single complex physical process. Nor can one ignore dynamic versus photochemical processes as has been done in the past. The complexity of the causal links between all of these processes make it necessary to utilize an experimental approach rather than rely on theoretical predictions alone. In this way one can examine in detail the temporal and spatial variation of specific phenomena and hopefully separate the dominant physical processes.

One must, however, always combine sufficient information to successfully identify the major cause and effects. Any experimental study of the atmosphere must rely on a complement of experiments, and not be directed toward a single major observation. The basic information required must include at least the following complement of measurements:

1. Composition

   a. Neutral gas
   b. Ion gas
   c. Electron density

2. Energy budget

   a. Electron temperature
   b. Ion temperature
   c. Neutral temperature

3. Airglow

   a. Atomic Oxygen (5577 Å, 6300 Å)
   b. Molecular Nitrogen (3914 Å)

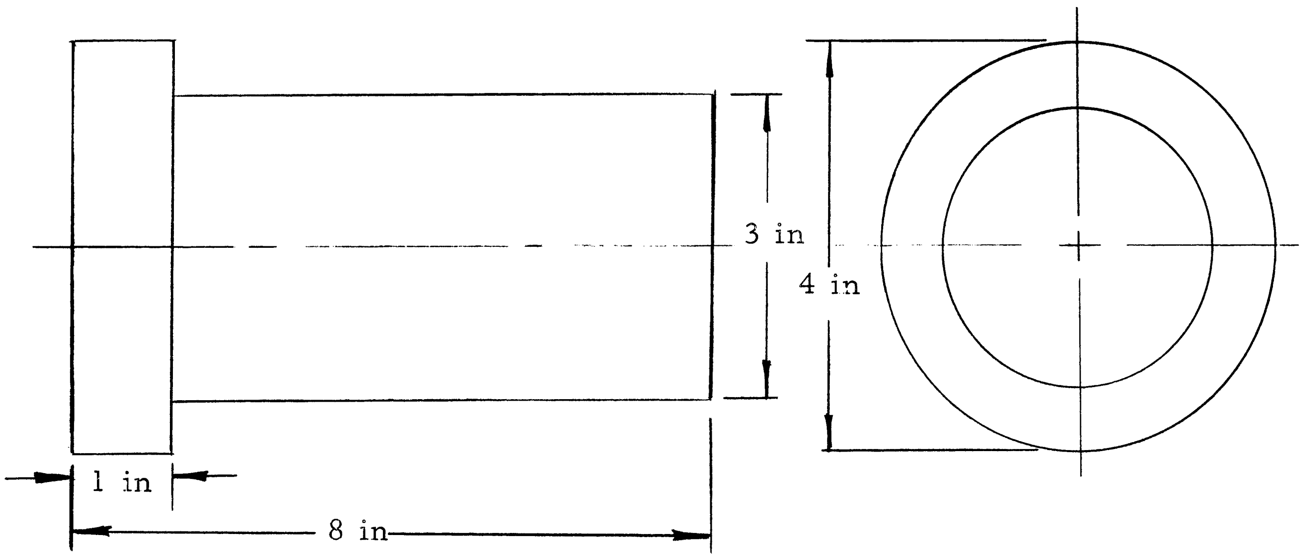
The addition of the airglow complement has been made to illustrate the importance of excited species in determining the energy budget and the ultimate fate of the energy deposited in the thermosphere. All three of these airglow spectral regions are related to fundamental physical processes and will be of vital interest in understanding the behavior of the atmosphere. The atomic oxygen 6300 Å emission is a monitor of the dominant recombination processes near the F_2 peak at night, the 5577 Å is related to low level atomic oxygen recombination and is strongly influenced by the dynamics near the lower boundary of the thermosphere, and 3914 Å is a direct measure of the role of high energy electron precipitation in the energetics of the lower thermosphere. Any justification for these particular choices is relative, but these do represent the major airglow processes and should be included in the proposed experimental package.

APPENDIX B

INSTRUMENT SPECIFICATIONS

<u>Experiment No. 1</u> Quadrupole mass spectrometer, neutral and ion mode

Size: 8" x 8" x 7" (combined sensor and electronics)
Wt: 8 lbs
Sensor only



Wt: 2 lbs
Electronics card size 6 1/2" x 7"
Note: Total volume must equal 8" x 8" x 7" minus sensor volume
Wt: 5.5 lbs

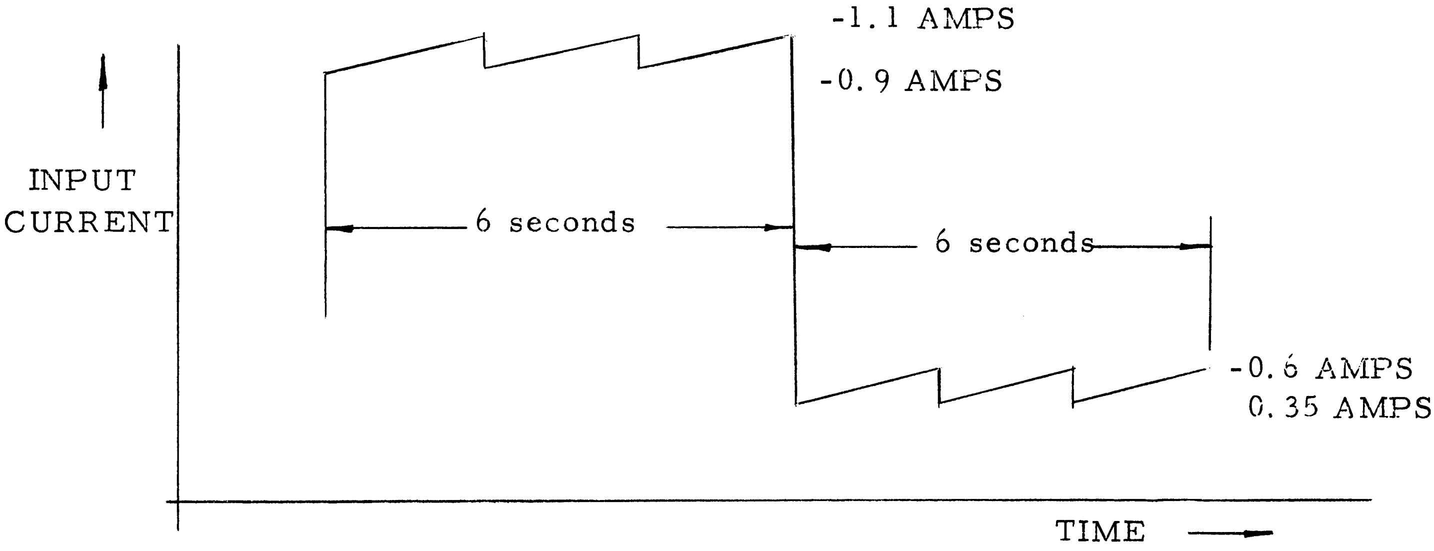
Data outputs:

2 Spectrom. Chans. 100 s
1 Monitor Chan. 30 s

Commands

1. Fil selection
2. Standby
3. On-off
4. High voltage
5. Cover deploy
6. Ion mode only

Power profile



Average power: 21 watts at 28 volts.

Experiment No. 2 Retarding potential mass analyzer.

Sensor size: 2.0" dia x 4" long
Sensor wt: 2.0 lbs
Electronics size: 4" x 4" x 4"
Electronics wt: 2.0 lbs

Data outputs:

1 Current Channel 0-15 cps, $\pm$ 1%
1 Range Channel 0-15 cps, $\pm$ 1%
1 Monitor Channel 0-3 cps, $\pm$ 1%

Commands

1 On-off
1 Ion mode
1 Electron mode
1 Sequenced mode
1 ∇V Command

Power 2.8 watts at 28 volt steady.

Experiment No. 3 Langmuir Probe

Size: 6.55" x 4.18" x 1.24"
Wt: 1.80 lbs

Data outputs

1 Current chan	0-16 cps	± 1%
1 Ramp sample	0-16 cps	± 1%
1 Monitor chan	0-3 cps	± 1%
1 Current range	0-16 cps	± 1%

Commands

1 On-off
1 ∇ V Mode

Power 2.4 watts at 28 volts steady.

<u>Experiments No. 4 and 5.</u> Resonance probe and ion spectrometer

Size: 4" x 4" x 4" (2 boxes)
Wt: 2 lbs each box

Data outputs:

1 Ant current	0-15 cps	± 1%
1 Ant voltage	0-15 cps	± 1%
1 Resonance detection	0-100 cps	± 5%
1 Excitation frequency	0-15 cps	± 1%
1 Monitor	0-3 cps	± 1%

Commands

1 On-off
1 Electron mode
1 Ion mode
1 Sequenced mode

Power 3.0 watts at 28 volts steady.

