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CFSTI PRICE(S) \$ \_\_\_\_\_

Hard copy (HC) 3-00

Microfiche (MF) 65

# 653 July 65

MILESTONE REPORT OF A STUDY OF  
SPACE MISSION DURATION  
EXTENSION PROBLEMS  
VOLUME 1

15 April 1967



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N 68-32160

(ACCESSION NUMBER)

(THRU)

(PAGES)

(CODE)

(NASA CR OR TMX OR AD NUMBER)

(CATEGORY)

FACILITY FORM 602

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## FOREWORD

This is Volume 1 of a four-volume report; Volumes III and IV will be available after 1 October 1967. This volume reflects the results of the first six months of study effort.

The material presented herein was developed under a Company-funded effort to determine the requirements and constraints imposed on long-duration, manned space flights by the potential unreliability of spacecraft systems, and the need to take corrective action in both the design stage and during the projected mission. The study was conducted in conjunction with the Manned Planetary Flyby Missions Study NAS8-18025 in order to use the mission and systems designs developed therein.

The study was conducted during the first half of CFY 1967 by the Systems Engineering Management Division of the Space and Information Systems Division of North American Aviation, Inc., under Research Authorizations (RA) 02195-15400 and 02195-15100. Documentation of the study is contracted under NAS2-4214, by the Mission Analysis Division, NASA/OART, Ames Research Center, Moffett Field, California

The work was performed under the direction of Roy B. Carpenter, Jr., the program manager, Advanced Operations Analysis, Systems Engineering Management. Substantial contributions were made by H. L. Steverson, R. F. Wadsworth, L. K. Relyea, E. M. Murad, J. P. Goggins, J. A. Roebuck and I. Streimer.



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## I. BACKGROUND

### 1.1 INTRODUCTION

Because extended, manned, earth orbital and interplanetary flights pose a challenge to the nation's technological capability, reasonable assurance of mission success and crew safety is a basic prerequisite to these flights. It is self-evident that a need exists to create an integrated design and operational concept that will yield the desired safety assurance, and yet be feasible, economical, and acceptable to engineering and management. However, the no-failure-allowed approach to achieving mission assurance is unrealistic for the longer manned missions within the 1970-1980 period. Since the ability to improve mission success has already been demonstrated, the demand has grown for a long-mission-duration manned spacecraft design that will make maximum use of both man and machine. Thus it is evident that the concept should include man as a maintenance expert, a trouble anticipating sensor, a backup operator, a backup computer, and perhaps other functions as yet undefined. In sum, such an approach is embodied in the availability concept.

This concept, originally applied to ground electronic systems, has been developed under a prior study for application to manned space missions, under a design and analysis technique deriving an optimum man-machine-mission relationship. It assures the required operational availability of spacecraft system functions while remaining within the constraints imposed by the crew, equipment, and mission commitments. The concept requires that the crew shall recognize degraded performance, isolate its cause, and perform the required maintenance action.

The maintenance aspect of the concept has been the subject of several former studies conducted at S&ID which indicate that man, given adequate preparation, can probably perform the required activities. Indications are that the maintenance workload would not exert a pronounced influence on crew utilization and it appears that the maintenance requirements can be identified with reasonable accuracy. Uncertainty relating to failure rates imposes a small weight penalty in increased spares, and its effect appears to be well below that of the uncertainty in propellant requirements.

The Apollo spacecraft or its subsystems were not specifically designed to facilitate in-flight maintenance. But they do represent what may be considered the 1970 technology. Apollo subsystems were used as the study



baseline, as modified by the flyby mission requirement. Commonality of functions of primary, or crew-sensitive spacecraft systems\* for diverse missions (planetary, lunar, or orbital) makes it reasonable to assume that the system components would be similar, if not interchangeable, for a given time period such as the 1970's or 1980's. In addition, the development problems associated with new designs for each new mission make such a philosophy unattractive because of both cost and risk.

The extended-mission Apollo represents the best contemporary source of detailed system design data upon which to base the proposed study. These data, in conjunction with the configurations established in the Mars/Venus Flyby Study (Ref. 1.1), provide the basis for the mission and systems for the planetary mission module. The planetary, lunar landing modules and earth recovery modules are expected to use the same system components, but the need for maintainability is less crucial because of the relatively short duty cycles. By application of the same reasoning, the Apollo Extension Systems (Ref. 1.2) form a realistic base for the extended earth-orbital and lunar missions.

The study was conducted in three phases. Phase I identifies the maintenance problem in terms of expected requirements and constraints; Phase II will include design sensitivity, and the required trade-off analysis; and Phase III will analyze the effects on spacecraft design and establish a mission system description as it applies to the reliability and crew safety problems.

## 1.2 REQUIREMENT FOR MAINTENANCE

S&ID has studied for several years reliability problems associated with extended manned space travel—a portion of the work being accomplished under NASA contracts (Refs. 1.1, 1.2, 1.3 and 1.4), and in addition, S&ID, through company-sponsored studies, has continued these efforts. These, in conjunction with the Apollo Program and its extension studies, have provided for the time period of interest a wealth of data on the reliability/crew safety aspects of manned spacecraft.

As shown in Figure 1.1, the data indicates that, during the next decade, it will be impractical to attempt design of a spacecraft for maintenance-free operations for missions in excess of about 45 days. The practical mission limits for a non-maintainable design for a manned spacecraft probably vary between 30 to 45 days, depending on the mission profile and objectives. As missions are extended in duration and the abort profiles become more complex and time consuming, equipment failure becomes virtually certain. Further, a point is reached where adding redundancy no longer compensates

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\*A system wherein loss of function would be deleterious to the crew.

