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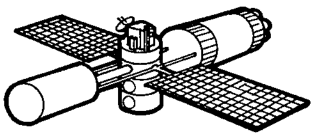
PRELIMINARY TECHNICAL DATA FOR EARTH ORBITING SPACE STATION



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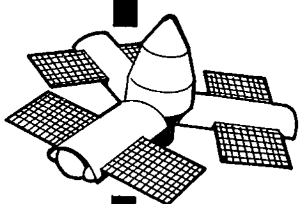

VOLUME IV

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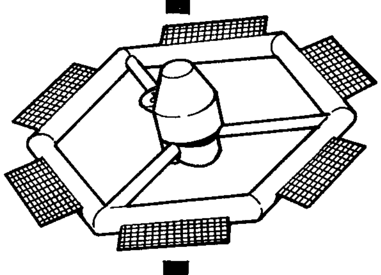

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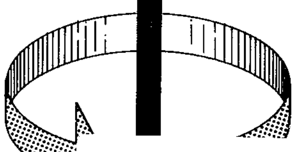

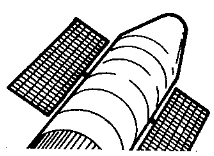

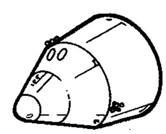

PREFACE

This document, Volume IV of IV, contains the Manned Spacecraft Center's technical data on configurations, integration, and weights for the Earth Orbital Manned Space Station Study. The data is concerned with orientation, stability, design integration, spacecraft concepts, and the associated weights. A section which compares the Space Station and Mars Missions is also included. This data is submitted in response to a NASA Headquarters' initiated study which includes requirements data from Langley Research Center, and experiment integration data from Marshall Space Flight Center. The complete integrated study will include the data from all three Centers.

The contributions of the various organizations within the Manned Spacecraft Center are acknowledged at the beginning of each section. Some of the data within these sections may differ slightly from the summary document since the summary presents the technical data in an integrated form.

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EARTH ORBITING SPACE STATION

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FOR EARTH ORBITING SPACE STATION

VOLUME IV

CONFIGURATIONS, INTEGRATION, AND WEIGHTS

SECTION 1.0

ORIENTATION

ADVANCED SPACECRAFT TECHNOLOGY DIVISION

ENGINEERING AND DEVELOPMENT DIRECTORATE

MANNED SPACECRAFT CENTER

1.0 ORIENTATION

1.1 DISTURBANCES

1.1.1 ORBIT PRECESSION

The oblateness of the earth causes the orbit plane of a satellite to precess relative to inertial space about the earth's polar axis. The rate of precession is determined by the altitude, eccentricity, and inclination of the orbit, and is in the direction opposite to the motion of the satellite. This rate, expressed as the motion of the ascending node of the orbit, is shown in Figure 1.1 for a 260 n.m. circular orbit as a function of inclination. For a 60° inclination, the node travels westward approximately 3.9° per day. Thus, the orbit plane makes about four complete revolutions per year around the earth's axis.

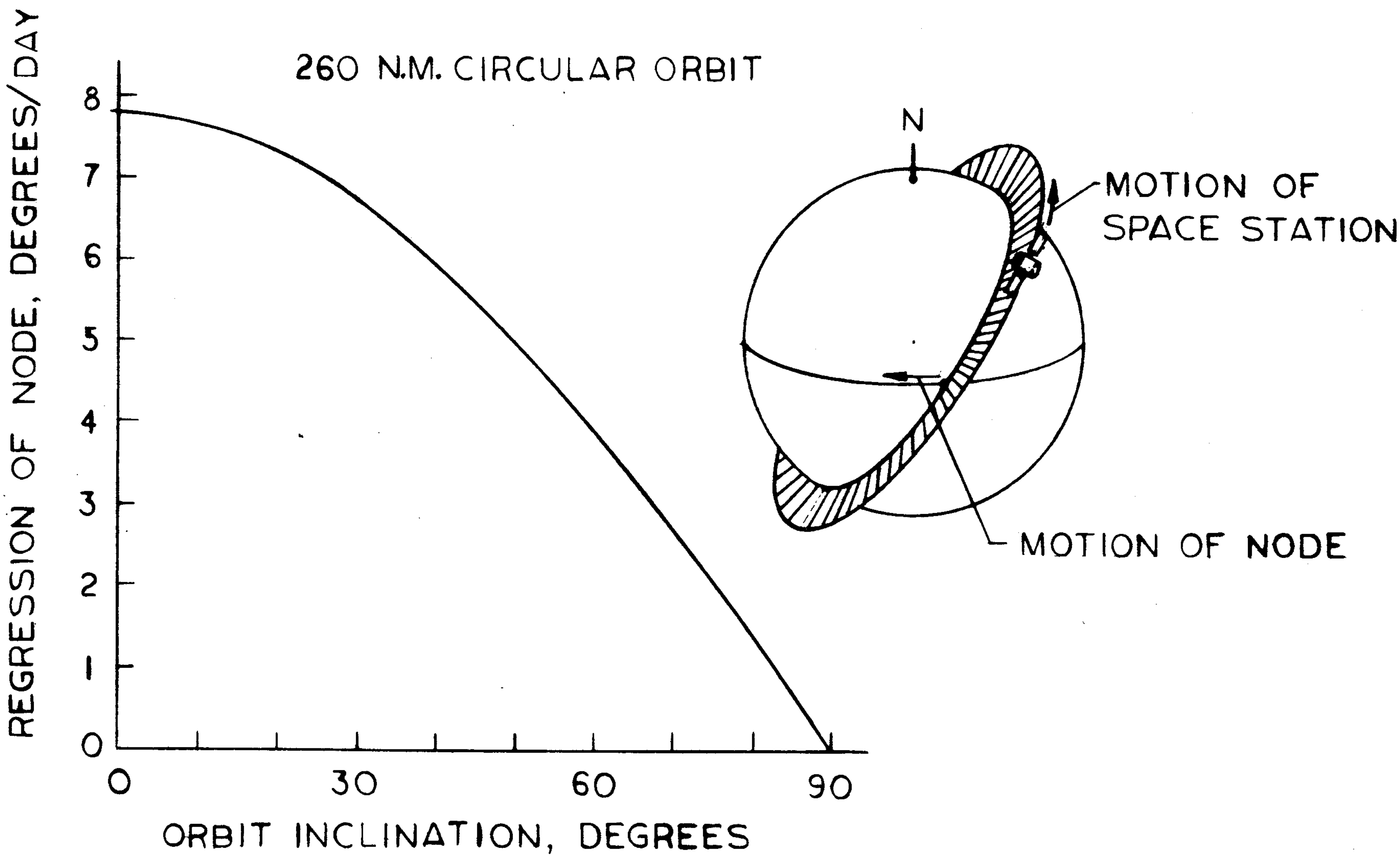
This precession, together with the inclination of the equator to the ecliptic, causes a large variation in the angle between the spacecraft-sun line and the orbit plane as the earth moves around the sun, as shown in Figure 1.2. Figure 1.3 illustrates a typical variation of this angle over a period of a year for the orbit specified in the preceding paragraph. The initial position of the ascending node was selected arbitrarily for this example. Other initial positions will shift the curve to the left or right within the indicated envelope.

1.1.2 GRAVITY GRADIENT TORQUE

A torque is applied to a spacecraft in orbit whenever the principal axes of inertia do not coincide with the local vertical and the orbit plane. This torque results from the combined effects of the variation of gravitational acceleration with distance from the center of the earth and the centrifugal acceleration of the spacecraft in its orbit. Although the torque is zero for any orthogonal orientation of the principal axes with the orbit plane and local vertical, the only stable orientation is that which has the axis of minimum inertia parallel to the local vertical and the axis of maximum inertia normal to the orbit plane.

1.1.3 OTHER DISTURBANCES

Torques are also applied to the spacecraft as a result of disturbances other than the gravity gradient discussed above. These include such forces as aerodynamic drag, solar pressure, and the earth's magnetic field. The most important of these is aerodynamic drag. Calculations indicate that aerodynamic torque is a second order effect at the altitude under consideration. Selection of a symmetrical configuration, in which



ORBIT PRECESSION

FIGURE 1.1

ORBIT PRECESSION ILLUSTRATION

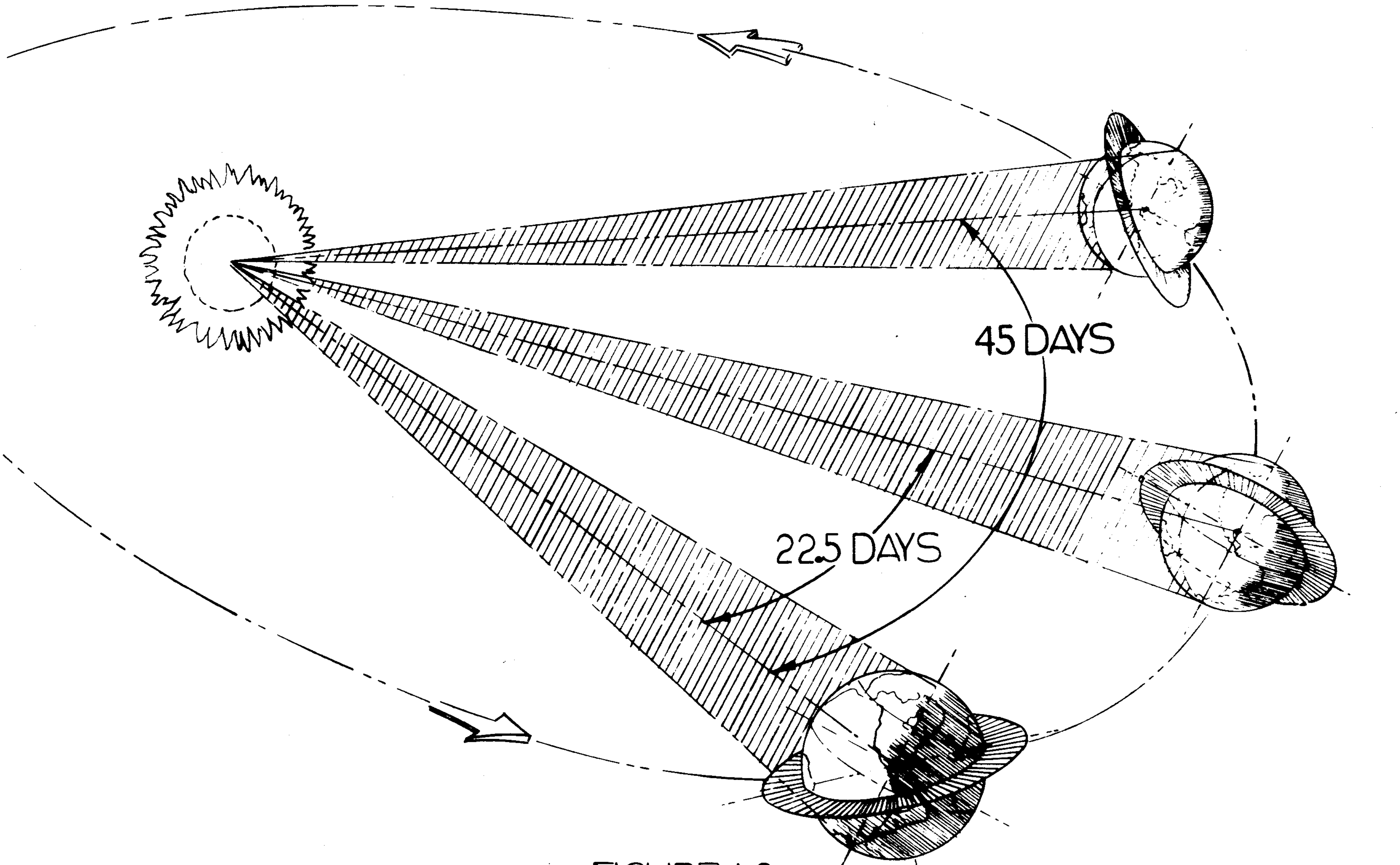
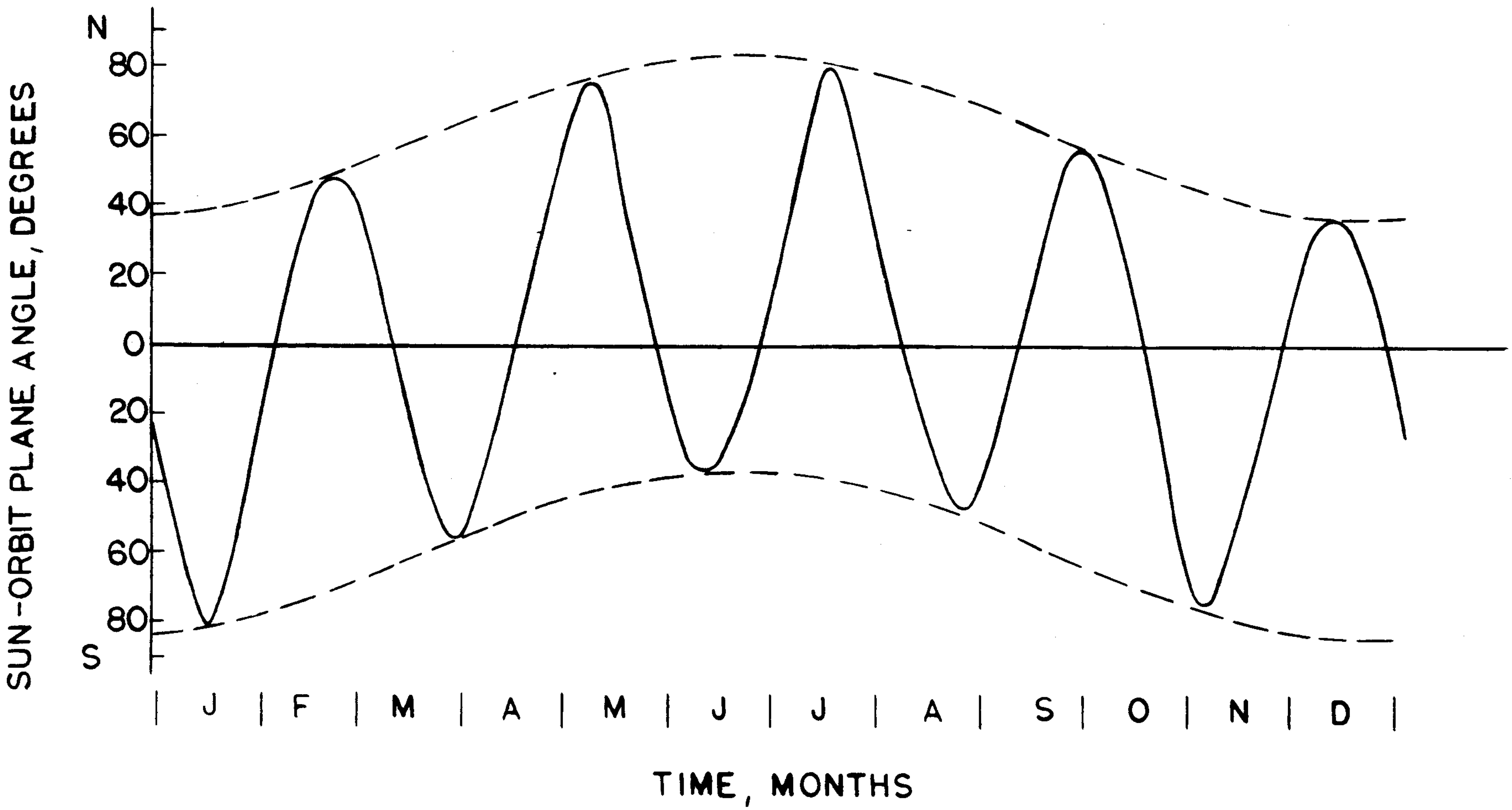


FIGURE 1.2

260 N.M., 60° CIRCULAR ORBIT



SUN POSITION RELATIVE TO ORBIT PLANE

FIGURE 1.3

the resultant of aerodynamic forces passes through or near the center of mass, will further reduce or eliminate this torque.

Because the disturbances mentioned in this section are small compared to the gravity gradient torque, they have not been taken into account in the orientation considerations which follow.

1.2 ORIENTATION CONSIDERATIONS

Space station orientation is influenced primarily by three conflicting pointing requirements, viz., solar cells, astronomical sensors, and earth sensors, together with the disturbances discussed in Section 1.1. Spacecraft thermal control, docking, and orbit maintenance must also be considered although these factors are not as constraining as those previously mentioned.

Since a 10° solar cell array pointing error causes a performance degradation of only 1.5%, precise solar cell pointing is not required. However, the large size and moments of inertia involved can cause substantial disturbances in the spacecraft attitude if the solar arrays are continuously rotated in an oscillatory manner relative to the spacecraft. Reliability considerations also favor the elimination of slip rings for power transmission. Therefore, total cumulative rotation of the solar array relative to the spacecraft should be less than 360° if possible.

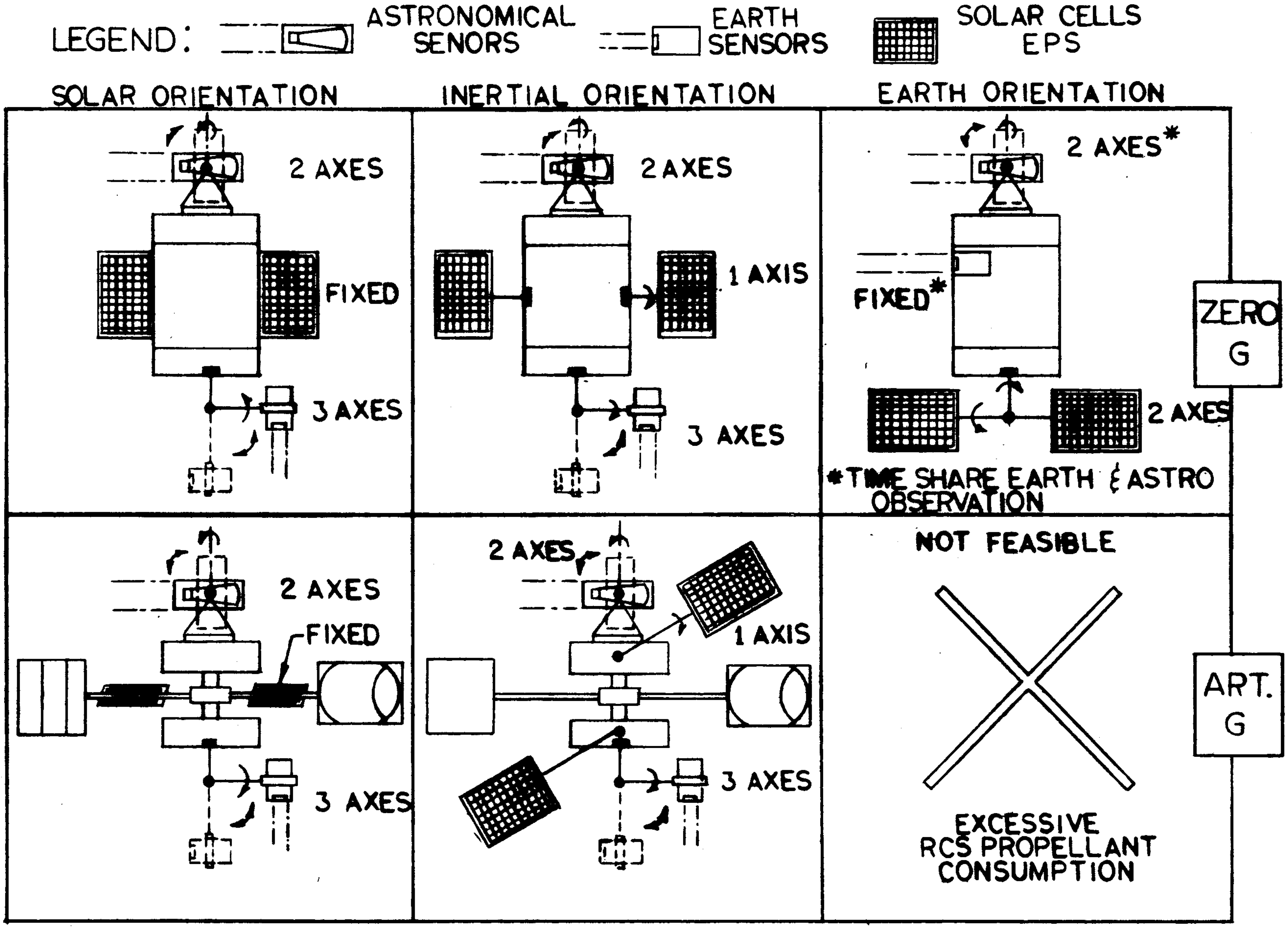
Astronomical sensors should be mounted so as to view the entire celestial sphere during the mission. Viewing of any given point should be available as long and as frequently as possible. Solar instruments represent a special case, since they must view the sun. Pointing accuracy requirements preclude mounting these on the solar cell arrays. They represent, therefore, an additional constraint on station orientation.

Earth sensors are, in most cases, aligned to the local vertical. Their orientation is, therefore, changing continuously, a condition entirely opposed to solar cell and telescope pointing requirements. Since substantial volumes are required for both earth sensor and astronomical sensor installations, it appears necessary to resolve the orientation problem by locating the two groups of sensors in separate sections of the station with the capability of independent motion about one or more axes.

Figure 1.4 illustrates schematically some possible arrangements of zero and artificial gravity space stations with the gimbal axes needed to satisfy the various pointing requirements.

1.2.1 ZERO GRAVITY SPACE STATION

The first factor to be considered in zero gravity station orientation is the gravity gradient problem. As pointed out in Section 1.1.2, torques exist whenever the principal axes of inertia are



SENSOR MOUNTING
ZERO G/ARTIFICIAL G SPACE STATIONS

FIGURE 1.4

not aligned with the local vertical and the orbit plane. Thus, if the station is held in an arbitrary inertial orientation for some period of time, reaction control propellant must be consumed to counteract the torque produced. However, a special case exists if one principal axis is normal to the orbit plane. Inertial orientation will then result in a periodic gravity gradient torque with no secular component. These periodic torques can be absorbed by control moment gyros without propellant expenditure. It will, therefore, be desirable to orient the station with one principal axis normal to the orbit plane to avoid large propellant usage.

Because of the variation in sun angle relative to the orbit plane as shown in Figure 1.3, orientation in accordance with the preceding paragraph requires two degrees of freedom for solar cell pointing. The pointing mechanism can be simplified, however, by using the longitudinal axis of the station as one of the required solar cell axes. If the earth sensors are mounted at one end of the station in a module which can be rotated about the longitudinal axis, both earth sensor and solar cell requirements can be satisfied by orienting the longitudinal axis normal to the orbit plane as illustrated in Figure 1.5. The earth sensors can then track local vertical continuously. The solar cell arrays must be mounted on an axis perpendicular to the longitudinal axis with a rotation capability of $\pm 83\frac{1}{2}^{\circ}$ from the central position shown in Figure 1.5. This rotational requirement results from the sun angle variation illustrated in Figure 1.3.

Because of the precession of the orbit, the longitudinal axis must be repositioned at a rate of approximately 3.3°/day to maintain the station attitude normal to the orbit plane. A roll rate of approximately 1°/day is also required for solar tracking, although it need not be continuous since solar pointing accuracy is not critical.

Astronomical sensors can be mounted on the end of the station opposite the earth sensors. This provides a platform that is inertially fixed (within the stability limits of the station) during observation periods. Observation would probably be interrupted while station reorientation is being performed as described above. The interval between reorientations will depend primarily on the capability of the earth sensor package to compensate for orientation errors.

This concept imposes a limitation on astronomical sensors in that the region near one celestial pole is never within view. The problem can readily be solved by pitching the station 180° as intervals, such as once a year, to permit viewing of the obscured area.

EARTH ORIENTATION-ZERO G STATION
AXIS ALIGNED NORMAL TO ORBIT PLANE

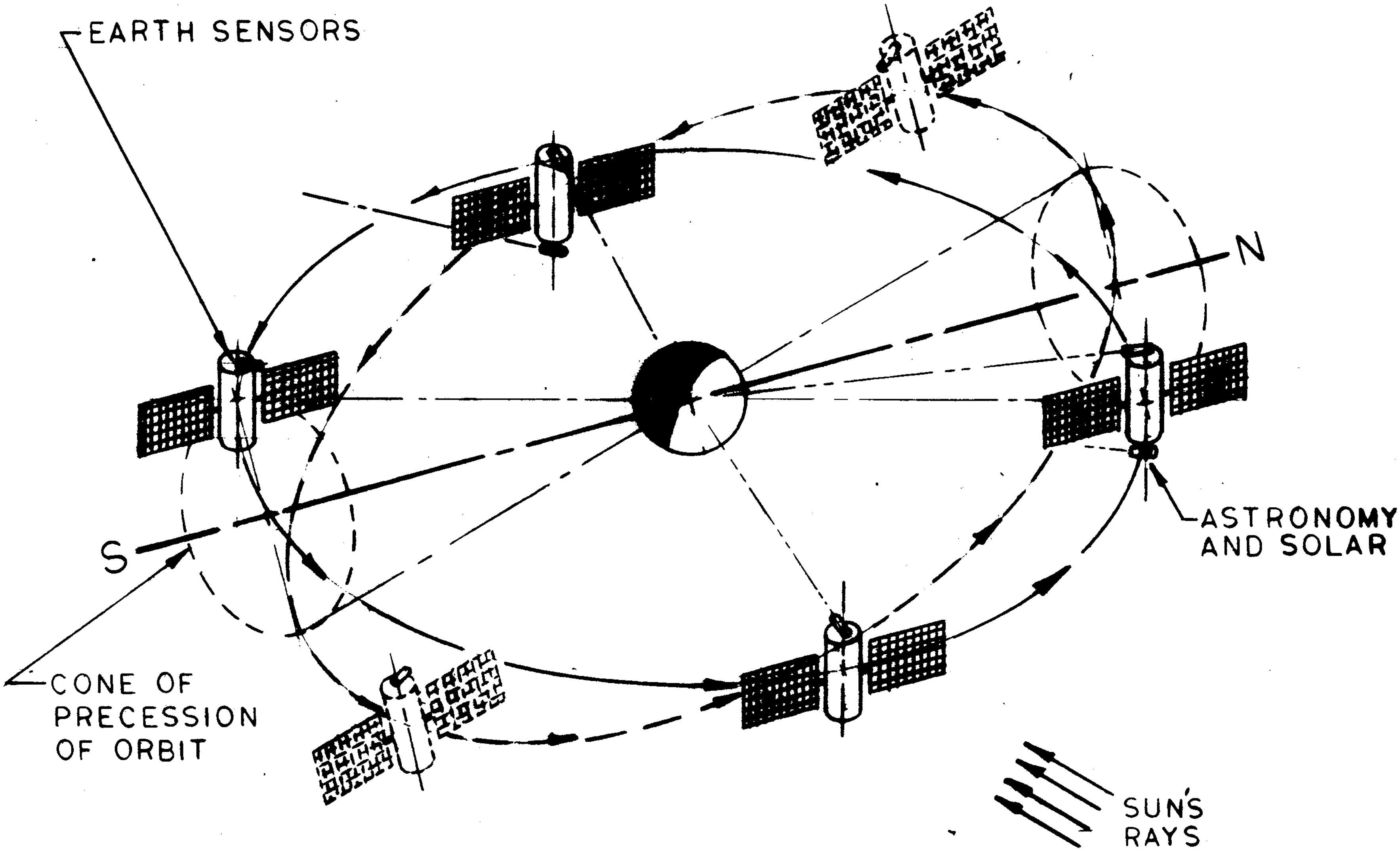


FIGURE 1.5

1.2.2 ARTIFICIAL GRAVITY SPACE STATION

The large angular momentum of a rotating space station reduces the effect of disturbances such as gravity gradient torque to a small amount. This is discussed more fully in Section 2.0. It is noted here because it essentially eliminates such disturbances from consideration in selecting the orientation of a rotating station.

1.2.2.1 INERTIAL ORIENTATION

In considering the artificial gravity space station, a fixed orientation of the spin axis immediately appears desirable to avoid the large amounts of reaction control propellant required to precess the spin axis. If this is done, the solar cells, earth sensors, and astronomical sensors must all be mounted on a counter-rotating, zero gravity hub (see Figure 1.4). In this case, astronomical sensor viewing is restricted to a hemisphere unless the station is periodically inverted to view alternate hemispheres in turn. This, however, would defeat the purpose of the fixed orientation. Earth sensors will be interrupted by the rotating modules during part of each orbit. This can be overcome by suitable timing in most cases. It still represents an inconvenience and would seriously hamper some sensors such as a mapping radar. Solar cells would require at least one degree of freedom in addition to the zero gravity hub bearing axis and would be subject to some intermittent shadowing by the rotating modules.

1.2.2.2 SOLAR ORIENTATION

An alternate to a fixed orientation is alignment of the sign axis toward the sun (Figure 1.6). The most obvious advantage is the ability to fix the solar cells to any part of the station without gimbals. This will improve reliability because, once deployed, the solar cell arrays will be completely static. If the arrays are suitably arranged on the rotating module, a significant improvement in rotational stability is also possible for some configurations.

Solar orientation is advantageous for astronomical sensors as well. By mounting these on the shaded side of the hub, the need for protection from the sun is eliminated. The entire celestial sphere is available for observation in the course of a year. Solar instruments will, of course, be located on the sunlit side of the hub.

The superior planets other than Mars will be available for viewing on roughly the same schedule as the fixed stars, as discussed above. Mars can be observed about every 26 months for a period of several months. Venus and Mercury, however, will not be

SPIN AXIS ORIENTATION
PARALLEL TO SUN'S RAYS

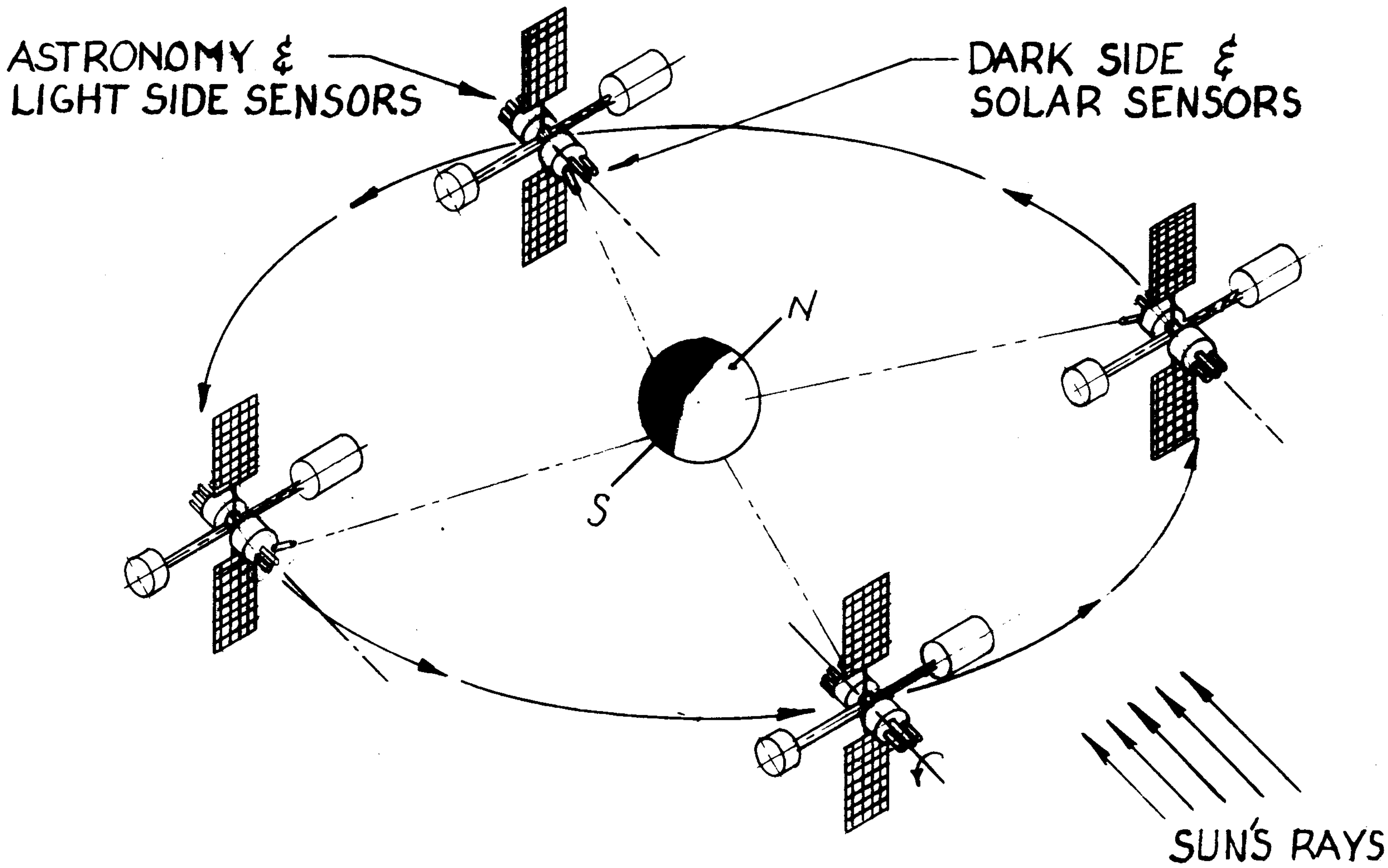


FIGURE 1.6

observable by sensors on the dark side of the station and will require an additional instrument or instruments on the sunlit side of the hub.

In general, a single earth sensing instrument will view either the sunlit side of the earth or the dark side but not both, although there are exceptions. For the sun-oriented artificial gravity space station, division of earth sensors into these two groups is attractive because the shaded side of the station always faces the earth during the sunlit half of each orbit and vice versa. If light-side earth sensors are mounted on the shaded side of the hub and dark-side sensors on the sun side, the rotating module will never interfere with either group.

Solar orientation requires that the spin axis be precessed an average of approximately 1^{o}/day to follow the sun. The frequency of the reorientation maneuvers does not appreciably affect propellant requirements, which are about 4,000 pounds/year for a typical configuration.

1.2.2.3 EARTH ORIENTATION

Two spin axis orientations may be considered: parallel to the local vertical and normal to the orbit plane. The first of these can be eliminated immediately by reason of the excessive propellant required (on the order of 60,000 pounds/day).

Orientation normal to the orbit plane offers some advantages to the earth sensors, which can be mounted in the hub for continuous viewing with no interference from the rotating modules.

Astronomical sensors can view the entire celestial sphere during the year with the exception of the region near one celestial pole. As with the corresponding zero gravity case, this could be overcome by precessing the axis 180^{o} from time to time. However, the propellant cost would be substantial.

The solar cell installation would require two degrees of freedom and would be subject to station shadowing at times.

Because of the precession of the orbit, the spin axis must be precessed to maintain correct orientation. Propellant requirements for this purpose will be approximately 13,000 pounds/year calculated on the same basis used in Section 1.2.2.2.

PRELIMINARY TECHNICAL DATA

FOR EARTH ORBITING SPACE STATION

VOLUME IV

CONFIGURATIONS, INTEGRATION, AND WEIGHTS

SECTION 2.0

STABILITY

ADVANCED SPACECRAFT TECHNOLOGY DIVISION

ENGINEERING AND DEVELOPMENT DIRECTORATE

MANNED SPACECRAFT CENTER

2.0 STABILITY

2.1 INTRODUCTION

This section will consider the effects on the space station of internal and external disturbances of the attitude of either a zero gravity or artificial gravity station. These disturbances will influence the attitude of the zero gravity station in some manner and that of the artificial gravity station in, perhaps, a totally different manner. The variation in the attitude of either space station is considered to be of prime importance because of the celestial, terrestial and solar equipment requirements.

Several types of internal and external disturbances and their implications to the station subsystems will be discussed. Table 2.1 indicates concepts currently under consideration to maintain the required attitude and stability for a rotating station. Probably the same concepts will be required for use in a zero gravity station; however, there may be a considerable difference in the size and operation of the subsystems and components.

2.2 DISCUSSION

The following paragraphs discuss some of the internal forces that influence the attitude of a space station.

2.2.1 INITIAL BALANCE

The artificial gravity space station will require accurate initial balance as well as center of gravity compensation to provide maximum stability for astronomical and earth sensors. Accurate placement of components and systems can help reduce the initial balance problem; however, a balance system will be required for precise trim and for compensation of mass movement within the space station.

2.2.2 INTERNAL MOVEMENTS

The internal movements that will affect the stability of the space station can be separated into three categories: men, cargo, and experiment deployment. Each category is discussed briefly and individually.

2.2.2.1 MAN MOVEMENT

For long duration missions involving a varied experimental program, man will be required to move about the station in order to work effectively in space. That is, man in space will require working, control, living, recreational, and sanitation areas much like man on earth.

TABLE 2.1

DISTURBANCES ACTING ON ROTATING SPACE STATION

DISTURBANCE	EFFECT			COMPENSATION					
	Wobble	C. G. Shift/ Tilt	Reso-nance	Station Momentum	Design	Balance System	Wobble Damper	Control Moment Gyro	RCS
INTERNAL									
INITIAL BALANCE	X	X			X	X			
MOVEMENT:									
Men	X	X		X		X	X	(X)	
Cargo	X	X		X	X	X	X	(X)	(X)
Experiment Deployment	X	X		X	X	X	X	(X)	(X)
MACHINERY	X			X	X		(X)		
FLEXIBILITY			X		X		(X)	(X)	
EXTERNAL									
GRAVITY GRADIENT	X			X					
DRAG	X			X	(X)				
SOLAR PRESSURE	X			X					
MAGNETIC	X			X					
METEOROID	X			X					
DOCKING	X	X		X	X		X		(X)
LEAKAGE	X			X					

( ) Possible additional compensation techniques

2.2.2.1.1 Man Movement in a Zero Gravity Station

For man movement in a zero gravity station the angular displacement in degrees is estimated by using the equation

$$\theta = \frac{m\ l\ d \times 57.3}{I} \tag{1}$$

where:

m	=	Mass of man in slugs
l	=	Moment arm
d	=	Distance moved
I	=	Moment of inertia of station

The results of three possible movements are shown in Table 2.2 and are illustrated in Figure 2.1. The values used for substitution into Equation (1) were:

Case I			Case II and III		
m	=	6 slugs	m	=	6 slugs
l	=	90 feet	l	=	45 feet
d	=	10 feet	d	=	20 feet
Ix-x	=	1.84×10^6 slug ft^2	Iy-y	=	4.52×10^6 slug ft^2
			Iz-z	=	2.95×10^6 slug ft^2

2.2.2.1.2 Man Movement in Zero Gravity Hub of Rotating Station

For man movement in a zero gravity hub of a rotating station, the equation for θ in degrees is developed as follows:

$$\theta = \frac{I_m \omega_m}{I_s \omega_s} \tag{2}$$

$$I_m \omega_m = ml^2 \times \frac{v}{l} \times 57.3 \tag{3}$$

Simplifying equation (3) and substituting for $I_m \omega_m$ in equation (2), one obtains

$$\theta = \frac{m\ l\ v \times 57.3}{I_s \omega_s} \tag{4}$$

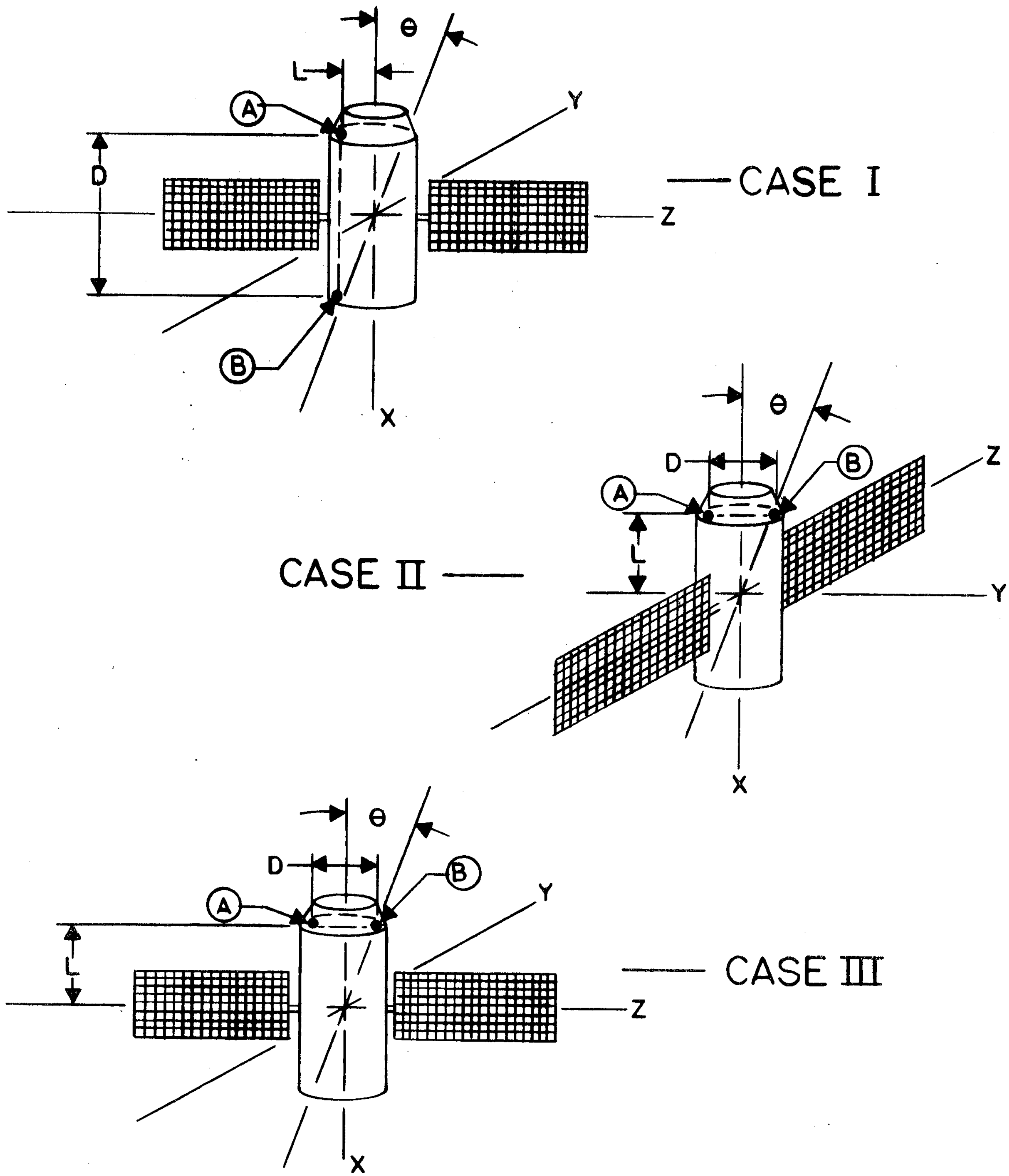
where:

m	=	Mass of man in slugs
l	=	Moment arm
v	=	Velocity
I_s	=	Moment of inertia of station about x-x axis
ω_s	=	Angular velocity about x-x axis.

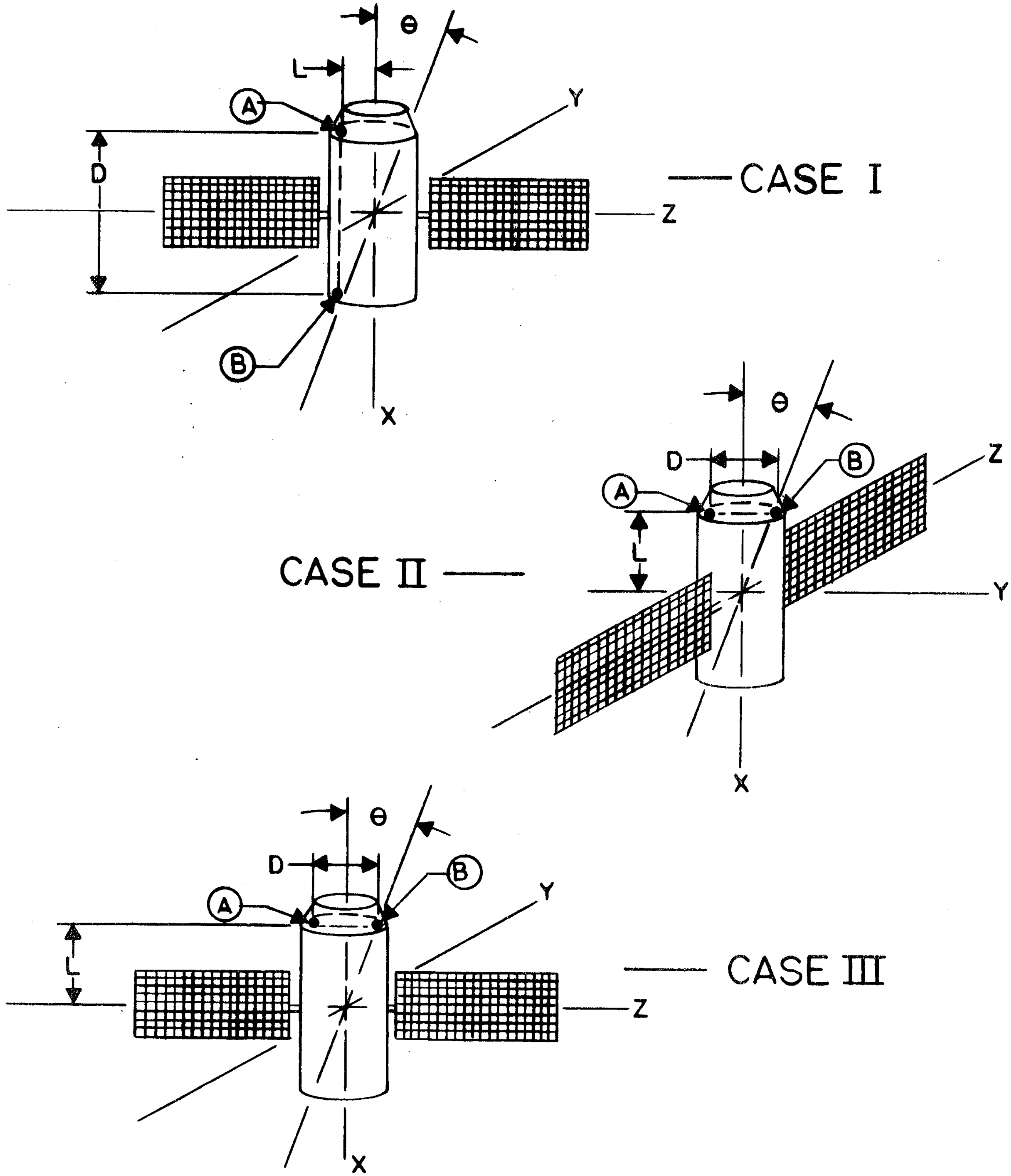
TABLE 2.2

MAN MOVEMENT - STABILIZATION

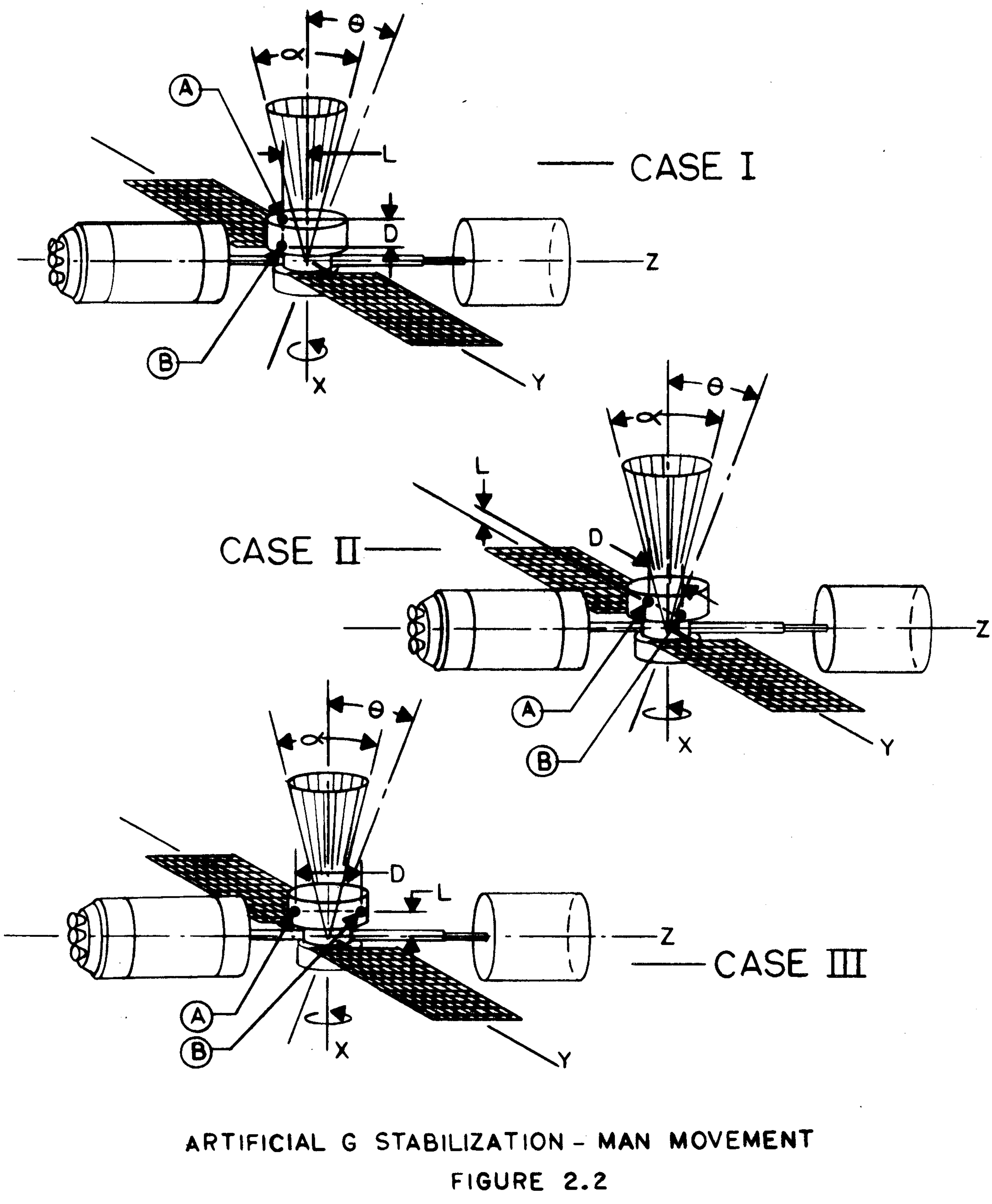
θ DEGREES				α DEGREES		
Direction of Movement	Man Moves	Artificial "G" (Transient)	Zero "G"	Plane	Man Moves	Artificial "G"
X - X	A-B Case I	.0018	.168	XY	C-D Case I	.136
Y - Y	A-B Case II	.0012	.068	XZ	E-F Case II	.011
Z - Z	A-B Case III	.0012	.105	YZ	G-H Case III	.012



ZERO G STABILIZATION – MAN MOVEMENT
FIGURE 2.1



ZERO G STABILIZATION-MAN MOVEMENT
FIGURE 2.1



ARTIFICIAL G STABILIZATION - MAN MOVEMENT
FIGURE 2.2

Moments about the three axes are considered individually. The results **are shown in Table 2.2. Figure** 2.2, **Case I, shows the movement along the x - axis from A to B.** The values substituted in equation (4) are as follows:

$$m = 6 \text{ slugs}$$
$$l = 15 \text{ feet}$$
$$v = 5 \text{ feet per second}$$
$$I_s = 36 \times 10^6 \text{ slug feet}^2$$
$$\omega_s = .4 \text{ radians per second}$$

Figure 2.2 also shows the movement along the y axis (Case II) and z axis (Case III). The values substituted in equation (4) are as follows:

$$m = 6 \text{ slugs}$$
$$l = 10 \text{ feet}$$
$$v = 5 \text{ feet per second}$$
$$I_s = 36 \times 10^6 \text{ slug feet}^2$$
$$\omega_s = .4 \text{ radians per second}$$

2.2.2.1.3 Man Movement in Rotating Portion of Station

For movement in the rotating portion of an artificial gravity station, the man is assumed to move as shown in Figure 2.3. Three cases are shown assuming movement in the xy, xz, and yz planes of the living quarters. The resulting wobble angle for each case is included in Table 2.2. Each movement produces product of inertia changes. The equation used to compute the wobble angle is

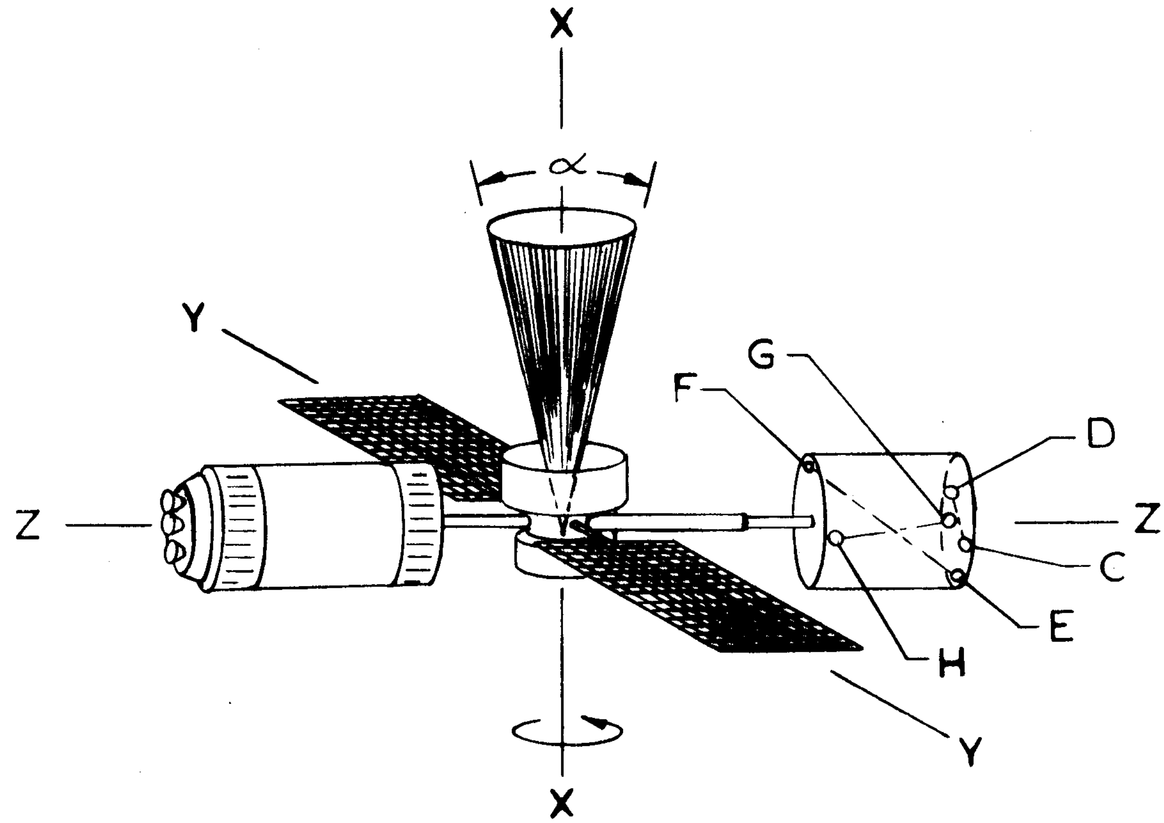
$$\alpha = \tan^{-1} \frac{2 I_{xy}}{I_x - I_y} \qquad (5)$$

where α is the angular displacement from the existing principal axis. Referring to Figure 2.3, the value of α is twice the angle obtained by using equation (5). This equation is used in each case by changing the axes to correspond to the particular plane of interest.

