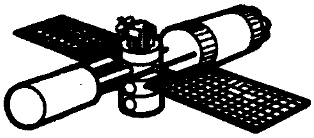
29P MSC-EA-R-66-1

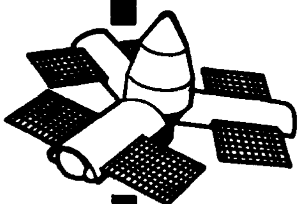
Vol. I



PRELIMINARY TECHNICAL DATA FOR EARTH ORBITING SPACE STATION

~~SUMMARY REPORT~~

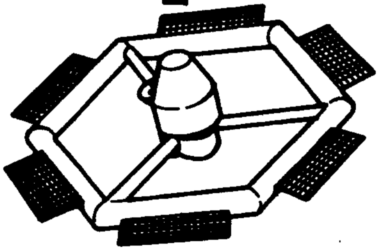
VOLUME I



NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

MANNED SPACECRAFT CENTER

Houston



NOVEMBER 7, 1966

Rq/43834

PREFACE

This document contains the Manned Spacecraft Center's summary technical data on the Earth Orbital Manned Space Station. These data are concerned with the human factors, environment, logistics, systems, weights, and configurations. This document is submitted in response to a NASA Headquarters initiated study which includes experiment 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.

This document was integrated by the Systems Engineering Branch (SEB) of the Advanced Spacecraft Technology Division. The major contributions to the study were made by the following organizations:

Medical Research & Operations Directorate	Human Factors
Space Science Division	Environment
Flight Crew System Division	Operations
Flight Operations Directorate	Operations
Propulsion and Power Division	Electrical Power, Reaction Control, Cryogenic Storage
Crew Systems Division	Environmental Control
Instrumentation & Electronic Systems Division	Communications, Data Management, Instrumentation
Advanced Spacecraft Technology Division	Habitability, Configurations, Systems Integration, Weights, Logistics

LIST OF VOLUMES

EARTH ORBITING SPACE STATION

VOLUME NUMBER	TITLE
I	Summary Report - Preliminary Technical Data for Earth Orbiting Space Station
II	Technical Data - Standards and Criteria for Earth Orbiting Space Station
III	Technical Data - Systems for Earth Orbiting Space Station
IV	Technical Data - Configurations, Integration, and Weights for Earth Orbiting Space Station

TABLE OF CONTENTS

EARTH ORBITING SPACE STATION

SUMMARY REPORT

SECTION	SUBJECT
4.4	Weights
4.4.1	Major Contributing Factors
4.4.2	General Configuration Weights

LIST OF FIGURES

EARTH ORBITING SPACE STATION

LIST OF TABLES

EARTH ORBITING SPACE STATION

4.0 SPACE STATION CONFIGURATION AND SYSTEMS APPLICABLE TO ANY PHYSICAL CONFIGURATION

To establish a basic set of technical data applicable to any space station configuration requires that each element and its relationship to the constraints for the total system be understood. Few of the many space station elements can be studied independently from the sum total. The flow diagram shown on the following page illustrates the interdependency of the major elements and the primary constraining factors.

The most important groundrules are as follows:

Crew Size: 9 to 24 men.

Altitude: 260 nautical miles (nominal).

Orbital Inclination: 50° - 70°.

Orbital Life: 5 years.

Gravity: Zero and artificial.

Launch Expendables: Sufficient for resupply interval plus 50% margin.

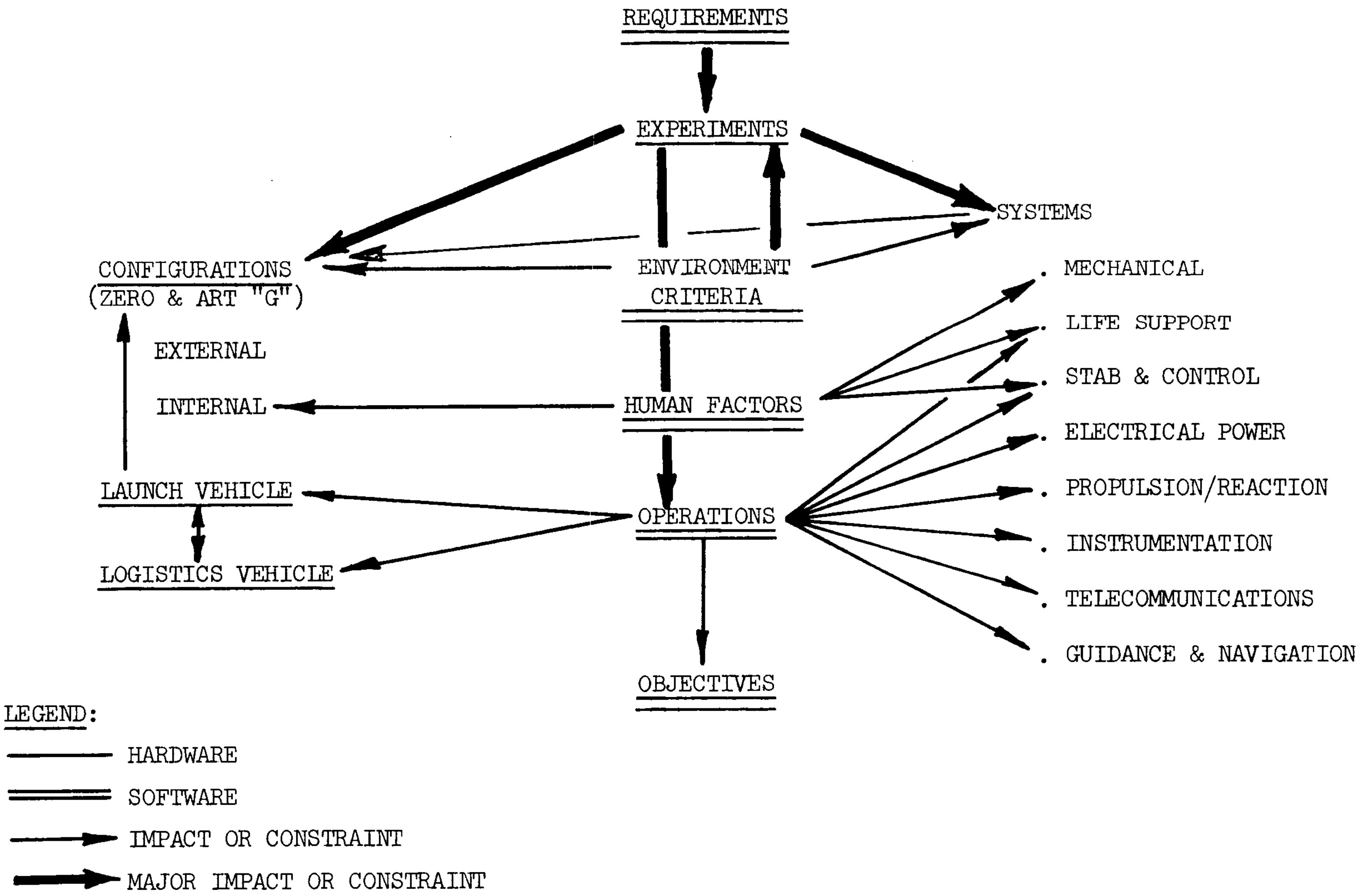
Habitable Quarters: Individual sleeping quarters and other provisions to approach earth-like conditions.

Work Quarters: Sufficient to allow on-board maintenance.

The general design approach is to establish a station with near earth-like total environment and a maximum independence from earth support. This self-sustaining capability will include

FLOW DIAGRAM

SPACE STATION STUDY ELEMENTS



provisions to cope with on-board emergencies resulting from system failures. Emergency return devices are provided in the event the station must be abandoned. The design will also allow for the accommodation of a multitude of experiments.

Section 4.0 is based on the foregoing design philosophy.

4.1 STANDARDS AND CRITERIA

4.1.1 HUMAN FACTORS

4.1.1.1 NUTRITION

A 2800 K cal per man-day diet will be provided that furnishes a variety of familiar foods. The most realistic data available indicates that the diet should have the following distribution of calories:

15 percent protein

33 percent fat

52 percent carbohydrate

The distribution of types of stored food should be:

75 percent dried (rehydratable and bites)

15 percent heat processed

10 percent frozen

Drinking water will be provided at 6.5 pounds/man-day.

4.1.1.2 PERSONAL HYGIENE

The social aspects of personal hygiene require frequent and routine body cleansing. Personal hygiene provisions will be as follows:

Handwashing and whole body washing

Clothes washing

Oral hygiene

Shaving, haircutting, etc.

4.1.1.3 WASTE MANAGEMENT

The waste management system must prevent the buildup of toxic gases, odors, and microorganisms.

4.1.1.4 HABITABILITY

The space allocated for individual and group habitability should be functionally compatible with the normal earth environment. To provide for this, the following allotments were derived.

4.1.1.4.1 Wardroom

The wardroom will be used for eating and recreation. The size was based on an allocation of 21 square feet per man, assuming occupancy by two-thirds of the crew at any one time.

4.1.1.4.2 Food Preparation

The food preparation area will be adjacent to the wardroom. The allotted area is shown in Table 4.1. This area does not include the food storage requirement.

4.1.1.4.3 Personal Quarters

Individual private quarters have been provided for each crew member. These compartments are used for sleep, relaxation and study. A total floor area of approximately 35 square feet is allotted to each compartment as indicated in Figure 4.1.

4.1.1.4.4 Gymnasium

The crew exercise area is shown in Table 4.1.

4.1.1.4.5 Sick Bay

The sick bay provides space for biomedical experiments and treatment of sickness or injuries. The allotted area will vary with the crew size as shown in Table 4.1.

4.1.1.4.6 Hygienic Area(s)

The hygienic area(s) provides facilities for cleansing and waste collection. There will be one toilet for each four men and a shower for each twelve men; each station will have a minimum of two toilets and one shower. The allotted area will vary with crew size as shown in Table 4.1

4.1.1.4.7 Command Station

The command station provides systems control and monitoring. The allotted area is as shown in Table 4.1.

CREW QUARTERS

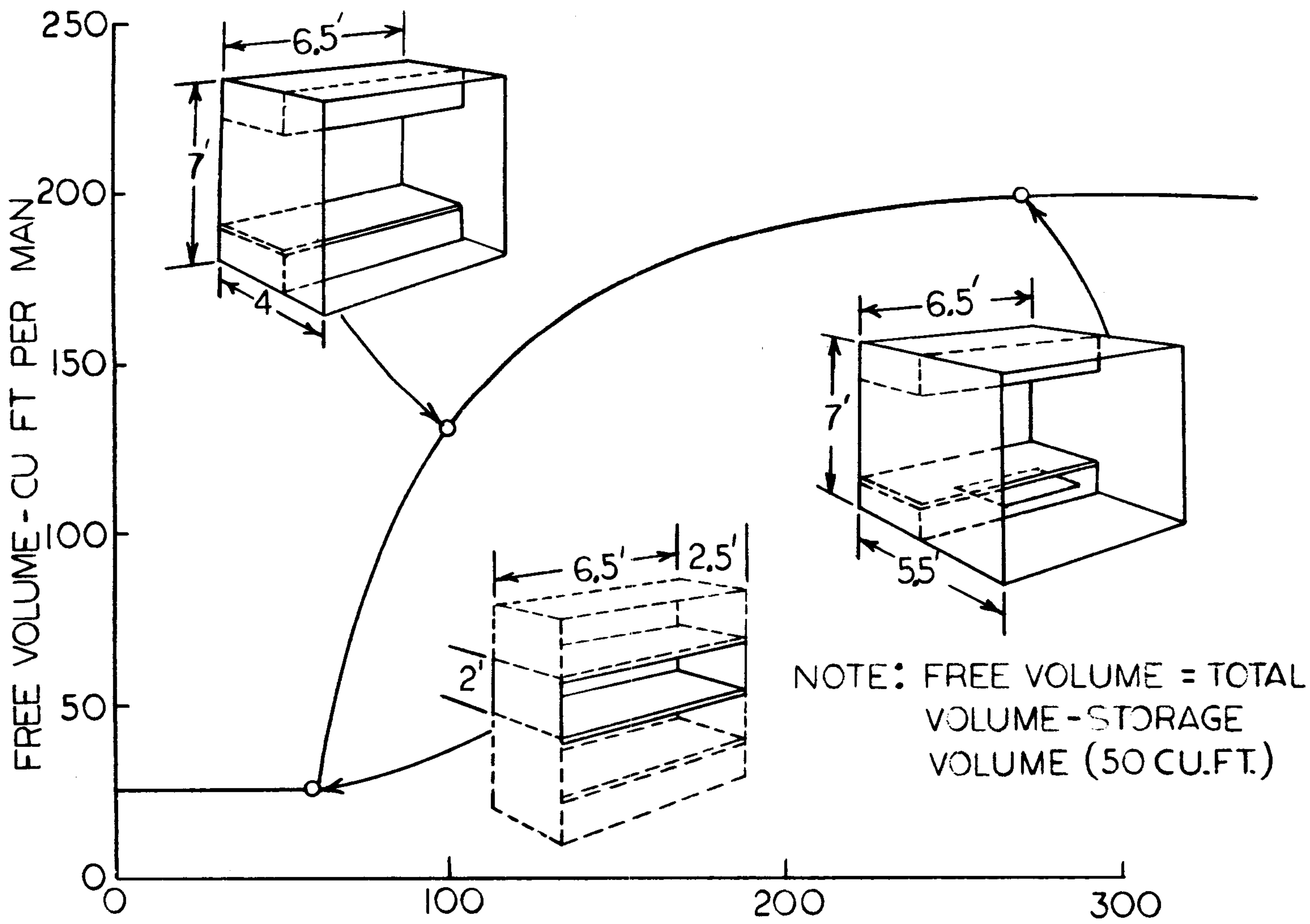


FIGURE 4.1

TABLE 4.1

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

4.1.1.4.8 **Summary**

Table 4.1 summarizes habitability allotments for 9, 12, and 24 man crews.

4.1.1.5 **ATMOSPHERE**

The following are recommended requirements for the station atmosphere:

Pressure	14.7 psia
Composition	
Oxygen	21 percent
Nitrogen	79 percent
Contaminant Criteria	
Carbon Dioxide	less than 0.5 percent
Carbon Monoxide	
Nominal	10 ppm
Maximum	25 ppm
Water Vapor Pressure	
Minimum	10 mm Hg absolute
Maximum	18 mm Hg absolute
Hydrocarbons	
Total	100 ppm or less
Individual Gas	1.0 ppm or less
Ozone	0.1 ppm or less
Aerosols (Maximum size)	0.3 micron

4.1.1.6 MICROBIOLOGY

The occurrence of infectious disease during a prolonged space mission will be minimized by preventive and surveillance methods. This imposes a requirement for frequent medical examinations and biological control of the spacecraft equipment and environment.

4.1.1.7 THERMAL

The temperature should be adjustable between $65^{\circ}F$ and $80^{\circ}F$ with an accuracy of $\pm$ $3^{\circ}F$ at any selected temperature within this range. The transcompartment temperature gradient should not exceed $5^{\circ}F$. Humidity control is required in parallel with temperature control to permit selection of the optimal temperature/humidity ratio for comfort. The absolute water content should not be less than 10 mm Hg nor exceed 18 mm Hg water vapor pressure. Air velocity of at least 15 feet per minute is recommended.

4.1.1.8 NOISE

Noise should not exceed 125 db for a period of 30 seconds and should be less than 115 db if the duration is for 300 seconds. Noise levels must not interfere with voice communication and must not constitute a chronic annoyance factor. Limitation of total white noise levels to 75 db (with a 50 db limit from 600 to 4800 cps) in station work spaces and 50 db in living quarters is recommended.

4.1.1.9 ARTIFICIAL GRAVITY CONSIDERATIONS

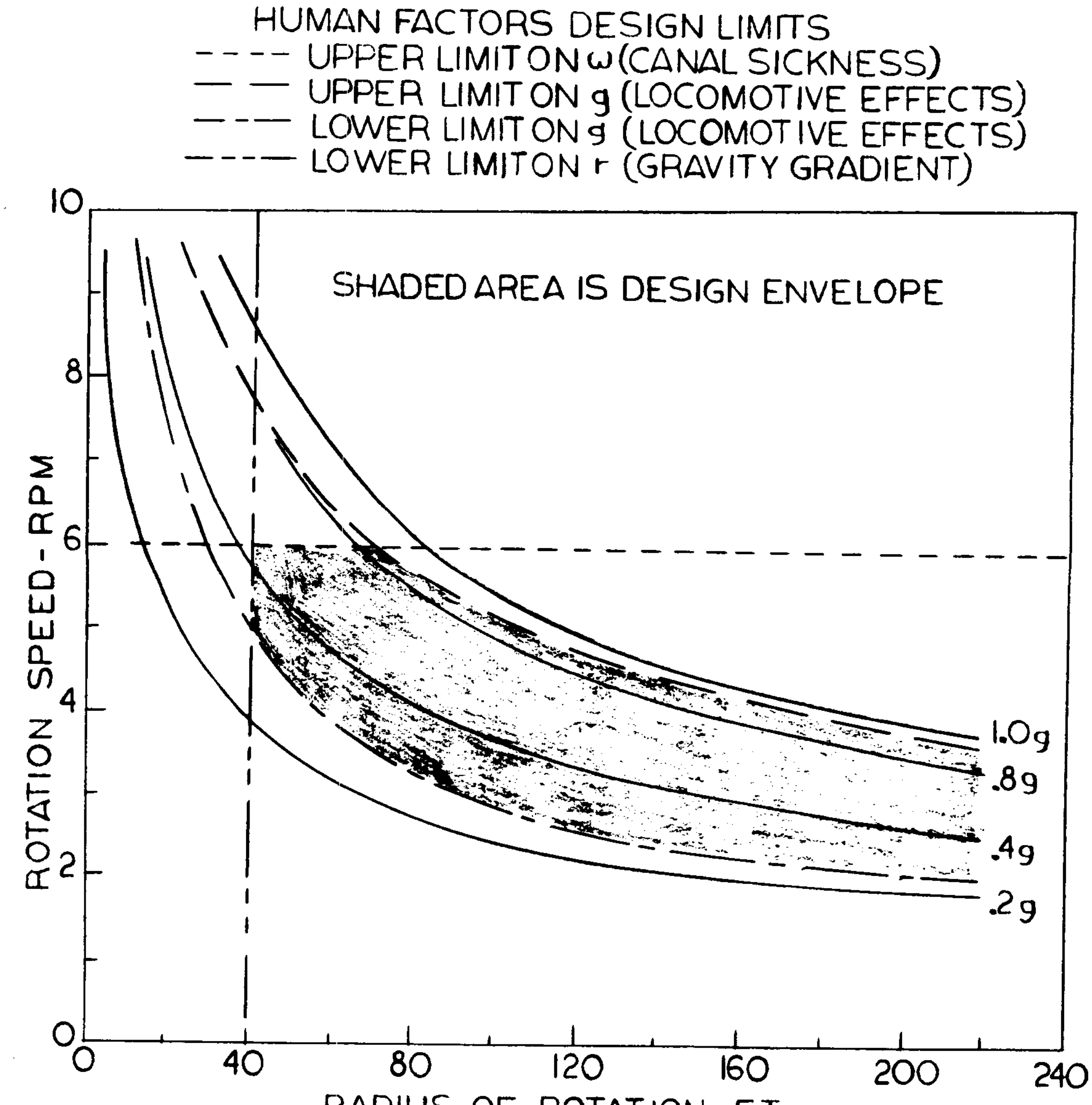
It is assumed that flight tests during the early AAP missions will provide the answer to whether or not the station must be rotated to provide artificial gravity. The present state of knowledge can be discussed in two parts; first, the possibility of medical problems due to long term zero gravity and secondly, the myriad of habitability problems which zero gravity can cause. The feeling of the medical people closest to the Gemini flights is that the body will probably adjust to zero gravity quite satisfactorily. At the present time, this opinion is based on one 14 day and several three to four day flights. One of the major purposes of the AAP program will be to investigate this problem in greater depth and over longer periods of time.

The habitability problem is, at present, the reason for considering artificial gravity. In essence, the argument is that it may prove cheaper to rotate the entire station than to design the spacecraft to handle the nearly countless special engineering tasks associated with operating a wide variety of experimental devices at zero gravity. This aspect of the problem must be answered before any complex station can be rationally created. Fortunately, this problem may be resolved more readily than the medical question because it doesn't require long flight durations.

If a zero gravity station is to be designed, the basic problem is that of selecting design criteria. The data available, upon which such criteria could be based, are not sufficient to do more than suggest certain probable design limits. Early flight tests must then be made to resolve this problem and to validate the small amount of ground based data available. It should be noted that this ground based data, by necessity, contains a one "g" component which may invalidate any initial conclusions. A summary of the criteria situation follows.

A human factors design envelope is presented in Figure 4.2. These data are discussed below.

a. Experimental data obtained by Dr. Graybiel at the U. S. Naval School of Aviation Medicine at Pensacola, Florida, have indicated the threshold of the occurrence of "canal sickness" to be approximately 3.82 rpm. However, the Life Sciences Department at General Dynamics Convair, San Diego, California, under the direction of Dr. Newsom has obtained data on a rotating vehicle that indicates man can adapt and function effectively at 6 rpm. Tests by both Dr. Graybiel and Dr. Newsom at higher rpm values have indicated problem areas. Therefore, a maximum rotation speed of 6 rpm is selected.



HUMAN FACTORS DESIGN ENVELOPE

FIGURE 4.2

b. To date, a majority of the aeromedical specialists and design engineers has selected a maximum value of 15 percent for the variation of the force acting on a man's head relative to the force acting at his feet. The effects produced by this head to foot gravity gradient are unknown at the present time. A six-foot man will experience this 15 percent variation at a 40-foot rotational radius. Based on this arbitrary value, a minimum rotational radius of 40 feet is selected.

c. Since man's natural environment is 1"g," this value was selected as the maximum force man should experience in a rotating space station. However, when man moves tangentially in the direction of rotation, the resultant force will be greater than that experienced when stationary. Hence, a maximum"g"level less than 1 "g" is shown in Figure 4.2.

d. Experimental data obtained by Beebe at Wright-Patterson Air Force Base have indicated a level of 0.2 "g" to be the minimum for walking. The decrease in the resultant force acting on man when tangential movement is opposite to the direction of rotation dictates the selection of a minimum "g" level slightly greater than 0.2 "g" as shown in Figure 4.2.

e. The human factors design envelope is open-ended since the maximum radius of rotation is based on practical engineering design rather than human factors.

f. An examination of the tolerance limit curves of Figure 4.2 indicates that the human factors design envelope is prescribed on three sides by the upper g limit, the lower g limit, and the upper limit on rotational speed. Since other human factors stress-limit curves, such as the curve for minimum rim velocity, lie outside the envelope, the stress limits they represent will not normally be exceeded.

g. In addition to the human factors parameters associated with the design envelope presented in Figure 4.2, the following design considerations are presented.

(1) Radial movement should be minimized due to the variation of the tangential velocity with the radial path.

(2) Activity at a rotating spin axis should be minimized since the centrifugal force component will be equal to zero and the resultant force will be equal to the Coriolis force.

(3) The living-working compartment should be oriented so the direction of traffic is parallel to the rotation axis. The crew duty-station positions should be oriented in a manner so that, during normal activity, the lateral axis through the crew member's ears is parallel to the spin axis. In addition, the work console instruments and controls should be designed to minimize the left-right head

rotations and up-down arm movements. The considerations will minimize the Coriolis effects.

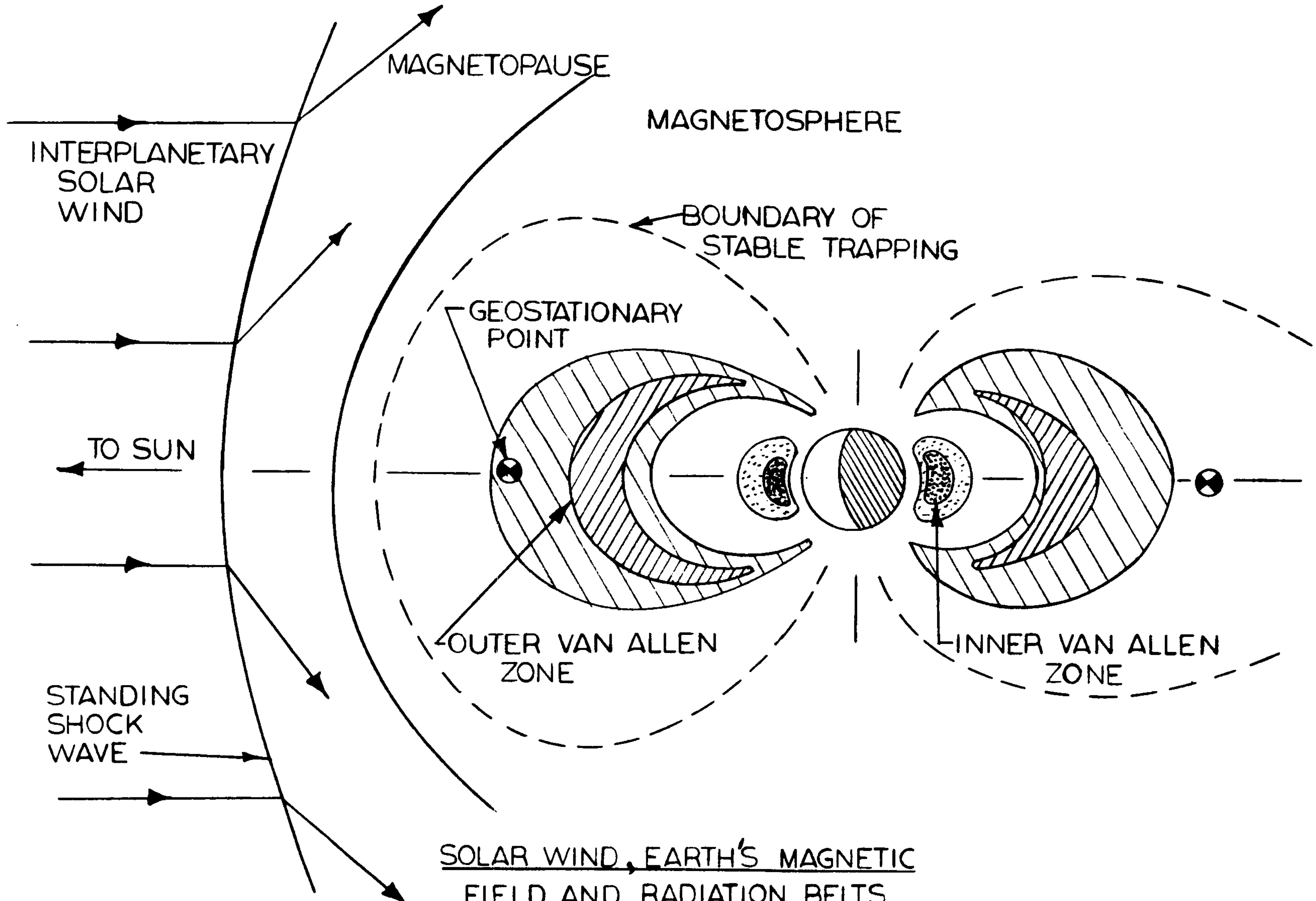
4.1.2 ENVIRONMENT

4.1.2.1 RADIATION ENVIRONMENT

4.1.2.1.1 Sources of Harmful Radiation

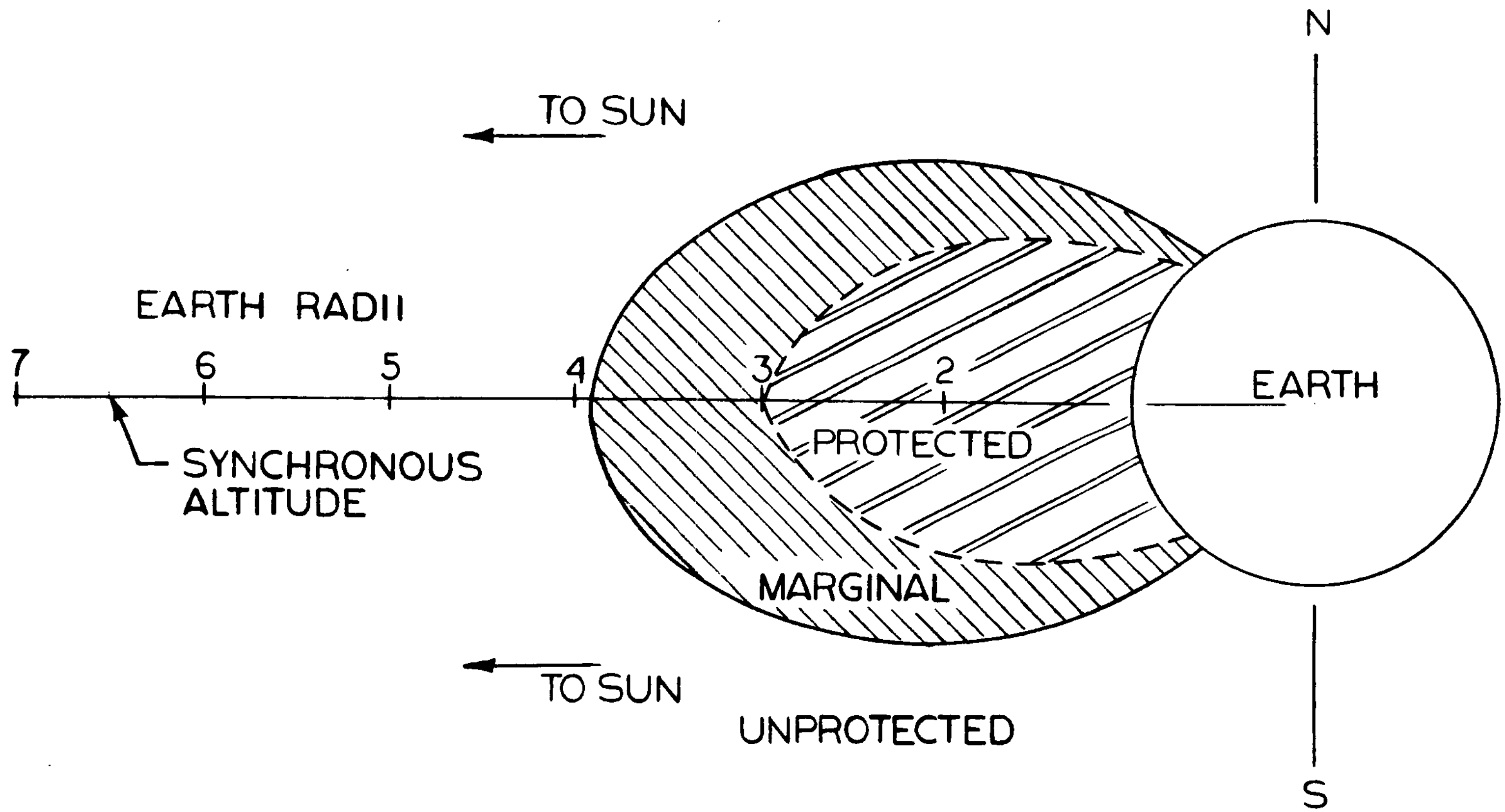
The two major sources of harmful radiation which must be considered in the selection of a space station orbit are the solar particle events and the radiation belts. The solar particle events consist of charged particles emitted by the sun which, upon encounter with earth, are deflected by the magnetic field. As a result of the partial shielding provided, solar radiation can be neglected for low altitude, low inclination orbits. The same forces which tend to divert solar radiation also tend to contain particles around the earth within so-called radiation belts. The level of harmful radiation in these belts is sufficient to preclude extended operation between the altitudes of 500 and 2500 nautical miles at the equator. The spatial relationship between the earth, the magnetosphere and the radiation belts is illustrated in Figure 4.3. Figure 4.4 shows the areas which are protected from solar particles.

Anomalies in the radiation pattern around the earth are created by a variation in the magnetic flux which in turn is caused by the asymmetry of the field. This asymmetry can be approximated



SOLAR WIND, EARTH'S MAGNETIC FIELD AND RADIATION BELTS

FIGURE 4.3



AREA PROTECTED FROM SOLAR PARTICLES BY EARTH'S MAGNETIC FIELD

FIGURE 4.4

by a dipole field which is displaced toward the Pacific Ocean. The resulting hazard is located in the South Atlantic Ocean as shown in Figure 4.5.