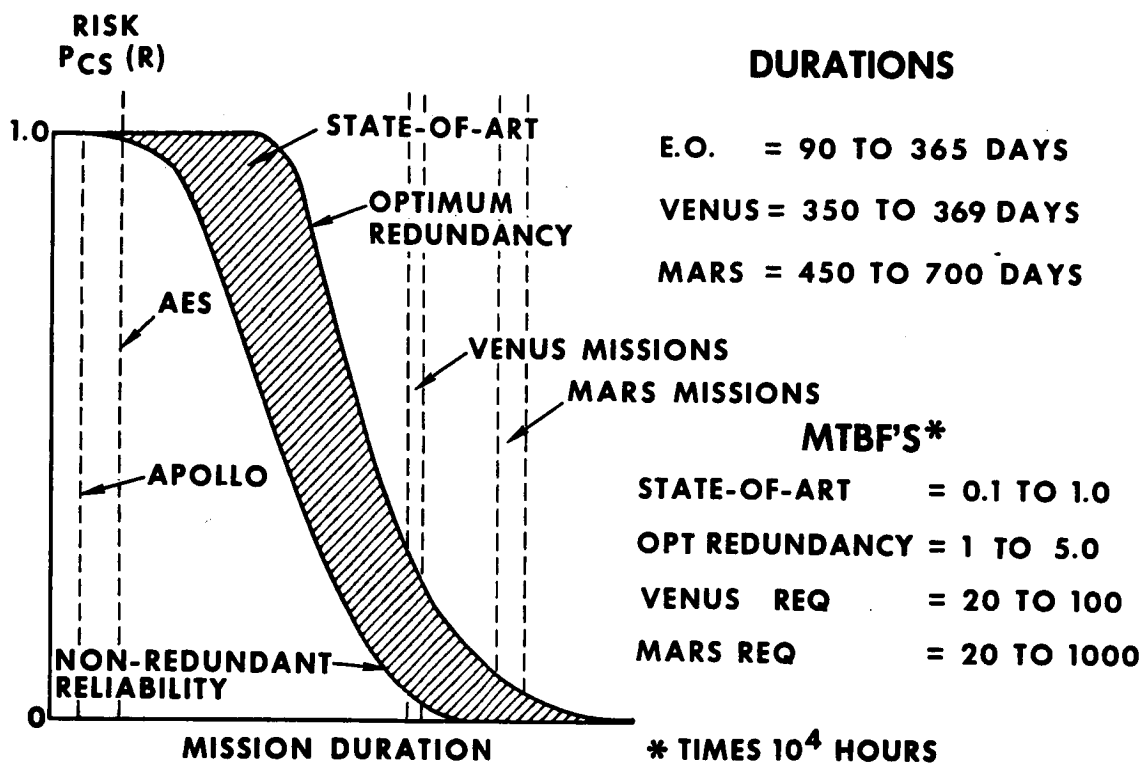


Figure 1.1. The Mission Extension Problem

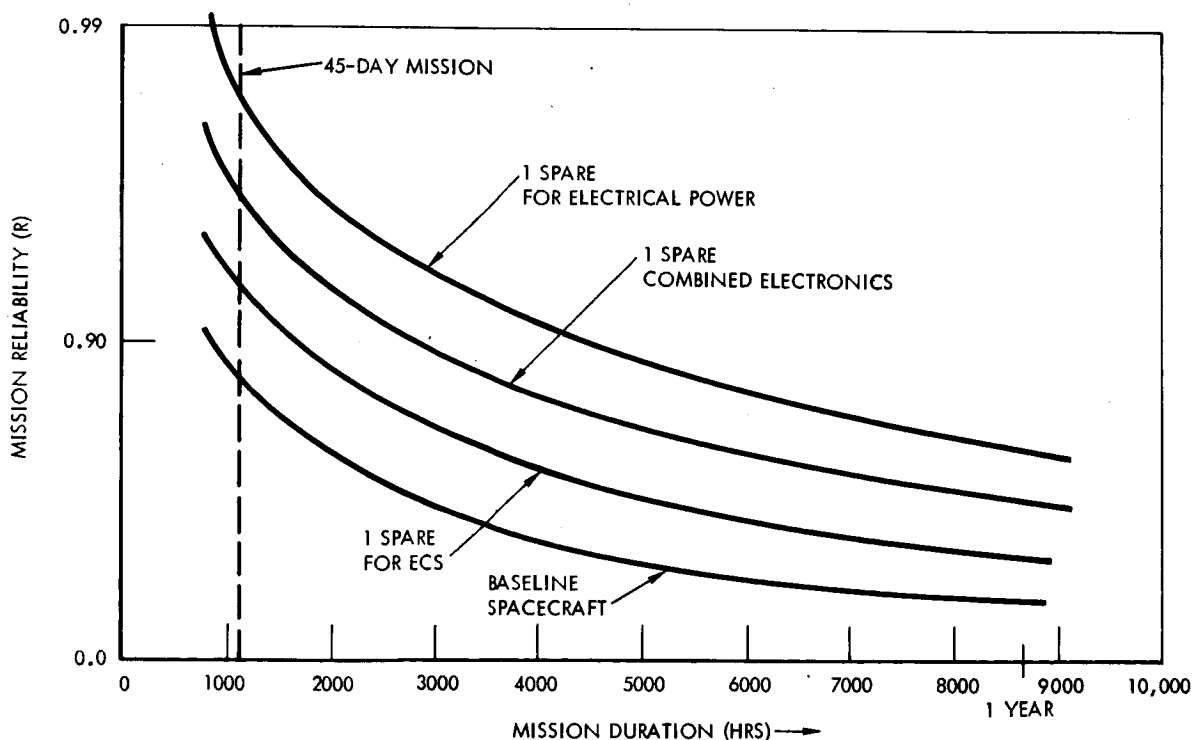


Figure 1.2. Spacecraft Probability of Mission Success as a Function of Mission Duration and Simple Sparring



for potential failures, but rather adds to the failure hazard. This technology limit is created by the need to include switching devices, performance monitors and voting circuits as well as the connections to the system in the function reliability assessment. The practical limit seems to be between one and two additional components in simple redundancy. Maintenance beyond this point must be considered as a more reasonable alternative.

This study is concerned with mission durations measured in years. The approximate mission reliability requirements in terms of mean time before failure (MTBF) without maintenance are:

1. Venus Flyby      20 to 100 x 10<sup>4</sup> hours
2. Mars Flyby      20 to 1000 x 10<sup>4</sup> hours
3. Mars Landing    20 to 600 x 10<sup>4</sup> hours

The state-of-the-art capability has been shown (Ref. 1.1) to be as follows:

1. Without redundancy, approximately 0.1 to 1.0 x 10<sup>4</sup> hours
2. With optimum redundancy, approximately 1.0 to 5.0 x 10<sup>4</sup> hours.

If no failures are to be tolerated these estimates obviously fall far short of the expressed requirements, literally by orders of magnitude. Further, this same study indicated that, on the average, system MTBF can be improved by factors of between 5 and 10 over any decade. The effect of applying those systems to the longer space missions are demonstrated in Figure 1.1, where a state-of-art spacecraft is applied to the missions of interest without programmed maintenance.

Clearly, the longer missions must be prepared for failures. At this point, some study results have suggested that a possible alternative would include abort, spacecraft replacement, escape capsules, or rescue. But, S&ID studies have shown that, for the planetary and most lunar area missions, none of these will assure crew survival (Refs. 1.1, and 1.5). These are reviewed later in this study report and shown to be ineffective. Conversely, provision for even the simplest of maintenance provides very startling results in terms of increases in mission success and crew survival.

For example, the typical state-of-the-art spacecraft MTBF is estimated to be about 2800 hours. Now, assume that the mission duration is about 400 hours. Without any repair the probability of mission success (no failure) is only:



$$R = e^{-\frac{400}{2800}}$$

$$= 0.870$$

By making provisions for just one repair, reliability, R, is increased to at least:

0.933 at the lower boundry

or

0.990 at the upper boundary,

depending on the assumed distribution, the method of calculation, and how the provision for the repair was implemented. Adding provisions for one more repair (in the critical system), or a total of two, raises the lower boundary estimate for mission reliability, R, to more than 0.99. These data indicate that providing for maintenance for the longer missions possesses a very attractive potential for increasing probability of mission success. Further, this is one case where the mathematics present a very conservative picture of the actual gains derived. This effect is dramatically shown in Figure 1.2 which presents an estimate of mission reliability as a function of mission duration and spares application. The lower curve, the baseline spacecraft, is representative of the latest AES reliability estimates derived from Apollo data. The curves above the base spacecraft represent the effects of adding one spare to the previous state for replacement of a critical component in the listed system. Note that a marked effect on mission reliability is achieved by adding only three spares.

The effects of sparing on the probability of safe return are not as dramatic for many earth orbital missions because of the abort capability. However, for the extended lunar and planetary missions, the results of M&R actions are essentially the same as shown for mission success. This condition prevails because of the abort criteria applied to the Apollo missions and the very high initial probability of crew survival. But, as the missions are extended in distance away from the earth, the abort time delay exercises an increasingly more significant influence on the survival characteristics of a nonmaintainable manned spacecraft design.

From the foregoing, it is evident that a few spares and the associated M&R actions have a profound effect on both missions safety and success. The next most natural question is how many M&R actions are required to achieve a reasonable level of safety for the missions of interest? This is



actually a major subject of the study. However, the number was estimated in Ref. 1.1. Using the prior estimated spacecraft MTBF, the number of M&R actions to be prepared for can be calculated as a function of the risk to be taken in meeting the need.

Since we are dealing with a statistical problem, the answer will be statistical in nature; if the mission risk or reliability is to be 0.95, this means that a 5-percent risk of not having the required part for a failure is acceptable. In Figure 1.3, the number of M&R actions to be prepared for has been estimated for the state-of-the-art spacecraft, as a function of mission duration and acceptable risk. With only a 5-percent risk, less than 85 M&R actions need be expected. Thus, this adds up to less than one activity per week of mission time, a very modest workload indeed. A basic intention of this study is to identify which specific components in a typical contemporary spacecraft will require attention and in what form.

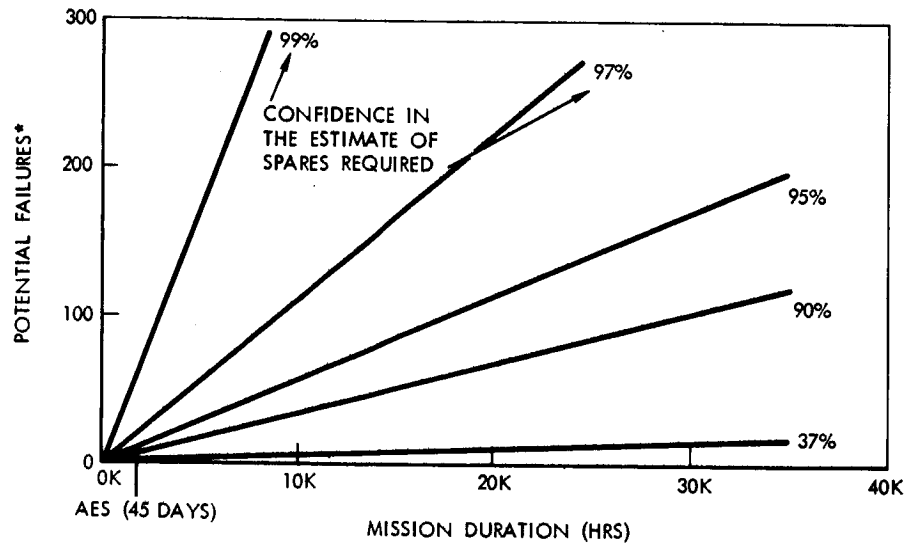
In the past, maintenance has often been accomplished despite the design, rather than as a result of designing for maintenance. But now, crew safety, political, and cost aspects make it mandatory to take advantage of the potential reduction in risk inherent in the maintainable design. Since in-flight maintenance is dependent, to a large extent, on an amenable system configuration and hardware, full consideration must be given to the design for maintenance at the onset of the program. Thus, in-flight maintenance becomes an integral part of system design, and requires coordination with the concerned disciplines such as engineering, reliability, human factors, maintainability, logistics, and operations analysis. This systems approach has led to the availability concept detailed in Ref. 1.6.