For Case I, the values substituted in equation (5) are

$$I_x = 36 \times 10^6 \text{ slug feet}^2$$
$$I_y = 34 \times 10^6 \text{ slug feet}^2$$
$$I_{xy} = 6 \times 14 \times 14 \text{ slug feet}^2$$

In this case, the man is moving as shown in Case I of Figure 2.3 (C to D) resulting in a distance of 14 feet along both the x and y axis.



CASE I	C-D
CASE II	E-F
CASE III	G-H

STABILIZATION

MAN MOVEMENT-ARTIFICIAL "G" MODULE

FIGURE 2.3

For Case II, equation (5) is written as follows:

$$\alpha = \tan^{-1} \frac{2 I_{xz}}{I_x - I_z} \qquad (6)$$

The values for this case are:

$I_x = 36 \times 10^6$ slug feet ^{2}
$I_z = 3 \times 10^6$ slug feet^2
$I_{xz} = 6 \times 14 \times 20$ flug feet^2

From Case II, Figure 2.3, the distance moved in the x direction is 14 feet and in the z direction is 20 feet.

Case III is similar to Case II; however, equation (5) is written

$$\alpha = \tan^{-1} \frac{2 I_{yz}}{I_y - I_z}$$ where

$I_y = 34 \times 10^6$ slug feet^2
$I_z = 3 \times 10^6$ slug feet^2

and
$I_{yz} = 6 \times 14 \times 20$ slug feet^2

For this case, the distance moved in the y direction is 14 feet and in the z direction 20 feet.

2.2.2.1.4 Summary

Table 2.2 indicates that there are disturbances to either type of station of approximately the same magnitude.

2.2.2.2 TELESCOPE MOVEMENT

In addition to man movement, there will also be movement of scientific equipment such as an astronomical telescope. To indicate the magnitude of the disturbance of repositioning scientific equipment on a zero and artificial gravity station, it is assumed that an astronomical telescope of 481 slugs mass (approximately 15,500 pounds) is moved through 90 degrees in 12.5 minutes. Figure 2.4 indicates the rotation of the telescope with respect to each of the station axes. The angular disturbance is shown in Table 2.3 for both station concepts.

STABILIZATION

TELESCOPE ROTATION

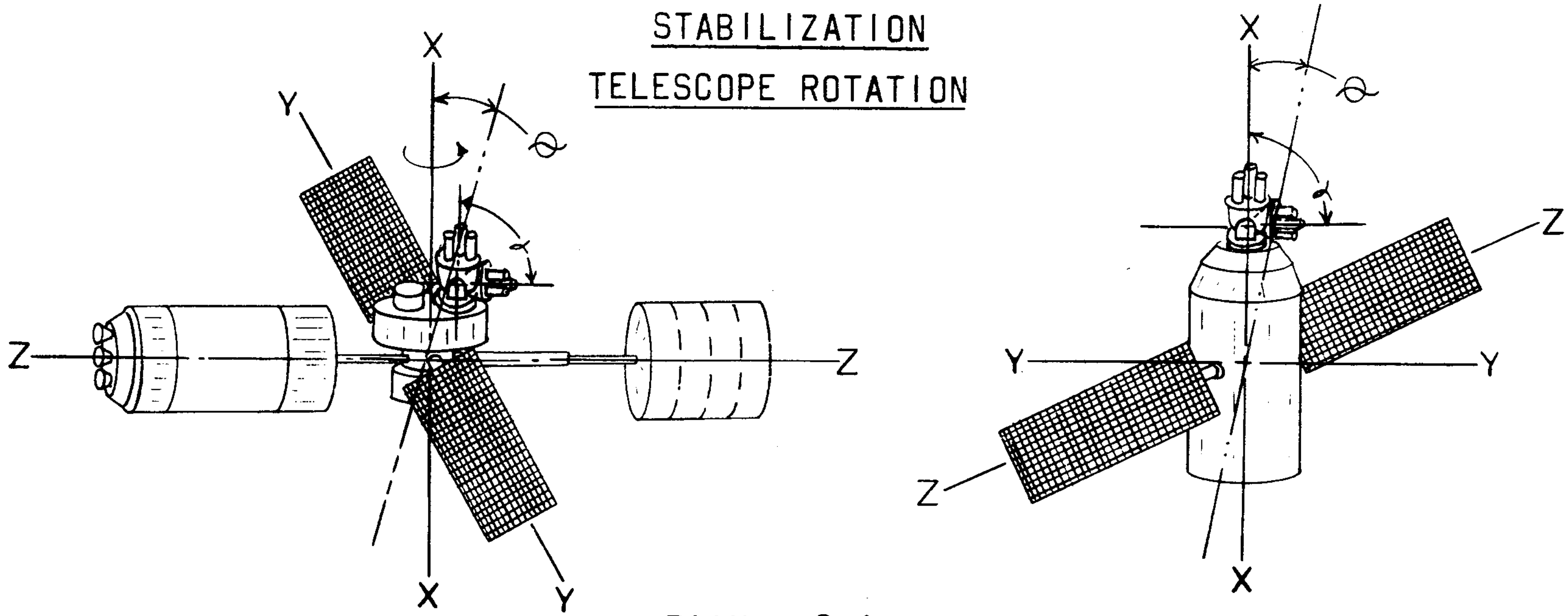


FIGURE 2.4

θ (DEGREES)

	ARTIFICIAL "G"	ZERO "G"
X-X	.00000402	.0235
Y-Y	.00000426	.0096
Z-Z	.00004830	.0147

TELESCOPE ROTATION - 90^{0}/12.5 MINUTES

TELESCOPE ANGLE - $\alpha = 90^{0}$

TABLE 2.3

2.2.2.2.1 Telescope Movement on Zero Gravity Station

The angular disturbance is expressed by

$$(\theta \text{ station})\ (I \text{ station}) = (\theta \text{ telescope})(I \text{ telescope}) \qquad (7)$$

$$\therefore\ \theta \text{ station} = \frac{(\theta \text{ telescope})(I \text{ telescope})}{I \text{ station}} \qquad (8)$$

where

θ telescope = 90°
I telescope = 481 slug feet^2
Ix-x station = 1.84×10^6 slug feet^2
Iy-y station = 4.52×10^6 slug feet^2
Iz-z station = 2.95×10^6 slug feet^2

For a given angular rotation, the moment of inertia about the axis of rotation of the telescope determines the magnitude of disturbance - hence, the higher the station inertia the smaller the disturbance.

2.2.2.2.2 Telescope Movement on Artificial Gravity Station

The angular disturbance resulting from the rotation of an astronomical telescope on an artificial gravity station is expressed by

$$\theta = \frac{(I \text{ telescope})\ (\quad \text{telescope})}{(I \text{ station} \quad)\ (\quad \text{station})}$$

where

I telescope = 481 slug feet^2
telescope = .0021 radians per second

astronomical telescope on an artificial gravity station is expressed by

$$\theta = \frac{(I \text{ telescope})(\omega \text{ telescope})}{(I \text{ station})\ (\omega \text{ station})}$$

where

I telescope = 481 slug feet^2
ω telescope = .0021 radians per second
I station = Ix-x, Iy-y, or Iz-z
ω station = .4 radians per second

The artificial gravity station, due to its inherent stability , is hardly affected by this type of movement as shown in Table 2.3, and Figure 2.4. In addition, the station returns to its initial position upon completion of the telescope movement.

2.2.2.3 CENTER OF GRAVITY EXCURSIONS

Center of gravity location has no significant effect on the stability of a zero gravity space station. This is also true of the non-rotating hub of an artificial gravity station. However, center of gravity excursion in the rotating module of an artificial gravity space station will substantially affect its usefulness as a platform for sensors.

2.2.2.3.1 Man Movement

Crew movements within the rotating module will cause a continual shifting of the center of gravity, resulting in a cylindrical motion of the axis of the non-rotating hub unless compensation is provided. As an example, movement of a man from one extremity of the station to the other will cause a center of gravity shift of approximately 0.1 inch. The threshold value which creates disturbances to experiments is not presently known.

2.2.2.3.2 Cargo Movement

The movement of cargo from the docking port to the interior of the space station will cause disturbances of a magnitude similar to those previously discussed in man movement.

2.2.3 EXTERNAL DISTURBANCES

The external disturbances that influence the stability of the station are discussed briefly in this section. These disturbances are gravity gradient torques, drag, solar pressure, magnetic effects, meteoroid impacts, docking, and leakage.

2.2.3.1 GRAVITY GRADIENT TORQUES

A cursory analysis has been conducted of comparative control requirements of an artificial gravity space station and a zero gravity station. Each station was studied in two basic orientations: (1) the X axis continuously pointed at the sun, and (2) the X axis continuously normal to the orbital plane. The space station was assumed to be in a 260 n.mi. earth orbit inclined at 55 degrees to the equator.

Two categories of control requirements were considered: (1) control to compensate for gravity gradient induced torques, and (2) control to maintain the X axis in its proper alignment. Aerodynamic torques are a second order effect at the altitude being considered and can be neglected. Recently published calculations have shown that gravity gradient torques based on a spherical earth are accurate within one percent. Earth oblateness will, however, affect the motion of the orbit about the earth and can impose a significant requirement on category (2) above. This will be explained in more detail in the analysis which follows.

The analysis assumes that the station is being held to the required alignment. Gravity gradient torques about the individual body axes are presented for a single orbit of each case considered. The summation of these torques represents the control energy required per orbit. The single orbit data have been obtained for spatial geometries representing the earth in four positions (90 degrees apart) about the sun. While these data, developed by a computer program, are not sufficient to estimate the energy requirements for a complete year, they are considered adequate to represent comparative requirements. Control requirements to maintain X axis orientation were hand calculated.

2.2.3.1.1 Earth Orbit Motion

If the zonal hormonic (J_2) is introduced to modify the earth model from a sphere to an oblate spheroid, the rate of regression (in a direction opposite to the station motion) of the node may be expressed as:

$$\dot{\Omega} = 10.05 \left(\frac{R_E}{R_O}\right)^{3.5} \cos i \text{ , Deg/Day}$$

Evaluating this for the orbit being considered, the regression rate is about 4.4 degrees/day. Therefore, in 82 days the node will have traversed one revolution; in 1/4 year, the node will appear displaced about 45 degrees from its initial position. These calculations were used to (1) establish the station orientation to evaluate the gravity gradient torques, and (2) assess the the torques required to slue the vehicle X axis to the desired orientation in space.

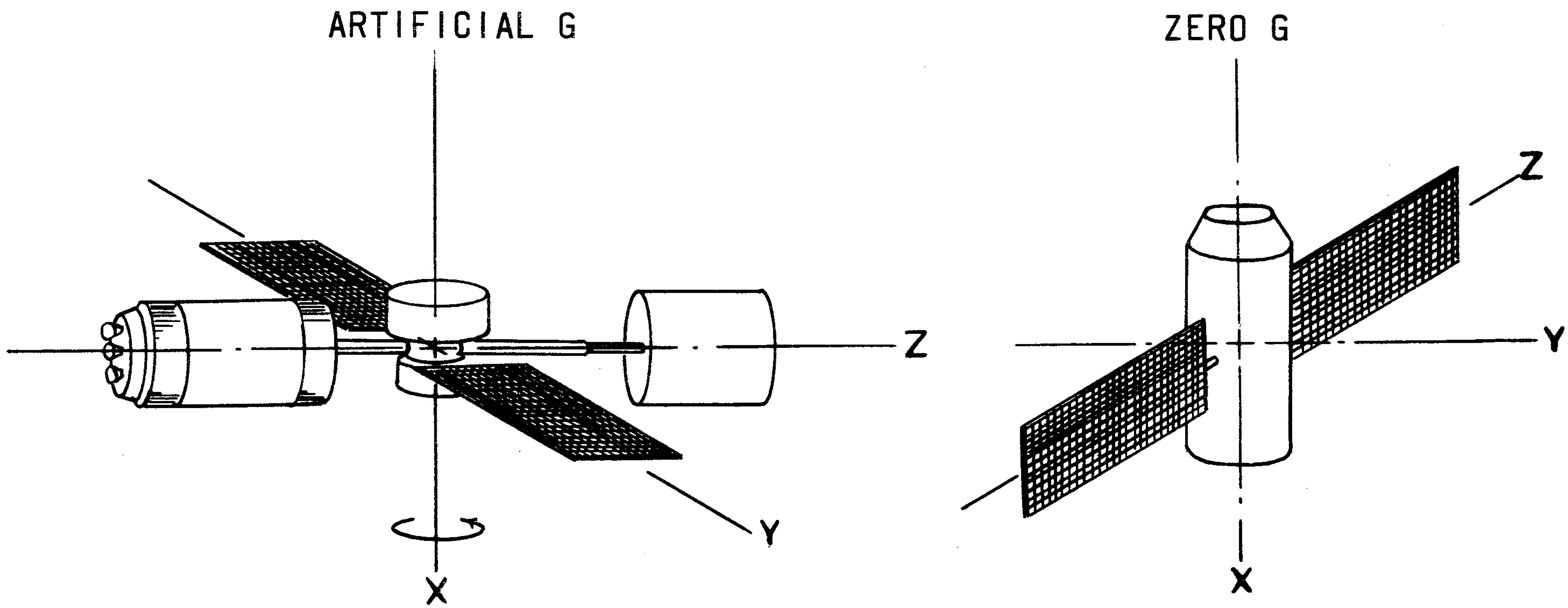
2.2.3.1.2 Artificial Gravity Configuration - Sun Oriented

This confiugration (Figure 2.5) spins about the X axis at 24 degrees/second and develops an angular momentum of $I\omega = 15 \times 10^6$ ft-lb-sec. The Z axis inertia is an order of magnitude smaller than the X and Y inertias so that, for arbitrary orientation, small torques can be expected about the Z axis. This can be seen from the following relationships for gravity gradient torques:

$$\begin{aligned} \Upsilon_x &= f(\psi, \theta, \phi) \quad Iz - Iy \\ \Upsilon_y &= f(\psi, \theta, \phi) \quad Ix - Iz \\ \Upsilon_z &= f(\psi, \theta, \phi) \quad Iy - Ix \end{aligned}$$

where ψ , θ , and ϕ relate the vehicles' axes to the orbital plane.

Figure 2.6 presents typical time histories of gravity gradient torques for a single orbit about the earth. The position of the earth about the sun represents one in which the station's spin axis lies in the orbital plane and remains parallel to the ecliptic.



PARAMETER	ARTIFICIAL GRAVITY	ZERO GRAVITY
STATION I ABOUT X AXIS	36×10^6 Slug Ft^2	1.84×10^6 Slug Ft^2
STATION I ABOUT V AXIS	34×10^6 Slug Ft^2	4.52×10^6 Slug Ft^2
STATION I ABOUT Z AXIS	3×10^6 Slug Ft^2	2.95×10^6 Slug Ft^2

ARTIFICIAL AND ZERO GRAVITY STABILITY PARAMETERS

FIGURE 2.5

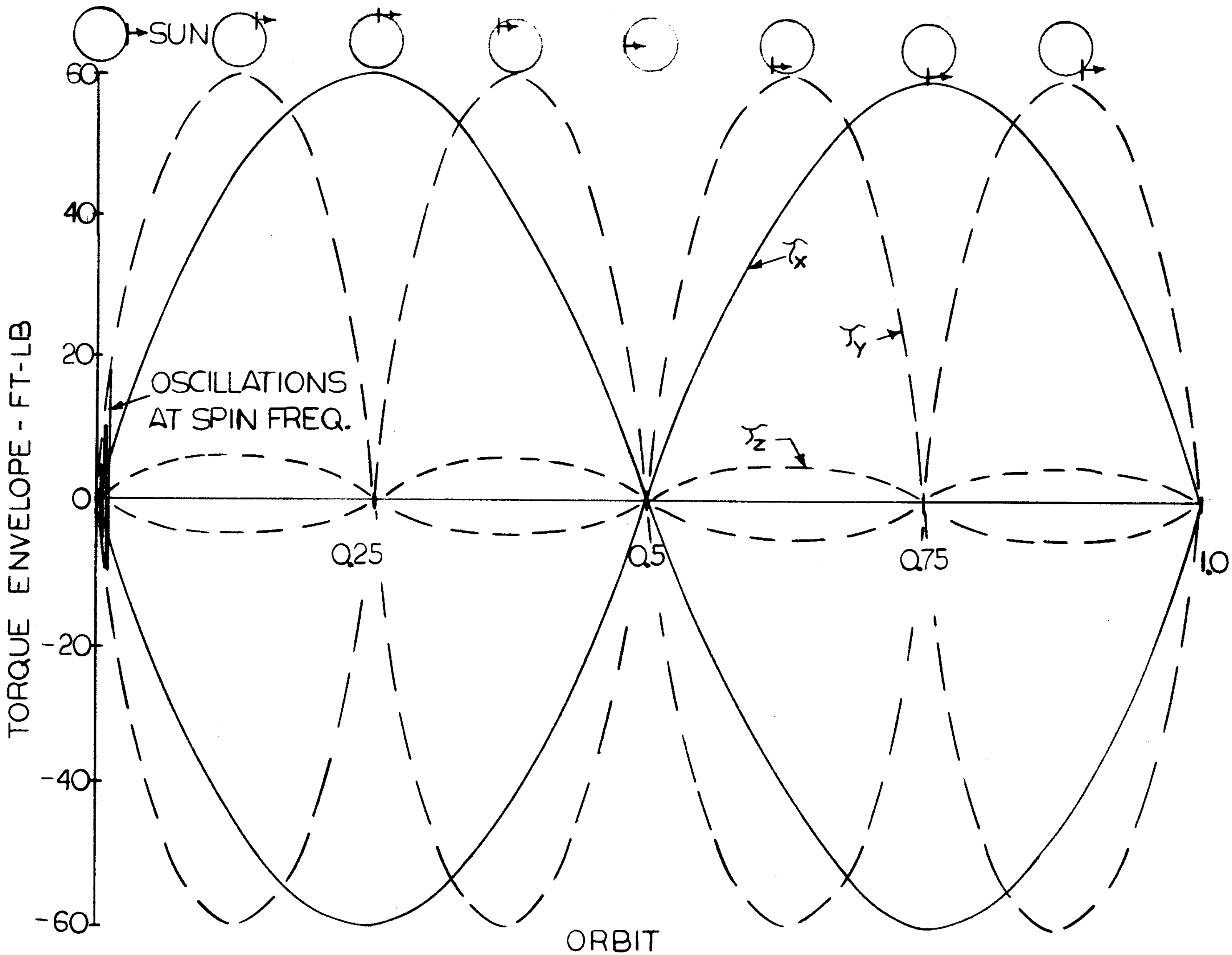


FIGURE 2.6 - SPIN STABILIZED SUN ORIENTED GRADIENT TORQUES

Other solar positions were investigated which accounted for the tilting of the spin axis with the orbital plane which is caused by precession: the data were similar to Figure 2.6 with a slight reduction in peak torques and different phasing relationships among the axes. The data represent envelopes of cyclic torques caused by the rotation of the Y and Z axes.

The stability of the artificial gravity station in maintaining a sun orientation was quite good for the single orbit considered. Deviation of the spin axis during an orbit was only a few tenths of a degree in the worst case.

Because of the angular momentum of the station, control power will be required to slue the spin axis through 360 degrees in one year to maintain the sun orientation. If continuous control is applied, the torque required is the angular momentum times 0.985 degrees per day, or about 3 ft -lb constant torque.

2.2.3.1.3 Zero Gravity Configuration - Sun Oriented

The moments of inertia of this configuration are of the same order of magnitude about all axes (Figure 2.5) so that the gravity gradient torques, for an arbitrary orientation, will be similar. However, because of the lack of spin stability, the inertial orientation must be controlled. Efforts to evaluate this configuration with the aforementioned computer program were unsuccessful because of the angular divergence experienced during a given orbit. Hand calculations were made to estimatethe torques at a solar poistion in which the X axis was parallel to the ecliptic and neither Y or Z axes lay in the plane of the orbit. The maximum torques calculated in this condition were 1.80, 1.61 and 4.56 ft-lb about the X, Y, and Z axes respectively.

2.2.3.1.4 Artificial Gravity - Spin Axis Normal to Orbit Plane

Since this configuration is also spin stabilized, the spin axis will have to be slued in a coning motion to account for the precession of the orbit. At the computed precession rate, the energy requirement will be nearly four times the energy required to slue the sun oriented station, since the momentum vector must be re-directed through 3.6 deg/day as compared to 0.985 deg/day.

For spin axis orientation normal to the orbital plane, no disturbances will be seen about the Y and Z axes. An oscillatory torque, whose frequency is twice the spin rate, of about 60 ft-lb will be seen about the X axis; however, negligible variations will be seen in the spin rate. The torque required to account for gravity gradient disturbances, therefore, is essentially zero for this orientation.

2.2.3.1.5 Zero Gravity - Longitudinal Axis Normal to Orbit Plane

This configuration was initially positioned such that the Z axis was directed toward the earth. Of the two axes in the orbital plane, this axis has the minimum inertia and is therefore gravity gradient stabilized. With proper initial alignment, torque was developed about the X axis only and oscillated between $\pm$ 3 ft-lb. This caused an excursion about this axis of $\pm$ 38 degrees. No other deviations were experienced.

2.2.3.1.6 Summary

Spin stability was sufficient to maintain both artificial gravity station orientations with no control for a given orbit. Because of the apparent insensitivity to the gradient torques, the sun orientation would appear preferrable from the standpoint of control energy.

The zero gravity configuration has its minimum distrubances from gravity gradient torques in the orbit orientation with the solar panels (Z axis) directed toward the earth. Although this orientation requires that the X axis be slued to compensate for the precession of the orbit, the energy requirement will be small and this orientation is preferred from a control energy standpoint.

2.2.3.2 DRAG EFFECTS

The drag effect on the station stability is a function of the $\frac{C_D A}{W}$, resulting primarily in orbit decay. Since the station requires periodic resupply, it is anticipated that the maintenance of the prescribed orbit will be accomplished by utilizing the logistic vehicle propulsion system. The onboard stabilization system will be required to maintain the proper attitude during the orbit correction maneuver.

2.2.3.3 OTHER DISTURBANCES

Currently, there is no indication that the station stability will be significantly affected by solar pressure, magnetic effects, meteoroid impacts, docking impact, or leakage.

PRELIMINARY TECHNICAL DATA

FOR EARTH ORBITING SPACE STATION

VOLUME IV

CONFIGURATIONS, INTEGRATION, AND WEIGHTS

SECTION 3.0

DESIGN INTEGRATION

ADVANCED SPACECRAFT TECHNOLOGY DIVISION

ENGINEERING AND DEVELOPMENT DIRECTORATE

MANNED SPACECRAFT CENTER

3.0 DESIGN INTEGRATION

This section of the report is concerned with the process of converting experiment and system requirements into configuration design concepts, insuring compatibility of the configuration with the overall objectives of the mission, and identifying trade-off areas. The integration of experiment requirements and system requirements is discussed in separate sections, but the two requirements are closely interrelated.

Because this study basically encompassed the conceptual phase, only those requirements which were judged to significantly affect the space station general arrangement and mechanization were investigated.

3.1 EXPERIMENT INTEGRATION

Experiment requirements were obtained primarily from the Marshall Space Flight Center's (MSFC) Space Station Working Group. Seven categories of experiments were established by the Space Station Requirements Steering Committee:

a. Astronomy
b. Earth Resources
c. Meteorology
d. Biology
e. Long-term Flight
f. Research & Development in Advanced Technology
g. Orbital Operations and Logistics

Table 3.1 summarizes the gross experiment requirements. The 9 man, "small" station experiment volume requirements are doubled for the 24 man, "large" station or for the dual station concept.

3.1.1 ASTRONOMY

Table 3.2 lists the astronomical instruments which have been accommodated in the configuration study. As pointed out by MSFC, not all instruments may be included on the initial launch. However, for the configuration study it was assumed that all hardware and interface provisions would be initially provided on the station. If an instrument were actually carried to the station on a later logistics flight, it would be installed on its mounting provisions by the crew. The configuration drawings (see Section 4.0) will show all the instruments included in Table 3.2.

Station orientation and instrument pointing considerations, as discussed in Section 1.0, have established the requirement for a two axis gimbal mount for the astronomical sensor installation.

TABLE 3.1

GROSS EXPERIMENT REQUIREMENTS

9-MAN SPACE STATION

EXPERIMENT DISCIPLINE	STATION VOL, CU FT	LAUNCH WT., POUNDS	AVG. POWER KW	ZERO G RQMTS	STATION STABILI-ZATION	MANPOWER, MH/YR
Astronomy	1,200	9,000	.40	-	$\pm 1/4^{\circ}$	4,600
Earth Resources	1,200	11,800	1.50	-	$\pm 1/4^{\circ}$	3,000
Meteorology	1,200	2,700	1.00	-	$\pm 1/4^{\circ}$	3,500
Biology	1,200	13,400	1.50	10^{-5}G	-	4,600
R&D in Advanced Tech.	1,600	3,000	.70	Need	-	3,200
Long-Term Flight Biomed/Behavioral	800	3,400	.30	Need	-	4,100
Orbital Operations & Logistics	- -	3,000	.10	-	-	-
TOTALS	7,200	46,300	5.5	-	-	23,000

24-MAN SPACE STATION, or
DUAL SPACE STATION CONCEPT

EXPERIMENT DISCIPLINE	STATION VOL, CU FT	LAUNCH WT., POUNDS	AVG. POWER KW	ZERO G RQMTS	STATION STABILI-ZATION	MANPOWER, MH/YR
Astronomy	2,400	10,000	.75	-	$\pm 1/4^{\circ}$	9,200
Earth Resources	2,400	14,000	1.50	-	$\pm 1/4^{\circ}$	5,400
Meteorology	2,400	2,700	1.00	-	$\pm 1/4^{\circ}$	7,000
Biology	2,400	21,500	2.50	10^{-5}G	-	9,200
R&D in Advanced Tech.	2,400	3,000	.70	Need	-	3,200
Long-Term Flight Biomed/Behavioral	2,400	9,000	.60	Need	-	9,000
Orbital Operations & Logistics	- -	8,500	.20	-	-	3,000
TOTALS	14,400	68,700	7.25	-	-	46,000

TABLE 3.2

ASTRONOMICAL INSTRUMENTS

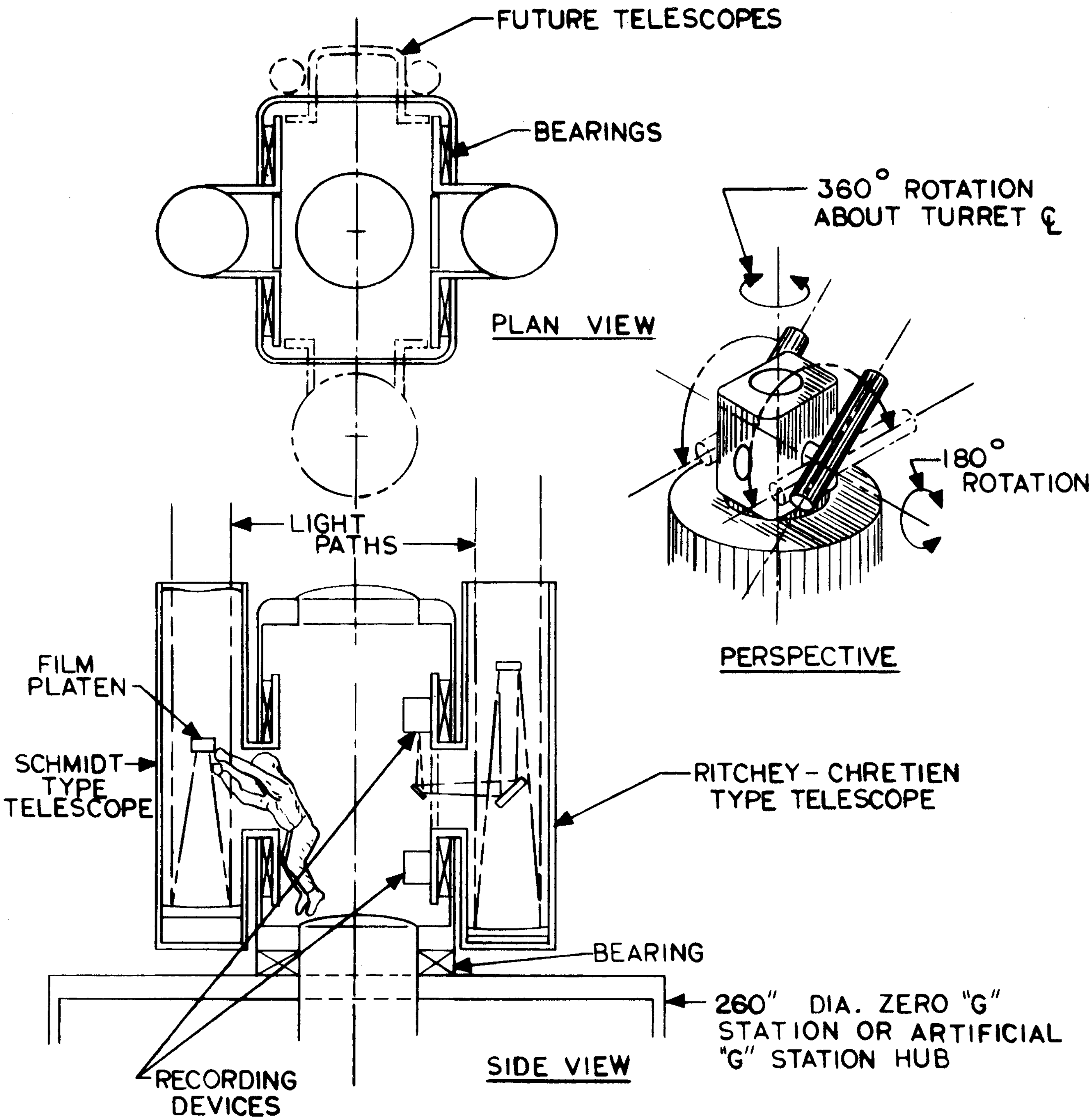
	WEIGHT, LBS.	SIZE	POWER, WATTS	DATA RECOVERY MODE	SPECIAL TECHNOLOGY REQUIRED
RADIO ASTRONOMY					
.05 to 10 MM Wave Length Parabolic Reflector	1,500	10' Dia.	40	Mag. Tape	Cryogenic Coolers (.25-10°K) Close Tolerance Structures, Detectors
OPTICAL					
IR Thru UV Wide Angle Schmidt	1,000	40" x 160"	50	Film, TV	
Moderate Field-IR Thru UV General Purpose	1,000	40" x 160"	50	Film, TV	Cryogenic Coolers (10° - 80°K)
High Resolution Diffraction Limited IR Thru UV	6,000	40" x 160"	150	Film, TV	Spectrally Selective Film
SOLAR					
Coronograph	1,500	12" x 240"	50	Film, TV	
Spectroheliograph	2,000	36" x 240"	50	Film, TV	
X-RAY & GAMMA-RAY					
Arrays: 10 MEV	2,000	50 Ft.^{2}	15	Mag. Tape	Spark Chamber, Cevenkov Array
0.2-20 MEV	1,800	50 Ft.^{2}	15	Mag. Tape	Scintillator, Solid State Arrays
0.2-20 KEV	900	50 Ft.^{2}	15	Mag. Tape	Proportional Counter Arrays
X-Ray Imaging Telescope (Stellar & Solar)	1,000	12" x 240"	50	Film, TV	
TOTALS	22,700		485		

This allows the instruments to be pointed in the required direction independent of station orientation. Pointing accuracy requirements are dependent on the resolution desired from each instrument, generally on the order of arc seconds. It is feasible to stabilize the station to within $\pm \frac{1}{4}^{\circ}$. Thus, fine pointing must be provided within the instrument or gimbal mount. Two possibilities for accomplishing fine pointing are: (1) provide a "fine" mechanical drive system for the gimbal mount, or (2) incorporate a "semi-released" or "soft" attachment system between the instrument and the gimbal mount which will allow the instrument to be independently controlled. Pointing error signals would be derived from the celestial source being observed. The "soft" attachment system appears to afford the maximum pointing accuracy. Figure 3.1 shows a telescope installation concept based on a turret type of mount to allow shirt sleeved access to the sensors by the crew. The turret atmosphere is evacuated during instrument operation. The turret is basically the same for both zero and artificial gravity configurations. It is installed on the non-rotating hub of the artificial gravity station. As the figure indicates, the instruments may be aimed at any point within one-half of the celestial sphere with a given station attitude. Within a period of time the entire celestial sphere can be viewed because of orbit precession or movement of the earth about the sun depending on the station orientation mode as discussed in Section 1.0.

In addition to the sensor installation, appropriate laboratory volume is required for instrument maintenance, auxiliary equipment, experiment set-up, film development, data reduction, etc. Volume is required adjacent to the sensor turret for equipment directly associated with the sensors themselves. Other equipment may be located away from the sensors if desirable, such as in the artificial gravity module of the artificial gravity station. A minimum laboratory pressurized volume of 1200 cubic feet has been specified in addition to the sensor installation. Because of the nature of the data from the astronomy experiments, it is envisioned that data reduction equipment could be designed that would also support the earth resources and meteorology experiments.

3.1.2 EARTH RESOURCES AND METEOROLOGY

The earth resources and meteorology experiments are directly related as to pointing direction and accuracy, data processing, sharing of sensors, etc.; therefore, they are grouped together for this study phase. Table 3.3 lists the equipment items considered for these experiment categories. For an artificial gravity station, sensor pointing and station orientation requirements (see Section 1.0) dictate a three axis gimbal installation of all sensors for gross pointing, plus a fine



TYPICAL TELESCOPE MOUNTING TURRET

FIGURE 3.1

TABLE 3.3

EARTH RESOURCES & METEOROLOGY EQUIPMENT

	VOLUME, CU.FT.	WEIGHT, LBS.
EARTH RESOURCES - INSTRUMENTATION		
Metric Cameras	50	1,000
Typical High Resolution Panoramic Cameras	50	1,400
Multi-Spectral Tracking Telescope	50	900
Synoptic Multi-Band Cameras	50	1,000
High Resolution Radar Imager	13	210
Radar Altimeter/Scatterometer	18	50
Wide Range Spectral Scanner	2	115
Electrical Support Equipment for above	4	*
IR Long Wavelength Spectrometer	4	60
Infrared Radiometer	4	50
Passive Microwave Imaging (Stereoscopic)	8	150
Passive Microwave Spectrum Emissions	8	100
UV Spectrometer	2	100
Laser Altimeter/Scatterometer	4	200
Electrical Support Equipment for above	6	*
Absorption Spectroscopy	6	35
Radio Frequency Reflectivity	2	33
Magnetometer	4	50
Gravity Gradiometer	4	15
EARTH RESOURCES - SUPPORT EQUIPMENT		
Display Equipment/Associated Electronics	60	1,200
Film Storage Canisters	90	5,000
Spare Parts	40	500
Magnetic Tape Units (6 Req'd)	6	300
Film Processing Equipment	30	250
METEOROLOGY - INSTRUMENTATION		
IR Spectrometer	1	53
IR Inteferometer Spectrometer	1	56
IR Radiometer	0.2	38
Polarimeter	0.5	100
UV Spectrometer	1	97
UV Photometer	0.2	50
Microwave Radiometer	0.1	49
Star Tracker	5	125
Radio Propagation	10	224
Spherics Detector	2	35
Camera	2	55
TV Camera	2	84
Radiation Detector	2	40
Micrometeoroid Detector	3	50
Ionospheric Sounder	2	25
Sounding Probes	10	250

TABLE 3.3 - CONTINUED

	VOLUME, CU.FT.	WEIGHT, LBS.
METEOROLOGY-SUPPORT EQUIPMENT		
Electronic Support Equipment	38	*
Sounding Probe Launcher	4	150
Magnetic Tape Units (16 Req'd)	16	800
Deployment Chamber	15	60
TOTALS	630 $Ft.^3$	15,059

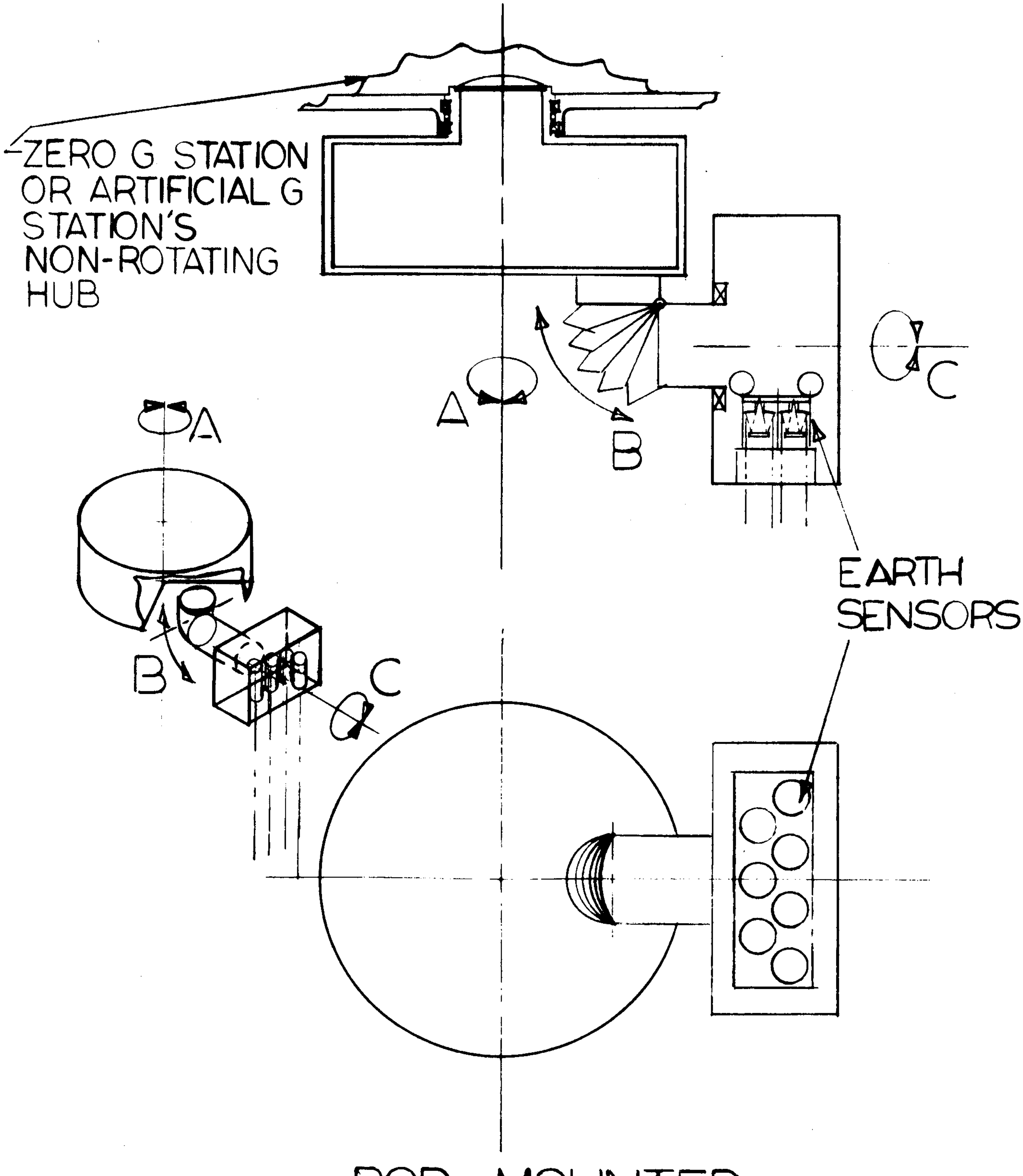
* Included in weight of appropriate sensors

pointing control for certain sensors to achieve the desired resolution. The major impact on the configuration is indicated by Figures 3.2 and 3.3 which show schematics of pod mounted and turret mounted earth sensor concepts, respectively. For the pod mounted concept, all sensors are installed within a pod which is mounted to the station through a two axis gimbal (axes A and B). This gimbal allows the pod axis C to be positioned normal to the orbit plane. As the station circles the earth, the pod is rotated about axis C to maintain the sensor pointing along the local vertical. The relative velocity vector between the station and the earth's surface does not always lie within the orbit plane because of the earth's rotation; therefore, it may be required to skew axis C to provide image motion compensation. This may be accomplished by a biased movement about axes A and B during each orbit revolution.

The turret mounted sensors utilize axes A and B to point the sensors along the local vertical and use axis C to maintain sensor alignment with the ground track. The axis geometry difference between the two concepts is indicated in the figures.

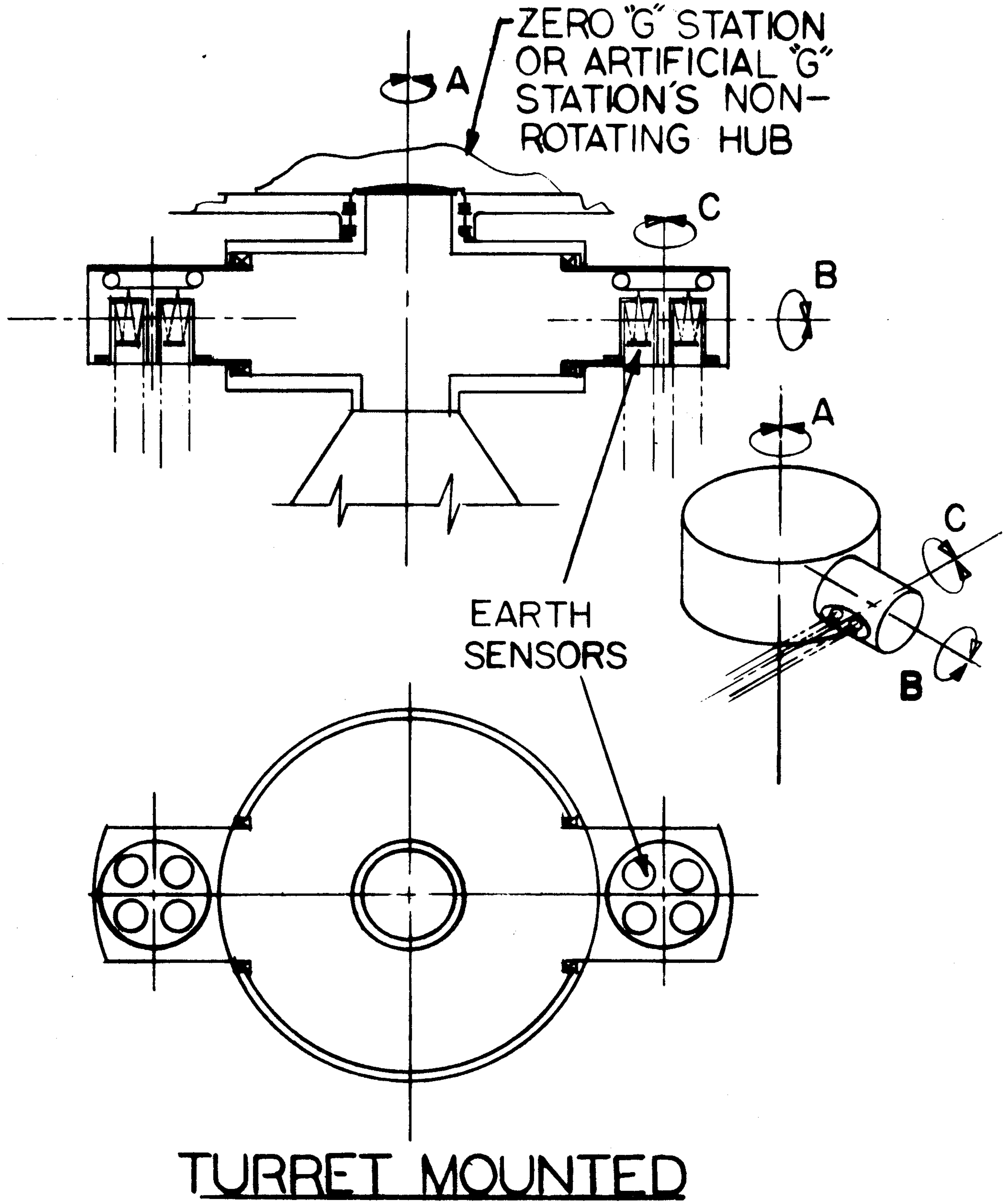
An apparent advantage of the pod mounted concept is that the primary motion required for pointing is a uniform rotation about axis C, once the pod is aligned to the orbit plane. However, because of a probable slight eccentricity of the orbit the motion about C may not be uniform. Some bias movement about axes A and B may be necessary as previously described. For the turret installation, the motion about axes A and B is complex to maintain the pointing along the local vertical throughout an orbit revolution; the motion about axis C is a simple bias to compensate for the earth's rotation. Further analysis of the required motion and mechanization is necessary to aid in concept selection; however, basic packaging and structural considerations may be the dominant factors.

The zero gravity station would appear to eliminate the requirement for earth sensor gimballing except that solar panels and astronomical sensors must be considered. If it were desired to operate both astronomy and earth viewing equipment simultaneously, the earth sensors would require the same type of installation as for the artificial gravity station. Solar panels must always be gimballed unless the station is sun oriented. The best compromise appears to be an inertial station orientation. A special case is to fix the zero gravity station's longitudinal axis normal to the orbit plane. Because of the low orbit precessional rate, this can be an inertially fixed attitude for several hours to allow astronomical observation. For this orientation, the solar panels require only a single axis of freedom in conjunction with rotation about the station's longitudinal axis. Astronomical sensor operation is unencumbered. The earth sensors require a single axis of freedom, parallel to



POD MOUNTED
EARTH SENSORS

FIGURE 3.2



TURRET MOUNTED
EARTH SENSORS

FIGURE 3.3

the station's longitudinal axis. However, because of the high resolution of certain earth sensors, two more axes of freedom appear to be necessary for "trimming" the effects of orbit precession, orbit eccentricity, and the earth's oblateness. The turret installation is, therefore, recommended for earth sensor installation on the zero gravity station.

Fine pointing control to achieve high resolution may be accomplished by connecting the sensors to the turret or pod through a stabilized platform. This platform is controlled to the necessary accuracy, which may be on the order of arc seconds for certain instruments.

Other requirements such as special windows, vacuum operation, cryogenic temperatures, film changing, maintenance, etc. are readily met by either the pod or turret sensor mount concept.

A total station volume allotment of 2400 cubic feet, including sensor installation, has been made for a 9 man station. This provides for installation of auxiliary equipment and allows for maintenance, etc. Certain functions can be performed in the artificial gravity area of an artificial gravity station. These functions include data reduction and transmission and equipment repair.

3.1.3 BIOLOGY

A minimum biology laboratory pressurized volume of 1200 cubic feet is required. Zero gravity is the primary reason for having biology experiments in orbit. A gravity level of 10^{-5} g or less is required. The following data are pertinent to this low level of acceleration:

a. For a station weighing 200,000 pounds, an applied force of 2 pounds will produce 10^{-5} gravity linear acceleration.

b. Aerodynamic drag, even with large solar panels, will not exceed approximately 3/8 pound force for 260 n.m. altitude.

c. Movements of the crew and other masses within the station will produce transient linear accelerations of the space station exceeding 10^{-5} gravity. Some degree of isolation of the biology experiments from the effects of mass movement may be required. A spring-damper suspension system for critical experiments may suffice. The biology laboratory should be located as near the station's mass center as possible to minimize effects of angular motion of the station.