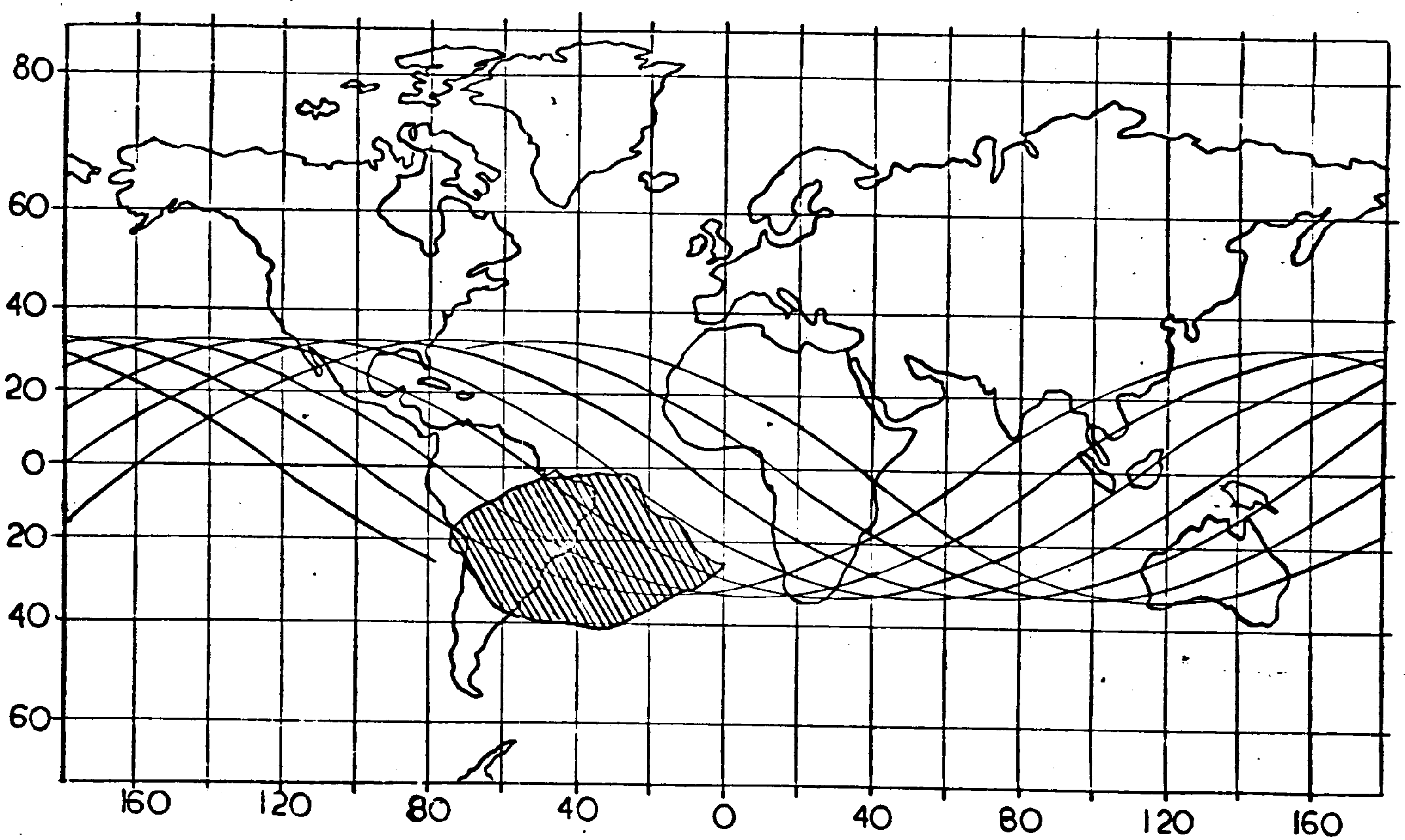
4.1.2.1.2 The Effect of Orbital Inclination

At altitudes of 500 nautical miles or less, the spacecraft is below the high radiation levels in the belt. Referring to Figures 4.4 and 4.5, it can be seen that for a zero degree inclination, the space station does not encounter the South Atlantic anomaly hazard and is well shielded from solar radiation by the magnetic field. The shielding effect of the magnetic field is essentially constant up to inclinations of about sixty degrees. At higher inclinations the shielding is not as predominant and the solar particles become an important part of the total radiation. The contribution of the South Atlantic hazard to the total radiation dose becomes significant at inclinations greater than zero degrees.

Preliminary calculations of total radiation dose for an altitude of 300 nautical miles, various shielding weights, orbital inclinations and exposure times are presented in Figures 4.6 through 4.8.

Figure 4.6 shows that the total dose received in a 30^{o} inclined orbit is greater than for a 60^{o} orbit. This is because the vehicle spends less total time in the South Atlantic hazard at 60^{o} than at 30^{o} and the magnitude of the radiation in the belts is not as great at 60^{o} as at 30^{o} for a 300 nm orbit.

28.5° ORBITAL TRACE



LOCATION OF RADIATION HAZARD IN SOUTH ATLANTIC ANOMALY
FOR 300 NM ALTITUDE

FIGURE 4.5

SHIELD WEIGHT VS RADIATION DOSE

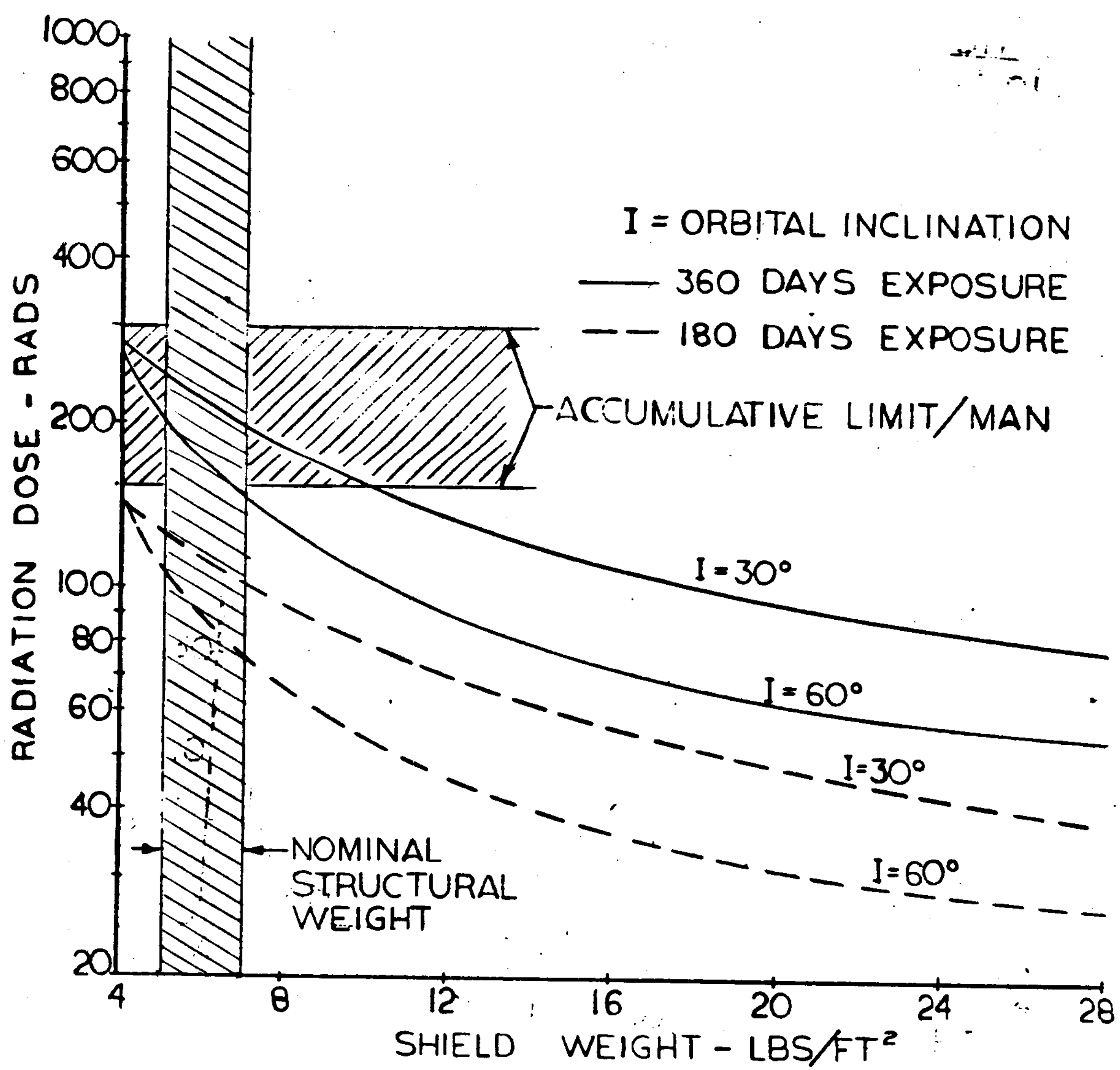


FIGURE 4.6

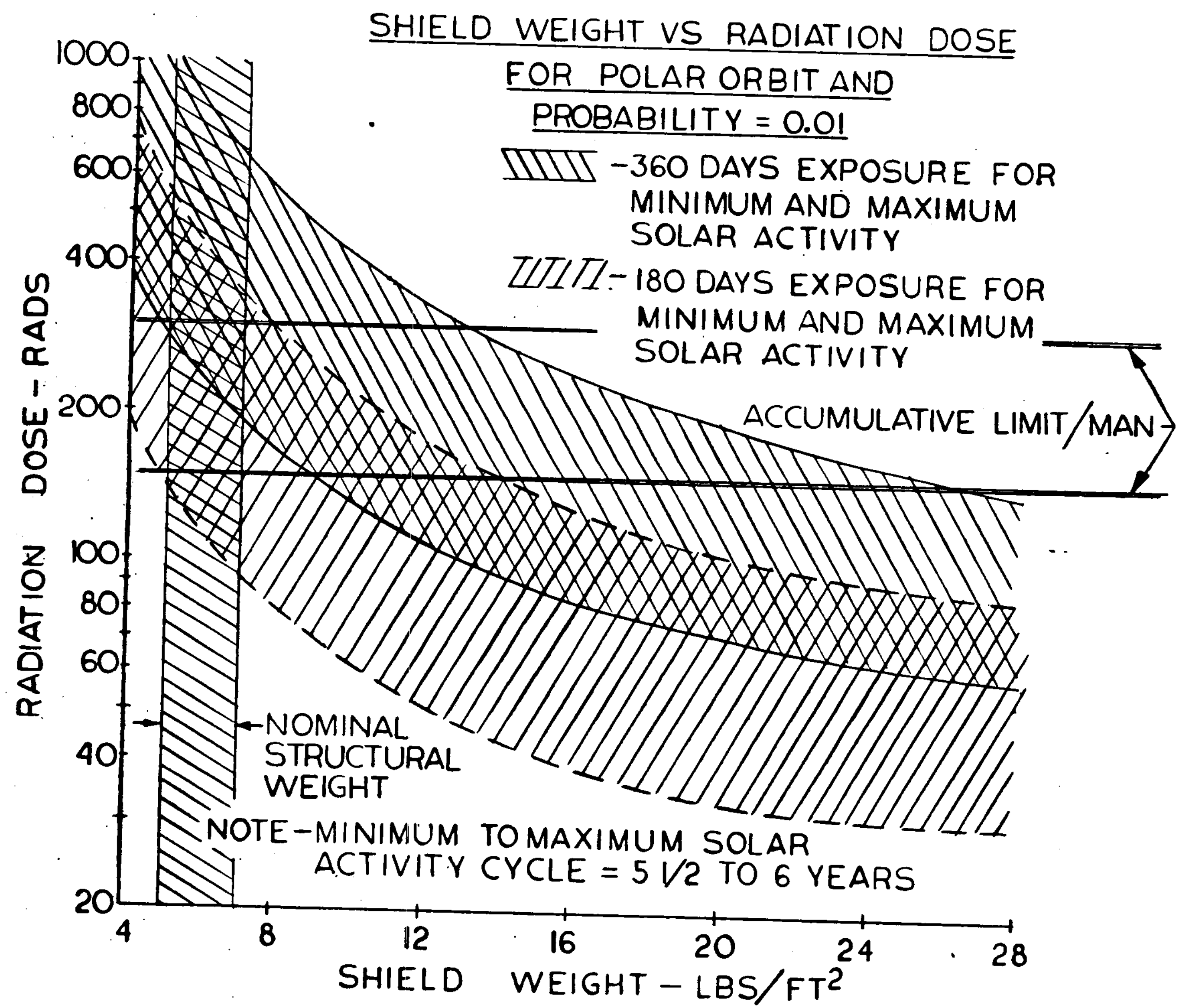

SHIELD WEIGHT VS RADIATION DOSE
FOR POLAR ORBIT AND
PROBABILITY = 0.01
-360 DAYS EXPOSURE FOR MINIMUM AND MAXIMUM SOLAR ACTIVITY
-180 DAYS EXPOSURE FOR MINIMUM AND MAXIMUM SOLAR ACTIVITY
ACCUMULATIVE LIMIT/MAN
NOMINAL STRUCTURAL WEIGHT
NOTE-MINIMUM TO MAXIMUM SOLAR ACTIVITY CYCLE = 5 1/2 TO 6 YEARS
RADIATION DOSE - RADS
1000
800
600
400
200
100
80
60
40
20
4
8
12
16
20
24
28
SHIELD WEIGHT - LBS/FT2

FIGURE 4.7

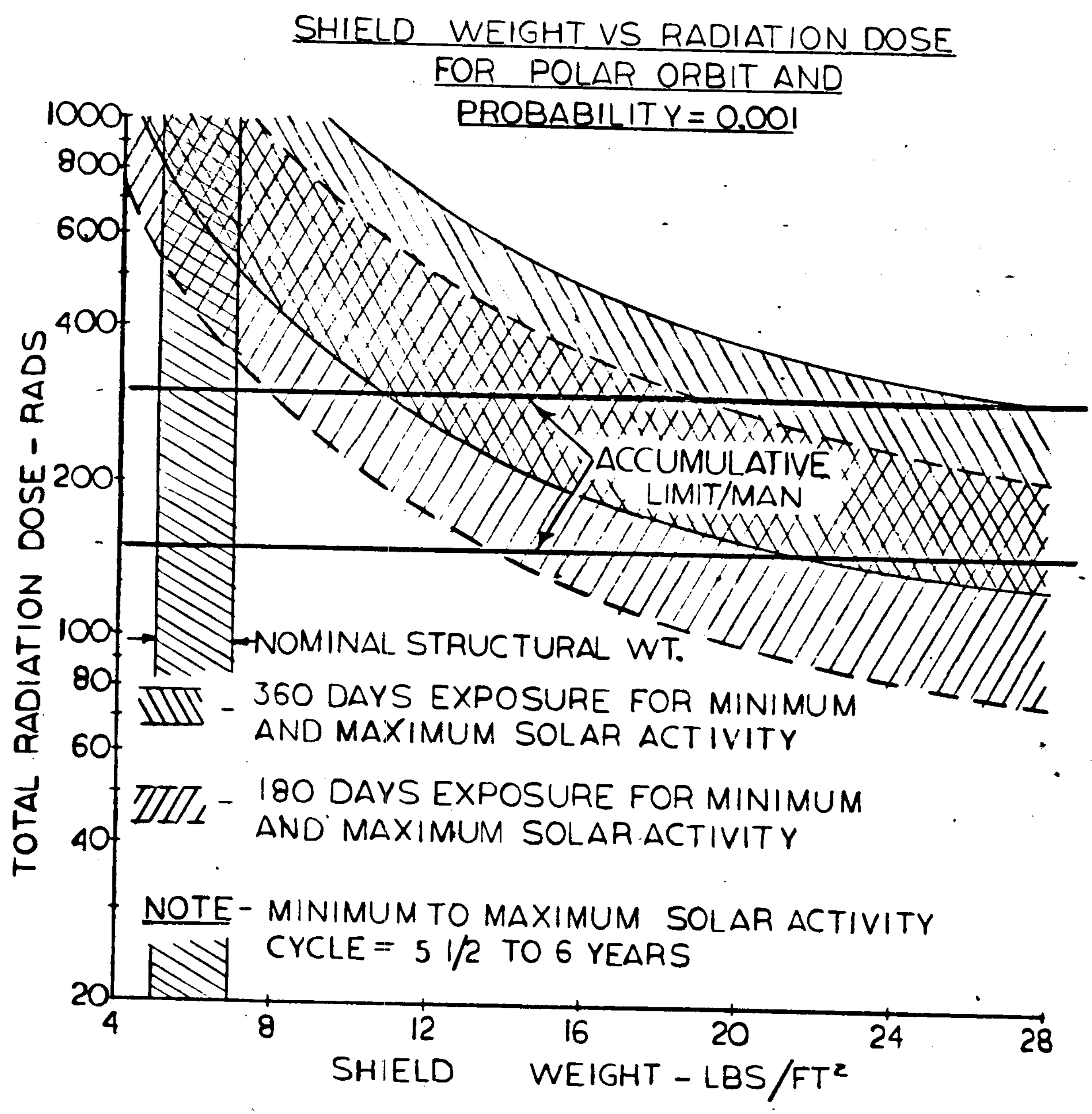

SHIELD WEIGHT VS RADIATION DOSE
FOR POLAR ORBIT AND
PROBABILITY = 0.001
TOTAL RADIATION DOSE - RADS
1000
800
600
400
200
100
80
60
40
20
4
8
12
16
20
24
28
SHIELD WEIGHT - LBS/FT²
ACCUMULATIVE
LIMIT/MAN
NOMINAL STRUCTURAL WT.
- 360 DAYS EXPOSURE FOR MINIMUM AND MAXIMUM SOLAR ACTIVITY
- 180 DAYS EXPOSURE FOR MINIMUM AND MAXIMUM SOLAR ACTIVITY
NOTE - MINIMUM TO MAXIMUM SOLAR ACTIVITY CYCLE = 5 1/2 TO 6 YEARS

FIGURE 4.8

A comparison of Figures 4.7 and 4.8 shows that for polar orbit a large weight penalty is incurred as the probability of exceeding a given dose level is decreased from 0.01 to 0.001. An additional weight penalty is incurred for operating during maximum solar activity. Four alternatives which would alleviate the weight problem are:

a. Plan the mission around a minimum solar activity time period.

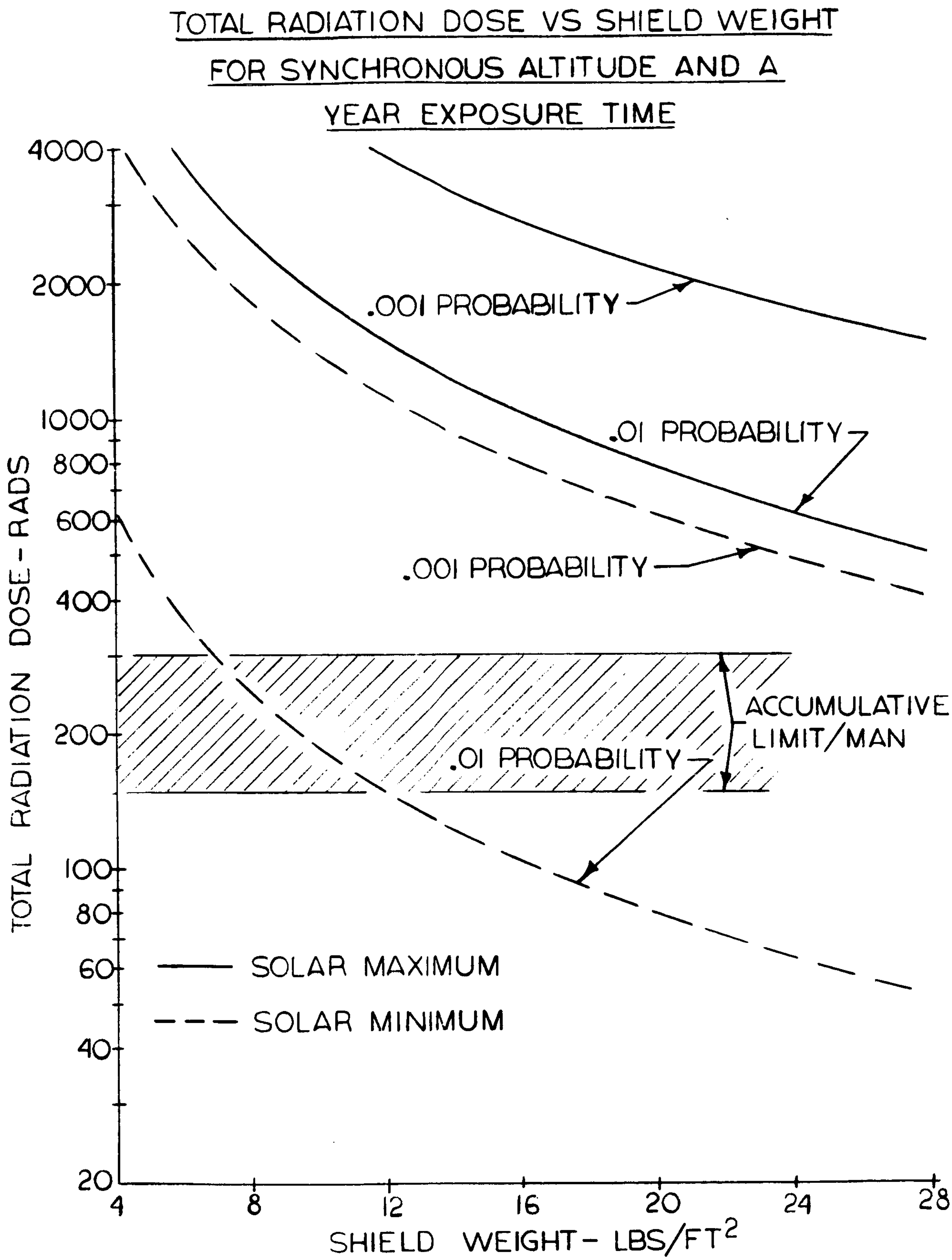
b. Frequent crew rotation.

c. Plan for station abort and relax probability numbers (several hours advance warning can be expected). The capability to abort the station may be necessary in any case due to the lack of confidence in solar data.

d. Minimize station shielding weight and provide an internal shelter for protection during an intense solar event. (Occupancy for approximately three days would be required.)

4.1.2.1.3 Considerations for Synchronous Orbit

A spacecraft in synchronous orbit (approximately 6.5 earth radii) is well beyond the protection of the earth's magnetic field as indicated in Figure 4.4. Since the exact nature of the interaction between solar particles and the magnetic field is not precisely defined at this altitude, it is recommended that the interplanetary environment be used for design purposes. The shield weight versus radiation dose based upon this assumption is shown in Figure 4.9.



TOTAL RADIATION DOSE VS SHIELD WEIGHT FOR SYNCHRONOUS ALTITUDE AND A YEAR EXPOSURE TIME
4000
2000
1000
800
600
400
200
100
80
60
40
20
4
8
12
16
20
24
28
TOTAL RADIATION DOSE - RADS
SHIELD WEIGHT - LBS/FT2
.001 PROBABILITY
.01 PROBABILITY
.001 PROBABILITY
ACCUMULATIVE LIMIT/MAN
.01 PROBABILITY
SOLAR MAXIMUM
SOLAR MINIMUM

FIGURE 4.9

The statements made in the previous section concerning the weight penalties for polar orbit are also valid for this case.

4.1.2.1.4 Human Tolerances and Operational Considerations

There are several factors which contribute to the definition of the maximum allowable radiation dose for the crew. Among these are the type of radiation, the area of the body which is exposed and the rate at which the dose is received. Sufficient information is not presently available to specify such a maximum value; however, for the purpose of this study a nominal value of 300 rads total dose in one year is recommended.

The only operational constraint imposed by the radiation environment (other than possible crew rotation requirements) will be certain restrictions on Extra-Vehicular Activity (EVA). In low inclination (up to 60°), low altitude orbits, EVA must not be scheduled while the spacecraft is passing through the South Atlantic hazard. This will amount to a few minutes during each revolution. For polar orbit, the same criterion applies; but, in addition, the occurrance of a solar event will require that EVA be avoided for that portion of the orbit which is not protected by the magnetic field (see Figure 4.4).

4.1.2.2 METEOROID ENVIRONMENT

The flux of meteoroids encountered in space is composed of varying sporadic flux and brief but intense showers. Of the space station missions considered, synchronous orbit is slightly more hazardous due to the reduced effect of earth shielding. This mission, along with an assumed surface area of 10,000 ft^2, was used for the shielding calculations.

The model assumed for calculation of the near earth meteoroid flux, including sporadic flux and showers, and compensating for gravitational concentration, was: $N = 10^{-3.83} m^{-1.34}$ (for $m \geq 10^{-2}$) and $N = 10^{-3.15} m^{-1}$ (for $m < 10^{-2}$) where N is the flux per $10^4 ft^2$-years of meteoroids of mass greater than or equal to m grams.

4.1.2.2.1 Definition of Probability

Once a flux model has been established and an area-time product chosen, the probability that a meteoroid of a given mass or larger will impact the station can be calculated. To associate this probability with that of penetration, the structure is designed such that a meteoroid of a given mass (associated with a given probability of occurrence) will not penetrate, but one of a slightly larger size will penetrate. The smaller of the two is then defined as the threshold mass for a stated probability of penetration.

The terminology associated with the probabilities is then;

P_o is the probability of no penetrations,

P_1 is the probability of no more than one penetration,

P_5 is the probability of no more than five penetrations,

and

P_o = .999 means that there is one chance in a thousand of encountering one or more meteoroids larger than the P_o threshold size,

P_1 = .999 means that there is one chance in a thousand of encountering two or more meteoroids larger than the P_1 threshold size,

P_5 = .999 means that there is one chance in a thousand of encountering six or more meteoroids larger than the P_5 threshold size.

4.1.2.2.2 Calculation of Shield Weights

An average meteoroid velocity of 30 km/sec and mass density of 0.5 gm/cm^3 was used in determining the shielding requirements. The aluminum shielding S (lb/ft^2) to prevent penetration of an impacting mass m (grams) is given as

$$S = 41.51\ K\ m^{0.352} + C_K.$$

For multiwall or bumper configurations the structural efficiency factor K is defined as the ratio of the total thickness of the number of sheets required to prevent penetration to the theoretical single sheet thickness. For a double wall of two inch spacing with a filler material (low density open-celled foam) K = 1/7 and C_K = .225, where C_K represents the added weight of the filler material. These values were used in Figure 4.10 to calculate the shielding weights.

4.1.3 OPERATIONS

For the purpose of this study, only the broad categories of

METEOROID SHIELDING - NEAR EARTH AND SYNCHRONOUS ORBITS

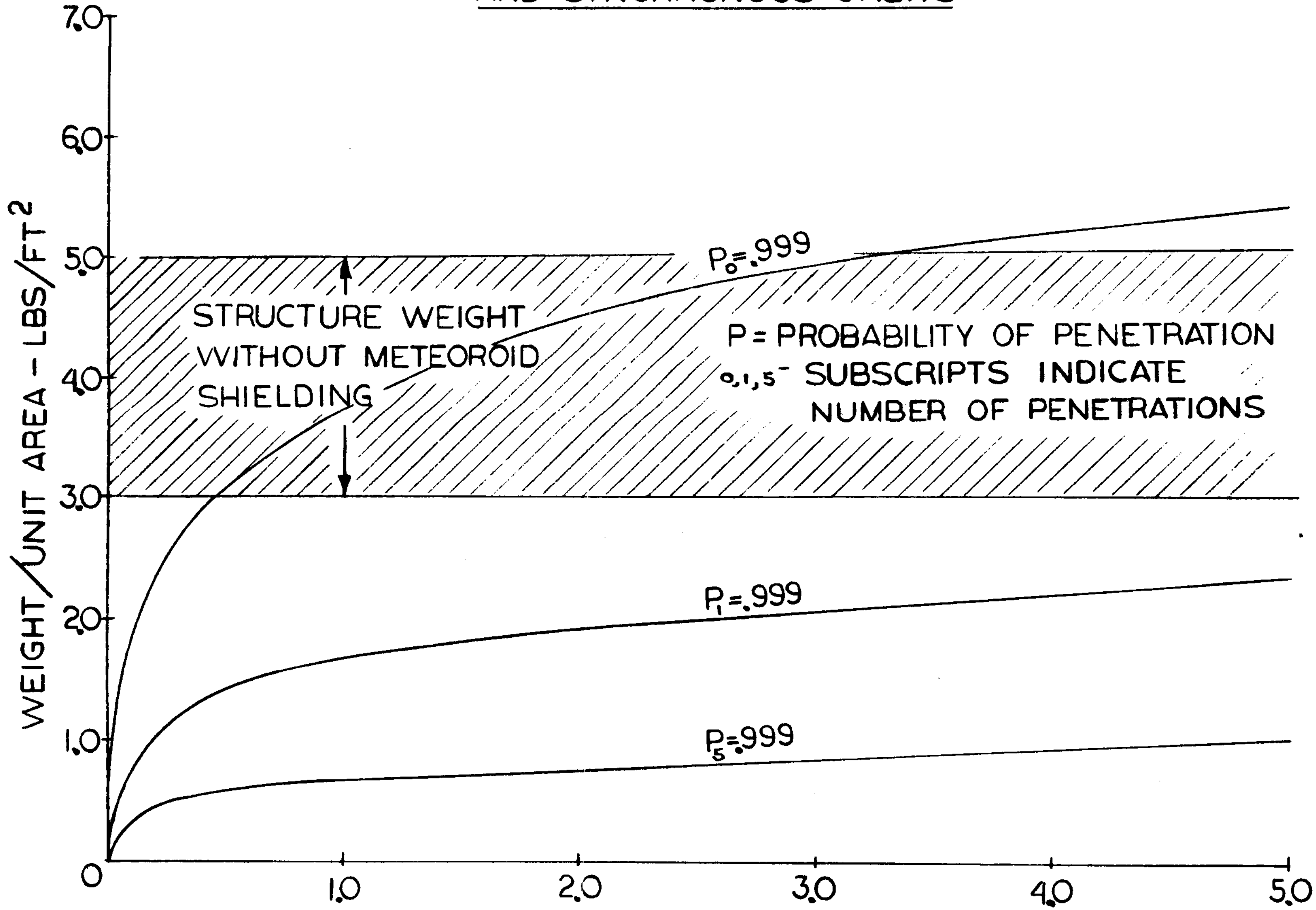


FIGURE 4.10

logistics and crew time allotments were considered. Administration duties, experiment planning and implementation, and the crew designation by scientific disciplines are to be derived when more data are available. Also, the interface between the ground and the station will be deferred, and the technical implications of the station operations will be emphasized.

4.1.3.1 LOGISTICS REQUIREMENTS

The following paragraphs describe the categories of logistics and provide some preliminary estimates of station housekeeping requirements. Quantitative logistics estimates for experiments are not included in this section.

4.1.3.1.1 Logistics for Experiments

Experiment logistics is primarily concerned with the transportation of experiment equipment, specially trained personnel and scientists, and special consumables and/or reactants. Special experiment-related test, maintenance and installation equipment are also included in this category. Resupply for some experiments may be stringent depending on the duration of the experiments and the consumption rate of special experiment-related consumables.

4.1.3.1.2 Logistics Requirements for Space Station Stabilization and Orbit Maintenance

Propellant and pressurant resupply is determined from reaction control and orbit decay requirements, the size and activity of the crew, and attitude hold and orientation requirements. Periodic propellant tank replacement may be required because of bladder recycle limitations.

If orbit maintenance corrections are to be performed with the logistics vehicle's propulsion system, additional propellant will have to be provided for this operation.

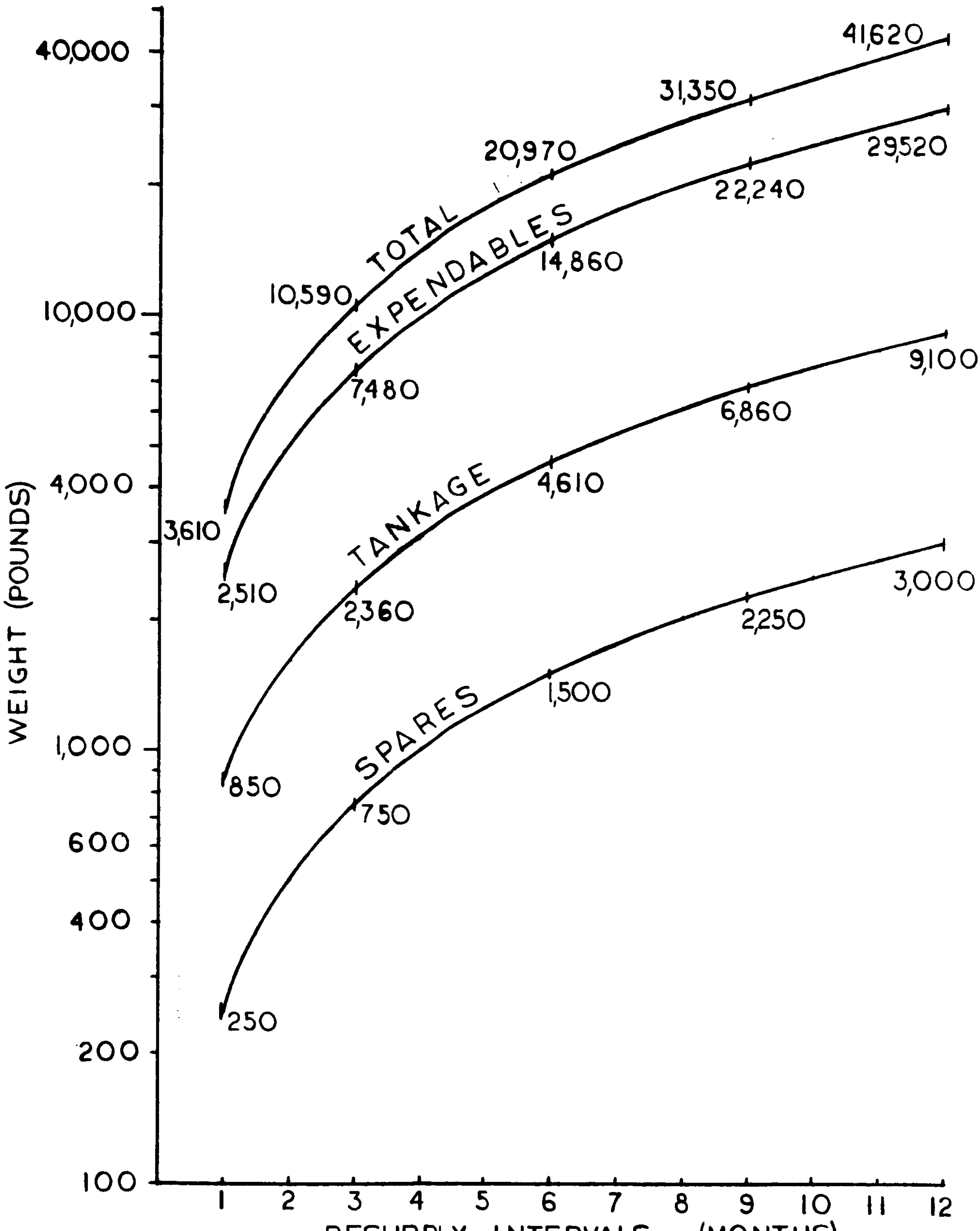
4.1.3.1.3 Logistics Requirements for Space Station Housekeeping

The major resupply items of this area are cryogenics, food, spares, and space station personnel. Logistics requirements for cryogenics require sufficient fluid quantities to account for space station resupply plus venting and transfer line losses. The cryogenics will require positive expulsion transfer or complete tank replacement and subsequent disposal of the replaced tanks.