### 1.3 THE AVAILABILITY CONCEPT

#### 1.3.1 Description

While it is essential that the probability of failures on long space missions be recognized and accepted, it does not necessarily follow that they need to be catastrophic. For example, space flights to date have encountered failures with no crew loss. This factor led S&ID to the development and application of the availability concept to the manned planetary missions in which abort was impractical.

The availability concept is a design or mission analysis technique that facilitates the determination of an optimum man-machine relationship. Mission effectiveness is maximized through establishment of a safe and reasonable balance between system and mission performance, operation control, reliability, and maintainability. Application of this concept can result in a design that provides maximum operational availability of the



\*AT THE INDICATED PERCENTILE, USING STATE OF ART ESTIMATES AND ASSUMING AN EXPONENTIALLY DISTRIBUTED FAILURE HAZARD.

Figure 1.3. Expected Number of Failures for the Vehicle System of a Contemporary Spacecraft as a Function of Mission Duration

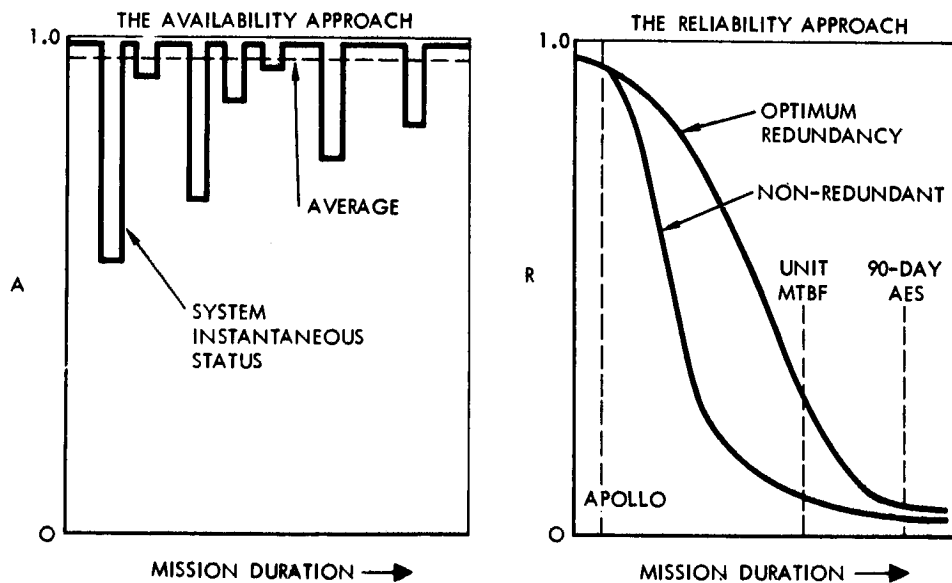


Figure 1.4. Reliability Concept Versus Availability Concept for Long Missions



system functions within the constraints imposed by crew capabilities, mission requirements, and existing technology, thus maximizing the potential mission success.

Application of this design concept virtually eliminates the usual risk patterns associated with the reliability design and replaces it with controllable parameters, not limited to the state-of-the-art and any given time period. The resultant risk pattern is demonstrated in Figure 1.4 where the two design concepts are contrasted. In both concepts, the ordinates R, reliability, and A, availability, express the probability of mission success. Ordinate A is independent of time and is dependent only on maintenance and the ability to meet the downtime constraints. In the situations portrayed, the mission time approaches or is in excess of the system MTBF. The longer the mission duration, the more probable failure becomes. The nonmaintainable approach is destined to near-certain failure for the longer missions. Contrasting this with the availability design, there is no appreciable change found in the probability of mission success within the mission duration indicated. Individual system functions may be down for short periods, as demonstrated subsequently within this report.

### 1.3.2 Application

The availability concept as an analytical/design tool is presented in logic form in Figure 1.5. Before application, the system reliability/safety logic has been prepared in simplified form, with the weak links\* identified in order of weakness. Then, starting with the weakest, the analytical logic is applied to each block ( $x.x.x_i$ ) in sequence, until the safety/success goal is achieved, or surpassed. A detailed explanation is given in Refs. 1.1, and 1.6 along with sample applications.

The key to the analysis is to determine what level of assembly to work on, and the most effective/safe corrective action required to reduce a failure hazard. Each weak link must be treated as an individual case; the most probable failure modes should be isolated, and then appropriate action determined. Computers can only be used in a bookkeeping role because they are ineffective at this point.

Selection criteria to be considered include accessibility, least number of spares per weak link, least number and complexity of repairs per weak link, ease of maintenance, least redundancy and simple monitoring and diagnosis. Redundancy is a less desirable alternative because interchangeability of spares is reduced. The process of selecting the level of assembly for maintenance can have a profound influence on the resulting mission requirements. To assure maximum mission efficiency, it is necessary to

\*Weak Links are the more failure prone components of a system.