3.1.4 RESEARCH AND DEVELOPMENT IN ADVANCED TECHNOLOGY

A minimum laboratory pressurized volume of 1600 cubic feet has been specified for this experiment category. A volume of 2400 cubic feet is desirable. Included in the specified laboratory volume is the requirement for a test cell of 600-900 cubic feet which must have the following capabilities:

a. Vacuum or pressurized operation with provisions for an inert atmosphere of nitrogen or helium.

b. Accessibility to the outside through a large hatch (5 feet by 6 feet) which can be remotely operated.

c. Remote movement of large experiment apparatus in and out of the test cell.

d. One-man airlock access.

The general lab area contains instrumentation, racks for experiment storage, displays and controls, tools, etc. This area will be connected to the station environmental control system. The majority of the actual experiment operations will be performed within the test cell or exterior to the station.

Because approximately 55 percent of the experiments are insensitive to gravitational requirements, a portion of the R&D lab facilities may be located in the artificial gravity module of an artificial gravity station. Certain experiments require a low level of gravity (from a few thousandths of a g up to .1 g); therefore, the rotating hub of the artificial gravity station may be utilized as a centrifuge for these experiments.

Table 3.4 summarizes the requirements for the R&D laboratory.

3.1.5 LONG TERM FLIGHT

This experiment category involves the development of procedures, systems, etc. for future long duration missions and encompasses the biomedical and behavioral experiments. Provisions for conducting this program are integrated with the station systems and medical facilities. Special laboratory volume of a minimum of 800 cubic feet at zero gravity is required.

The biomedical portion of the experiment program involves assessing the long term effects of zero gravity on the crew; therefore, for the artificial gravity station, two to four crewmen will be required to live and work exclusively in the zero gravity hub.

GENERAL

- Total Logistics Requirements: 44,000 lb
- Equipment to be launched with initial station: 3000 lb
- Data to be returned to Earth: 1300 lb
- Power Requirements: 13000 KW-HR
  700 Watts - average - $10\frac{1}{2}$ hour/day
  6 KW - peak
- Crew Time: 16000 manhours
- Orbital Support Equipment
  - Space Suits
  - AMU
  - Portable Life Support Systems
  - Tethers
- Zero Gravity Required
- Accessibility to Centrifuge

LABORATORY REQUIREMENTS

General R & D Lab

- Environmentally Controlled by Space Station
- 1000 ft^3 (minimum) - 1500 ft^3 (desirable)
- Work Table/Peripheual work area
- Experiment Storage Racks
- General Purpose Equipment Storage Area (185 ft^3)
- Viewport and Airlock to Test Cell
- Small Equipment Airlock (8 ft^3)
- Fluid Experiment Area
- Pressure Door (3 ft. in diameter)
- Display Console/Computers
- Outside Viewport

Test Cell

- Separate Environment Control Capability
- Vacuum Operating Capability
- 600 ft^3 (minimum) - 900 ft^3 (desirable)
- Large Door to outside
- Test Fixture
- Equipment Boom

Outside Station

- 10-12 Mounting Brackets
- 4 equipment mounting pads
- Small boom

RESEARCH AND DEVELOPMENT IN ADVANCED TECHNOLOGY LABORATORY

REQUIREMENTS SUMMARY

TABLE 3.4

3.1.6 ORBITAL OPERATIONS AND LOGISTICS

This phase of the experimental program involves the development of efficient operational and logistics procedures and equipment. The space station is considered to be the laboratory with no special areas required. Normal storage areas within the station are used for stowage of equipment such as pressure suits, maneuvering units, etc.

3.2 SYSTEM INTEGRATION

This section discusses the integration of only those system requirements which have a major effect on the space station configuration.

3.2.1 CREW QUARTERS

The habitability requirements of a space station dictate to a great extent the overall size of the vehicle. Area requirements are shown in Table 3.5.

The private quarters will be large enough to provide one man with sleeping facilities, personal storage compartments, and enough free volume for relaxation. Noise, vibration or any other annoyance factors that may filter through to the private quarters must be considered in the overall system integration.

For convenience, the private quarters should have a hygienic compartment located nearby. The hygienic compartment will consist of toilet and body cleaning (shower-type) facilities. A hygienic compartment containing only a toilet and hand washing facilities will be located near the wardroom.

The wardroom, or kitchen and dining area, may be adjacent to the gymnasium so that they can be made into one large recreation room. The wardroom can also be used as a recreation room and will be analogous to the kitchen-den in a modern home.

The sick bay may be used in the biomedical experiment program as well as serving its primary purpose. This compartment should be located near one or two private compartments in order to utilize the private compartments as "hospital" rooms. Increased space for treatment might be provided by having a folding "wall" between the sick bay and an adjacent private compartment.

Furnishings and decor are envisioned to be highly efficient and very attractive.

3.2.2 ELECTRICAL POWER SYSTEM

Integration of a solar cell electrical power system into a space station requires consideration of storage during the

TABLE 3.5

HABITABILITY AREA REQUIREMENTS

Crew Size	9	12	24
	Floor Area, Ft^2		
Sleeping Quarters	315	420	840
Wardroom	125	165	330
Food Preparation	16	16	36
Hygiene	28	28	56
Sick Bay	108	108	135
Gymnasium	60	60	90
Command Station	32	32	48
Total	684	829	1535

launch phase, deployment to the operational position, and operation of the system. Two concepts, folding panels and rolling sheets (window shades), have been considered. Each concept has advantages and may integrate into a particular space station configuration better than the other. The basic orientation of the space station affects the position and operation of a solar cell electrical power system as discussed previously.

Integration of a nuclear reactor electrical power system into a space station requires shielding and physical separation to protect the crew, experimental equipment, and subsystems from radioactivity. The physical separation and shielding weight may be integrated into an artificial gravity space station by using the nuclear electrical power system as a counterweight. Launch and prelaunch constraints are not considered to be severe if the nuclear reactor is not activated until the space station is in orbit.

3.2.3 VOLUMETRIC CONSIDERATIONS

The Mercury, Gemini, and Apollo spacecraft have been designed for minimum in-flight maintenance. However, long duration missions will require volumetric considerations for in-flight maintenance and repair. The working volume provided in past and present space vehicles is relatively small in comparison to the systems volume or the total volume.

The space Station configurations for long duration missions must account for both equipment integration volume and associated crew work space. For the purposes of this discussion, the following assumptions are made:

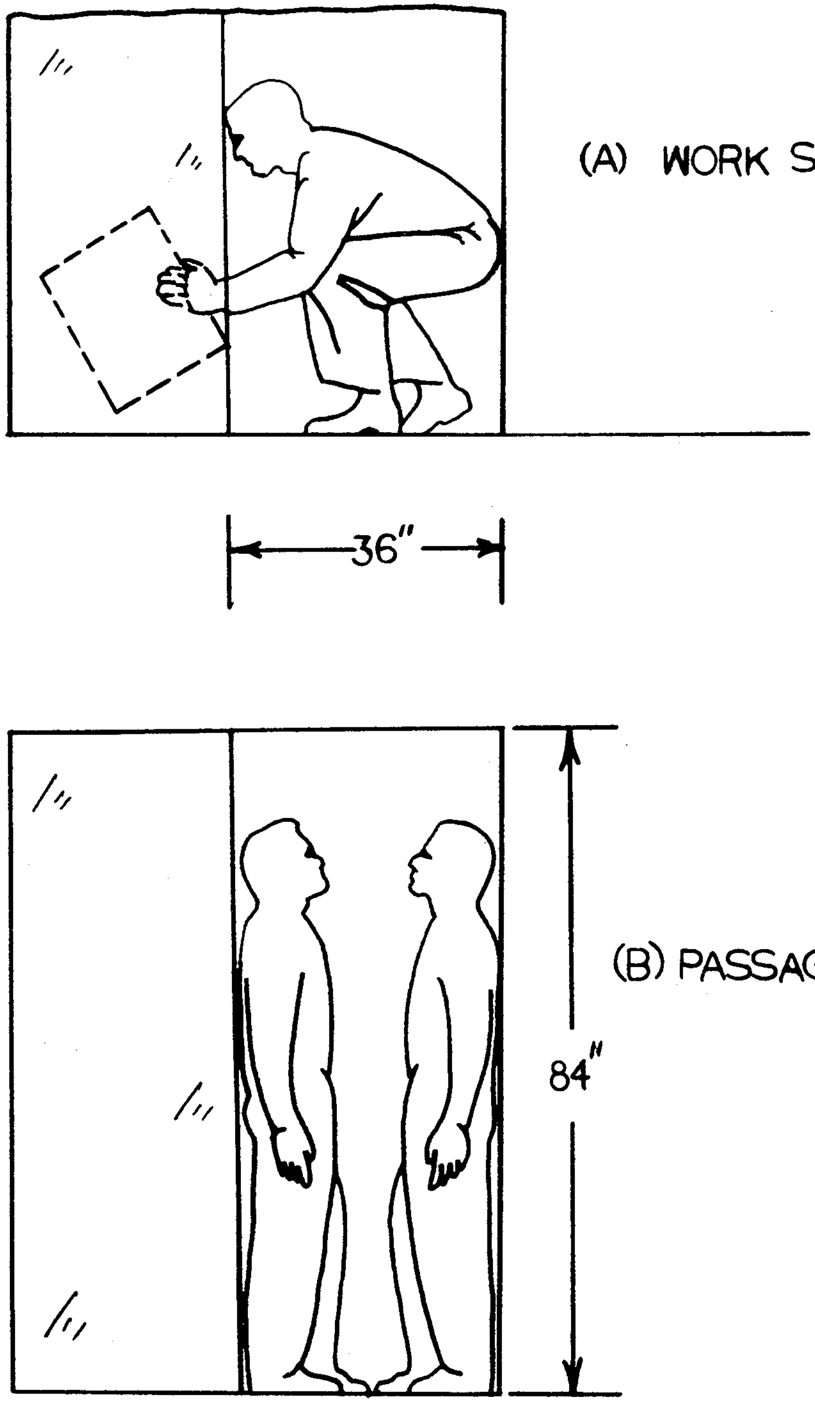
a. The equipment is integrated so that it can be either pulled out in the manner of a desk drawer or rotated out in the manner of a cabinet door.

b. The working spaces are also used for passageways. The passageways should be 36 inches wide. Refer to Figure 3.4 . It is assumed that passage of two or more crew members in the same passageway will occur with each man facing the other.

c. A 5 foot diameter hatch is provided in the center of each configuration to facilitate movement of equipment and personnel.

Two different diameter modules are shown in Figure 3.5 . The 33 foot module contains enough volume to provide equipment racks and passageways as follows. The outer equipment rack is 26 inches deep and the two inner racks are each 27 inches deep . The equipment volume V_e represents approximately 54 percent of

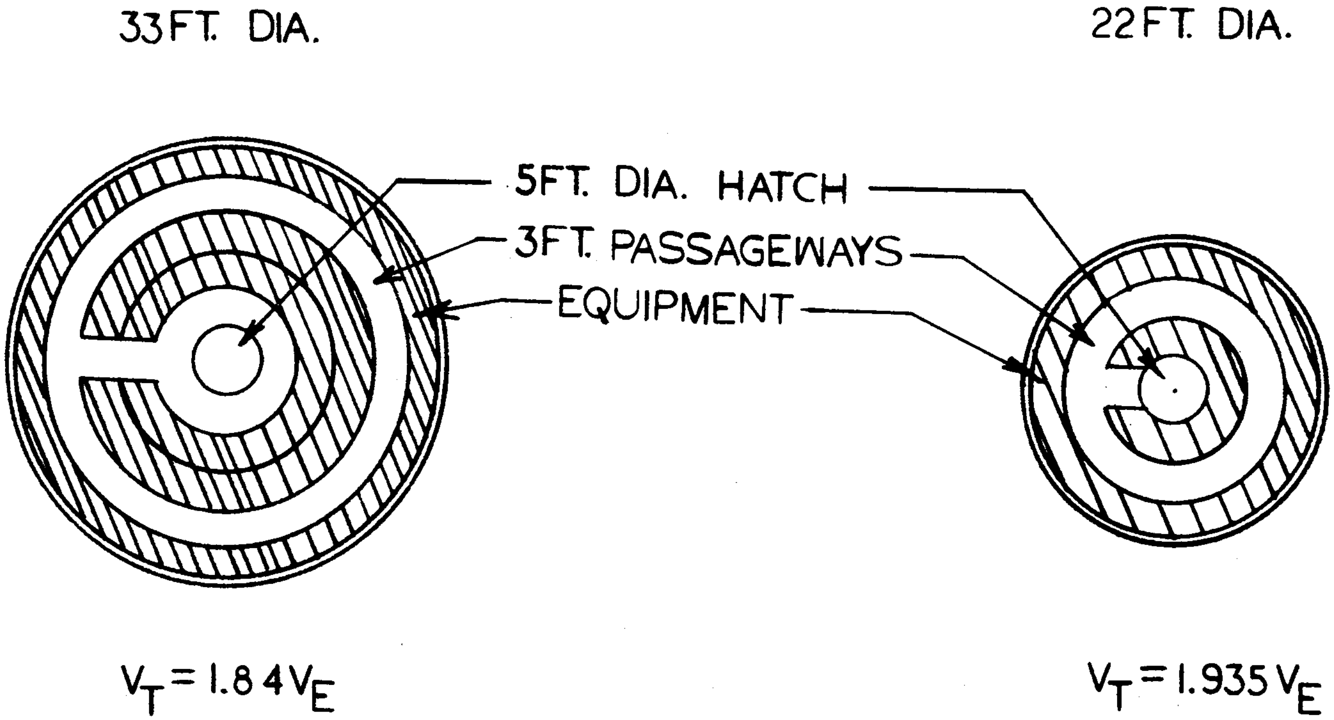
ANTHROPOMETRIC CONSIDERATIONS



(A) WORK SPACE

(B) PASSAGEWAY

FIGURE 3.4



INTEGRATION CONCEPTS

FIGURE 3.5

the total volume. The 22 foot module provides enough volume for two equipment racks, each 30 inches deep. The equipment volume for the 22 foot module represents approximately 52 percent of the total volume.

Considering the ratios of equipment volume to total volume for each of the modules shown in Figure 3.5 , it seems that:

$$V_{t33} = 1.8\ V_e \quad \text{or} \quad V_{t22} = 1.9\ V_e$$

Even though the two configurations shown in Figure 3.5 are simplified, the principles employed are applicable to a more detailed analysis of equipment volume requirements as they are related to the total volume. This simplified example indicates that the total volumetric requirements are approximately twice the equipment volume.

This is illustrated in Figure 3.6 by the "2 V_e" line, which may be considered minimum total volume required. The shaded area represents uncertainties which result from non-optimum design. The "3 V_e" line was used as nominal for this study.

3.2.4 LOGISTICS

Crew transfer and resupply requirements dictate that the station provide for docking with a logistics spacecraft or an Apollo Command Module.

Because of the type of cargo to be transferred from the logistics spacecraft's cargo module to the station, a pressurized interface tunnel of approximately 5 feet diameter is required. For transfer of fluids and gases, lines may be connected between the space station and logistics spacecraft.

The docking interface should be located near the station's mass center to minimize angular impulses applied during docking. Because of the differences in spacecraft configurations and interface requirements with existing spacecraft, a new docking mechanism design is required. Figure 3.7 illustrates a concept identified as a Universal Ring Docking Mechanism. The mechanism and docking tunnel interface on each vehicle is identical, so that a logistics spacecraft is capable of docking with another logistics spacecraft as well as with the station. The mechanism is external to the transfer tunnel, so it need not be removed to allow crew or cargo transfer. The particular arrangement of the ring and shock absorbers has been thoroughly analyzed dynamically in a prior study of docking mechanisms by the Advanced Spacecraft Technology Division of MSC. It is adaptable to a wide range of configurations and interface diameters.

The standard Lunar Module docking mechanism can be incorporated either at an adjacent docking port or at the primary docking

EQUIPMENT INSTALLATION VOLUME

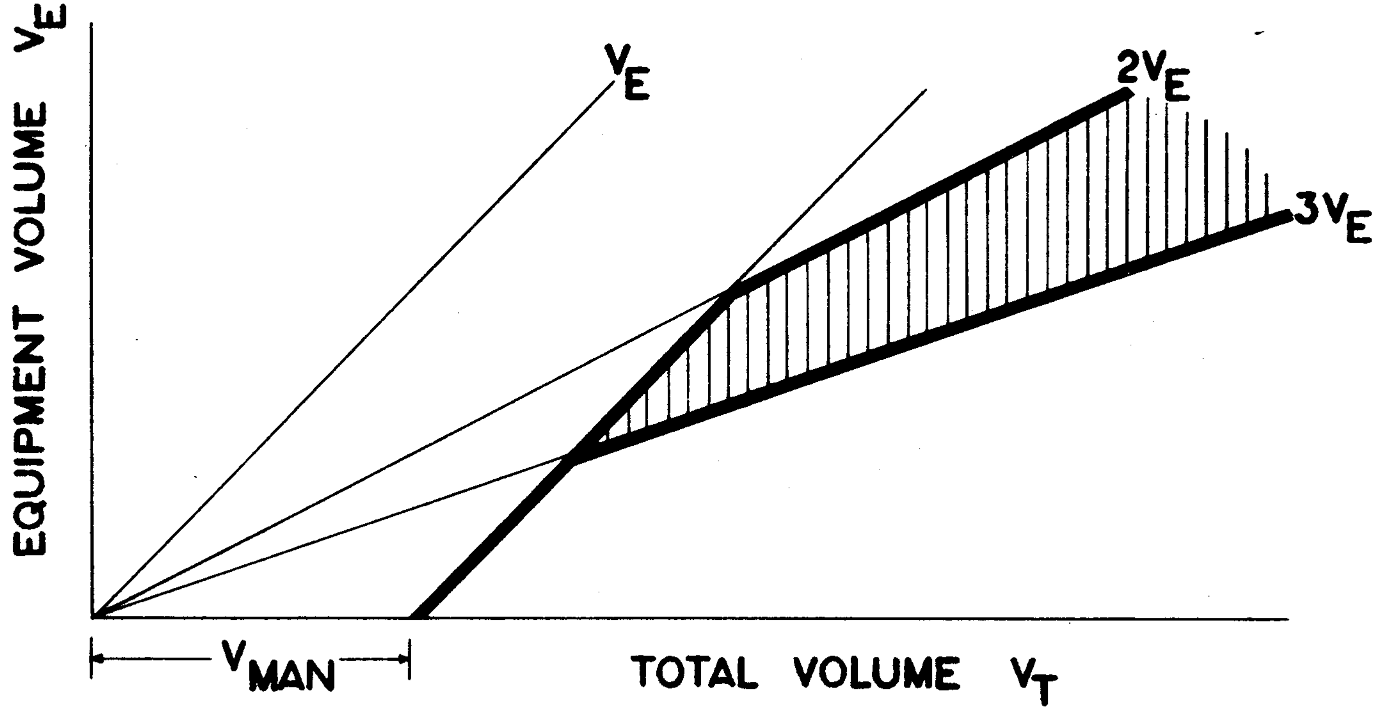
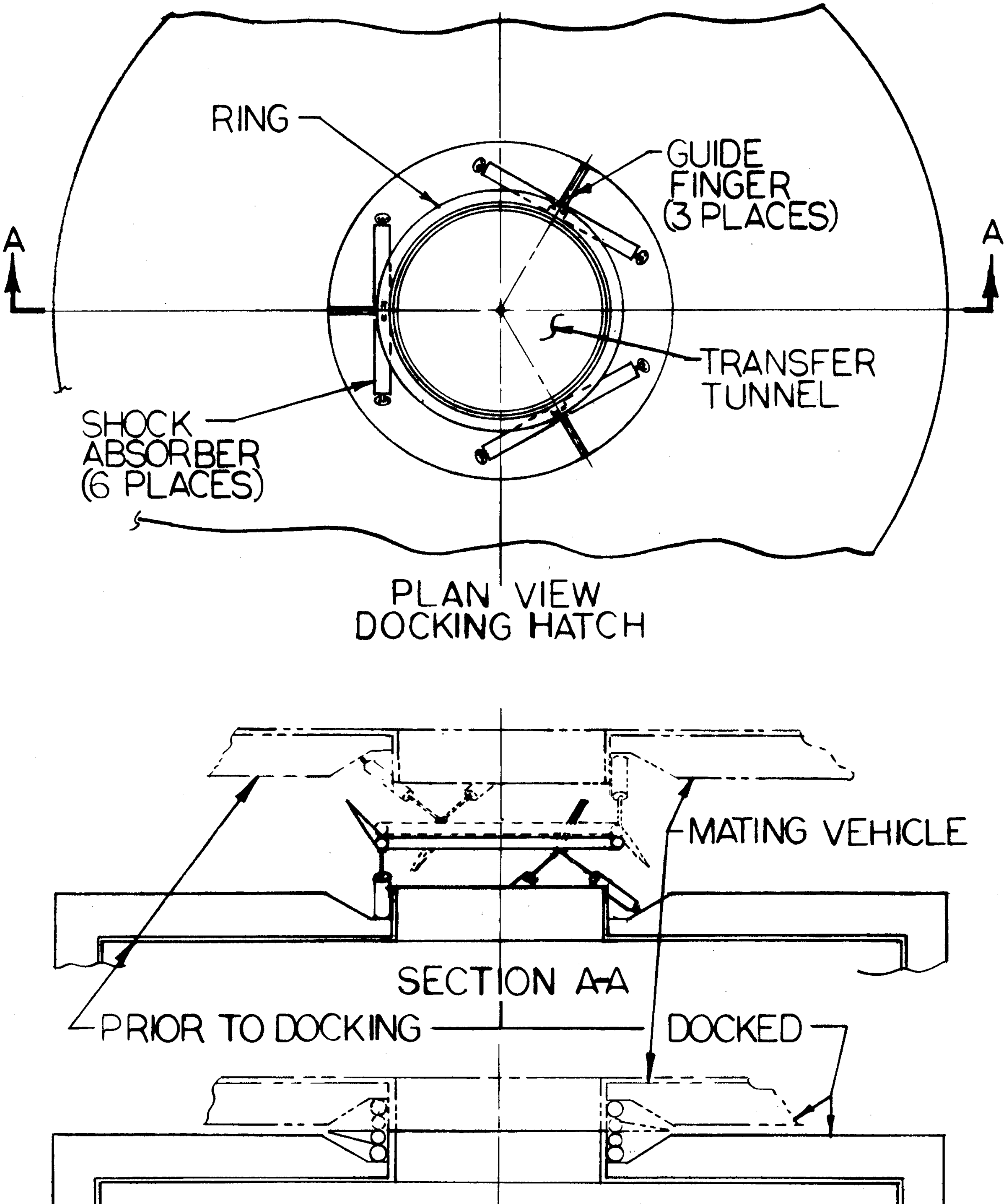


FIGURE 3.6



UNIVERSAL RING DOCKING MECHANISM

FIGURE 3.7

port using an adapter designed to be manually installed concentrically with the primary docking tunnel. This will allow docking with an Apollo or Apollo Applications Program spacecraft.

For the artificial gravity station the docking interface must be located on the non-rotating hub and should be concentric with the spin axis of the station.

3.2.5 ARTIFICIAL GRAVITY DESIGN REQUIREMENTS

Through analysis of the mission and experiment requirements, certain basic configuration design requirements have been established for the artificial gravity station. Those having a major impact on the station general arrangement are summarized as follows:

a. The nominal spin radius must be 75 feet or greater.

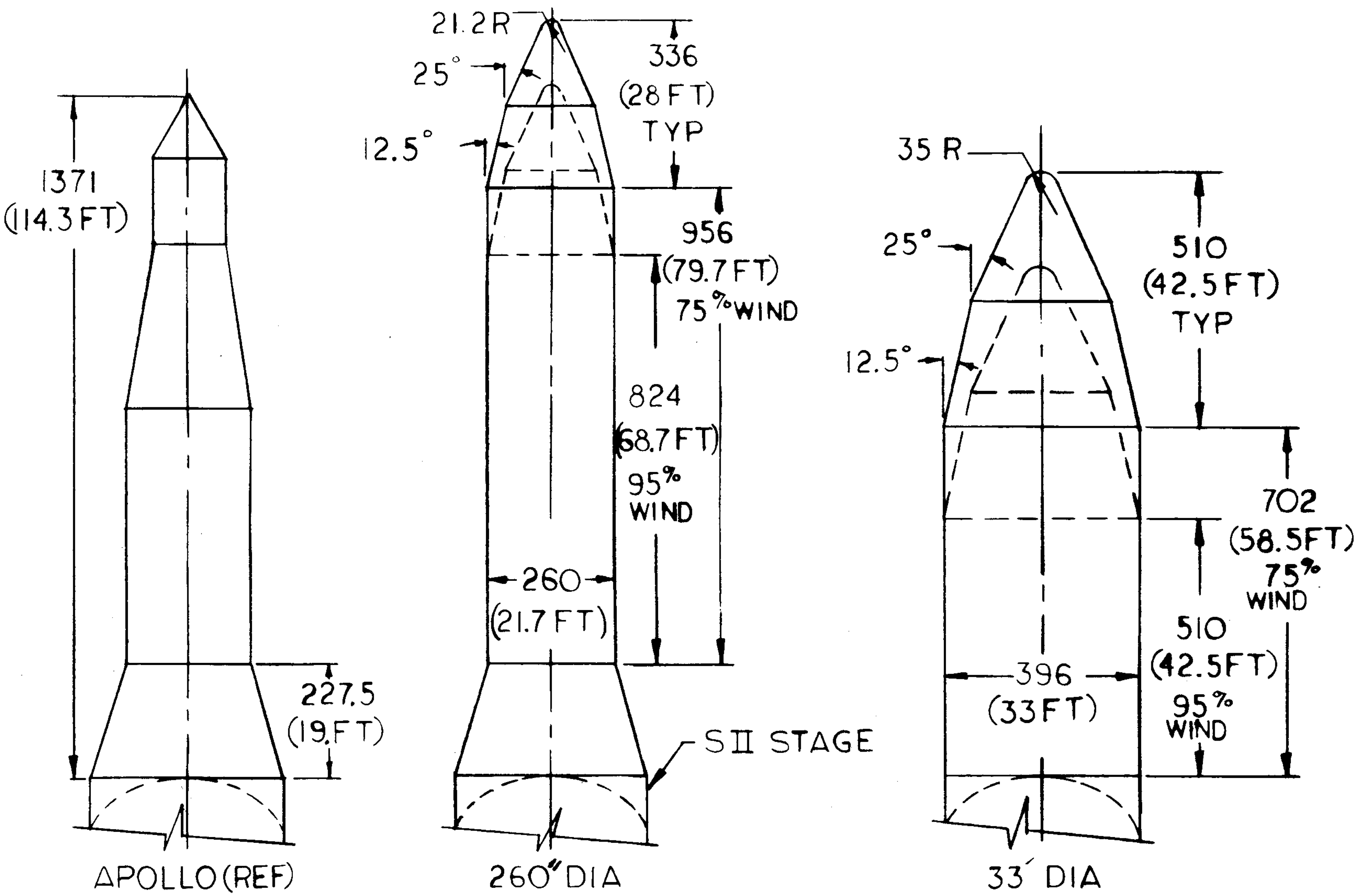
b. The spin axis must be a principal axis of maximum inertia by a significant margin.

c. The configuration must provide a non-rotating or zero-gravity volume of adequate size, arrangement, and stability to accommodate experiments and logistics requirements.

d. The station must be capable of being launched on a single two-stage Saturn V vehicle.

Because of launch vehicle payload envelope constraints as indicated in Figure 3.8, the artificial gravity station must be deployed into its orbital configuration after insertion into orbit. Also, to achieve the desired spin radius, deployment is likely to require telescoping arms (though not necessarily) to provide the proper separation of modules about the desired spin axis.

The non-rotating volume (hub) of the station must be located at the station's spin axis and connected to the rotating portion of the station through bearings concentric with the spin axis. A drive mechanism between the rotating and non-rotating portions of the station is required to overcome bearing friction. It must have a controlled variable speed capability to compensate for variations in the angular velocity of the rotating portion of the station. To provide the proper stability of the zero gravity portions of the station, the mass of the rotating portion must be balanced to the required accuracy about the desired spin axis. To allow normal crew activities and mass transfer within the rotating station, an active mass balance control is necessary. Mass balance may be accomplished by mechanically



ALLOWABLE PAYLOAD ENVELOPES
2-STAGE SATURN V LAUNCH VEHICLE

FIGURE 3.8

changing the spin radius of the rotating module(s), repositioning the hub to maintain alignment with the spin axis, or transferring a compensating mass, such as a fluid, between the hub and extremities of the station.

As discussed in Section 2.0, if the mass moment of inertia of the station about its spin axis is sufficiently greater than the inertia about any other axis, the station is inherently stable. That is, its spin axis tends to remain fixed in inertial space, and normal activities on board the station will have negligible effect on the stability of the station. To utilize the gyroscopic effect to the maximum advantage, the configuration must not only have the correct mass distribution but must be relatively rigid structurally. Therefore, configurations based on cable interconnected modules have not been considered.

It is desirable to integrate the entire artificial gravity area into a single module. This eliminates duplication of facilities and equipment as would be the case for a station that had three separate gravity modules, for example.

PRELIMINARY TECHNICAL DATA

FOR EARTH ORBITING SPACE STATION

VOLUME IV

CONFIGURATIONS, INTEGRATION, AND WEIGHTS

SECTION 4.0

CONFIGURATIONS

ADVANCED SPACECRAFT TECHNOLOGY DIVISION

ENGINEERING AND DEVELOPMENT DIRECTORATE

MANNED SPACECRAFT CENTER

4.0 CONFIGURATIONS

Configuration concepts were developed for both zero gravity and artificial gravity space stations. The general arrangement of each concept was evolved in response to system and experiment requirements which were established during the space station study. The concepts presented in this report are not necessarily optimum. However, it is believed that they are representative of the basic size and arrangement necessary for the experimental mission as it is presently defined.

It is important to point out that all concepts possess several or all of the characteristics described as follows:

a. The basic arrangement is modular, in that separate compartments are provided for most major functional categories (living, experimental disciplines, and supporting systems).

b. Each compartment has a 7 foot head height. The orientation of all floors is normal for the ground crew when the station is on the launch pad.

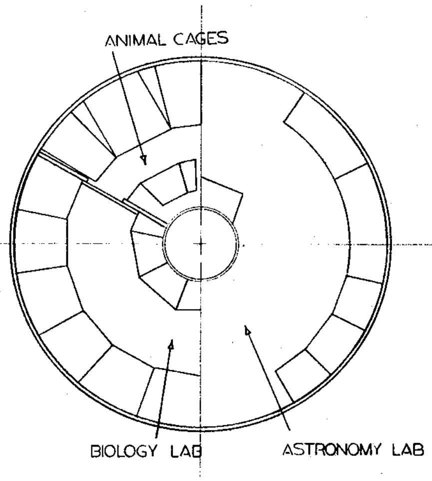
c. Each compartment is structurally designed to hold internal pressure with adjacent compartments depressurized.

d. Generally, all internal pressure bulkheads are coincidental with the compartment floors or ceilings and are flat. Tension ties between floors reduce the weight penalty of using flat pressure bulkheads instead of domed bulkheads. The tension ties may be integrated with equipment support structure.

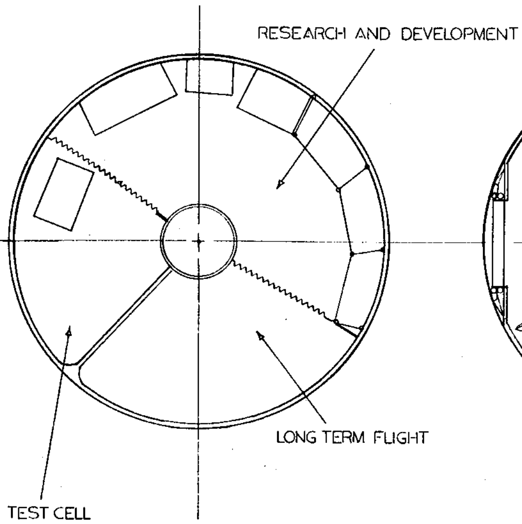
4.1 ZERO GRAVITY CONFIGURATIONS

Figure 4.1 shows the general arrangement of a 9-man, 260 inch basic diameter, zero gravity space station designed for launch on a two-stage Saturn V vehicle. The configuration is arranged to allow the station's in-orbit attitude to be such that its solar panels are maintained normal to the sun's rays, or its longitudinal axis is maintained normal to the orbit plane.

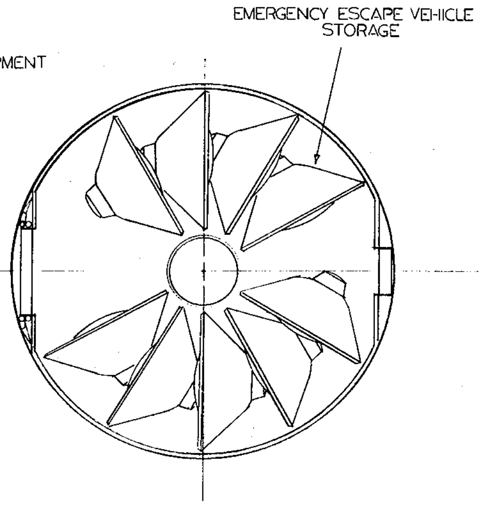
The only difference between the two orientations is that the solar panels may be fixed to the sun-oriented station but require one axis of freedom if the station is oriented normal to the orbit plane. In either case, the attitude would be held inertially fixed during astronomical sensor operation (approximately half an orbit). Earth sensor operation could proceed simultaneously with astronomy or other experiments because of the turret installation of earth sensors which is described in



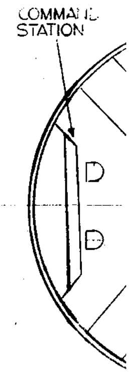
SECTION A-A

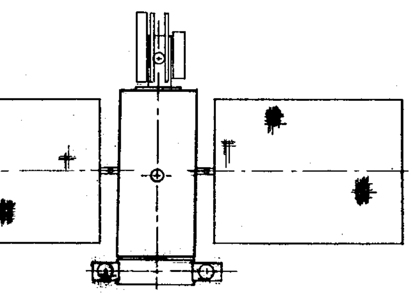


SECTION B-B

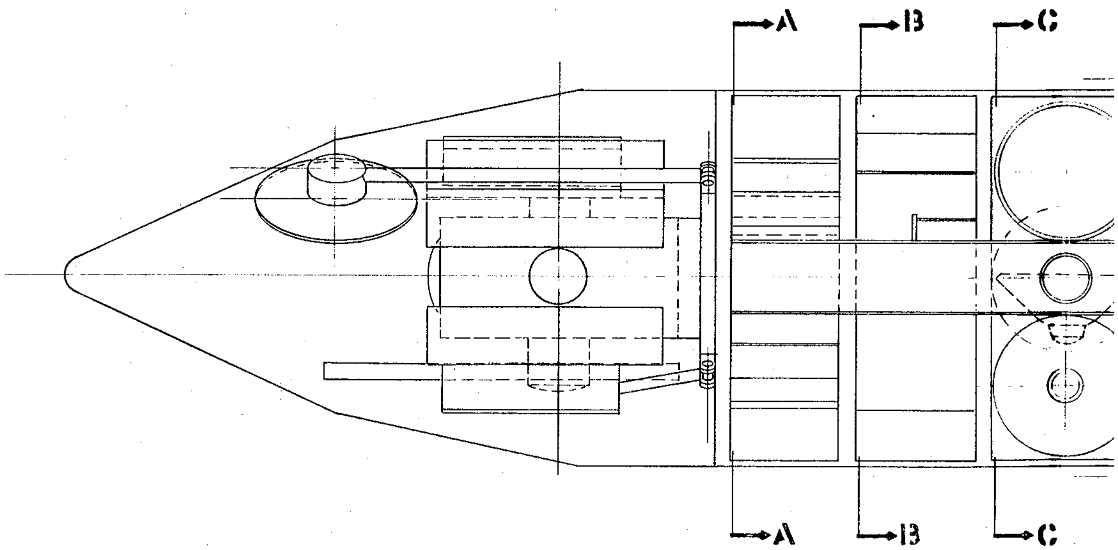


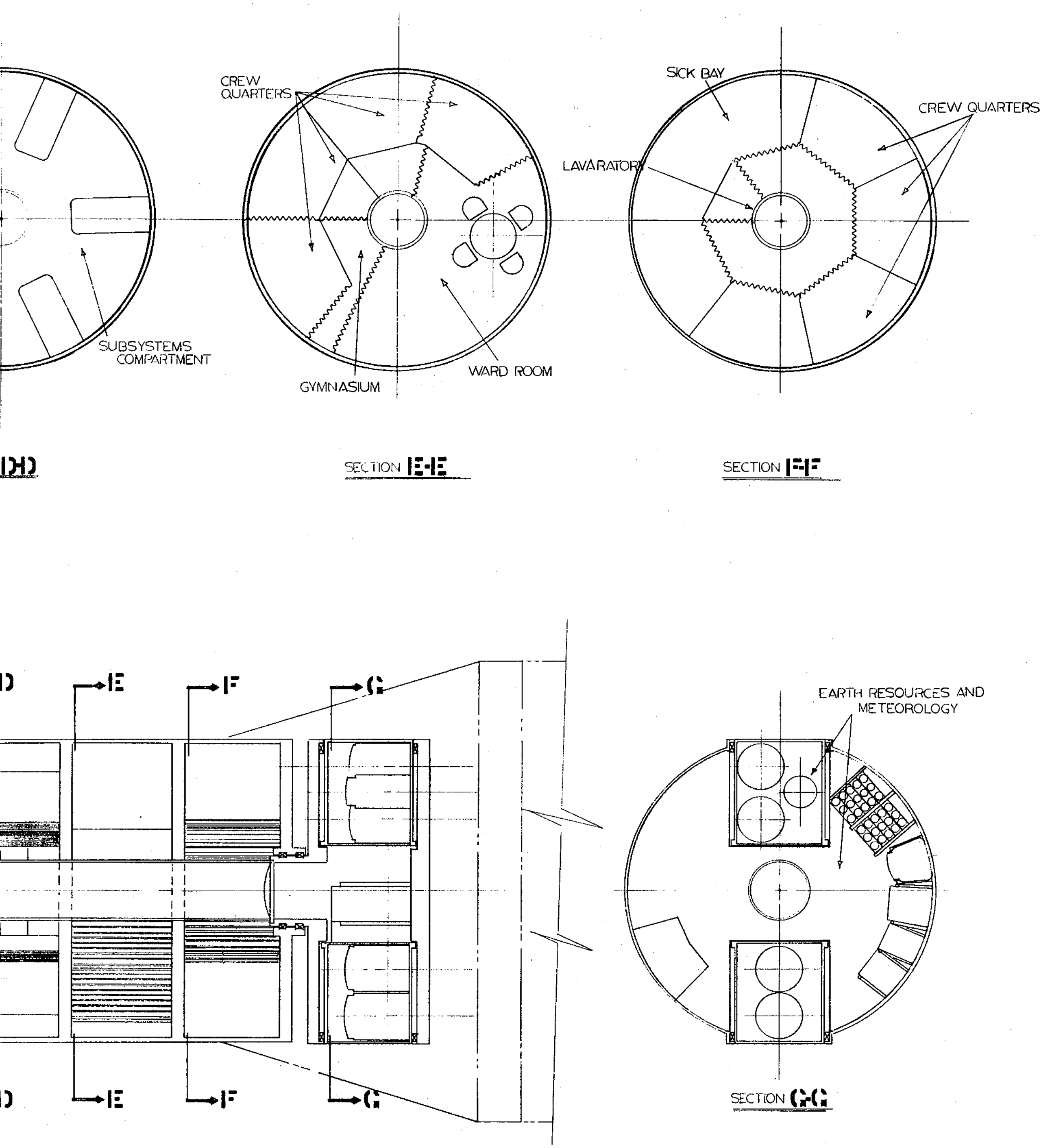
SECTION C-C





ORBITAL CONFIG





GENERAL ARRANGEMENT
ZERO 'G', 260 IN. DIA., 9 MAN
SPACE STATION

FIGURE 4.1

Section 3.1.2. Astronomical sensor installation is as described in Section 3.1.1.

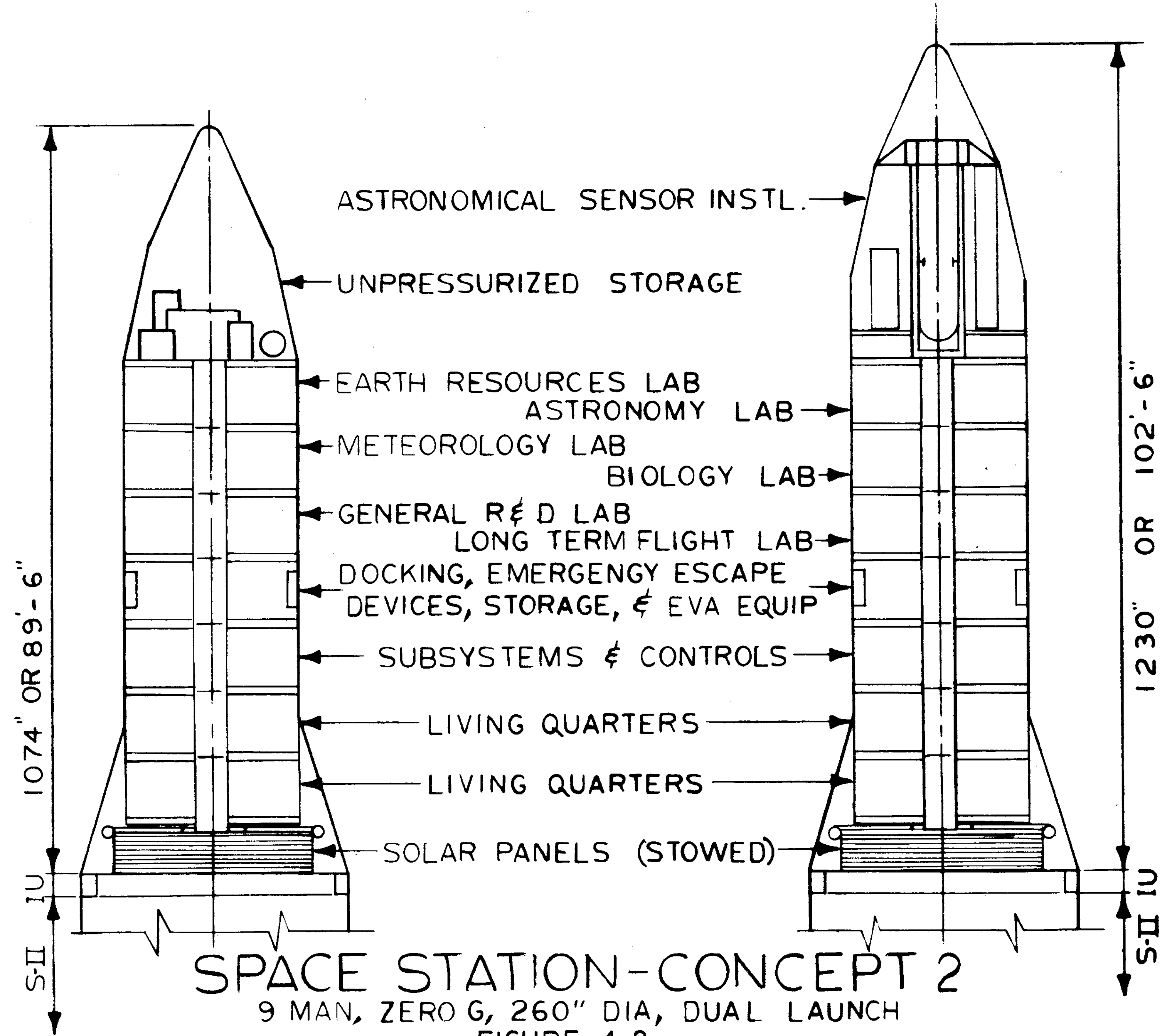
Three compartments of the station are devoted to the experiments. A centrally located compartment provides space for emergency escape devices, consumables (including non-pressurized storage areas where desirable), other cargo, and docking. This compartment may exceed the basic 7 foot height. The remaining compartments contain the crew quarters and systems hardware.

Figure 4.2 shows the schematic of a concept which utilized two 9-man stations. Each contains basically the same volume, support systems, crew quarters, and structure, and is designed for approximately one-half of the experiment disciplines. Theoretically, this allows more optimum operational procedure and simplifies the overall requirements by dividing the experiments into two groups. Each group contains those experiments which are most compatible from the standpoint of station design, orientation in orbit, and operation. It also provides essentially double the available experiment man-hours, weight, and volume as compared to a single station.

4.2 ARTIFICIAL GRAVITY CONFIGURATIONS

Figure 4.3 shows the general arrangement (launch configuration) of a 9-man, 260 inch basic diameter, artificial gravity space station designed for launch on a two-stage Saturn V vehicle. Figure 4.4 shows the station deployment sequence and the in-orbit configuration. This is a typical "I" configuration artificial gravity concept which utilizes the spent S-II stage as a counterweight for its single artificial gravity module. This station size and arrangement allows deployment to a spin radius in excess of 75 feet without telescoping the structural linkage between the S-II counterweight and the artificial gravity module. This is a significant mechanical feature which can inherently allow the deployed configuration to be structurally rigid with minimum weight and complexity. The rigidity of the linkage could also be radically changed during final design and development without affecting the design of the deployment mechanism. Also, as the hub is moved into its deployed position, it can be located at the precise mass center of the actual in-orbit rotating portion of the station. This provides an inherent trimming capability.