Figures 4.11 and 4.12 indicate the estimated housekeeping resupply quantities required versus time for crew sizes of 9 and 24 men. Included in the curves are spares, tankage and expendables.

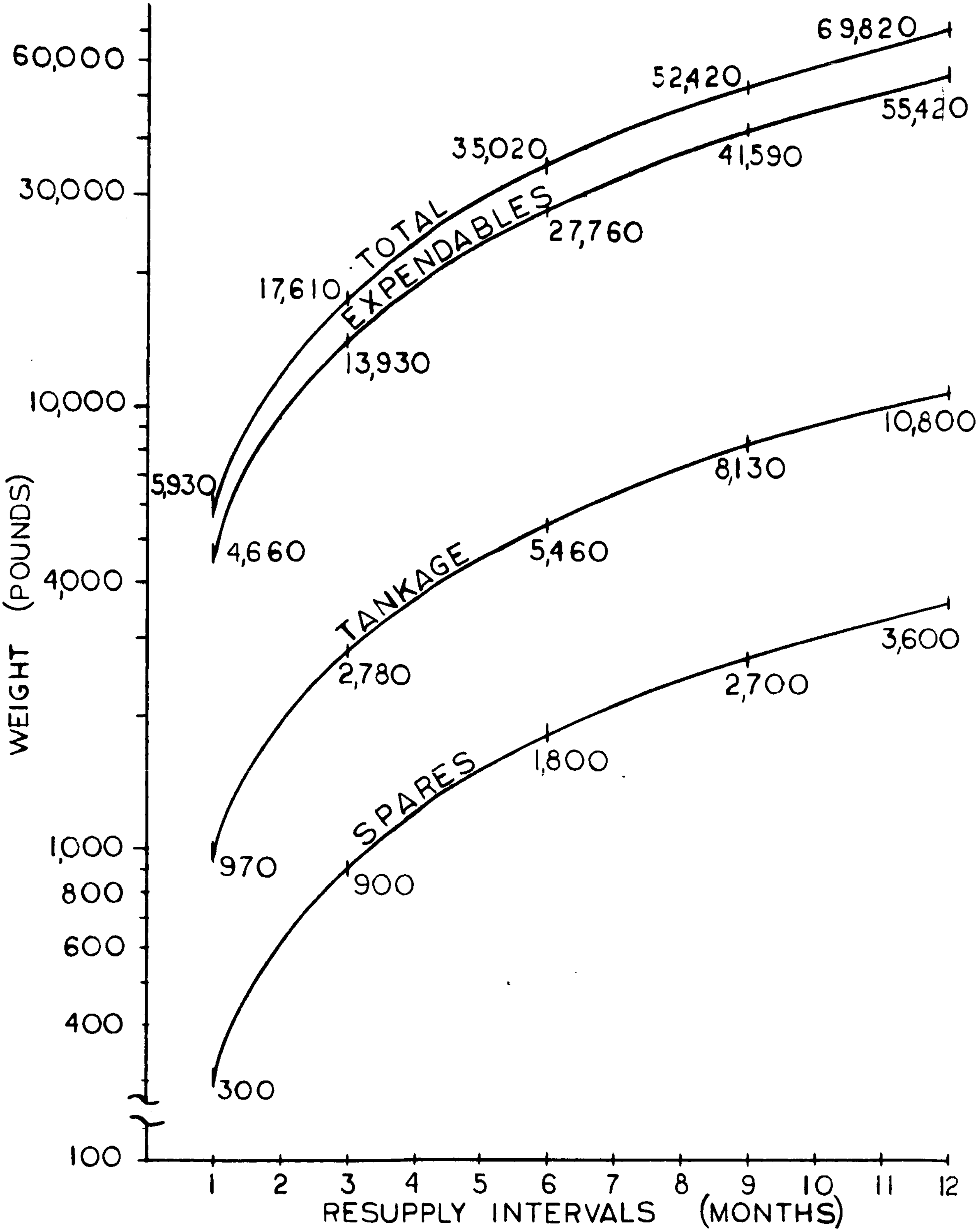
4.1.3.2 CREW TIME ALLOTMENT

The crew tasks are defined as personal, station operation and maintenance and experiment activity. Initial time allotments are provided in the following sections.



9 MAN SPACE STATION
RESUPPLY REQUIREMENTS

FIGURE 4.11



24 MAN SPACE STATION RESUPPLY REQUIREMENTS

FIGURE 4.12

4.1.3.2.1 Personal Tasks

Personal tasks consist of sleeping, personal hygiene, physical fitness, health checks, recreation, relaxation, and eating. An allotment of 787 minutes per day for each man has been estimated for these activities.

4.1.3.2.2 Station Operation and Maintenance

Station operation and maintenance consists of the following activities:

- Station Management
  - Orbit Keeping
  - Navigation
  - System Monitoring
  - Rendezvous/Docking
  - Station/Experiment Planning
- Communication/Data Management
  - Formating
  - Screening
  - Sequencing
- Maintenance
  - Systems
  - Structure

An allotment of 1920 man-minutes per day has been established for a nine man crew.

The EVA time allotment has been based on the assumption of one EVA per 21 man-days. An allotment of 27 man-minutes per man-day has been established for this activity.

4.1.3.2.3 Experiment Activity

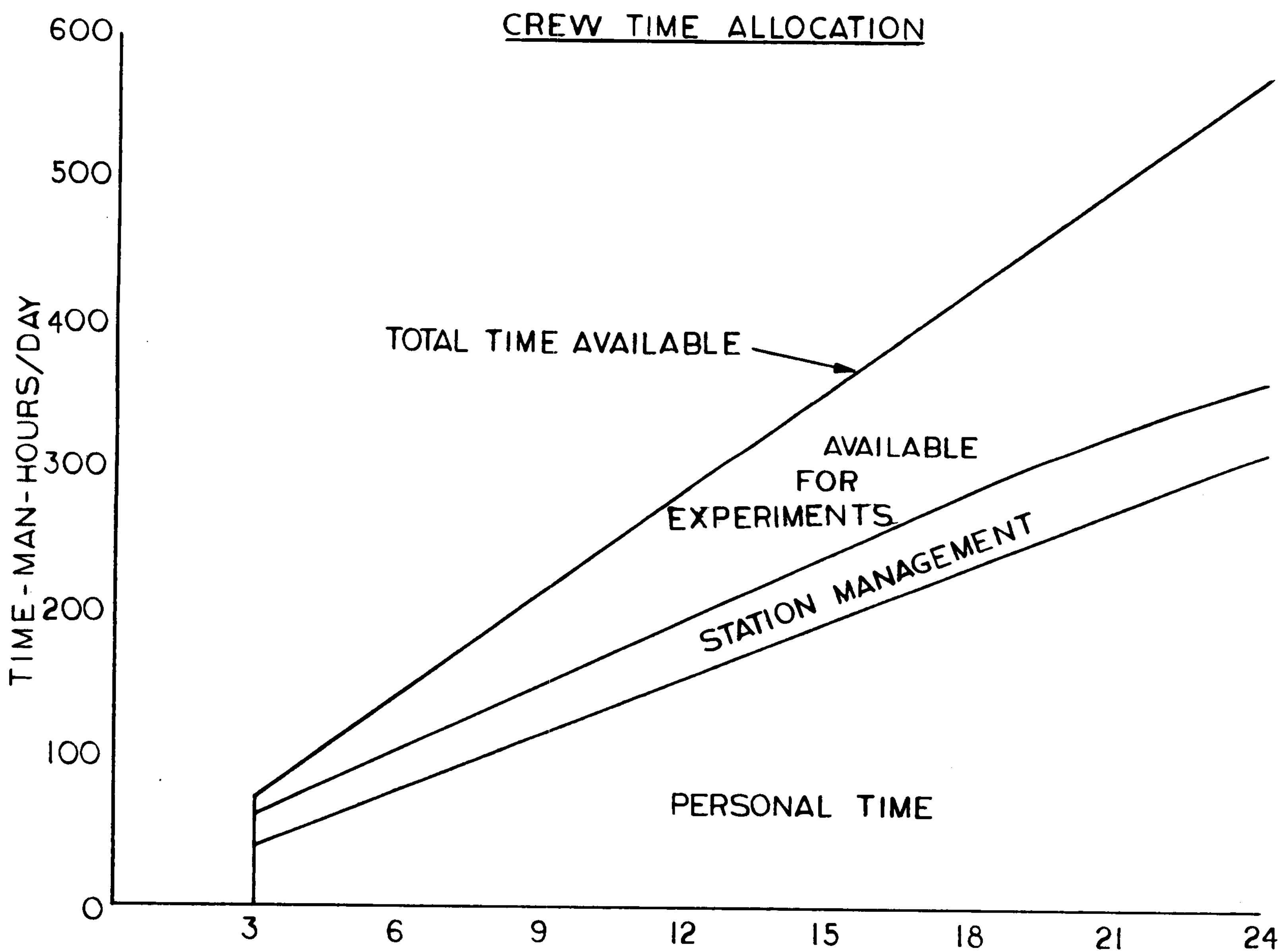
The time available for experiments (including EVA) is dependent on the crew size and the crew activities defined above. The data in Figure 4.13 indicate the variation of available experiment time with crew size using the above allotments.

4.1.4 RELIABILITY AND MAINTENANCE

Achievement of a high probability of mission success in a complex spacecraft, such as a large space station, dictates very high reliability of all components, extensive redundancy, onboard spares, or some combination of these. The reliability/maintenance philosophy is a function of the mission; that is, the systems approach for a planetary spacecraft will differ from that for a space station to which spares can be readily resupplied. In the present case, although resupply of spares will be available, it must be considered that the same systems will probably be used for a planetary mission; whole modules, in fact, may be duplicated for the space station and planetary missions. It follows that the reliability approach should represent a combination of the two.

Accordingly, components will be designed to operate for the full length of the space station life wherever possible. This will minimize spares requirements. At the same time, these components will not be greatly overdesigned for planetary missions, since a five-year component does not in general differ significantly from a two-year component.

Regardless of design wear-out life, failures will occur prematurely. In addition, it is not practical to conduct real-time



CREW TIME ALLOCATION
TOTAL TIME AVAILABLE
AVAILABLE FOR EXPERIMENTS
STATION MANAGEMENT
PERSONAL TIME
TIME-MAN-HOURS/DAY
0
100
200
300
400
500
600
CREW SIZE
3
6
9
12
15
18
21
24

FIGURE 4.13

life tests on systems prior to launch due to the mission length involved. Consequently, in-flight maintenance will be required and must be provided for during design. These provisions will include redundancy in those systems which must operate continuously so as to permit component replacement without interruption of necessary functions.

4.2 SYSTEMS

The guidelines for systems selection are: use projected state-of-the-art technology, obtain maximum maintainability, and provide redundancy where required. The following sections provide a brief description of each system.

4.2.1 ELECTRICAL POWER SYSTEM

The Electrical Power System (EPS) for a nine-man space station has been sized for an average power output of 15 kw based on the requirements shown in Table 4.2.

The primary systems considered for this study were silicon solar cells, radioisotopes and nuclear reactors. Thermoelectric, Brayton cycle and mercury Rankine cycle conversion systems were studied for the isotope and nuclear systems. Regenerative fuel cells were considered for secondary and peaking requirements but were not competitive with batteries on a weight, cost, or volume basis.

Figures 4.14, 4.15 and 4.16 illustrate comparative weights, radiator areas and internal volumes for the systems considered. The radioisotope/Brayton cycle system is competitive with all others, but isotope availability for the 1973 time period restricts average power to about 10 kw. Thermoelectric conversion lowers this limit to 4 kw. Radioisotope power systems are, therefore, not recommended for this application.

TABLE 4.2

ELECTRICAL POWER REQUIREMENT

	Average Power, kw
Environmental Control System	2.0
Guidance & Control	0.6
Crew Systems	0.1
Communications & Data Management	1.9
Lighting	0.8
Instrumentation	0.5
Experiments	5.7
Contingency	3.4
Total	15.0

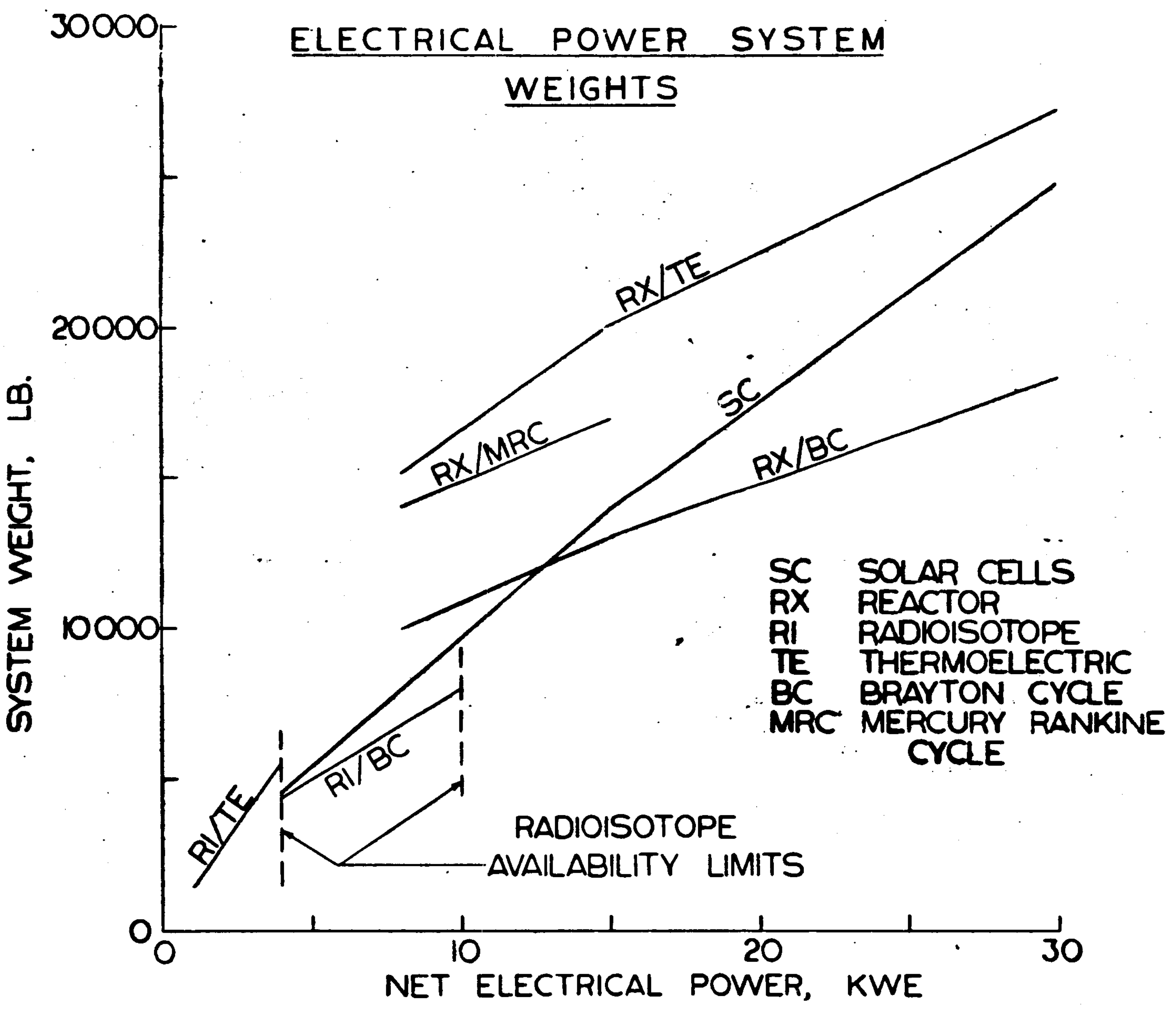

ELECTRICAL POWER SYSTEM
WEIGHTS
30000
20000
10000
0
SYSTEM WEIGHT, LB.
0
10
20
30
NET ELECTRICAL POWER, KWE
RX/TE
RX/MRC
SC
RX/BC
RI/TE
RI/BC
RADIOISOTOPE
AVAILABILITY LIMITS
SC SOLAR CELLS
RX REACTOR
RI RADIOISOTOPE
TE THERMOELECTRIC
BC BRAYTON CYCLE
MRC MERCURY RANKINE CYCLE

FIGURE 4.14

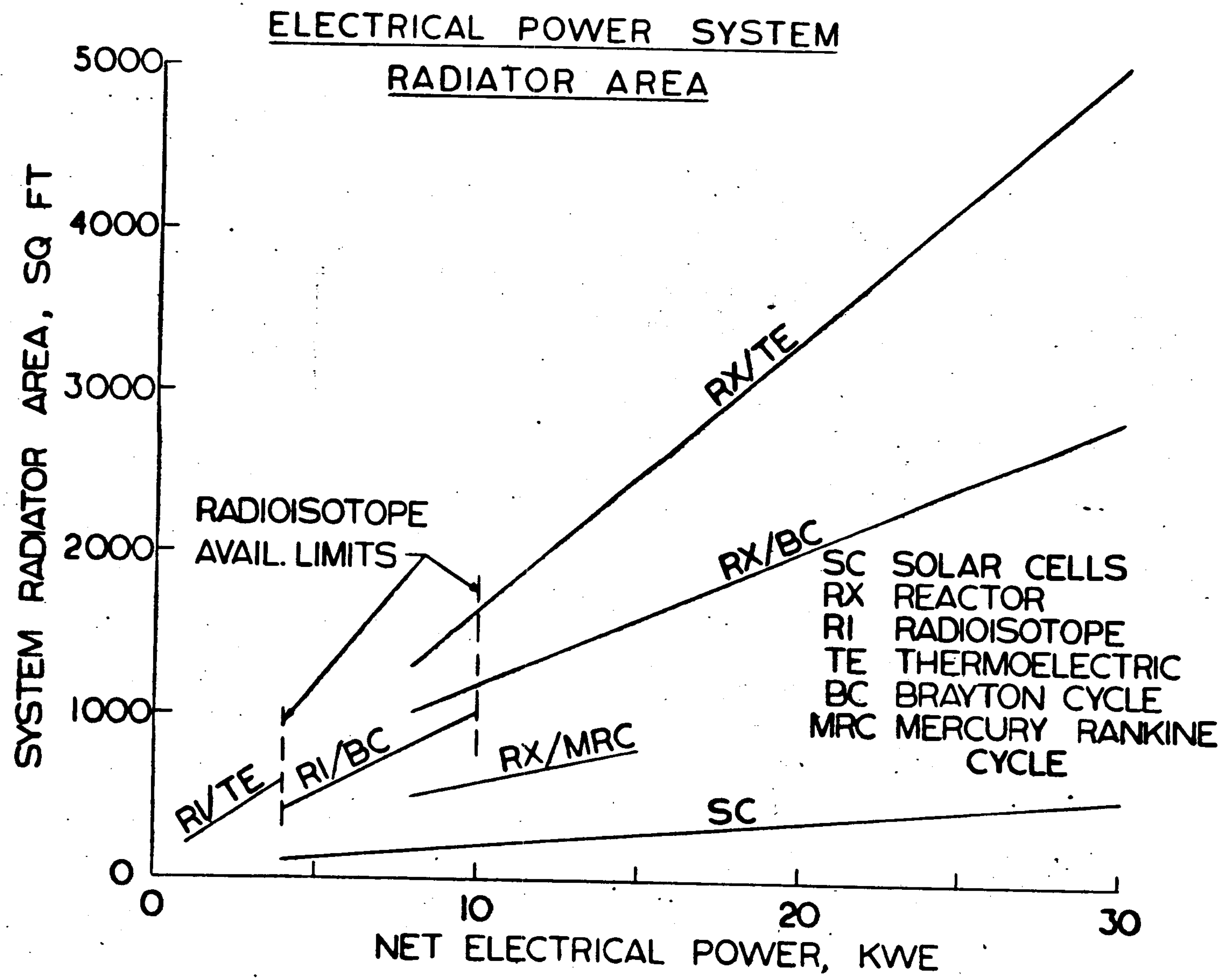
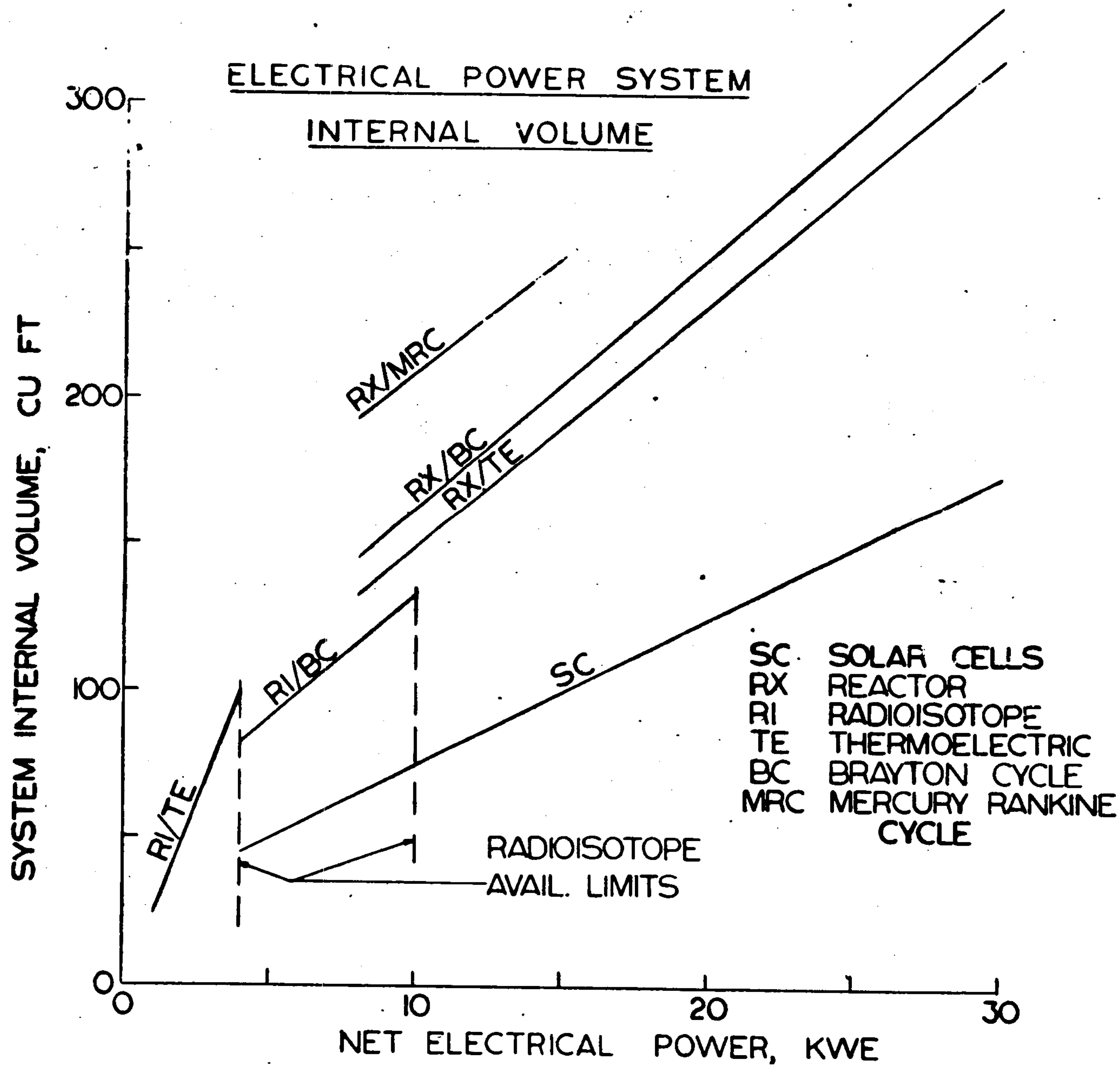

ELECTRICAL POWER SYSTEM
RADIATOR AREA
SYSTEM RADIATOR AREA, SQ FT
5000
4000
3000
2000
1000
0
0
10
20
30
NET ELECTRICAL POWER, KWE
RADIOISOTOPE
AVAIL. LIMITS
RX/TE
RX/BC
RI/TE
RI/BC
RX/MRC
SC
SC SOLAR CELLS
RX REACTOR
RI RADIOISOTOPE
TE THERMOELECTRIC
BC BRAYTON CYCLE
MRC MERCURY RANKINE CYCLE

FIGURE 4.15



ELECTRICAL POWER SYSTEM
INTERNAL VOLUME
SYSTEM INTERNAL VOLUME, CU FT
300
200
100
0
0
10
20
30
NET ELECTRICAL POWER, KWE
RX/MRC
RX/BC
RX/TE
RI/BC
RI/TE
SC
RADIOISOTOPE
AVAIL. LIMITS
SC SOLAR CELLS
RX REACTOR
RI RADIOISOTOPE
TE THERMOELECTRIC
BC BRAYTON CYCLE
MRC MERCURY RANKINE CYCLE

FIGURE 4.16

Of the nuclear reactor systems, Brayton cycle conversion appears most attractive. It is competitive with solar cells from a standpoint of weight and is substantially lighter if high power levels are required (see Figure 4.14). Internal volume and radiator area are not prohibitive. However, availability of a nuclear energy source by 1973 is questionable, and system cost will be substantially higher than a solar cell system if the reactor cost is included. In addition, some experiments require very low radiation levels which may be difficult to achieve with a nuclear system.

The principal disadvantage of the solar cell system is the large deployed area (4300 ft^2 for a 15 kw system in a 260 n.m. orbit), which requires an estimated 1400 lb or more per year additional propellant for orbit maintenance. The requirement for solar orientation places an additional constraint on station attitude.

Considering all factors, solar cells appear to have the advantage at this time and have been selected for purposes of this study. However, the reactor/Brayton cycle system warrants further study before a final decision is made. The remainder of the EPS need not depart significantly from current technology.

4.2.2 ENVIRONMENTAL CONTROL/LIFE SUPPORT SYSTEM

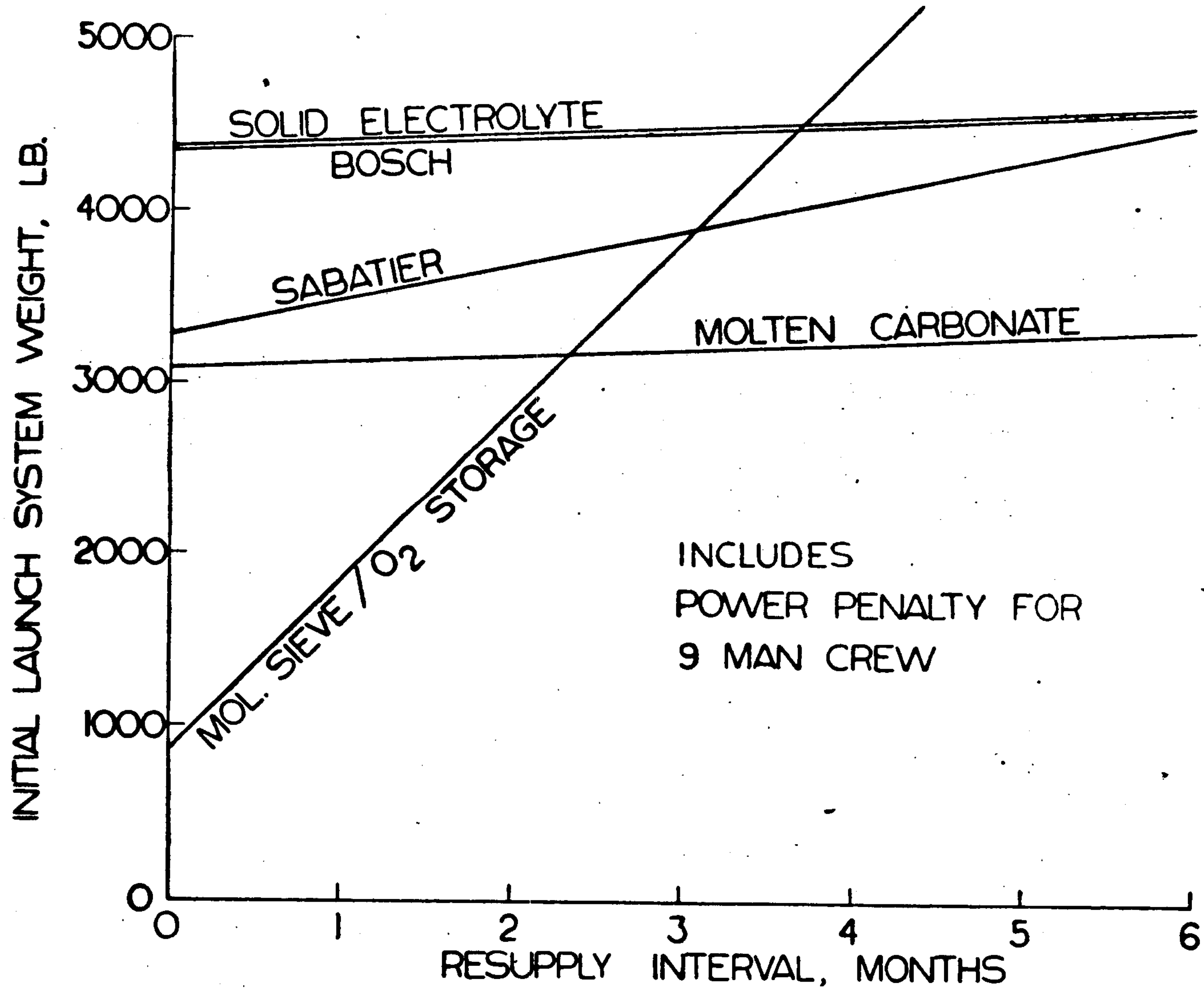
The Environmental Control/Life Support System (EC/LSS) consists of an atmospheric and thermal control circuit and a water and waste management circuit.

Water reclamation is virtually mandatory to avoid excessive logistic requirements. Oxygen regeneration from carbon dioxide (CO_2) is also attractive from the logistic standpoint. However, electrical power requirements are high (approximately 1.1 kw for nine men) and an extensive hardware development program will be necessary. In addition, the Sabatier process requires cryogenic storage of hydrogen. Due to the cost and complexity involved, oxygen regeneration is not believed to be warranted at this time. Comparative weight estimates for several possible regeneration systems as a function of resupply interval are shown in Figure 4.17, based on data by D. C. Popma of Langley Research Center. At this time, the Sabatier process appears to be the most practical, but the others should also be considered in the event a decision is made to employ oxygen regeneration.

The following sections describe the components of the system, their function and some of the integration factors.

4.2.2.1 ATMOSPHERIC REGENERATION CIRCUIT

The cabin gaseous environment is revitalized by means of carbon dioxide (CO_2) absorption, noxious and toxic gas removal, filtering, water vapor control, and thermal dissipation. This atmospheric regeneration circuit utilizes a blower system, condenser-heat exchanger, mechanical water separator, contaminant removal circuit, the CO_2 management circuit, filters and the necessary controls. Makeup for leakage is also provided through the regeneration circuit from cryogenic stores. The system



CO_2 REDUCTION VS. O_2 STORAGE

FIGURE 4.17

also contains a separate gas compressor, a lithium hydroxide CO_2 removal system and suit connectors for emergency crew support.

Removal of odors and trace contaminants from the cabin atmosphere is accomplished by an absorption bed. Contaminants which are not readily absorbed (e.g., hydrogen, carbon monoxide, and methane) are controlled by subsequently directing a small portion of the atmospheric flow through a catalytic oxidizer.