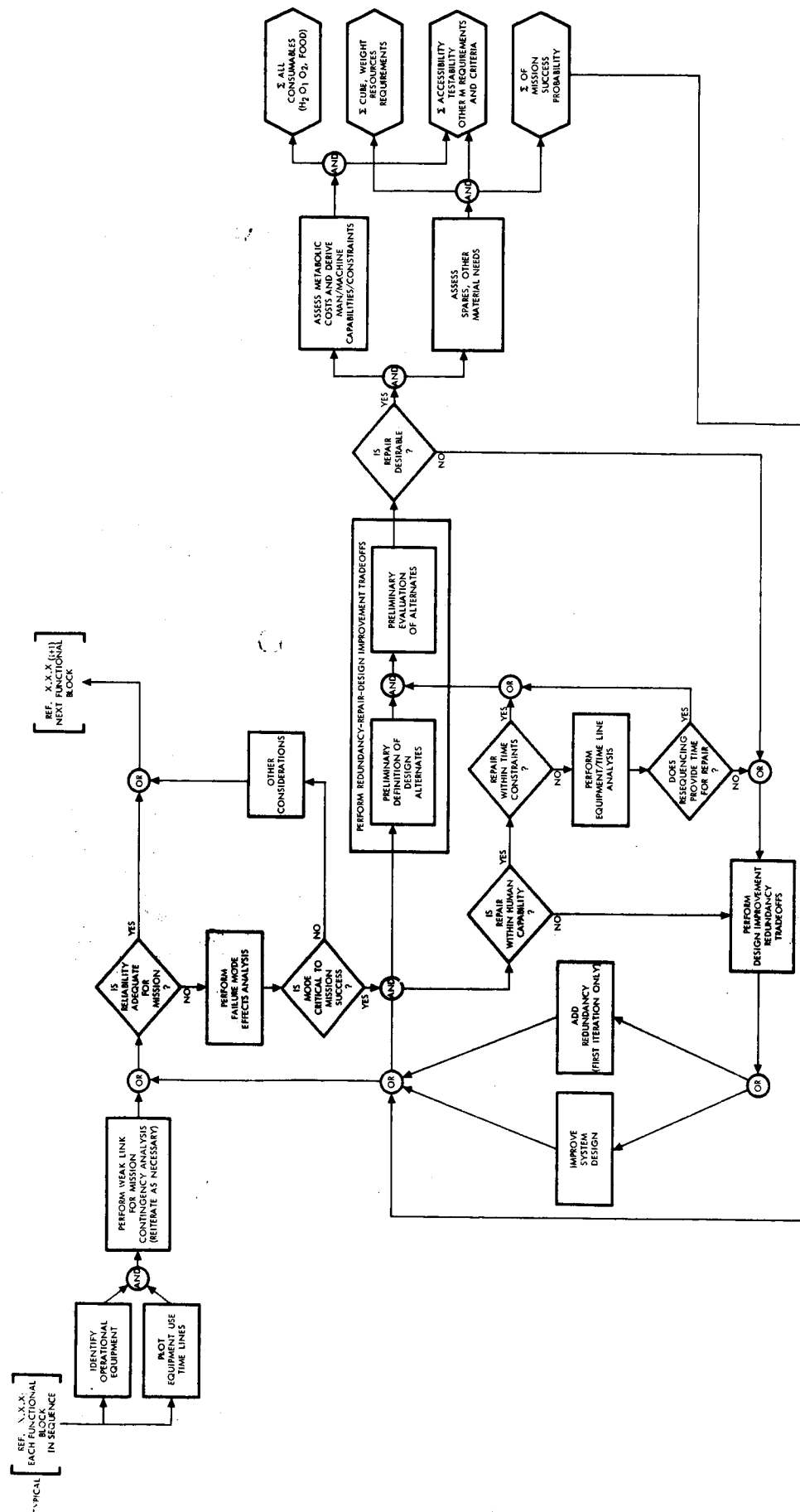


Figure 1.5. Availability - Analysis Logic

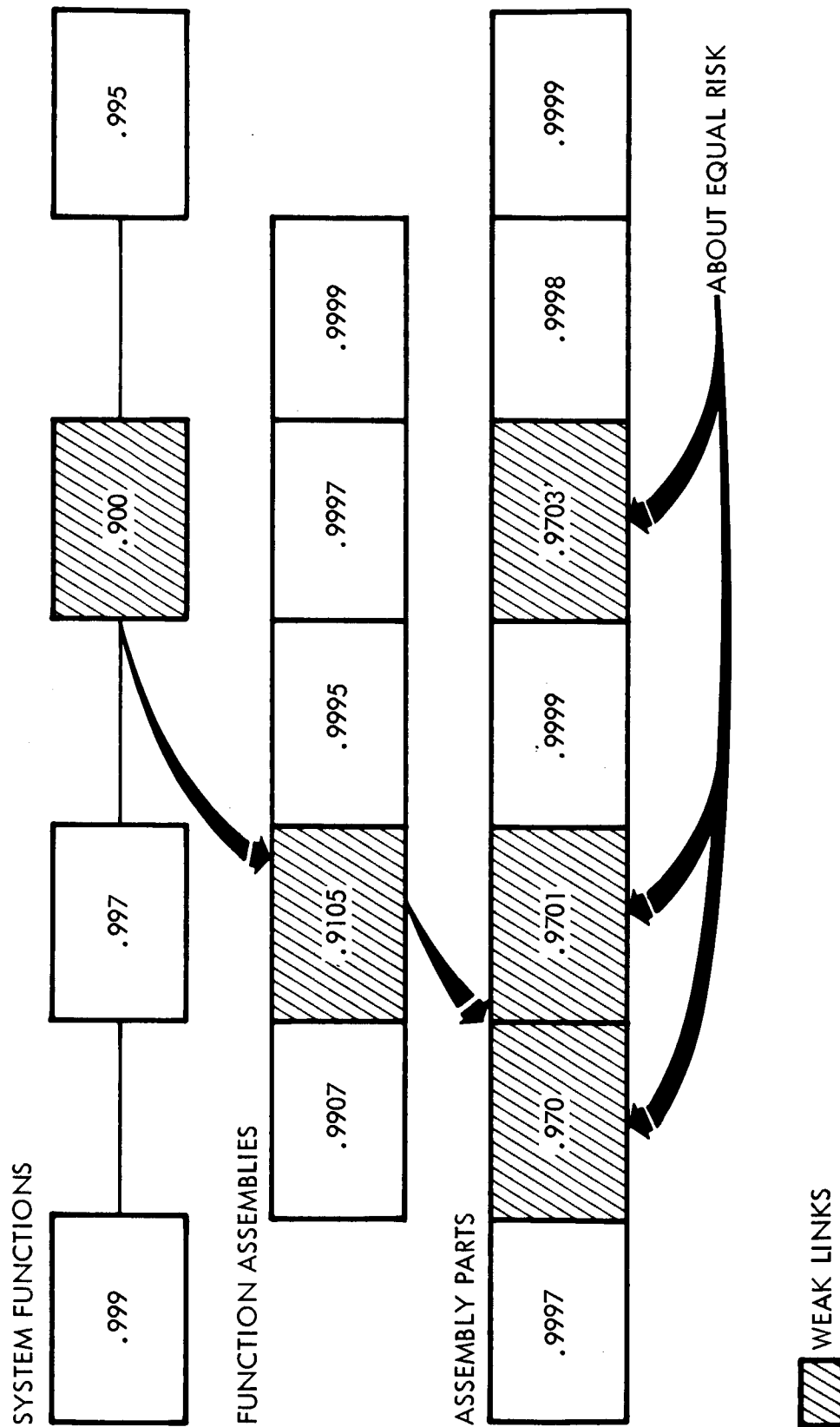


Figure 1.6. Weak-Link Isolation and Maintenance Level Determination



determine how failure risk is distributed within the specific system, functions, assemblies, or parts. From the example of Figure 1.6, note that one function displays a low reliability only at the system level. Therefore, one assembly at this level still contributes most of the failure hazard. However, at the part level, three assemblies exhibit equal risk of failure. The one assembly which contains all those parts could be spared, or the three parts could be spared. Thus, the choice is an obvious one if the spare assembly is small and lightweight, easy to diagnose, and easy to replace.

### 1.3.3 Application Requirements and Constraints

To apply these concepts on manned spacecraft and mission design some requirements and constraints should be imposed. And a basic objective encompassing this study is to identify and bound them and demonstrate further the feasibility of long duration manned space ventures. Therefore, these will be discussed in detail throughout the report.

REQUIREMENTS are imposed by the need to perform the M&R actions. Some examples of these are:

1. A spare parts complement - provided to meet the repair and replacement needs to a risk level compatible with the mission goals.
2. A Performance Monitor - designed to facilitate identification of system malfunctions where they are most likely to occur.
3. Diagnostic Equipment - designed to isolate malfunctions in the potentially weak system functions.
4. Tools - selected to aid the crewman in making the M&R action within the constraints imposed on the crewman and the mission environment.
5. Backup Support - support systems and/or redundant systems necessary to assure performance of critical functions during the M&R cycles.
6. Maintainable systems - designed to facilitate the maintenance and/or repair of those functions identified as potential weak links.



CONSTRAINTS are imposed on the mission and systems designs which establish a boundary on some of the design parameters. Some examples of these are:

1. Mission System Downtime Constraints - are restrictions imposed on the length of time the total system functions. For example, the velocity correction can be out of service. These are imposed by mission profile and spacecraft attitude requirements.
2. Crew System Downtime Constraints - are restrictions imposed on the length of time a crew system function can be inoperative at any one time. For example, the CO<sub>2</sub> reduction. These are imposed by the need to provide the crew with a habitable environment.
3. Workman Constraints - are imposed by the physical limitations of man performing a given activity within a specific environment.

#### 1.4 BACKGROUND REFERENCE

| <u>Reference No.</u> | <u>Title and Source</u>                                                                                                                                                             |
|----------------------|-------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|
| 1.1                  | <u>Manual Mars and/or Venus Flyby Study,</u><br>NAS9-3499, SID 65-761-5, June, 1965.                                                                                                |
| 1.2                  | <u>Preliminary Definition Phase, Apollo</u><br><u>Extension Systems,</u> SID 65-1547,<br>16 December 1965.                                                                          |
| 1.3                  | <u>Manual Mars Mission Module Study,</u> NAA/<br>S&ID SID 64-1, January, 1964.                                                                                                      |
| 1.4                  | <u>Radio Isotope Dynamic Electrical Power</u><br><u>System Study for Manned Mars/Venus</u><br><u>Mission,</u> NAS9-3520, NAA/AI,<br>19 February 1965.                               |
| 1.5                  | <u>Space Rescue and Escape,</u> SID 66-1341,<br>NAA/S&ID, September, 1966.                                                                                                          |
| 1.6                  | <u>A Proposal for Availability Extension Studies</u><br><u>as a Means of Extending the Useful Life of the</u><br><u>Apollo Spacecraft,</u> SID 66-10, NAA/S&ID,<br>17 January 1966. |



## II. MISSION REQUIREMENTS

### 2.1 MISSION OBJECTIVES AND CONSIDERATIONS

This study was conducted in close coordination with the NAA/S&ID Study of Manned Planetary Flyby Missions conducted under contract NAS8-18025 to NASA/MSFC, Ref. 1.1. Since the objective of this study is the demonstration of the feasibility of safe, long-duration manned space flight, it seemed reasonable to apply it to a mission of primary interest. For that reason, much of the data developed under the NASA contract provided guidance for this study.

Mission requirements and the associated design considerations evolve from the desire to achieve a given objective. For purposes of this study, the objective is to identify the factors which affect the probability of safe return of a manned mission designed to fly by the planet Mars; and to identify the combination of mission and vehicle design concept that will assure the highest probability of safe return.

This study is concerned with the safety and success aspects of the mission, to the exclusion of systems design. However, it is impractical to attempt to separate these considerations completely. To avoid a design study per se, the design alternatives identified in the referenced NASA study will be used as a baseline, and other design alternatives will be considered only when necessary to provide the desired mission success assurance.