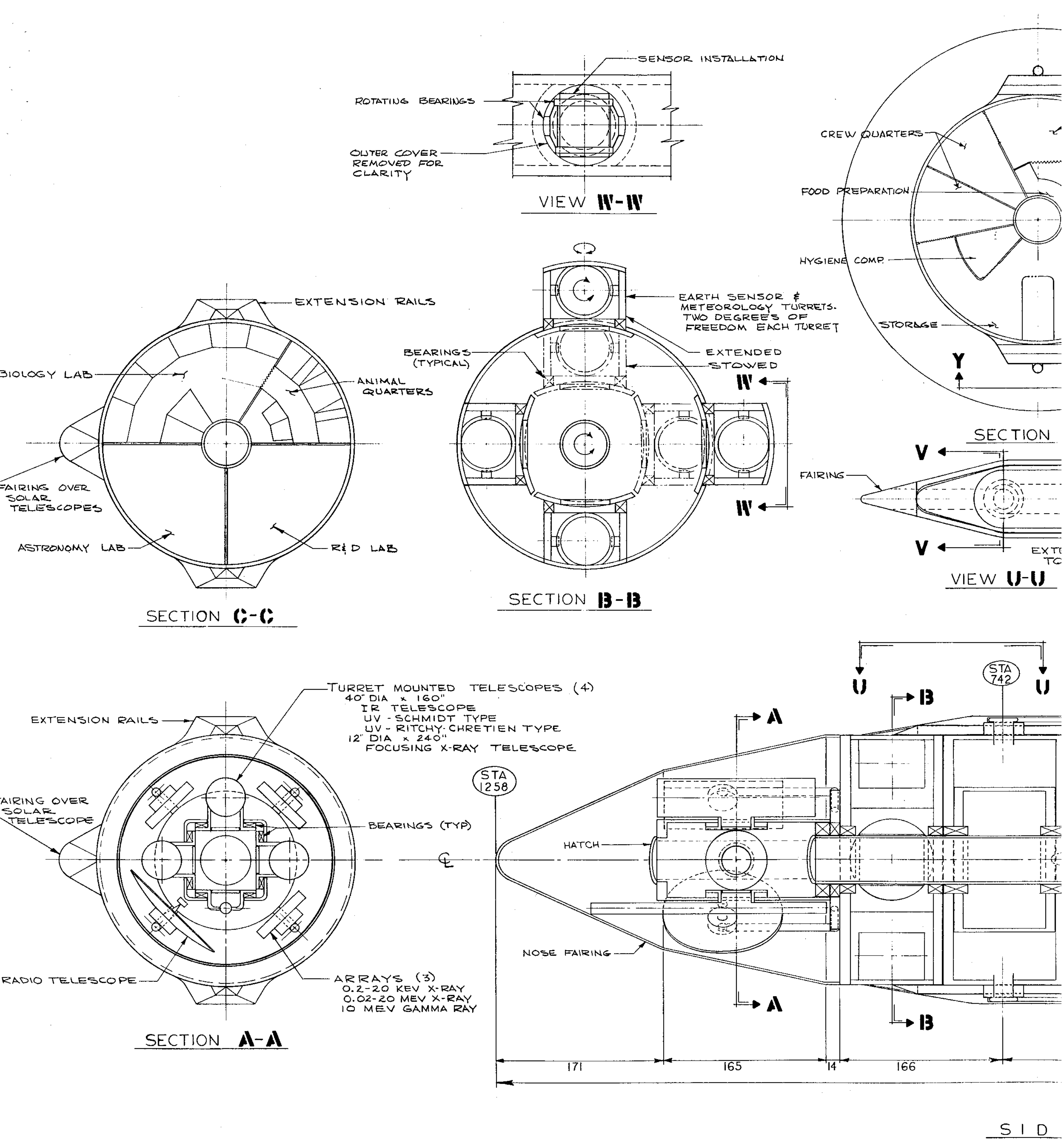
The station arrangement shown by the drawing includes living quarters for two (2) crewmen in the zero gravity hub. This is a biomedical experiment requirement. Separate support systems are provided for the zero gravity and artificial gravity modules to minimize the complexity of transferring system functions across the rotating interface. Laboratory volume is provided in



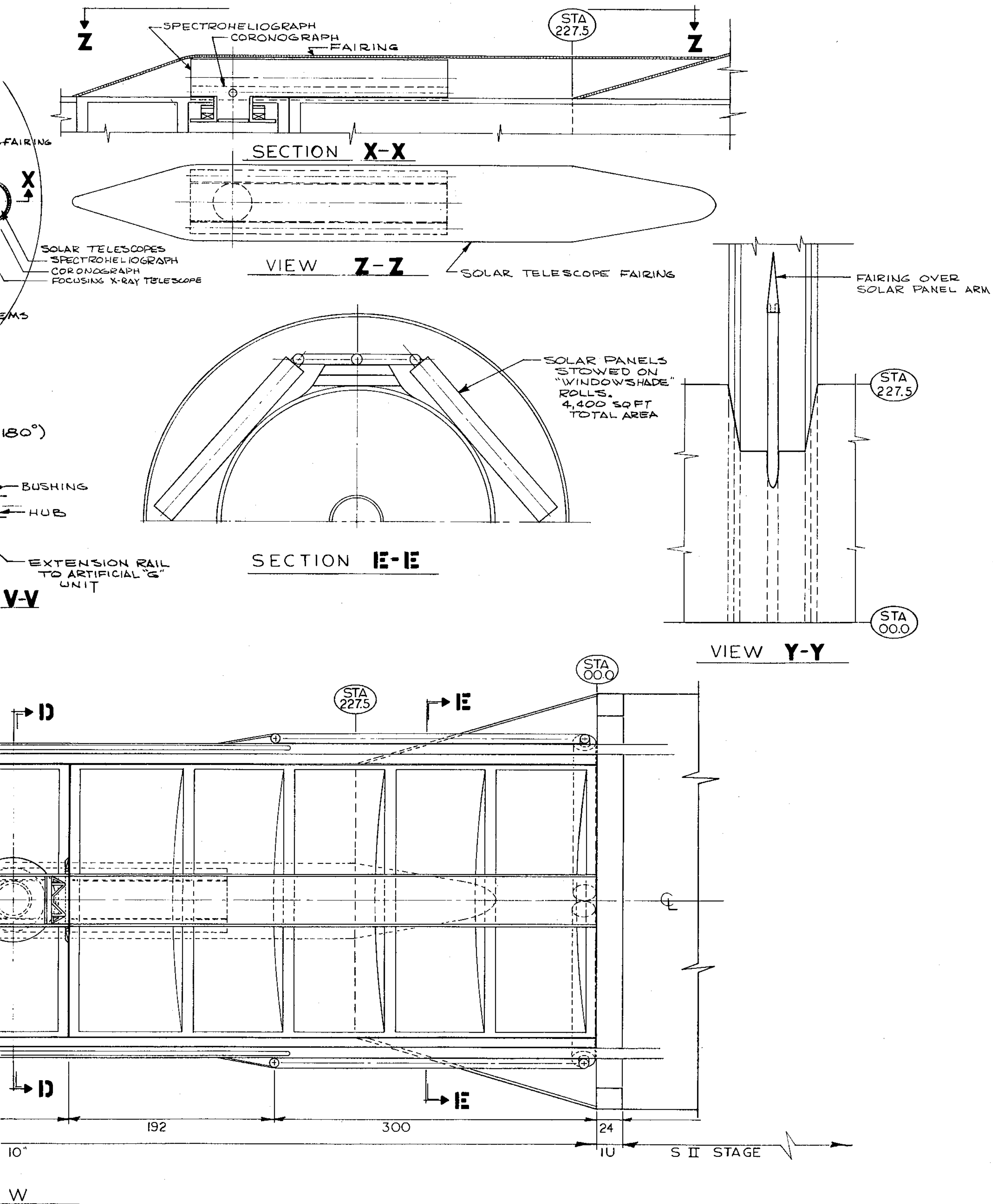
SPACE STATION-CONCEPT 2

9 MAN, ZERO G, 260" DIA, DUAL LAUNCH

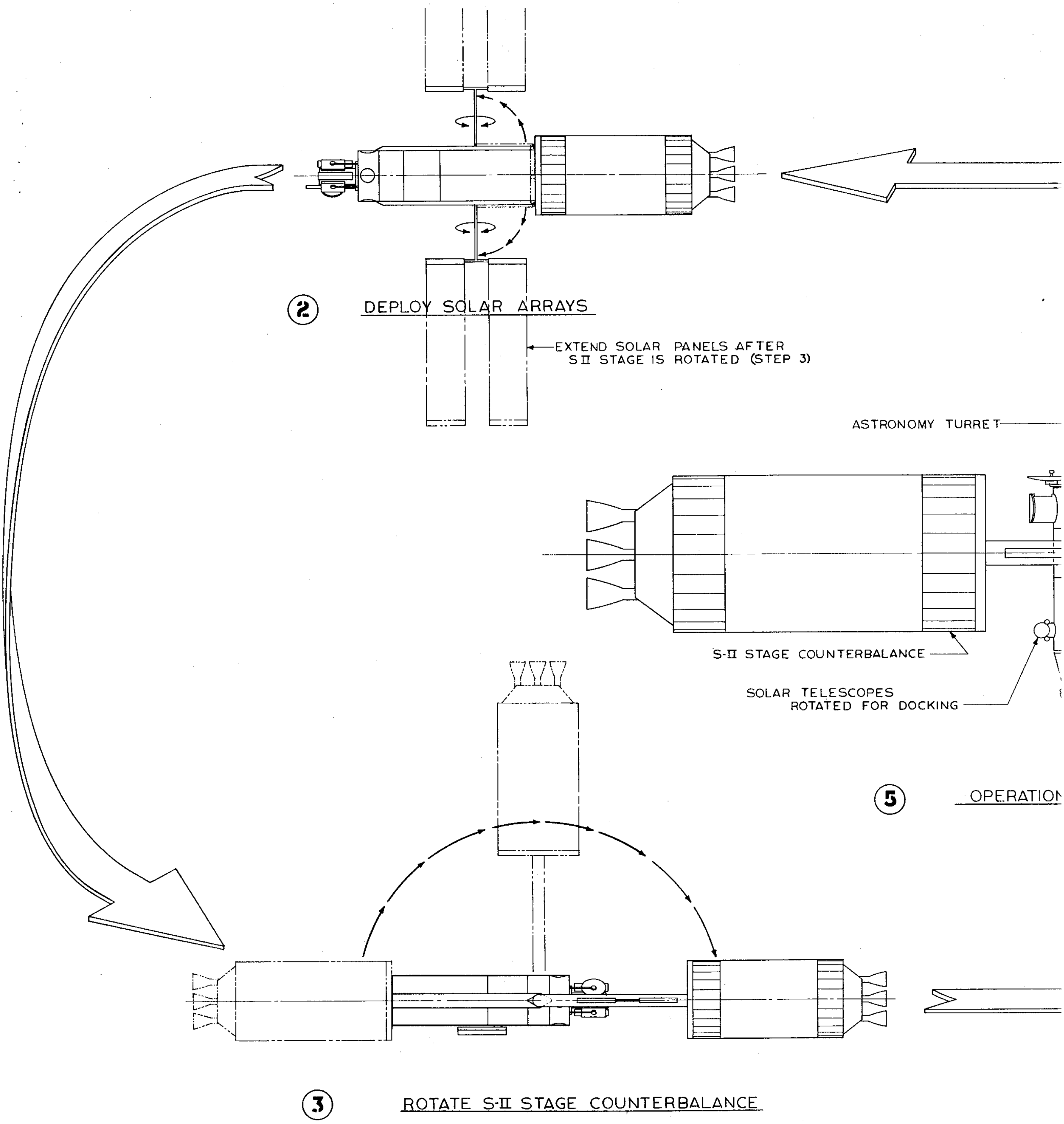
FIGURE 4.2

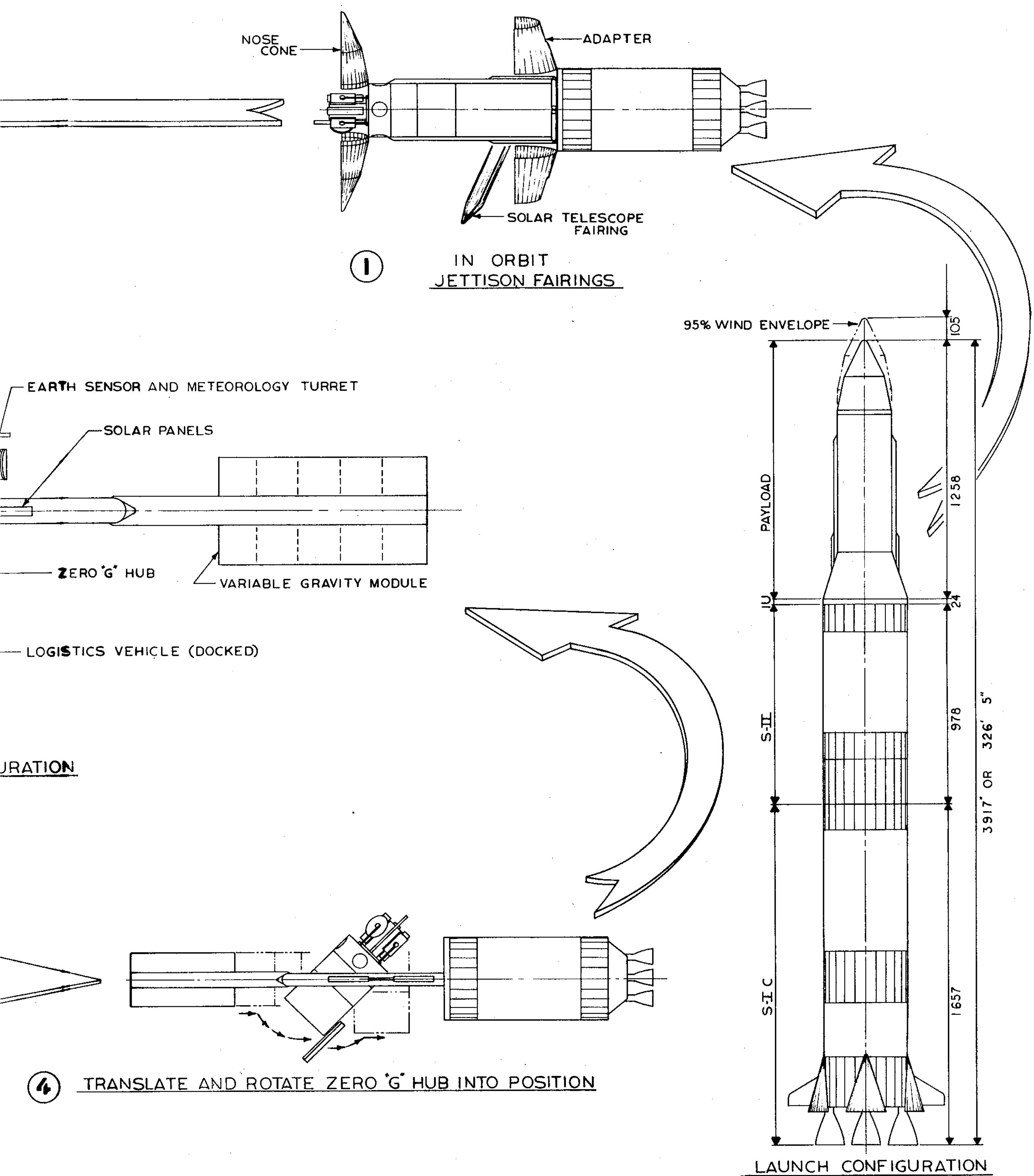


SENSOR INSTALLATION
ROTATING BEARINGS
OUTER COVER REMOVED FOR CLARITY
VIEW W-W
CREW QUARTERS
FOOD PREPARATION
HYGIENE COMP.
STORAGE
Y
SECTION
EXTENSION RAILS
BIOLOGY LAB
ANIMAL QUARTERS
FAIRING OVER SOLAR TELESCOPES
ASTRONOMY LAB
R&D LAB
SECTION C-C
EARTH SENSOR & METEOROLOGY TURRETS. TWO DEGREES OF FREEDOM EACH TURRET
BEARINGS (TYPICAL)
EXTENDED
STOWED
W
W
SECTION B-B
V
FAIRING
V
VIEW U-U
TURRET MOUNTED TELESCOPES (4)
40" DIA x 160"
IR TELESCOPE
UV - SCHMIDT TYPE
UV - RITCHY-CHRETIEN TYPE
12" DIA x 240"
FOCUSING X-RAY TELESCOPE
EXTENSION RAILS
FAIRING OVER SOLAR TELESCOPE
BEARINGS (TYP)
RADIO TELESCOPE
ARRAYS (3)
0.2-20 KEV X-RAY
0.02-20 MEV X-RAY
10 MEV GAMMA RAY
SECTION A-A
STA 1258
HATCH
NOSE FAIRING
U
U
STA 742
B
A
A
B
171
165
14
166
SID



GENERAL ARRANGEMENT
ARTIFICIAL 'G', 260' DIA., 9 MAN
SPACE STATION
FIGURE 4.3


2
DEPLOY SOLAR ARRAYS
EXTEND SOLAR PANELS AFTER S II STAGE IS ROTATED (STEP 3)
ASTRONOMY TURRET
S-II STAGE COUNTERBALANCE
SOLAR TELESCOPES ROTATED FOR DOCKING
5
OPERATION
3
ROTATE S-II STAGE COUNTERBALANCE



DEPLOYMENT SEQUENCE
ARTIFICIAL "G", 260" DIA., 9 MAN
SPACE STATION
FIGURE 4.4

the artificial gravity module for all experimental functions which can tolerate the artificial gravity field. These may include data reduction, instrument adjustment and maintenance, planning, etc.

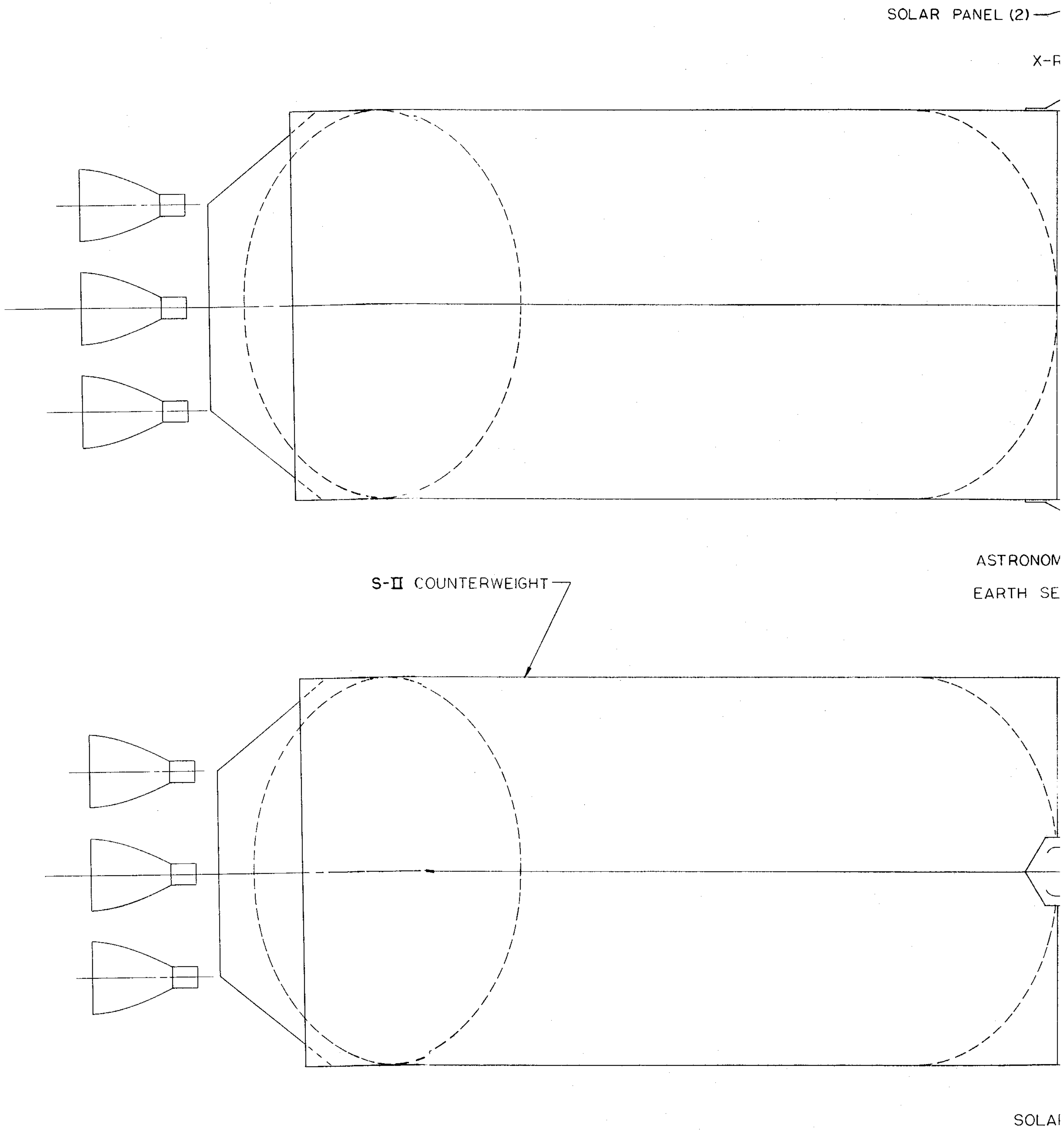
The station is designed to have its spin axis aligned toward the sun. This allows the solar panels to be fixed to the rotating structure of the hub. Unregulated direct current is transferred to the non-rotating hub through slip rings. With this arrangement, the solar panels add to the station stability by increasing the mass moment of inertia about the spin axis.

Figure 4.5 shows the general arrangement of a 396 inch diameter artificial gravity station. The basic concept is similar to the 260 inch diameter station previously described. Deployment utilizes telescoping tubes between the hub and artificial gravity module to provide both structural interconnections and access. As shown by the drawing, the S-II stage counterweight must move toward the hub during station deployment. To accomplish this, a truss linkage between the S-II stage and hub is retracted during deployment. All module interconnecting elements can be designed to be quite stiff in the deployed configuration.

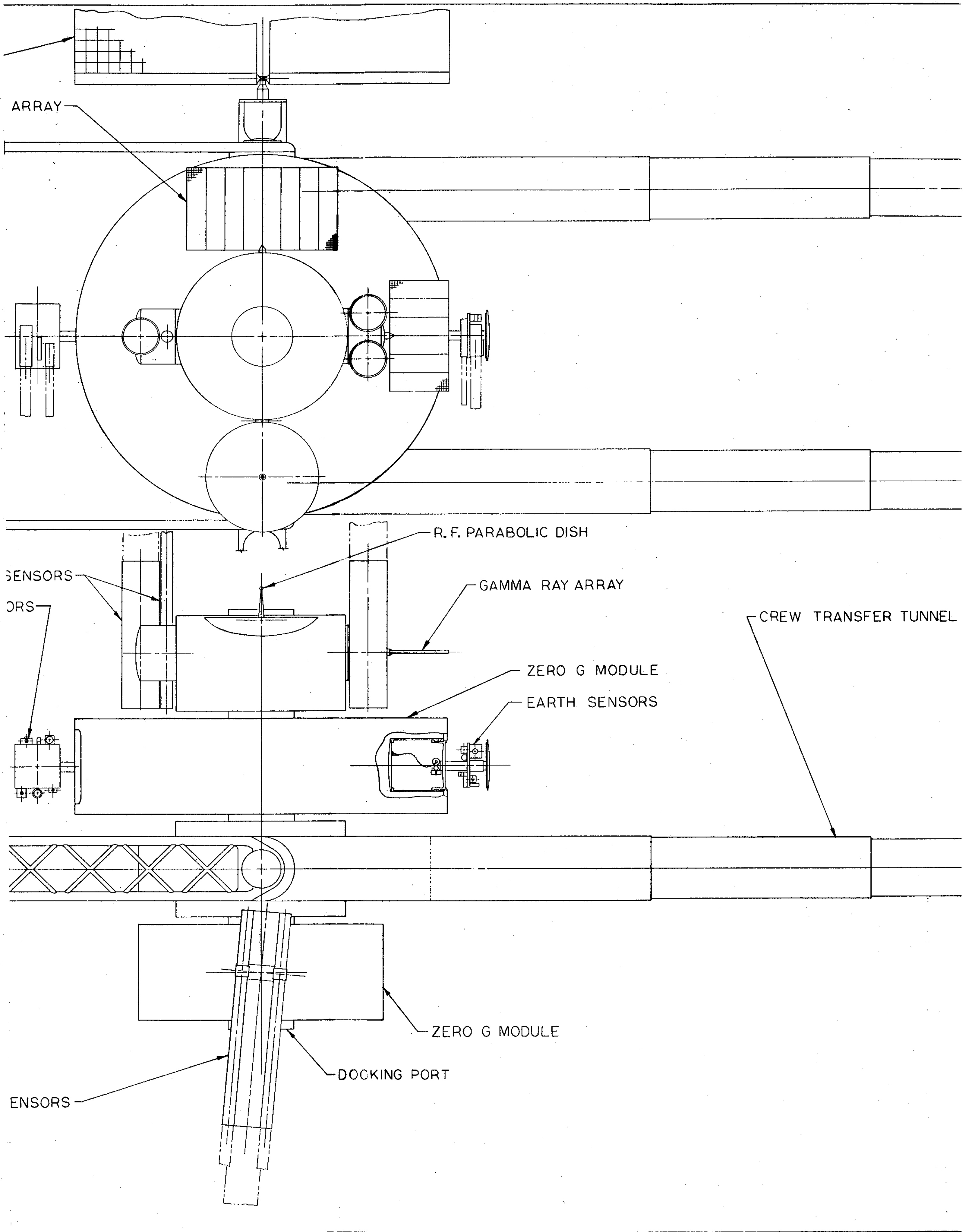
The 396 inch diameter configuration has approximately 20 percent more pressurized volume than the 260 inch configuration described previously. This is inherent because of the modular arrangement and does not reflect a volume requirement difference. The 396 inch configuration can readily accommodate additional crewmen and/or laboratory equipment.

A feature of this configuration, which is indicated on the drawing, is a gimbal mounted non-rotating hub. The gimbal essentially provides a "soft" interconnection between the rotating and non-rotating modules to provide a degree of isolation from angular wobble and the effects of mass imbalance of the rotating portion of the station. Springs and dampers in parallel would be utilized at the gimbal axes. Another basic effect of this hub mechanization would be passive wobble damping. The non-rotating hub has a large mass and inertia which would react station wobble through the spring-damper system to cause an inherent dissipation of wobble energy. The study results presented in Section 2.0 indicate that the gimbal mount may not be necessary; however, it is presented as a concept.

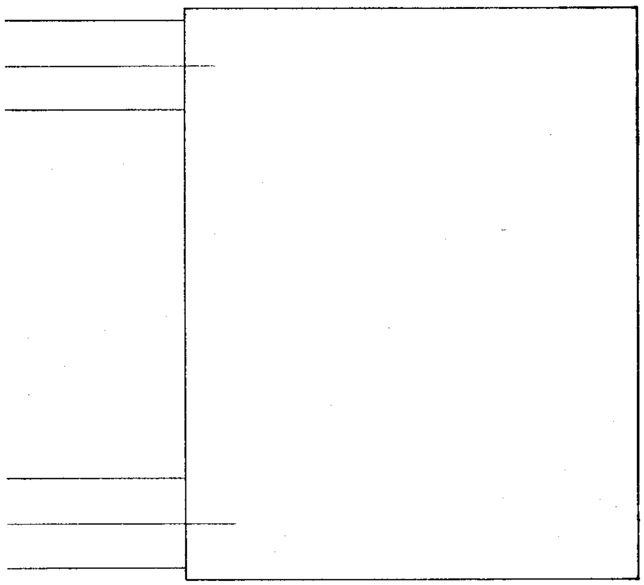
Figure 4.6 shows the general arrangement of a 24-man artificial gravity station. This configuration is basically identical to the 396 inch diameter station except for the additional laboratory, system, and living volume provided by additional zero gravity and artificial gravity compartments.

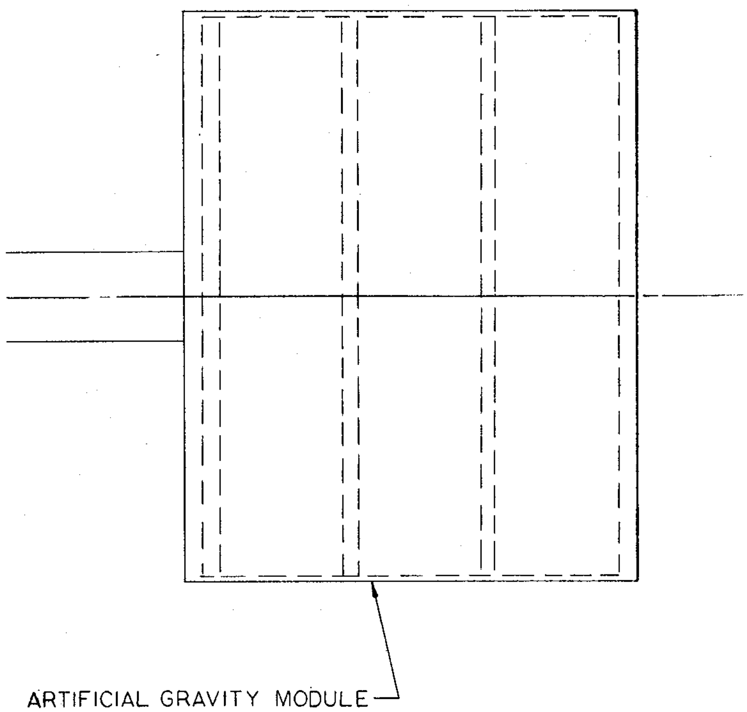


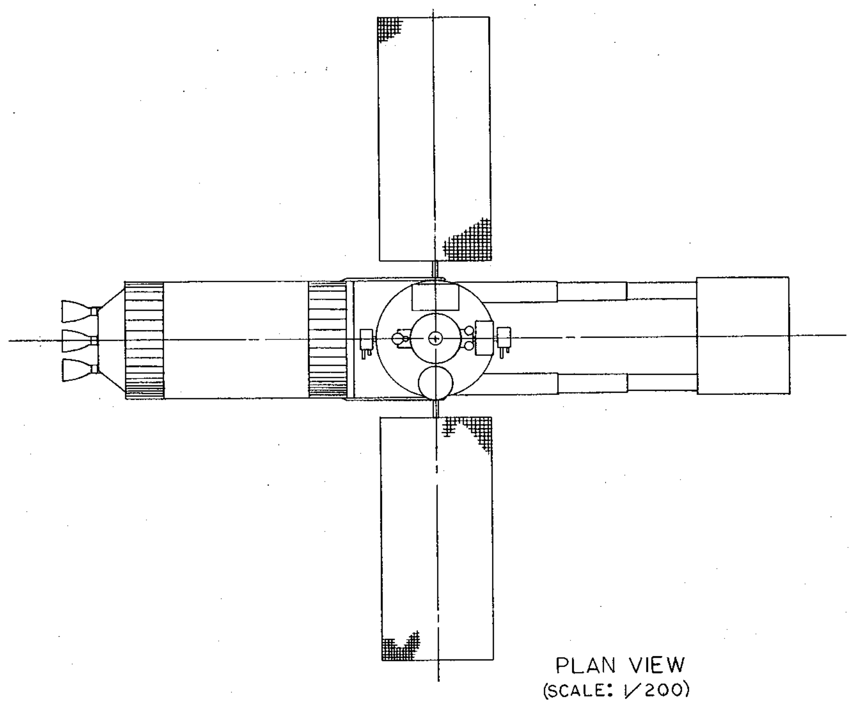
SOLAR PANEL (2)
X-R
ASTRONOM
EARTH SE
S-II COUNTERWEIGHT
SOLA



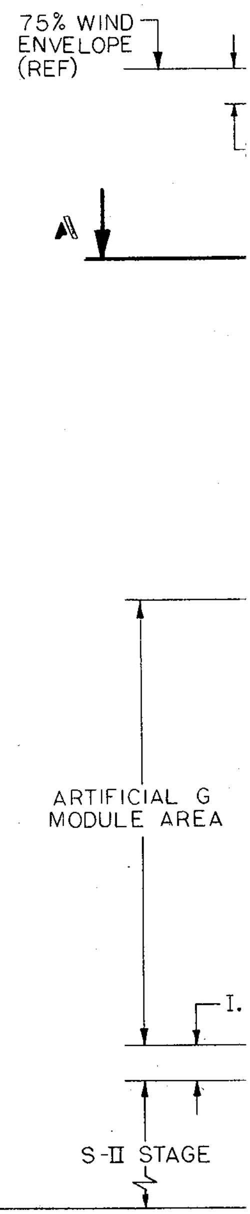
ARRAY
R.F. PARABOLIC DISH
SENSORS
ORS
GAMMA RAY ARRAY
CREW TRANSFER TUNNEL
ZERO G MODULE
EARTH SENSORS
ZERO G MODULE
DOCKING PORT
ENSORS

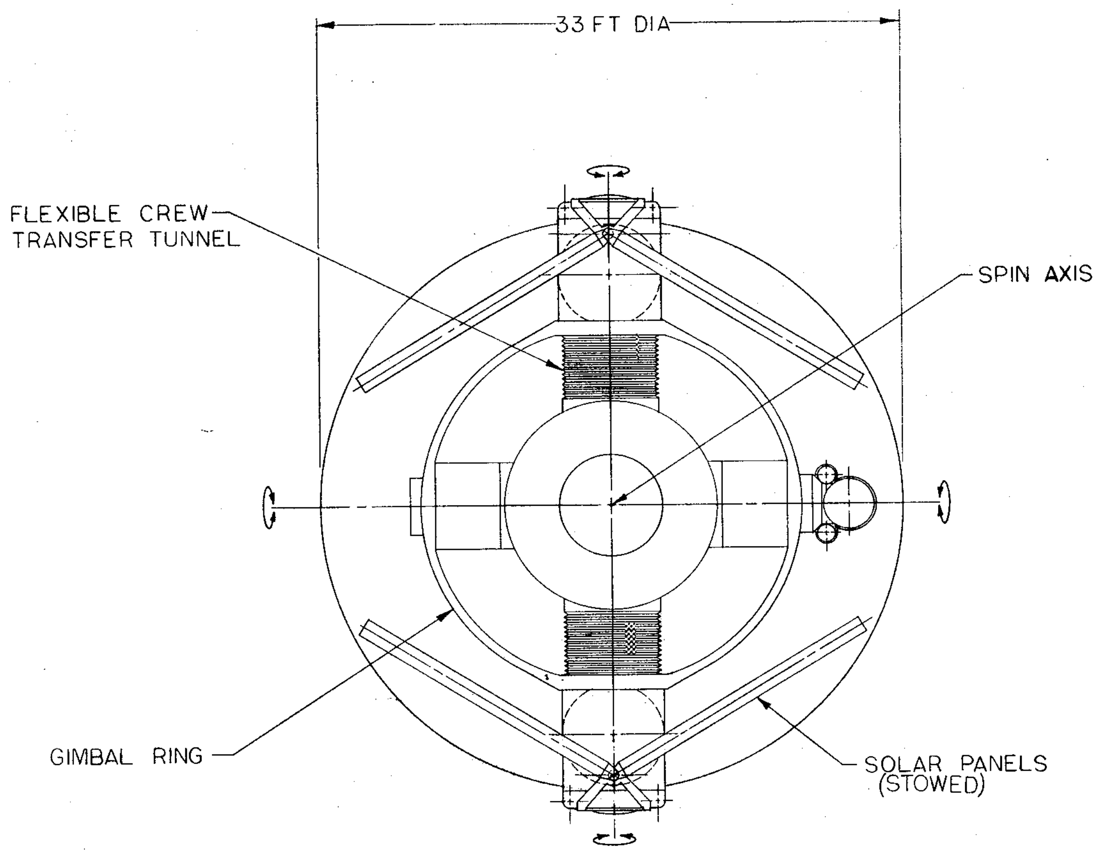



ARTIFICIAL GRAVITY MODULE

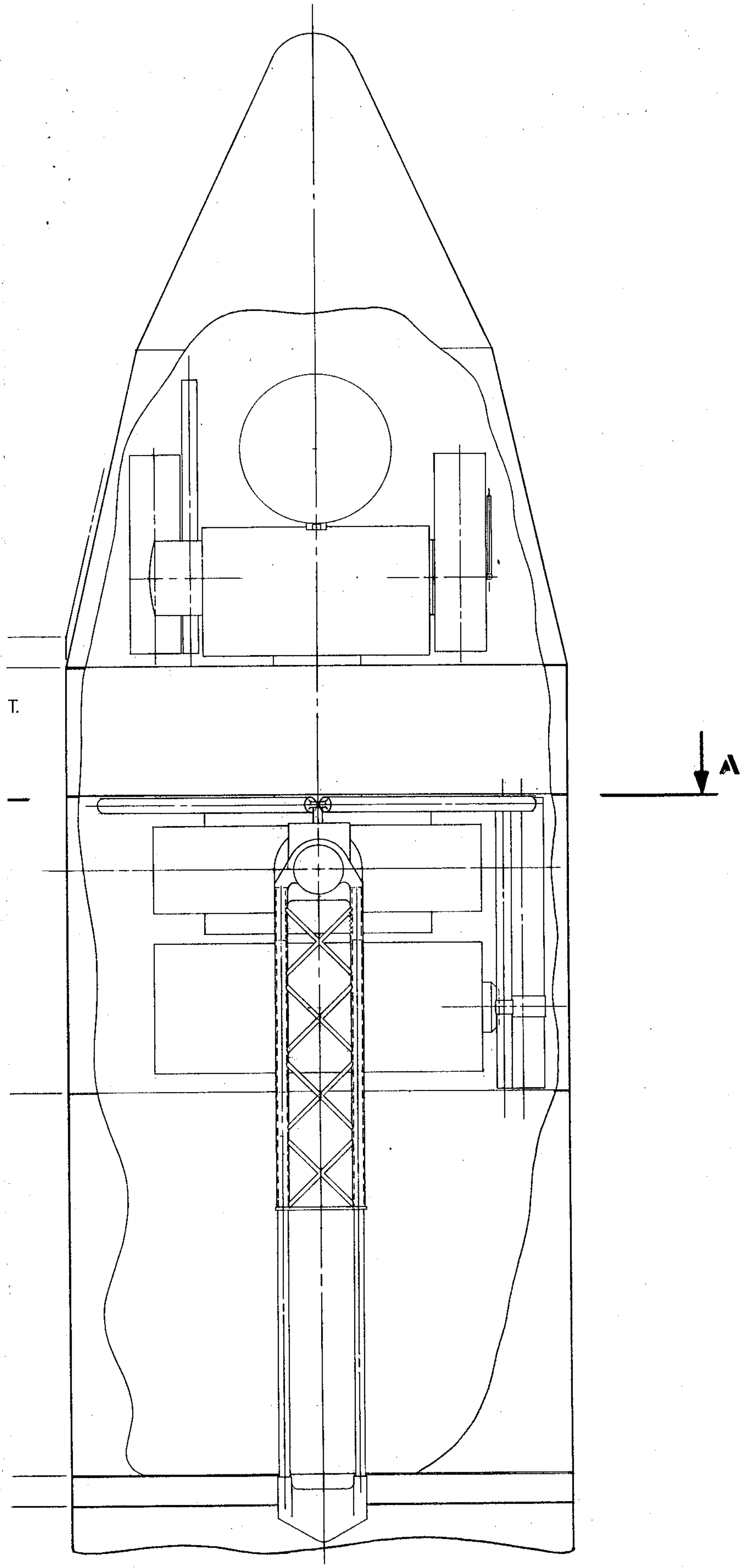


PLAN VIEW
(SCALE: 1/200)





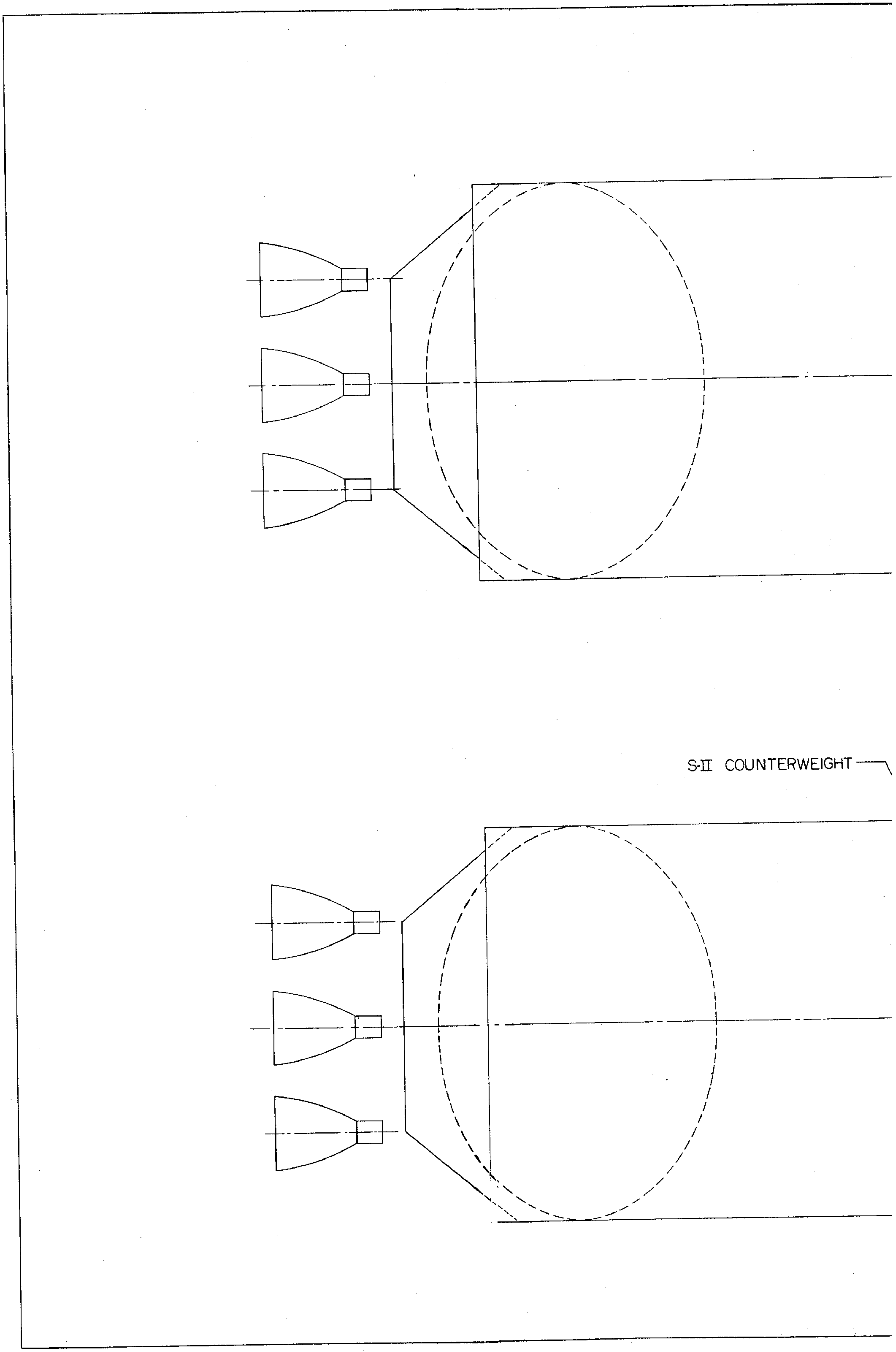
SECTION A—A

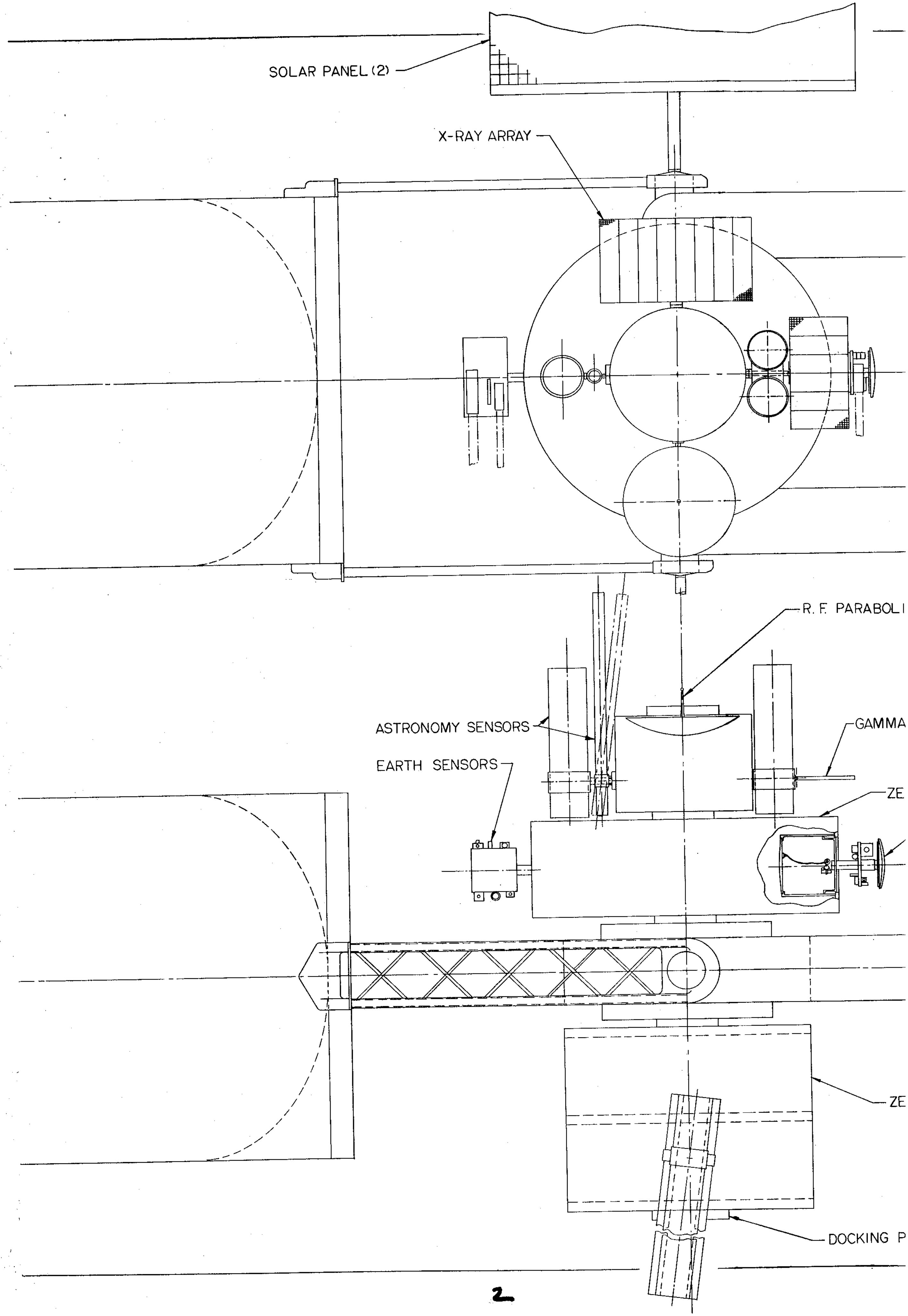


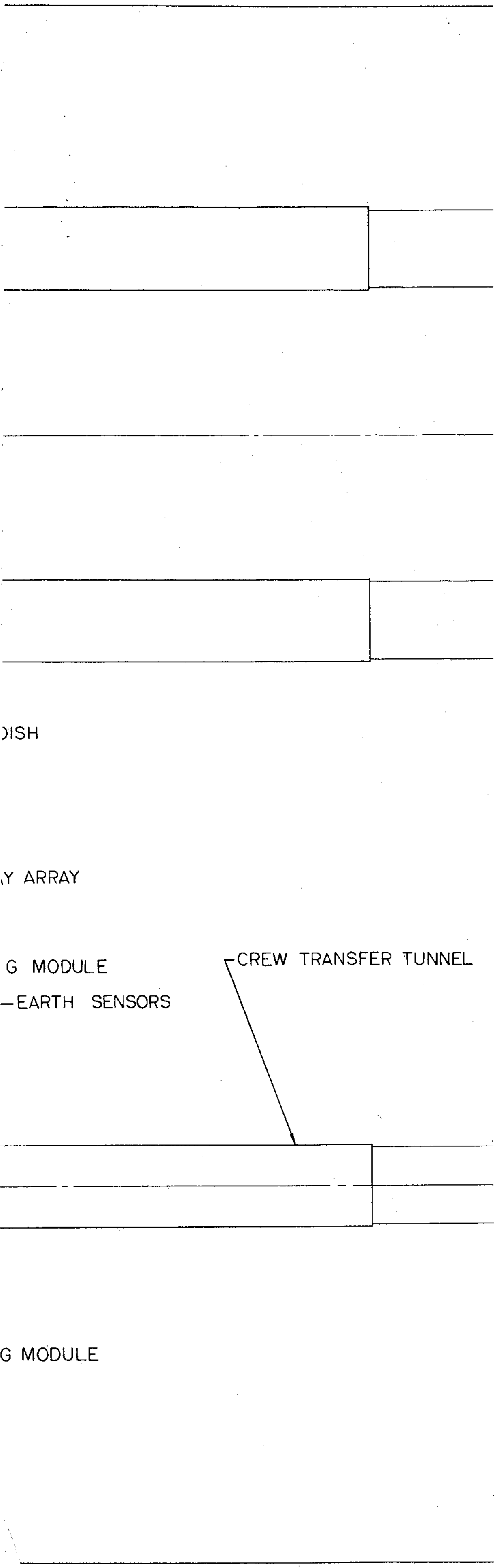
LAUNCH CONFIG

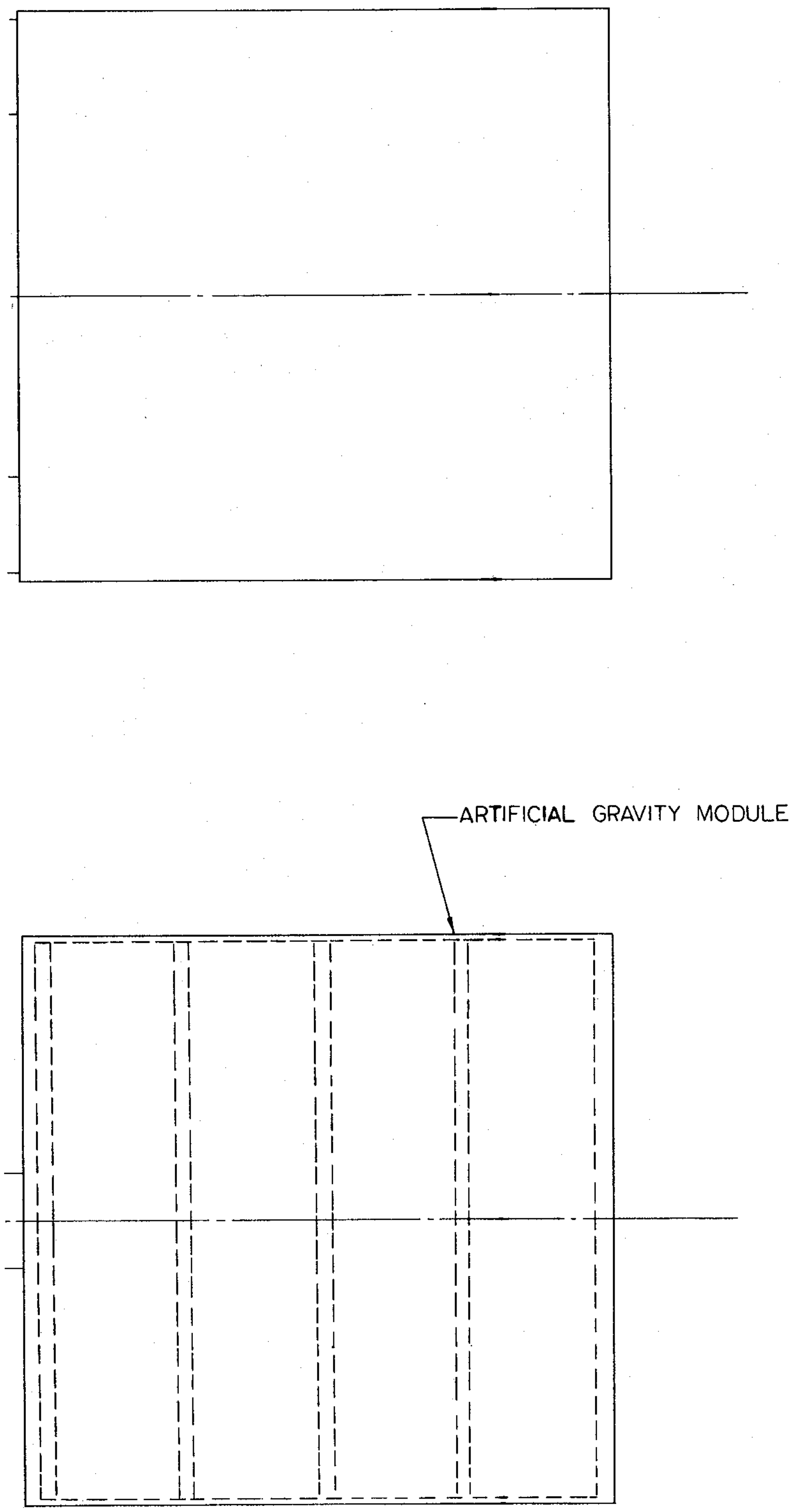
GENERAL ARRANGEMENT
ARTIFICIAL G, 33 FT. DIA, 9-12 MAN
SPACE STATION

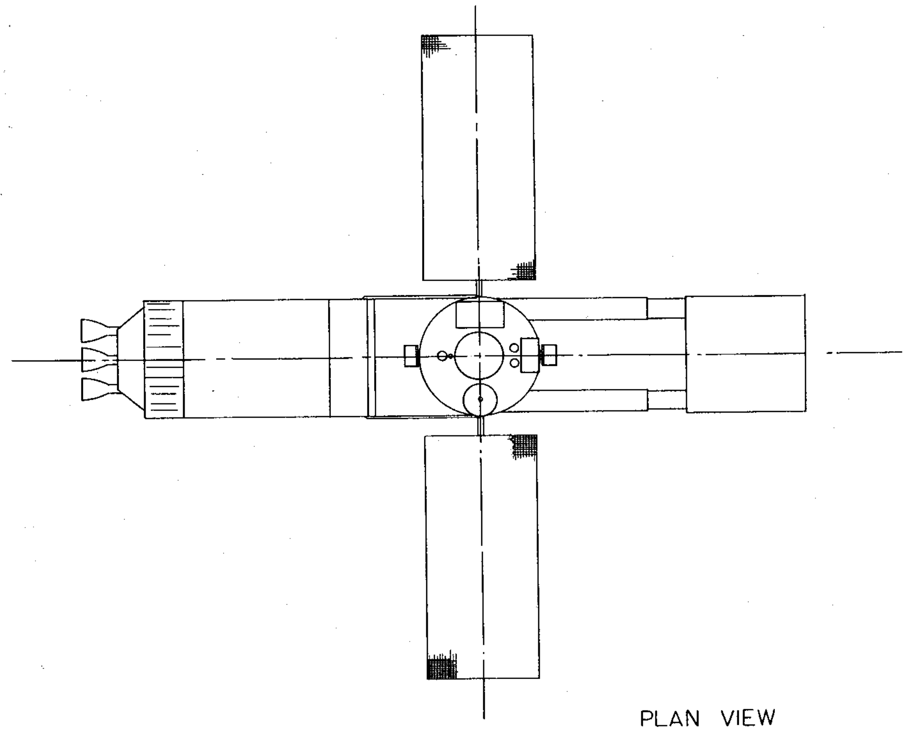
FIGURE 4.5


S-II COUNTERWEIGHT


SOLAR PANEL (2)
X-RAY ARRAY
R. F. PARABOLI
ASTRONOMY SENSORS
EARTH SENSORS
GAMMA
ZE
ZE
DOCKING P

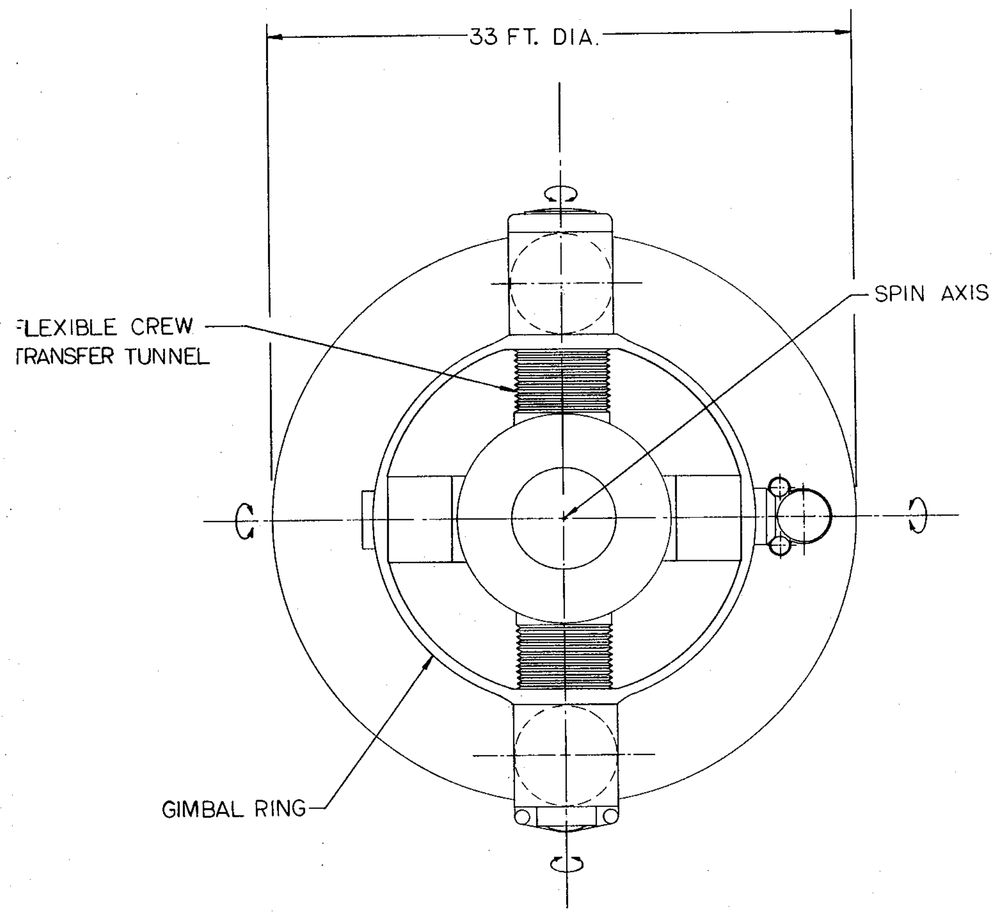

ISH
Y ARRAY
G MODULE
CREW TRANSFER TUNNEL
EARTH SENSORS
G MODULE


ARTIFICIAL GRAVITY MODULE

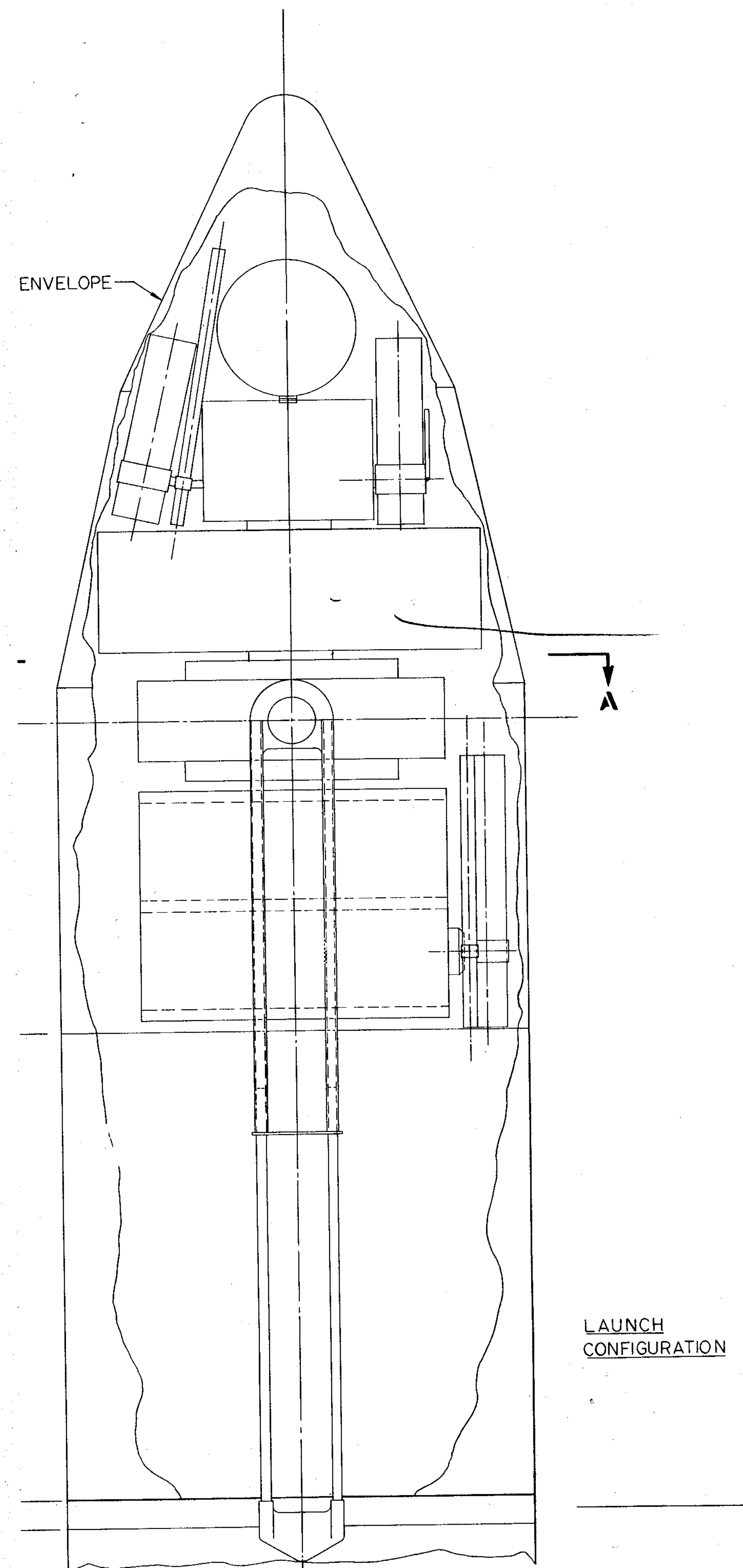


PLAN VIEW

(SCALE- 1:200)



SECTION A-A



GENERAL ARRANGEMENT
ARTIFICIAL G, 33 FT. DIA.,
24 MAN SPACE STATION

FIGURE 4.6

The interior arrangements of the artificial gravity modules for a 9-man, 260 inch diameter station; a 9-man, 396 inch diameter station; and a 24-man, 396 inch diameter station are shown in Figures 4.7, 4.8, and 4.9, respectively. The number of compartments shown for the 396 inch diameter modules differs from the number shown in the general arrangement drawings of the 396 inch diameter configuration because of launch packaging considerations and the fact that emergency escape device storage volume requirements are not defined. Also, the laboratory volume suitable for the artificial gravity module is only an estimate at this time. The arrangements shown are considered to be representative.