4.2.2.1.1 CO_2 Removal Circuit

CO_2 removal is accomplished by a four-bed regenerable solid absorption system which utilizes silica gel as a dessicant and molecular sieves (or zeolites) for CO_2 removal. The CO_2 is rejected to vacuum. The water is desorbed from the silica gel and returned to the cabin atmosphere.

4.2.2.1.2 Atmospheric Thermal Circuit

The cabin atmospheric thermal circuit maintains a reasonable environment for the crewman while dissipating heat from the sun, non-coldplated electronics and atmospherically cooled experiments. This is accomplished with two high flow blowers in conjunction with a plate fin/integral wick heat exchanger. A heating mode is also included in the cabin heat exchanger for the case where there is no solar heat and low internal thermal loads exist.

4.2.2.1.3 Equipment Cooling Circuit

Thermal control is provided by a coolant loop which serves the atmospheric regeneration circuit, cabin cooling, CO_2 removal system, and the electronic equipment. Heat rejection is accomplished with a space radiator. Contingency EC/LSS cooling will be provided by a water evaporative heat exchanger.

4.2.2.2 WATER AND WASTE MANAGEMENT CIRCUIT

The integrated water and waste management circuit reclaims body wash water, collects and processes human liquid and solid wastes to provide potable water, and sterilizes the condensed respired and perspired water for drinking and/or washing. The operation is largely automatic except during an actual defecation or urination when the flush and rinse valves must be cycled by the user to clean himself and the equipment.

The water from the four major contaminant sources is processed with separate systems; however, the resulting system is an integrated water and waste management circuit. The majority of the wash water is reclaimed in a membrane diffusion unit which retains the brine after processing. This brine and the feces flush is purified in a vacuum distillation system which provides makeup for the wash and drinking water, and also replenishes the fecal flush.

The urine and urine flush are also processed in an identical vacuum distillation water system to provide drinking water and sustain the cycle.

The humidity condensate is only sterilized before storage since no chemical impurities will be present in this water except those that are absorbed from the atmosphere.

4.2.3 CREW SYSTEMS

Crew systems consist of personal and support equipment required for the comfort and well-being of the crew. This includes food preparation and storage, living accommodations, personal care, clothing, pressure suits, Portable Life Support Systems, medical kits, etc. These systems will be similar in many areas to those in current programs; however, some new requirements arise from the mission length being considered.

Laundry facilities will be provided for clothing and bedding, saving approximately 10 lb/man-month in resupply weight. Development does not appear difficult, particularly for the artificial gravity station. A shower is provided for crew bathing. Since water is reclaimed, the weight penalty for these appliances is primarily the equipment itself.

Refrigerated storage and an oven will also be required to accommodate the frozen foods included in the diet.

4.2.4 GUIDANCE AND CONTROL SYSTEM

The function of the Guidance and Control (G&C) system will be to accept navigation data from either the ground stations or the crew, display data to the crew, and provide attitude control for all phases of the mission. The following is a functional description of the G&C.

Initial stabilization of either the zero gravity or artificial gravity space station will be accomplished by referencing the station to the sun. The sun sensor provided for initial "lock-on" and subsequent solar tracking will also be used for periodic alignment of the Inertial Measurement Unit. The spin axis of a solar oriented artificial gravity station will be precessed periodically to maintain proper orientation.

When the station has attained the desired orientation, a set of control moment gyros will be activated for attitude control. The control moment gyros can remove small perturbations over a long period of time and, thereby, reduce the reaction control propellant requirements.

During normal orbital operations, the computer will receive attitude and/or navigation data from either the Inertial Measurement Unit or the Display Keyboard. After processing the data, the computer feeds the appropriate signal back to the inertial unit or to the Control Electronics Assembly. The control electronics in turn provide the necessary signals to the attitude control system.

Circuit margin calculations show that omni antennas can be used acceptably for the 260 n.m. orbit. The use of the Lunar Module (LM) high gain antenna is recommended for synchronous orbit; however, the use of a directional antenna will present an integration constraint on the space station. Pointing requirements for the directional antenna will necessitate either earth orientation of the station or gimballing of the antenna axes during transmission.

4.2.5.1 EXTRA VEHICULAR ACTIVITY

Voice and biomedical data will be handled by the standard Apollo EVA communications system.

4.2.5.2 TELEVISION

EVA and on-board TV signals can be generated by the modified Apollo TV camera and monitor which are presently space qualified. However, circuit margins are sufficient to permit commercial broadcast quality and the development of a new system to take advantage of this feature is recommended.

4.2.5.3 DATA MANAGEMENT SYSTEM

Infrequent earth-space station contacts in a 60° orbit make a very efficient Data Management System (DMS) necessary. The DMS will consist of the equipment necessary to receive experiment and housekeeping sensor outputs and efficiently process, sort, select, format, program, route, control, and/or display these data. The DMS will provide the following functions:

a. Data Acquisition

b. Spacecraft Monitoring

c. Data Processing and Control

4.2.5.4 DATA STORAGE SYSTEM

The main function will be to augment the data management system in order to optimize the storage capacity or the telemetry down-link bandwidth.

The data storage system will consist of the following units:

a. Video bandwidth recorders

b. Multichannel variable speed wide bandwidth recorders

c. Digital recorder

d. Portable recorders

(1) Portable EVA recorder

(2) A reproduce unit in the vehicle

The anticipated operational life of the recorder units is one year, at which time they should be replaced.

4.2.6 INSTRUMENTATION

The instrumentation system consists of several major components. These are: measurement systems, signal conditioning systems, displays and controls, caution and warning systems, timing, and the lighting system.

4.2.6.1 MEASUREMENT SYSTEM

The function of the measurement system is to sense all physical stimuli for which measurement is required, and to provide a repeatable, proportional electrical signal which is functionally related to the variable.

The problems are similar to those inherent in other systems: namely, those related to long life and reliability. In addition, absolute calibration concepts must be devised. New installation techniques must be developed which will allow replacement of sensors without disrupting system operation.

4.2.6.2 SIGNAL CONDITIONING SYSTEM

The signal conditioning system will be used for amplifying, shaping, mixing, or otherwise processing or modifying the raw transducer signals. The conditioned signals will then be recorded and in many cases also telemetered and displayed. Some of the signals will be combined or integrated into the caution and warning system to alert the crew to conditions which require response.

4.2.6.3 DISPLAYS AND CONTROLS SYSTEM

The Displays and Controls (D&C) system will provide a centralized station designed to monitor the condition or status of the operational systems and control or alter appropriate variables as required.

The system will consist of panels on which are mounted meters, displays, switches, circuit breakers, indicators, and other hardware necessary for monitoring or manual control.

New designs are required which will permit servicing or replacement of components without disruption of system operation.

Standardization will permit the direct interchange of various subassemblies or components when required by emergency conditions.

4.2.6.4 CAUTION AND WARNING SYSTEM

The function of the Caution and Warning (C&W) system is to alert the crew to conditions which, if not corrected in reasonable time, will prove detrimental to the welfare of the station occupants and/or the mission.

The C&W electronics package will contain the logic circuitry and level sensors which will energize the Master Alarm, flags, tones, and annunciators used to indicate out of tolerance or unsafe conditions, failures, or potential failures.

The C&W system interfaces with all other systems and the final configuration is dependent on the mission complexity. Even so, the C&W hardware should be basic and would differ from Apollo primarily in magnitude and in types of systems monitored.

4.2.6.5 CENTRAL TIME AND FREQUENCY STANDARD AND ASSOCIATED EQUIPMENT

This system will provide the space station with a highly accurate time reference for use by the on-board navigation and guidance system and for other general timekeeping. It will also provide on-board experiments with time and/or interval measurements as needed.

4.2.6.6 SPACE STATION LIGHTING

The light environment in the space station must be controlled to a comfortable and constant level that will allow visual acuity for controlling and operating the station. The control of light entering the windows will be accomplished by shades and filters similar to those used on the Apollo Command Module.

Lighting will be accomplished by means of electroluminescent panels supplemented by incandescent lamps where required. Additional lighting will be provided in the controls and displays area by means of flood lights directed on the console. The station will also have an external light system consisting of the following:

a. Docking lights (running lights)

b. Rendezvous beacon light

c. Portable lighting

An auxiliary emergency lighting system will be provided in all areas of the space station. This system will be connected to an emergency battery system and will provide illumination intensities of approximately 5-foot candles.

4.2.7 CRYOGENIC STORAGE SYSTEM

Cryogenic storage of oxygen and nitrogen is required for makeup of atmospheric losses due to leakage, depressurization, and metabolic consumption of oxygen. Suitable storage systems are not currently available and must be developed.

Subcritical storage of oxygen and nitrogen will be used for optimum performance. System parameters such as operating pressure, tank size, insulation type, etc., require further study before recommendations can be made.

If the Sabatier oxygen regeneration system is used, hydrogen must also be stored. In this case, refrigeration or vehicle orientation will be necessary in order to achieve the required storage times. The refrigeration loads are low and the refrigeration temperatures are in the range from $70^{\circ}F$ to approximately minus $60^{\circ}F$.

Resupply of cryogens can be accomplished by fluid transfer or by tank replacement. The preferred technique has not yet been determined.

4.2.8 REACTION CONTROL SYSTEM

The Reaction Control System (RCS) provides the following functions in a rotating space station:

a. Spin-up and de-spin

b. Control moment gyro realignment

c. Spin axis precession and attitude control

The RCS will utilize pressure-fed, earth storable, hypergolic propellants. Thrusters, tanks, and valves will be modularized for simplicity of replacement in space. Positive expulsion propellant tanks will be pressurized by a volatile liquid, such as Freon, and a reversible thermal control loop. This will permit propellant replenishment without loss of pressurant.

4.2.9 STRUCTURES

The primary structure of the station will be semi-monocoque and consists of stiffened, load carrying skin, circumferential frames and longitudinal beams (longerons). Skin in pressurized areas will have integrally machined, waffle pattern stiffeners. All joints that must be pressure tight will be welded. An external, non-structural micrometeoroid bumper skin will surround all areas requiring micrometeoroid protection. Multilayer, reflective insulation and/or low density, open cell, plastic foam will be installed between the bumper and structural skin. Nose fairings and interconnect structure will be sheet-stringer type construction and will be coated with ablative material where required for thermal protection during launch. Structural fairings in certain areas may utilize honeycomb sandwich construction.

Bulkheads between compartments will normally be flat and designed to carry the station internal pressure in case of depressurization of a compartment. These bulkheads, or compartment "floors," will utilize radial and intercostal beams. The bulkhead pressure skin will either have integrally machined stiffeners or be of honeycomb sandwich construction.

Aluminum alloy will be used for the major portion of the primary structure. For all structural elements to be welded, new, higher strength, weldable aluminum alloys under development at present will probably be used.

4.2.10 SYSTEM TECHNOLOGY STATUS

The general status of system technology as it relates to space station requirements is summarized in Tables 4.3 through 4.8. For this purpose, technology status has been divided into three categories:

a. Current, indicating that the technology, if not the hardware, is available at the present time.

b. Improved, implying that the technology is not available now but can be foreseen within the time frame anticipated for the space station.

c. Advanced, referring to those items which may not be available without additional effort.

TABLE 4.3

ELECTRICAL POWER SYSTEM TECHNOLOGY STATUS

FUNCTION	TECH.	COMPONENTS	REQUIREMENTS
. POWER GENERATION	2	SOLAR CELL ARRAY	POWER FOR SUB-SYSTEMS & EX-PERIMENTS
	1	BATTERIES	
. POWER DISTRIBUTION	1	INVERTERS	
	1	CONTROLS	
	1	WIRING	

SYSTEMS CONSIDERED:

. RADIOISOTOPE THERMOELECTRIC
. " BRAYTON
. " MERCURY RANKINE
. NUCLEAR THERMOELECTRIC
. " BRAYTON
. " MERCURY RANKINE
. SOLAR CELL/BATTERY
. " " /REGEN. FUEL CELL

1. Current Technology
2. Improved Technology
3. Advanced Technology

TABLE 4.4

ENVIRONMENTAL CONTROL/LIFE SUPPORT SYSTEM TECHNOLOGY STATUS

FUNCTION	TECH	COMPONENTS	CREW REQUIREMENTS
. THERMAL CONTROL	1	RADIATORS	. METABOLIC & EQUIPMENT HEAT DISSIPATION
	1	COOLANT LOOPS	. TEMPERATURE REGULATION
	1	COLD PLATES	
	1	HEAT EXCHANGERS	
	1	CONTROLS	
. ATMOSPHERIC CONTROL	1	VALVES & REGULATORS	. CO_2 REDUCTION
	1	WATER SEPARATOR	. HUMIDITY CONTROL
	1	COMPRESSORS & FANS	. PRESSURE CONTROL
	1	CONTROLS	. ODOR & TRACE CONTAMINANT REMOVAL
	2	MOLECULAR SIEVE	
	2	CATALYTIC BURNER	
	3	SABATIER REACTOR	
	3	ELECTROLYSIS UNIT	
. CRYOGENIC STORAGE	1	VALVES & PLUMBING	. BREATHABLE ATMOSPHERE
	1	CONTROLS	. PLSS RECHARGE
	2	TANKS	
. WATER RECLAMATION & WASTE MANAGEMENT	2	TOILETS	. POTABLE WATER
	1	STORAGE TANKS	. WASTE REMOVAL
	2	WATER PURIFICATION SYSTEM	

SYSTEMS CONSIDERED:

. OXYGEN REGENERATION VERSUS STORAGE

1. CURRENT TECHNOLOGY
2. IMPROVED TECHNOLOGY
3. ADVANCED TECHNOLOGY

TABLE 4.5

CREW SYSTEMS TECHNOLOGY STATUS

FUNCTION	TECH	COMPONENTS	CREW REQUIREMENT
. FOOD	2 2	OVEN STORAGE	. NUTRITION
. REST & RELAXATION	1 1	BUNKS BOOKS, GAMES, ETC.	. SLEEP . ENTERTAINMENT
. EXERCISE	1	EXERCISE DEVICES	. PHYSICAL FITNESS
. PERSONAL CARE	1 1 1 1 2 2	SHAVES HAIR & NAIL CLIPPERS TOOTHBRUSHES CLEANSING PADS SHOWER LAUNDRY	. CLEANLINESS . ODOR CONTROL . GROOMING
. MEDICAL	1 1 1	MEDICAL KIT RADIATION DOSIMETERS BIOMEDICAL INSTRU-MENTATION	. FIRST AID . MONITOR CREW CONDITION
. SPACE ENVIRONMENT PROTECTION	1 1	PRESSURE SUITS PLSS	. EVA . UNPRESSURIZED OPERATION
. WORK STATIONS	1 1	SEATS RESTRAINTS	. COMFORT . CONVENIENCE

1. CURRENT TECHNOLOGY
2. IMPROVED TECHNOLOGY
3. ADVANCED TECHNOLOGY

TABLE 4.6

GUIDANCE & CONTROL SYSTEM TECHNOLOGY STATUS

FUNCTION	TECH.	COMPONENTS	CREW REQUIREMENTS
. STABILIZATION	2	CONTROL MOMENT GYROS	. ATTITUDE HOLD . WOBBLE DAMPING
. ORIENTATION	1	REACTION CONTROL JETS	. ATTITUDE MANEUVERS
	1	CONTROLS	. CMG DESATURATION
	3	TANKS	. SPIN-UP
. NAVIGATION	1	COMPUTER	. NAVIGATION DATA FOR EXPERIMENTS & STATION OPERATION
	1	INERTIAL MEASUREMENT UNIT	
	1	CONTROLS & DISPLAYS	

1. Current Technology
2. Improved Technology
3. Advanced Technology

TABLE 4.7

COMMUNICATIONS AND DATA MANAGEMENT SYSTEM TECHNOLOGY STATUS

FUNCTION	TECH.	COMPONENTS	CREW REQUIREMENTS
. COMMUNICATIONS	1	S-BAND TRANSMITTERS & RECEIVERS	. STATION-GROUND VOICE COMMUNICATION
	1	ANTENNAS	. TELEMETRY
	1	AUDIO CENTERS	. TRACKING
	1	RADAR TRANSPONDER	. UP-DATA
	1	TV CAMERA	. RENDEZVOUS
			. EVA COMMUNICATION
			. TELEVISION
. DATA MANAGEMENT	2	RECORDERS	. DATA STORAGE
	2	PCM, ETC.	. DATA PROCESSING
	2	GEN. PURPOSE COMPUTER	

1. Current Technology
2. Improved Technology
3. Advanced Technology

TABLE 4.8

INSTRUMENTATION SYSTEM TECHNOLOGY STATUS

FUNCTION	TECH.	COMPONENTS	CREW REQUIREMENTS
. MEASUREMENT	1 1	TRANSDUCERS SIGNAL CONDITIONERS	. SENSE SYSTEM OPERATING PARAMETERS
. DISPLAY & CONTROL	1 1 1 1	GAUGES & INDICATORS SWITCHES, ETC. CLOSED CIRCUIT TV LOGIC CIRCUITS	. CREW READOUT OF SYSTEM PARAMETERS . CONTROL SYSTEM OPS. . MONITOR EVA, DOCKING ETC. . CAUTION & WARNING
. CENTRAL TIMING	1 1	HIGH PRECISION OSCILLATOR FREQUENCY DIVIDERS	. FREQUENCY STANDARD FOR NAVIGATION, COMMUNICATIONS, OTHER SYSTEMS
. LIGHTING	1 1 1	LIGHTING FIXTURES EXTERNAL LIGHTS PORTABLE LIGHTS	. GEN. ILLUMINATION . PANEL ILLUMINATION . RENDEZVOUS & DOCKING ILLUMINATION . EVA ILLUMINATION

1. Current Technology
2. Improved Technology
3. Advanced Technology

4.3 SPACE STATION CONFIGURATIONS

The major factors which affect the configuration of a space station are shown below.

Experiments
Launch Vehicle
Systems
Operations
Deployment Requirements
Crew

Impacts:

- Volume
- Weight
- Performance

Other considerations include orientation and stability requirements.

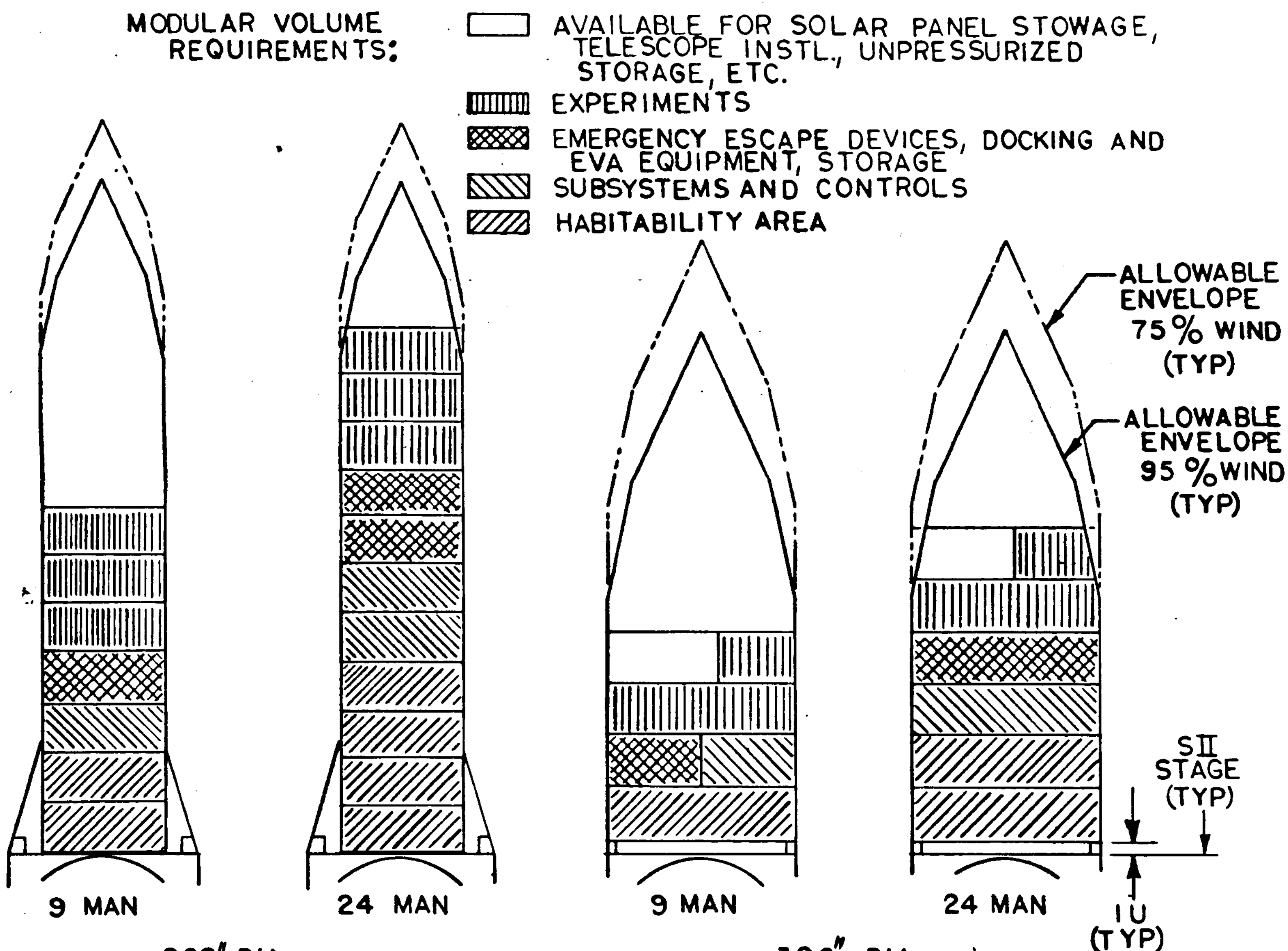
The following sections discuss the general requirements imposed by these factors.

4.3.1 INTEGRATION CONSIDERATIONS

It is not possible to specify all of the integration constraints at the current level of design; however, the following sections provide a description of the major factors and some of the alternatives available.

4.3.1.1 LAUNCH CONSTRAINTS

The launch constraints include the allowable payload envelope (volume and shape), the launch loads, and the prelaunch service and checkout requirements. The envelope is shown schematically in Figure 4.18. As shown, the envelope must accommodate crew, systems, emergency return devices, and experiments.



VOLUME COMPARISON

FIGURE 4.18

Aerodynamic factors establish the size and shape of the launch envelope which in turn limits the total volume available. Launch loads are the critical design factor for some structural, experimental and systems components. Other components such as optical lenses and mirrors favor a particular orientation relative to the launch loads. In some cases, additional structural support provisions are required.

4.3.1.2 EXPERIMENT INTEGRATION

The majority of the earth, celestial and solar sensors require exposure to the space environment for optimum, unobstructed operation and, therefore, must be placed in unpressurized areas or areas capable of being depressurized. Major maintenance will normally be accomplished in a pressurized area and, possibly, in the artificial gravity module of a rotating space station. Experiment electronics and data reduction equipment should be located in a pressurized area. Sources of gaseous effluent, such as RCS thrusters, should be located as remote from the sensors as possible to prevent "clouding" of optical surfaces. Installation provisions and physical locations of all sensors must satisfy pointing, stabilization, thermal, launch loads, pad access requirements, etc.

Of all sensor installation requirements, the pointing requirements will affect the station arrangement most. The space station will have to provide for pointing solar sensors toward the sun, astronomical sensors toward the desired point in the

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

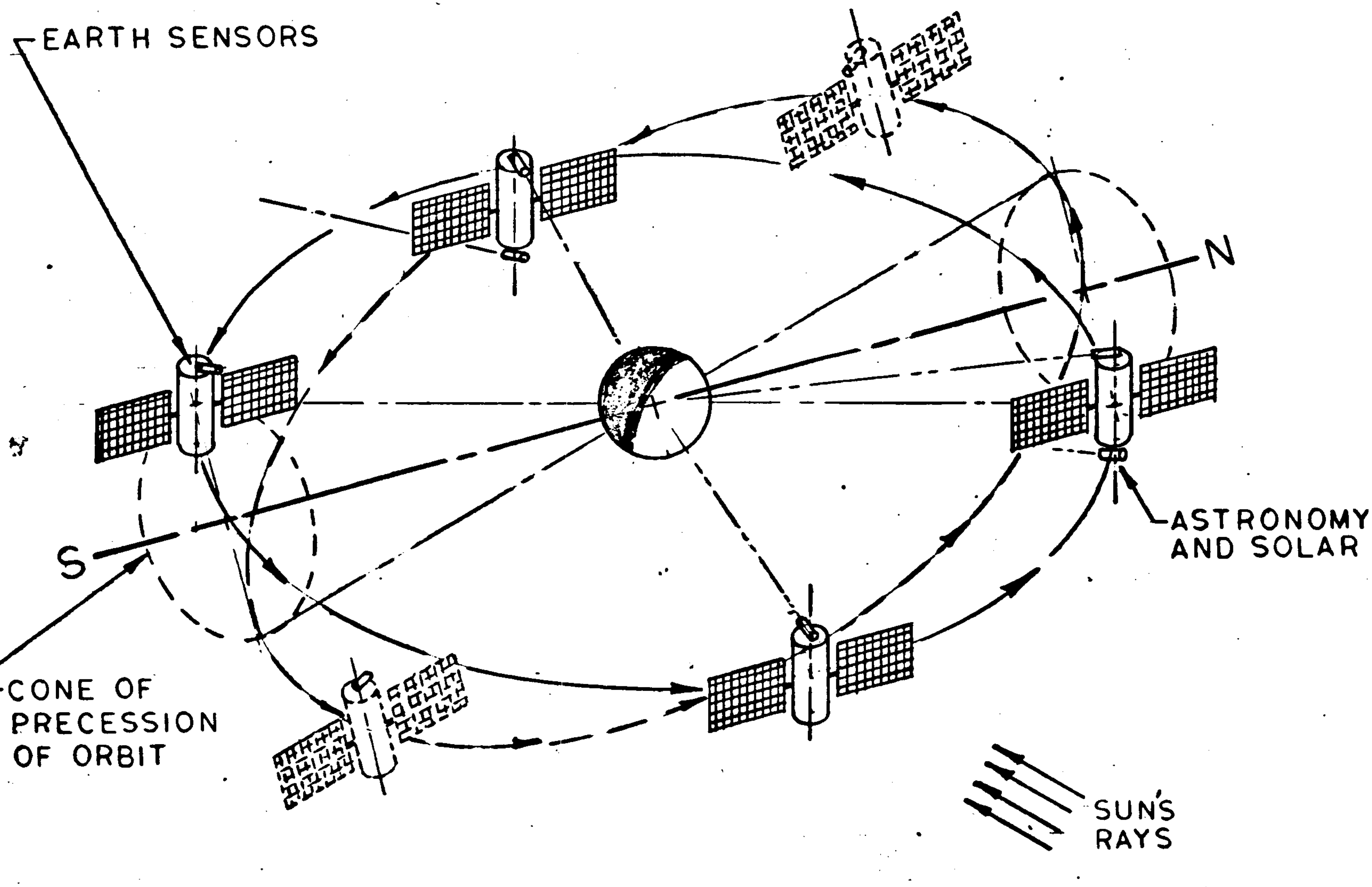


FIGURE 4.19

Astronomical sensors require a 2-axis gimbal mount and an inertially fixed space station attitude. The gimbal mount is required to provide telescope stabilization. This gimbal mount can also provide basic directional pointing which eliminates the necessity of pointing the entire station and allows the station to be oriented to best satisfy other requirements. Telescope operation with the station orientated toward the earth is not practical because the gimbal mount cannot readily track and stabilize simultaneously. Earth sensor and astronomical sensor operation could be time shared, however, to utilize orientation for each.

Earth sensors will require the equivalent of a 3-axis gimbal mount if operation is required when the station is solar or inertially oriented.

Solar orientation eliminates the requirement to gimbal solar cells and is similar to inertial orientation with respect to telescope stabilization. This is because the sun angle relative to the solar cells can be allowed to vary on the order of 10 degrees without substantially reducing solar cell output. Since the sun angle changes only one degree per day, the station can be reoriented relative to the sun at approximately 20-day intervals (assuming the station is initially oriented 10 degrees ahead of the sun). During this interval, the station attitude is inertially fixed relative to the mean sun position.

celestial sphere, and earth viewing equipment toward points on the terrestrial sphere, and, possibly, allow simultaneous operation of all. These three different pointing requirements are in basic conflict as to direction and directly affect the station orientation requirements. Another factor which must be considered is the precession of the orbital plane about the earth's polar axis, which is caused by the earth's oblateness. The precessional rate of a low altitude, 60° inclined orbit, for example, is approximately 4° per day.

Figure 4.19 indicates the general relationship between the pointing requirements for a zero gravity station. In this example, the station is oriented to facilitate earth viewing with its longitudinal axis normal to the orbit plane. Because of the precession of the orbit plane and the varying position of the earth relative to the sun, the solar pancls must be gimballed. If the earth sensors were fixed to the station, the station would be required to roll about its longitudinal axis at the rate of one revolution per orbit revolution. In this case, the solar panels would require a 2-axis gimbal mount. If the station were not rotated about its longitudinal axis for earth viewing, the earth sensors would have to be mounted with one axis of freedom parallel to the station longitudinal axis. The solar panels would then require only one axis of freedom - normal to the station longitudinal axis. The station itself would be positioned about its longitudinal (roll) axis to provide the other axis of freedom for the solar panels.

Figure 4.20 illustrates the pointing considerations for an artificial gravity station. In this case, the station spin axis is pointed toward the sun. If continuous earth orientation were desired, an excessive amount of RCS propellant would be required to precess the station's angular momentum vector. Therefore, either solar or inertial orientation is desirable. If solar orientation is selected, the solar panels may be fixed to the rotating portion of the station. Astronomical and earth sensors must be mounted to a non-rotating portion of the station and will require 2-axis and 3-axis gimbal mounts respectively.

A desirable location of each type of sensor relative to the sun is shown in Figure 4.20. Locating the earth sensors on the end of the hub that is pointing toward the earth during the light side passage eliminates having these sensors view through the "spokes" of the station. The astronomical sensors, which would be operated during dark side passage, can view any point within half the celestial sphere during approximately half of any orbit. Within a 6-month period the entire celestial sphere will be accessible for viewing. One limitation for this arrangement is that the planets Mercury and Venus cannot be viewed. If the astronomical sensors were located on the end of the hub pointed toward the sun, no planet outside the earth's orbit can

SPIN AXIS ORIENTATION
PARALLEL TO SUN'S RAYS

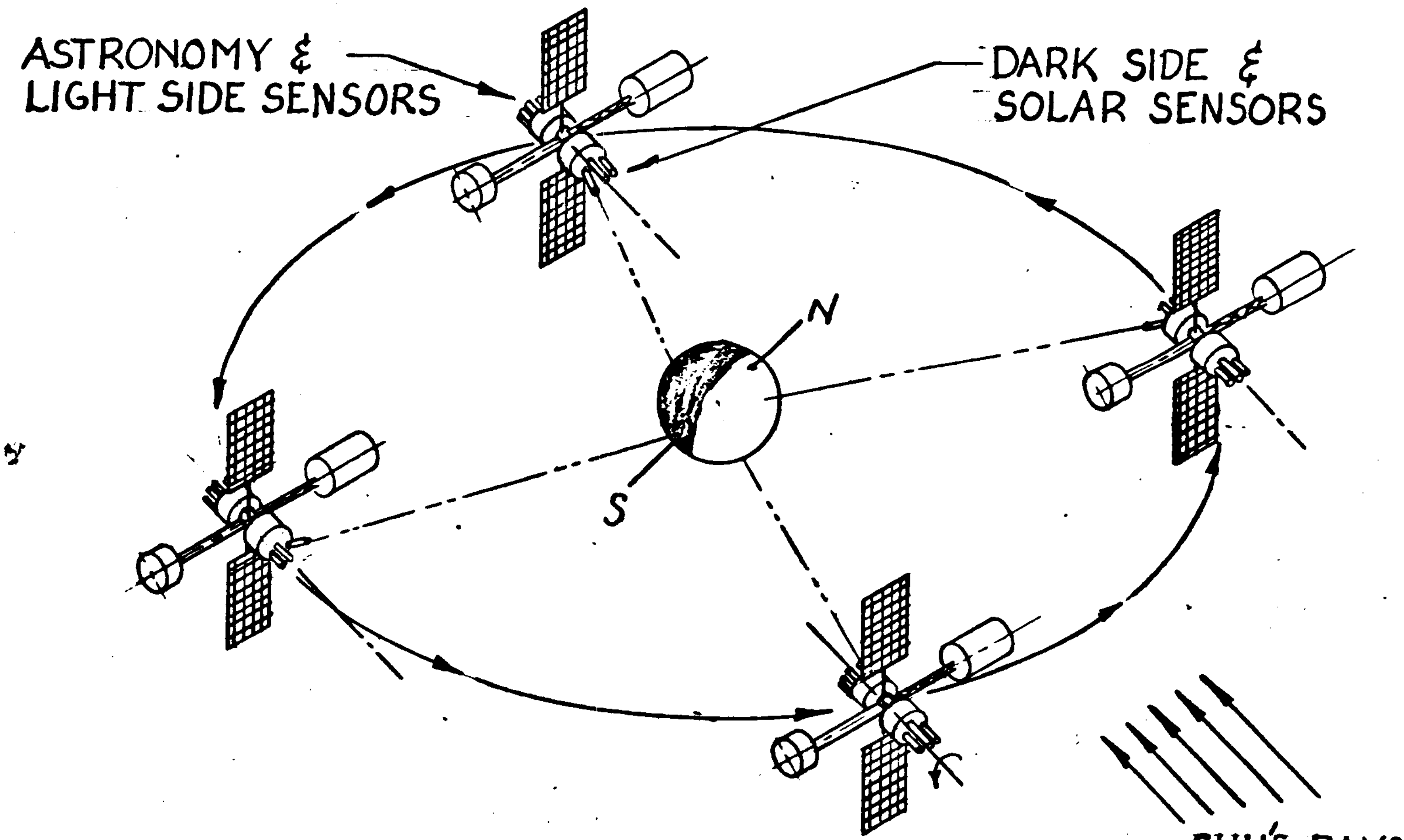


FIGURE 4.20

be viewed at or near its conjunction with earth. However, the station can be despun for a period to allow either of these limitations to be overcome.