Mission success and safety is dependent on both the mission profile and the mission systems. The more complex the mission profile and the individual phases, the lower the chances of success. Also, the longer the mission, the more difficult the mission assurance problem becomes. This may be seen from the basic reliability, R, equation:

$$R = e^{-\left[\lambda_1 t_1 + \lambda_2 t_2 + \dots + \lambda_n t_n\right]}$$

where:

t = phase duration

$\lambda$  = (f) phase complexity (because the support systems are more complex)

n = mission phase



This becomes even more obvious when a comparison is made between the landing mission and the flyby missions as shown in Figure 2.1. For the longer missions note that at least 38 discrete and major operations exist, many of which are time critical. Yet, for the flyby missions, there are only 16, most of which are not time critical. This relationship will be explored in detail later in the report.

Several potential missions were identified in the referenced study as potential candidates for the time period of interest. They vary considerably in duration and character, as a function of both planet objective and time of departure. These are illustrated in Figure 2.2. The missions do display many features in common, specifically in the profile functions. For that reason, selecting a representative baseline mission for the remainder of this study will not compromise its usefulness, particularly if the baseline is nearly a "worse case."

## 2.2 THE BASELINE MISSIONS AND SPACECRAFT

Since mission success and safety are sensitive to the mission profile characteristics, it is essential to this study to identify a recommended profile. Further, since these objectives are also sensitive to the required functions and associated equipment, it is equally necessary to identify those functions that are essential to each mission phase or portion thereof.

For the purposes of this study, a representative manned Mars flyby mission was selected as the baseline mission, as a result of consultation with the NASA study team. The following are some of its basic characteristics, which will be detailed in functional flow analysis to follow:

|                    |                          |
|--------------------|--------------------------|
| Departure date     | 1977                     |
| Total duration     | 690 days or 16,560 hours |
| Outbound leg       | 140 days or 3360 hours   |
| Return leg         | 530 days or 12,720 hours |
| Planetary vicinity | 20 days or 480 hours     |

For additional details see the NAS8-18025 report.

The baseline spacecraft comes from the same source and is shown in Figure 2.3. The subsystems are Apollo derivatives, where applicable. Where this was not practical the components were used to make up the new subsystems. The details are given later in this report. The baseline spacecraft layout is as shown in Figure 2.4 and was derived from Ref. 1.1, 1.2 and 2.1.

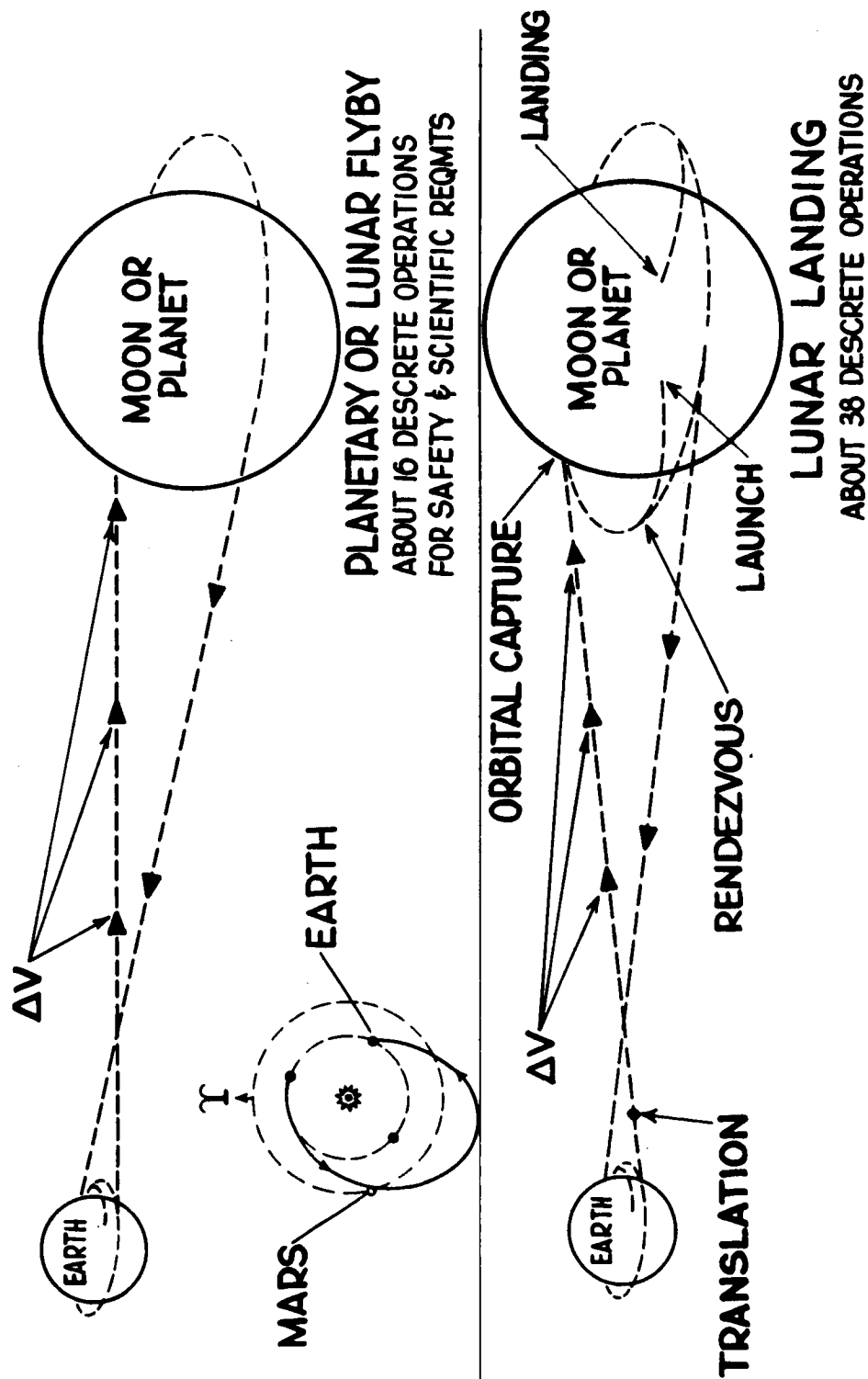
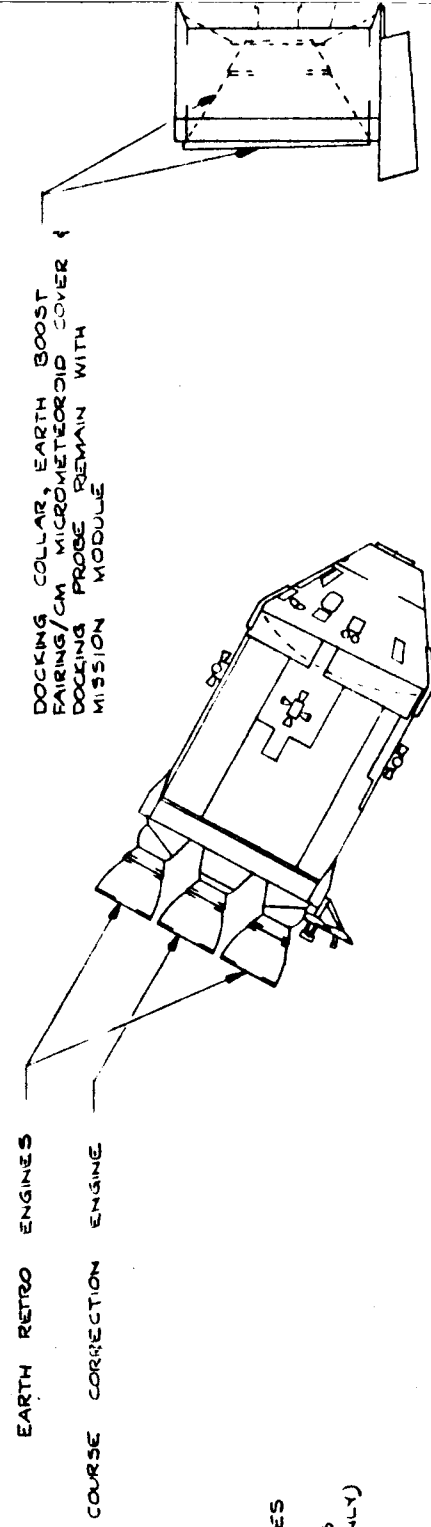
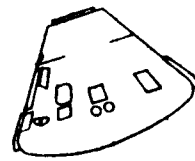
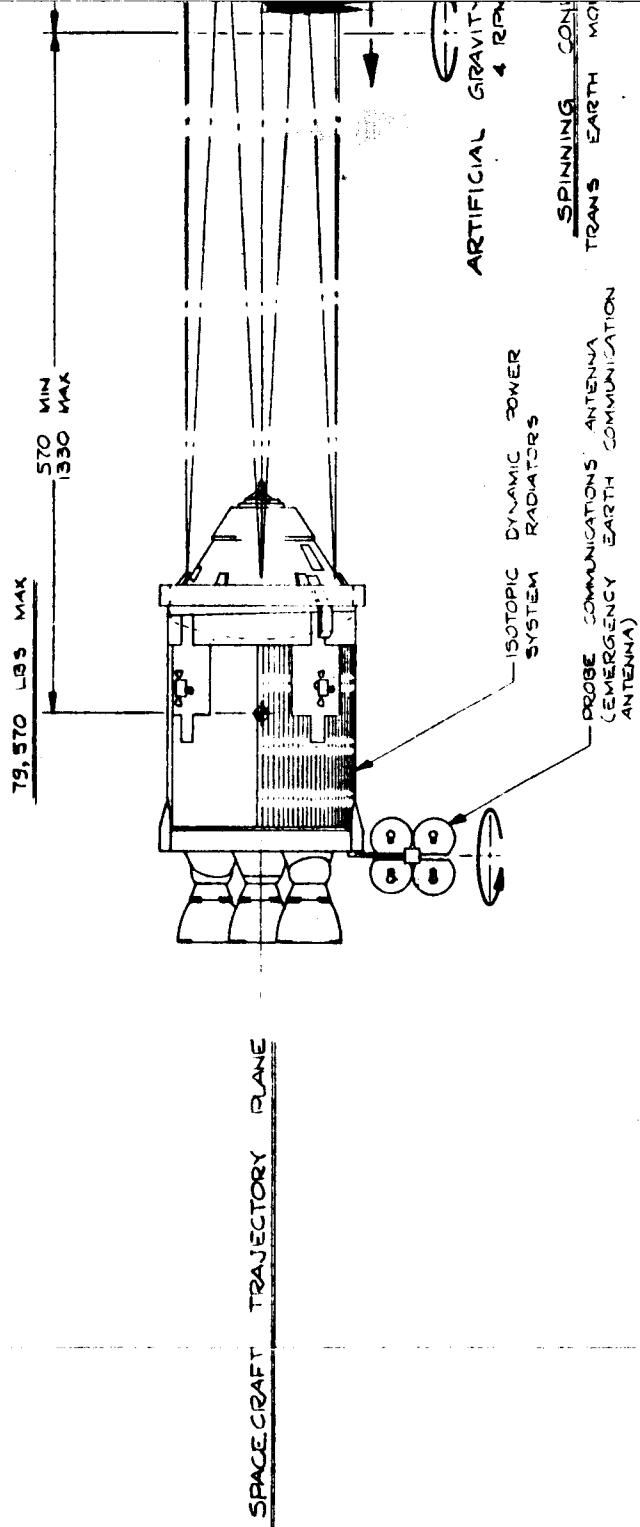