4.3 ARTIFICIAL GRAVITY CONFIGURATION EVOLUTION

Figures 4.10, 4.11, and 4.12 show general arrangements of the initial artificial gravity stations conceived during this study. It was determined that there are four basic arrangements of the "I" configuration station relative to the launch configuration as indicated in Figure 4.13. These four arrangements involve locating the artificial gravity module either above or below the hub for launch packaging and having the hub centerline (corresponding to the deployed station spin axis) located either vertically or horizontally while on the launch pad. Arrangement 4, as shown in Figure 4.13, was selected for further study for the following reasons:

a. All compartment floors are horizontal and in normal down position on the launch pad to facilitate check-out.

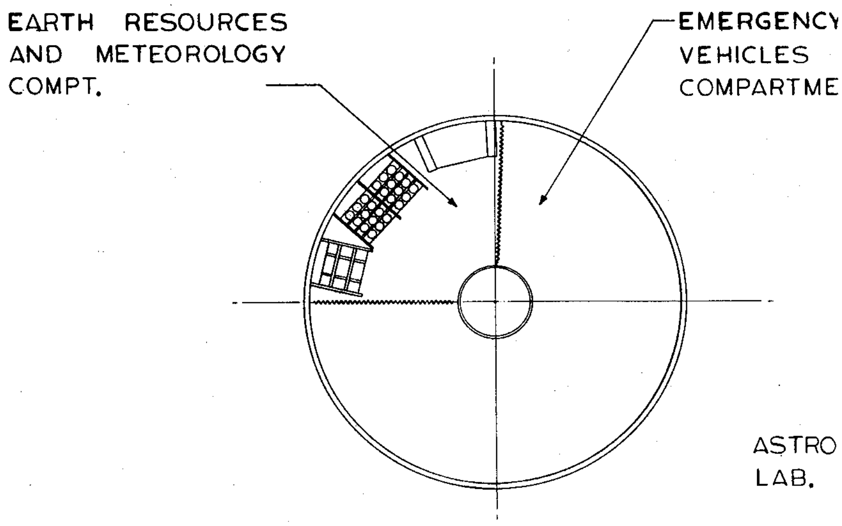
b. Maximum utilization is made of the conically shaped nose required for the external launch configuration by installing telescopes, antennas, etc. in this area.

c. Aerodynamic fairing structural requirements are minimized (i.e., no massive station modules are supported by fairings).

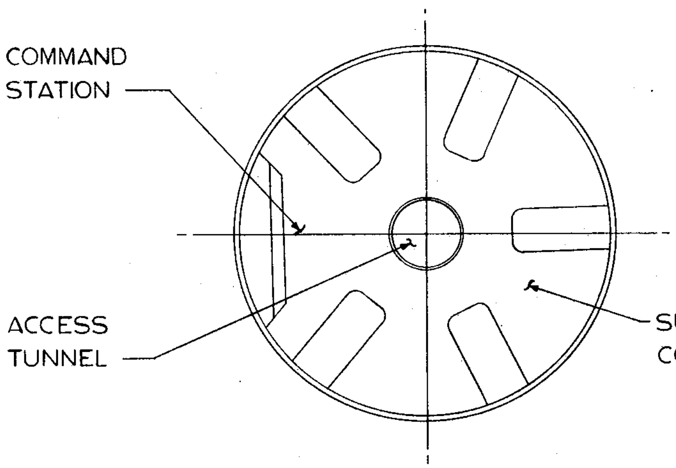
d. Maximum flexibility of non-rotating hub arrangement is provided.

e. Deployment linkages proved to be either simplified or not unduly affected because of minimum telescoping requirements.

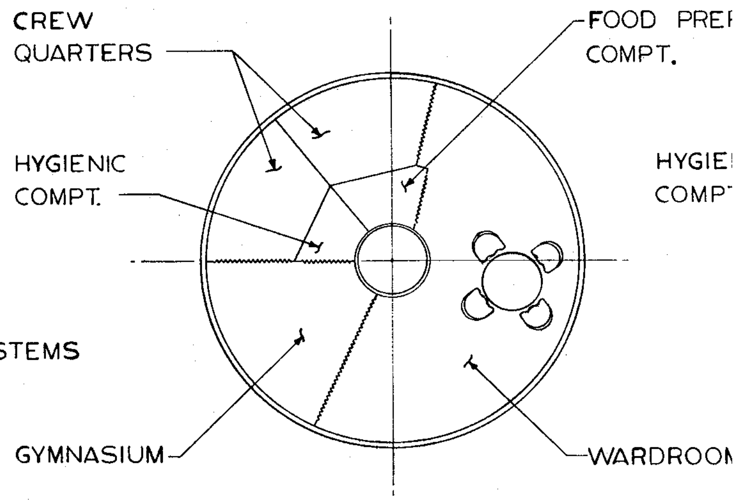
The configuration shown in Figure 4.10 does not utilize the S-II stage as a counterweight. Instead, a small module is deployed which may either be ballast or provide a more useful function such as to house a nuclear electrical power system. The weight of the counterweight could be small relative to the artificial gravity module by deploying it at a large spin



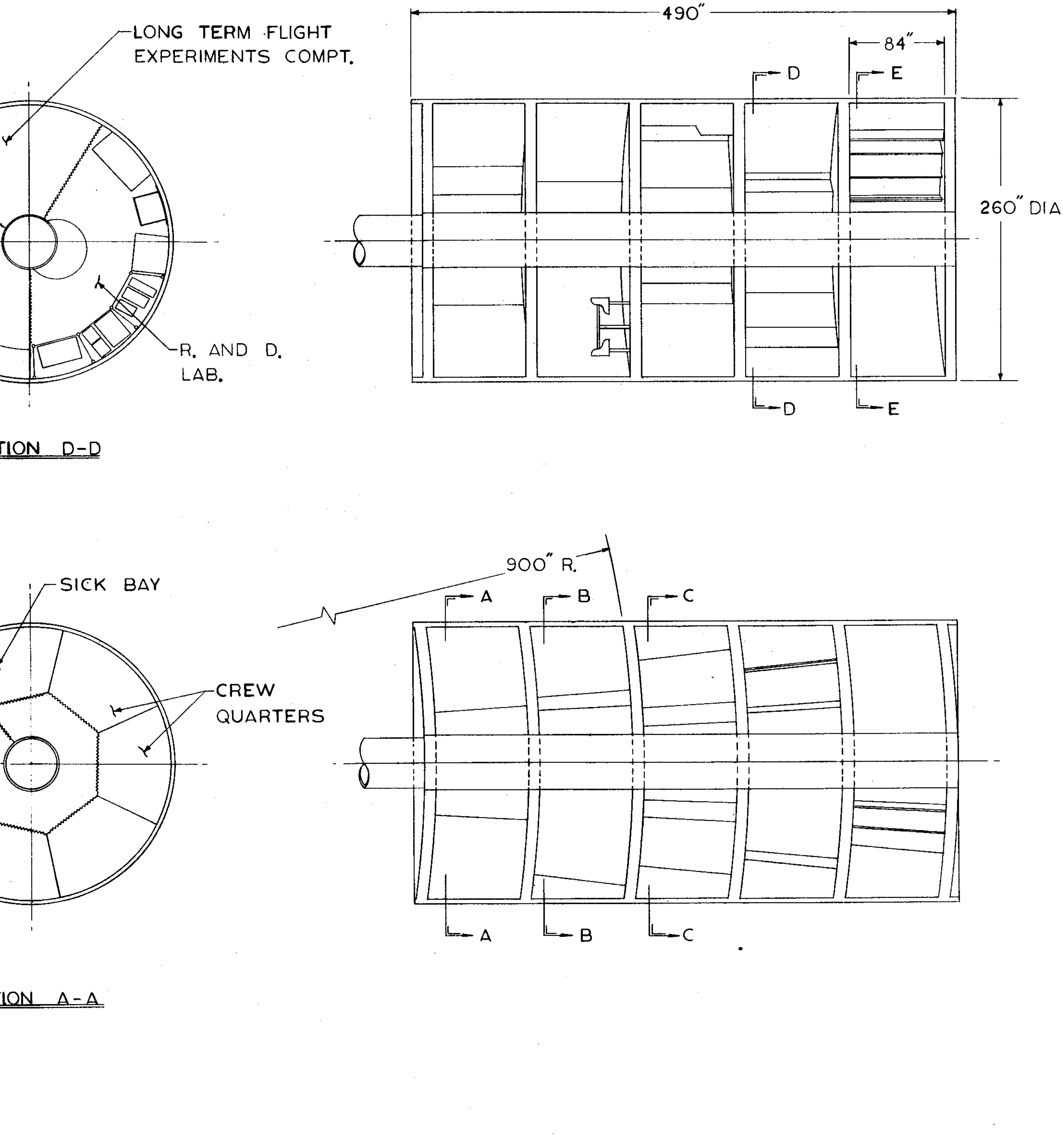
SECTION E-E



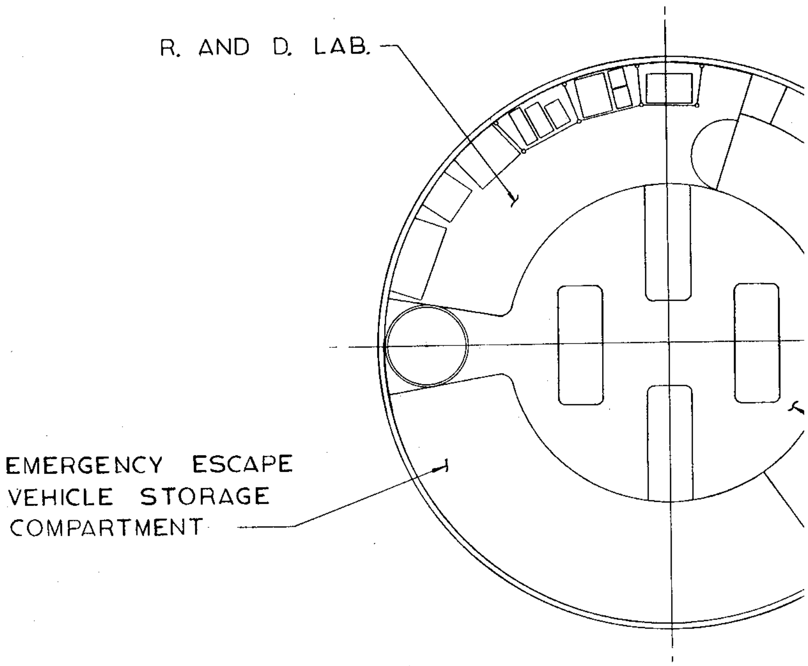
SECTION C-C



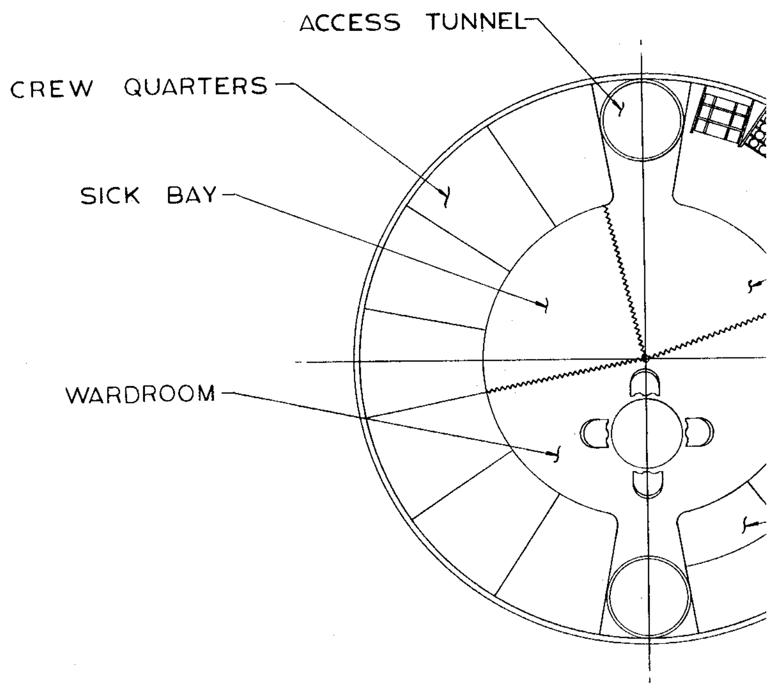
SECTION B-B



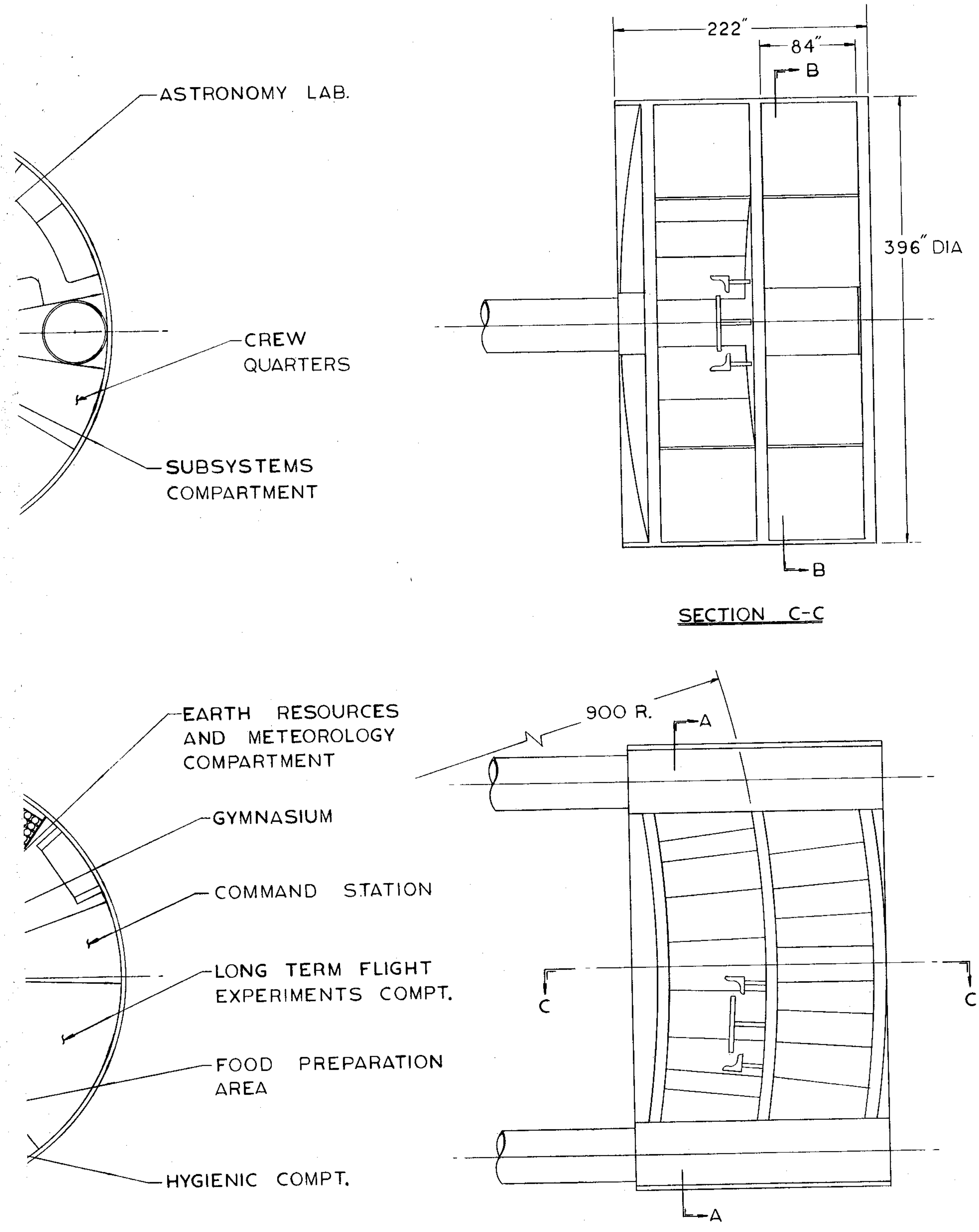
SPACE STATION — ART. "G" MODULE
9 MAN CREW — 260" DIA
FIGURE 4.7



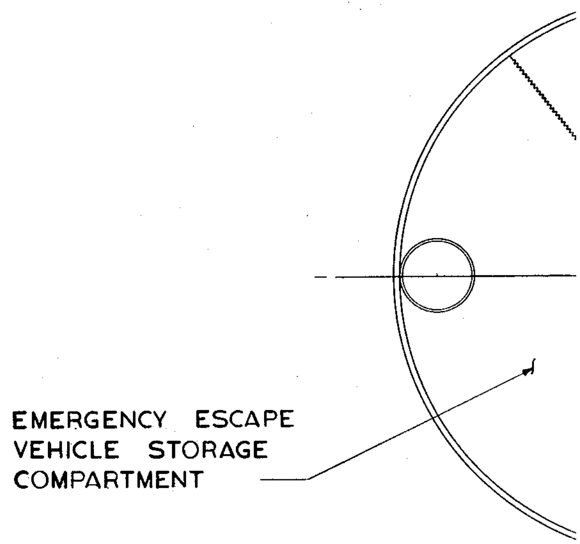
SECTION B-B

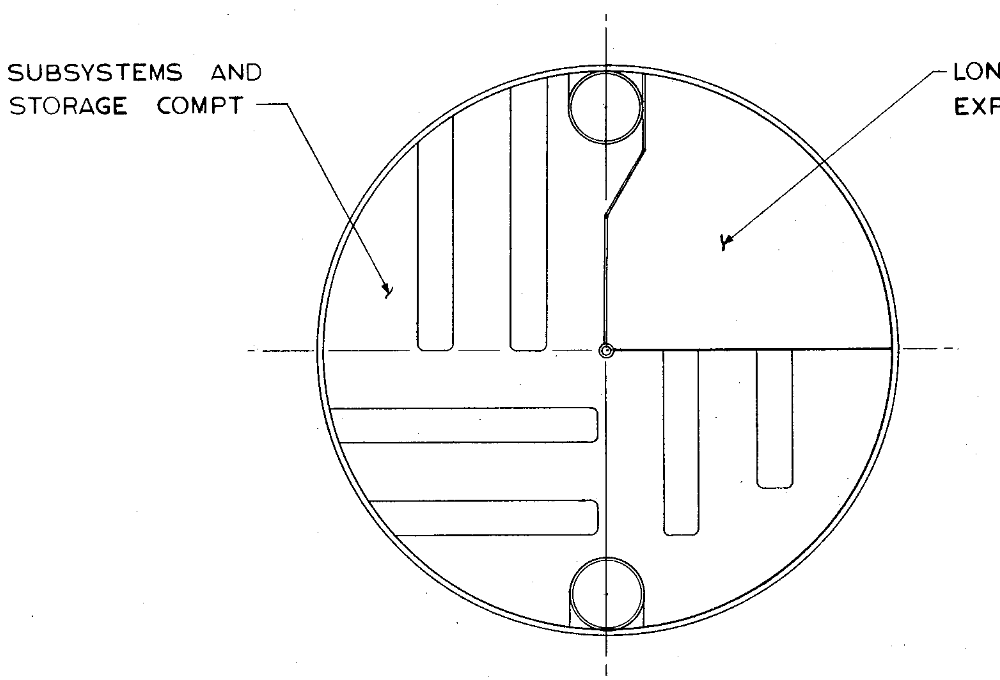


SECTION A-A

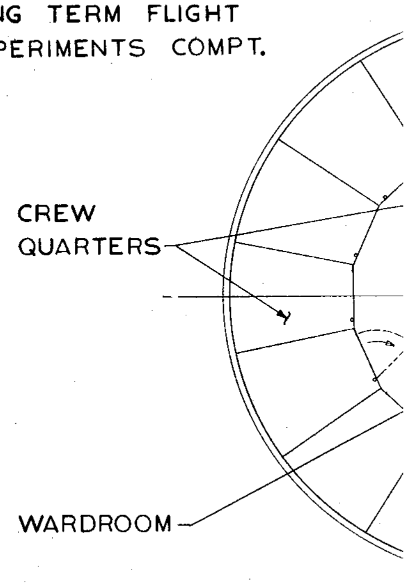


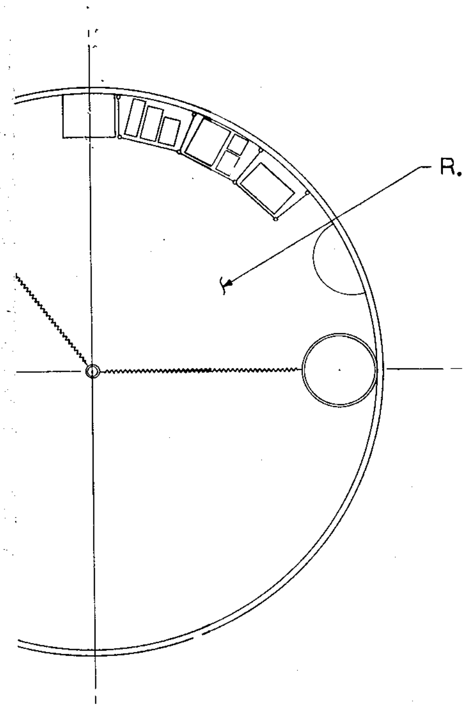
SPACE STATION — ART. "G" MODULE
9 MAN CREW – 33' DIA.
FIGURE 4.8


EMERGENCY ESCAPE
VEHICLE STORAGE
COMPARTMENT

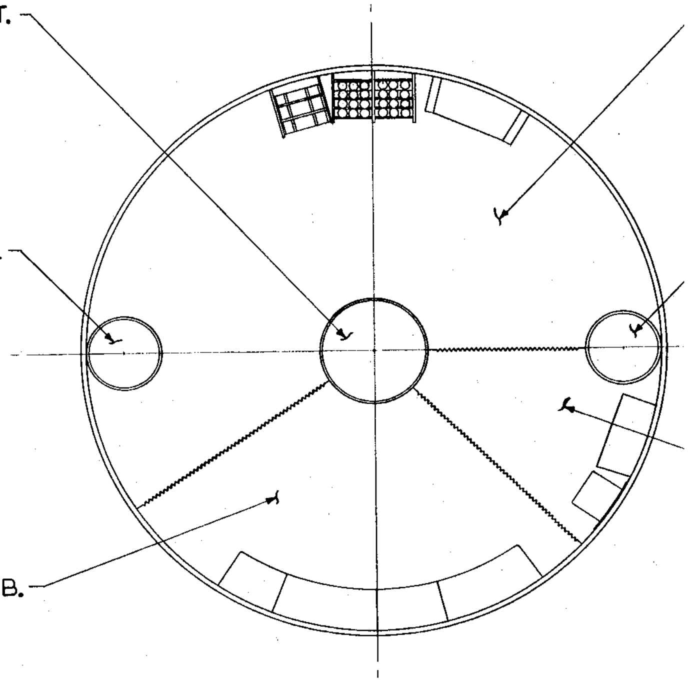

SUBSYSTEMS AND
STORAGE COMPT
LONG TERM FLIGHT
EXPERIMENTS COMPT.

SECTION C-C

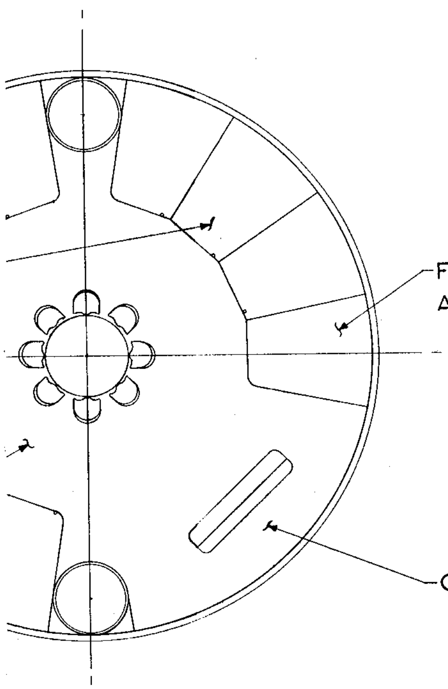

CREW
QUARTERS
WARDROOM



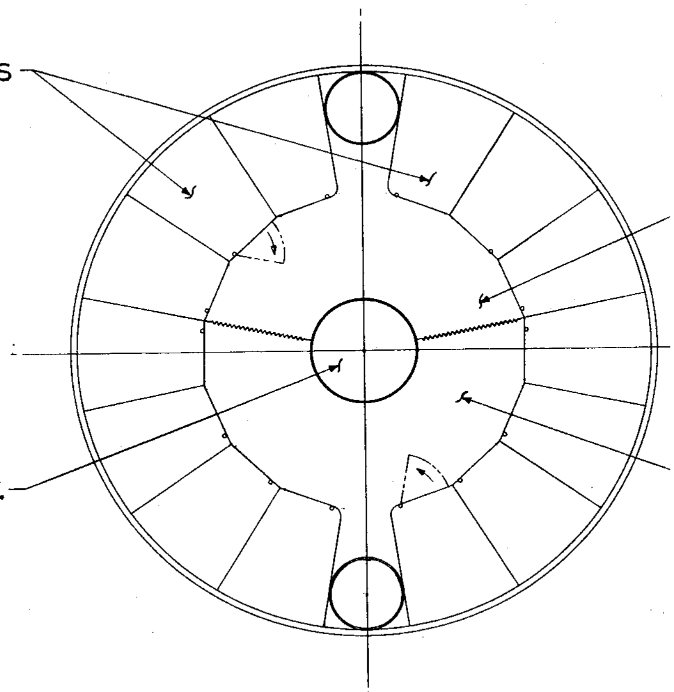
CTION E-E



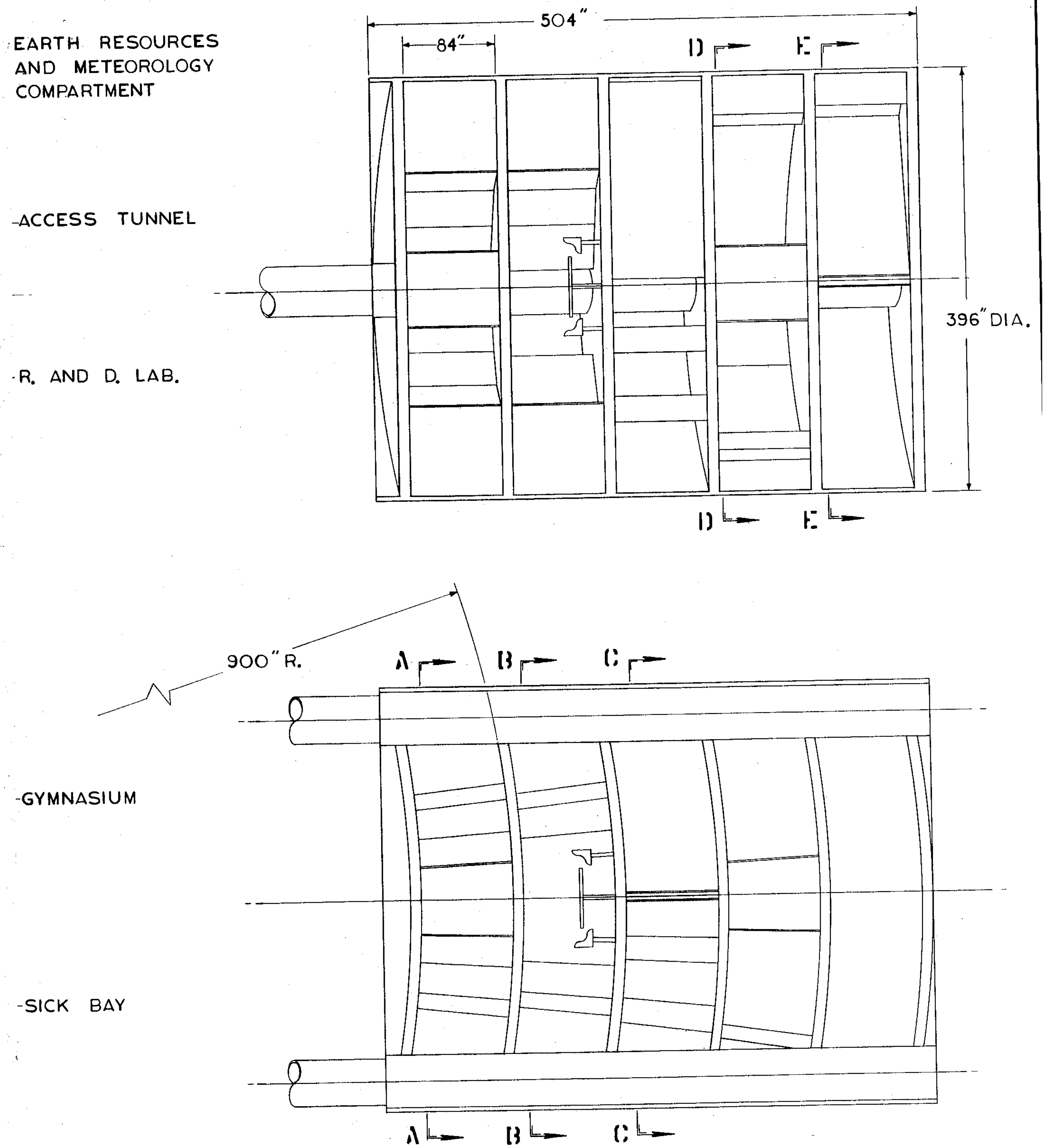
SECTION D-D



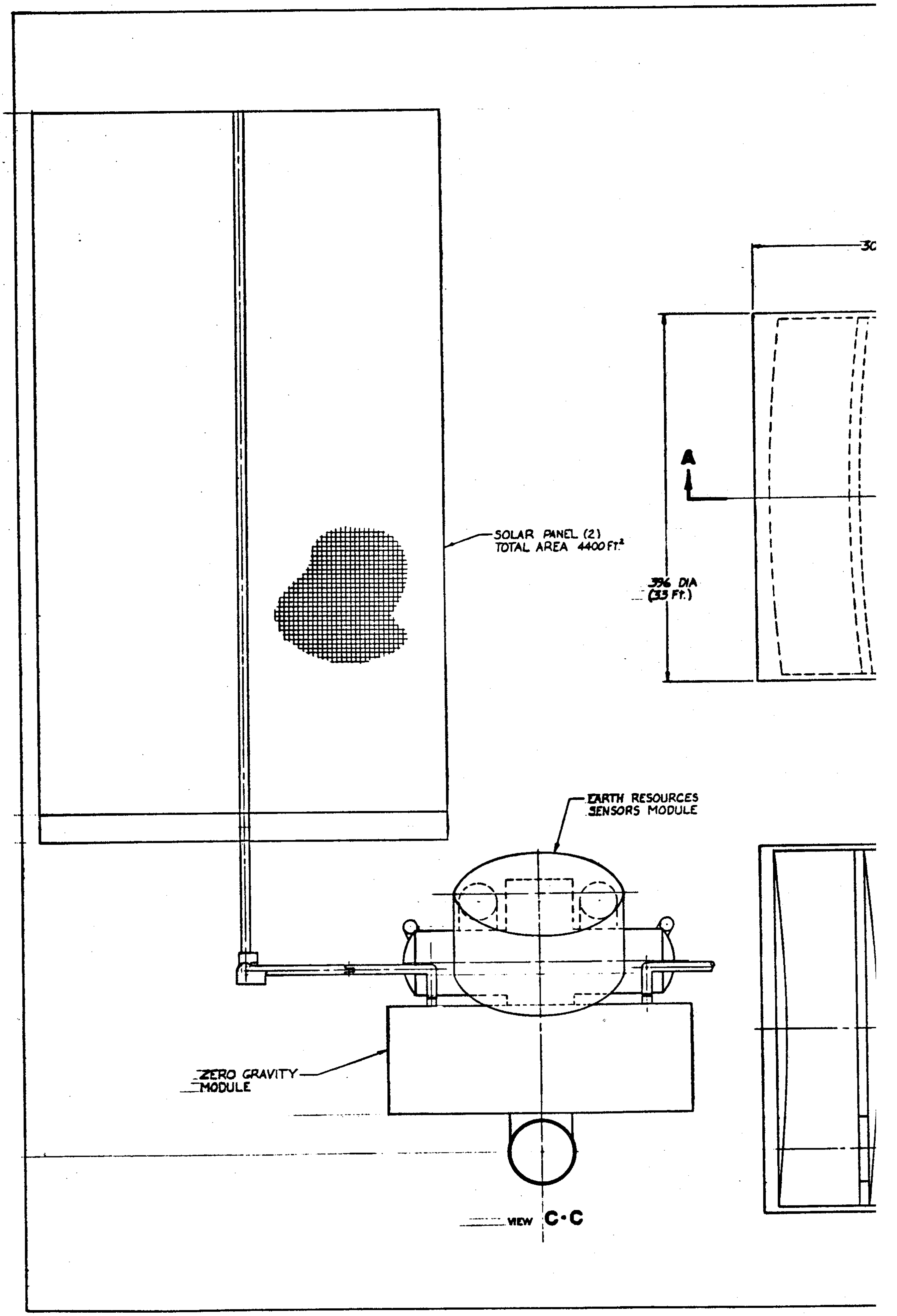
CTION B-B

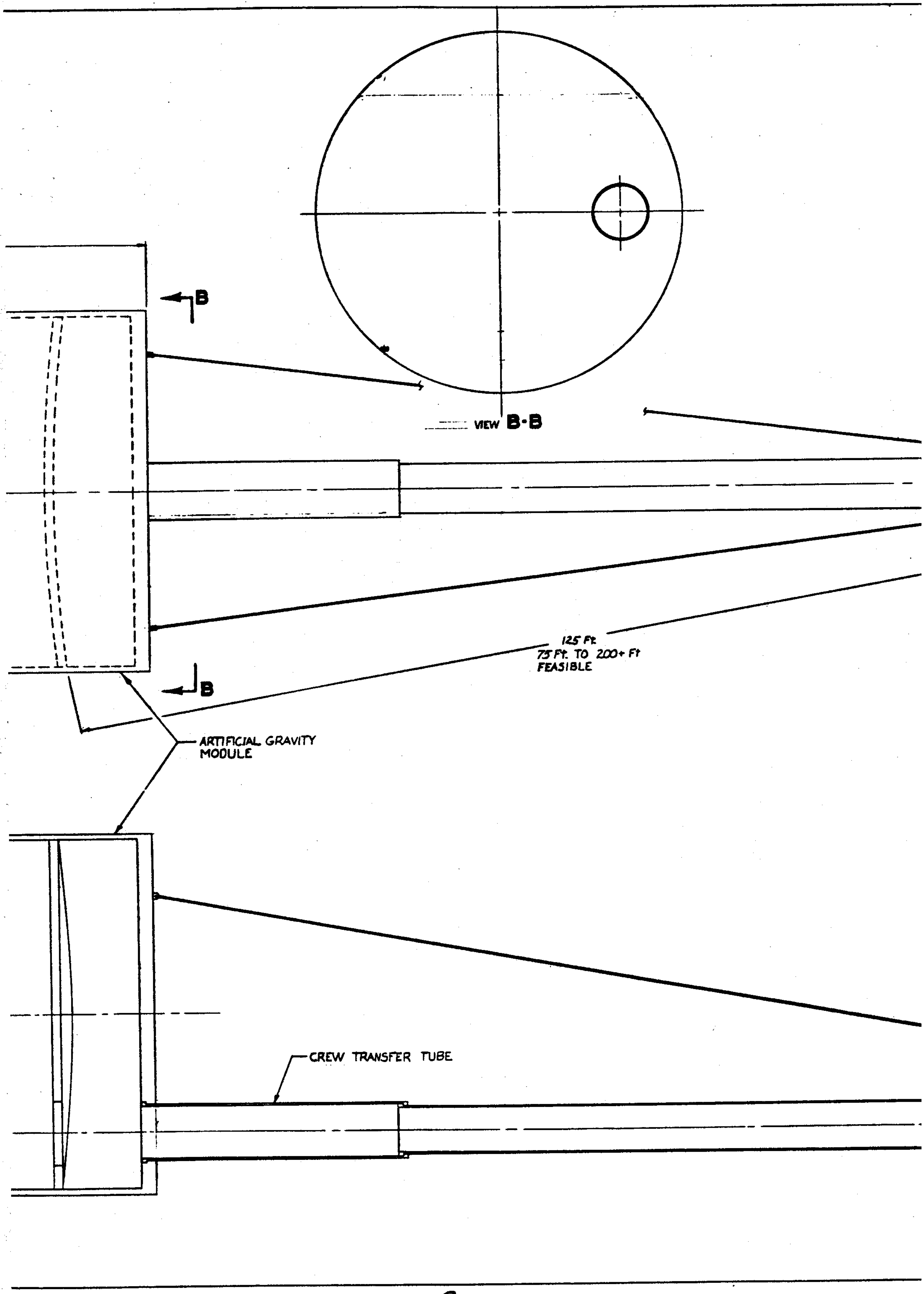


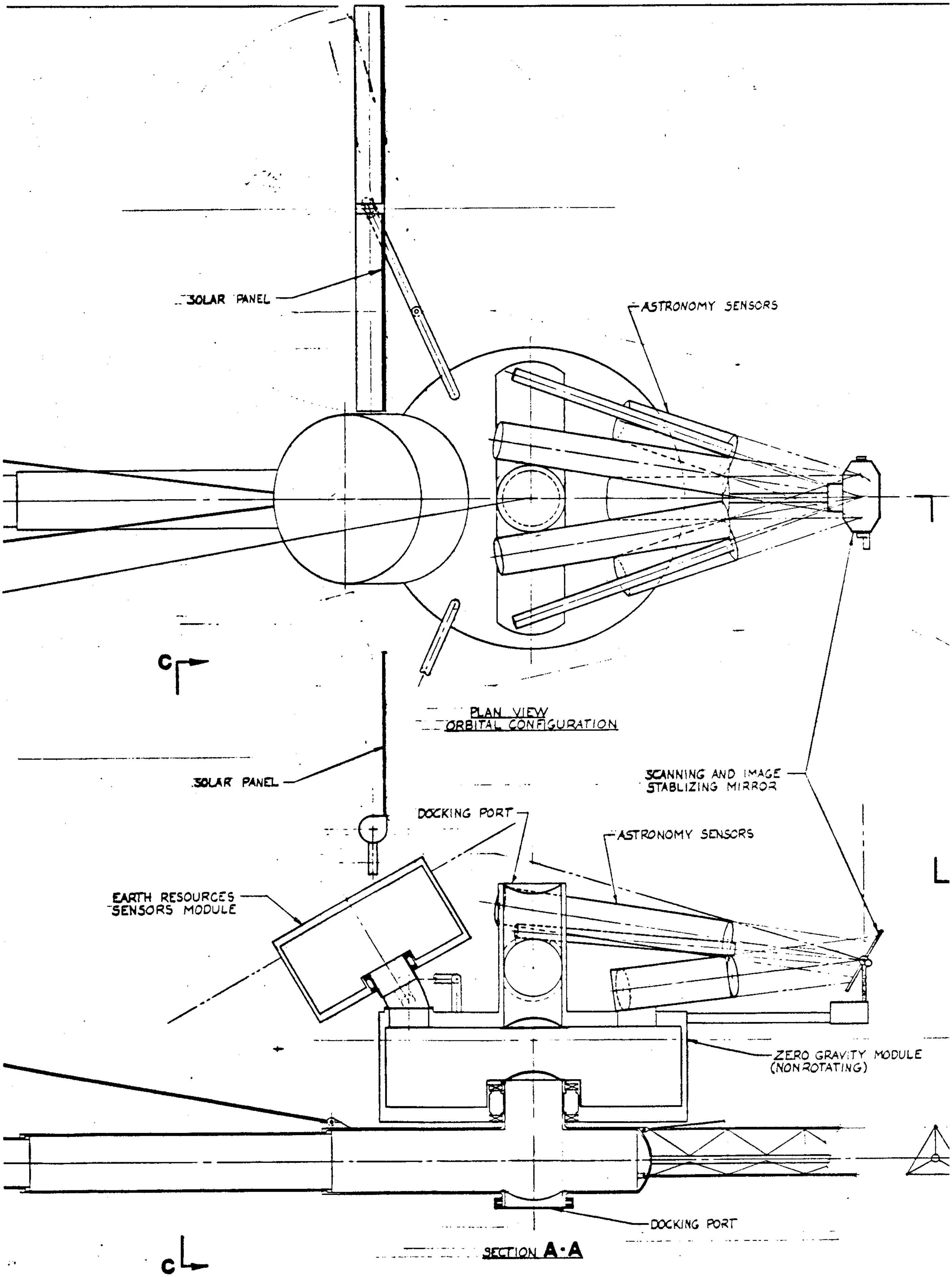
SECTION A-A



SPACE STATION — ART. "G" MODULE
24 MAN CREW — 33' DIA.
FIGURE 4.9


SOLAR PANEL (2)
TOTAL AREA 4400 FT.²
396 DIA
(33 Ft.)
A
30
EARTH RESOURCES
SENSORS MODULE
ZERO GRAVITY
MODULE
VIEW C-C


B
VIEW B-B
B
125 Ft
75 Ft. TO 200+ Ft
FEASIBLE
ARTIFICIAL GRAVITY
MODULE
CREW TRANSFER TUBE


SOLAR PANEL
ASTRONOMY SENSORS
C
PLAN VIEW
ORBITAL CONFIGURATION
SOLAR PANEL
SCANNING AND IMAGE
STABLIZING MIRROR
DOCKING PORT
ASTRONOMY SENSORS
EARTH RESOURCES
SENSORS MODULE
ZERO GRAVITY MODULE
(NONROTATING)
DOCKING PORT
SECTION A·A
C

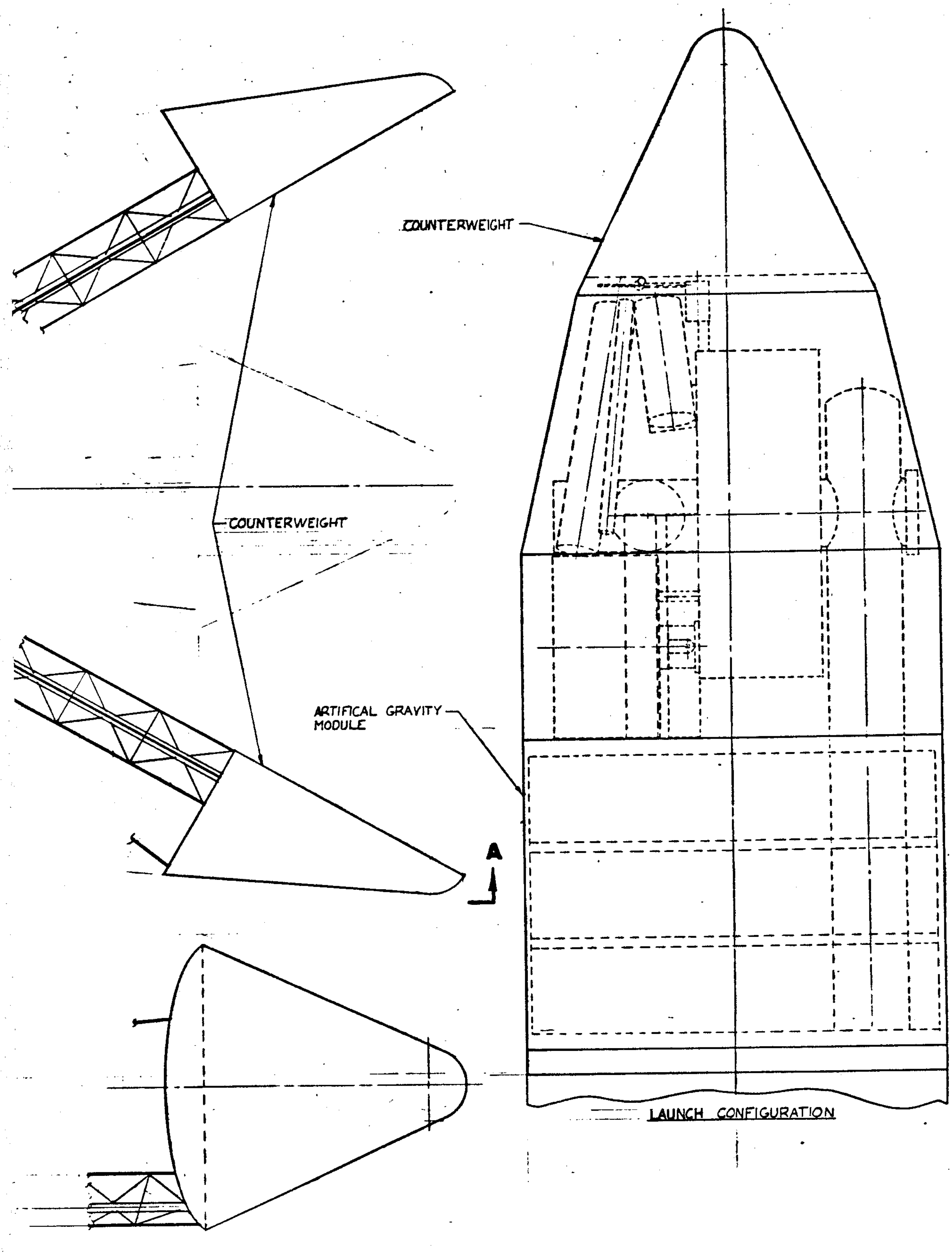
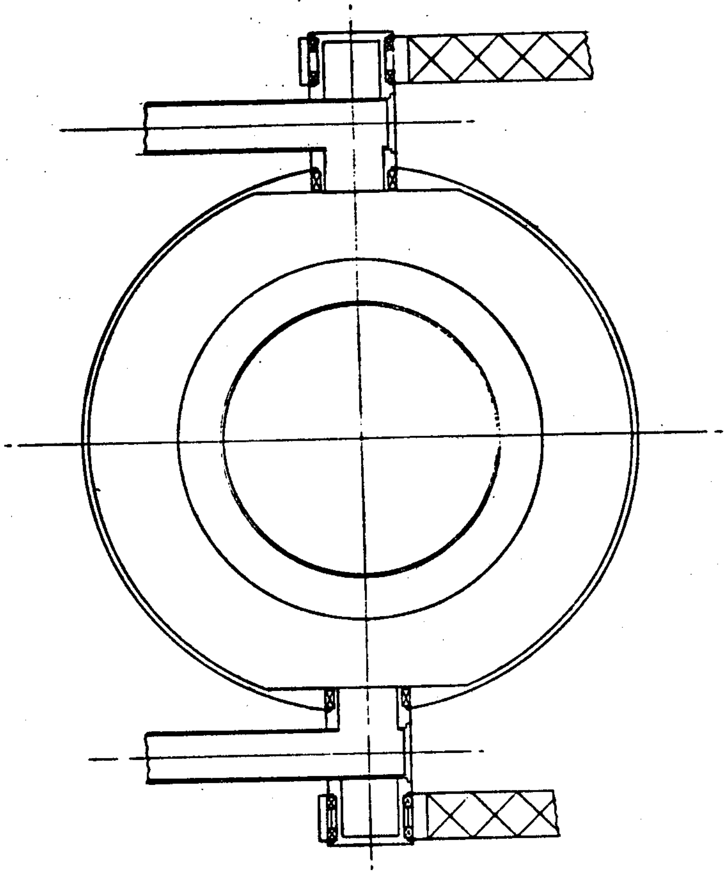
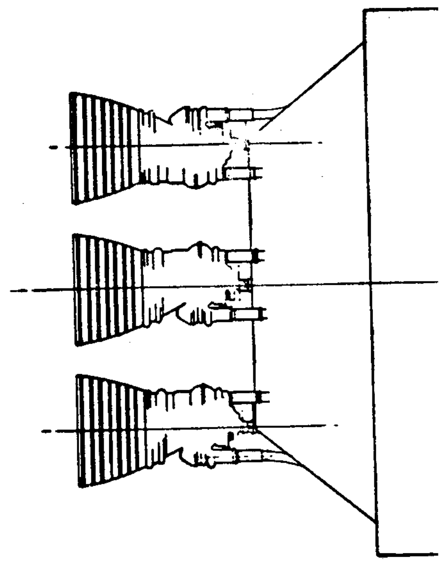