<u>Experiment No. 6</u> Photometer

Sensor size: 6" dia. x 1.5" x 1.5"
Wt: 2.0 lbs
Electronics size: 3.5" x 3.5" x 1.5"
Wt: 0.5 lbs

Data outputs

1 Photometer output	0-5 cps	± 1%
1 Photometer range	0-5 cps	± 1%
1 Monitor		

Commands

1 On-off
1 H.V. on
1 5577 Å
1 6300 Å
1 5577-6300 Å sequence

Power 3.0 watts at 28 volts steady

General Considerations

A. Attitude control $\pm 20^{\circ}$, three axis
B. Experiment sensor locations:
   #1 Must point into velocity vector.
   #2 Must point into velocity vector.
   #3 Probe axis must be perpendicular to velocity vector.
   #4 & 5 Sensor is 2 booms of attitude control system.
   #6 Must point toward earth.

C. Environmental
   Temp. limits while operating:
   $-20^{\circ}C$ to $+60^{\circ}C$
   Temp. limits while non operating:
   $-40^{\circ}C$ to $+80^{\circ}C$

Vibration must not exceed published levels for Scout launch vehicle.

APPENDIX C

ORBITAL CALCULATIONS

The following is the method of prediction used to determine the lowest altitude attainable for a one year lifetime

$$L_T = L_N (1 + \Delta S_o - \Delta I)$$

where L_T = total lifetime estimate (365 days)
L_N = nominal lifetime based on the ballistic parameter
ΔS_o = percent change (+ or -) from the 1959 ARDC atmosphere as a function of time of launch, taking into account density variations
ΔI = per cent loss due to the diurnal bulge

$$\frac{w}{C_D A_{eff}} = \text{ballistic parameter}$$

where w = spacecraft weight = 233.8 lb
C_D = coefficient of drag = 2.2
A_{eff} = effective frontal area of spacecraft = 14.4 ft^2
13.7 ft^2 actual frontal area + 5% additional area due to spacecraft oscillation.

Thus,

$$\frac{w}{C_D A} = 7.37 \text{ lb/ft}^2$$

Referring to nominal lifetime versus altitude curves in NASA TN D-1995
L_N = 514 days.

ΔS_o = -.15 (constant for late 1969 launch)
ΔI = -.14

Thus,

$$L_T = 514 (1 - .15 - .14) = 365 \text{ days.}$$

This calculation of the predicted lifetime corresponds to an altitude of 295 nautical miles.

Using the precession rate of -.986 degrees/day the inclination can be computed as follows:

$$i = \arccos \frac{-r^{\frac{7}{2}} (1 - e^2)^2 \dot{\Omega}}{J R_o^2 u_e^{1/2}}$$

$r = h + R_o$ = altitude of orbit plus radius of the earth
$R_o = 20.9 \times 10^6$ ft $\qquad h = 1.79 \times 10^6$ ft
e = eccentricity = 0
J = oblateness factor = 1.62×10^{-3}
$\dot{\Omega}$ = precession rate = -.986 degrees/day
u_e = gravitational parameter = 1.40×10^{16} ft^2/sec^2
i = inclination

i = 82.4 degrees

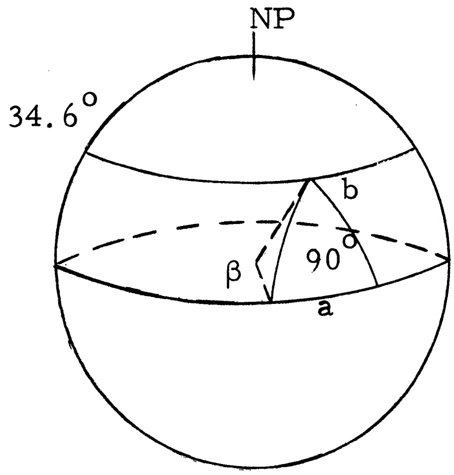
The orbital period is calculated from the equation

$$T = \frac{2\pi}{u_e} r^{3/2}$$

where T is the period

T = 95.3 min.

Applying spherical trigonometry, the subsequent error in longitude at the equator is computed as follows,



$\beta = 82.4^\circ = i$

$b = 34.6^\circ$ = latitude of WTR

$\sin a = \tan b \cot \beta$

$a = 5.3^\circ$

$\cos a \quad \Delta a = -\tan b \csc^2 \beta \Delta \beta$

assuming latitude equals a constant

$$\Delta a = \frac{-\tan b}{\cos a \sin^2 \beta} \Delta\beta$$

for $\Delta\beta = \Delta i = .95$ degrees injection error $\qquad \Delta a = .67^\circ$

FIGURE 36
LAUNCH WINDOW

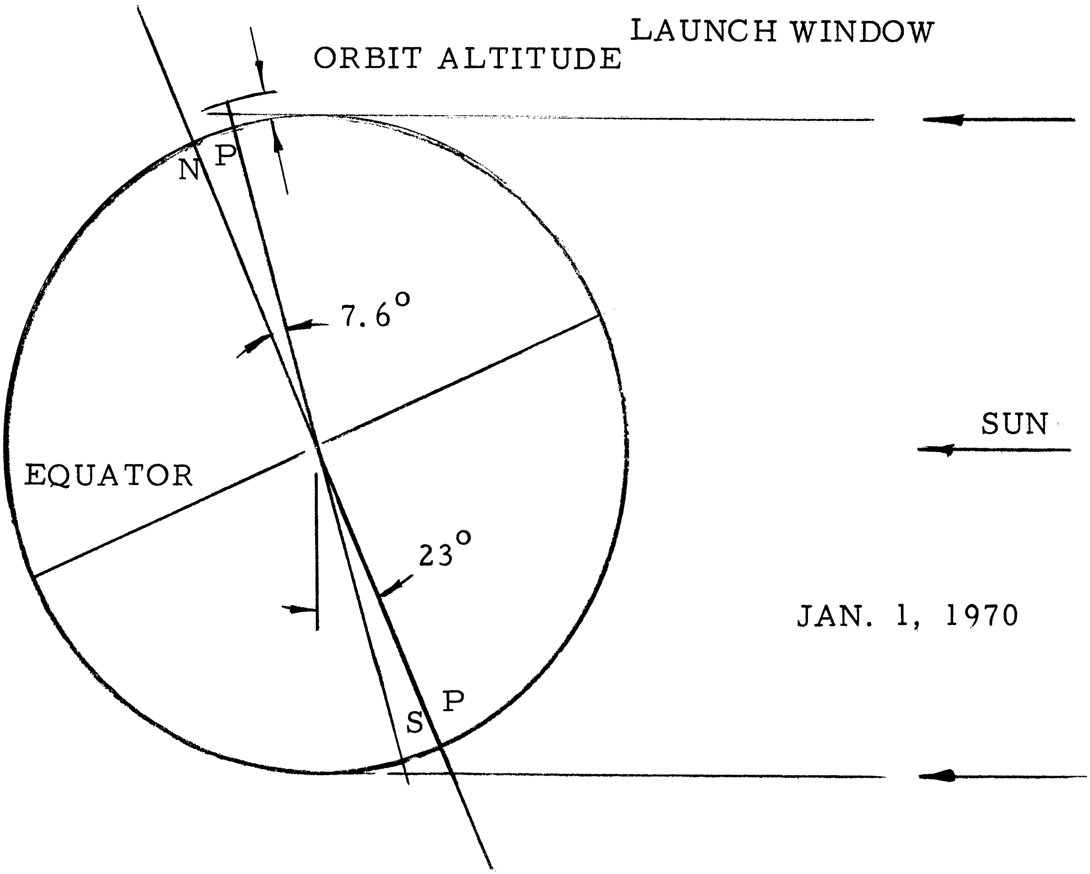
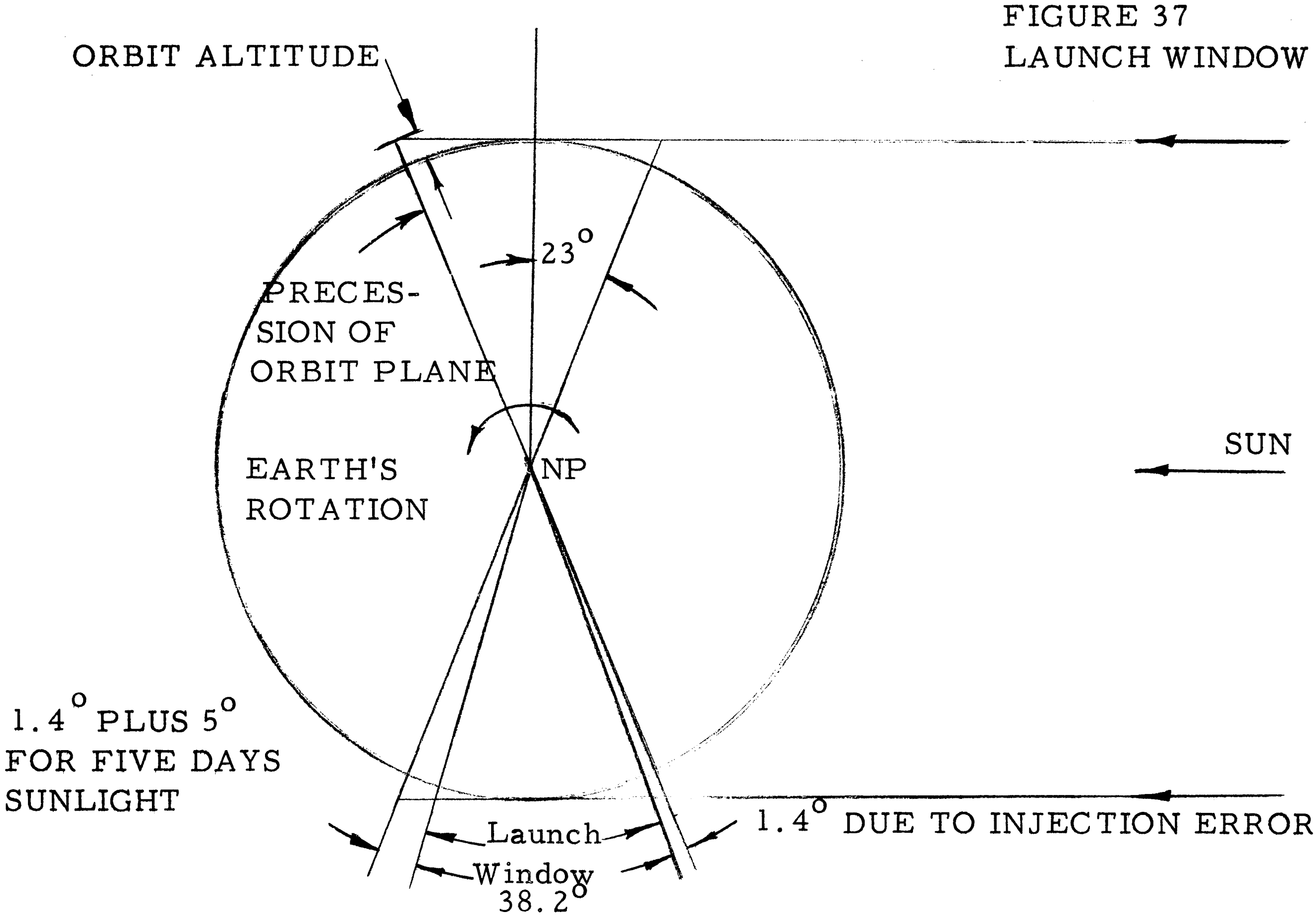