Figure 2.1. Comparison Between Planetary Lunar Flyby and Lunar Landing



| TARGET PLANET(S)      | 1975         | 1976        | 1977 | 1978         | 1979 | 1980 | 1981 | 1982 | 1983 |
|-----------------------|--------------|-------------|------|--------------|------|------|------|------|------|
| <b>VENUS</b>          |              |             |      |              |      |      |      |      |      |
| <b>MARS</b>           |              |             |      |              |      |      |      |      |      |
| <b>PERIHELION</b>     |              |             |      |              |      |      |      |      |      |
| <b>TWILIGHT FLYBY</b> |              |             |      |              |      |      |      |      |      |
| <b>DUAL PLANET</b>    |              |             |      |              |      |      |      |      |      |
| <b>POWERED</b>        |              |             |      |              |      |      |      |      |      |
| <b>UNPOWERED</b>      |              |             |      |              |      |      |      |      |      |
| <b>TRIPLE PLANET</b>  |              |             |      |              |      |      |      |      |      |
| <b>LEGEND:</b>        | <b>EARTH</b> | <b>MARS</b> |      | <b>VENUS</b> |      |      |      |      |      |

Figure 2.2. Manned Planetary Flyby Mission Opportunities



EARTH AERODYNAMIC ENTRY

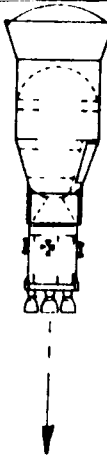
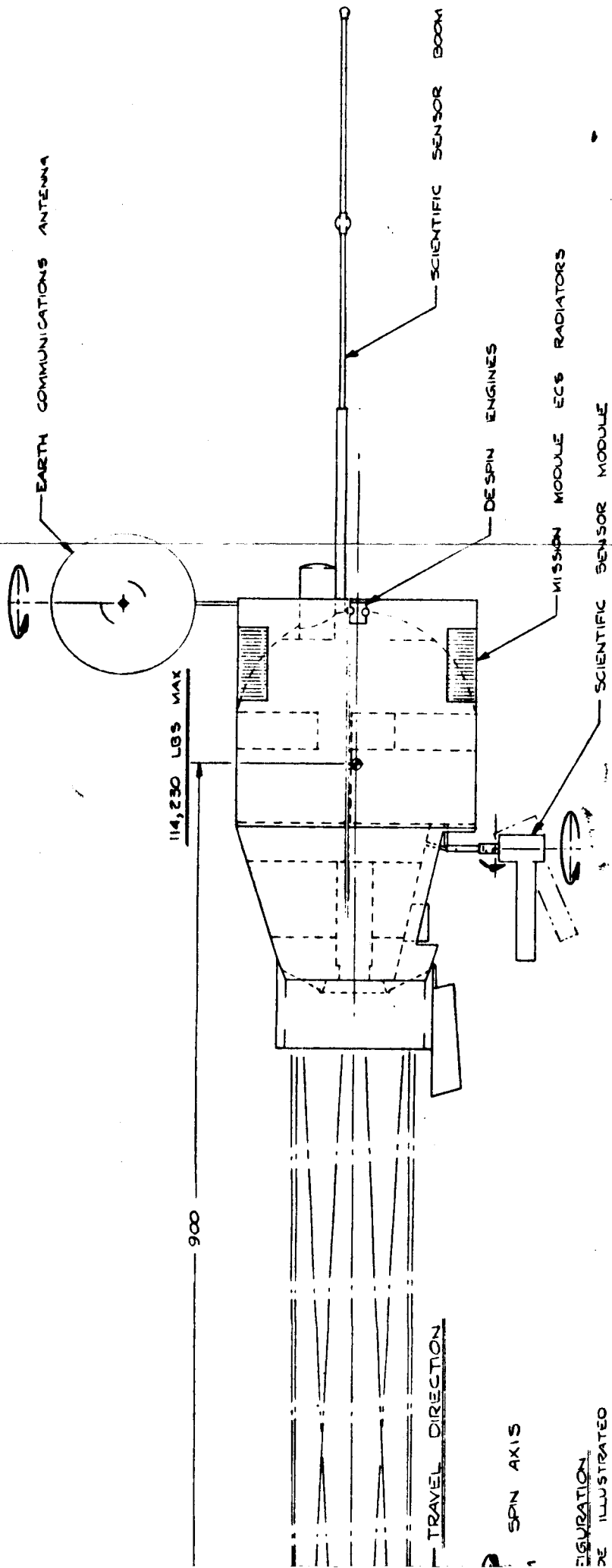
TERMINAL ENTRY SEQUENCE:

1. JETTISON AFTERBODY HEAT SHIELD
2. FIRE DROGUE CHUTE MORTARS & DEPLOY 2 DROGUE CHUTES
3. RELEASE DROGUE CHUTES
4. FIRE MAIN CHUTE MORTARS & DEPLOY 3 MAIN CHUTES
5. JETTISON ISOTOPIC POWER SOURCE (LAND LANDING ONLY)
6. IMPACT