Because of sunlight reflection from spacecraft structure, etc., a telescope may be limited to viewing objects located 45 degrees or more from the sun. However, if there were an object of interest near the sun at a given time, it would be located outside the 45 degree limitation three months later.

Concepts to satisfy sensor installation and pointing requirements are presented in Section 4.3.2.

Laboratory volume, specialized equipment, appropriate environment, etc., are requirements imposed on the station design by all experiment categories. Separate laboratory compartments for each experimental discipline are generally desirable. Isolation from the overall space station environment, especially the atmosphere, is required by certain experiments. Gravity levels of essentially zero are required by a majority of the experiments. Achieving less than 10^{-5} g's, as necessary for certain biological experiments, may require special gimbal mounts in either the rotating or non-rotating stations. The requirement for zero gravity necessitates a large non-rotating hub in the artificial gravity configuration.

4.3.1.3 SYSTEMS INTEGRATION

4.3.1.3.1 Electrical Power System

A solar cell system has tentatively been selected for the station power source. However, some discussion of the integration of radioisotope and nuclear reactors is presented.

A solar cell EPS affects the space station configuration because of deployment and pointing requirements. The deployment method is constrained by available launch volume and mechanical ingenuity. Solar cell pointing is further complicated by the requirements of the experiments and the basic orientation of the space station. The installation of a solar cell EPS is dependent upon the orientation of the station. Should the station be orientated toward the sun, the solar cells may be fixed relative to the space station. Other orientations require the solar panels to be gimballed with respect to the station.

If a radioisotope EPS were used, the waste heat generated presents a problem because of the large radiator area required. The location of personnel, systems and experimental equipment relative to the heat source is also constrained.

A nuclear reactor EPS constrains the configuration by requiring shielding and physical separation to protect the crew, experimental equipment and systems. This problem can be minimized on an artificial gravity space station by using the nuclear EPS as a counterweight. Launch and prelaunch constraints are not considered to be great if the nuclear reactor is not activated until the station is in orbit.

One problem common to all power sources is the requirement for transmission of power between the rotating and non-rotating modules of an artificial gravity station. This requirement presents a potential design and development problem to achieve reliability over long periods of time. One possibility is to provide separate conditioning and control functions so that only unregulated DC is transferred across the rotating joint.

4.3.1.3.2 Environmental Control

Separate modular systems should be used in the hub and the artificial gravity module of a rotating station to avoid sealing, insulation, and rotating joint problems involved with transfer of atmospheric gas, water, coolant, etc.

4.3.1.3.3 Communications

The primary communications system (earth-to-space station voice communications, telemetry and TV) will be installed in the artificial gravity module of a rotating station. Communication between the hub and the artificial gravity module will be by radio frequency link. The use of slip rings does not appear to be desirable because of static and the number of separate conductors that would be required.

4.3.1.3.4 Guidance and Control

Guidance and navigation sensors may require installation within the non-rotating portion of the artificial gravity station. Auxiliary equipment will be located at the command station in the artificial gravity module. A reaction control system will be installed in the artificial gravity module to spin-up, despin, and change spin axis orientation. Control moment gyros may be used to damp station wobble caused by internal mass movement and external torques.

4.3.1.4 OPERATIONS

4.3.1.4.1 Logistics Interface

Logistics operations require that the space station be capable of docking with a logistics spacecraft to allow material and personnel transfer. An artificial gravity space station must have the docking port located on the non-rotating hub to avoid despinning the station. An additional logistic constraint is created by the requirement to transfer items from the logistic spacecraft to both the rotating and non-rotating parts of the artificial gravity station.

4.3.1.4.2 EVA Interface

EVA will be required to satisfy certain operational and experimental demands. For an artificial gravity station, the EVA port should be located on the non-rotating hub. An airlock will be used to avoid excessive loss of atmospheric gases and may also be used as a decompression chamber.

4.3.1.4.3 Escape and Emergency Return

The space station crew safety requirements will be provided on-board the space station for all emergencies except a catastrophic failure. Should this occur, emergency return devices which have the capability of re-entry and safe landing will be provided for each crewman. The use of the emergency return device allows a reduction in complexity of the logistics vehicle system since it will not have to survive extended orbital storage.

4.3.1.5 SPACE STATION ACTIVATION AND MECHANIZATION

To transform an artificial gravity space station from the launch configuration to the orbital configuration will require deployment of the station modules, systems and experimental equipment, activation of the systems and spin-up of the station for artificial gravity. Should a spent booster stage be used for a counterweight, it must be passivated by venting residual propellants and pressurants, deactivating destruct systems, etc. Important activation factors that will affect the configuration are manned versus unmanned launch and the degree of automatic activation before the station is manned.

A manned launch restricts a space station configuration because of the required crew facilities and launch abort capability. Part of the structural system, and other systems to a degree, must be designed for launch loading to crew safety requirements rather than the less stringent mission success requirements.

An artificial gravity space station has two unique mechanical functions as follows:

a. A portion of the station must be rotated to achieve artificial gravity while the center hub is maintained inertially fixed to provide a zero-gravity volume.

b. The artificial gravity configuration must be achieved by deploying the artificial gravity module and a counterweight or opposing module(s) in relation to the center hub. Preliminary study has indicated that the mechanical functions of rotating, sealing and deploying the modules are feasible.

4.3.1.6 CREW ACCOMMODATIONS

In order to fulfill the habitability requirements, the crew must be provided with private quarters, wardroom, gymnasium, hygienic compartments, and a sick bay.

The private quarters will be large enough to provide sleeping accommodations, personal storage and volume for relaxation. Each compartment will have approximately 35 square feet of floor area.

For convenience, the hygienic compartment will be located near the private quarters. This compartment will consist of toilet and body cleansing facilities. A hygienic compartment containing only a toilet and lavatory facilities will be located near the wardroom.

The wardroom may be adjacent to the gymnasium to allow temporary conversion into a single large room. The wardroom can also be used as a recreation room and will be analogous to the kitchen-den in a modern home.

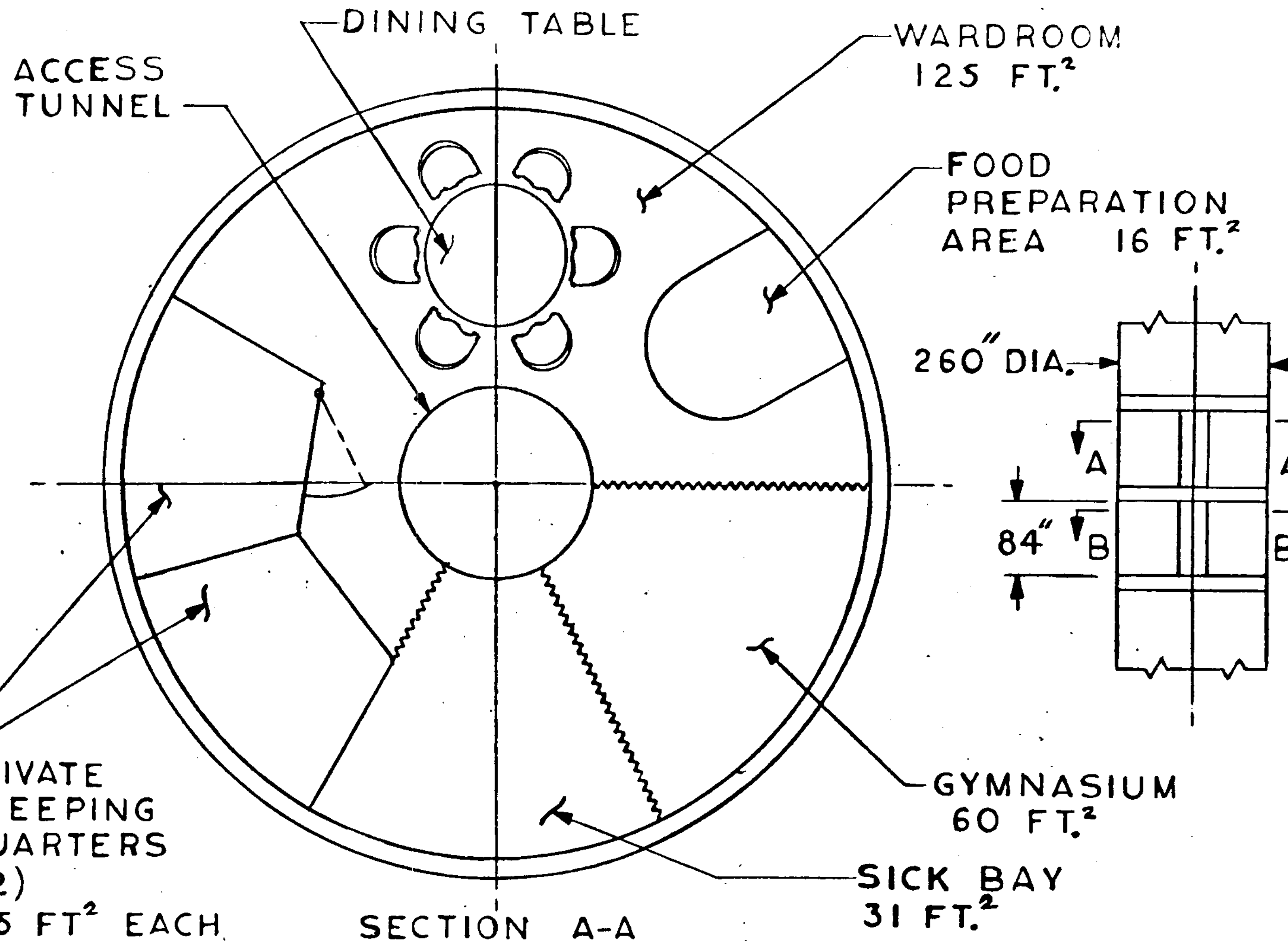
The sick bay may be used in the biomedical experiment program, as well as serving its primary purpose. This compartment should be located near one or two of the private quarters in order to utilize them as "hospital" rooms. Increased space for treatment might be provided by a folding "wall" between the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

In general, approximately 75 square feet of floor area per man with seven feet head height is sufficient to fulfill the habitability requirements. Figures 4.21 and 4.22 illustrate a conceptual arrangement for a 260-inch diameter module. Figures 4.23 and 4.24 are for a 396-inch diameter module.

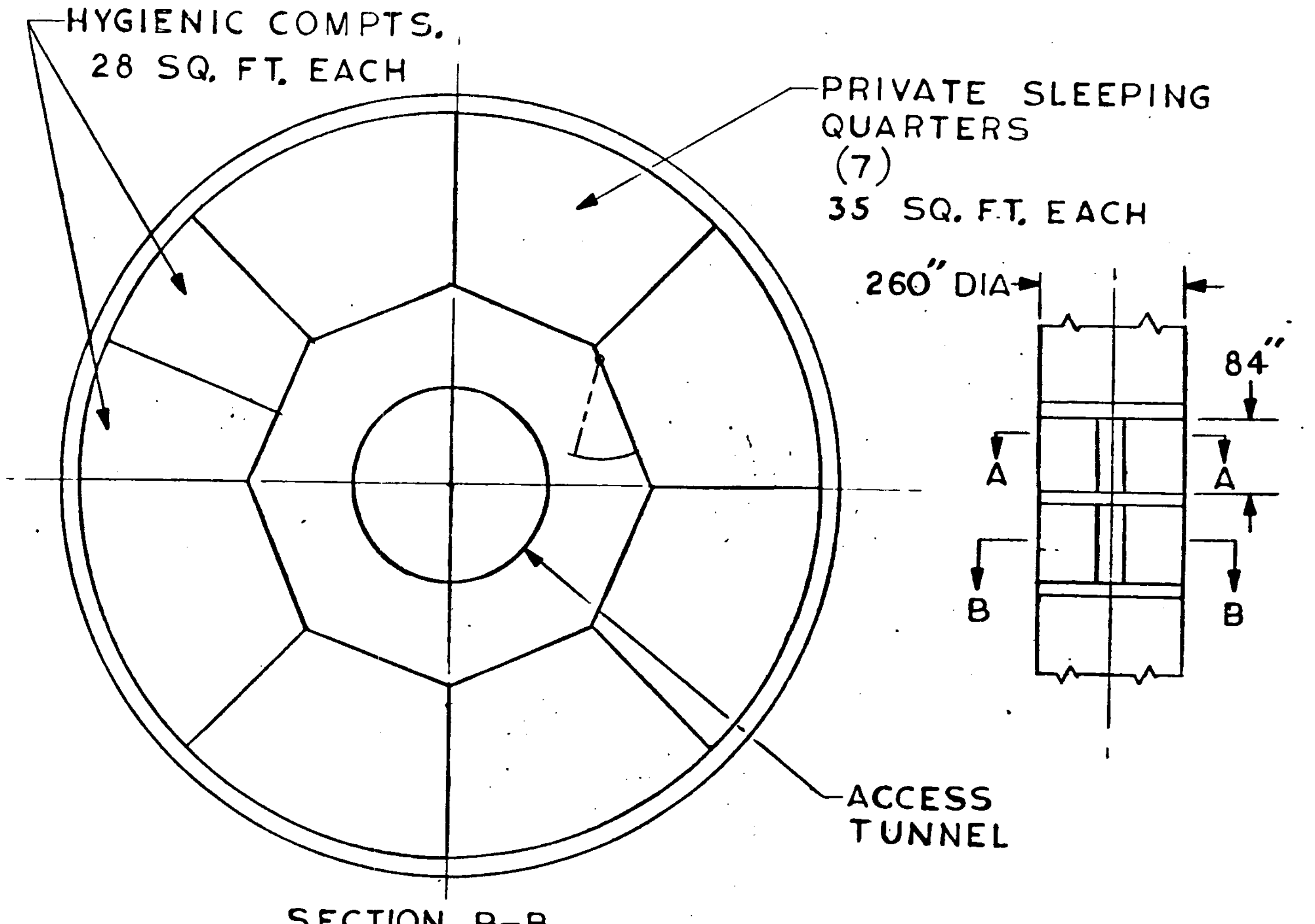
4.3.2 CONFIGURATION CONCEPTS

4.3.2.1 ORIENTATION DESIGN PHILOSOPHY

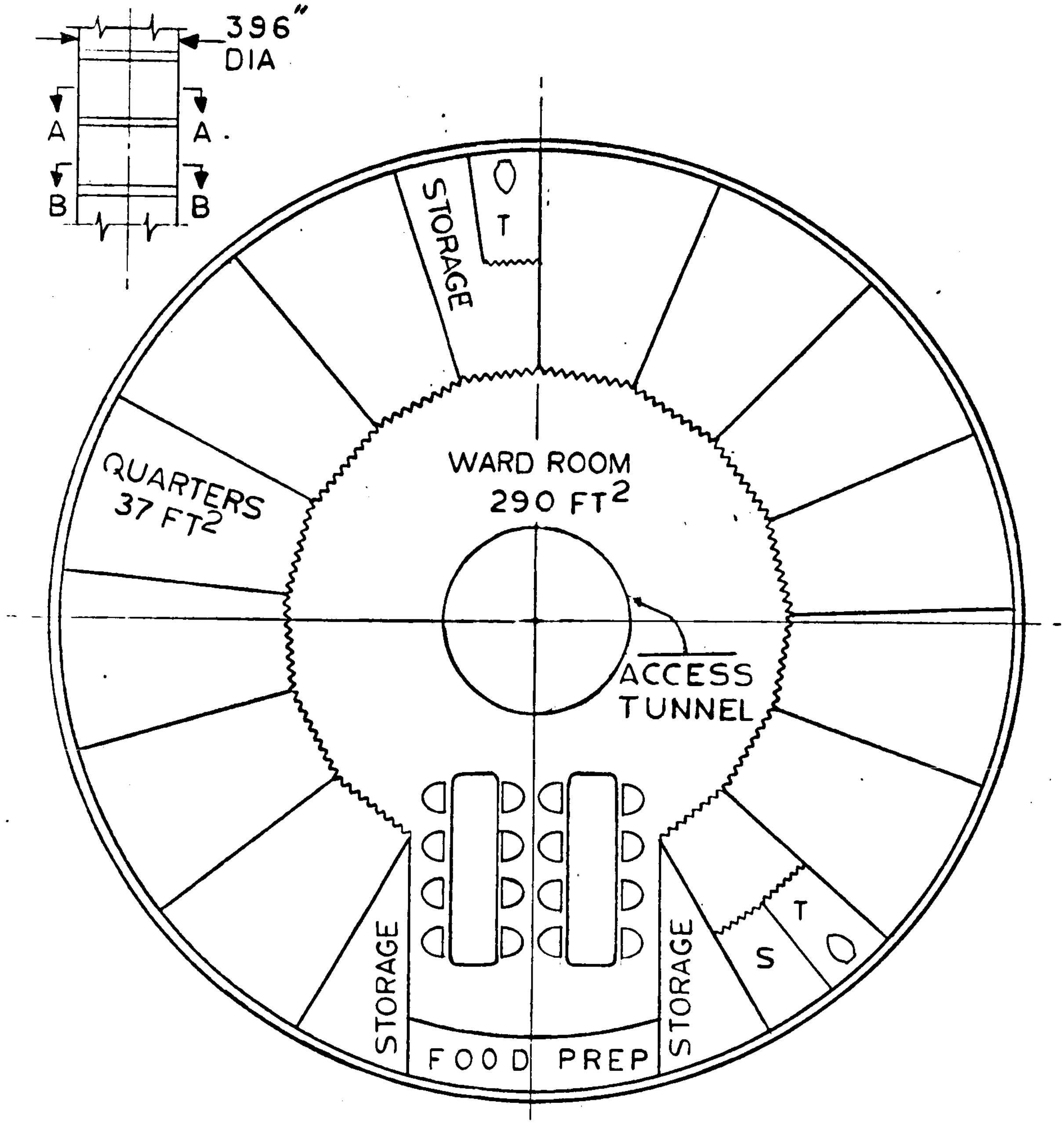
Orientation requirements are derived from astronomical, earth and solar sensors, and solar cells as discussed in Section 4.3.1.2.



CREW QUARTERS
FLOOR PLAN
FIGURE 4.21



CREW QUARTERS
FLOOR PLAN
FIGURE 4.22

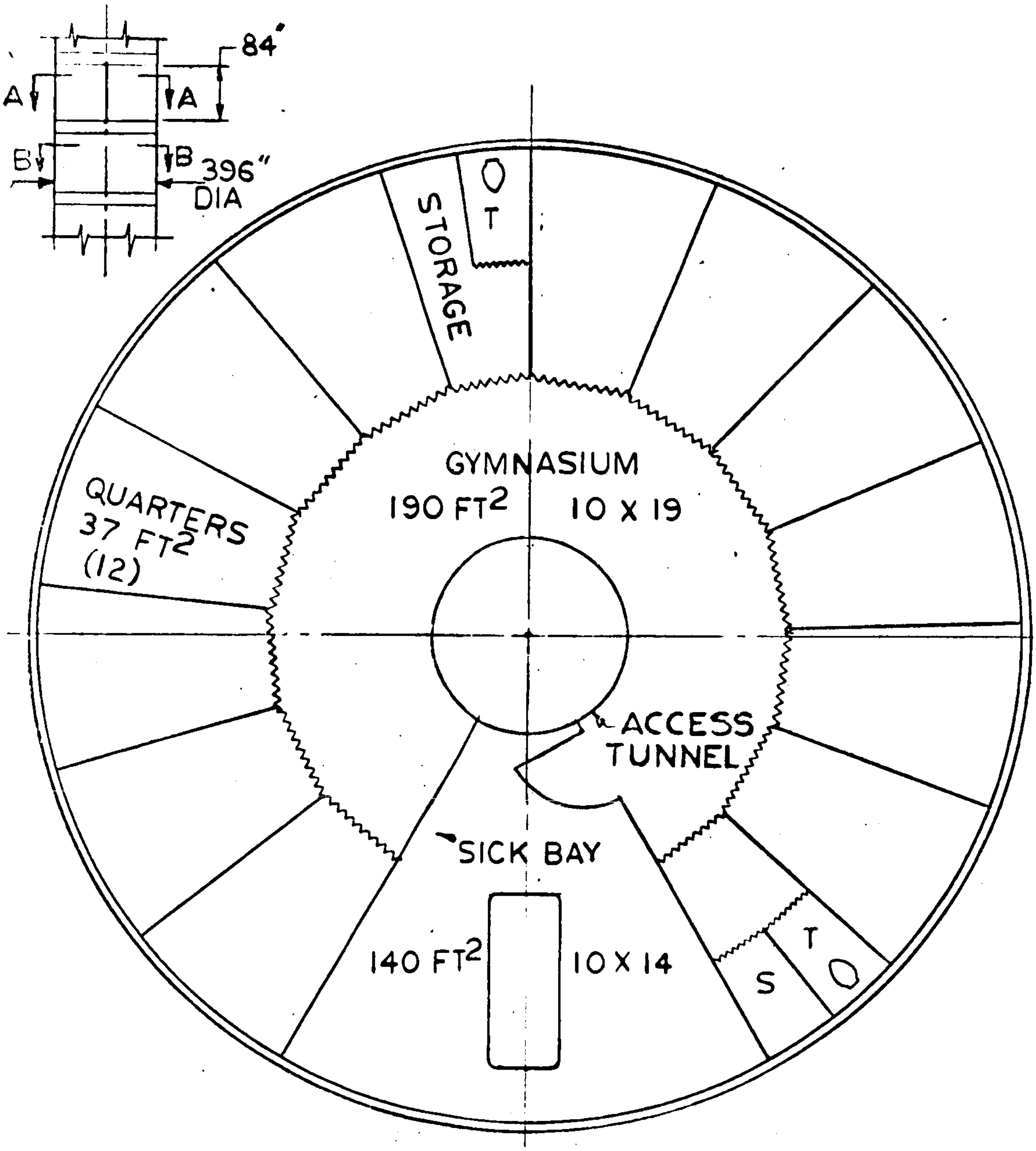


SECTION A-A

CREW QUARTERS FLOOR PLAN

24 MEN-396" DIA. MODULE

FIGURE 4.23



SECTION B – B

CREW QUARTERS FLOOR PLAN

24 MEN-396" DIA MODULE

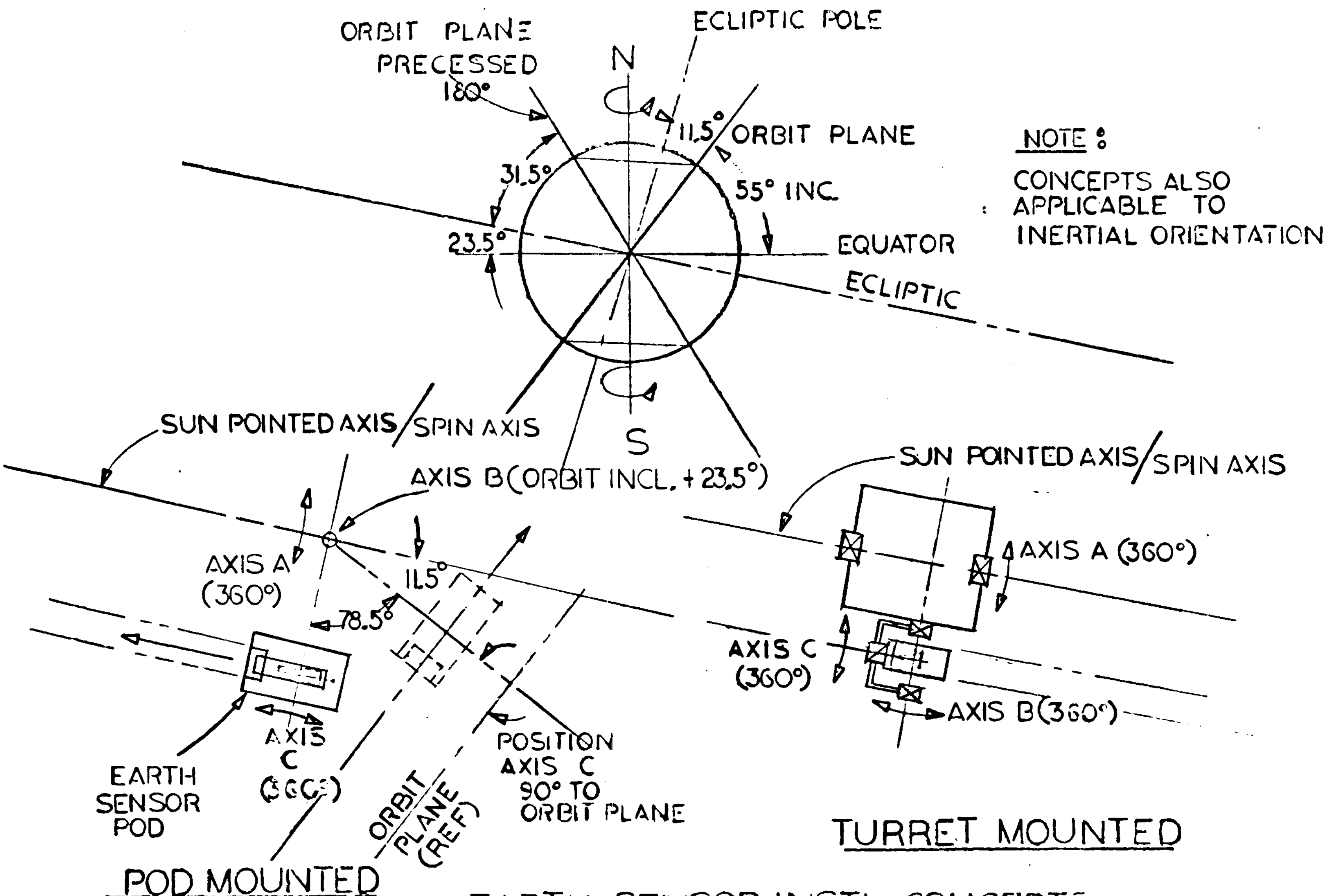
FIGURE 4.24

All of these pointing requirements plus the general desirability of having a fixed relationship to the sun for thermodynamic considerations has indicated that solar orientation is a "prime" choice for both zero gravity and artificial gravity stations.

A primary advantage of solar orientation is that, once deployed, the EPS solar panels may be fixed relative to the station. The elimination of gimbal mechanisms should improve station reliability, and will enhance attitude stability. This is particularly true for artificial gravity stations.

Astronomical sensor installation is relatively unaffected by solar orientation of the station, as discussed in Section 4.3.1.2. The earth sensors will require gimbal mounts. Two possible concepts for earth sensor installation are shown in Figure 4.25. For the pod installation, the sensors are mounted within a module or "pod" which is provided with three axes of freedom relative to the station. The basic principle is to position the pod with with its axis "C" normal to the orbit plane. Then, a relatively uniform rotation of the pod about axis "C", at a rate of approximately four degrees per minute, keeps the sensors pointed toward the earth. For an artificial gravity station, axis "A" may be combined with the non-rotating hub principal axis.

The turret mounted installation utilizes three axes having a different relationship to the spin axis. The turret mounted



EARTH SENSOR INSTL CONCEPTS
FOR SOLAR ORIENTED SPACE STATION

FIGURE 4.25

sensors are pointed along the local vertical using axis "A" and axis "B". However, axis "C" is required to orient the sensor with respect to the relative velocity vector between the sensor and the earth for proper image motion compensation. Pointing will require a continuous and coordinated movement about both the "A" and "B" axes. Further study is necessary to define all of the trade-offs between these two concepts.

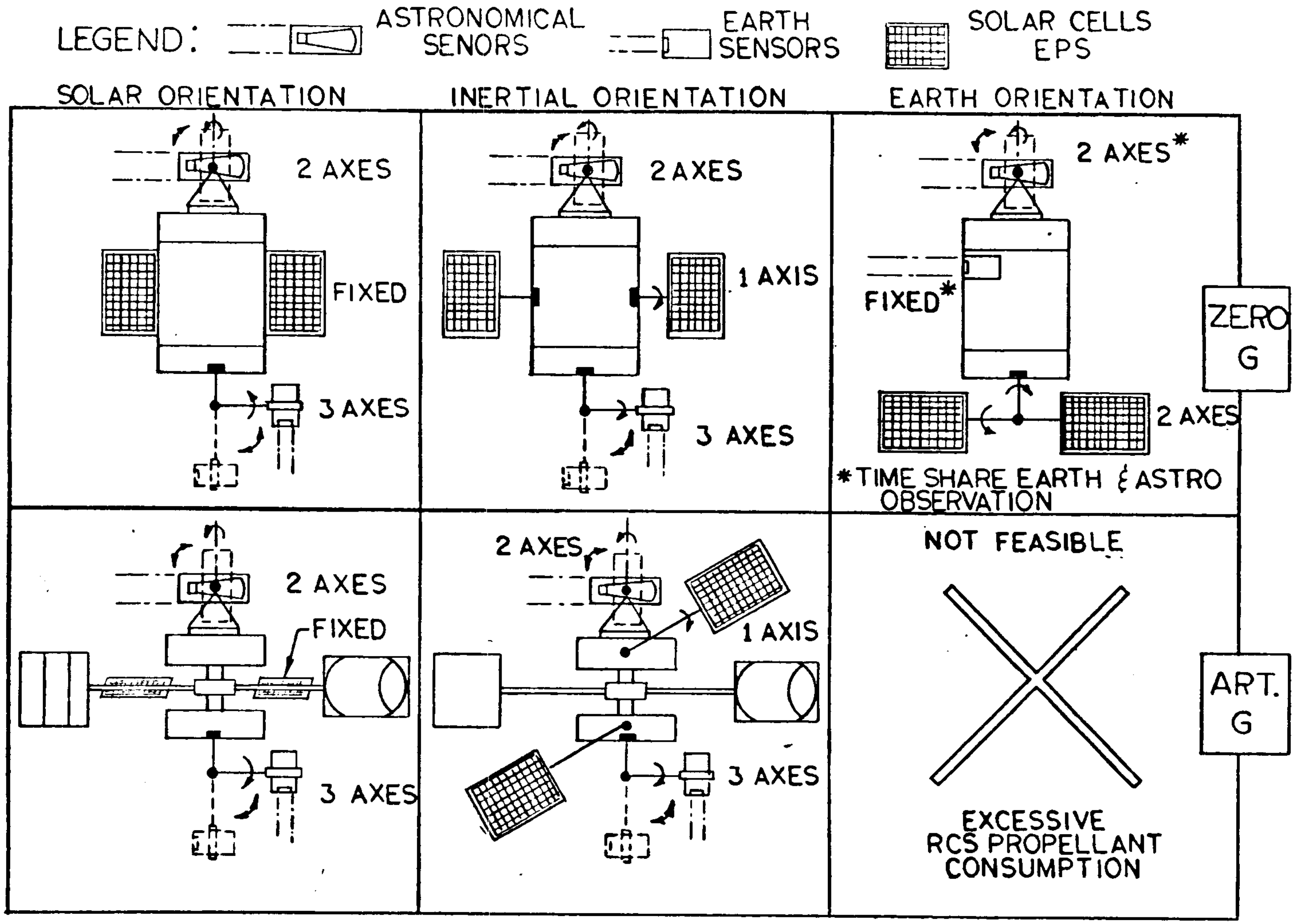
Figure 4.26 presents a summary of sensor mounting requirements for zero gravity and artificial gravity stations for the three station orientation modes. Because the artificial gravity station has a non-rotating, zero gravity hub, there is no significant difference between zero gravity and artificial gravity stations as to sensor mounting. Earth orientation for an artificial gravity station of the Saturn V-launched class would require several thousand pounds of RCS propellant per day to continually precess the angular momentum vector.

4.3.2.2 BASIC CONCEPTS

Three basic concepts for a space station have been established and will be described in the following sections. All concepts are compatible with the experiment integration requirements presented in Section 3.0 of this report.