FIGURE 37
LAUNCH WINDOW



APPENDIX D

SAMPLE CALCULATIONS FOR STRUCTURAL ANALYSIS

For a circular tube carrying an axial load, the stress calculations are as follows:

$$\sigma_L = \frac{F}{A} = \frac{ma}{2\pi rt}$$

$$\sigma_A = \frac{\pi^2 E}{12\,(1 - \mu^2)} \left(\frac{t}{L}\right)^2$$

$$\text{safety factor} = \frac{\sigma_A}{\sigma_L}$$

σ_L = limiting stress applied during flight qualifying tests
σ_A = allowable stress which the tube can withstand before yielding
t = tube wall thickness
L = tube length
r = tube radius
m = mass of payload
a = maximum acceleration of payload
E = modulus of elasticity
M = Poisson's ratio

For STRATUM's main support tube, these values become:

t = 0.05 in
L = 21.5 in
r = 4.5 in
m = 250 lbs
a = 25.5 G
E = 1×10^7 PSI
μ = 0.33
σ_L = 457 PSI
σ_A = 970 PSI

safety factor = 2.12

The honeycomb design calculations are based upon a procedure appearing in the March 1960 issue of "Materials in Design Engineering". For a sandwich panel clamped on all edges and subjected to a distributed transverse load, the following equations may be used:

$$M = \beta \quad q\,a^2$$

$$\sigma_L = \frac{M}{t_c\, t_s}$$

$$\text{safety factor} = \frac{\sigma_A}{\sigma_L}$$

σ_L = limiting stress
σ_A = allowable stress
M = maximum experienced moment
q = load in PSI
a = length of panel
b = width of panel
β = factor dependent upon ratio of length to width
σ_L = limiting stress applied
σ_A = allowable stress
t = honeycomb sandwich thickness
t_s = honeycomb skin thickness
t_c = honeycomb core thickness

For STRATUM's equipment platform the following values were used:

q = 7.98 PSI
a = 20 in
b = 20 in
β = 0.125
t = 1.00 in
t_s = 0.02 in
t_c = 0.96 in
M = 395 in - lbs/in
σ_L = 20,700 PSI
σ_A = 40,000 PSI

safety factor = 1.94

To obtain approximate vibrational frequency calculations for honeycomb sandwich, an equivalent plate thickness must be calculated. By equating the formulas for moments of inertia for the two cases, we calculate a theoretical h for the honeycomb which can be used in frequency equations for solid plates.

$$I = \frac{E\, t_s\, d^2}{2} = \frac{E\, h^3}{12}$$

$$h = (6\, t_s\, d^2)^{1/3}$$

For panels which are clamped on all four sides, the frequency equation is:

$$f_n = K \left[\frac{E}{12\,(1 - \mu^2)\,\rho} \left(\frac{h^2}{a^4} \right) \right]^{1/2}$$

f_n = natural frequency of panel
K = fixity coefficient
E = modulus of elasticity
μ = Poisson's ratio
ρ = density
h = equivalent thickness of a plate
a = width of panel
t_s = honeycomb skin thickness
d = distance between centers of honeycomb skins

For the side panels on STRATUM, these values become:

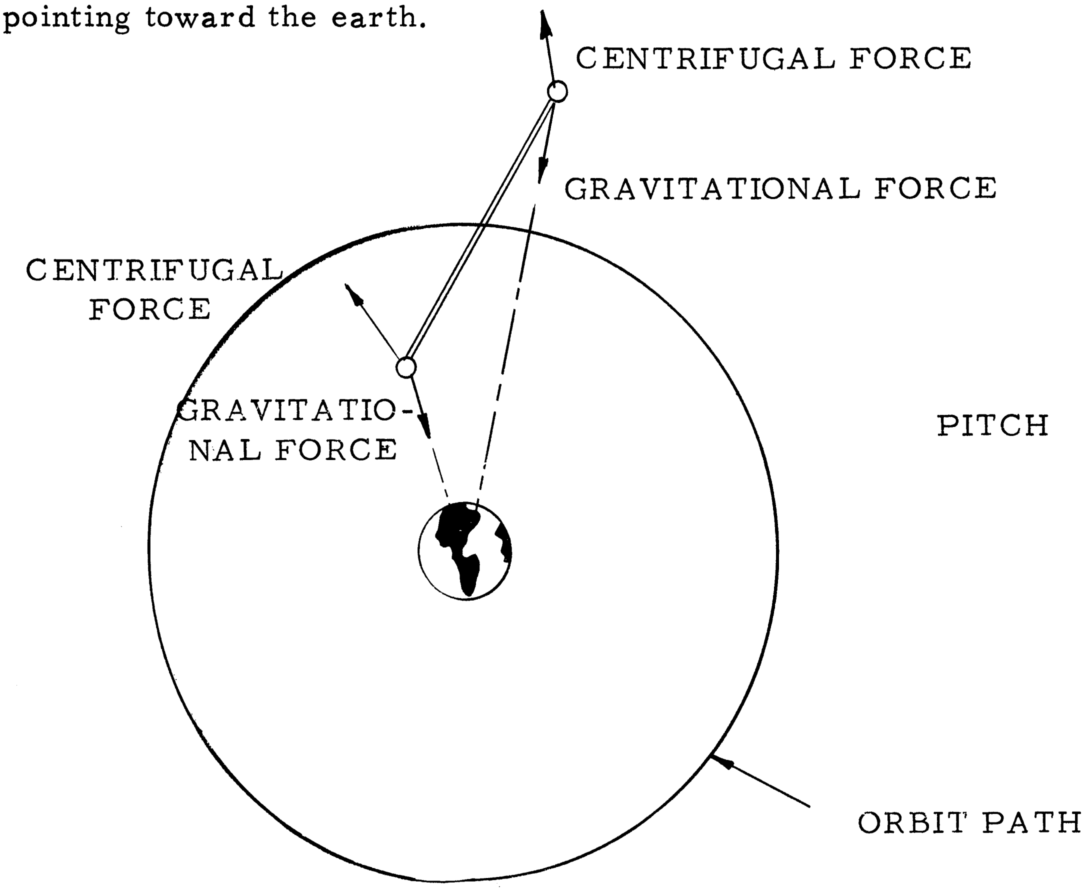
K = 3.69
E = 1×10^7 PSI
μ = 0.33
ρ = 2×10^{-5} PSI
h = 0.174 in
a = 9.12 in
t_s = 0.016 in
d = 0.234 in
f_n = 690 CPS