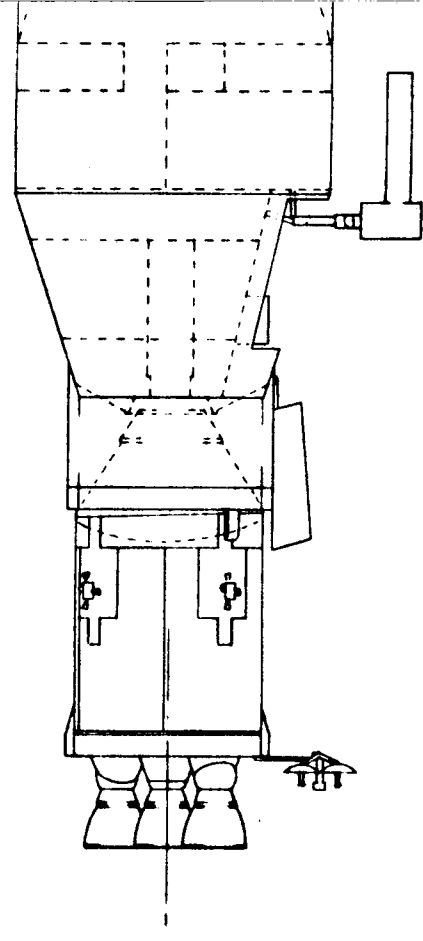
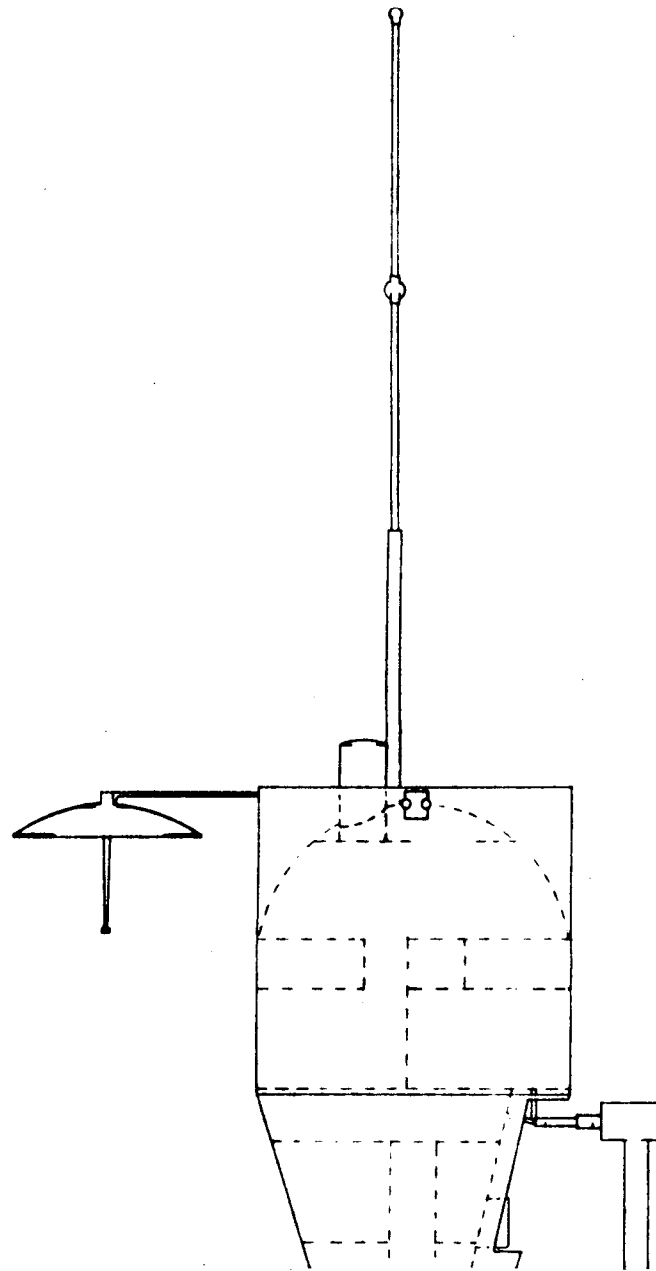
FOLDDOUT FRAME 1