4.3.2.2.1 Concept 1

Concept 1 is a zero gravity station designed to accommodate all



SENSOR MOUNTING
ZERO G/ARTIFICIAL G SPACE STATIONS

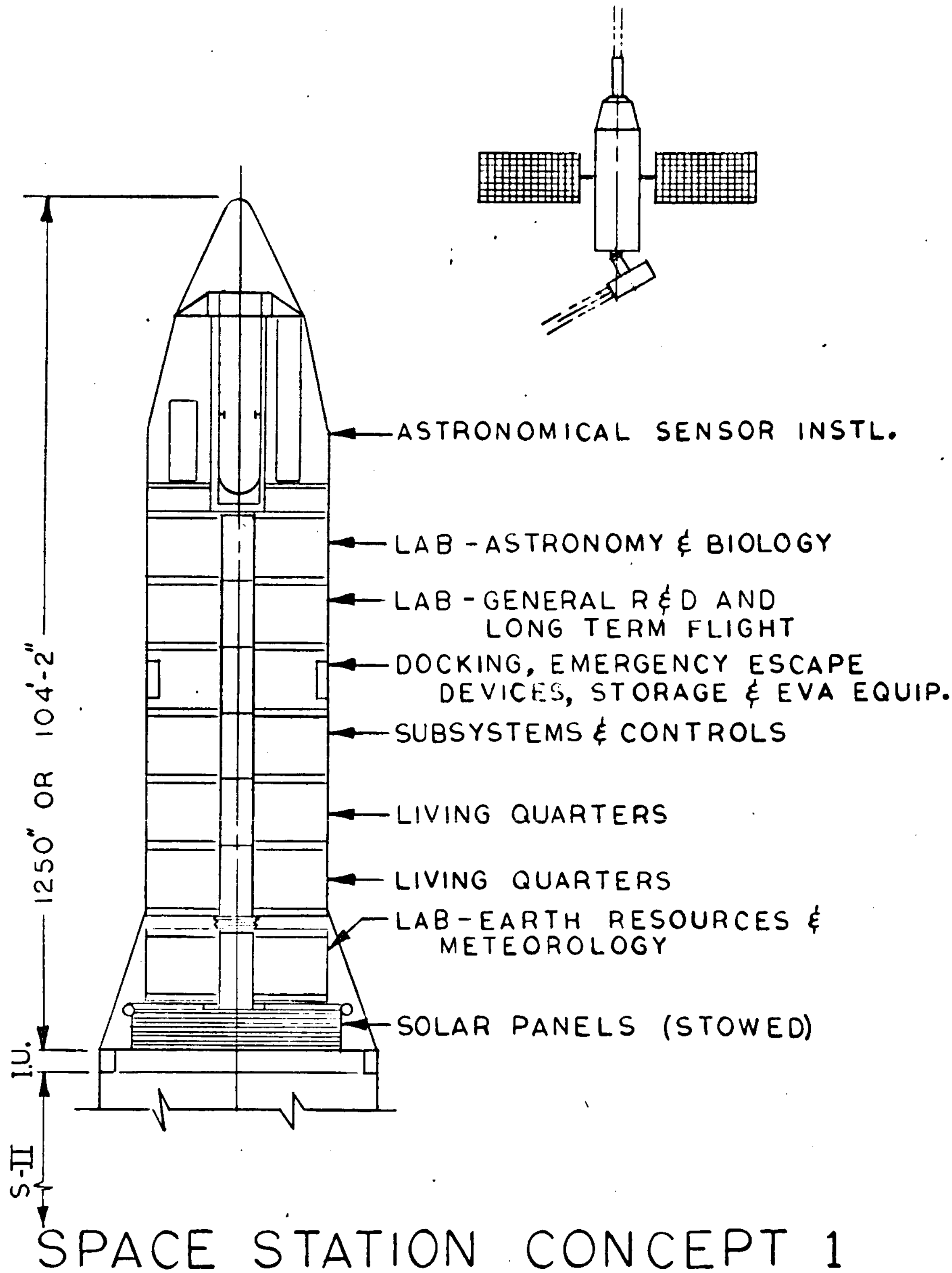
FIGURE 4.26

categories of experiments, and has a normal crew complement of 9 men. It would be launched on a 2-stage Saturn V launch vehicle, unmanned. Figure 4.27 shows the basic general arrangement. The station design approach is modular in that all major functions are provided for within separate compartments. Earth sensors are in the earth resources and meteorology lab compartment which is located at the lower end of the station and gimballed to allow the solar cells to be fixed to the station. If these sensors were not gimballed, the entire station would have to be earth oriented during their operation, thereby requiring that the solar cells be gimballed.

4.3.2.2.2 Concept 2

Concept 2 utilizes two separate 9 man space stations to accomplish the experiment program. Figure 4.28 illustrates the two configurations. Basically, one-half of the experiments are accommodated in each station. The total laboratory volume for each of the Concept 2 stations is the same as for the single, Concept 1 station. Therefore, essentially twice as much volume, weight and crew time is available for each experiment category.

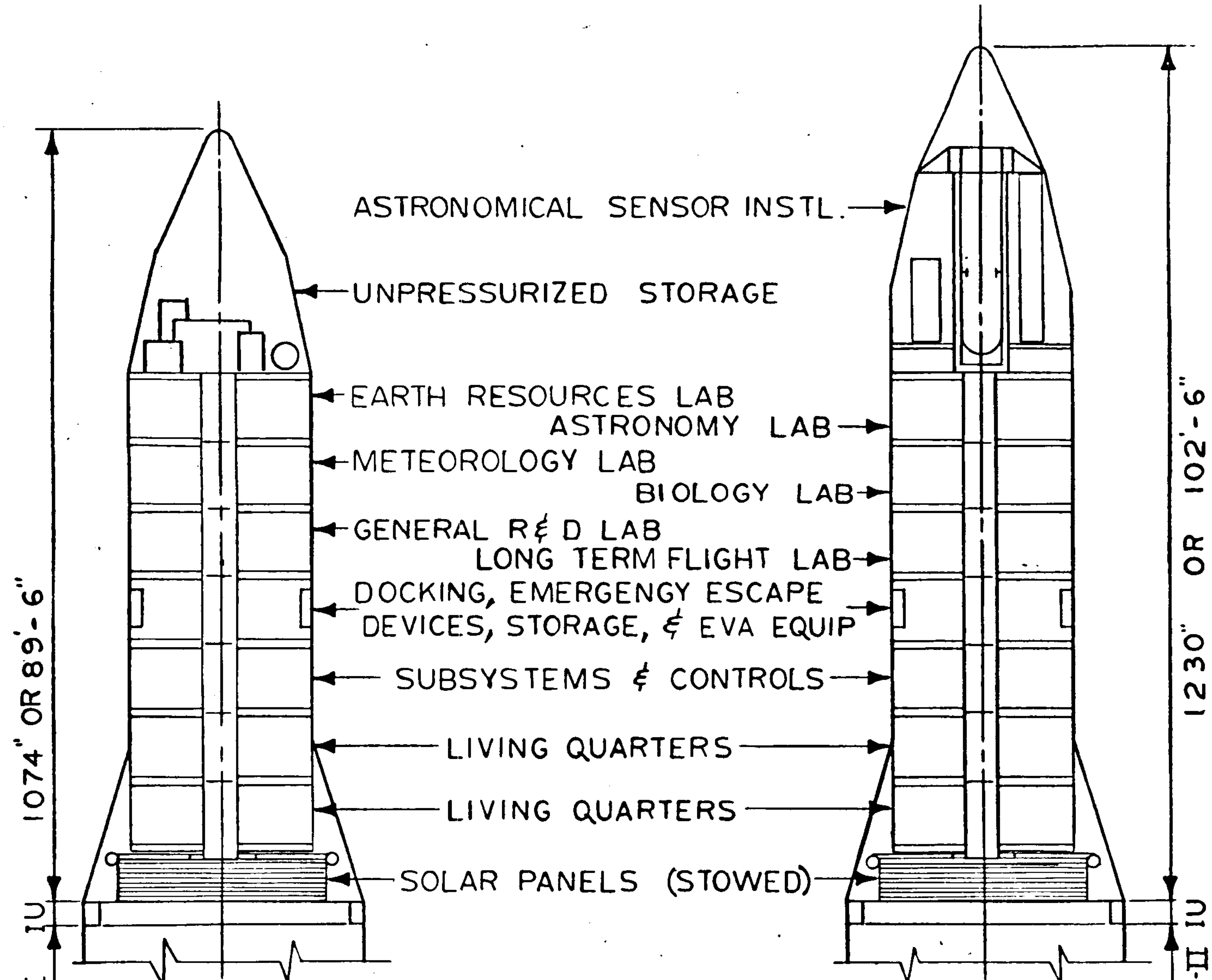
The basic structure, crew accommodations and systems are identical for each station. Because the astronomy experiments are on one station and the earth resources and meteorology experiments are on the other, the respective orientation modes can be optimized. The astronomy station will be solar or inertially



SPACE STATION CONCEPT 1

9 MAN, ZERO G, 260" DIA

FIGURE 4.27



SPACE STATION-CONCEPT 2
9 MAN, ZERO G, 260" DIA, DUAL LAUNCH
FIGURE 4.28

oriented, and the earth resources and meteorology station will be earth oriented. One of the reasons for two stations is to divide the experiments into two groups, each group containing those experiments that are the most compatible with respect to station design and operation.

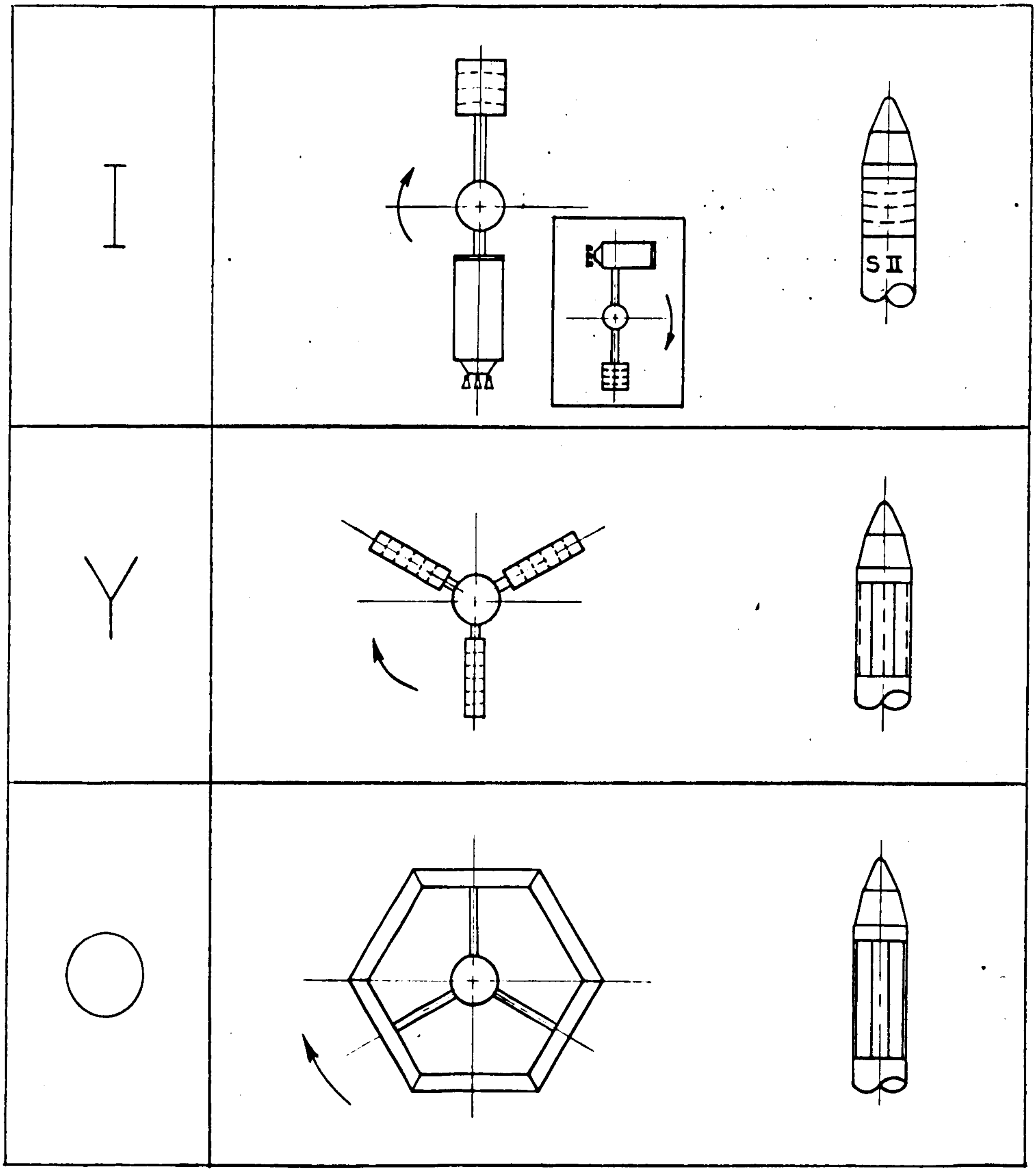
4.3.2.2.3 Concept 3

Concept 3 is a single, Saturn V launched, artificial gravity space station designed to accommodate all categories of experiments. Configurations for crews of 9 men and 24 men have been developed.

Three basic configuration concepts have been identified to date and are illustrated in Figure 4.29. To differentiate between them, nomenclature has been chosen that is a function of their orbital shapes viewed normal to their spin planes. These shapes will be referred to as the I, the Y, and the O. Each has a non-rotating hub located at its mass center to provide a zero gravity volume, and a habitable volume located at a distance from its spin axis to produce artificial gravity.

It was not within the scope of this study to evaluate configuration concepts; therefore, any tendency to compare the I, Y, and O configurations in the subsequent discussion should not be construed as an evaluation.

To be dynamically stable, the space station must spin about a principal axis of inertia. It is apparent that, for the same



BASIC CONFIGURATION CONCEPTS

ARTIFICIAL GRAVITY

FIGURE 4.29

station mass and maximum spin radius, the O has the most stability - the maximum possible. The Y has adequate stability. The I can be made adequately stable by distributing the station mass to concentrate as much as possible in the plane of rotation and to either side of the longitudinal axis.

The hub arrangement for each concept can be basically identical. As indicated by Figure 4.29, the same launch packaging and deployment constraints apply in each case.

Substantial differences exist in the arrangement of the artificial gravity area of each concept. For the I, the entire artificial gravity volume is a single integral cylindrical module. The Y has three separate modules radially extended from the hub, and the O may have as many as six modules, circumferentially located about the hub. The I and O have the most commonality since the entire gravity area is accessible without traversing the hub.

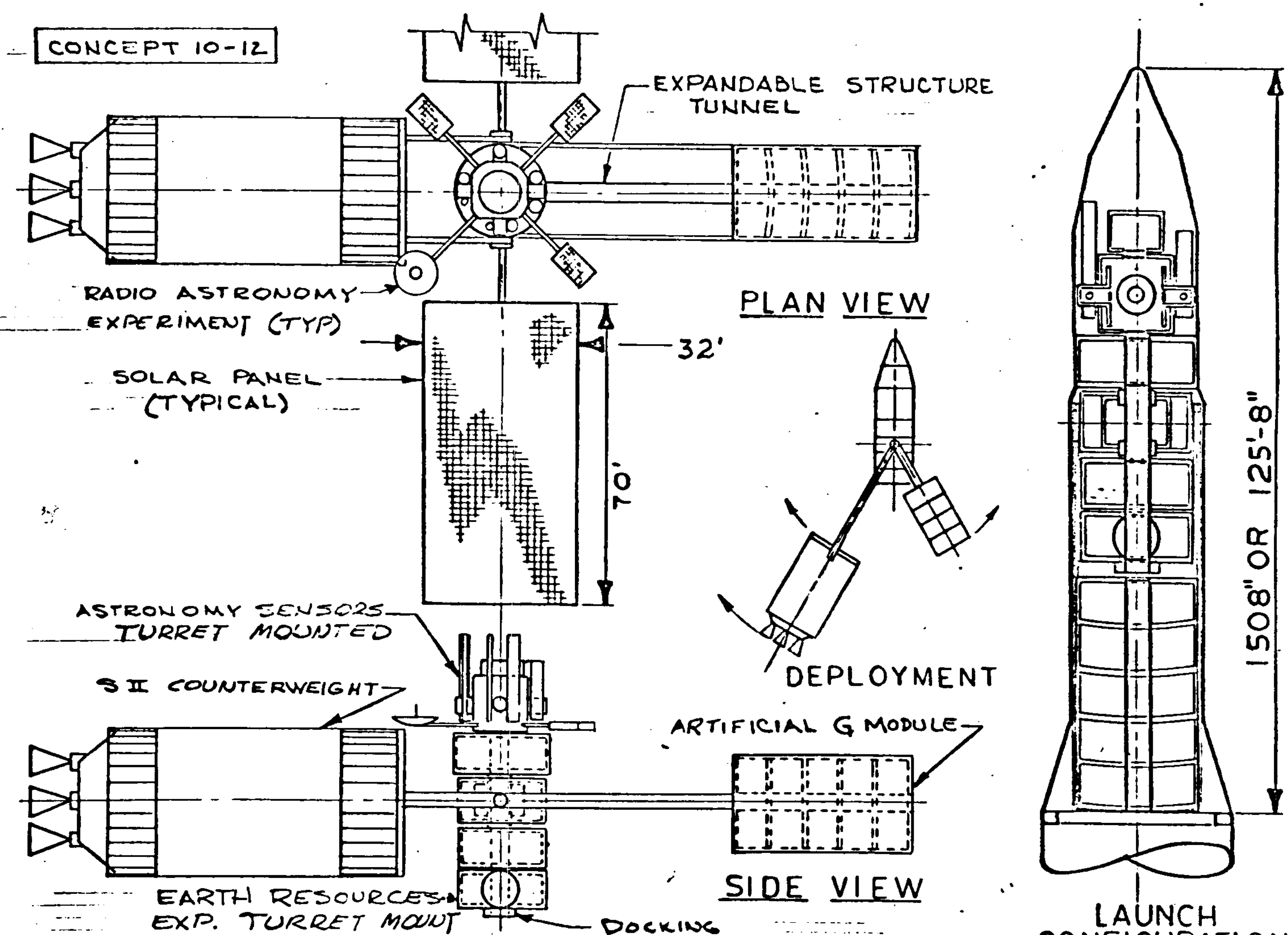
The entire gravity area of the O has a relatively uniform gravity level. The Y has a varying gravity level because of the radial arrangement of compartments. The I is "in-between" with a moderate range of gravity levels.

The maximum achievable spin radius is more limited for the O because, as the spin radius is increased, each of its artificial gravity modules must be lengthened. This tends to increase station volume and/or launch envelope length. Both the Y and I can achieve a larger spin radius with a minimum effect on volume and launch envelope by adding telescoping elements to the tubes connecting the hub and artificial gravity modules. The volume of the Y and I can be radically changed without affecting the basic concept.

In summary, it is believed that these three concepts represent feasible and attractive approaches to an artificial gravity space station which can accommodate all presently known crew, system, and experiment requirements.

The I configuration was chosen as a baseline for this study. The basic characteristics of the I configuration relative to experiment integration are applicable, in principle, to the Y and O configurations.

Figure 4.30 shows the general arrangement of a 9-man, 260 inch diameter, I configuration, artificial gravity station. The station consists of the hub, a cylindrical artificial gravity module and the spent S-II stage counterweight. The station is deployed in orbit from the launch configuration by rotating the artificial gravity module 90° in one direction about an axis normal to the hub centerline, and rotating the S-II stage 90° in the other direction about the same axis as indicated in the figure. The truss linkages that attach the artificial gravity module and the S-II stage to the hub are then telescoped to the proper length. An expandable structure tunnel allows transfer of crew or equipment between the hub and artificial gravity module in a pressurized



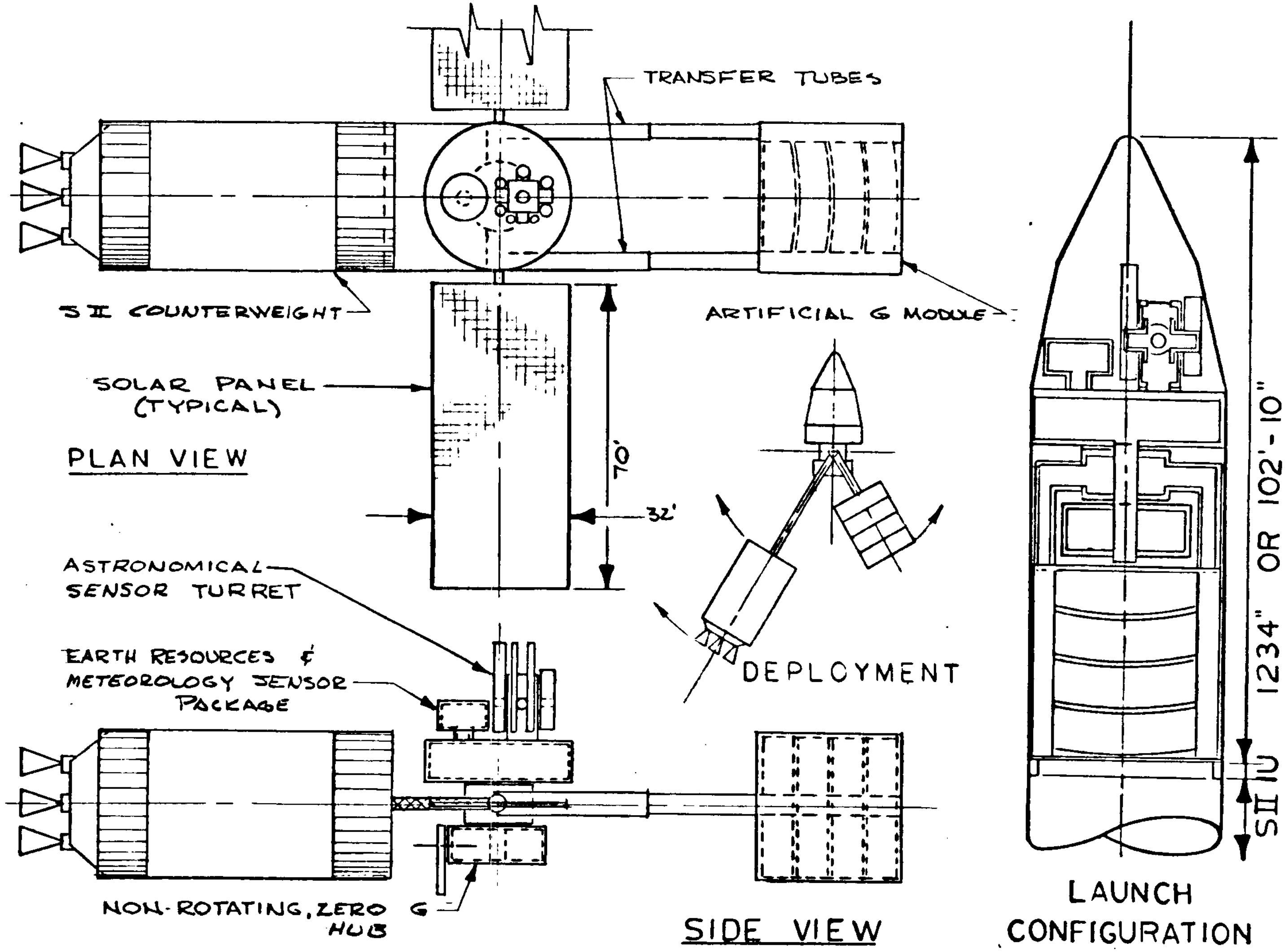
SPACE STATION - CONCEPT 3

ARTIFICIAL G, 9 MAN, 260" DIA

FIGURE 4.30

environment. The artificial gravity module contains crew living accommodations, systems for its operation, the command and control center, emergency escape devices, lab volume for data reduction, and experiments that are compatible with the artificial gravity environment. The non-rotating hub contains most of the lab volume, astronomy and earth sensors, systems for its operation and living accommodations for crewmen who are zero gravity test subjects. The drawing shows a nominal rotational radius of 75 feet for the middle of the artificial gravity module. A considerably larger radius is feasible, if required. The astronomical sensors are turret mounted to the non-rotating hub to provide 2-axis gimballing and allow access for maintenance and experiment setup. Earth sensors are also turret mounted at the opposite end of the hub. The earth sensor installation allows them to be retracted within the earth resources and meteorology lab module for maintenance, film changing, etc., and for launch packaging. The basic orientation of this configuration is to nominally maintain the "astronomy" end of the spin axis pointed toward the sun. The solar panels are fixed to the rotating portion of the hub. This provides a desirable increase in the mass moment of inertia of the rotating portion of the station about the spin axis.

Figure 4.31 shows the general arrangement of a 24 man, 396 inch diameter, I configuration artificial gravity station. It is



SPACE STATION CONCEPT 3
ARTIFICIAL G, 24 MAN, 396" DIA
FIGURE 4.31

basically similar to the 9-man, 296 inch diameter station in arrangement and concept. The larger diameter (same as the S-II stage) allows an increase in volume within the allowable launch envelope and more flexibility in sensor installation.

Figure 4.32 shows more detail of the hub of the 396 inch diameter station. Both astronomical and earth sensors are located on the same end of the hub. This allows the spin axis to be pointed toward the sun with the sensors on the end of the hub opposite the sun. The solar sensors, however, are on the sun end of the hub to avoid having their view of the sun interrupted periodically by rotating portions of the station. An important detail shown in the figure is that the non-rotating hub is attached to the rotating portion of the station through a 2-axis gimbal. Conceptually, the gimbal prevents the transfer of wobble motion from the rotating protion of the station to the hub. Springs and dampers may also be incorporated into the gimbal mount so that the mass and inertia of the hub is utilized to aid in passively damping wobble in the rotating station.

4.3.2.3 ROTATIONAL STABILITY

The rotating space station is a classical application of the physical laws governing its rotational motion, since it functions in an environment almost entirely without resultant, externally applied forces. Preliminary data indicates that the effects of aerodynamic drag, gravity gradient torque, solar pressure, etc. on the rotational stability are

DETAIL—ARTIFICIAL G SPACE STATION
HUB (396" MAX DIA)

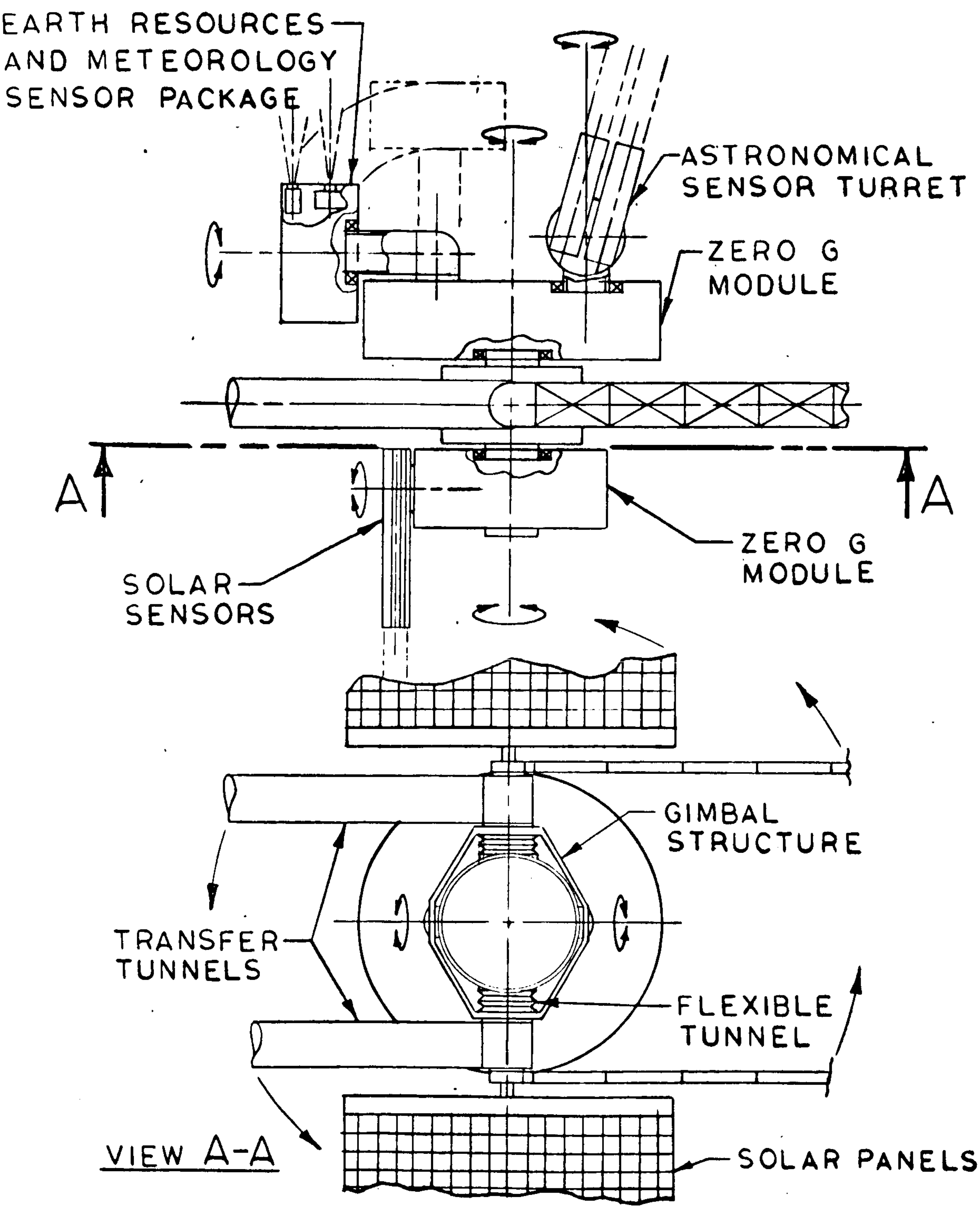


FIGURE 4.32

negligible. Once the station is spinning in a uniform manner it will tend to remain in this state and afford a very stable platform with inertially fixed attitude.

The primary disturbances that affect the motion of the station are torques applied during docking and internal mass movements. External torques change the angular momentum of the station by changing the spin rate or attitude of the spin axis or both. Random external torques and internal mass movements will produce "wobble," a complex angular motion. The most common internal mass movement will be that of the crew moving from one location to another. This mass relocation changes the principal axis of inertia and the center of gravity location of the space station. The result is that the spin axis shifts toward a new principal axis. If the old and the new principal axes lie at some angle to each other, the spin axis will rotate through that angle and then "overshoot" an amount equal to the angle. Thus, a wobble is introduced which will continue until removed by passive or active damping systems.