APPENDIX E

PRINCIPLES OF THE GRAVITY GRADIENT SYSTEM

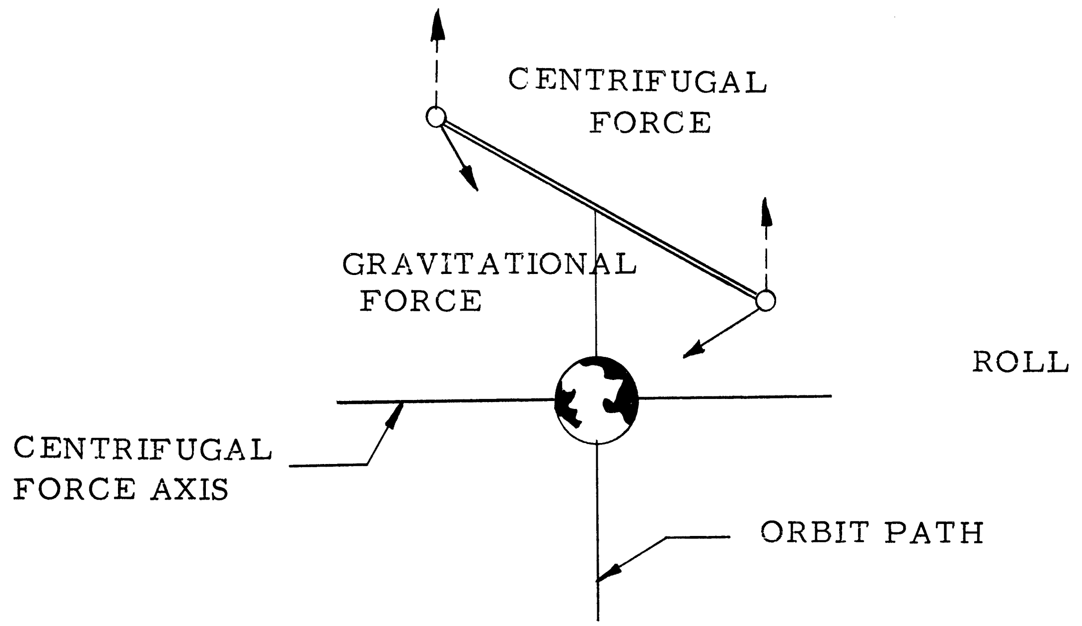
Consider a dumbell shaped figure orbiting around the earth. Both gravitational and centrifugal forces act on the ends of the dumbell. Torque is the product of a force and the length of a lever arm--a lever perpendicular to the force, through the dumbell's center of mass.

The torques generated in pitch are simple and all act in the orbit plane. When the vehicle is disturbed in pitch, the centrifugal force on the end of the dumbell pointing away from the earth is greater than that of the one pointing toward the earth.



The lever arms are unequal in such a proportion as to cancel out the torques. The gravitational force is greater on the end of the dumbell pointing toward the earth because it is nearer the earth. Since these forces always point toward the earth, the lever arms are unequal, the largest being the one associated with the earth-pointing end. Therefore the end with the greatest force also has the greatest lever arm, and the torque is said to restore rather than null the satellites preferred orientation.

The torques generated in roll are also easily seen. The centrifugal forces are unequal, the greatest being associated with the outward end. These are parallel and thus have identical lever arms. The resultant torque therefore



tends to restore the vehicle to local vertical. The gravitational torques act in the same way as in pitch. The yaw torques are generated entirely by centrifugal forces. Figure 38 shows that the components of centrifugal force in the yaw plane produce a couple that swings the dumbell parallel to the orbit path.