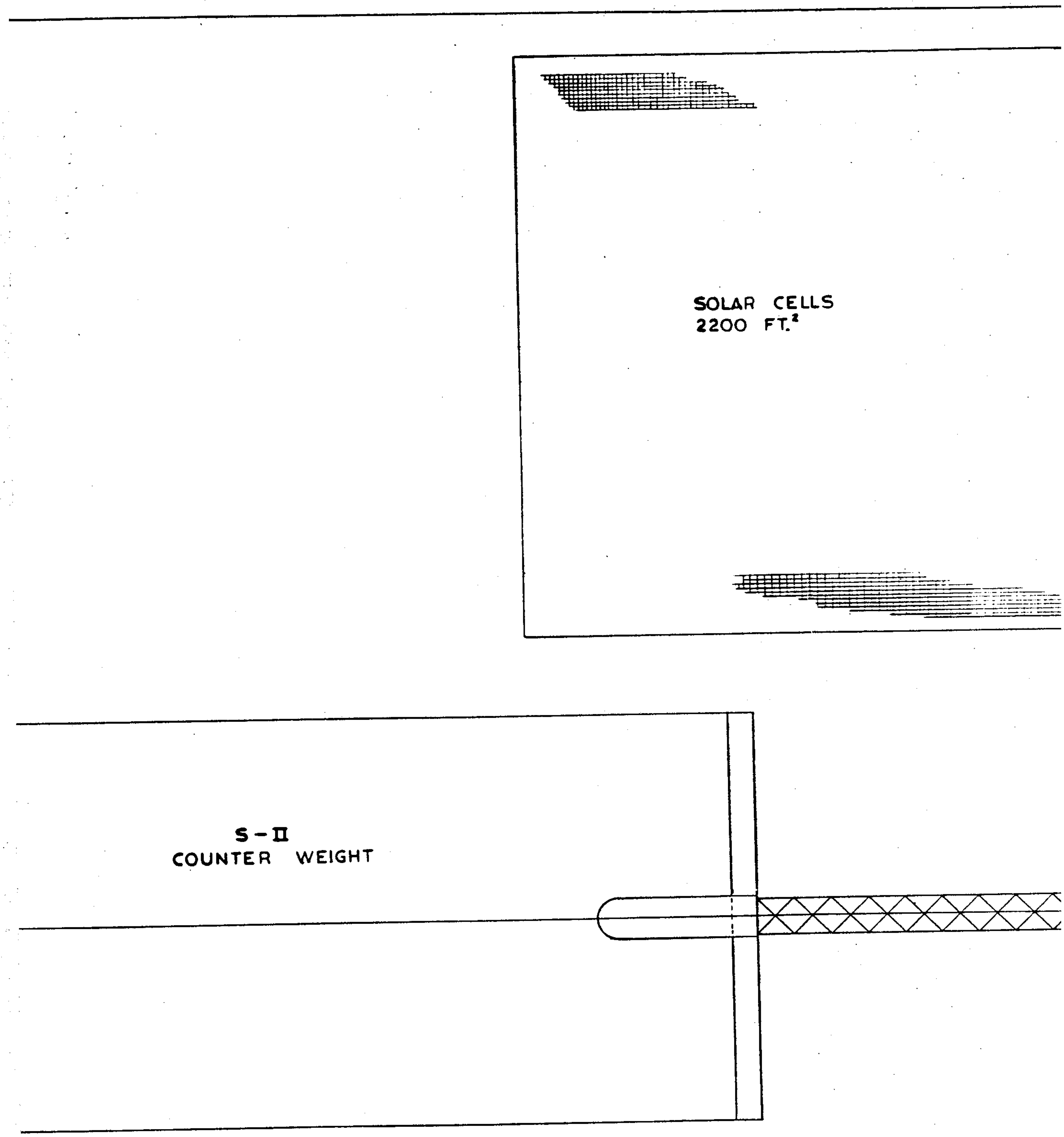
FIGURE 4.10

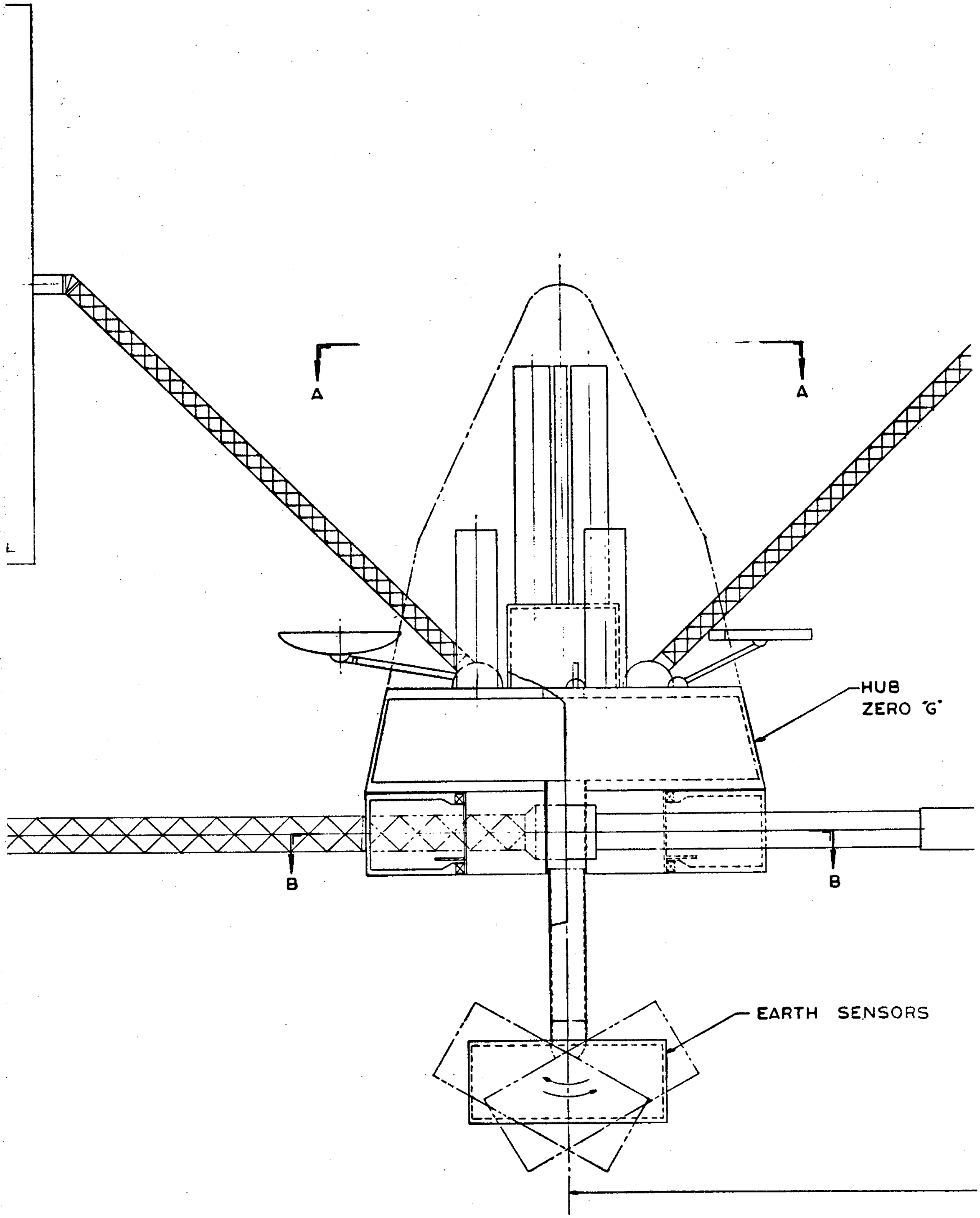
SCALE 1:40	MANNED SPACECRAFT CENTER HOUSTON, TEXAS
	SPACE STATION - ARTIFICAL GRAVITY 24 MAN CREW CONFIGURATION NO. 1
	IPD6-21001

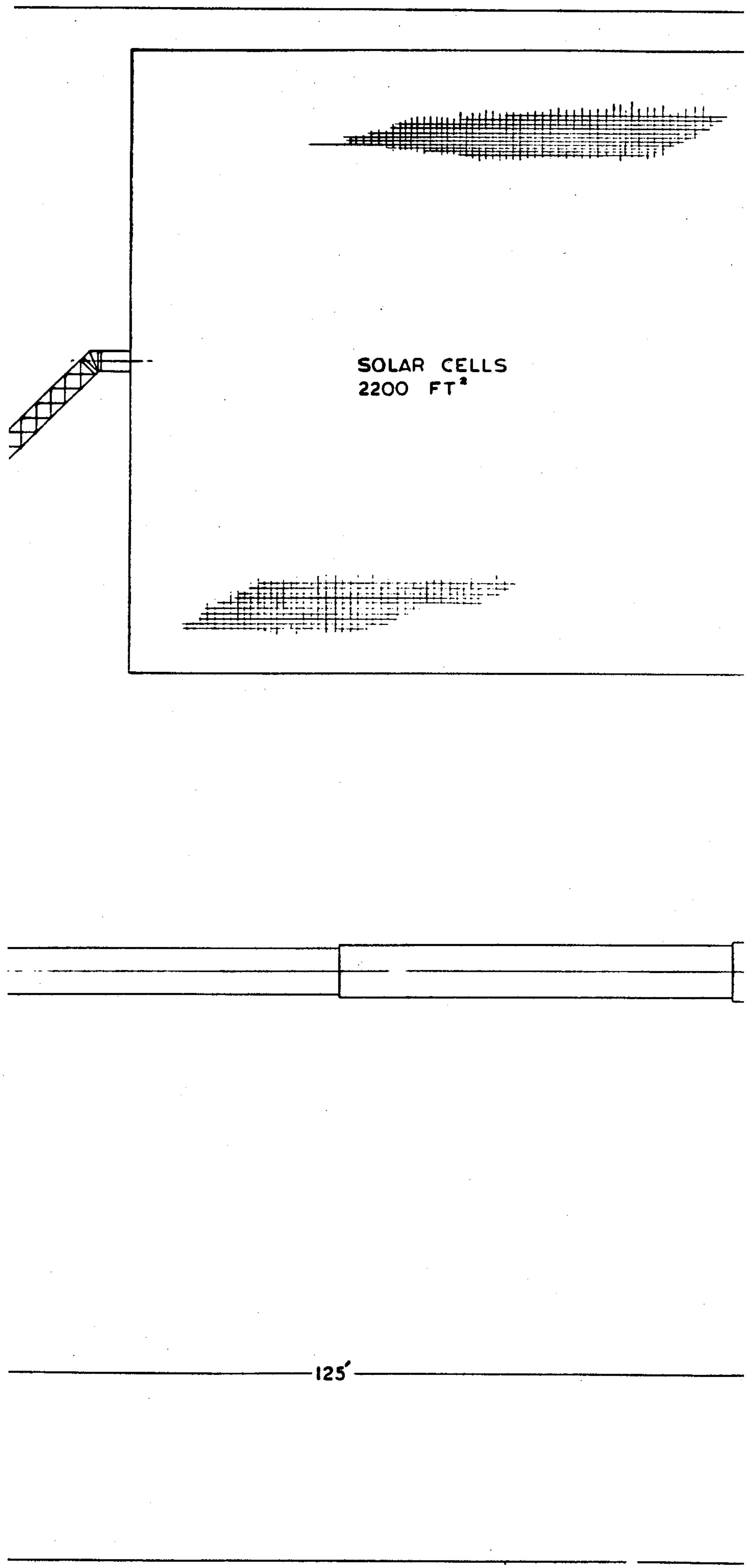


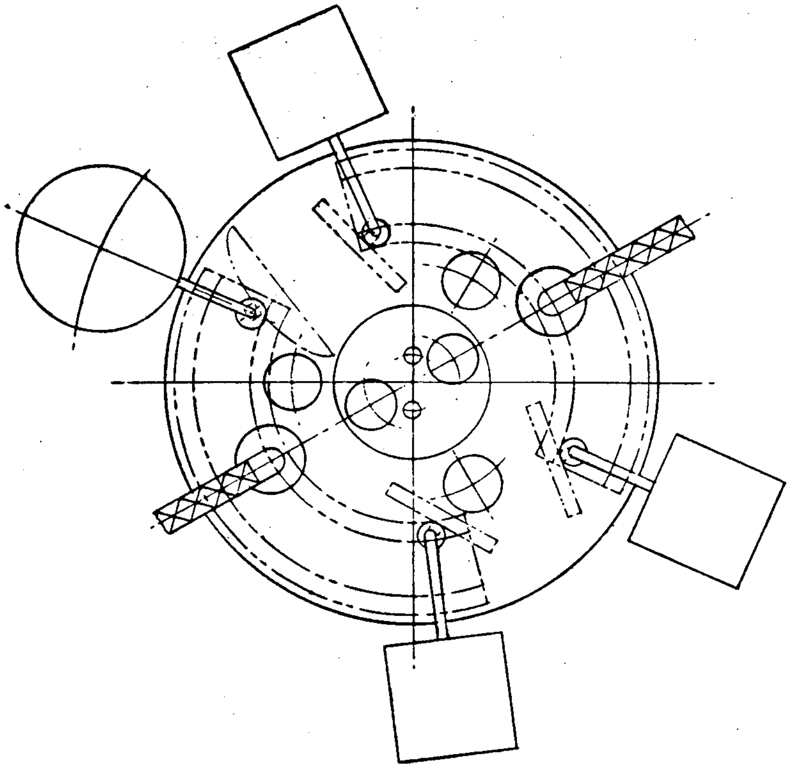
(ROTATED 180°)
SECTION B-B



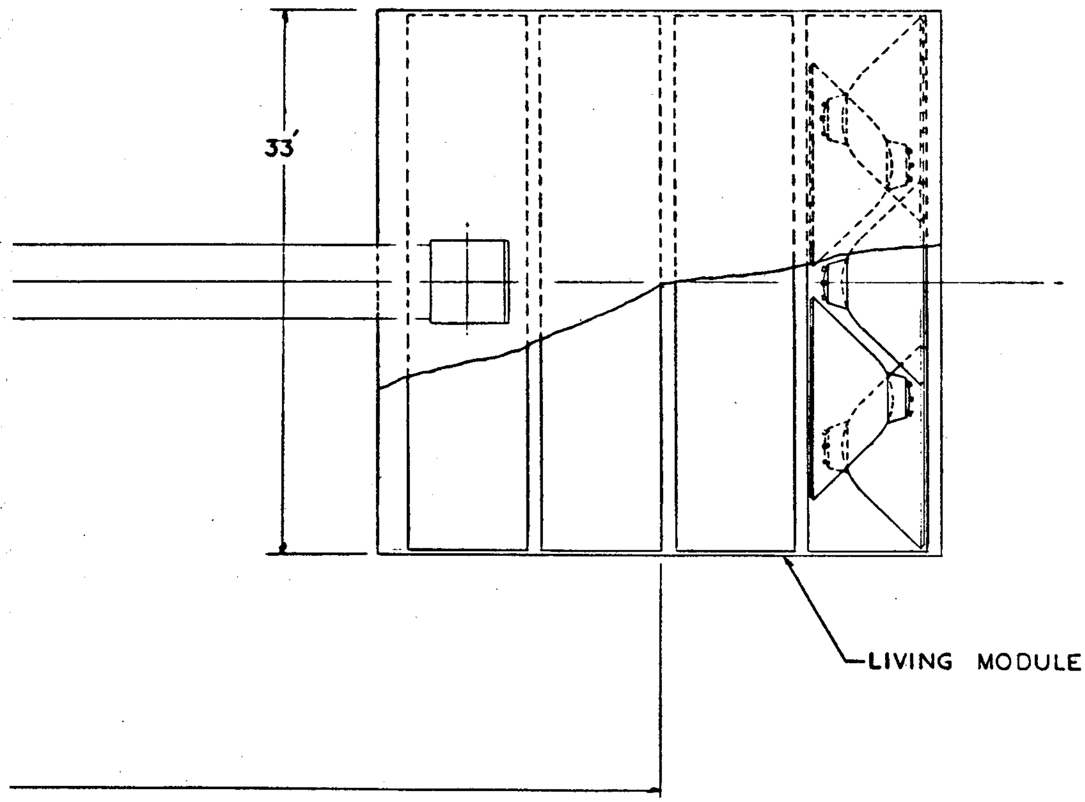

SOLAR CELLS
2200 FT.²
S - II
COUNTER WEIGHT

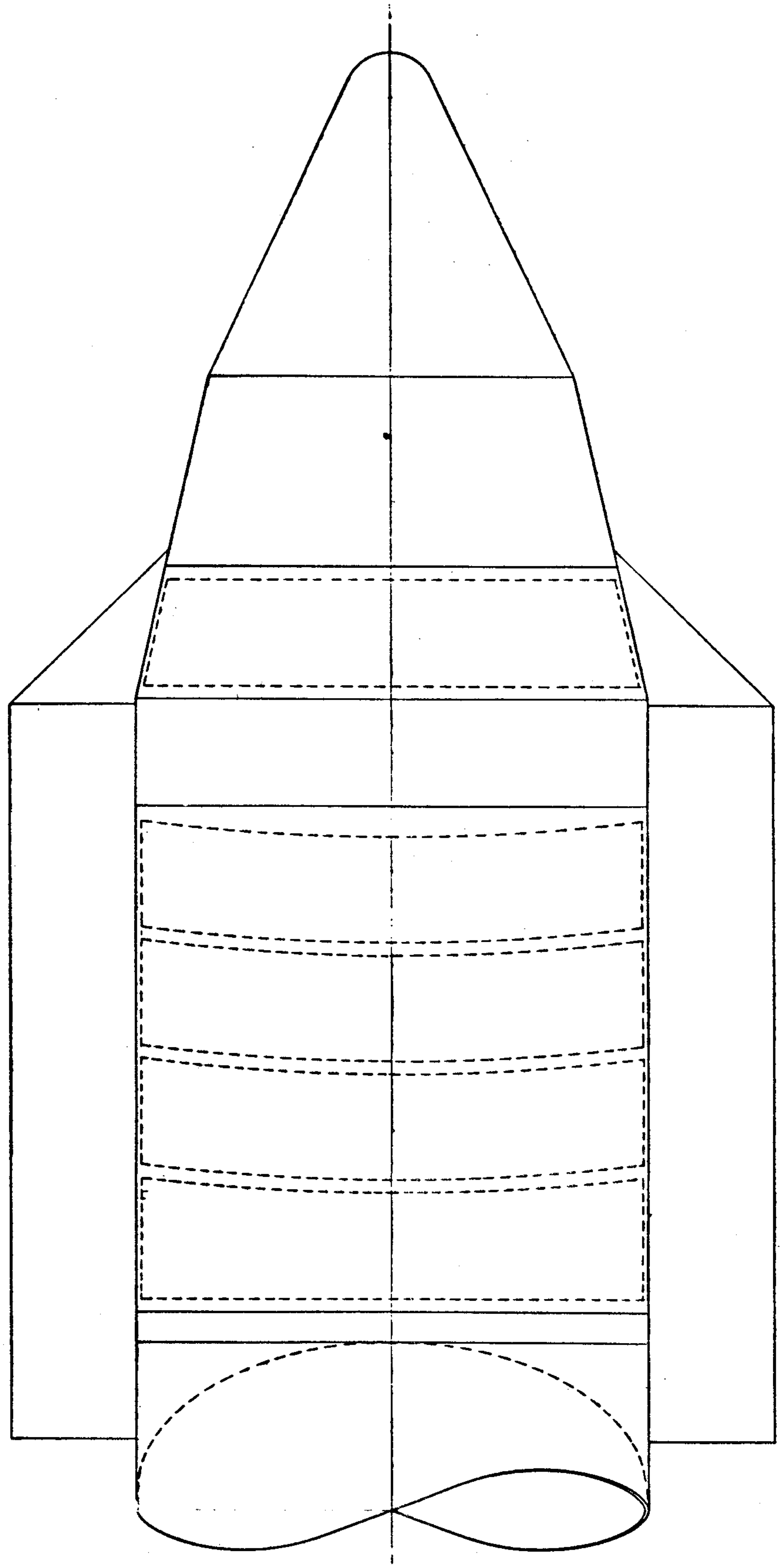

A
A
HUB
ZERO "G"
B
B
EARTH SENSORS


SOLAR CELLS
2200 FT²
125'



VIEW A-A

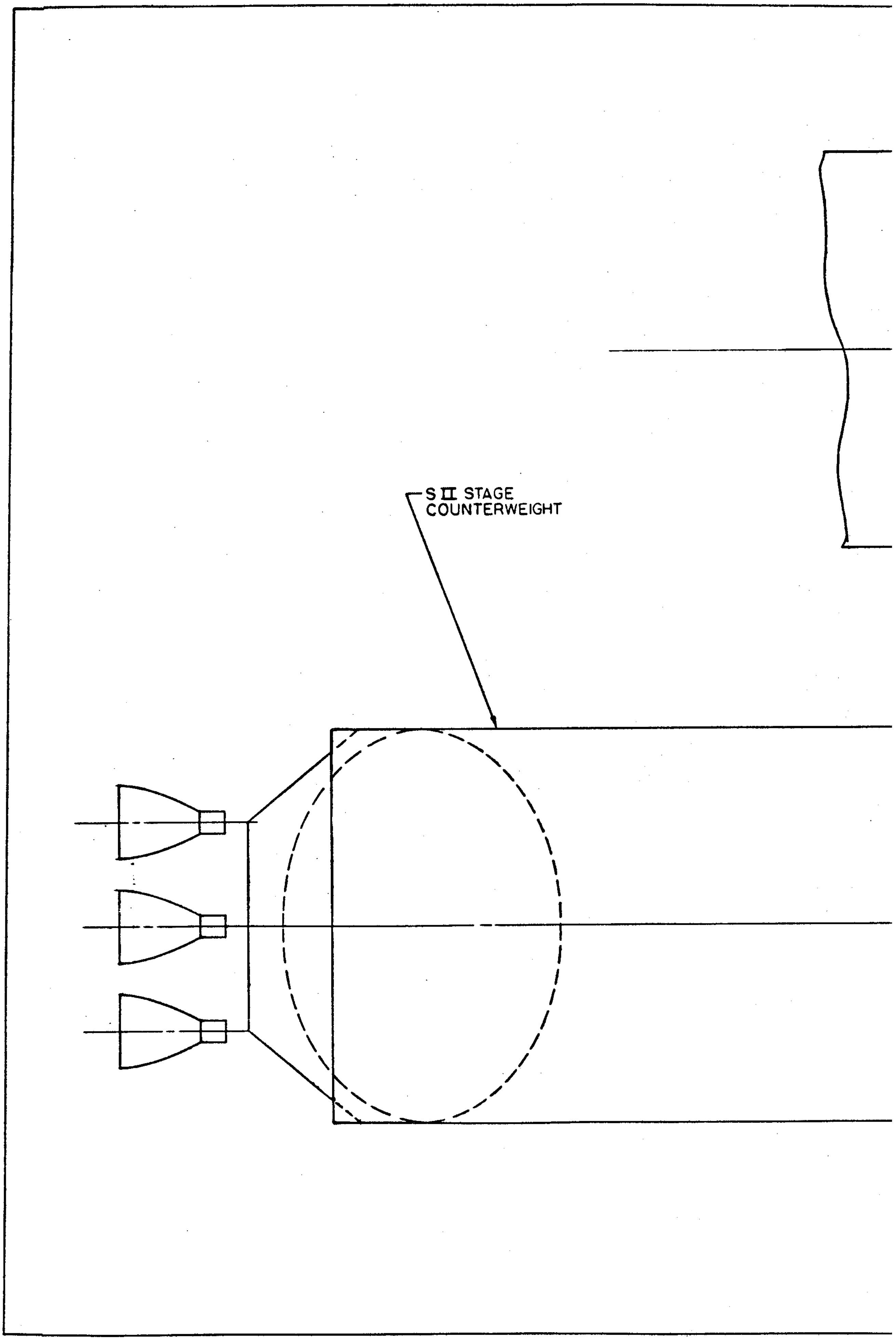


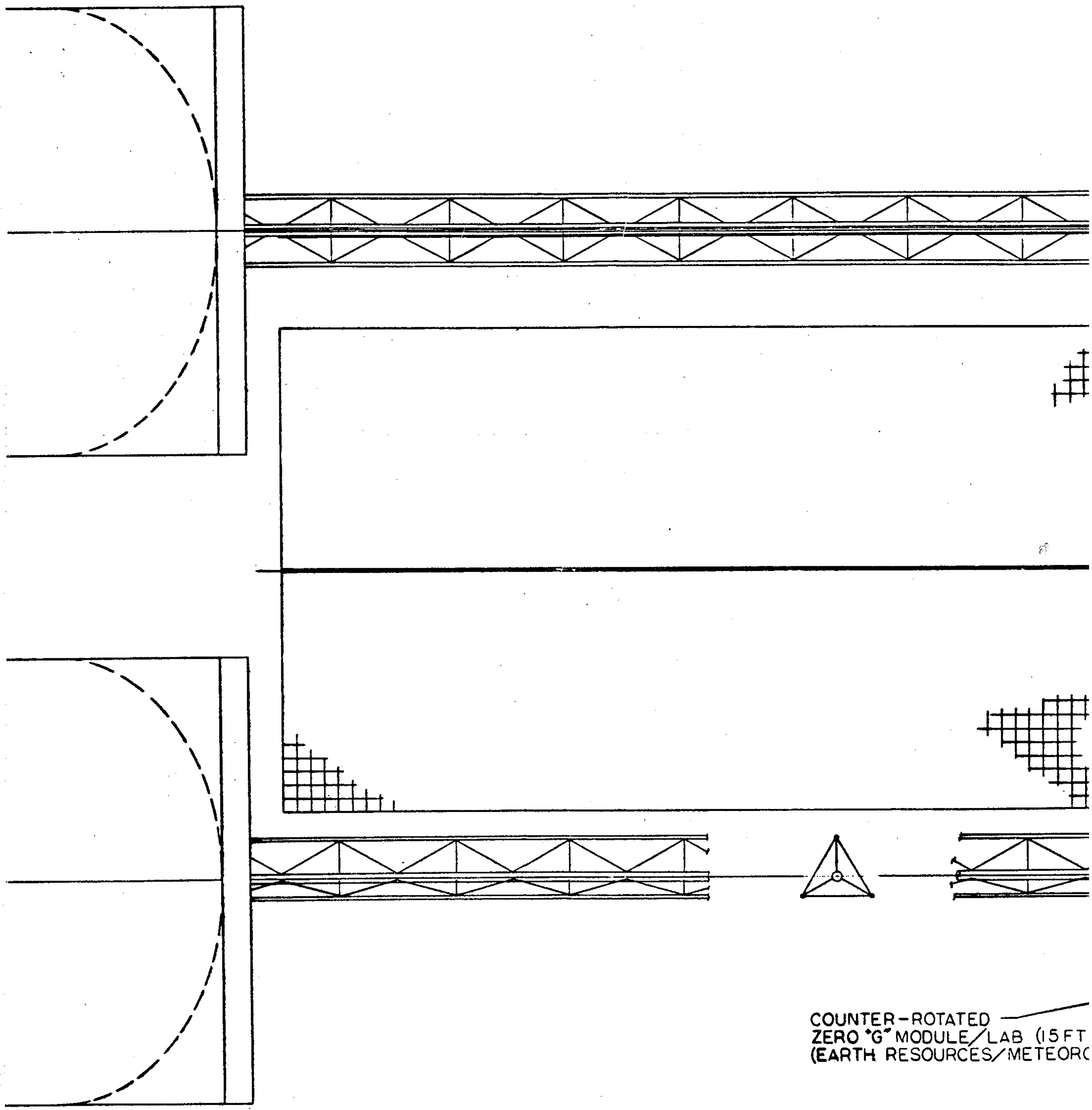


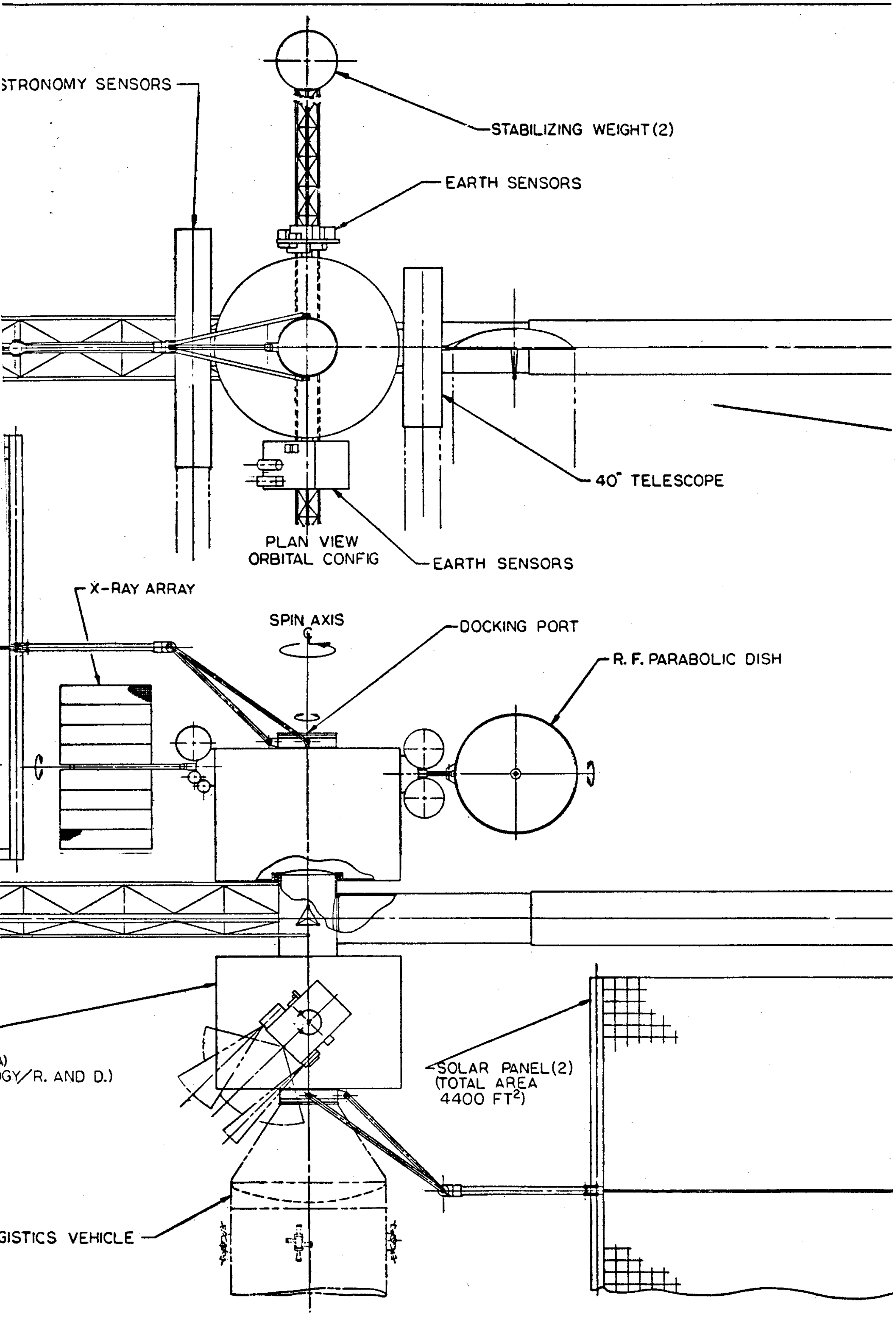
LAUNCH CONFIGURATION

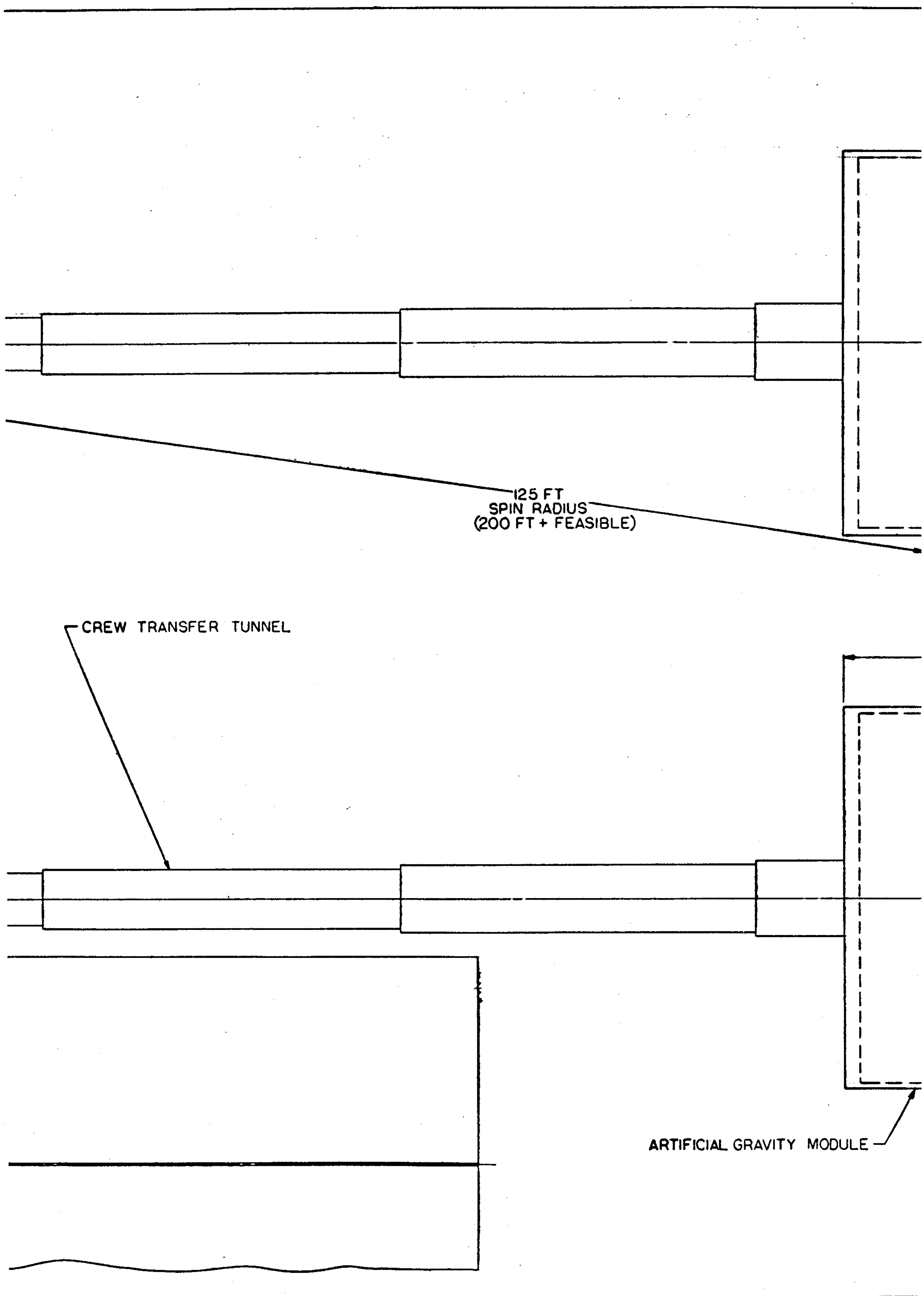
FIGURE 4.11

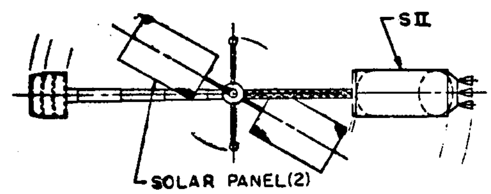
SCALE 1/40	MANNED SPACECRAFT CENTER HOUSTON, TEXAS
	CONCEPT — ART "G" SPACE STATION CONFIG. #2
9/22/66	PD6-21002


S II STAGE
COUNTERWEIGHT

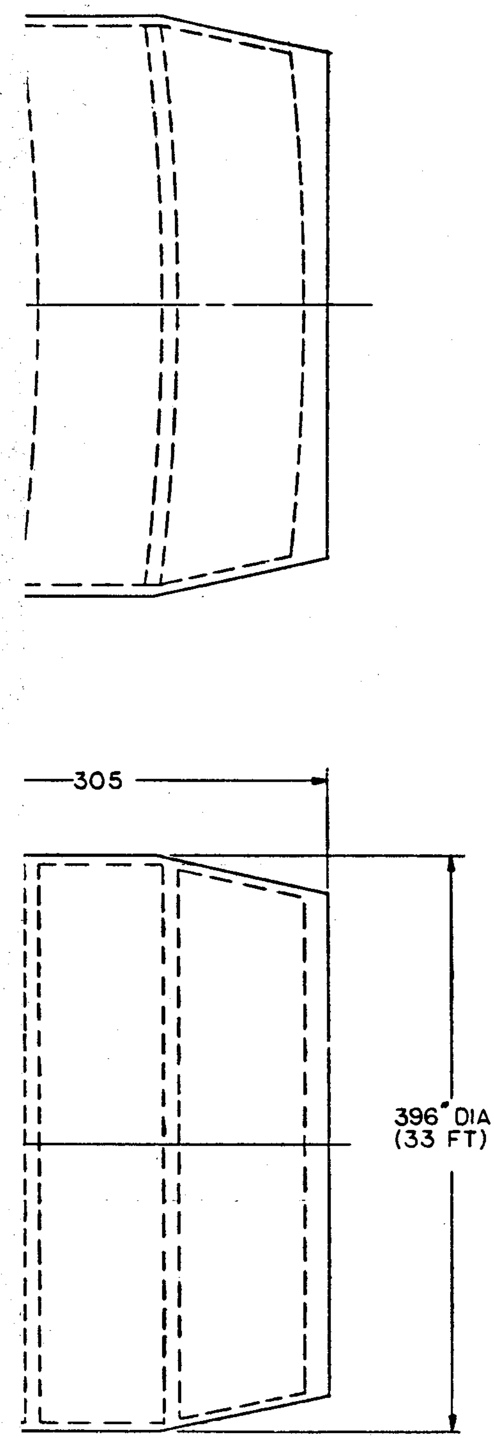

COUNTER-ROTATED
ZERO "G" MODULE/LAB (15FT
(EARTH RESOURCES/METEORC

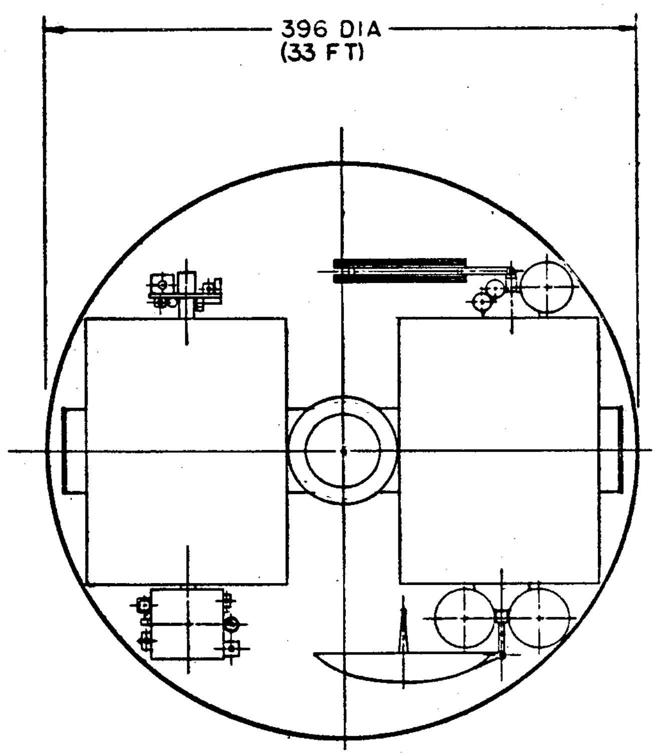

STRONOMY SENSORS
STABILIZING WEIGHT (2)
EARTH SENSORS
40" TELESCOPE
PLAN VIEW
ORBITAL CONFIG
EARTH SENSORS
X-RAY ARRAY
SPIN AXIS
DOCKING PORT
R. F. PARABOLIC DISH
A)
GY/R. AND D.)
SOLAR PANEL (2)
(TOTAL AREA
4400 FT²)
GISTICS VEHICLE


125 FT
SPIN RADIUS
(200 FT + FEASIBLE)
CREW TRANSFER TUNNEL
ARTIFICIAL GRAVITY MODULE



PLAN VIEW
(SCALE: 1/500)





SECTION A-A

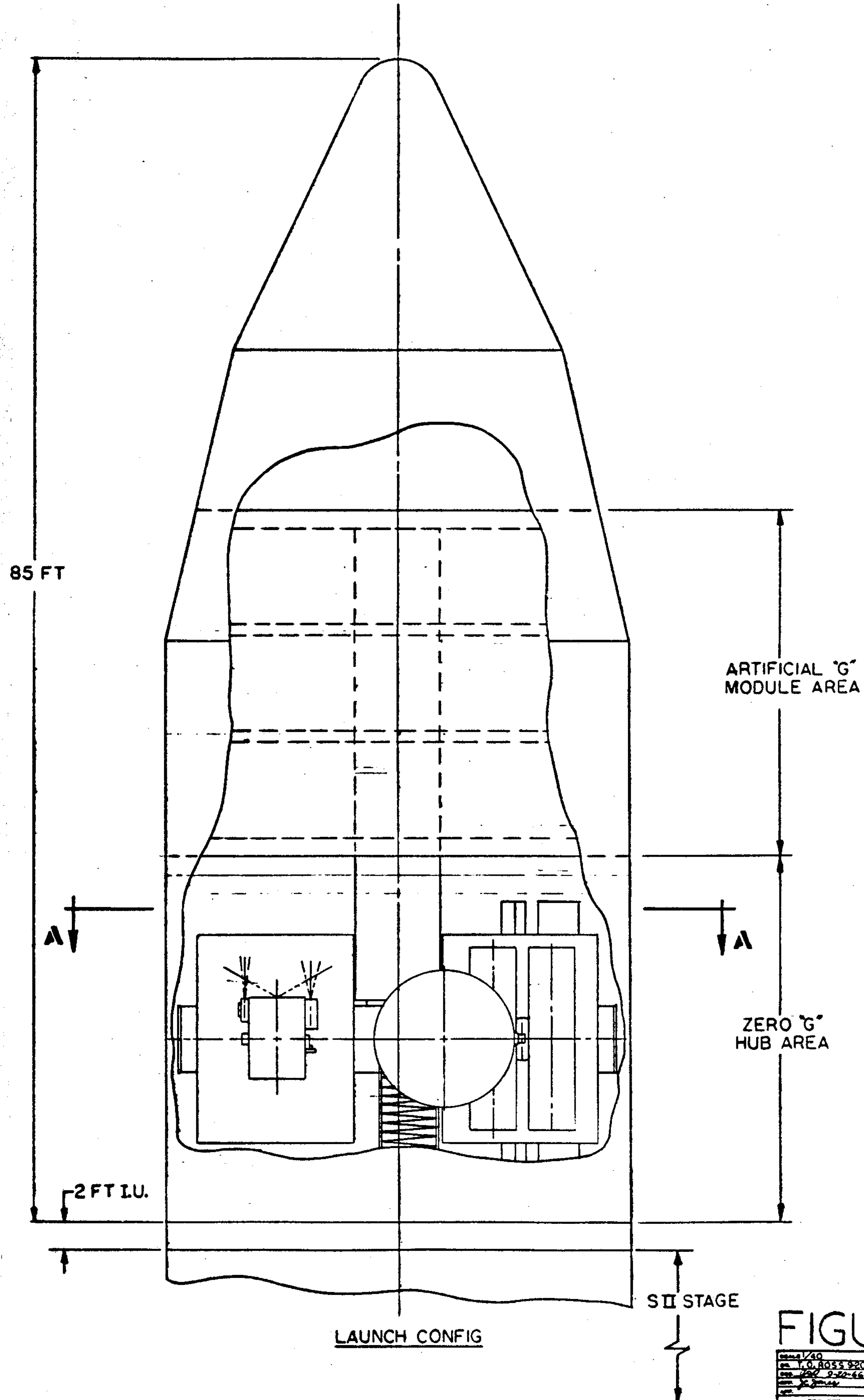


FIGURE 4.12

SCALE 1/40	MANNED SPACECRAFT CENTER HOUSTON, TEXAS
DR. T. O. ROSS 920	CONCEPT - ARTIFICIAL "G" SPACE STATION CONFIG #3
CHK. 9-20-66	
PROJECT	PD6-21003

BASIC LAUNCH CONFIGURATION ARRANGMENTS FOR I CONFIGURATION ARTIFICIAL GRAVITY SPACE STATION

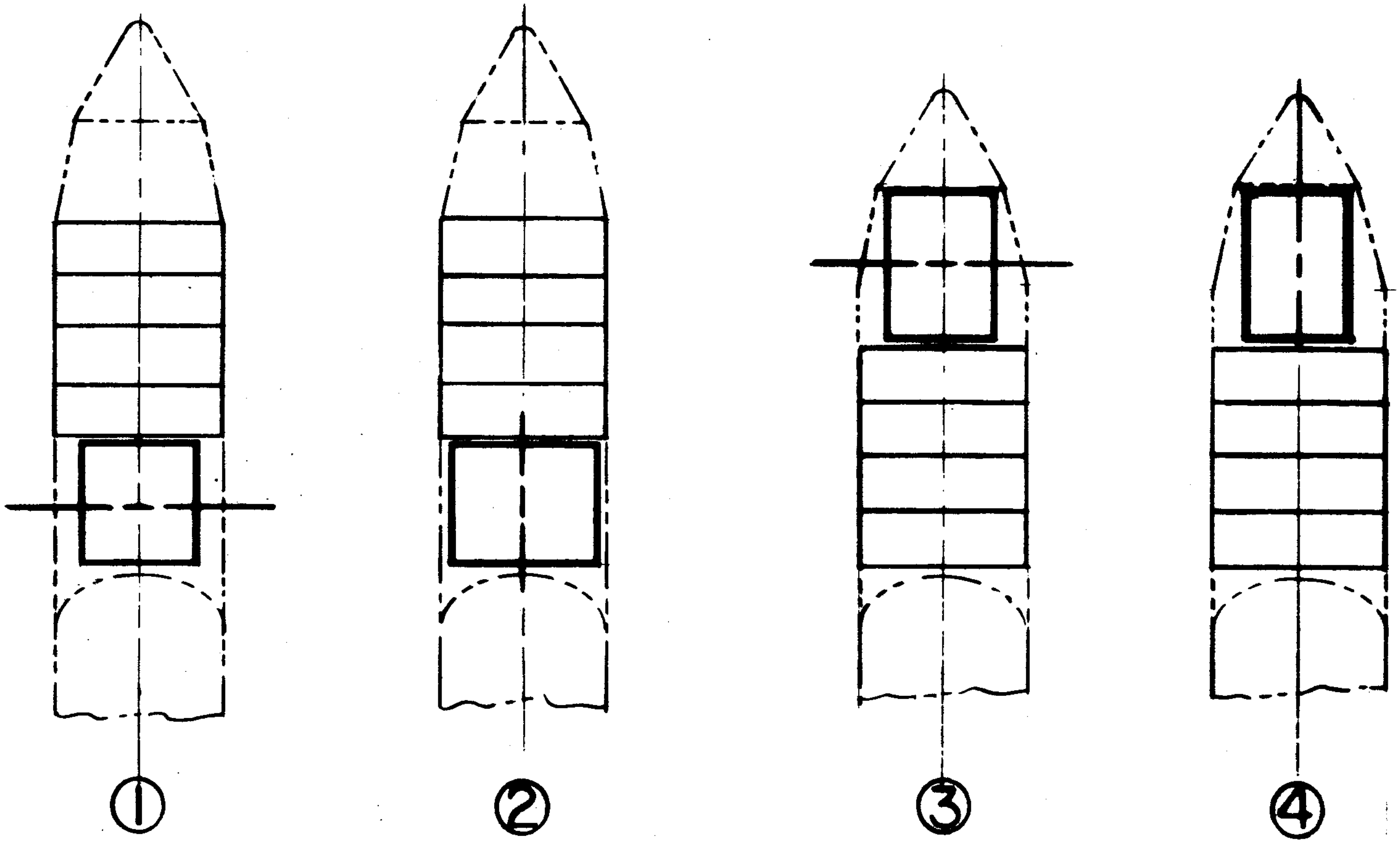


FIGURE 4.13

radius. An expandable truss could be used to connect the counterweight to the hub. The weight of a ballast counterweight can be traded off versus the additional orbit-keeping propellant caused by the added aerodynamic drag of the S-II, plus the reduction in payload caused by transferring the S-II stage into the final desired orbit. A considerable gain in payload can be realized by using the two-stage Saturn V to inject the station into a parking orbit of approximately 100 n.m. altitude and providing a "kick" stage to perform a Hohmann transfer maneuver to obtain the desired operational altitude. Thus, an overall payload gain can result by leaving the S-II stage in the parking orbit. This trade-off is shown in Figure 4.14. Spin-up and attitude control reaction thrusters should be located on the counterweight, however, to reduce propellant requirements. Because of the possible usefulness of the large volume of the spent S-II stage and the additional complexity involved, the ballast counterweight concept was not pursued further. However, if a nuclear electric power system were employed, the concept is worthy of additional study. It is also worth noting that the use of a small amount of ballast can greatly increase the gyroscopic stability of the station.