The infrequent occurrence of docking will minimize its impact on the stability requirements of the overall experimental mission. However, crew movements and actions will occur continually and must be accommodated. The gross effect of a crewman moving within the confines of a typical space station was computed as shown in Figure 4.33. The figure also indicates the nature of the wobble which is produced. As shown, the total

$$\theta_{LIM} = TAN^{-1} \frac{2I_{XY}}{I_X - I_Y}$$

IF: $I_{XY} = 6 \text{ SLUGS} \times 14 \text{ FT} \times 14 \text{ FT} = 1176 \text{ SLUG-FT}^2$

$I_X = 35.64 \times 10^6 \text{ SLUG-FT}^2$

$I_Y = 33.83 \times 10^6 \text{ SLUG-FT}^2$

$\theta_{LIM} = 0.0745° = 5$ ARC MINUTES

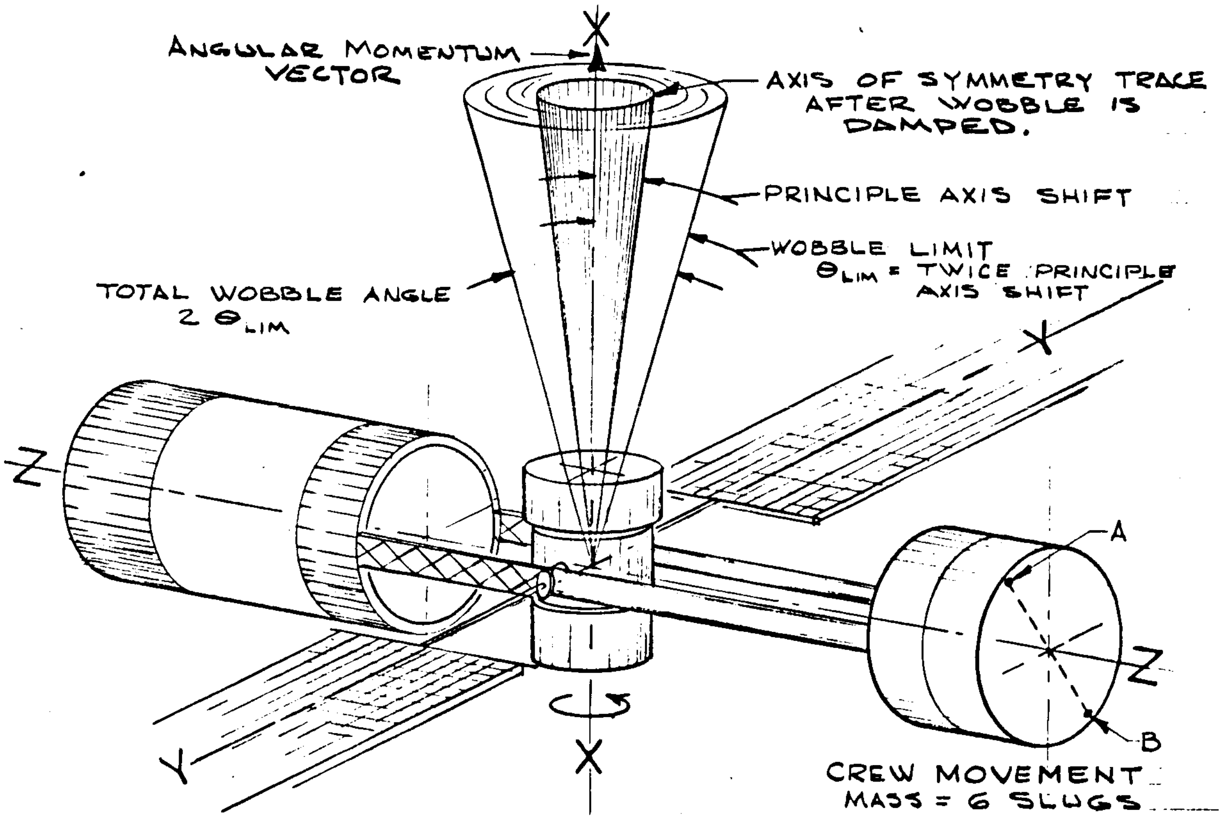
$\boxed{2\theta_{LIM} = 10 \text{ ARC MINUTES}}$

WHERE: I = MASS MOMENT OF INERTIA

I_{XY} = PRODUCT OF INERTIA

θ_{LIM} = WOBBLE ANGLE LIMIT

SUBSCRIPTS X, Y, Z = STATION AXES



ARTIFICIAL GRAVITY SPACE STATION
RESPONSE TO INTERNAL MASS MOVEMENT
FIGURE 4.33

wobble angle resulting from a crewman moving diagonally across the artificial gravity module from position A to position B is computed to be approximately 10 arc minutes. If the crewman remained at position B and the wobble damping system did not change the total angular momentum, the station will assume a new attitude with respect to the centrifugal force vector. That is, the axis of symmetry will be moving about the angular momentum vector. The total angular motion would be approximately 5 arc minutes. If the man returned to position A, or if a mass balance system compensated for his movement, the station would return to its original state of motion. Station "tilt" induces an angular motion of the non-rotating hub because the hub axis no longer coincides with the spin axis. If large amounts of cargo or equipment were added to the station, or relocated within it, the station "tilt" could become excessive. Also, the center of mass, and thus the axis of rotation, may be moved away from the center line of the hub bearings producing excessive lateral "run-out" of the hub. Therefore, a mass balance control system will be required to compensate for large mass additions or relocations.

Basic system requirements relative to station rotation are summarized as follows:

a. The desired station spin axis must be a principal axis about which the mass moment of inertia is a maximum.

b. A system must be included which will provide active, static and dynamic balance control of the station about the desired spin axis.

c. A system must be provided which will damp wobble.

d. A system must be provided to position the spin axis in the desired inertial attitude. If the spin axis is to be sun oriented, a system is required to periodically precess the spin axis to maintain the sun pointing attitude.

A summary of potential stability disturbances, resultant effects and potential compensating techniques is provided in Table 4.9. The types of compensation systems shown are indicative of the concepts being considered for the artificial gravity space station.

TABLE 4.9

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

4.4 WEIGHT

4.4.1 MAJOR CONTRIBUTING FACTORS

4.4.1.1 MANNED SPACECRAFT HISTORY

A brief survey of this country's manned spacecraft programs shows that early design weights have always been exceeded. The general range of design weight increase is between 20 and 40 percent. Fortunately, boosters developed during the same time period improved their payload capability to overcome the spacecraft weight increase.

4.4.1.2 MANNED SPACE STATIONS

A survey of various space station studies (past contractor studies) reveals that a wide discrepancy exists in the weights estimated for the various systems. The discrepancy is of such a magnitude that essentially no confidence level can be established. It should be noted that several of the systems are especially prone to unexpected weight growth: structure, environmental control, crew accommodations, electrical power and experiment systems.

The fact that space stations have volumes much greater than current spacecraft is important in the estimation of weights. It should be considered that if volume is available, it will be used. The absence of this consideration is likely to induce unexpected weight growth.

The primary considerations in the establishment of weight goals are as follows:

a. Booster payload capability

b. Space station design weight margin

c. Space station weight growth

(1) Spacecraft history

(2) Volume/system considerations

4.4.2 GENERAL CONFIGURATION WEIGHTS

A generalized approach for estimating spacecraft weight by system is used to obtain the weight data. Tables 4.10 and 4.11 are summary comparisons of four conceptual space stations and include some of the prime parameters that influence the weight data. Table 4.12 is a weight breakdown of the systems and expendable items for 9 and 24-man crews. It is assumed that each system will have a capability to operate for 3 months with a 50 percent additional margin. Resupply of expendables and selected spares is assumed to occur at 3-month intervals. Table 4.13 provides comparable data for a 6-month resupply interval.

4.4.3 GENERALIZED STRUCTURAL WEIGHT APPROACH

The generalized approach for estimating spacecraft structural weight is shown in Figure 4.34. This figure indicates the variation of structural weight in pounds per cubic foot with total body volume. The data points (excluding 3A, 4, and 4A)

TABLE 4.10

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 4.11

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 4.12

SYSTEMS AND EXPENDABLE WEIGHT - INITIAL LAUNCH

(3 Months Resupply plus 50% Margin)

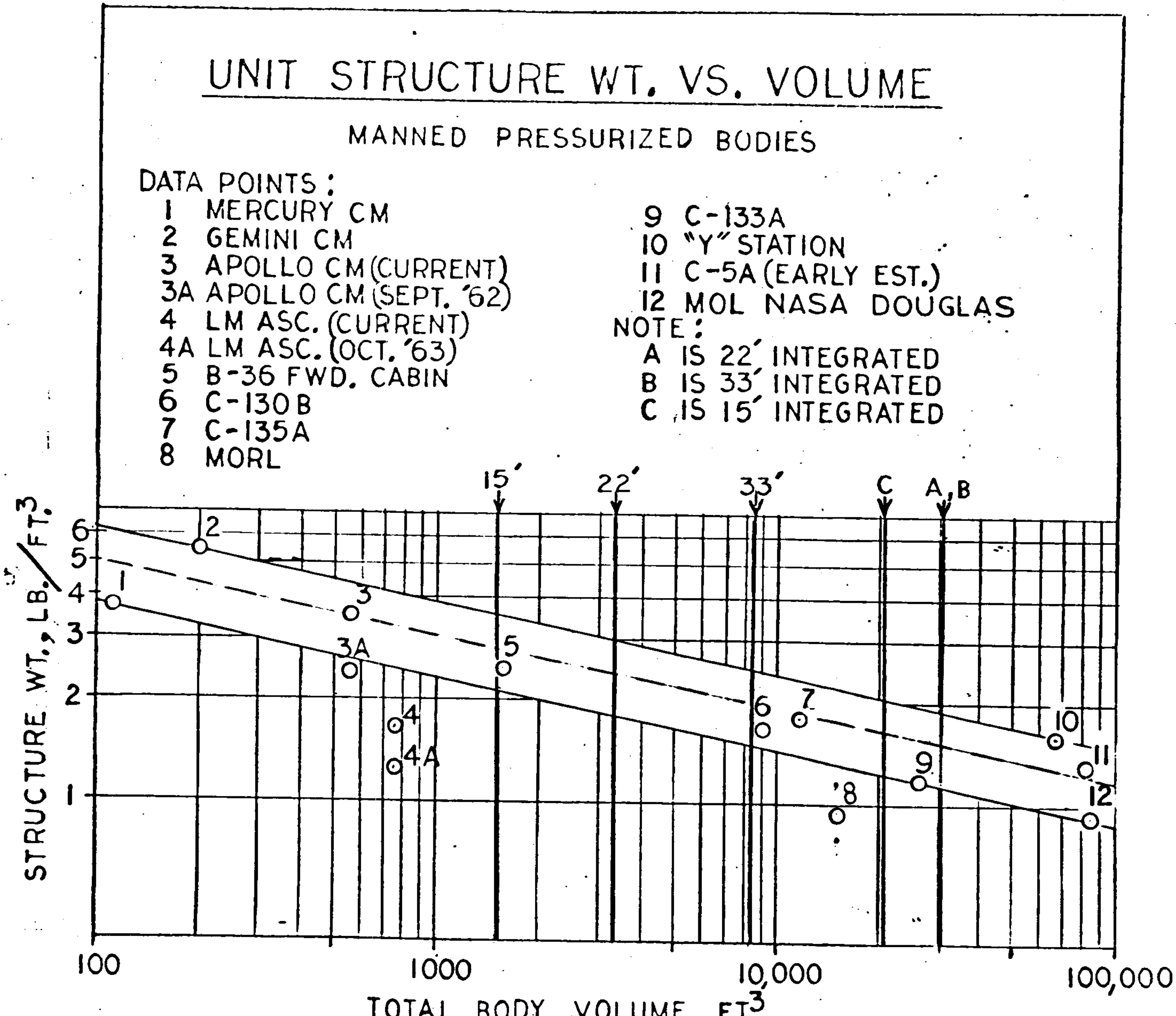
SYSTEM	9-MAN CREW	24-MAN CREW
INERT	(Pounds)	(Pounds)
Environmental Control	6220	12880
Crew Support	1800	4600
Electrical Power	11330	18100
Communications & Data Management	1500	1560
Instrumentation	500	500
Guidance and Control	1300	1300
Reaction Control	2700	2750
Cryogenic Tankage	2640	3450
TOTAL INERT	27990	45290
EXPENDABLES		
Oxygen	3380	7370
Nitrogen	3570	4130
Food	2680	7130
Environmental Control	140	380
PLSS Water	430	1140
PLSS LiOH	210	550
Attitude Control Propellant	3460	3540
TOTAL EXPENDABLES	13870	24240

TABLE 4.13

SYSTEMS AND EXPENDABLE WEIGHTS - INITIAL LAUNCH

(6 Months Resupply plus 50% Margin)

SYSTEM	9-MAN CREW	24-MAN CREW
INERT	(Pounds)	(Pounds)
Environmental Control	6220	12880
Crew Support	3600	9200
Electrical Power	11330	18100
Communication & Data Management	1500	1560
Instrumentation	500	650
Guidance and Control	1300	1300
Reaction Control	4780	4900
Cryogenic Tankage	4450	5720
TOTAL INERT	33680	54310
EXPENDABLES		
Oxygen	6240	14050
Nitrogen	5430	6000
Food	5360	14260
Environmental Control	280	760
PLSS Water	860	2280
PLSS LiOH	420	1100
Attitude Control Propellant	6280	6440
TOTAL EXPENDABLES	24870	44890


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}
TOTAL BODY VOLUME, FT.^{3}
FIGURE 4.34

are for vehicles in which the pressurized volume is approximately 50 percent or more of the total volume. The dashed line represents a first estimate of the structural weight for a zero gravity space station. It is recommended that the upper solid line on this figure be used for artificial gravity stations.

4.5 MARS MISSION - SPACE STATION COMPARISON

The study ground rules, the mission, system and subsystem requirements for the Mars Flyby Mission and for the Earth Orbiting Space Station were compared to identify their commonalities and differences.

4.5.1 MAJOR DIFFERENCES

The important differences were extracted to show where additional study would be needed to allow common program definition, design, development, testing and, to some degree, hardware. Table 4.14 lists the differences under two categories; ground rules and system requirements.

4.5.1.1 Groundrule Differences

The differences in ground rules represent those items that may be adjusted to make the requirements of the two missions more compatible. The items and how they may affect the compatibility of the two missions are briefly discussed in the following paragraphs.

4.5.1.1.1 Crew Size

Crew size affects the overall size and to some degree the shape of a vehicle because of the necessary areas and volumes required to provide a habitable interior. The sizing of environmental control, life support and crew systems is also affected.

TABLE 4.14

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

4.5.1.1.2 Zero-Artificial Gravity

The Mars mission study was groundruled to use zero gravity while the space station study used both zero and artificial gravity. Should one mission use artificial gravity and the other use zero gravity, a lesser degree of compatibility is possible. This difference would result in the development of systems and functions that would be used on only one mission. The deployment from launch to flight configuration, the wobble damping and part of the stabilization system, the non-rotating lab for experiments, and the interface between the non-rotating and rotating parts are examples of the additional development required for an artificial gravity vehicle. The difference in crew system requirements would affect the compatibility of design, development, testing and use of a crew compartment for both artificial and zero-gravity conditions.

4.5.1.1.3 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 components, and crew rotation. Mission characteristics make resupply for a planetary mission impractical. Resupply is included under groundrules because resupply for the space station can, to some extent, be adjusted to make the requirements of the two missions more compatible. Expendable storage time and capacity, which is dependent upon the resupply interval, is one example which may help achieve compatible requirements.

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

4.5.1.1.5 Experimental Payload

Although some experiments can and will be identical for both missions, others will differ greatly. For example, the Mars mission requires that surface probes and planet orbiting sensors be launched from the spacecraft at planetary encounter, while the space station will contain Earth resources and meteorological sensors permanently attached and specifically oriented to the Earth. Experiments impose requirements on almost all subsystems and the crew; therefore, the largest detriment to compatible vehicle requirements may be the experimental payload.

4.5.1.2 SYSTEM REQUIREMENTS

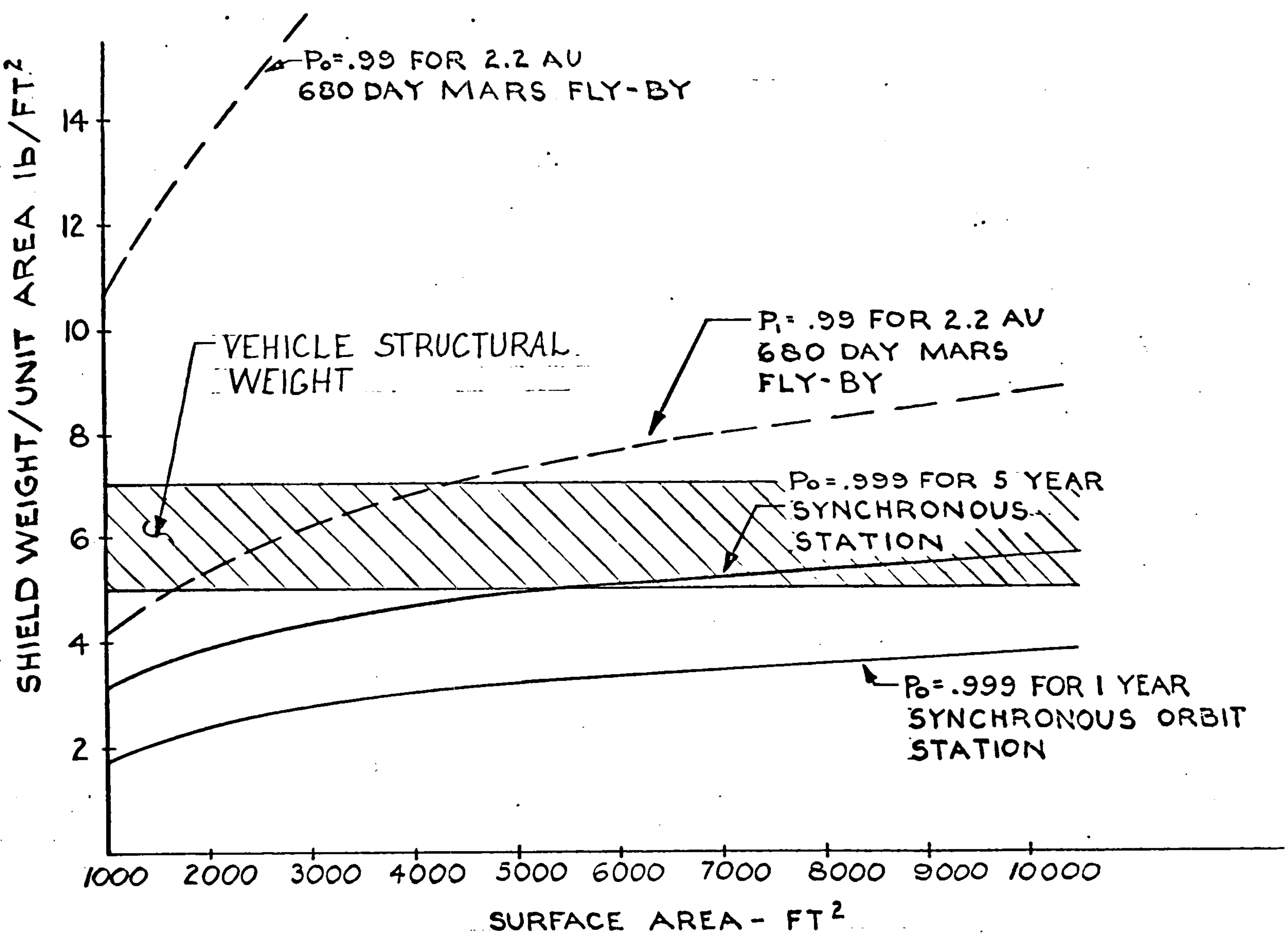
The differences in system requirements represent those items that are imposed by mission requirements. Mission changes and design techniques may be used to make the two mission requirements more compatible.

4.5.1.2.1 Meteoroid Environment

Figure 4.35 shows a comparison of space station and Mars mission meteoroid shield weight requirements. The band bounded by 5 and 7 pounds per square foot represents the expected vehicle structural weight. A probability of .999 for no penetration is shown for a space station in synchronous orbit (worse than low earth orbit) for periods of one year and five years. The top curve represents a probability of .99 for no penetration for a 680-day Mars flyby mission which goes to 2.2 Astronomical Units (A.U.). A probability of .99 for one penetration for the same mission is also shown. Vehicle design which will allow meteoroid shielding to be easily varied and a mission change which would provide a propulsive turn at Mars, therefore avoiding the asteroid belt, are possible ways to reduce the large difference in shielding.

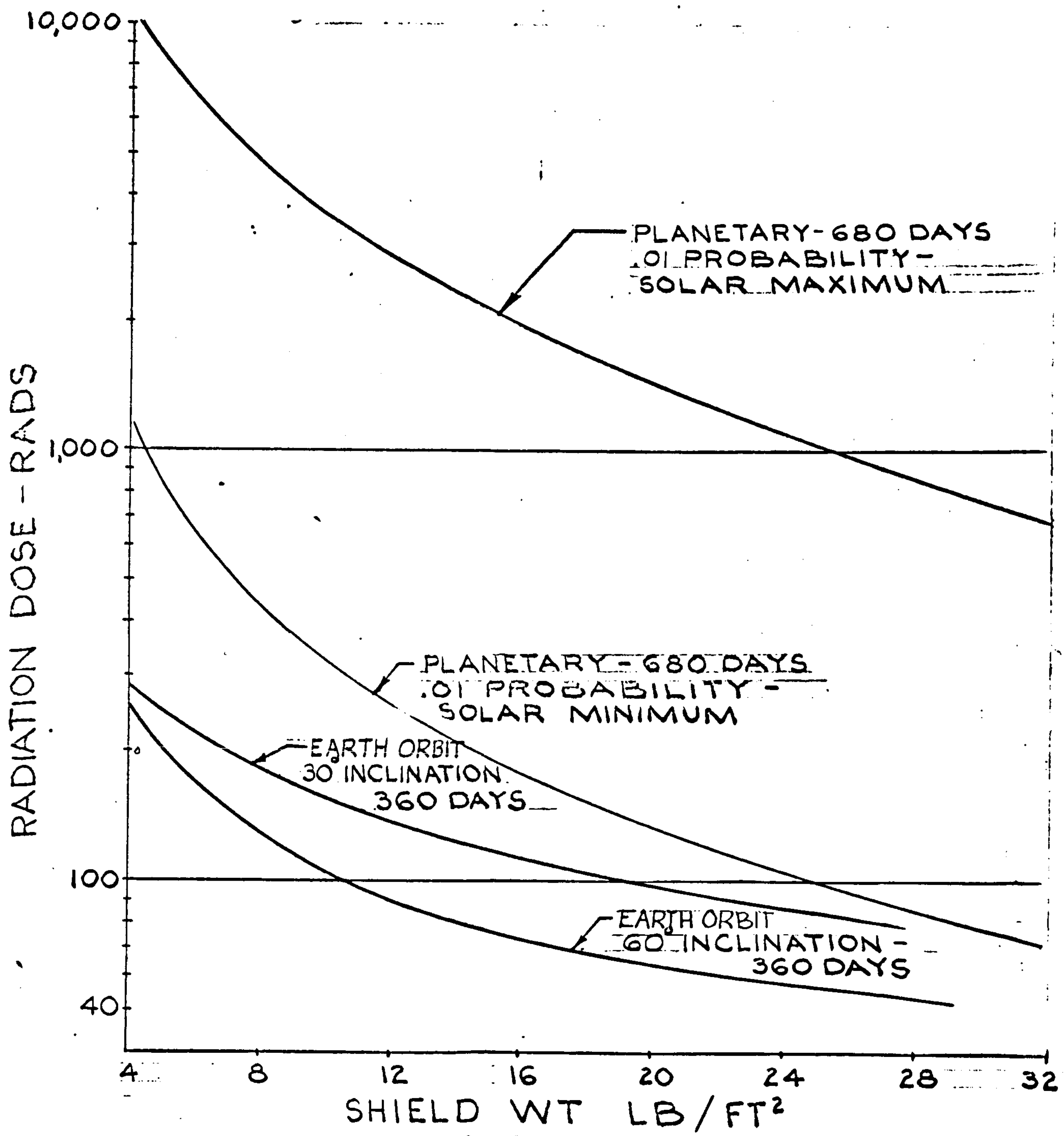
4.5.1.2.2 Radiation Environment

Figure 4.36 shows a comparison of space station and planetary mission radiation protection requirements. The two lower curves show data for a 30° and a 60° inclined low earth orbit. The



SHIELD WT. VS. SURFACE AREA-EARTH ORBIT & PLANETARY

FIGURE 4.35



RADIATION DOSE VS SHIELD WEIGHT

FIGURE 4.36

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, thereby imposing a requirement that is not necessary for the space station in low altitude, low inclination orbits.

4.5.1.2.3 Thermal Environment

The thermal environment for the two missions differ because of the 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 units (A. U.) from the sun, while the Mars mission is such that the vehicle's distance from the sun varies from .6 A.U. to 2.2 A.U. The space station is within the influence of the high earth albedo while the Mars mission vehicle is influenced by planetary albedo for only short periods.

4.5.1.2.4 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 throughout the mission, therefore requiring a propulsive force to maintain the orbit.

4.5.1.2.5 Earth Entry

Earth entry from the earth orbiting space station is a proven operation. Earth entry from a Mars mission will require much higher entry velocities and precise guidance to acquire the entry corridor. The difference is another example of a development required for the Mars mission only.

4.5.1.2.6 Mission Time

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

MANNED LOGISTIC SYSTEM

STUDY

TABLE OF CONTENTS

6.0 MANNED LOGISTICS SYSTEM

For operation of future space stations in earth orbit, there is a requirement for an efficient, versatile logistic system. The influencing factors of various space station programs which affect the logistics system are cargo delivery requirements, personnel delivery requirements and the length of stay at the space station of the personnel. Except for these factors, individual space station configurations have no unique design effect on the logistics system. Each station must have a docking port and some method or mechanism for cargo transfer.

Before a logistic spacecraft system could be described, it was necessary to investigate the launch rate requirements, since the booster cost is a dominant recurring cost of a logistic system. Figure 6.1 shows the factors which determine the number of logistic launches required for a space station program.

6.1 OVERALL CONFIGURATION CHARACTERISTICS

The logistic system configuration characteristics are shown below:

- Low L/D re-entry module
- Saturn IB launch (Saturn V for polar and synchronous missions)
- Land/water landing capability
- Design to reflect reuse
- Simplicity of design

LOGISTIC DIAGRAM — LOGISTIC LAUNCH REQUIREMENTS

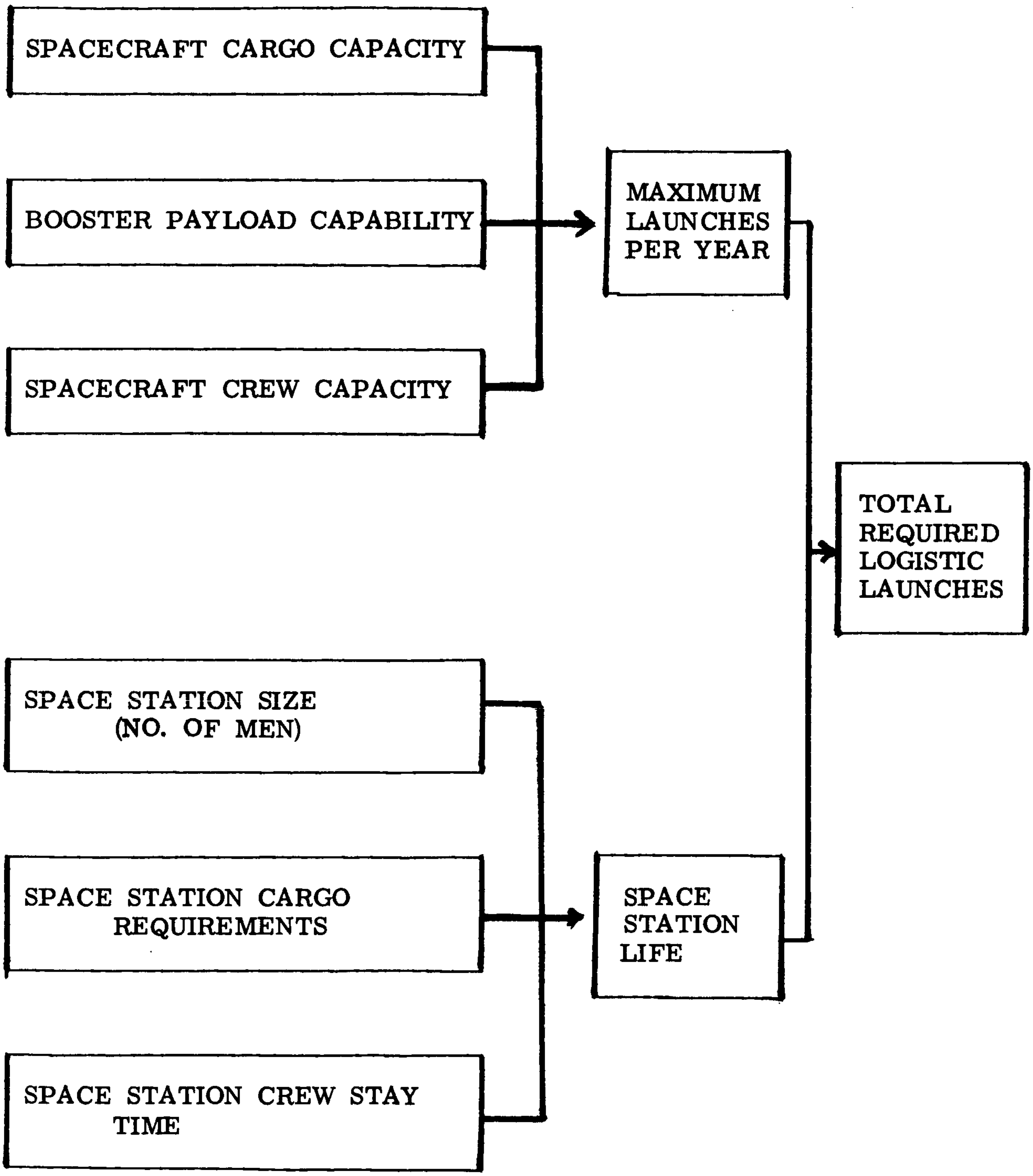


FIGURE 6.1

- Separate crew and cargo modules
- No EVA transfer as routine operation

The major reasons for the choice of the low L/D semi-ballistic re-entry module are as follows:

- Technology developed in Apollo program
- Lightest system weight
- Most cost effective configuration
- Easily integrated with Saturn launch vehicle
- Adequate re-entry maneuver capability for land or water landing
- Maximum cargo capacity
- Minimum development risk
- Abort system developed
- Minimum system complexity and crew participation

Figure 6 .2 shows the various configurations considered, the basic weight of each and the amount of useful cargo that can be delivered to a 260 n.m., 50 degree inclined orbit. The cargo capability is a maximum for the advanced low L/D type (hypersonic L/D = 1/4 to 1/2) vehicle. Even though the vehicle is sized for 6 or 9 men, it has the most cargo delivery capability since it is optimized for the low earth orbital mission. The logistics vehicle will deliver men and cargo to the space station, then return to earth after a short time (up to 4-5 days) consistent with cargo unloading and in orbit waiting times for deorbit to land at selected landing sites.

LOGISTIC VEHICLE CONFIGURATIONS

	3 MAN		6 MAN			9 MAN
DESCRIPTION	BLOCK II CM BLOCK II SM	BLOCK II CM NEW CARGO-PROPULSION MODULE	MODIFIED CM BLOCK II SM	MODIFIED CM NEW CARGO-PROPULSION MODULE	ADVANCED LOW L/D TYPE CM NEW CARGO-PROPULSION MODULE	ADVANCED LOW L/D TYPE CM NEW CARGO-PROPULSION MODULE
WEIGHT	28, 500	23, 000	29, 500	24, 000	19, 000	20, 000
CM	(12, 000)	(12, 000)	(13, 000)	(13, 000)	(8, 000)	(9, 000)
SM	(16, 500)	(11, 000)	(16, 500)			
CARGO MODULE				(11, 000)	(11, 000)	(11, 000)
*CARGO DELIVERED TO 260 N. M. 50 DEGREE ORBITAL INCLINATION	7, 500	13, 000	6, 500	12, 000	17, 000	16, 000

*TWO STAGE DIRECT ASCENT TO 100 N. M.
LOGISTIC SPACECRAFT PROPULSION TO 260 N. M.

FIGURE 6. 2

Figure 6.3 is a plot of annual personnel launch rate requirements for a 9-man and 24-man space station for 3, 4, and 6-month crew duty cycle periods. The number of annual launches is plotted against logistic vehicle crew size. The horizontal line, indicating a 10 limit, is the launch capability at KSC with Pads 34 and 37B operational. A more realistic limit of 4 to 6 a year is shown as a black band. As shown, a 6- or 9-man crew module can significantly reduce the number of annual launches.

Figures 6.4 and 6.5 show annual cargo delivery capability of the various logistic vehicle configurations consistent with the personnel launch rate requirements shown in Figure 6.3. The fourth line is an estimate of the space station resupply requirements per year; the fifth line indicates the amount available for experiments (the difference between cargo capability and space station requirements).

Figure 6.6 is a graphic plot of Figure 6.4 showing cargo delivered versus logistic vehicle size. The one-year estimated requirement of 44,000 pounds for a 9-man space station is shown. The solid lined curve represents 3-month duty cycles, the line with cross marks represents 6-month duty cycles. The number of launches for each case is shown at the end of the curves. Obviously, twice the cargo can be carried on a 3-month duty cycle basis, since twice as many launches are made.