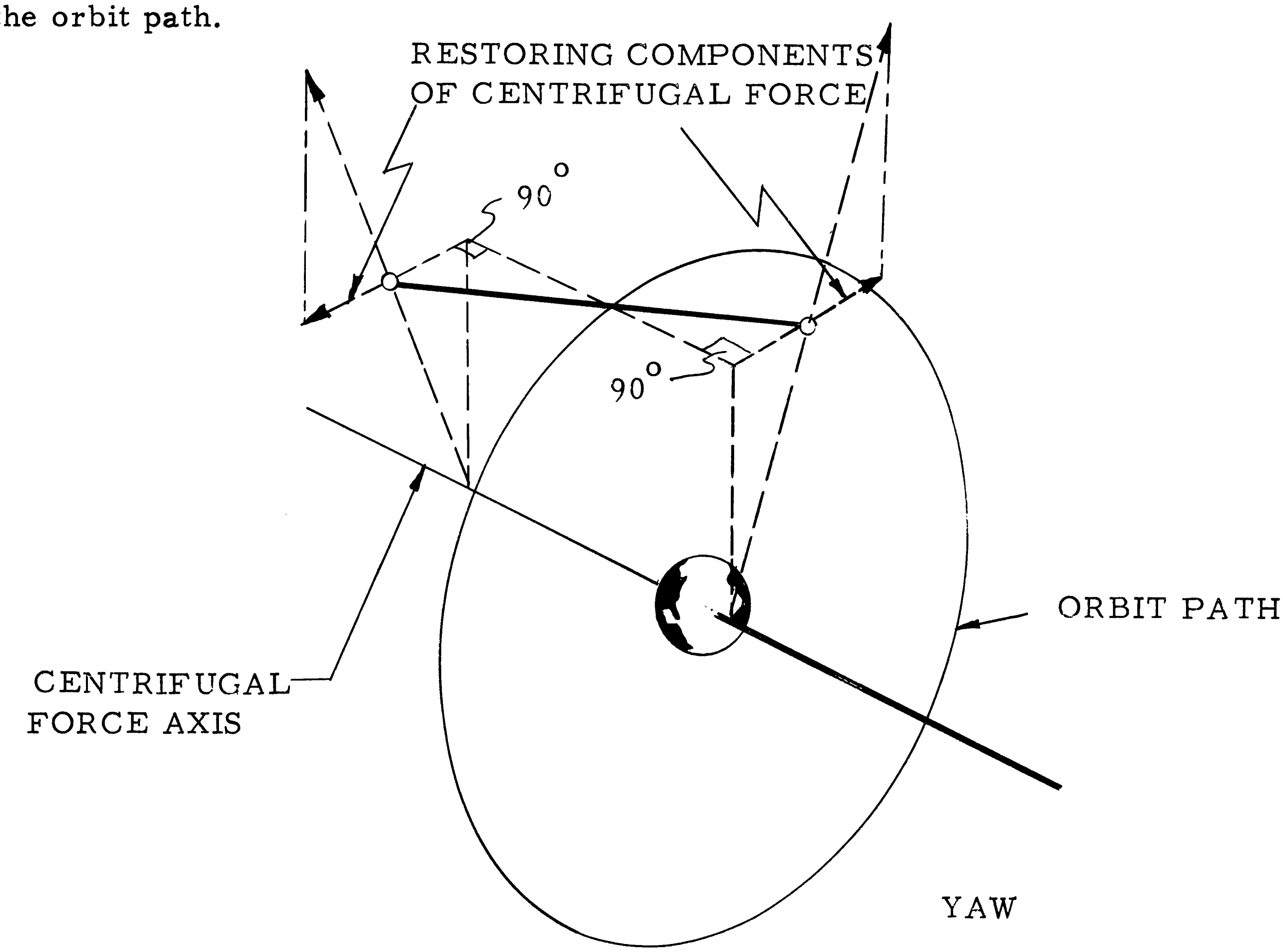


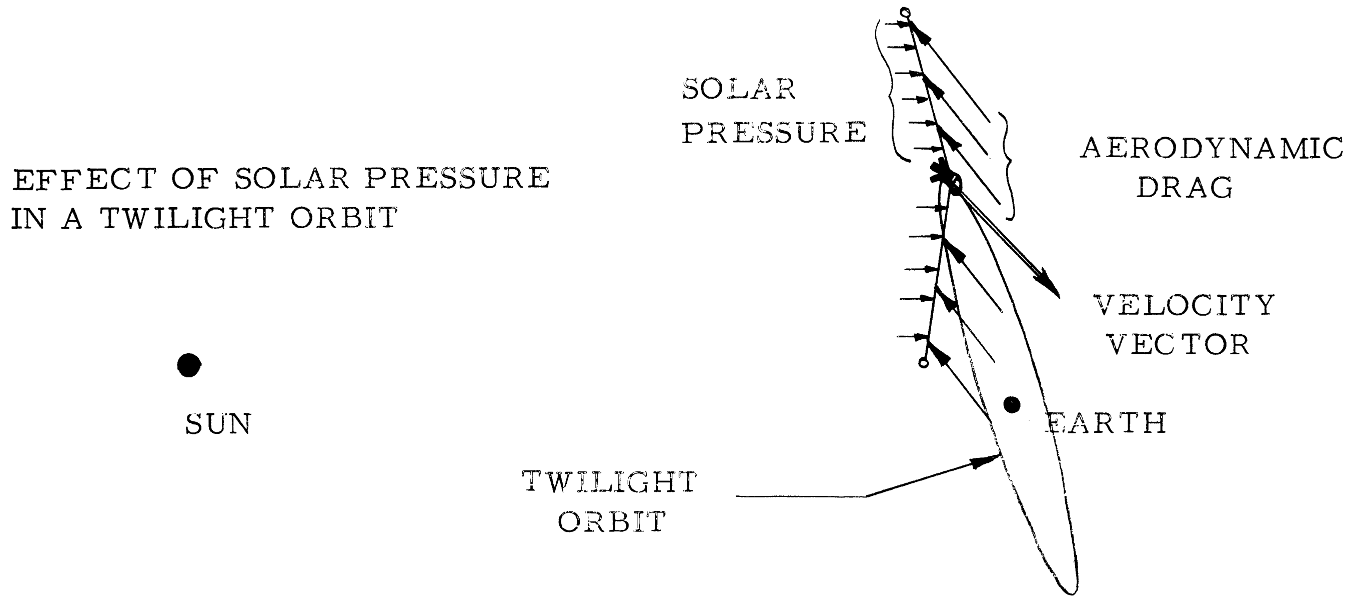
FIGURE 38

APPENDIX F

ANALYSIS OF DISTURBING TORQUES

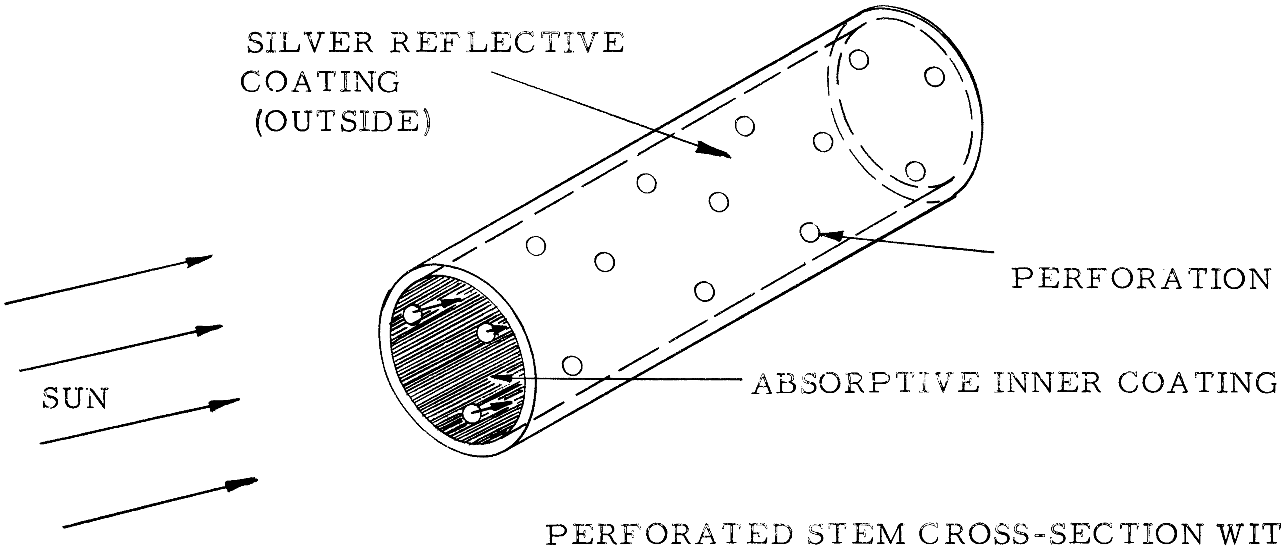
Since little information about micrometeoroids is known at the present time, let it suffice to say that STRATUM expects no exceptional micrometeoroid activity.

Solar pressure is a very real problem; however, especially in a twilight orbit. Solar pressure of 1×10^{-7} lbf/ft^2 is standard for usage in near-earth orbits. A solar pressure yaw torque of $.22 \times 10^{-5}$ lbf/ft, calculated for a twilight orbit, is small as compared to the value of



aerodyanmic torque. The aerodynamic torque is calculated at 26×10^{-5} lbf/ft for a yaw angle of 20^{o}. This implies that the effect of solar radiation is not negligible but is certainly dominated by the restoring torques.

Thermal bending of the gravity gradient rods can be lessened if the rods are coated with a highly reflective material such as silver. Also, if the rods are perforated and the inner surface is coated with a highly absorptive material, there is less chance of a temperature gradient across the rod.



PERFORATED STEM CROSS-SECTION WITH SILVER OUTER COATING AND ABSORPTIVE INNER COATING

Since the only available power on board is electrical, a residual magnetic dipole moment results from the interaction of the earth's magnetic field with any permanent or induced magnetic dipole from permeable material in the satellite. For analytical computations a dipole of 1 pole-cm per pound of spacecraft is usually the rule of thumb. Therefore a dipole of 300 pole-cm is used. After all instruments are on board, small magnets are placed inside the spacecraft to orient the magnetic dipole in a particular direction. Preferably this orientation will coincide with the yaw axis. Since the hardest axis to stabilize is the yaw axis, the dipole moment will be located so as not to disturb the spacecraft in the yaw plane. The residual dipole moment varies according to the relation:

$$T_M = M H \sin \phi$$

where M = satellites magnetic dipole (chosen as 300 pole-cm for analytical reasons)

H = earth's magnetic intensity at the satellite (approximately 1/2 oersted for 300 nautical miles)

ϕ = angle between the earth's magnetic field and the magnetic dipole of the satellite.

No torques in excess of 1.8×10^{-7} lbf/ft are expected for a disturbance of less than 20 degrees. The aerodynamic torques are enough to restore the vehicle to its preferred orientation.

The torques generated from highly eccentric orbits tend to divert the spacecraft from its desired orientation. As the vehicle decreases its altitude in flight, two things happen: 1) the gravity gradient torques increase because of decreasing altitude, and 2) centrifugal torques increase due to larger orbital rates. Since STRATUM is in a circular orbit, it has been estimated that if the insertion errors are within that stated for Scout, the perturbing torques due to eccentricity can be neglected. The relative motion between the poles of the magnets in the dampers and the poles of the earth require the magnets to "flip" over twice every orbit. Therefore, the rate of the dampers motion is twice orbital rate, causing a disturbing torque in pitch. The errors induced by this disturbance are not negligible but are self-damping.

APPENDIX G

YO-YO DESPIN RELEASE MECHANISM

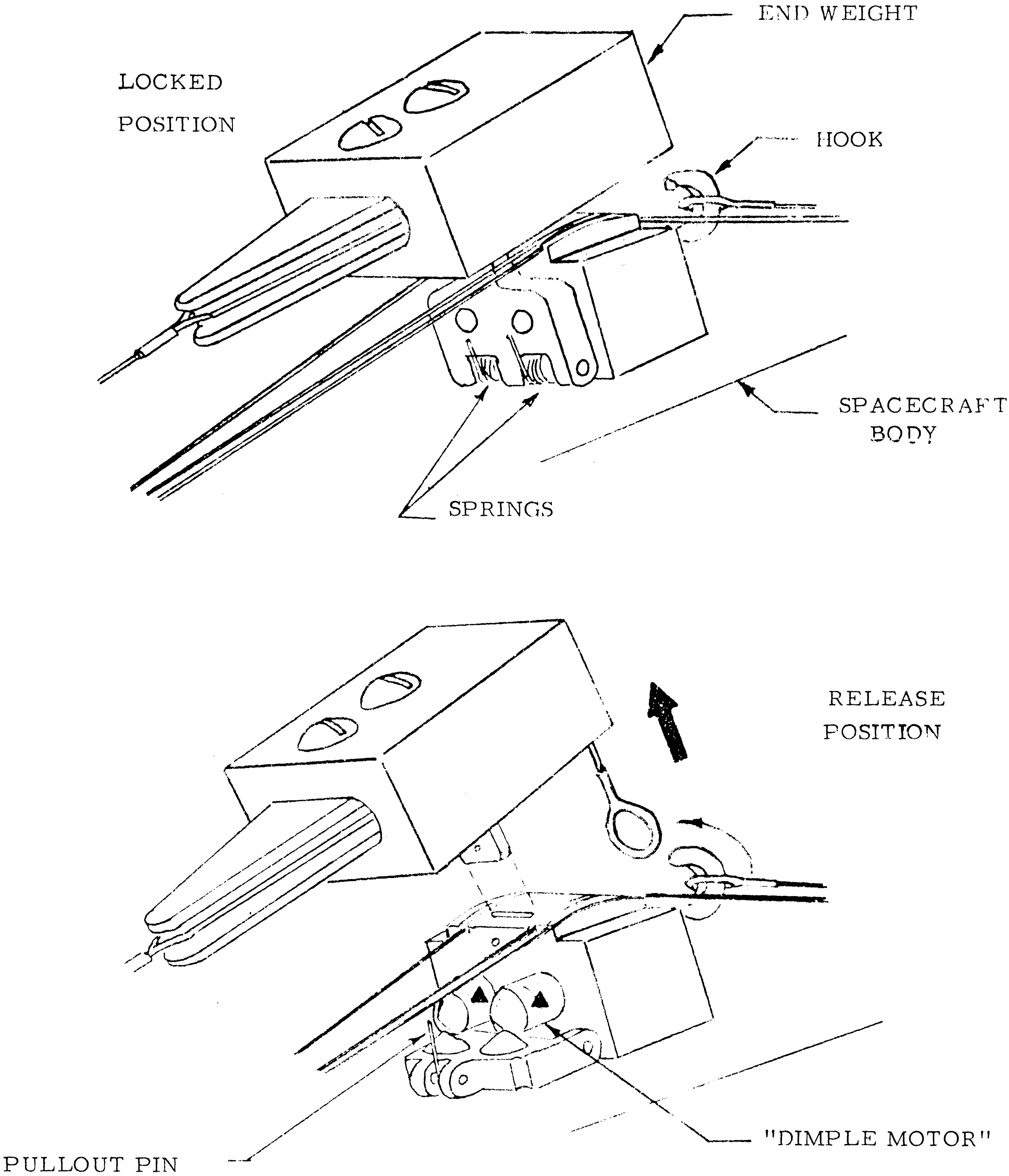


FIGURE 39. DESPIN WEIGHT RELEASE MECHANISM

APPENDIX H

ATTITUDE SENSING DATA

MAGNETOMETERS:

Schonstedt RAM-5C Heliflux Magnetic Aspect Sensor

Input voltage: 24 to 32 v dc
Input current: 11 ma typical
Range of field: 600 millioersted plus or minus
Stability: 3%
Output impedance: 20K
Output load: 100K
Weight: 5 ounces
Power: .31 watts required per sensor
Size: 3 x .625 diameter

INFRARED SENSORS:

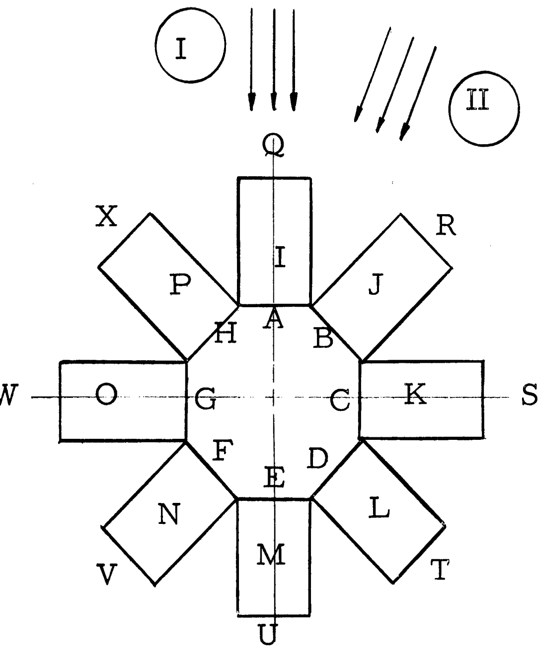
Barnes Engineering Model 13-181-X Radiation Balance Earth Sensor

Optical band: 14 to 16 micron
Number of packages: 2
System weight: 3 lbs
Power required: 1/2 watt
Unit can be modified optically to fit any particular orbit

SOLAR ASPECT SENSORS:

Original design using vapor-deposited discrete photovoltaic surfaces

4 sensors per system
Weight: 2.5 lbs total
Power: 1 watt for system
Sensor size: 1/2 x 1 x 2 1/2



APPENDIX I
POWER SYSTEM

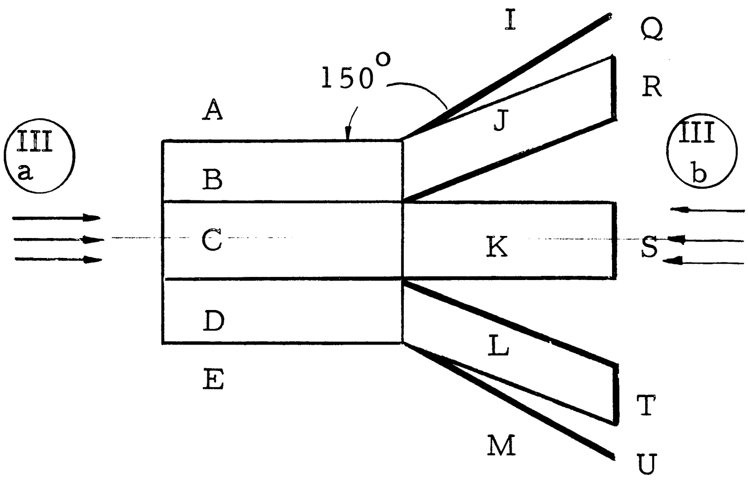


FIGURE 40 . EQUIVALENT FACE CALCULATIONS

SUN DIRECTION	FACES REFERRED TO	F CORRECTION FACTOR	θ ANGLE OF INCIDENCE	COS θ	F COS θ	EF
I	A	1.0	0°	1.0	1.0	1.00
	H, B	1.0	45	.707	.707	1.414
	I	1.0	30	.865	.865	.865
	P, J	.95	63.5	.446	.424	.848
		TOTAL EF FOR I				4.127
II	A, B	1.0	22.5°	.924	.924	1.848
	C, H	.92	67.5°	.383	.352	.704
	I, J	.94	63	.414	.390	.780
	P, K	.87	70.5	.334	.290	.580
		TOTAL EF FOR II				4.912
III						
a	I J K L M N O P					
		.96	60°	.576	.555	
b	Q R S T U V W X					
		TOTAL EF FOR EITHER IIIa or III b				4.44

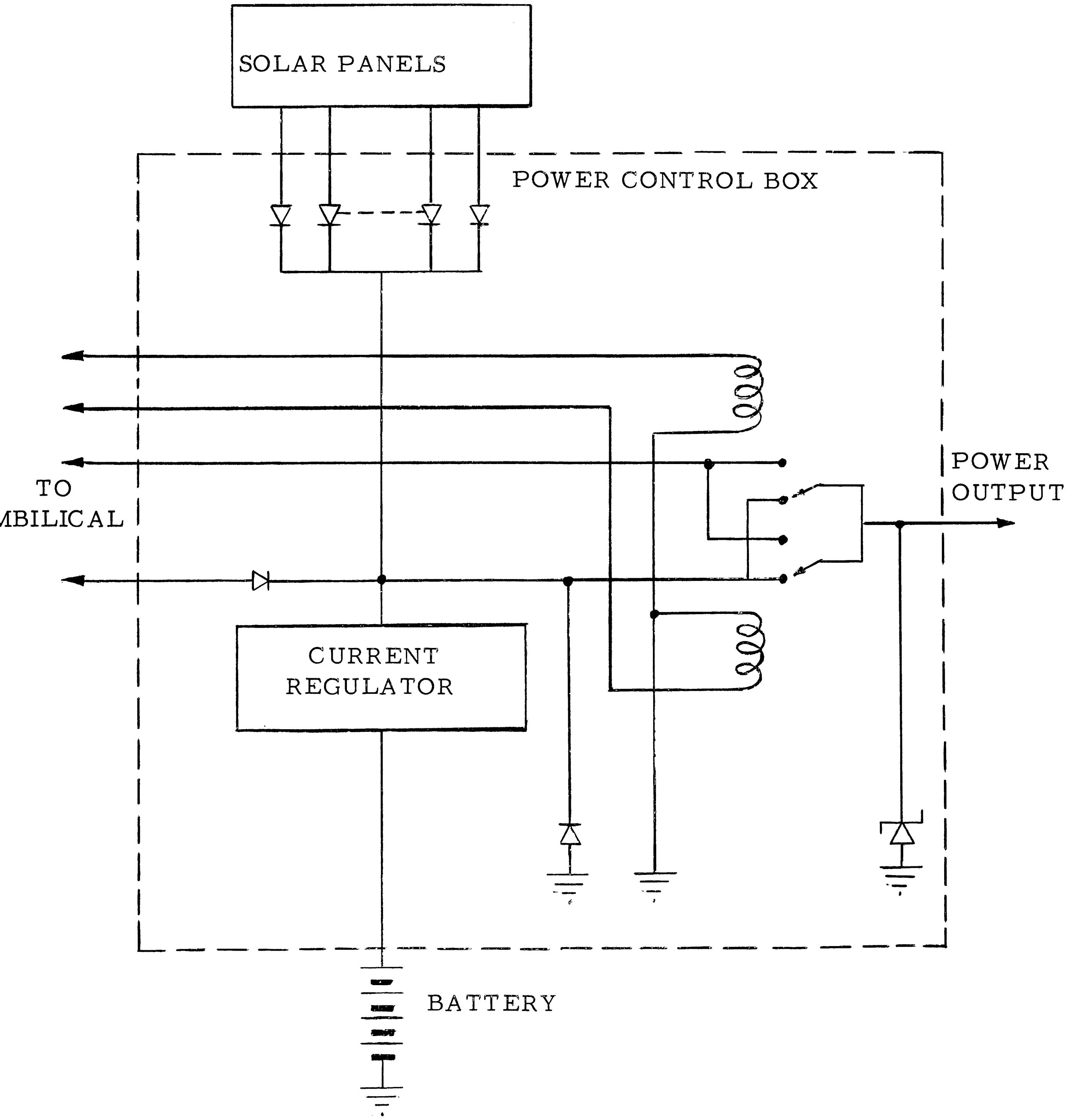


FIGURE 41 . POWER SYSTEM SCHEMATIC DIAGRAM

APPENDIX J

THERMAL CONTROL

A brief outline of the way we solved our thermal control problem is as follows: (1) find temperatures of each side assuming no internal conduction or radiation; (2) find heat absorbed by each side; (3) determine how lateral conduction affects temperature distribution; (4) determine how use of heat pipes affects temperature distribution; (5) find average internal temperature; (6) find thermal time constant.