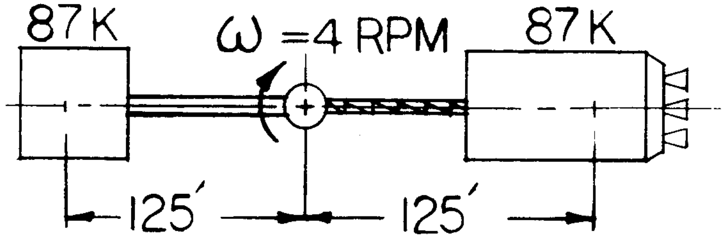
The configuration shown in Figure 4.10 is also noteworthy in that telescope pointing is accomplished with a plane mirror. The telescopes are arranged so that a single mirror can direct the light into any one of the telescopes; however, only one at a time may be operated. The mirror can also incorporate the fine pointing function to allow the telescopes to be fixed to the non-rotating hub. The earth sensors are pod mounted. This configuration is designed to be oriented with its spin axis parallel to the earth's polar axis. Therefore, the solar panels must have two axes of freedom relative to the non-rotating hub.

The configuration shown in Figure 4.11 is similar in basic arrangement to the one in Figure 4.3 except that the orientation mode is inertial (not sun oriented). Thus, the solar panels are gimbal mounted to the non-rotating hub. All module interconnect structure is stowed external to the basic 396 inch diameter. This may be undesirable when considering the aerodynamics associated with launch.

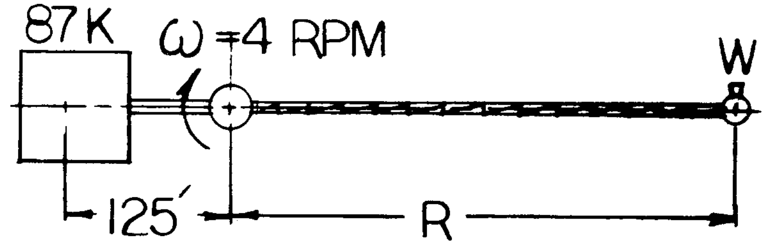
Figure 4.12 shows a configuration in which the hub is stowed between the artificial gravity module and the S-II stage at launch. Both astronomical and earth sensors are turret mounted. This configuration does not have adequate zero gravity hub volume and does not lend itself to providing adequate volume efficiently.

These three artificial gravity configurations provided background for the "I" configuration concept as depicted in Figures 4.3, 4.5, and 4.6.

ARTIFICIAL GRAVITY SPACE STATION
COUNTERWEIGHT TRADE-OFF



S-II COUNTERWEIGHT



BALLAST COUNTERWEIGHT

• HOHMAN TRANSFER OF S-II STAGE 100 N. MI. → 260 N.MI.	5,900 LBS. PROP. SYS. (I_{SP} = 313)
• AERODYNAMIC DRAG (5 YEARS)	13,100 LBS. PROP. SYS. (I_{SP} = 313)
• REDUCED SPIN-UP PROPELLANT	420 LBS. PROP. (I_{SP} = 300)
• REDUCTION IN PROP. FOR SOLAR ORIENTING SPIN AXIS (5 YEARS)	13,188 LBS. PROP. (I_{SP} = 300)

TRADE-OFF = 32,608 LBS. BALLAST + BOOM

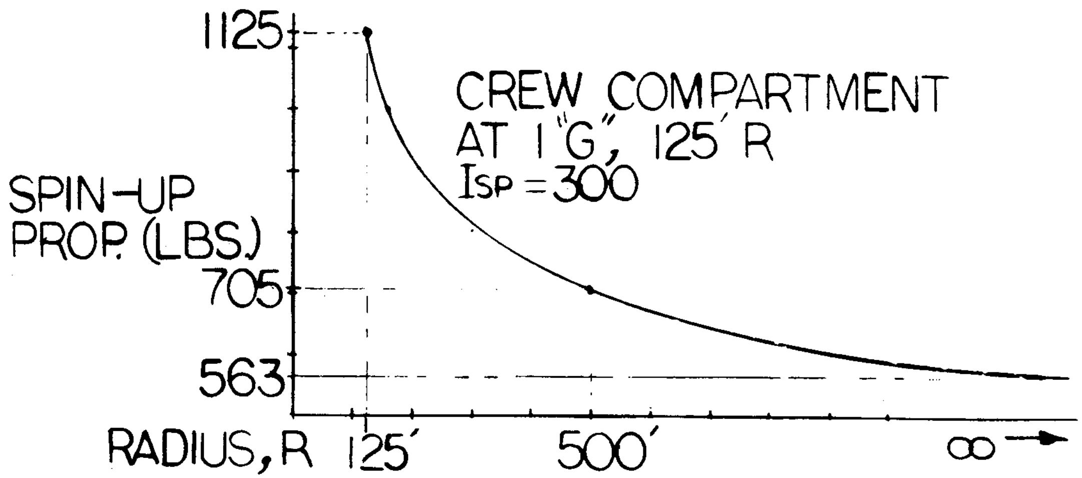


FIGURE 4.14

PRELIMINARY TECHNICAL DATA

FOR EARTH ORBITING SPACE STATION

VOLUME IV

CONFIGURATIONS, INTEGRATION, AND WEIGHTS

SECTION 5.0

SPACE STATION WEIGHTS

ADVANCED SPACECRAFT TECHNOLOGY DIVISION

ENGINEERING AND DEVELOPMENT DIRECTORATE

MANNED SPACECRAFT CENTER

5.0 SPACE STATION WEIGHTS

5.1 WEIGHT PREDICTION TECHNIQUES

5.1.1 INTRODUCTION

Many helpful weight engineering tools have been developed in the past, primarily for aircraft program use. Even in this field, however, it appears that the weight technology has lagged some of the other technologies that comprise the programs. The weight histories of recent aircraft and manned spacecraft programs point out the need for a greater effort to reduce the error of prediction.

Effective tools for manned spacecraft weight prediction are essentially nonexistent in terms of proven use. Limited use of some of the proven aircraft estimating and predicting techniques appears to offer some answers. The most apparent answer, although somewhat fundamental, is the basic approach of using (hardware) information to get reliable answers early in a program.

Although manned spacecraft information is available, the limited amount of hardware data restricts the development of reliable estimating and predicting techniques. The logical approach appears to be a blending of all sources of information, coupled with logical theory, and tempered by practical hardware data. This is the basic approach used for the space station weight predictions.

Past program weight histories indicate that the weight misprediction and/or weight growth are greatest during the earliest phases of a program. The earliest phase may be defined as that corresponding to this report.

During this phase, care must be taken to utilize the best possible tools of weight technology to prevent excessive misprediction. Error in specific configuration weight estimates plus error in the comparable weight growth estimates can combine in the study phase to cause excessive misprediction.

5.1.2 WEIGHT ESTIMATING AND PREDICTING

Estimating and predicting are discussed together to help define the difference between the two when related to weight engineering. Also, it is important that this relationship is held in proper perspective throughout the various phases of a program.

An estimated weight is defined as the weight of a spacecraft system, subsystem, or component that is based upon preliminary data such as mass fractions. A calculated weight is based upon detail design drawings. The actual weight is a measured quantity.

The estimate is made independent of time while the prediction is made with time as one of the prime parameters.

Figure 5.1 shows some generalized considerations that are important to a large manned space station. This band plot shows the status of a manned aerospace vehicle weight in relation to the first operational weight and time. A general milestone designation is used to fit most major programs. Essentially all manned aerospace vehicles have had weight histories that increase, i.e. fall within the borders of the total band. Moderately advanced, manned aircraft generally comprise the upper half of the band before operation time while manned spacecraft to date generally fall within the lower half of the band before operation time.

The total time span of Figure 5.1 is divided into two separate major phases: the definition phase and the acquisition phase. The dividing line in this case is the date of the initial contract for acquisition.

Weight estimating and predicting are prime working tools during the definition phase. Standardized means of reporting and commonly understood (between customer and contractor) weight groupings for control are techniques which will determine how soon and how nearly the shaded data band of Figure 5.1 becomes coincident with the unshaded band.

Weight estimating and predicting for the conceptual, request for proposal (RFP), and evaluation phases should reflect sufficient depth to become direct inputs to the contract. This depth is unobtainable without first determining the various weight effects during the conceptual, RFP, and evaluation phases. However, past program histories indicate that the weight engineering emphasis during this time period is almost insignificant as compared to later periods.

The RFP appears to set the pace for the destiny of the vehicle's weight and other mass properties. Therefore, the evaluation of various proposals becomes extremely important so that the proper contract weights may be negotiated. Unless the realistic tools of weight estimating and predicting are fully utilized before the contract, both the customer and the contractor are likely to start out on grounds of misunderstanding.

5.1.2.1 WEIGHT ESTIMATING

Weight estimating during the conceptual studies is subject to large error and oversight, even if the estimates are based on clear and substantial study guidelines. The biggest offender that causes these large errors and oversights appears to be the various techniques used to estimate. The shaded double arrow of Figure 5.2 represents, for the most part, this error of estimation.

FIGURE 5.1
GENERALIZED CONSIDERATIONS
AEROSPACE VEHICLE WEIGHT
(MANNED)

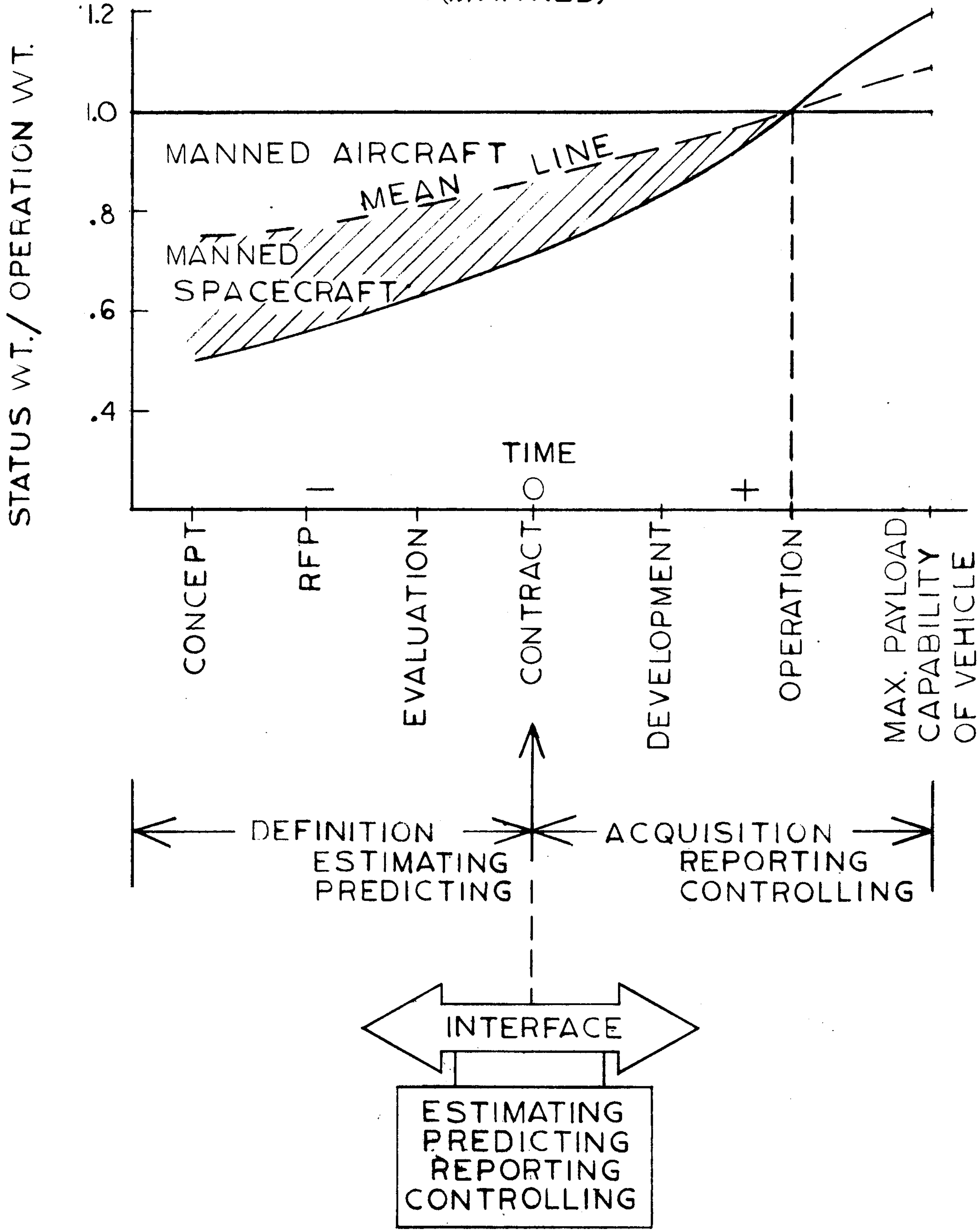
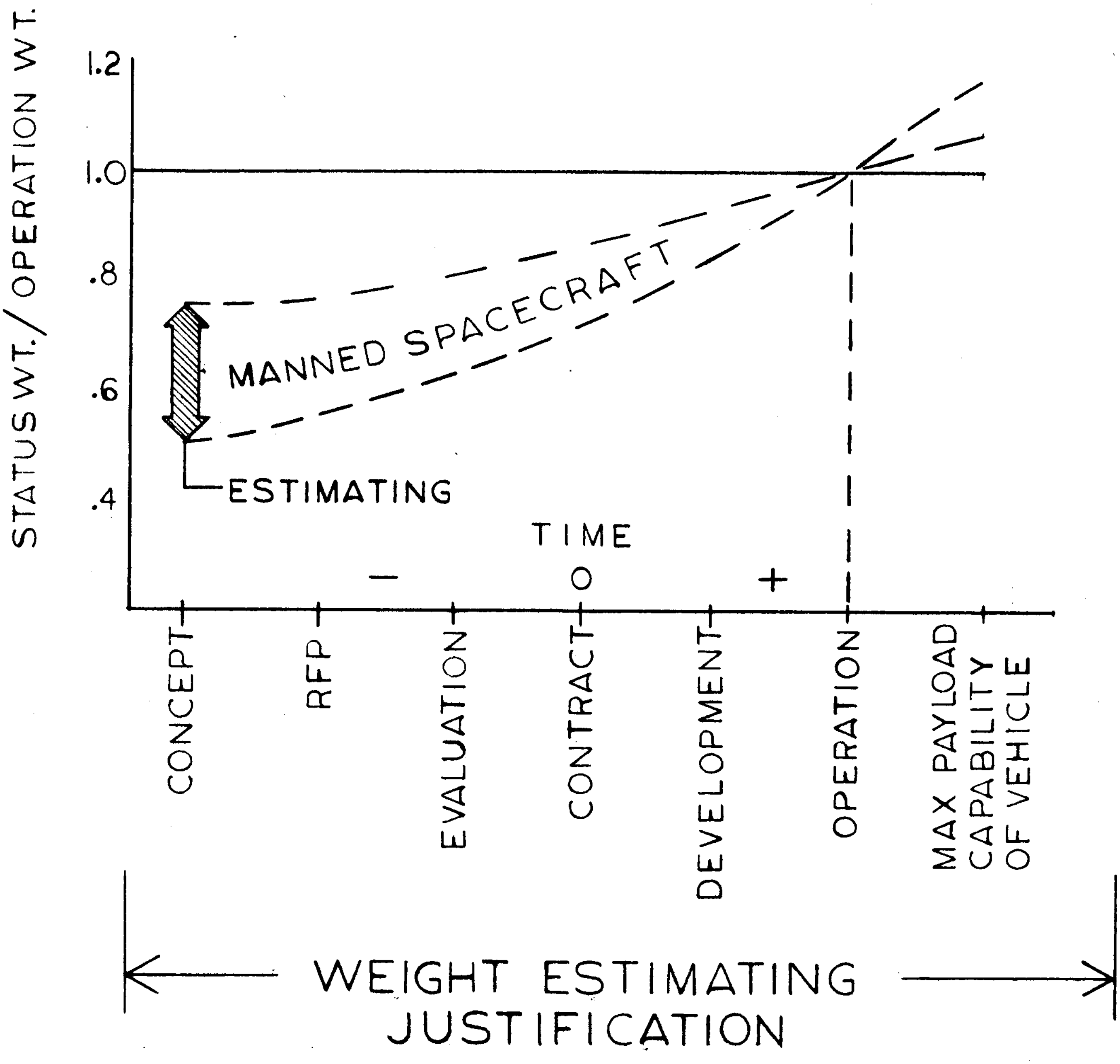


FIGURE 5.2
ESTIMATING CONSIDERATIONS
MANNED SPACECRAFT



(HARDWARE DATA UTILIZATION)
(TECHNIQUE DEVELOPMENT)
(FUNCTIONAL REPORTING)

A proper justification for the various weight estimating techniques should be a primary goal for the customer in-house conceptual studies. The RFP should be the ideal point of introduction for a requirement of justification from the contractor. The point to be made here is that there is an apparent large variation in contractor concern and/or capability for weight estimating, not to mention weight prediction.

A large variation of contractor estimated weight exists for most proposed systems and especially subsystems. Although the reasons for this are many, including competitive bias, it is felt that the primary reason is the various means of obtaining and justifying these estimates.

An example of variation in system weight is shown in Figure 5.3. This is a band plot of various body structure unit weights against body area. Over 50 hardware data points were used to obtain the band between Curve A and Curve C. The majority of the data points represents fighters, re-entry vehicles, bombers, and transports. Curve A represents various adapters, fairings, and booster segments.

If a structure unit weight is used or proposed that falls below Curve A for a manned spacecraft, justification should be required. Several past study and proposal weights do fall below Curve A as shown in Figure 5.3. This type of apparent mis-estimation must be sought out early in the definition phase to preclude starting on the bottom of the band plot shown in Figure 5.2.

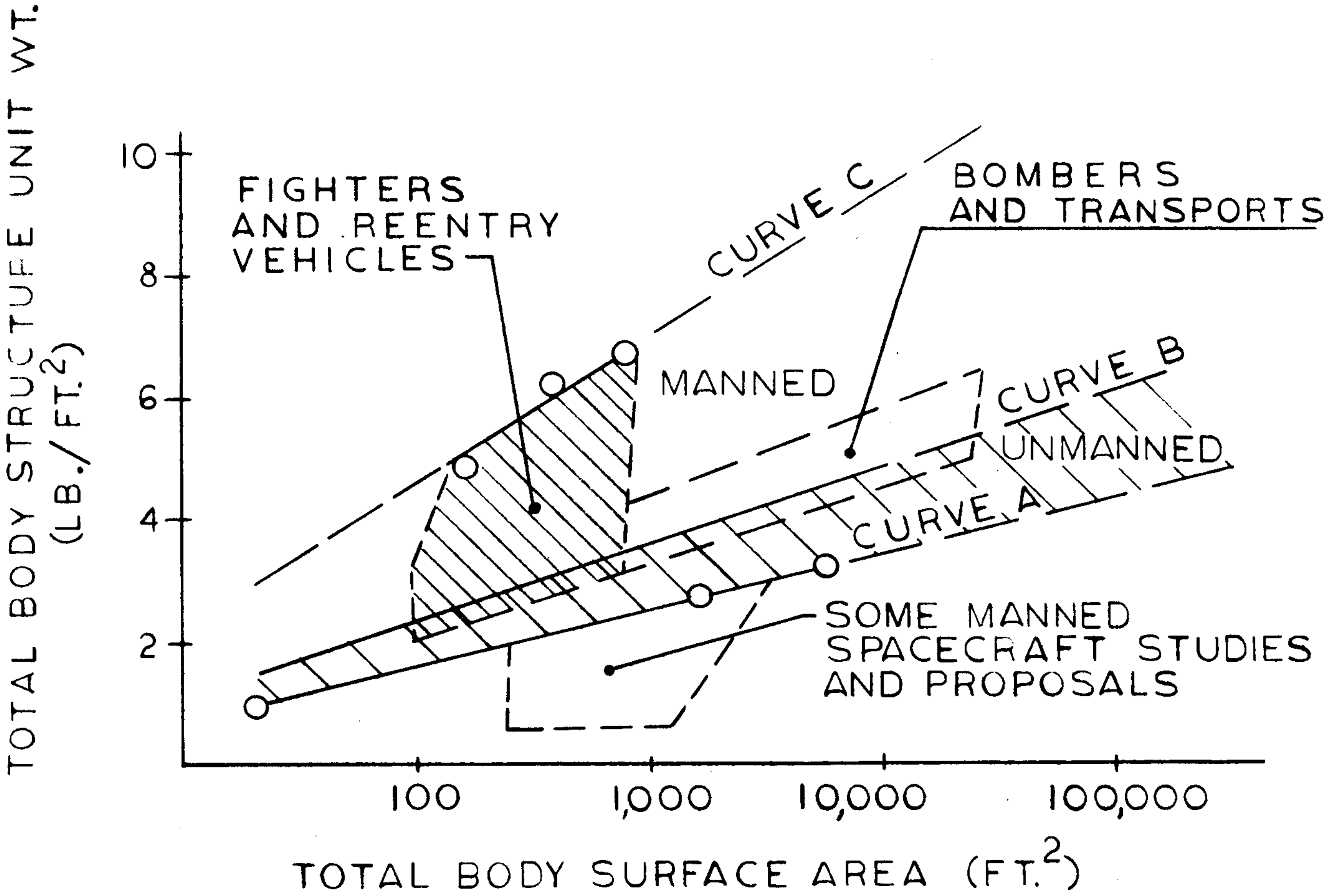
The fact that some study and proposal unit weights do fall below Curve A of Figure 5.3 does not rule out the possibility that these weights can be substantiated by analytical means. However, the analytical data must be confirmed by hardware statistics.

Figure 5.4 is a plot that involves hardware statistics. The actual weight of various sidewall structures is plotted against the estimated weight. The estimating technique used is common to all the data points shown and includes the following dependent parameters:

Material strength and density
Bending moment
Axial load

The mean value of the data points of Figure 5.4 is 80 percent greater than the "actual equals estimated" values. This means that estimated weights should be multiplied by 1.8 (non-optimum factor) to obtain weights in the hardware category. It is important to note the difference between the 1.8 non-optimum factor above and some of the frequently used 1.05 and 1.10 factors found in the various detailed, theoretical analysis approaches to weight estimating.

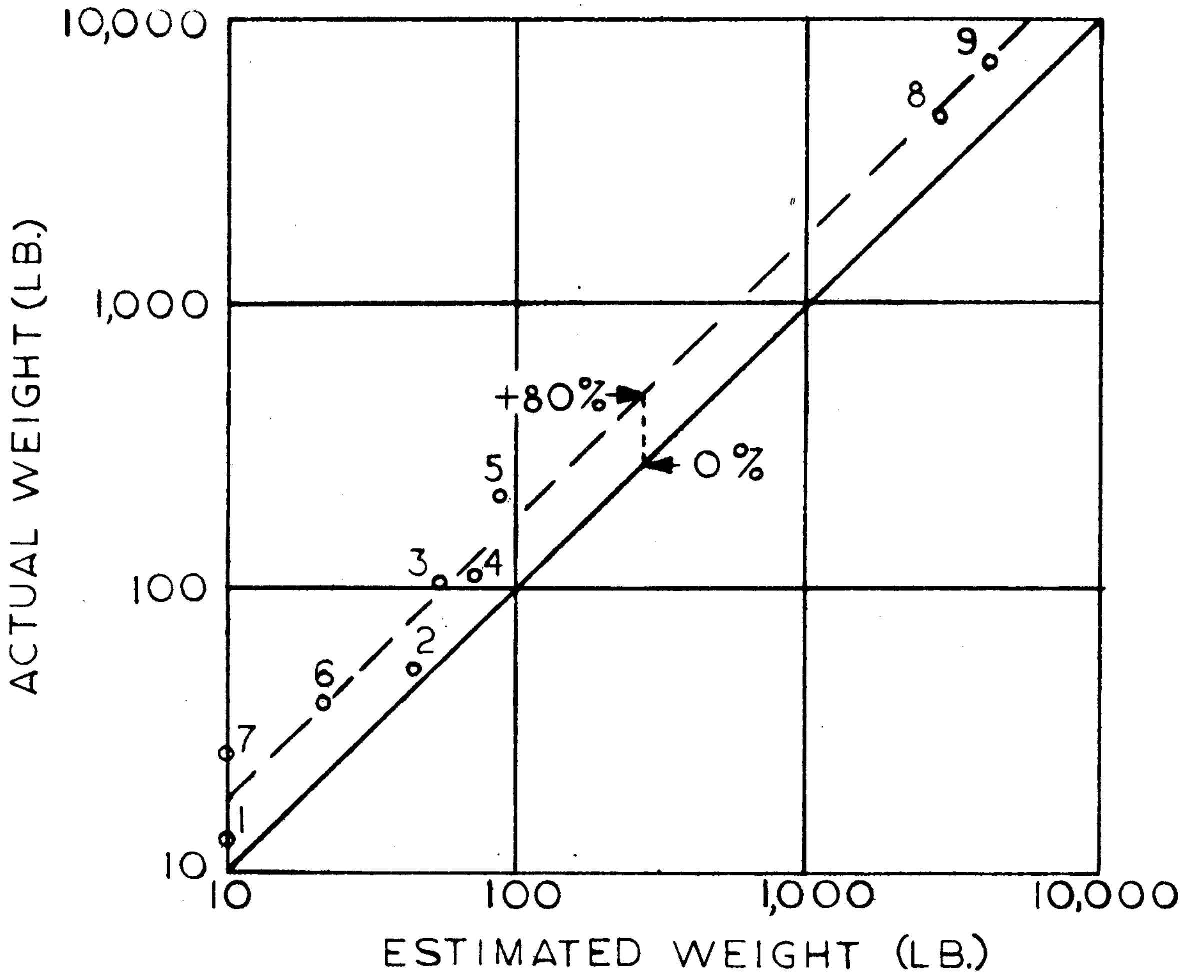
FIGURE 5.3
ESTIMATING CONSIDERATIONS
EXAMPLE — BODY STRUCTURE SYSTEMS



WEIGHT ESTIMATING JUSTIFICATION

(HARDWARE DATA UTILIZATION)
(TECHNIQUE DEVELOPMENT)
(FUNCTIONAL REPORTING)

FIGURE 5.4
ESTIMATING CONSIDERATIONS
EXAMPLE—BODY SIDEWALL STRUCTURE



DATA POINTS:

1. MERCURY FWD. CYLINDER
2. MERCURY CONE
3. MERCURY ADAPTER
4. GEMINI CONE
5. GEMINI ADAPTER
6. GEMINI R&R SECTION
7. GEMINI RCS SECTION
8. TITAN II (STA. 300-1200)
9. TITAN III (STA. 405-1210)

It is recognized that a large portion of the weight of any newly conceived spacecraft must be estimated by theoretical methods. Nevertheless, every attempt should be made to include the effects of hardware information wherever and whenever possible. It is important to point out that the trend in future spacecraft weight reporting will largely determine the effectiveness of the tools developed for the definition phase weight estimating and predicting.

5.1.2.2 WEIGHT PREDICTING

Weight predicting, as defined earlier, is based on the estimated weight but includes separate allowances for weight changes with time. The shaded arrow in Figure 5.5 shows a general range of predicting that should be considered in the definition phase. It should be remembered that definition also improves with time. Therefore, estimating accuracy should be thoroughly stressed during the definition phase.

Several additional major factors should be considered when developing the techniques for weight prediction. The first factor involves environment and/or state-of-the-art. Figure 5.6 shows the weight history of seven vehicles during the acquisition phase. There is noticeable difference in the overall slopes for vehicles that have faced advanced increases in the environment and/or state-of-the-art as opposed to those that have faced moderate increases.

The second factor involves experience gained on vehicle design. Figure 5.7 indicates that first generation vehicles have a higher growth rate than second generation vehicles. For example, the Gemini spacecraft tends to follow the second generation curve; whereas, the Mercury and the Apollo spacecraft tend to follow the first generation curve.

The weight implications from Figures 5.6 and 5.7 are difficult to predict at the start of the acquisition phase. They are more difficult to predict during the conceptual studies. Nevertheless, certain bands of information are beginning to appear and at least these should be analyzed for the best possible utilization.

5.1.3 WEIGHT REPORTING AND CONTROLLING

Reporting and controlling are discussed together because weight reporting is the baseline for any control program just as estimated weight is the baseline for prediction. There is a binding interface between all four of these disciplines of weight engineering.

5.1.3.1 WEIGHT REPORTING

Functional reporting, breakdowns that comprise the total functional system or subsystem, continues to be the most usable approach to weight engineering in advanced design. The other various disciplines of engineering that contribute do not necessarily follow

FIGURE 5.5
PREDICTING CONSIDERATIONS
MANNED SPACECRAFT

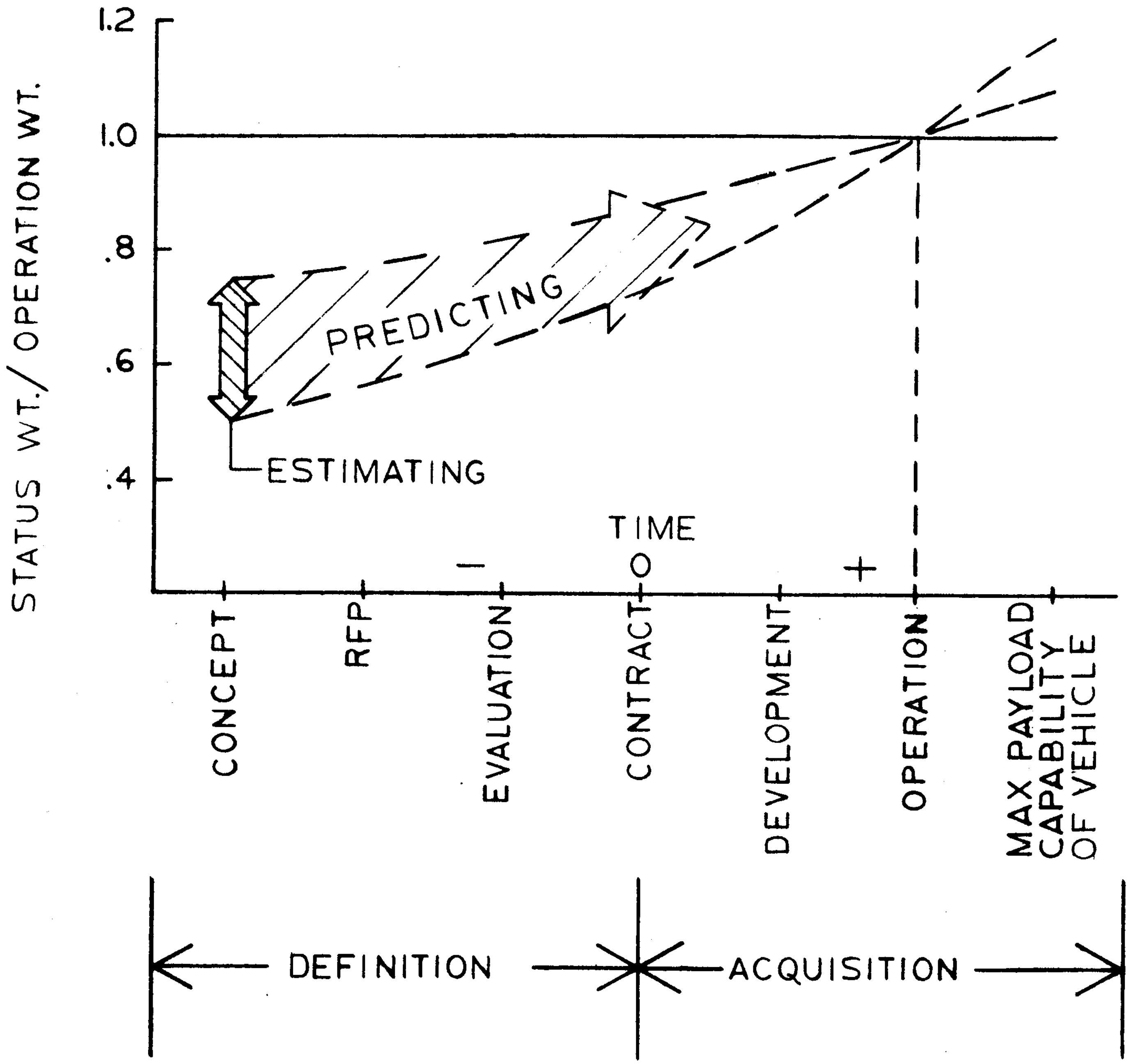


FIGURE 5.6

WEIGHT GROWTH VS. TIME

FOR INCREASE IN ENVIRONMENT AND/OR STATE-OF-THE-ART

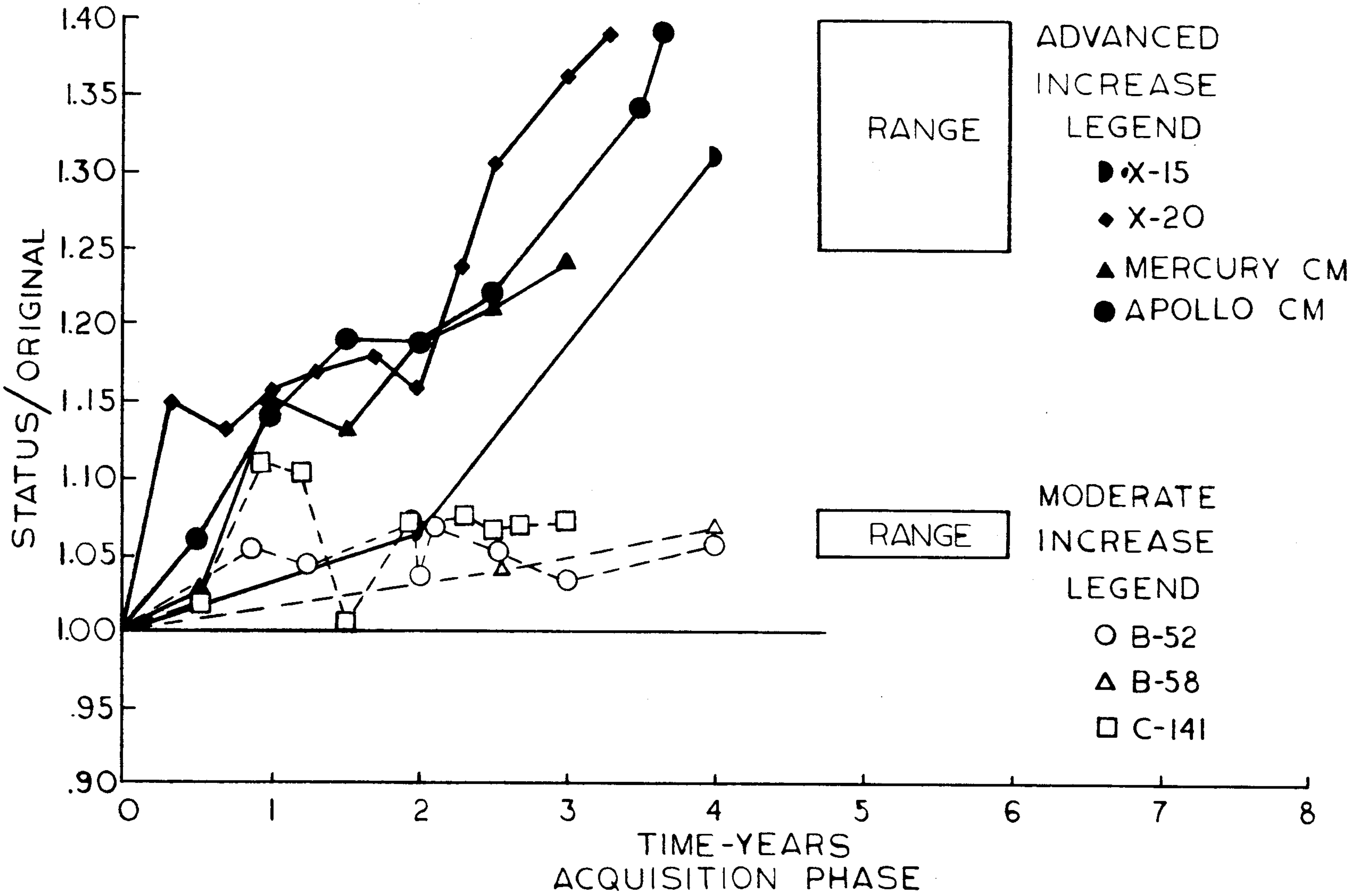
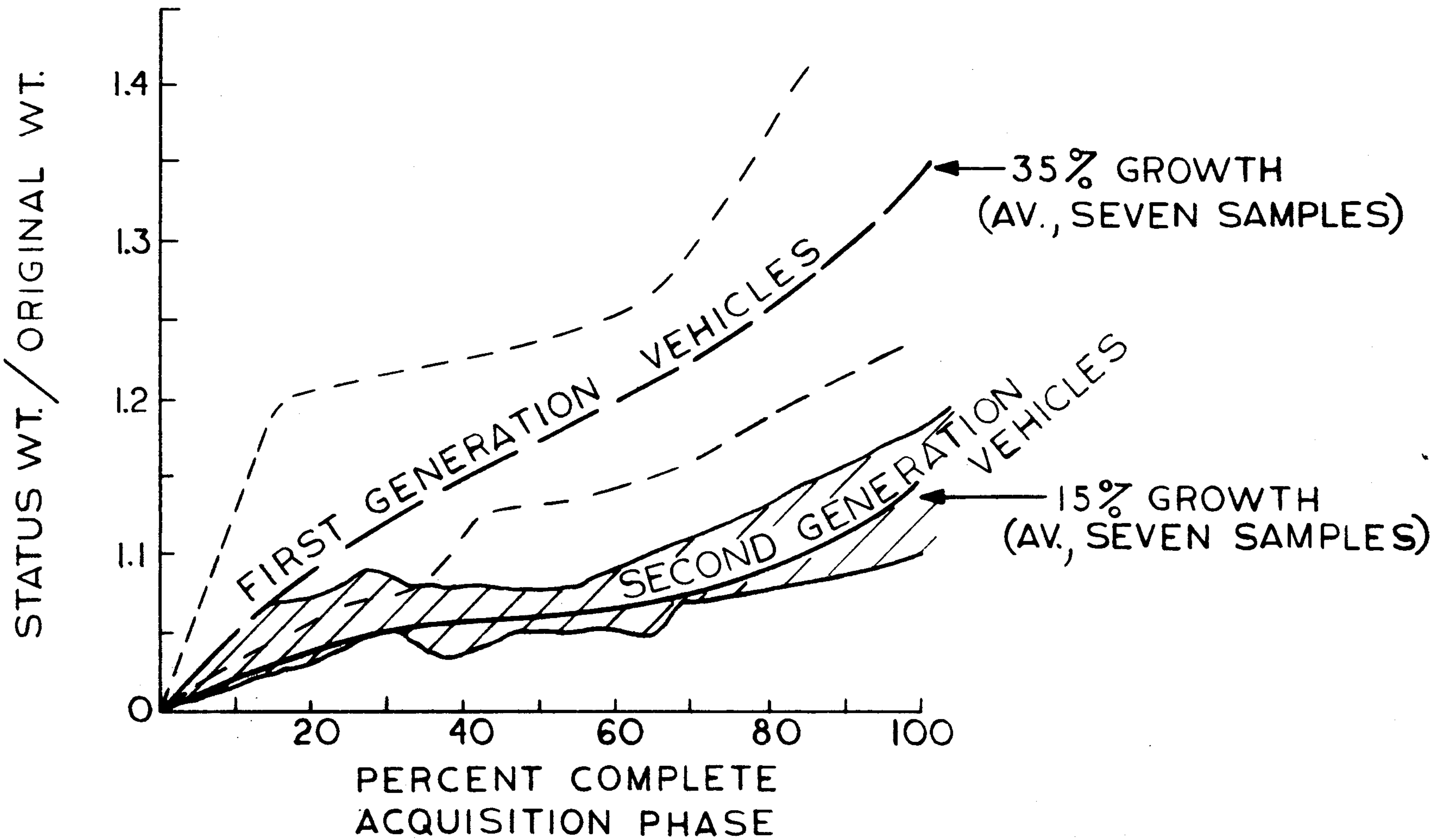


FIGURE 5.7

WEIGHT GROWTH VS. TIME
FOR FIRST AND SECOND GENERATION VEHICLES



the same approach. This leads to an immediate problem of reporting and accountability long before the acquisition phase begins.

The weight accountability should be handled by an early listing of responsibility from a functional as well as a design breakdown as shown in Table 5.1. The functional breakdown is the responsibility of design integration while the design breakdown is the area of the system specialist. A technique for accountability is illustrated in Table 5.1. For instance, in the category of body structure, the question is where does the responsibility lie for equipment mounting provisions, with the structural specialist or the equipment designer. In the case of electrical power systems, a questionable item is the wiring. As long as the flow of responsibility is both vertical and horizontal as shown in Table 5.1 the likelihood of ommissions or double changes is reduced.

5.1.3.2 WEIGHT CONTROLLING

Weight controlling, although generally recognized as occurring primarily during the acquisition phase, actually begins during the definition phase - more often inadvertently than by planning. The definition phase sets the tone and the pace for the type and the amount of controlling to be done. The fundamental considerations of Table 5.1 are the basic requirements for early weight control. Weight reporting is the focal point for effective weight estimating and controlling.

A great deal can be learned about weight control in the definition phase by applying some specific control efforts used in the acquisition phase of previous programs. In recent years it has become necessary to instigate special weight control efforts to meet some primary requirements of various aircraft and spacecraft. Unfortunately, this type of effort is usually initiated after the goals and requirements are well established. The cost, in dollars and schedule, is greatest at this point in time for achieving improvements in the weight, reliability, and performance. An earlier, less costly application of this effort is not altogether impossible. In fact, many of these efforts should become common working tools in the development of good weight engineering throughout the total program.

5.1.4 SUMMARY

Instilling the philosophy of effective weight engineering is perhaps the greatest single obstacle that blocks the paths to effective achievement. If the overall philosophy does not recognize weight engineering as an important discipline, it is most likely to cause later problems.

TABLE 5.1

REPORTING AND CONTROLLING CONSIDERATIONS

RESPONSIBILITY		RESPONSIBILITY					
		DESIGN BREAKDOWN (Subsystems)					
FUNCTIONAL BREAKDOWN (Systems/Subsystems)	Struct. Design	Electronic Design	Propulsion Design	Electrical Design	Crew Systems Design	FUNCTIONAL TOTALS (Status and Goals)	
BODY STRUCTURE							
Primary Struct.	✓	—	—	—	—		
Secondary Struct.	✓ ?	?	?	?	?		
Mounting Prov.	?	?	?	?	?		
ELECTRICAL POWER SYSTEM							
Power Generation	—	—	—	✓	—		
Power Distribution	—	✓?	—	✓ ?	—		
Wiring	?	?	?	?	?		
	DESIGN TOTALS (Status and Goals)					Cross Check TOTAL	

5.2 CONFIGURATION WEIGHT DATA

5.2.1 CONFIGURATION WEIGHT SUMMARY - INITIAL LAUNCH

A preliminary weight analysis has been performed in a generalized manner in order to determine the magnitude of the weight and mass properties for the "zero" and artificial gravity space stations. Table 5.2 is an estimated summary weight comparison between a nine man "zero" gravity station, Figure 4.1;9 man rotating station, Figure 4.3;24 man rotating station, Figure 4.6; and the MORL study. This table presents several parameters that influence the weight estimates and lists gross weights for experiments, subsystems, expendables, emergency vehicles, structure and a summation of these weights for a 3 month resupply interval. The counterweight is not included since the comparison is between initial launch weights. Table 5.3 is a similar table based on a 6 months resupply interval. Both tables include a 50 percent margin for expendables and subsystem weight.

A brief paragraph which discusses each category of weight shown in these tables follows.

5.2.1.1 EXPERIMENTS

The experiments weights have been developed by MSFC with the exception of the long term flight experiments. The 9 man crew experiment weight consists of 3400 pounds of long term flight experiments and 42,900 pounds for MSFC integrated experiment weight. It is assumed that MSFC weight includes the experiments and supporting equipment. The primary and secondary structure of the experiments module is included in the structure weight which is discussed in paragraph 5.2.1.4.

5.2.1.2 SYSTEM AND EXPENDABLE WEIGHT

The system weights presented in Volume III have been estimated by the system specialists. Experience gained from Mercury, Gemini, and Apollo programs in conjunction with aircraft data indicates that even though effort is expended to keep the system weight in line with early estimates, the system weight increases because of omissions, system integration effects, management decisions, cost and performance considerations. Tables 5.4 and 5.5 contain a weight breakdown of the systems discussed in Volume III for 9 and 24 man crews with resupply intervals of 3 and 6 months. Increments of weight have been added at the component level to account for the definition phase weight growth. These tables, also, include a preliminary approach to allocating the system weight to the rotating and non-rotating portion of the typical configurations (Figure 4.3 and 4.6).

The weight impact of increasing the number of crew men is about 20 percent greater than increasing the resupply interval. Doubling the resupply interval affects the expendables, tankage

TABLE 5.2

CONFIGURATION WEIGHT SUMMARY - INITIAL LAUNCH

	ZERO "G"	ART. "G"	ART. "G"	MORL
PARAMETERS				
NUMBER OF MEN	9	9	24	9
*RESUPPLY INTERVAL (MONTHS)	3	3	3	1
BASIC DIAMETER (FEET)	22	22	33	22
PRESSURIZED VOLUME (Ft^3)	30,000	30,000	44,000	10,000
ZERO "G" VOLUME (Ft^3)	41,500	10,000	14,000	15,000
LAUNCH ENVELOPE VOLUME (Ft^3)	41,500	41,500	53,400	18,000
ELECTRICAL POWER LEVEL (KWe)	15	15	25	11
WEIGHTS				
EXPERIMENTS (Lbs)**	46,300	46,300	63,100	470
SUBSYSTEMS (Lbs)	28,000	28,000	45,300	15,900
EXPENDABLES (Lbs)	13,900	13,900	24,300	2,300
EMERGENCY VEHICLES (Lbs)	9,000	9,000	24,000	--
STRUCTURE (Lbs)	62,300	95,500	107,000	13,850
TOTAL (LBS)	159,500	192,700	263,700	32,520

* 3-MONTH RESUPPLY INTERVAL CONSIDERS 50 PERCENT MARGIN FOR SUBSYSTEMS ON INITIAL LAUNCH.

** INCLUDES LONG TERM FLIGHT EXPERIMENTS (3400 POUNDS) - NOT APPLICABLE TO MORL.

TABLE 5.3

CONFIGURATION WEIGHT SUMMARY - INITIAL LAUNCH

PARAMETERS	ZERO "G"	ART. "G"	ART. "G"	MORL
Number of Men	9	9	24	9
*Resupply Interval (Months)	6	6	6	1
Basic Diameter (Feet)	22	22	33	22
Pressurized Volume (Ft^3)	30,000	30,000	44,000	10,000
Zero "G" Volume (Ft^3)	41,500	10,000	14,000	15,000
Launch Envelope Volume (Ft^3)	41,500	41,500	53,400	18,000
Electrical Power Level (KWe)	15	15	25	11
WEIGHTS				
Experiments (Lbs)**	46,300	46,300	63,100	470
Subsystems (Lbs)	33,700	33,700	54,300	15,900
Expendables (Lbs)	24,900	24,900	44,900	2,300
Emergency Vehicles (Lbs)	9,000	9,000	24,000	--
Structure (Lbs)	63,800	97,000	109,000	13,850
TOTAL (Lbs)	177,700	210,900	295,300	32,520

* 6-Month resupply interval considers 50% margin for subsystems on initial launch.

** Includes long term flight experiments (3,400 pounds) - not applicable to MORL.

TABLE 5.4

SYSTEM AND EXPENDABLE WEIGHT - LAUNCH WEIGHT 3-MONTH RESUPPLY INTERVAL

	9-Man System Weight			24-Man System Weight		
	Total	Non-Rotating	Rotating	Total	Non-Rotating	Rotating
EPS	(11330)	(1900)	(9430)	(18100)	(2800)	(15300)
Solar Cell Array	4800		4800	8000		8000
Power Conditioning and Control	830	400	430	1400	600	800
Wiring	4500	1500	3000	6700	2200	4500
Instrumentation	(500)	(200)	(300)	(650)	(270)	(380)
Comm. & Data Mgt.	(1500)	(610)	(890)	(1560)	(640)	(920)
R.F.	300	150	150	300	150	150
Terminal Equipment	20		20	20		20
Data Mgt.	160	100	60	160	100	60
Data Storage	610	210	400	600	210	400
Audio & PMP	80	40	40	140	70	70
TV	330	110	220	330	110	220
EC/LSS	(6220)	(3710)	(2510)	(12880)	(7440)	(5440)
Atmos. Regen.	180	90	90	540	270	270
CO_2 Removal	360	180	180	1080	540	540
Cabin Circ. Loop	240	120	120	480	240	240
Coolant Loop & Radiators	4160	2680	1480	6940	4470	2470
Water Supply	800	400	400	2400	1200	1200
Waste Mgt.	480	240	240	1440	720	720
Crew Systems	(1800)	(600)	(1200)	(4600)	(1200)	(3400)
RCS	(2700)		(2700)	(2750)		(2750)
Tanks & Press.	2550		2550	2600		2600
Thrusters, Etc.	150		150	150		150
G&C	(1300)	(1000)	(300)	(1300)	(1000)	(300)
CMG	1000	1000		1000	1000	
Electronics	300		300	300		(300)
Cryogenic Tankage	(2640)	(2640)		(3450)	(3450)	
O_2	460	460		1010	1010	
N_2	2180	2180		2440	2440	
Expendables	(13870)	(8260)	(5610)	(24240)	(14960)	(9280)
O_2	3380	3380		7370	7370	
N_2	3570	3570		4130	4130	
Food	2680	600	2080	7130	1580	5550
Misc. EC/LSS	140	70	70	380	190	190
Plss H_2O	430	430		1140	1140	
Plss LiOH	210	210		550	550	
RCS Propellant	3460		3460	3540	3540	
	(41860)	(18920)	(22940)	(69530)	(31760)	(37770)
	13870	8260	5610	24240	14960	9280
	27990	10660	17330	45290	16800	28490

TABLE 5.5
SYSTEM AND EXPENDABLE WEIGHT - LAUNCH WEIGHT
6-MONTH RESUPPLY INTERVAL

	9-Man System Weight			24-Man System Weight		
	Total	Non-Rotating	Rotating	Total	Non-Rotating	Rotating
EPS	(11330)	(1900)	(9430)	(18100)	(2800)	(15300)
Solar Cell Array	4800		4800	8000		8000
Batteries	1200		1200	2000		2000
Power Conditioning and Control	830	400	430	1400	600	800
Wiring	4500	1500	3000	6700	2200	4500
Instrumentation	(500)	(200)	(300)	(650)	(270)	(380)
Comm. & Data Mgt.	(1500)	(610)	(890)	(1560)	(640)	(920)
R.F.	300	150	150	300	150	150
Terminal Equipment	20		20	20		20
Data Mgt.	160	100	60	160	100	60
Data Storage	610	210	400	610	210	400
Audio & PMP	80	40	40	140	70	70
TV	330	110	220	330	110	220
EC/LSS	(6220)	(3710)	(2510)	(12880)	(7440)	(5440)
Atmos. Regen.	180	90	90	540	270	270
CO_2 Removal	360	180	180	1080	540	540
Cabin Circ. Loop	240	120	120	480	240	240
Coolant Loop & Radiators	4160	2680	1480	6940	4470	2470
Water Supply	800	400	400	2400	1200	1200
Waste Mgt.	480	240	240	1440	720	720
Crew Systems	(3600)	(1200)	(2400)	(9200)	(2400)	(6800)
RCS	(4780)		(4780)	(4900)		(4900)
Tanks & Press.	4630		4330	4750		4750
Thrusters, Etc.	150		150	150		150
G&C	(1300)	(1000)	(300)	(1300)	(1000)	(300)
CMG	1000	1000		1000	1000	
Electronics	300		300	300		300
Cryogenic Tankage	(4450)	(4450)		(5720)	(5720)	
O_2	350	250		1920	1920	
N_2	3600	3600		3800	3800	
Expendables	(24870)	(14290)	(10580)	(44890)	(27000)	(17890)
O_2	6240	6240		14050	14050	
N_2	5430	5430		6000	6000	
Food	5360	1200	4160	14260	3190	11070
Misc. EC/LSS	280	140	140	760	380	380
Plss H_2O	860	860		2280	2280	
Plss LiOH	420	420		1100	1100	
RCS Propellant	6280		6280	6440		6440
	(58550)	(27360)	(31190)	(99200)	(47270)	(51930)
	24870	14290	10580	44890	27000	17890
	(33680)	(13070)	(20610)	(54310)	(20270)	(34040)

and the crew system weights. The expendables contribute approximately two-thirds of the weight increase. Increasing the crew size from 9 to 24 men affects all systems but the Guidance and Control system. Depending upon the resupply interval, the expendable weight for a 24 man crew increases by approximately 75 to 80 percent of that for a 9 man crew. The system weight for a 24 man crew increases by approximately 66 to 69 percent of that for a 9 man crew. The major hardware weight increases occur in the electrical power system, environmental control and life support system and the tankage for RCS, oxygen and nitrogen. The major expendable weight increases occur in food, oxygen, nitrogen, RCS propellant and Portable Life Support System (PLSS) recharges (water and LiOH).