LAUNCH RATE REQUIREMENTS

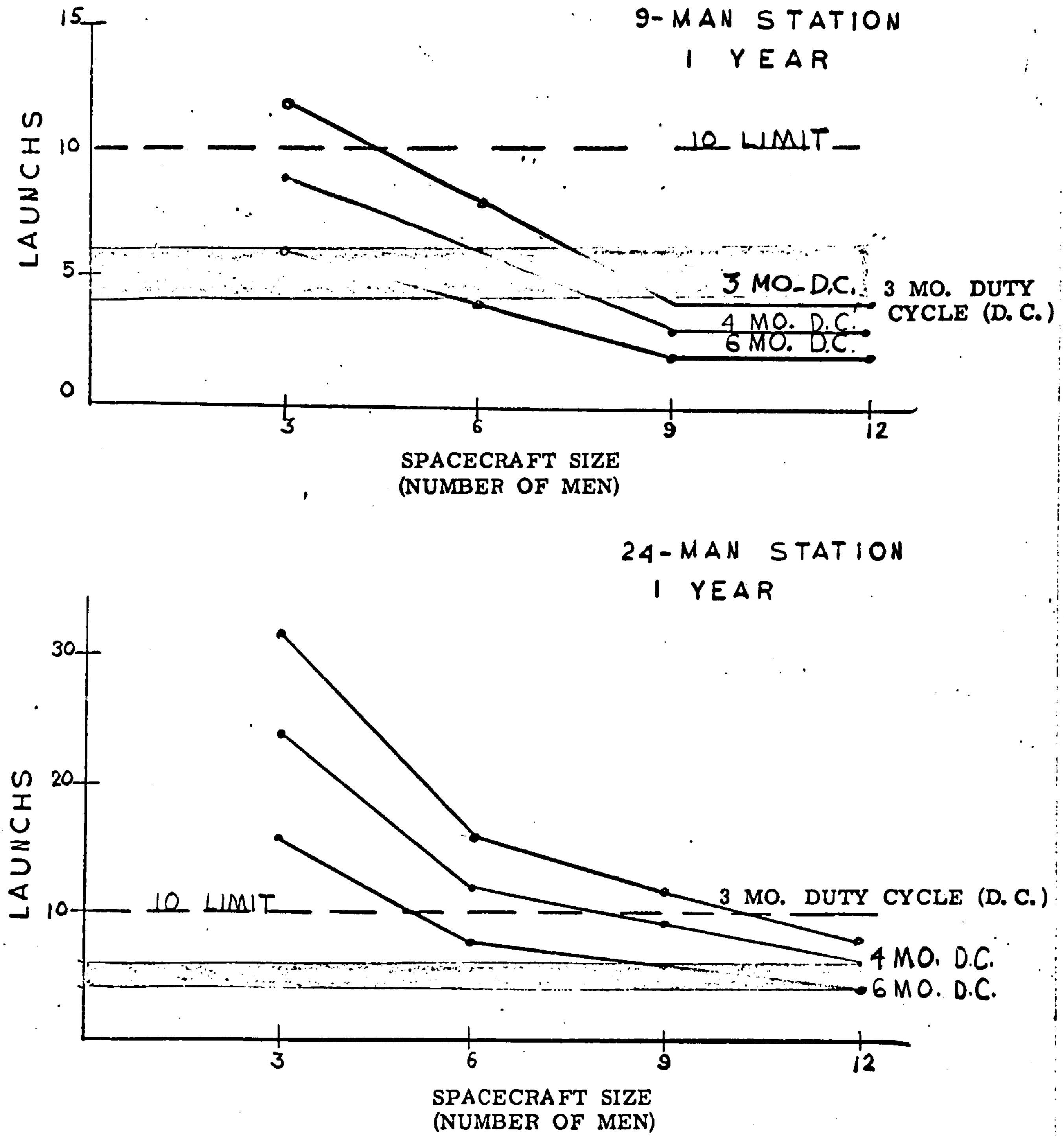


FIGURE 6.3

ANNUAL LOGISTICS CAPABILITY
9-MAN SPACE STATION

		3-MAN		6-MAN			9-MAN
	DUTY CYCLE	BLOCK II CSM	BLOCK II CM NEW CARGO-PROPULSION MODULE	MODIFIED BLOCK II CM BLOCK II SM	MODIFIED BLOCK II CM NEW CARGO-PROPULSION MODULE	ADVANCED BALLISTIC CM NEW CARGO-PROPULSION MODULE	ADVANCED BALLISTIC CM NEW CARGO-PROPULSION MODULE
NUMBER OF LAUNCHES PER YEAR	3 MO.	12	12	8	8	8	4
	6 MO.	6	6	4	4	4	2
CARGO CAPABILITY PER LAUNCH		7,500	13,000	6,500	12,000	17,000	16,000
CARGO DELIVERED PER YEAR	3 MO.	90,000	156,000	52,000	96,000	136,000	64,000
	6 MO.	45,000	72,000	26,000	48,000	68,000	32,000
SPACE STATION SUPPLIES REQUIRED PER YEAR		43,474	43,474	43,474	43,474	43,474	43,474
AVAILABLE FOR EXPERIMENTS	3 MO.	46,526	112,526	8,526	52,526	92,526	20,526
	6 MO.	1,526	28,526	-17,474	4,526	24,526	-11,474

FIGURE 6.4

ANNUAL LOGISTICS CAPABILITY
24-MAN SPACE STATION

	DUTY CYCLE	3-MAN		6-MAN			9-MAN
		BLOCK II CSM	BLOCK II CM NEW CARGO-PROPULSION MODULE	MODIFIED BLOCK II CM BLOCK II SM	MODIFIED BLOCK II CM NEW CARGO-PROPULSION MODULE	ADVANCED BALLISTIC CM NEW CARGO-PROPULSION MODULE	ADVANCED BALLISTIC CM NEW CARGO-PROPULSION MODULE
NUMBER OF LAUNCHES PER YEAR	3 MO.	32	32	16	16	16	12
	6 MO.	16	16	8	8	8	6
CARGO CAPABILITY PER LAUNCH		7, 500	13, 000	6, 500	12, 000	17, 000	16, 000
CARGO DELIVERED PER YEAR	3 MO.	240, 000	416, 000	104, 000	192, 000	272, 000	192, 000
	6 MO.	120, 000	208, 000	52, 000	96, 000	136, 000	96, 000
SPACE STATION SUPPLIES REQUIRED PER YEAR		75, 000	75, 000	75, 000	75, 000	75, 000	75, 000
AVAILABLE FOR EXPERIMENTS	3 MO.	165, 000	341, 000	29, 000	117, 000	197, 000	117, 000
	6 MO.	45, 000	133, 000	-23, 000	21, 000	61, 000	21, 000

FIGURE 6.5

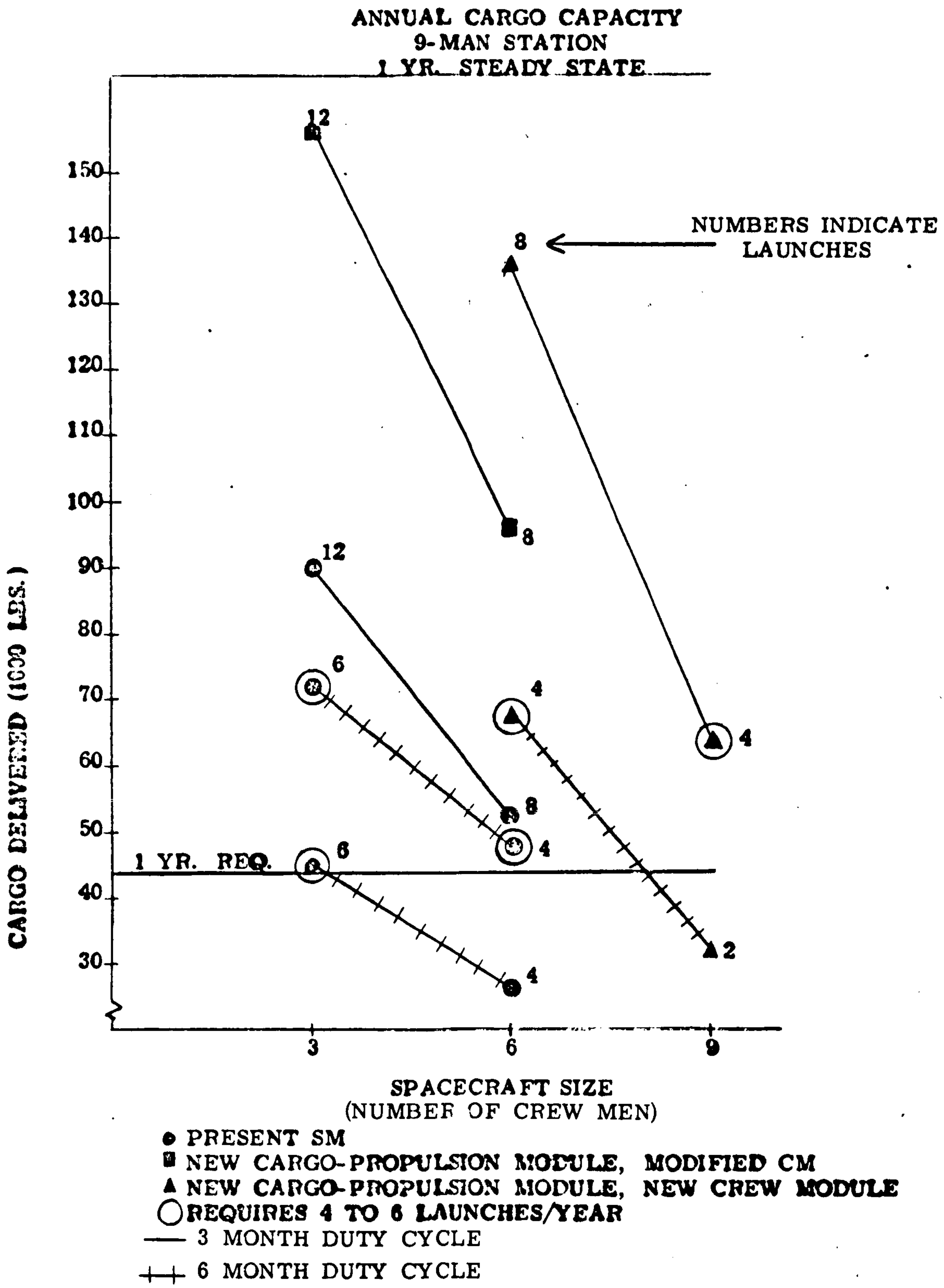

ANNUAL CARGO CAPACITY
9-MAN STATION
1 YR. STEADY STATE
NUMBERS INDICATE LAUNCHES
CARGO DELIVERED (1000 LBS.)
150
140
130
120
110
100
90
80
70
60
50
40
30
1 YR. REQ.
12
8
12
8
6
4
6
4
8
4
4
2
3
6
9
SPACECRAFT SIZE
(NUMBER OF CREW MEN)
PRESENT SM
NEW CARGO-PROPULSION MODULE, MODIFIED CM
NEW CARGO-PROPULSION MODULE, NEW CREW MODULE
REQUIRES 4 TO 6 LAUNCHES/YEAR
3 MONTH DUTY CYCLE
6 MONTH DUTY CYCLE

FIGURE 6.6

6.2 RE-ENTRY VEHICLE

Figures 6.7, 6.8, and 6.9 show a 9-man crew module with three different cargo-propulsion module configurations. The initial structural approach used is that of the shell-stringer-frame type with the stringers outside the pressurized skin. The aft heat shield uses ablative material, while radiative metallic heat shield would be used for the conical forebody. An estimated weight summary for the re-entry module is shown below.

Structure and thermal	3400
Crew and furnishing	2760
Navigation and Guidance	320
Communications	210
Displays and controls	295
Earth landing system	730
Electrical power	450
Environmental control system	390
Stabilization and control	415
Total	8970

6.3 LOGISTICS/CARGO MODULE

The purpose of the cargo module is to transport a variety of cargos such as supplied, fuel, experiments, etc. to the space station on a scheduled basis. The major requirement is to have sufficient volume to accommodate maximum cargo consistent with the payload capability of the launch vehicle. A typical cargo breakdown is shown below.

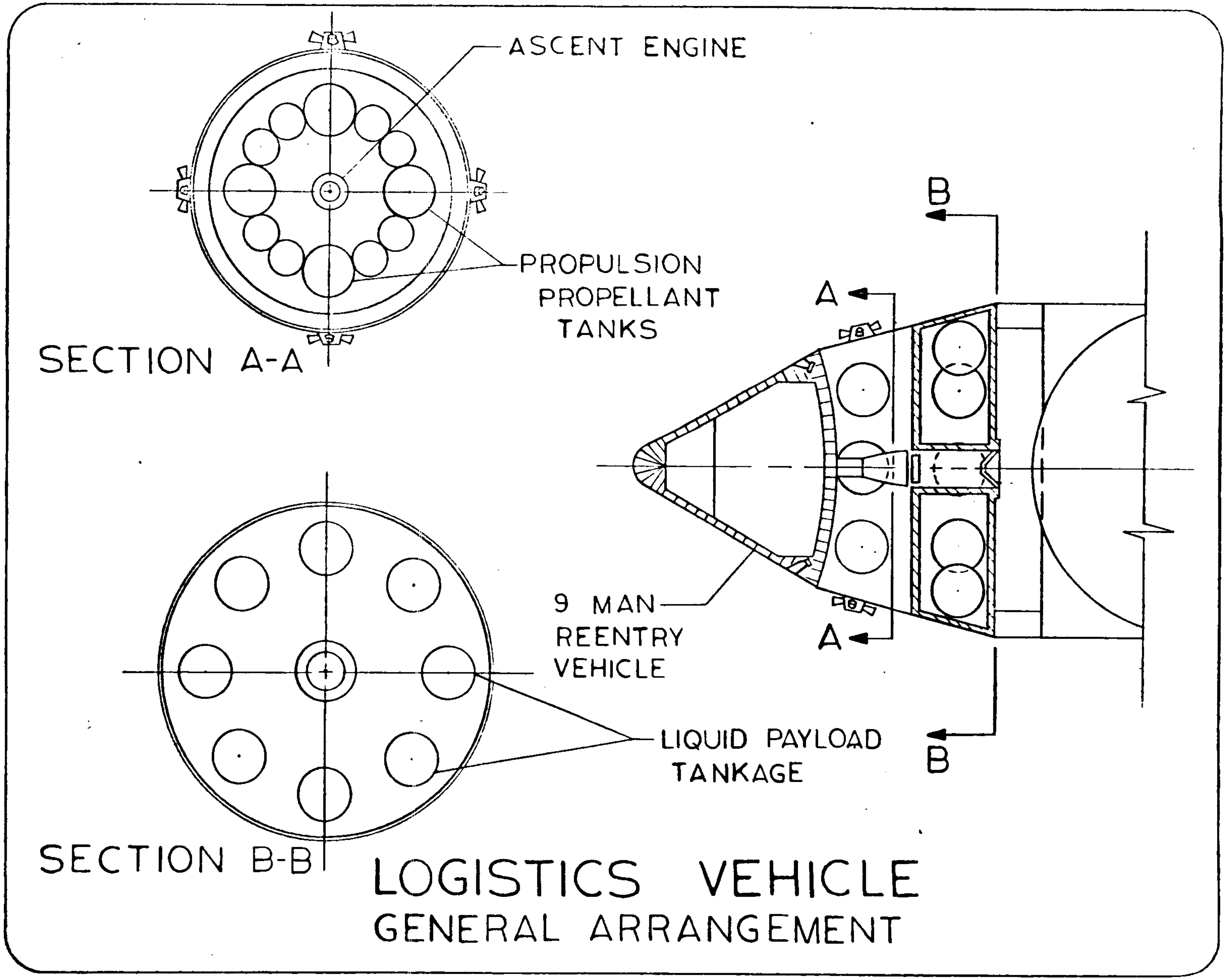

ASCENT ENGINE
PROPULSION PROPELLANT TANKS
SECTION A-A
B
A
9 MAN REENTRY VEHICLE
LIQUID PAYLOAD TANKAGE
SECTION B-B
LOGISTICS VEHICLE
GENERAL ARRANGEMENT

FIGURE 6.7

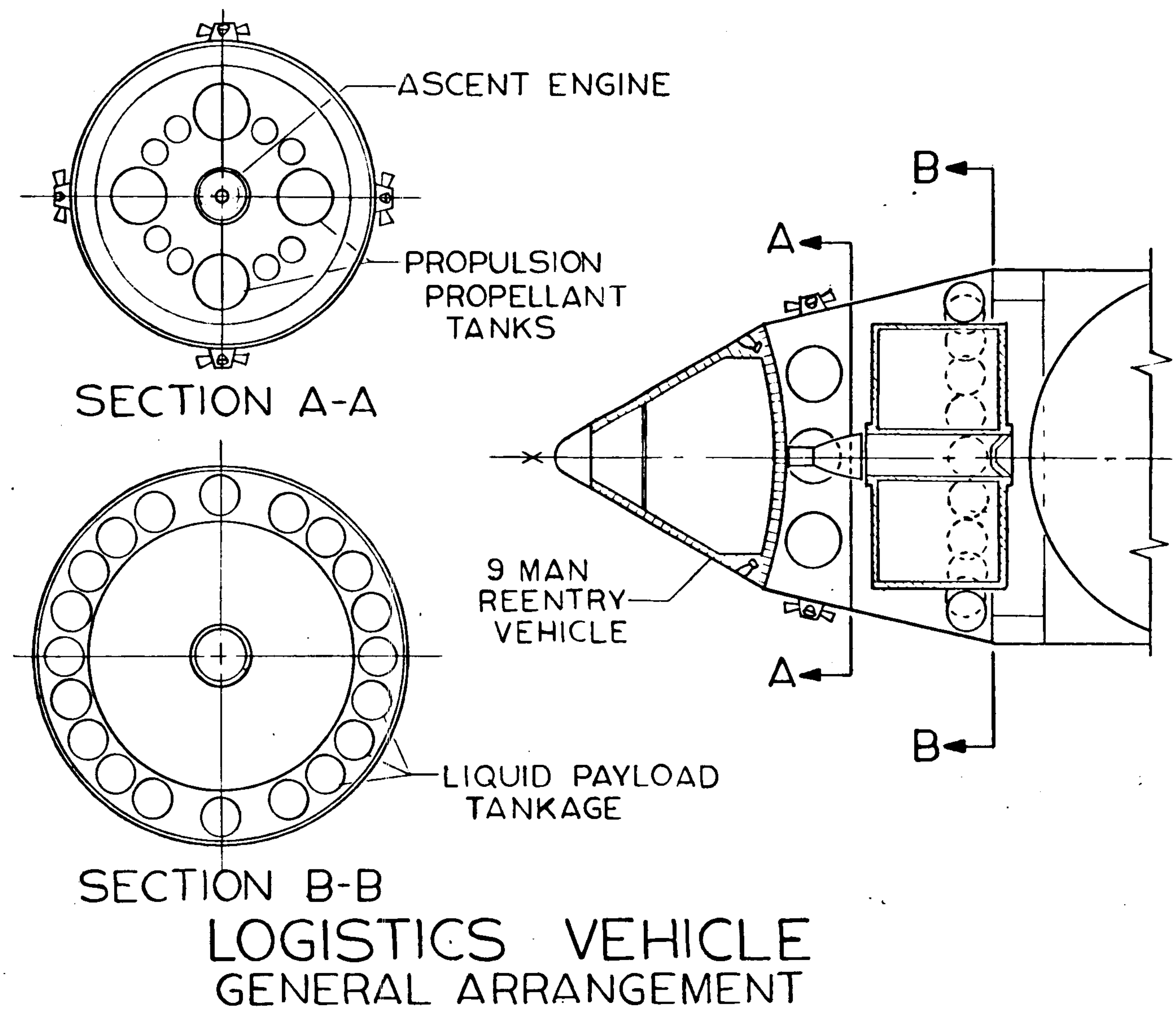
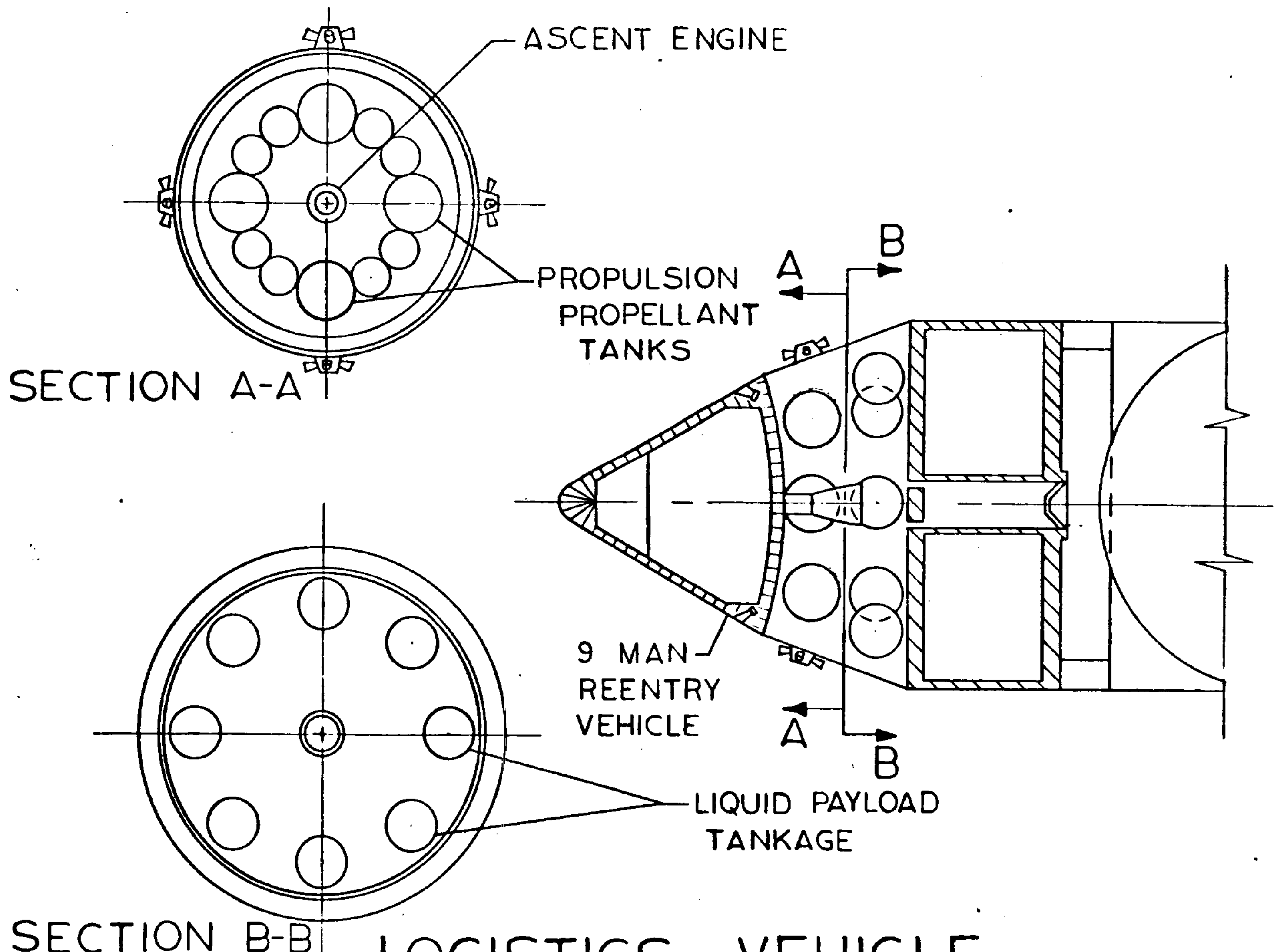

ASCENT ENGINE
PROPULSION
PROPELLANT
TANKS
SECTION A-A
A
A
B
B
9 MAN
REENTRY
VEHICLE
LIQUID PAYLOAD
TANKAGE
SECTION B-B
LOGISTICS VEHICLE
GENERAL ARRANGEMENT

FIGURE 6.8



LOGISTICS VEHICLE
GENERAL ARRANGEMENT

FIGURE 6.9

Type	Weight	Density #/Ft^3	Volume Ft^3
Food	1800	22	81
O_2	2900	72	40
N_2	1150	50	23
Propellant	3600	70	51
Mission Support	1450	22	66
Experiments	5100	20	26
Total	16000		287

Figure 6.9 shows an arrangement which can accommodate a 260 inch diameter cylindrical module. It is the heaviest approach of the three since it consists of the same adaptor as the minimum weight approach plus the weight of the larger can. The useable volume of the can is 2400 cubic feet.

Figure 6.10 shows the weight comparison of the three cargo module approaches. In each case the LM ascent engines have been provided plus the necessary fuel for the required orbital operations to 260 n.m. and retrograde.

6.4 PROPULSION MODULE

The spacecraft on-board propulsion systems required to perform the define logistics mission include launch escape, ascent maneuver, attitude control, retrograde re-entry and landing. The total required delta V budget for orbital maneuver is shown below:

WEIGHT COMPARISON

OF

LOGISTIC CARGO MODULE APPROACHES

ITEM / APPROACH	MINIMUM WEIGHT	183 INCH DIAMETER MODULE	260 INCH DIAMETER MODULE
STRUCTURE & THERMAL	2920	5120	8520
ELECTRICAL POWER	500	500	500
ECS	580	580	580
ORBIT ATTITUDE CONTROL	970	970	970
MAIN MANUEVER SYSTEM	5690	5690	5690
DOCKING STATION	340	340	340
TOTAL	11,000	13,200	16,600

FIGURE 6.10

Circularization	73
Hohmann transfer (100 n.m. to 260 n.m.)	560
One degree plane change	135
Rendezvous	250
Retrograde	400
Actual required	1418
10% contingency	142
Total	1550

Various propulsion module configurations are shown in Figures 6.7, 6.8, and 6.9.

6.5 LAUNCH AND DEPLOYMENT REQUIREMENTS

Prelaunch and launch loads for the logistic vehicle are in all cases within the Apollo configuration nominal load curve, and no redesign of the launch vehicle, nor new launch requirements for the logistic vehicle are apparent at this time.

6.6 OPERATIONS

The logistic vehicle recovery posture for a high inclination mission is comparable to the recovery posture for a low inclination mission. For this study, no consideration was given to the possibility of repeating orbits or propulsive plane change capability in determining the number and location of landing sites.

The operational guidelines are:

- Land landings are primary
- Logistics vehicle and booster qualified for routine operations
- Space station provides adequate emergency refuge for logistics vehicle
- Spacecraft lateral range used to get within landing zone ($L/D = 0.4$)
- Minimum DOD deployment (logistics)
- Landing environment
  1. Launch abort landing in water
  2. Land landing in selected areas
  3. Secondary planned water landings between 40°N 40°S
- Minimum number of land sites

The launch limits for a water landing are between 44° azimuth and 116° azimuth. All other launch azimuths from Cape Kennedy would be a land overfly. A land overfly and possible abort on land is undesirable from a recovery standpoint due to political problems, inaccessability of some land areas and possible damage of spacecraft on rough unknown terrain.

The environmental factors connected with a northern launch from Cape Kennedy reveal that the sea surface temperatures and wave heights would be undesirable for a winter time recovery in the North Atlantic.

In summary, the low L/D re-entry module is capable of accomplishing land landing with a minimum number of landing sites and reasonable waiting times in orbit, prior to retrograde.

6.7 CONCLUSIONS

- Logistics system costs are very sensitive to launch rate requirements.
- 6-man or more logistic vehicle will be required for a reasonable yearly logistic launch rate.
- 9-man logistic vehicle appears to be the optimum size. Larger size vehicles result in deficient cargo capability.
- Cargo-propulsion module replacement for service module required.

OPERATIONS SUPPORT IMPACT

OF SPACE STATION MISSIONS

FLIGHT OPERATIONS DIRECTORATE

MANNED SPACECRAFT CENTER

TABLE OF CONTENTS

9.0 OPERATIONS SUPPORT

The Flight Operations Directorate (FOD) has been engaged for over a year in a continuing study to determine the impact of post-Apollo programs which have been proposed at various levels of the NASA management structure. The technique utilized in this analysis is to investigate "mission classes," rather than each specific proposed mission, since most missions can be generally categorized according to flight operations requirements. Superimposed on this matrix of mission profile characteristics are three other operations factors: (1) experiment emphasis, (2) mission duration, and (3) multiple mission execution. These considerations must be weighed separately to determine a preliminary impact on flight operations support in the post-Apollo era. Mission classes through proposed long-range goals are discussed in the following sub-paragraphs.

9.1 SPACE STATION MISSION CLASSES

The following sections provide brief comments on the operations impact of the four basic classes of missions.

9.1.1 LOW-INCLINATION EARTH ORBIT

No operations impact.

9.1.2 POLAR ORBIT

This mission class is dubious from a performance standpoint for the uprated Saturn I, Saturn IB, logistics flights from Cape

Kennedy if the range safety constraint of no greater than a 140° launch azimuth is imposed. Initial studies of near polar launches indicate that a severe hazard to crew safety is imposed in the event of a launch abort. Early in the launch phase, land landings on both Cuba and the Panama area are possible without velocity correction by the spacecraft. It appears that insufficient time is available to execute either a propulsive or non-propulsive abort in certain launch phases unless performance trade-offs are made which might ultimately jeopardize payload objectives. Later in the launch phase, spacecraft landings in the frigid zones of the southern hemisphere are possible without major spacecraft velocity corrections using the main propulsion system. Since the mission profile would most likely depend on this propulsion system for its final insertion velocity, a single mission failure in this system would result in undesirable spacecraft landing areas near the antarctic. It is believed that neither tha launch vehicle nor spacecraft systems will be considered reliable enough to justify assumption of these risks. If the Saturn V launch vehicle is used, only the unlikely possibility of a land landing remains as a factor. It seems unjustified to develop a land landing capability for three or four missions, and the risk would be too great without it. From an orbital operations support standpoint, station coverage would be quite consistent unless high-latitude tracking sites could be added. The cost of installing permanent stations at high latitudes is again unjustified for the small number of polar missions which might be flown. Therefore, tracking ships and aircraft would be used to fill the coverage gaps, and the multi-mission support consideration would become important. That is, these mobile facilities with their reduced performance capabilities must be shared with adjacent or simultaneous missions, and the required locations might be incompatible. Finally, it is not immediately obvious that proposed mission or experiment objectives justify inclinations close to 90°. From a mapping and survey viewpoint, the areas near the poles are of least interest, and from a forestry/agricultural standpoint, these areas are barren for most or all of the year. From an operations development viewpoint, the ability to enter a polar orbit is no more demanding from a guidance standpoint than any other mission requiring ascent yaw steering.

9.1.3 HIGH INCLINATION EARTH ORBIT (45° - 70°)

The major impact here is the reduction in station coverage and contact times, but effects are not significant enough to preclude this mission class. The basic problems presented by the polar mission are essentially absent for orbital inclinations of 60° or less. Performance requirements are reduced to where the uprated Saturn I launch situation is practical, abort recovery problems vanish, and the need for high-latitude tracking sites becomes much less important. The IBM Orbiting Research Laboratory (ORL) Study of fruitful experiment areas revealed that nearly all civil applications of low-altitude earth orbit missions can be realized for inclinations in the 60° to 75° magnitude range.

9.1.4 EARTH SYNCHRONOUS

Because of current systems operation and performance limitations of the Saturn V launch vehicle, the synchronous mission is at best marginal from an experimental payload standpoint. In addition, the magnitude of the velocity decrement required for deorbit is such that backup propulsion is unavailable in the event of a primary system failure. If the S-IVB stage of the Saturn V can be modified to accommodate either an additional restart or an increased orbital lifetime, or both, the payload margin becomes greater. From an operations support standpoint, tracking coverage requirements of major events during the ascent to synchronous altitude become significant constraints. Recovery operations for direct descent from an American-continent-centered hover point become difficult, and an orbital correction maneuver would be required to effect landing in the western hemisphere. The radiation hazard to the crew with the present spacecraft has been emphasized in previous contractor documentation. Since the mission profile is in this marginal performance category, more definition of mission objectives and payload requirements is necessary before an operations support position can be taken. The limits, for example, which are acceptable for perigee/apogee and inclination must be known.

9.2 EFFECT OF MISSION DURATION

The major effect of increased mission duration, taken by itself, is to reduce flight control functions to a more routine nature. These functions include not only personnel support, but data processing and display, recovery control procedures, and computational services. Therefore, long duration missions would primarily require more normal work shifts and working conditions, as well as revised techniques to accommodate the increased amount and mundane nature of flight data. In addition, inflight rescheduling of flight activities would become the rule, rather than the exception, for longer duration missions with multiple objectives. Therefore, real-time flight planning, possibly using computerized techniques, will become a necessity.

9.3 EFFECT OF MULTIPLE MISSIONS

To apply the factor of multiple mission support to operations requirements, consideration must also be given to mission duration. Each mission requires, in addition to real-time support, a period of about six months of prelaunch preparation and three months of post-flight evaluation. If the assumption is made that the results of one flight will not require long-lead-time changes for the subsequent mission, then minor changes can be absorbed in the normal preparation activity. A simple minded approach, then, is to assume that operations planners and operations support personnel can be added linearly with the number of overlapping missions involved. The support facilities required are then based on the functional needs of these people and the time sharing factors involved. Since the same basic facilities are generally required for full scale simulations as are provided for the actual operation, a final summation of manpower and facilities can be made which varies directly with the number of missions in some simple mathematical expression. The actual support posture is, of course, more complicated than this approach. For example, the support for two different mission classes will undoubtedly be inconsistent; therefore, the support configuration must be structured to handle peak loads. The support requirements will not total to discrete integral support units in most cases, and the next larger integer must be assumed. Finally, if an unreasonable constraint to launch schedules is to be avoided, an additional factor to accommodate program contingencies and schedule slippages must be considered. If the entire analysis is conducted in a conservative manner, then the multiple mission support requirements will involve a minimum of wasted manpower, resources, and facilities. There are also areas of support which contain some degree of flexibility, particularly in the preflight phase. These flexible areas include control-center reconfiguration, mission class analysis, scheduling of initial simulations, and the like. If the support personnel are assumed to be mission oriented; that is, they pursue their particular responsibilities for a given mission from start to completion, a reasonable level of multiple-mission support can be derived for a prescribed mission assignment plan with stated objectives and flight durations.

9.4 EFFECT OF EXPERIMENT EMPHASIS

The degree with which inflight experiments are emphasized for post-Apollo missions affects largely the data handling aspects of operations support requirements. An experiment program no greater than twice the anticipated Apollo involvement would probably require only minor modifications to present mission support facilities and systems. Beyond this level, increased data processing and display equipment, computational capability, and experiment operations personnel will be required. In addition, the experimenter/operations interface will undoubtedly demand new facilities to accommodate experiment observers who are required to conduct real-time operations and evaluation of results. Both the preflight preparation activities and postflight operations and evaluation procedures are greatly complicated by involved and variant experimental objectives. Additional operations facilities and staffing would therefore be required for all operations support functions in a full scale experiment program.