Temperatures of sides: (see Figure 42 , page 104)

General heat balance equation is

$$q_E = q_S + q_e + q_r + q_i$$

where q_E = heat emitted = $A \sigma e T^4$
q_S = absorbed solar emission $A_s\ a\ G_s$
q_e = absorbed earth emission = $A_e\ a\ E$
q_r = absorbed earth reflection = $A_e\ a\ R$
q_i = internal power

Solve for T to get:

$$T = \left[\frac{a}{e\sigma}\left(\frac{A_s}{A} G_s + \frac{A_e}{A} E + \frac{A_e}{A} R\right) + \frac{q_i}{e\,\sigma}\right]^{1/4}$$

a = average absorptivity = .67, e = average emissivity = .77
σ = Stefan-Boltzman constant = $.1718 \times 10^{-8}$

$$\frac{BTU}{hr - ft^2 - {}^oR^4}$$

$\frac{A_s}{A}$, $\frac{A_e}{A}$ = view factors, projected area divided by total area

G_s = solar constant = $442 \frac{BTU}{hr - ft^2}$

E = earth emission = $75 \frac{BTU}{hr - ft^2}$

R = earth reflection = 0 for twilight orbit

We will neglect q_i since it is small.

Side 3: $\frac{A_s}{A} = .925, \quad \frac{A_e}{A} = .450$

$$T = \left(\frac{.67}{.77}\right)^{1/4} \left[\frac{.925 \times 442}{.1718} + \frac{.450 \times 75}{.1718}\right]^{1/4} 10^2$$

$$T = 690^\circ R$$

Side 4: $\frac{A_s}{A} = .382, \quad \frac{A_e}{A} = .794$

$$T = \left(\frac{.67}{.77}\right)^{1/4} \left[\frac{.382 \times 442}{.1718} + \frac{.794 \times 75}{.1718}\right]^{1/2} 10^2$$

$$T = 584^\circ R$$

Heat absorbed by each side:

$$q'_{tot} = A_s\, a\, G_s + A_e\, a\, E + \cancelto{0}{A_e\, a\, R} + \cancelto{0}{q_i}$$

$$q_{tot} = a\, A\left(\frac{A_s}{A}\, G_s + \frac{A_e}{A}\, E\right)$$

Side 3:

$$q'_{tot} = .67 \left(\frac{9.12}{12}\right)\left(\frac{27}{12}\right)\left[.925 \times 442 + .450 \times 75\right]$$

$$q_{tot} = 505 \frac{BTU}{hr}$$

Side 4:

$$q'_{tot} = .67 \left(\frac{9.12}{12}\right)\left(\frac{27}{12}\right)\left[.382 \times 442 + .794 \times 75\right]$$

$$q'_{tot} = 260 \frac{BTU}{hr}$$

Lateral conduction between sides 3 and 4.

We used an iterative process to determine final temperature. One iteration follows:

$$q_{3\to4} = K\left(\frac{A}{L}\right) \quad (T_3 - T_4)$$

$$K = 120 \ \frac{BTU}{hr - ft - {}^{o}R}$$, conductivity of aluminum

$\frac{A}{L} = .0048$ a sum of the value for the side panels and the honeycomb itself

T_3 and T_4 are known from above.

$$q_{3\to4} = 120 \quad (.0048) \quad (690 - 584)$$

$$q_{3\to4} = 61 \ \frac{BTU}{hr}$$

$$q_3' = q_3 - q_{3\to4} = A\, e\, \sigma\, {T_3'}^{4}$$

so

$$T_3' = \left[\frac{(q_3 - q_{3\to4})}{A\, e\, \sigma}\right]^{1/4} = \left[\frac{444}{\left(\frac{9.12}{12}\right)\left(\frac{27}{12}\right)\ (.77)\ (.1718)}\right]^{1/4} 10^2$$

$$T_3' = 670^{o}\,R$$

$$q_4' = q_4 + q_{3\to4} = A\, e\, \sigma\, {T_4'}^{4}$$

so

$$T_4' = \left[\frac{(q_4 + q_{3\to4})}{A\, e\, \sigma}\right]^{1/4} = \left[\frac{260 + 61}{\left(\frac{9.12}{12}\right)\left(\frac{27}{12}\right)\ (.77)\ (.1718)}\right] 10^2$$

$$T_4' = 615^{o}\,R$$

With these new temperatures, we can get a new $q_{3\to4}$ and continue iterating. This process is easily carried out on a computer.

Use of heat pipes.

Since the heat pipe is such an efficient device, we assume that it can transfer all the heat transferred to it at a given temperature. Hence the limiting factor is the axial conduction to the heat pipe. To find the operating temperature of the heat pipe, we find the satellite temperature assuming infinite conduction

$$q_{in,tot} = q_{out,tot}$$

$$\begin{aligned} q_{in_{tot}} &= a\, A_{P_s}\, G_s + a\, A_{P_e}\, E + q_i \\ &= .67\left(\frac{27 \times 23.8}{144}\right)\left[442 + 75\right] + 153\ \frac{BTU}{hr} \\ &= 1663\ \frac{BTU}{hr} \end{aligned}$$

$$q_{out,tot} = A_{tot}\, e\, \sigma\, T^4$$

$$T = \left(\frac{1663}{A_{tot}\, e\, \sigma}\right)^{1/4} = \left[\frac{1663}{\frac{8 \times 9.12 \times 27}{144}\,(.77)\,(.1718)}\right]^{1/4} 10^2$$

$$T = 560^\circ R$$

Axial conduction is determined in the same way as transverse conduction. From the spacing of the heat pipes, we see the hottest point will be at 5 in above the upper or 5 in below the lower heat pipe. This determines our value of L. Considering side 3, we get

$$q = K\left(\frac{A}{L}\right)\Delta T$$

The value of $\frac{A}{L}$ for axial conduction is .0041 ft.

$$q_{a\to b} = 120\ (.0041)\ (670 - 560)$$

$$q_{a\to b} = 54\ \frac{BTU}{hr}$$

$$q_{ab} - q_{a\to b} = \sigma\ A_{ab}\ \epsilon\ T^4 - q_{a\to b}$$

$$q_{ab} - q_{a\to b} = (1.718 \times 10^{-9}) \left[\frac{9.12 \times 5}{144}\right] (.77)\ (670)^4 - 54$$

$$q_{ab} - q_{a\to b} = 170 - 54 = 116\ \frac{BTU}{hr}$$

$$T_a = \left[\frac{q_{ab} - q_{a\to b}}{A_{ab}\ e\ \sigma}\right]^{1/4} = \left[\frac{116}{\left(\frac{5 \times 9.12}{144}\right)\ (.77)\ (.1718)}\right]^{1/4} 10^2$$

$$T_a = 610^\circ R$$

Internal temperature.

To find the temperature of the equipment platform, we assume it is subjected to a uniform load of 45 watts, has an $\frac{a}{e} = 1$ and an $\frac{A_P}{A} = 1$, where A_P is projected area.

$$T_i = \left(\frac{a}{e}\right)^{1/4} \left(\frac{A_P}{A}\right)^{1/4} \left(\frac{P_i}{\sigma}\right)^{1/4}$$

$$T_i = \left(\frac{153}{.1718}\right)^{1/4} 10^2 = 547^\circ R$$

Since this is less than the 560° temperature of the heat pipe, we see that the equipment platform will never exceed $560^\circ R$.

Thermal time constant.

In order to determine how fast the satellite will cool off when it goes into the earth's shadow, we must determine its time constant.

$$q_{in} = C_p m \frac{dT}{dt}$$

$$q_{in} = .67 \left(\frac{23.8 \times 27}{144}\right) \qquad 442 = 1320 \frac{BTU}{hr}$$

$$C_p = \text{specific heat} = .23 \frac{\text{watt hr}}{\text{lbm - } ^oR} = .78 \frac{BTU}{\text{lbm } ^oR}$$

$$m = \text{mass of satellite} = 250 \text{ lbm}$$

$$\frac{dT}{dt} = \frac{q_{in}}{C_p m} = \frac{1320}{.78\ (250)} = 7 \frac{^oR}{hr}$$

Since the satellite is in the shadow for a maximum of 36 minutes, it retains enough of its heat to maintain a sufficient internal temperature.

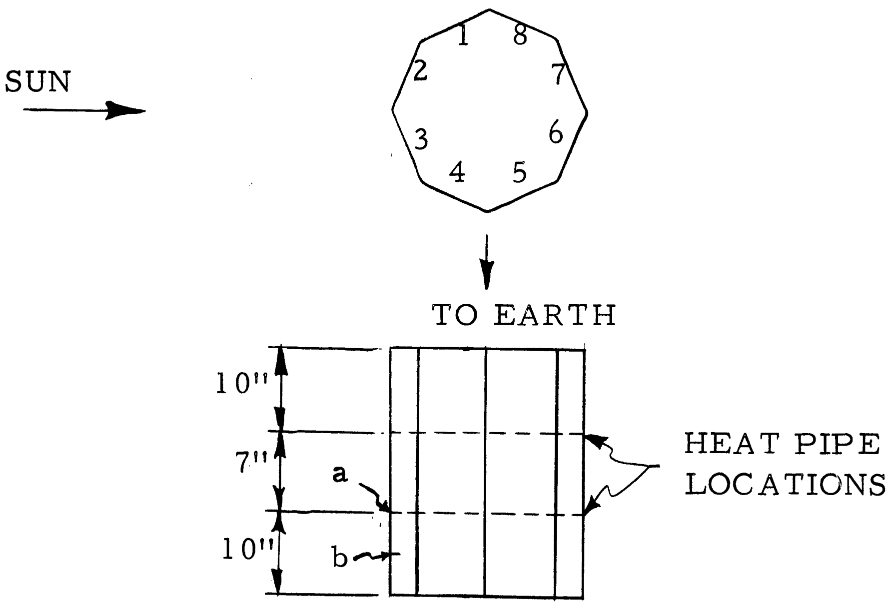


FIGURE 42 . SATELLITE MAIN BODY CONFIGURATION AND NUMBERING SYSTEM USED IN THERMAL CONTROL CALCULATIONS.