5.2.1.2.1 System Weight Changes

Each system will be discussed in the following sections. The basic assumptions are as follows:

a. The inert systems weight is independent of resupply interval with the exception of liquid stages.

b. Reliability and maintenance considerations are included by utilization of redundant components or spare components.

c. Current technology is utilized where possible.

5.2.1.2.2 Electrical Power System

5.2.1.2.2.1 Solar Cell Array

The electrical power requirement for a 9 man crew is 15kw and is 25kw for a 24 man crew. The solar cell array is assumed to weigh 320 pounds per kilowatt.

5.2.1.2.2.2 Batteries

Since batteries are used for peaking loads and when the solar cells are not generating, it is assumed that the battery weight will be increased in a ratio with the power requirements. The battery system weighs approximately 80 pounds per kilowatt.

5.2.1.2.2.3 Power Conditioning and Control

These components (inverters, regulators, battery chargers, controls, sequencers, and synchronizers) are assumed to be dependent upon the power requirement and for the 24 man crew will be approximately 1.67 times the estimated weight for the 9 man crew power conditioning and control. Estimated weight with growth for a 9 man crew is 830 pounds; thus, the estimated weight with growth for the 24 man crew is 1400 pounds.

5.2.1.2.2.4 Wiring

The electrical wire weight of 3300 pounds in Volume III, Section 3.0 is based on the Apollo CSM weight of 220 pounds per connected KW. The space station is a much larger vehicle than the CSM and the equipment inherently will require longer wiring runs. If an artificial gravity station is the selected configuration (Figure 4.3), at least 75 feet of additional cable will be required to service both the hub and the living quarters. It is estimated that a 9 man station will require 4500 pounds of wiring. For a 24 man station, 6700 pounds of wiring is required.

5.2.1.2.3 Instrumentation

The 9 man station equipment consists of measurement transducers, signal conditioners, display and control systems, caution and warning systems, timing equipment, event timers, and lighting system resulting in 500 pounds of system weight. The 24 man station will require an additional 150 pounds resulting from additional measurement transducers (20 pounds), signal conditioners (5 pounds) display and control system (60 pounds), caution and warning system (10 pounds), timing equipment (20 pounds), event timers (15 pounds), and lighting system (20 pounds).

5.2.1.2.4 Communications and Data Management

The weight increase of 60 pounds of the 24 man system over the 9 man occurs primarily in the Audio and Premodulation Processing equipment.

The 9 man system weight of 1500 pounds is obtained by the addition of the following weights to the weight data shown in Volume III, Table 4.3.

RF	2 Pounds
Terminal Equipment	2 Pounds
Data Storage	6 Pounds
Audio and PMP	2 Pounds
Total	12 Pounds

5.2.1.2.5 Environmental Control/Life Support System

This system is considered to be one in which a high growth in weight is possible. The 9 man system is based on the system data shown in Volume III, Table 2.2, with the changes shown in Table 5.6.

The 24 man system weight is derived by using three sets of the components for Atmosphere Regeneration, CO_2 Removal, Water Supply, and Waste Management; two sets of components for the Cabin Circulation loop; and increasing the coolant loop and radiators by approximately 67 percent due to the increased power load (25kw/

TABLE 5.6

EC/LSS WEIGHT

SYSTEM	WITH CO_2 REDUCTION ESTIMATED WEIGHT	WITHOUT CO_2 REDUCTION ESTIMATED WEIGHT	WITHOUT CO_2 REDUCTION PREDICTED WEIGHT
Atmospheric Regen.	175	175	180
Carbon Dioxide Removal	420	330	360
Carbon Dioxide Reduction	220	--	--
Cabin Circulation Loop	225	225	240
Coolant Loop	4050	4050	4160
Water Supply System	750	750	800
Solid Waste Management	440	440	480
SUBTOTAL	6280	5970	6220
Hydrogen & Tank	650	--	--
Other Expendables	140	140	*
TOTAL	7070	6110	6220

*Included under Expendables in Table 5.4

15kw=1.67).

The system weight for the 24 man crew is 12,880 pounds while weight for the 9 man crew is 6220 pounds.

5.2.1.2.6 Crew Systems

This system consists of bunks, clothing, extra spacesuits, body cleansing equipment, bedding, crew personal belongings, and recreational equipment. Approximately 200 pounds per man per three months is assumed. For the 24 man crew the assumption is that approximately 192 pounds per man per three months will suffice.

5.2.1.2.7 Reaction Control System

The reaction control system is estimated to be independent of the number of crew men and resupply interval with respect to the thrusters, valves, etc. The propellant tank and pressurization system weight is a function of the amount of propellant required. The 24 man crew tankage is slightly heavier because the weight and inertia of the 24 man station is greater than the 9 man station thus, requiring more spin up propellant. The tankage and pressurization system weight is assumed to be .69 of the propellant weight. The weight of 3460 pounds for a three month resupply interval is obtained by considering an initial station spin up to approximately 4 RPM (640 pounds), spin axis precession to align solar cells toward the sun every twenty days (1420 pounds), and attitude and stabilization (1410 pounds). For a six month resupply interval, additional spin up propellant is not required. However, an additional 2820 pounds is required for orientation and stabilization. Since the 24 man station is heavier it is estimated that 80 additional pounds of propellant is required for six month resupply interval.

5.2.1.2.8 Guidance and Control

This is the only system currently estimated to be the same for a 9 man or 24 man station. The control moment gyro is assumed to weigh 1000 pounds. It is assumed that wobble damping mechanisms will be included in the weight.

The electronics weight of 300 pounds is a computer, IMU, and associated equipment, essentially Apollo hardware. This system is essential to operation of the experiments and stabilization of the station.

5.2.1.2.9 Cryogenic Tanks

The weight of oxygen and nitrogen storage tanks is derived using the curves included in Volume III, Figure 5.36.

The oxygen storage tank weight is obtained from this equation:

Tank Wt. = n (R) W where

n = Number of tanks

W = Weight of liquid in each tank

$$R = \frac{1-K}{K}$$

where K is the ratio of fluid weight to wet system weight as shown in Figure 5.36, Volume III.

	R	N	W
9 Man, 3 Month Resupply	.136	2	1690
9 Man, 6 Month Resupply	.136	4	1560
24 Man, 3 Month Resupply	.136	5	1474
24 Man, 6 Month Resupply	.136	10	1405

The nitrogen storage tank weight is obtained using the above formula and figure.

	R	N	W
9 Man, 3 Month Resupply	.60	3	1200
9 Man, 6 Month Resupply	.60	5	1200
24 Man, 3 Month Resupply	.59	3	1375
24 Man, 3 Month Resupply	.59	6	1070

The tankage weight is a function of the number of tanks and the amount of fluid stored.

5.2.1.2.10 Expendables

5.2.1.2.10.1 Oxygen

The amount of oxygen to be stored includes the metabolic requirements of the crew, the leakage rate, one pressurization of the station, and the Extravehicular Activities (EVA) requirement. A margin of 50 percent of the resupply interval is included in the calculations for the oxygen consumption except for the initial pressurization which is determined by the volume pressurized. For the 9 man crew, the pressurized volume is assumed to be 30,000 cubic feet. The pressurized volume is increased by 10,000 cubic feet to provide for increasing the crew to 24 men. Table 5.7 contains the weight of oxygen for 9 and 24 man crews.

5.2.1.2.10.2 Nitrogen

The amount of nitrogen to be stored includes the leakage and initial pressurization requirements. Table 5.8 contains the weight of nitrogen required.

TABLE 5.7
WEIGHT OF OXYGEN REQUIRED

Crew (Number of Mem)	9		24	
Pressurized Volume (ft^3)	30,000		40,000	
EVA Per Week	3		8	
Resupply Interval (Months)	3	6	3	6
Use / Weight	Pounds	Pounds	Pounds	Pounds
Leakage (4.2#/day)	567	1134	567	1134
Initial Pressurization	519	519	692	692
Plss Recharge (.92#/charge)	54	108	144	288
Metabolic (1.84#/manday)	2237	4473	5962	11924
TOTAL	3377	6234	7365	14038

TABLE 5.8
WEIGHT OF NITROGEN REQUIRED

Crew (Number of Men)	9		24	
Pressurized Volume (ft^3)	30,000		40,000	
Resupply Interval (Months)	3	6	3	6
Use / Weight	Pounds	Pounds	Pounds	Pounds
Leakage (13.8#/day)	1863	3726	1863	3726
Initial Pressurization	1700	1700	2267	2267
TOTAL	3563	5426	4130	5993

5.2.1.2.10.3 Food

The food weight is a function of the number of men, the number of days, the calorie intake and the type of food. Since the Apollo type food may not provide the required selection and composition for a long mission, it is assumed that the food weight in pounds per man day is 2.2 instead of 1.65 to 1.85 used for Apollo.

5.2.1.2.10.4 Miscellaneous Environmental Control and Life Support System

It is assumed that charcoal filters and chemi-absorbent bed expendables will be consumed. These items are dependent on the number of men and days of operation. The weight for 9 men for 3 months is estimated at 140 pounds. The weight for 24 men for 3 months is approximately 2.67 times the 9 man weight or 380 pounds. These values are doubled to obtain the 6 month requirements.

5.2.1.2.10.5 PLSS Water

The PLSS water recharge is currently estimated to be 7.33 pounds per charge. The number of charges to be provided is a function of the number of EVA's required. The number of EVA's is assumed to be 3 per week for a 9 man crew and 8 per week for a 24 man crew.

5.2.1.2.10.6 PLSS LiOH

Each LiOH charge is currently estimated at 3.5 pounds.

5.2.1.2.10.7 RCS Propellant

This is discussed in paragraph 5.2.1.2.7.

5.2.1.2.11 Rotating and Non-Rotating Allocation

The allocation of systems to either the zero or artificial gravity modules of the artificial gravity station is necessary to allow determination of the mass properties impact upon stability and orientation.

The allocation of each component in Tables 5.4 and 5.5 is established by at least one of the following considerations.

a. Two crewmen live in the non-rotating portion for biomedical experiments.

b. Approximately 67 percent of the electrical power load is used in the non-rotating portion and therefore requires the major portion of the coolant loops and radiators in the environmental control and life support system.

c. EVA activity will orginate in the non-rotating portion, hence, PLSS recharges will be located in that area.

d. It is, also, assumed that systems such as EC/LSS, communications and data management, and instrumentation will be divided almost equally between the two areas.

5.2.1.3 EMERGENCY VEHICLES

These vehicles will provide an emergency return capability for each crewman. The weight represents a vehicle which is limited in capability and is potentially susceptible to weight growth due to the concept of "use for emergency only" and limited effort on the concept.

5.2.1.4 STRUCTURE

During the past five years some sporadic effort has been expended on determining an effective technique to estimate the weight of the structure of various types of spacecraft. Figure 5.8 indicates the variation of structural weight in pounds per cubic foot with the total body volume developed to date. Many other data points have been considered and included in the analysis but, for clarity, have been omitted from the graphs. The data points (excluding 3A, 4 and 4A) are for vehicles in which the pressurized volume is approximately 50 percent or more of the total volume. This figure has been utilized to obtain the structure weights included in Tables 5.2 and 5.3. The dashed line represents a first estimate of the structural weight for a zero gravity space station. This line considers that the structural weight estimated from this figure will include an adequate non-optimum factor for support and mounting of the system components, provisions for double skin (open cell foam filled) suitable for nominal meteoroid protection, and radiation protection for low earth orbit (under 300 nautical miles and up to 60° inclination). The upper solid line on this figure has been used for artificial gravity stations. This line is considered to include the additional structural weight penalty to provide rotating capability and mass property compensation devices that are not included in Guidance and Control, Stabilization, and Reaction Control Systems.

5.2.2 GENERALIZED MASS PROPERTIES

5.2.2.1 PRELIMINARY MASS PROPERTIES

Table 5.9 includes the weight, center of gravity, and moment of inertia values estimated for use by the Guidance and Control and Aerodynamics Groups in order to provide preliminary data on the control systems required for an artificial gravity station. For a zero gravity station the mass properties are as shown in Table 5.10.

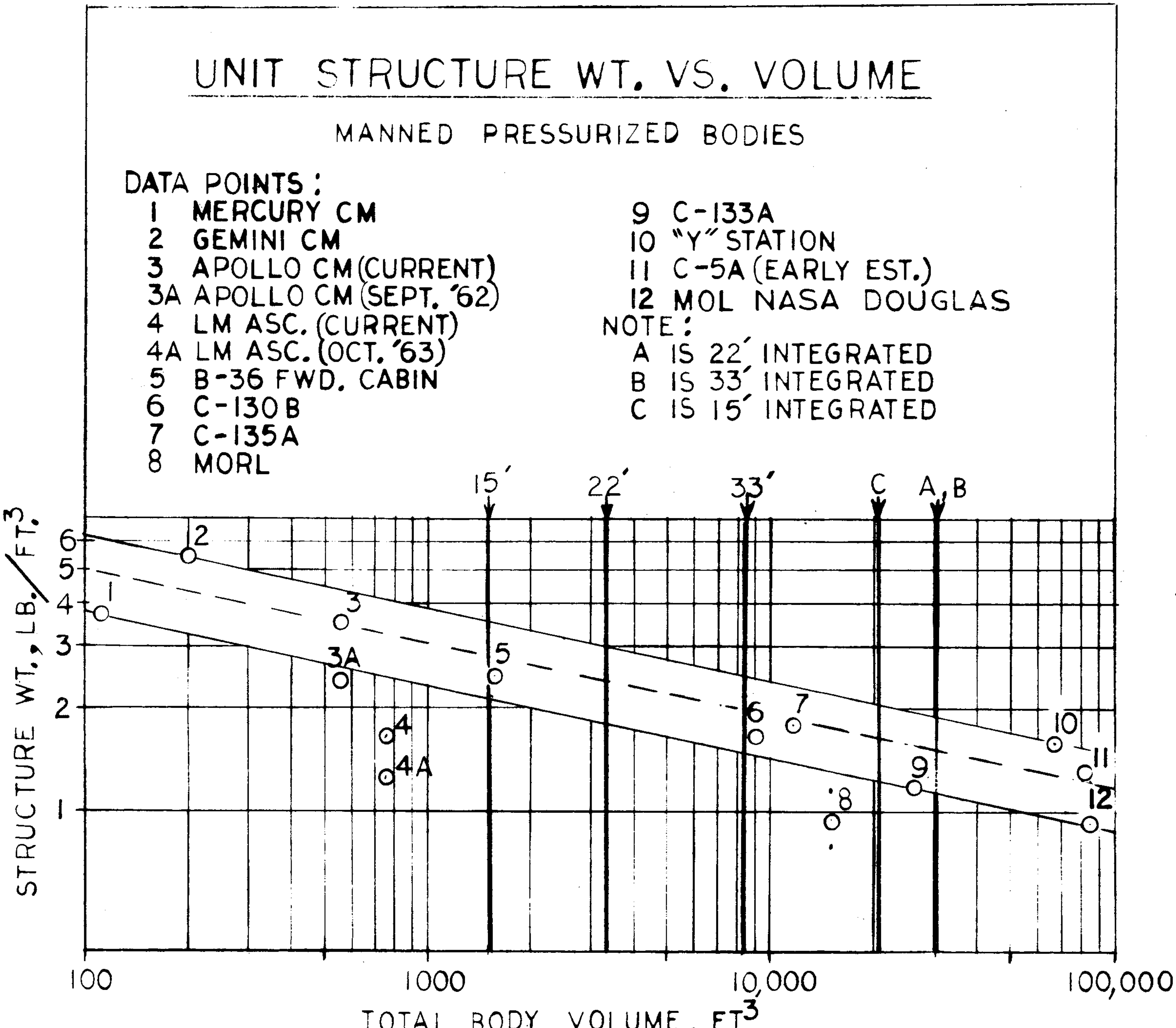

UNIT STRUCTURE WT. VS. VOLUME
MANNED PRESSURIZED BODIES
DATA POINTS:
1 MERCURY CM
2 GEMINI CM
3 APOLLO CM (CURRENT)
3A APOLLO CM (SEPT. '62)
4 LM ASC. (CURRENT)
4A LM ASC. (OCT. '63)
5 B-36 FWD. CABIN
6 C-130 B
7 C-135A
8 MORL
9 C-133A
10 "Y" STATION
11 C-5A (EARLY EST.)
12 MOL NASA DOUGLAS
NOTE:
A IS 22' INTEGRATED
B IS 33' INTEGRATED
C IS 15' INTEGRATED
15'
22'
33'
C
A, B
STRUCTURE WT., LB./FT.³
6
5
4
3
2
1
100
1000
10,000
100,000
TOTAL BODY VOLUME, FT.³

FIGURE 5.8

TABLE 5.9

PRELIMINARY MASS PROPERTY DATA - ARTIFICIAL GRAVITY STATION

COMPONENT	WEIGHT POUNDS	MOMENT ARM FEET X	Y	Z	MOMENT OF INERTIA SLUG $FT^2 \times 10^{-6}$ IX	IY	IZ
Living Quarters	87,300	0	0	75	.43	.43	.37
Arm	1,000	0	+14	-12	0	0	0
Arm	1,000	0	-14	-12	0	0	0
Arm	1,500	0	+13.7	+39	.01	.01	0
Arm	1,500	0	-13.7	+39	.01	.01	0
S II	83,000	0	0	-79.9	1.47	1.47	.43
Hub	20,000	0	0	0	.05	.03	.03
Solar Cells	4,400	0	+75	0	.12	.01	.11
Solar Cells	4,400	0	-75	0	.12	.01	.11
Rotating Part	204,100	0	0	0	35.64	33.83	2.59
Hub (Zero g)	78,100	12	0	0	.20	.31	.31
Total	282,200	3.3	0	0	35.84	34.40	3.15

TABLE 5.10
MASS PROPERTY DATA-ZERO GRAVITY STATION

Weight	159,000 Pounds
$\overline{X}$	45 Feet
$\overline{Y}$	0 Feet
$\overline{Z}$	0 Feet
Ix-X	1.84×10^6 slug ft^2
Iy-Y	4.52×10^6 slug ft^2
Iz-Z	2.95×10^6 slug ft^2

PRELIMINARY TECHNICAL DATA

FOR EARTH ORBITING SPACE STATION

VOLUME IV

CONFIGURATIONS, INTEGRATION, AND WEIGHTS

SECTION 6.0

MARS MISSION - SPACE STATION COMPARISON

ADVANCED SPACECRAFT TECHNOLOGY DIVISION

ENGINEERING AND DEVELOPMENT DIRECTORATE

MANNED SPACECRAFT CENTER

6.0 MARS MISSION - SPACE STATION COMPARISON

The study groundrules, the mission, and system requirements for the Mars flyby and for the earth orbiting space station were compared to identify their commonalities and differences. A comparison of requirements for the two missions was made, and the more stringent selected in each area to establish common requirements to be used for both missions. These requirements were used to establish a configuration suitable for both missions.

6.1 MAJOR DIFFERENCES

Differences were extracted to show where study would be needed to allow common program definition, design, development, testing, and hardware. Table 6.1 lists these differences under two categories: groundrules and system requirements.

6.1.1 GROUNDRULE DIFFERENCES

The differences in groundrules represent those items that may be adjusted to make the requirements of the two missions more compatible. These items and their effect on the compatibility of the two missions are discussed in the following paragraphs.

6.1.1.1 CREW CONSIDERATIONS

Crew size for the Mars flyby mission study was set at four men, while the space station study was required to consider both 9 and 24 man crews. The crew size affects the overall size and, to some degree, the shape of a spacecraft because of the necessary areas and volumes required to provide a habitable interior. The environmental control and life support subsystem and the crew systems requirements are also affected by crew size. Other important crew factor comparisons used to determine the common requirements for the two missions are as follows. The Mars mission crew must operate efficiently for the total mission while the space station crew may be rotated at intervals by a logistics spacecraft. Consequently, for the Mars mission, a very careful and efficient crew quarters and systems design is required. The space station may provide very useful data for such design. Specialists may be used for the conduct of most experiments aboard a space station, but the limited crew and many varied tasks require a more versatile crew for the Mars mission. Crew volume allocations for the two studies varied sufficiently to affect vehicle size. Section 6.3.2 discusses this variation and its effect in detail. Crew time allotments for the Mars mission were not determined; however, the time allotments for the space station study are compatible with the Mars mission.

TABLE 6.1

MAJOR DIFFERENCES BETWEEN MARS MISSION AND SPACE STATION

I. GROUND RULES

1. CREW SIZE
2. ZERO-ARTIFICIAL GRAVITY
3. RESUPPLY
4. LAUNCH & ORBITAL ASSEMBLY
5. EXPERIMENTAL PAYLOAD

II. SYSTEM REQUIREMENTS

1. METEOROID ENVIRONMENT
2. RADIATION ENVIRONMENT
3. THERMAL ENVIRONMENT
4. AERODYNAMIC DRAG
5. EARTH ENTRY
6. MISSION TIME

6.1.1.2 ZERO - ARTIFICIAL GRAVITY

The Mars mission study was groundruled to consider zero gravity while the space station study considered both zero and artificial gravity. Should one mission use artificial gravity and the other use zero gravity, a lesser degree of compatibility would exist. This difference would result in the development of systems and functions that would be used on only one mission.

The deployment operation includes extending the artificial gravity compartment and the counterweight in relation to the center hub, activation of seals and joint locks, and the connecting of umbilicals. Other functions required by an artificial gravity vehicle are wobble damping and the non-rotating hub for zero gravity experiments. The hub requires bearings, seals and transfer facilities between the rotating and non-rotating parts. Crew systems requirements for zero gravity are more stringent than for artificial gravity and include crew hold downs and aids for mobility, food preparation, crew tasks and personal hygiene.

6.1.1.3 RESUPPLY

The capability of providing a shuttle spacecraft to an Earth Orbiting Space Station allows resupply of expendables, addition of experiments, supply of spare parts or modular units, and crew rotation. Mission characteristics make resupply for a planetary mission impractical. Resupply for the space station can be adjusted to make the requirements of the two missions more compatible.

Although resupply represents a difference in mission requirements, it is an asset in the final development of the Mars mission, because repair, maintenance, operational procedures, and equipment tests may be performed aboard the space station which has a logistics link with earth.

6.1.1.4 LAUNCH AND ORBITAL ASSEMBLY

The earth orbiting space station is placed into orbit by a single launch. Although the Mars mission spacecraft is placed into earth orbit by a single launch, the complete trans-Mars injection configuration requires multiple launch and orbital assembly. The Mars mission, therefore, requires a more complex launch operation and additional operations to allow assembly in orbit. Hardware for docking of a logistics spacecraft to the space station is unlikely to be capable of being used for orbital assembly of a trans-Mars injection configuration, but operational procedures and design principles developed for both missions can be compatible.

6.1.2 SYSTEM REQUIREMENTS

The differences in system requirements represent those items that are imposed by mission requirements. These differences are discussed in the following paragraphs.

6.1.2.1 METEOROID ENVIRONMENT

Figure 6.1 shows a comparison of space station and Mars mission meteoroid shield weight requirements. The top curve is for a .99 probility of no penetration for a 680 day Mars flyby mission. The second curve is for a .99 probability of no more than one penetration for the same mission. The lower curve shows the requirement for a 2 year earth orbital mission. Total required protected area is approximately 4000 square feet for the Mars mission vehicle and approximately 10,000 square feet for the space station. Both mission studies assumed a meteoroid damage repair capability would be provided by arranging equipment to allow access to the pressure vessel.

A propulsive turn at Mars, to avoid the asteroid belt, could be used to reduce the large difference in shielding between the two spacecraft.

6.1.2.2 RADIATION ENVIRONMENT

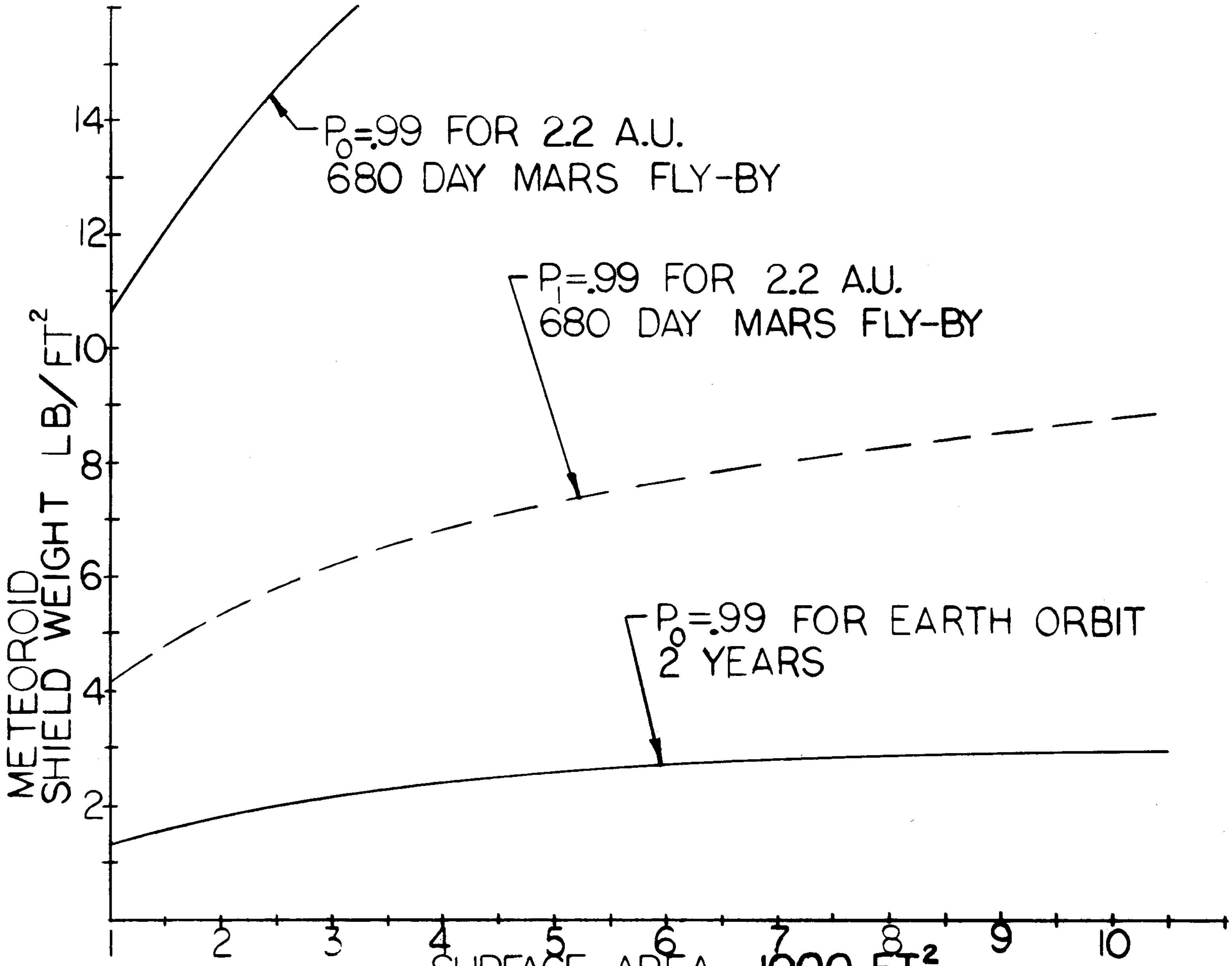
Figure 6.2 shows a comparison of space station and planetary mission radiation protection requirements. The two lower curves show data for 30^{o} and 60^{o} inclination earth orbits. The two upper curves enclose a band representing the predicted solar radiation for a planetary mission. The lower curve of the band represents the approximate values that may be expected for a 1975 Mars twilight flyby mission. A storm shelter is likely to be required for the planetary mission but is not necessary for the space station. Crew dose rates were limited to the following values for the Mars mission:

a. .999 probability of not exceeding 100 Rads when a storm shelter is used.

b. .99 probability of not exceeding 100 Rads when a storm shelter is not used.

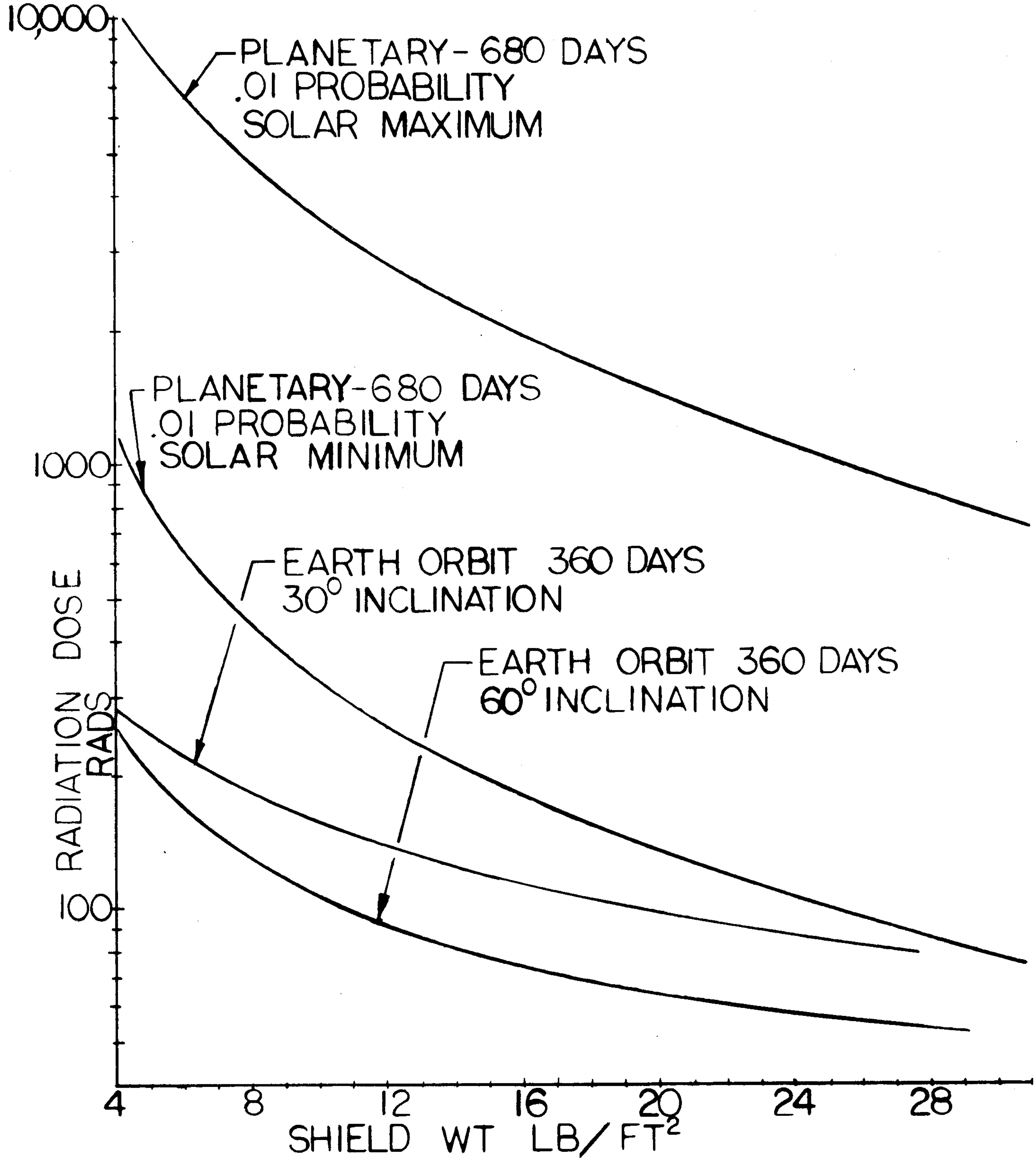
6.1.2.3 THERMAL ENVIRONMENT

Although the basic system problems imposed by the thermal environment are similar for the two mission, differences exist because of distances from the sun and the influence of planetary albedo. Solar flux for the space station is nearly constant as the vehicle remains at about 1.0 astronomical unit (A.U.) from the sun. For the Mars mission, distance from the sun



SHIELD WEIGHT VS SURFACE AREA

FIGURE 6.1



RADIATION DOSE VS SHIELD WEIGHT

FIGURE 6.2

varies from .6 A.U. to 2.2 A.U. The space station is within the influence of the high earth albedo during the entire mission while the Mars mission is influenced by planetary albedo for only short periods.

6.1.2.4 AERODYNAMIC DRAG

Aerodynamic drag affects the Mars mission vehicle only during launch and for the short time the vehicle is in earth orbit. The space station is acted upon by aerodynamic forces continuously through the mission, therefore requiring a propulsive force to maintain the desired orbit.

6.1.2.5 EARTH ENTRY

Entry into the earth's atmosphere from low earth orbit is a proven operation. Earth entry from a Mars mission involves much higher entry velocities (50,000 to 60,000 feet per second as compared to approximately 26,000 feet per second for entry from low earth orbit) and precise guidance is necessary to acquire the entry corridor. These differences represent developments which will be required for the Mars mission only.

6.1.2.6 MISSION TIME

Total mission time affects systems, subsystems, and the crew to various degrees. Reliability and maintainability are important systems aspects that are affected by mission time.

6.2 SUBSYSTEMS COMPARISON

6.2.1 ELECTRICAL POWER SYSTEM (EPS) COMPARISON

The EPS for the Mars flyby and the space station are the same basic type, viz., silicon solar cells for primary power with batteries for peak loads and operation in shadow. The systems differ as a result of three major factors: the maximum solar distance, the occultation of the sun in earth orbit, and the power profiles of the two missions, as outlined in Table 6.2.

The Mars mission array was sized for minimum housekeeping requirements at the maximum solar distance. Because of the variation in solar distance, and hence in solar array power output, ample additional power is then available in the vicinity of Mars for experiments and data transmission. Batteries are required for injection from earth, for Mars encounter (due to occultation of the sun by the planet), and for peak power requirements (largely communications) at maximum solar distance. A battery system to meet these requirements is relatively small.

Dark-side time per orbit for the space station will vary from zero to 39 percent because of orbit node regression and the

TABLE 6.2

MARS MISSION - SPACE STATION
ELECTRICAL POWER AND POWER REQUIREMENTS COMPARISON

	MARS FLYBY	SPACE STATION
MAXIMUM SOLAR DISTANCE	2.2 A.U.	1.0 A.U.
TIME IN SUNLIGHT	100%	61% MINIMUM
SOLAR ARRAY AREA	2000 FT^2	4400 FT^2
SOLAR ARRAY WEIGHT	0.7 LB/FT^2	1.4 LB/FT^2
AVERAGE POWER		
EC/LSS	1.0 KW	2.0 KW
GUIDANCE & CONTROL	0.4	0.6
COMM/DATA MGT. (MINIMUM)	0.1	1.9
INSTRUMENTATION	0.3	0.5
CREW SYSTEMS	0.1	0.1
EEM ECS	0.5	---
LIGHTING	0.4	0.8
EXPERIMENTS	0.2	5.7
CONTINGENCY	1.0	3.4
TOTAL	4.0 KW AVERAGE @ 2.2 A.U.	15.0 KW AVERAGE

inclination of the equator to the ecliptic. The batteries, however, must be designed for the worst case. Because of the large continuous power requirements and substantial peak loads, the space station battery system becomes a significant item in the total weight. The solar array must also be sized for the worst case.

The average power levels differ primarily in experiment requirements. It should be noted that the Mars mission power levels shown in Table 6.2 represent minimum loads for system sizing at 2.2 A.U.

The difference in solar array weights in Table 6.2 results from improved structural technology assumed to be available in time for the Mars mission.

6.2.2 ENVIRONMENTAL CONTROL/LIFE SUPPORT SYSTEM (EC/LSS) COMPARISON

The EC/LSS for both missions employs a molecular sieve for CO_2 removal, catalytic burner and chemisorbent bed for contaminant control, water reclamation, an oxygen-nitrogen atmosphere stored subcritically, and a coolant loop - radiator thermal control system. The principal differences are outlined in Table 6.3. Some of these are the result of variations in study groundrules. For example, a Mars flyby spacecraft designed on the same basis as the space station would have incorporated complete water reclamation.

Aside from the number of crew members, which affects component sizing, the fundamental difference is the lack of resupply capability for the Mars mission. The resulting longer storage time not only poses a more difficult cryogenic tank design problem, but requires larger storage capacity for oxygen, nitrogen, food, and other expendables.

6.2.3 COMMUNICATIONS AND DATA MANAGEMENT SYSTEM (C/DM) COMPARISON

Table 6.4 summarizes the differences in C/DM requirements for the Mars mission and the space station. The controlling factor in the Mars mission is the large transmission distance (up to 3.2 A.U.) together with high data rates, which requires a large directional antenna and high transmitter power.

For low altitude earth orbit missions, omnidirectional antennas and moderate transmitter power levels can provide sufficient data rates for presently defined requirements. Data storage requirements are substantial because of intermittent ground contact.

On-board data processing will be employed in both missions to minimize data transmission requirements.

TABLE 6.3

MARS MISSION - SPACE STATION
ENVIRONMENTAL CONTROL SYSTEM COMPARISON

	MARS FLYBY	SPACE STATION
OXYGEN REGENERATION	NOT REQUIRED	NOT REQUIRED
WATER RECLAMATION	ALL BUT FECAL	ALL
CREW SIZE	4	9
LEAKAGE	5.0 LB/DAY	16.7 LB/DAY
OXYGEN & NITROGEN CAPACITY	8940 LB	5950 LB
CRYOGENIC STORAGE TIME	680 DAYS	135 DAYS
FOOD ALLOWANCE	1.65 LB/MAN-DAY	2.2 LB/MAN-DAY
FOOD STORAGE CAPACITY	4500 LB	2680 LB

TABLE 6.4

MARS MISSION - SPACE STATION
COMMUNICATIONS COMPARISON

	MARS FLYBY	SPACE STATION
TRANSMITTER INPUT POWER	2000 WATTS	50 WATTS
ANTENNA	19' DISH	OMNI
CONTACT WITH GROUND	CONTINUOUS	INTERMITTENT
DATA TRANSMISSION RATE	10^6 bps AT 1.5 A.U.	10^6-10^7 bps

6.2.4 GUIDANCE AND CONTROL SYSTEM (G&C) COMPARISON

There are two principal differences in G&C requirements for the Mars flyby and the space station. One of these is the capability for on-board navigation and guidance in the vicinity of Mars to achieve better accuracy than earth-based tracking. Since space station navigation will be accomplished by ground tracking, on-board requirements can be satisfied by a computer generating instantaneous position information from ground-furnished orbit data. The Mars mission navigation and guidance system is therefore more complex than that for the space station.

The second major difference between the two spacecraft is the provision of artificial gravity in the space station, resulting in different control problems for the two missions. The control system in both cases will consist of reaction control thrusters and control moment gyros although the operation may not be identical. Requirements have not been determined in sufficient detail to establish the feasibility of common hardware for the control moment gyros, but it appears that the development program will be largely the same for both missions.

Artificial gravity will also require a means of compensating for crew moment and other imbalance within the spacecraft in order to avoid excessive accelerations in the zero-gravity hub. This will be accomplished by transfer of mass to maintain the center of mass and the principal axes of inertia in the correct position.

6.3 MARS MISSION CONFIGURATION EFFECTS

To determine the effect of applying common requirements to both missions, space station subsystem and crew allocations were applied to Mars Mission Configuration D-1. An altered set of requirements was also applied to configuration K-1 to take advantage of changes in Mars mission study groundrules and minimize the impact to the configuration.

6.3.1 SUBSYSTEM ALLOCATIONS

Table 6.5 shows the volume allocated to each subsystem for configuration K-1, configuration K-1 with the space station systems volumes sizing requirements applied, and with modified sizing requirements applied. The values listed in the table show minor or no differences except for unpressurized storage, food storage, and systems access. The difference in the unpressurized storage volume is caused by a larger leak rate (space station) being used in column 2. Food storage volume in columns 2 and 3 is larger because the space station study used more appetizing foods. The Mars mission study packaged

TABLE 6.5

SYSTEM ALLOCATIONS FOR MARS MISSION CONFIGURATION K-1 AND VARIATIONS

CONFIG.	K-1 (ON PRESS) PRESS (1)	K-1 WITH S.S. CREW & SYS. REQ. (2)	K-1 WITH MODIFIED REQ. (3)
EC/LSS	(147 FT^3) 48 FT^3	(270 FT^3) 53 FT^3	(122 FT^3) 53 FT^3
CREW SYSTEMS	80	71	71
FOOD	321	427	427
EPS	8	8	8
INSTRUMEN.	17	17	17
G&C	50	50	50
COMM. & DATA	25	25	25
ACCESS TO SYS.		448	448
TOTAL SYS. VOL.	(147 FT^3) 549 FT^3	(270 FT^3) 1099 FT^3	(122 FT^3) 1099 FT^3

the subsystems to be accessible to the crew by using living volume to perform maintenance and repair functions while a 200 percent service volume was allowed for space station subsystems. This factor accounts for the 448 ft^3 "Access to Systems" volume in columns 2 and 3.

6.3.2 CREW ALLOCATIONS

Table 6.6 shows the crew area and volume requirements for the same configurations discussed in the preceding section. Column 2 shows the total requirements for a 4 man crew. The food preparation area, head, sick bay, gym, and the command post are the same as a 9 man space station and all but the gym are considered to be about the minimum required. Since the Mars mission study did not separate the various living area requirements, the available volume was distributed as shown in column 1 for comparison. The ward room and sick bay were eliminated and the gym area was reduced; otherwise, the data are the same as column 2. A compromise to the above discussed allocations is shown in column 3. The ward room or the larger gym may double for a sick bay.

6.3.3 CONFIGURATION IMPACT

Figure 6.3 shows the results of applying the crew and systems allocations to Mars configuration K-1. The configuration on the left is K-1 and corresponds to the data in column 1 of Tables 6.5 and 6.6. The center configuration corresponds to column 2 of Tables 6.5 and 6.6 and shows the net external effect of a 15 foot increase in spacecraft length. The configuration on the right corresponds to the data in column 3 of Tables 6.5 and 6.6 and takes advantage of a Mars mission study groundrule change which eliminates trans-Mars injection abort capability. This configuration results in only a 5 foot increase in length over configuration K-1.

6.3.4 WEIGHT COMPARISON

Table 6.7 presents a comparison of the spacecraft weight summary for configuration K-1 using the mission criteria indicated in columns 1, 2, and 3 of Tables 6.5 and 6.6.

The change in structure weight results from the addition of approximately 15 feet vehicle length (190 inch diameter) in the second column and approximately 5 feet in the third column (5#/sq. ft. considered).

The change in EC/LSS weight represents increased tankage for cryogenic storage.

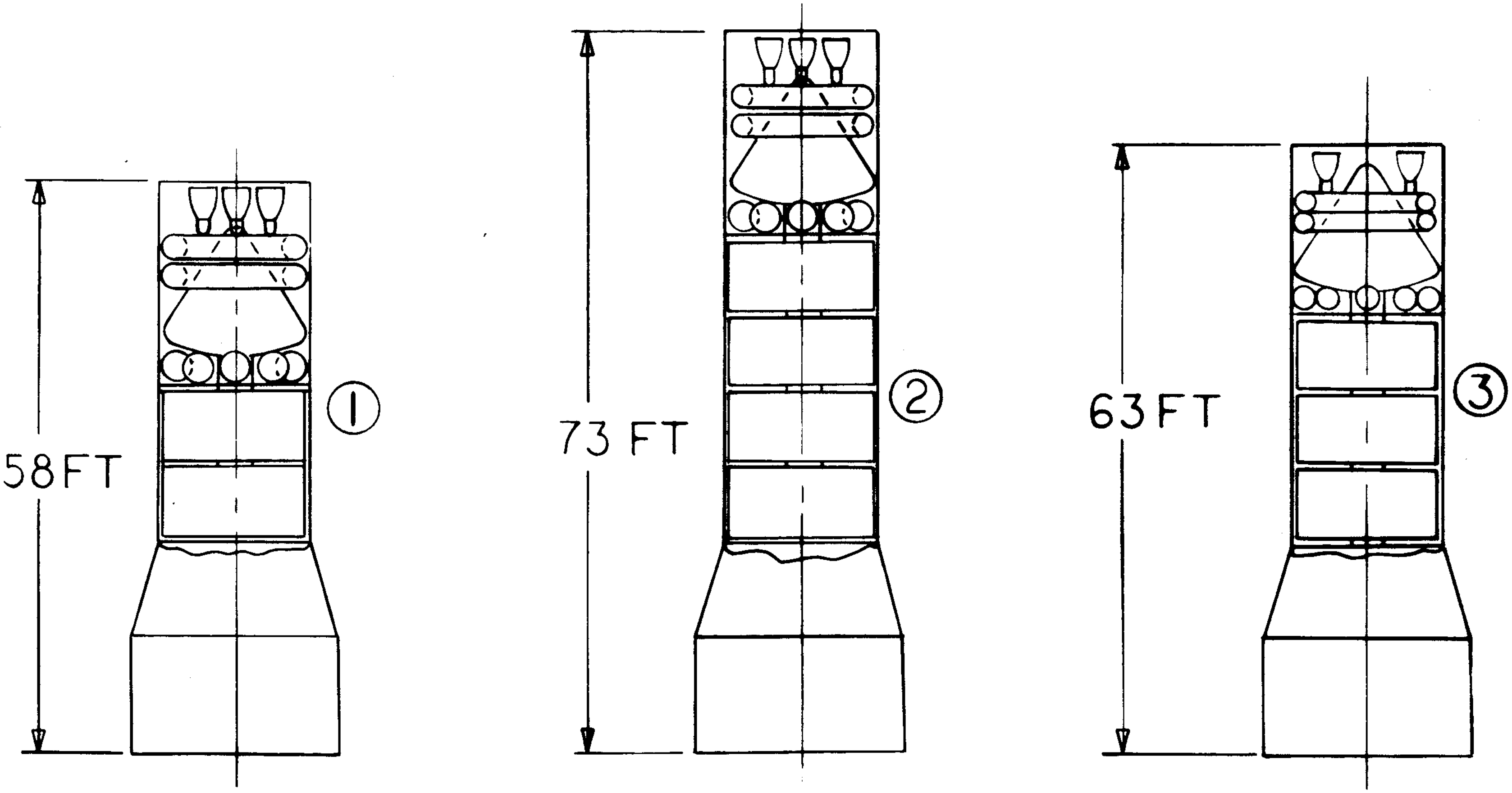
The change in oxygen, nitrogen, etc. results from a change in leakage requirements.

TABLE 6.6

CRU ALLOCATIONS

CONFIG.	1 K-1	2 K-1 WITH S.S. CREW & SYS. REQ.	3 K-1 WITH MODIFIED REQ.
SLEEP	140 FT^2	140 FT^2	140 FT^2
WARD ROOM		55	55
FOOD PREP	16	16	16
HEAD	28	28	28
SICK BAY		108	
GYM	38	60	60
COMMAND	32	32	32
CREW AREA TOTAL	254 FT^2	439 FT^2	331 FT^2
CREW VOL. TOTAL	1778 FT^3	3073 FT^3	2317 FT^3

EFFECT OF VARYING HABITABILITY & SYSTEMS REQUIREMENTS ON MARS MISSION CONFIGURATION



MARS FLYBY
CONFIG. K-1

PRESS VOL. 2330 $FT.^3$
INJECT WT. 162,830

MARS FLYBY
WITH SPACE STATION
CREW & SYS. CRITERIA

PRESS. VOL. 4660 $FT.^3$
INJECT. WT. 183,940

MARS FLYBY
MODIFIED CRITERIA

PRESS. VOL. 3495 FT^3
INJECT. WT. 153,640

FIGURE 6.3

TABLE 6.7
SUBSYSTEM WEIGHT SUMMARY

	K-1	Space Station Requirements	Modified Space Station Requirements
Structure	9000	12750	10250
EC/LSS	2980	8665	4855
Power System	3180	3180	3180
Guidance & Navigation	520	520	520
Comm. & Data Handling	3810	3810	3810
Personnel Accommodations	1830	1830	1830
Instrumentation	---	---	---
Controls & Displays	500	500	500
Spares	900	900	900
Total Empty Weight	22720	32155	25845
Oxygen, Nitrogen, Etc.	9320	12765	4525
Food	4500	6000	6000
Hygiene & Clothing	530	280	280
Total Mission (Less Meteoroid)	37070	51200	36650
Meteoroid Protection	480	480	480
Total Mission Module	(37550)	(51680)	(37130)
Envelope Meteoroid Shielding	(37770)	(44750)	(40100)
Midcourse Prop. Module	(6000)	(6000)	(6000)
Abort Capability	(11100)	(11100)	---
EEM	(15100)	(15100)	(15100)
Experiment Module	(39100)	(39100)	(39100)
Probe Compartment	12950	12950	12950
Probes	25000	25000	25000
Meteoroid Protection	1150	1150	1150
SM Propellant	(16210)	(16210)	(16210)
TOTAL	162830	183940	153640

The 1500 pound weight increase in food results from using 2.2 pounds per man-day instead of 1.65 pounds per man-day.

The decrease of 250 pounds in hygiene and clothing is the net effect of adding 150 pounds for laundry facilities and reducing the clothing allowance by 400 pounds.

The total weight change of the spacecraft between column 1 and column 3 is essentially the elimination of abort capabilities.

6.4 SPACE STATION CONFIGURATION EFFECTS

A survey of the artificial gravity space station configurations revealed that a part of the space station zero gravity hub may be used as a zero gravity Mars mission living module. The zero gravity hub of the space station configuration shown in Figure 4.3 , for example, contains two 260 inch diameter modules that house a biology laboratory, living quarters for two crewmen (zero gravity test subjects) and subsystems to support the zero gravity hub. The volume of the two 260 inch diameter modules is equivalent to the four smaller diameter modules shown in the center illustration of Figure 6.3. This provides adequate volume for a Mars mission living module. The allows the pressure vessel,the support subsystems, and possibly a large part of the integrated modules to be designed for both missions with the penalties being imposed upon the space station. Parallel study of the two missions will serve to enhance their potential compatibility.