APPENDIX K

SPECIFICATIONS ON INSTRUMENTS IN COMMAND AND TELEMETRY SUBSYSTEMS

Receivers

Frequency	136-138 (single frequency, crystal controlled)
Frequency change within band	change crystal and retune
Antenna impedance	50 ohms nominal
Tuning accuracy	$\pm$ 0.005%
Noise figure	6 db maximum
RF bandwidth	-3 db = 200 KC, - 60 db = 600 KC maximum
Maximum RF input	2 V RMS
Threshold sensitivity	1 microvolt nominal for correct relay closure, 2 microvolts maximum over temperature range
Signal level indication	0 to 4 volts DC into 10 K ohm load

Decoders

Number of channels	Up to 7 (optional) and IRIG channel 1 through 20
Simultaneous usable tones	3 (up to 7, by special request)
RF deviation required for proper relay operation (each channel)	30 KC peak
Channel bandwidth (-2 db)	$\pm$ 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 of relay contacts	42 (for 7 channels)
Total number of possible command functions	128 for 7 channels

Environmental Characteristics of Receivers & Decoders in a Single Capsule

Temperature	-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
Altitude	unlimited

Clock Oscillator

Event outputs	six independents outputs per module with outputs programmable for any time between 2 and 16378 seconds
Voltage	positive supply voltage, less an internal drop not exceeding four volts
Thermistor output	0 to 5 volts for range of -30 to 165 F
Time reference accuracy	0.005% over temp range 0 to 50 C 0.02% over temp range from -55 to 65 C
Size	7" x 6" x 4"

Diplexer

Size	3" x 1" x 2"
Weight	2.5 lbs

S-band Transmitter

Center frequency stability	0.01% under environmental operating conditions
Carrier deviation	+ 500 KHZ
Modulation - sensitivity	up to 250 KHZ p-p with nominal input impedance of 100 K ohms resistance shunted by 30 pf capacitance
Modulation - response	± 1.5 db from 5 HZ to 500 KHZ
Construction	modular construction. Boxes and covers are cast with integral shielding provided by the casting
Acceleration	100 G's in each direction of any three mutual perpendicular axes
Shock	100 G's 11 milliseconds duration (half-sine pulses) in each direction of any three mutually perpendicular axes

Subcarrier Oscillator

IRIG channels	1-21
Input	0 to 5 V ± 2.5 V
Input impedance	one megohm, minimum
Linearity	± 0.25% of bandwidth deviation from best straight line
Drift	less than ± 0.5% of bandwidth in 8 hour period after 5 seconds warmup at 25C

Output	1.5 V peak-to-peak into an 8 K ohm load
Output distortions	less than 1% at any frequency in the band
Shock	$\pm$ 1.0% of bandwidth for 100 G half sine shock pulse of 11 milliseconds duration in each major axis
Acceleration	linear accelerations up to 100 G's produce less than $\pm$ 1.0% of bandwidth frequency change
Vibration	$\pm$ 1.0% of bandwidth variation for 25 G's from 10 cps to 2000 cps in each major axis
Actual size	1.6" high by 0.81" thick by 1.35" wide

Mixer Amplifier

Voltage gain	2 to 20, continuously variable
Distortion	less than 0.5%
Noise	less than 10 mv PP with input shorted
Temperature stability	-20 to 85 C causes less than $\pm$ 1.0% gain variation based on best reference
Shock	less than $\pm$ 1.0% variations of specified performance for 200 G for 10 ms in each major axis
Acceleration	less than $\pm$ 1.0% variation of specified performance for linear performance up to 200 G
Vibration	less than $\pm$ 1.0% variation of specified performance for 30 G from 10 HZ to 2000 HZ

APPENDIX L

TAPE RECORDER SPECIFICATIONS

Weight of transport	6.75 lbs
Weight of logic	1.0
Transport size	9.3" x 7.7" x 3.5"
Logic size	6.5" x 4.5" x 1.3"
Required voltage	28 vdc
Transport power	2 w readin, 14w readout
Logic power	1w
Readout bandwidth	200 KHZ (projected to 1970 launch date)
Axial acceleration limit	18 G
Manufacturer	Kinelogic Corp.

APPENDIX M

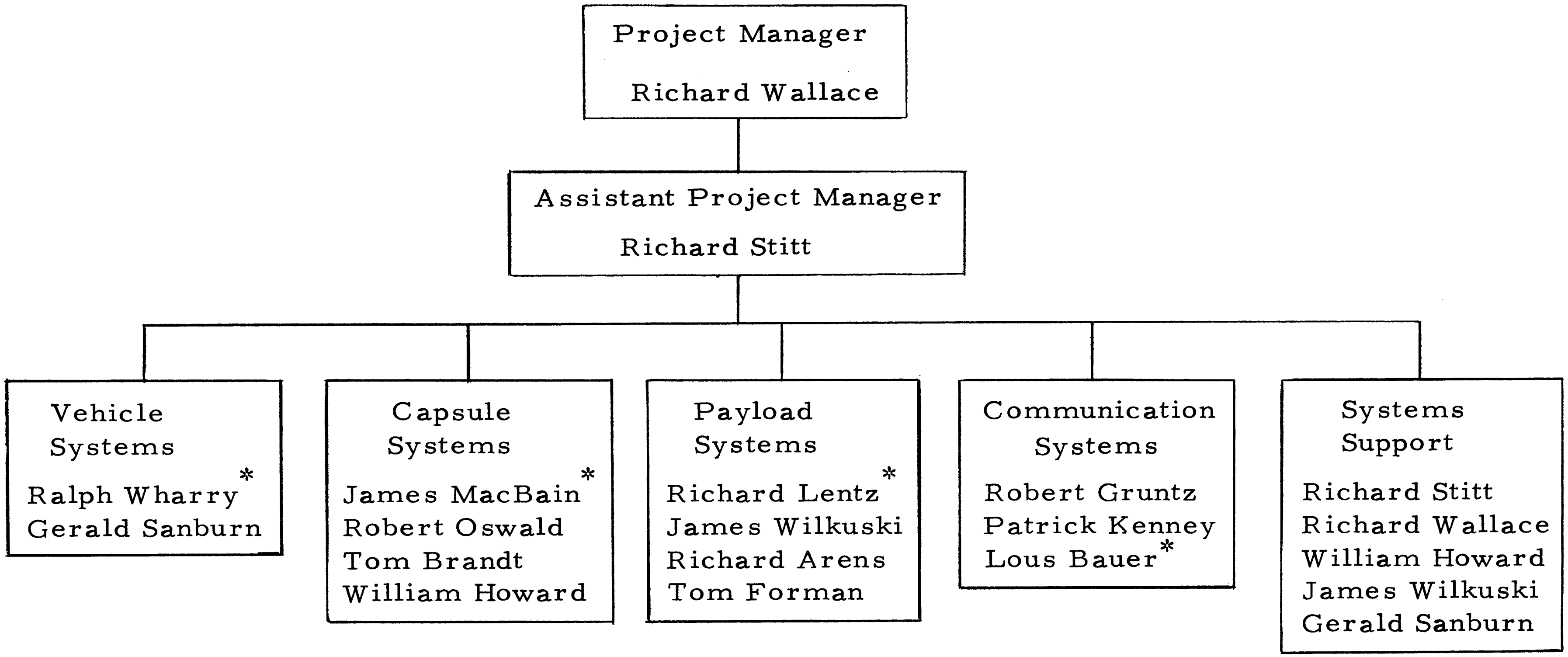
STRATUM WEIGHT BUDGET

Item	Weight	Cable	Fastener
E-Section of Scout	17.2	(b)	(b)
Scientific Instruments	21.5	1.8	2.0
Onboard Communications Equip.	35.8	4.0	3.0
Data Storage	25.3	.5	.5
Stabilization	16.8	(b)	1.5
Attitude Sensing	7.0	.8	.8
Body Mounted Cells	5.0	.4	(b)
Paddles and Mounted Cells	14.0	1.0	1.0
Paddle Deployment	1.6	.5	.5
Storage Batteries	15.0	.5	.5
Power Conditioning Equip.	1.5	.5	.5
Structure	35.4	(a)	1.0
Thermal Control	6.0	(a)	(b)
Yo-Yo Despin	2.0	(b)	(b)
Ballast	2.0	(a)	(a)
Total Weights	206.1	10.0	11.3
Total Spacecraft Weight	227.4		
Growth allowance for STRATUM	22.6		
Maximum allowable weight	250.0		

(a) Not applicable
(b) Included in Weight

APPENDIX N

STRATUM ORGANIZATION CHART



*Indicates group leader