EARTH RETRO MANEUVER

FOLDDOUT FRAME 2



INJECTION

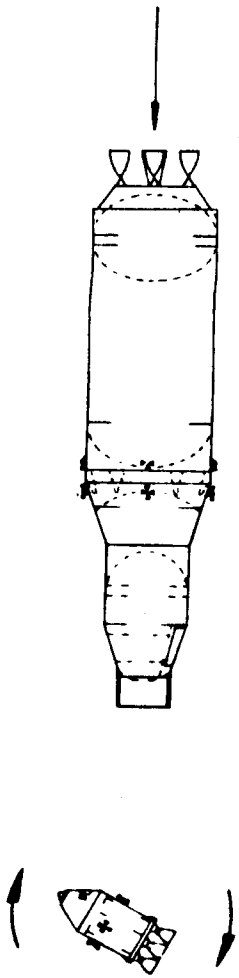


DOCKED CONFIGURATION  
TRANS WARE MODE ILLUSTRATED

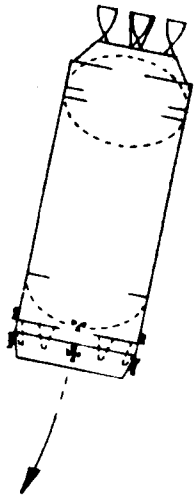
FOLDOUT FRAME

FOLDOUT FRAME 3

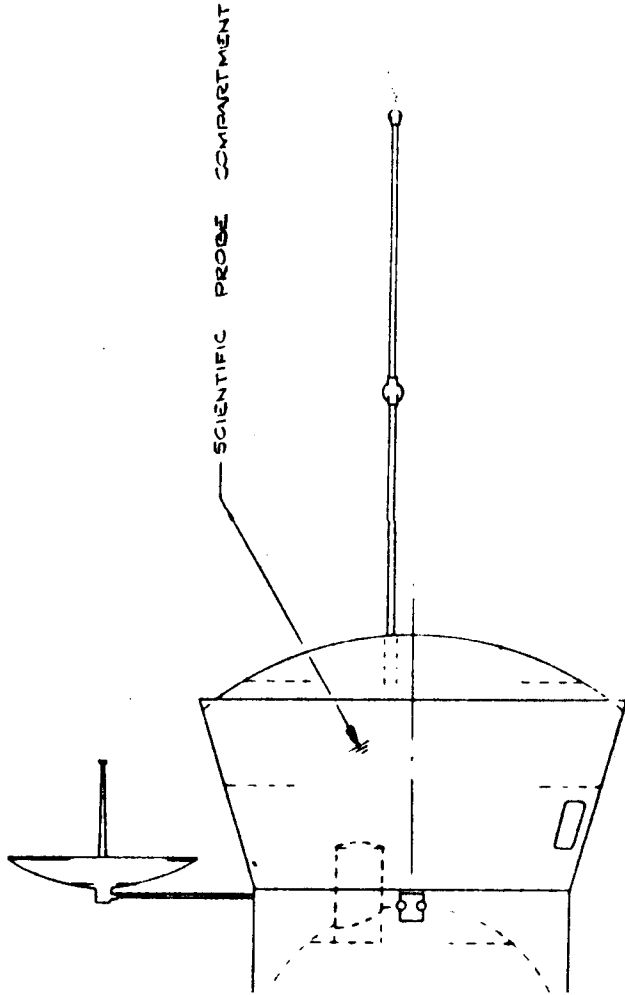
FOLDOUT FRAME



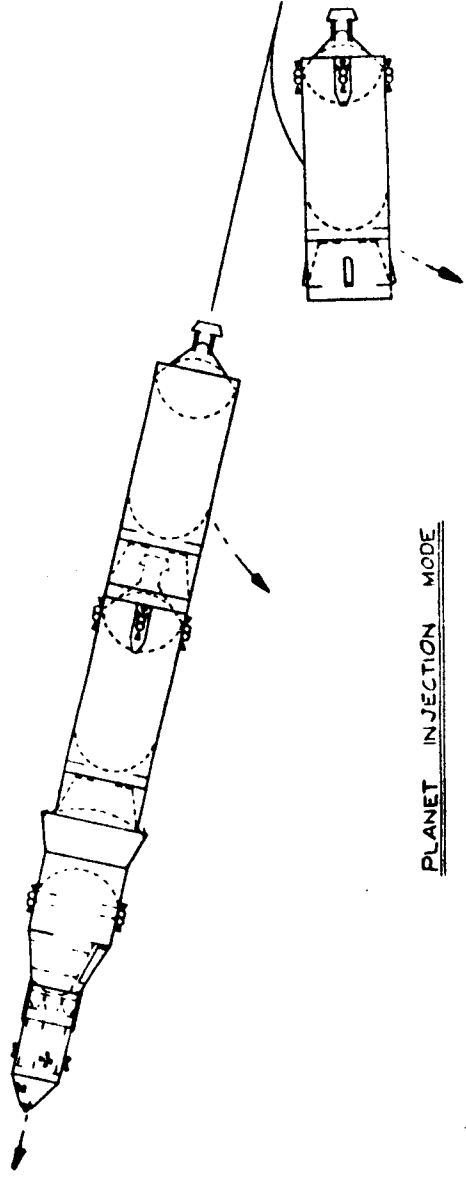
TRANSPOSITION & DOCKING



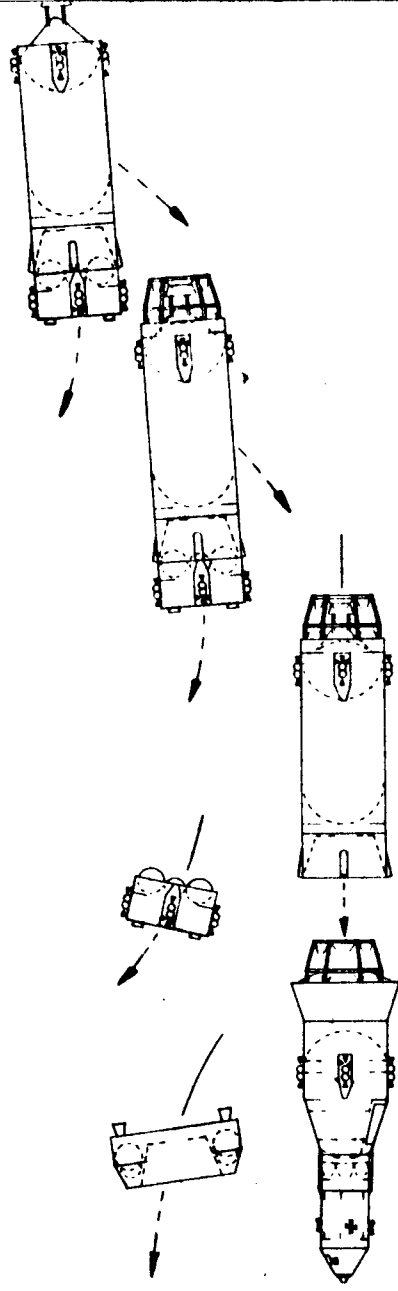
STAGE SEPARATION



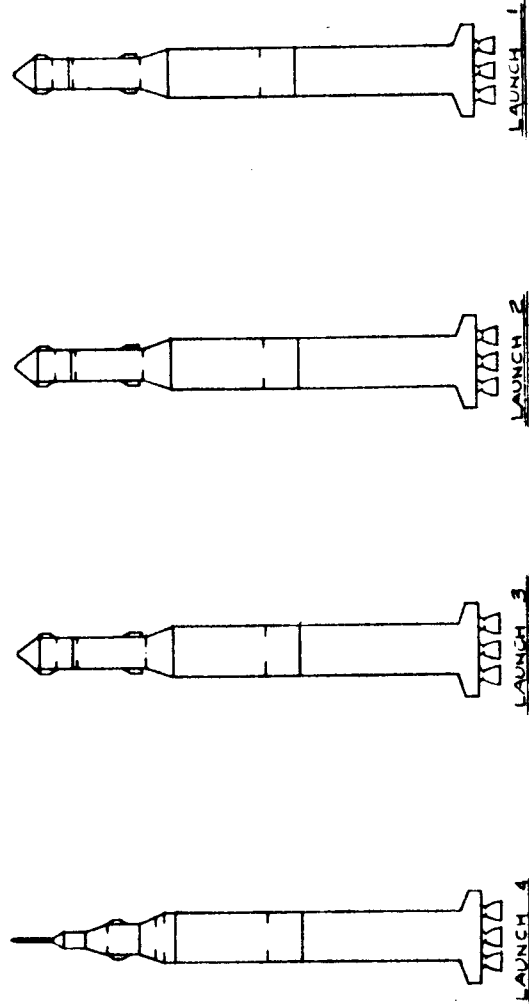
SCIENTIFIC PROBE COMPARTMENT



PLANET INJECTION MODE



EARTH ORBIT ASSY MODE



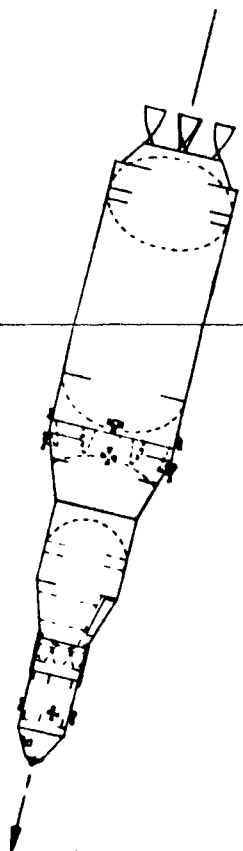
LAUNCH 4  
S-IC  
S-II  
S/C TRANSTAGE  
MANNED FLIGHT S/C

LAUNCH 3  
S-IC  
S-II  
S/C TRANSTAGE  
ELV TRANSTAGE

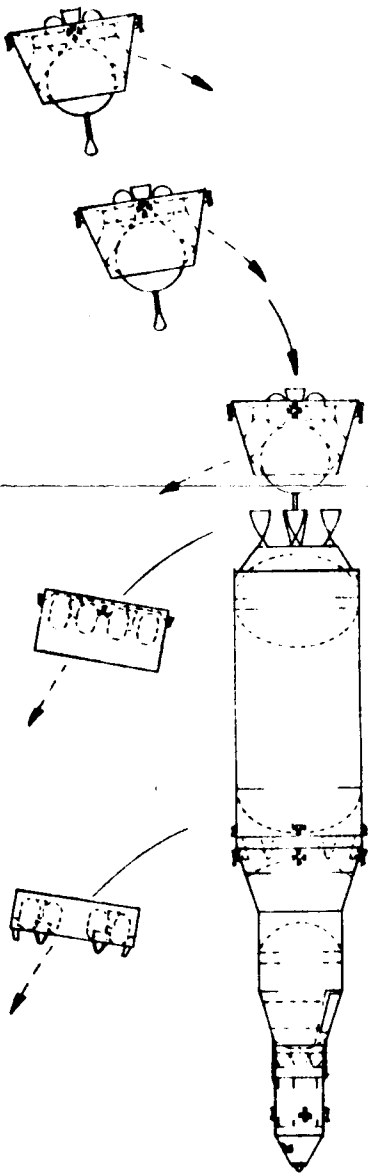
LAUNCH 2  
S-IC  
S-II  
S/C TRANSTAGE  
ELV TRANSTAGE

LAUNCH 1  
S-IC  
S-II  
S/C TRANSTAGE  
ELV TRANSTAGE

TANDEM S-IC INJECTION CONCEPT



PLANET INJECTION MODE



EARTH ORBIT ASSY & O<sub>2</sub> TRANSFER MODE

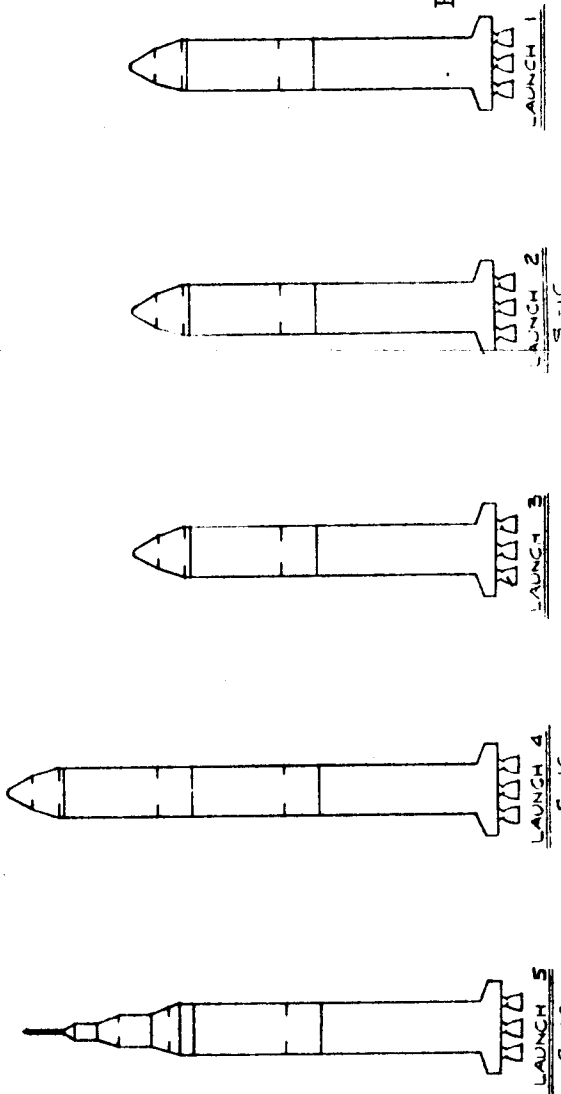


Figure 2.3. Mars/Venus Flyby Spacecraft, Baseline Concept

S/C TRANSTAGE  
MANNED FLYBY S/C

S-IIC  
S-II B TRANSTAGE

-O<sub>2</sub> TANKER & TRANSTAGE

S-IIC  
S-II

2-5, 2-6

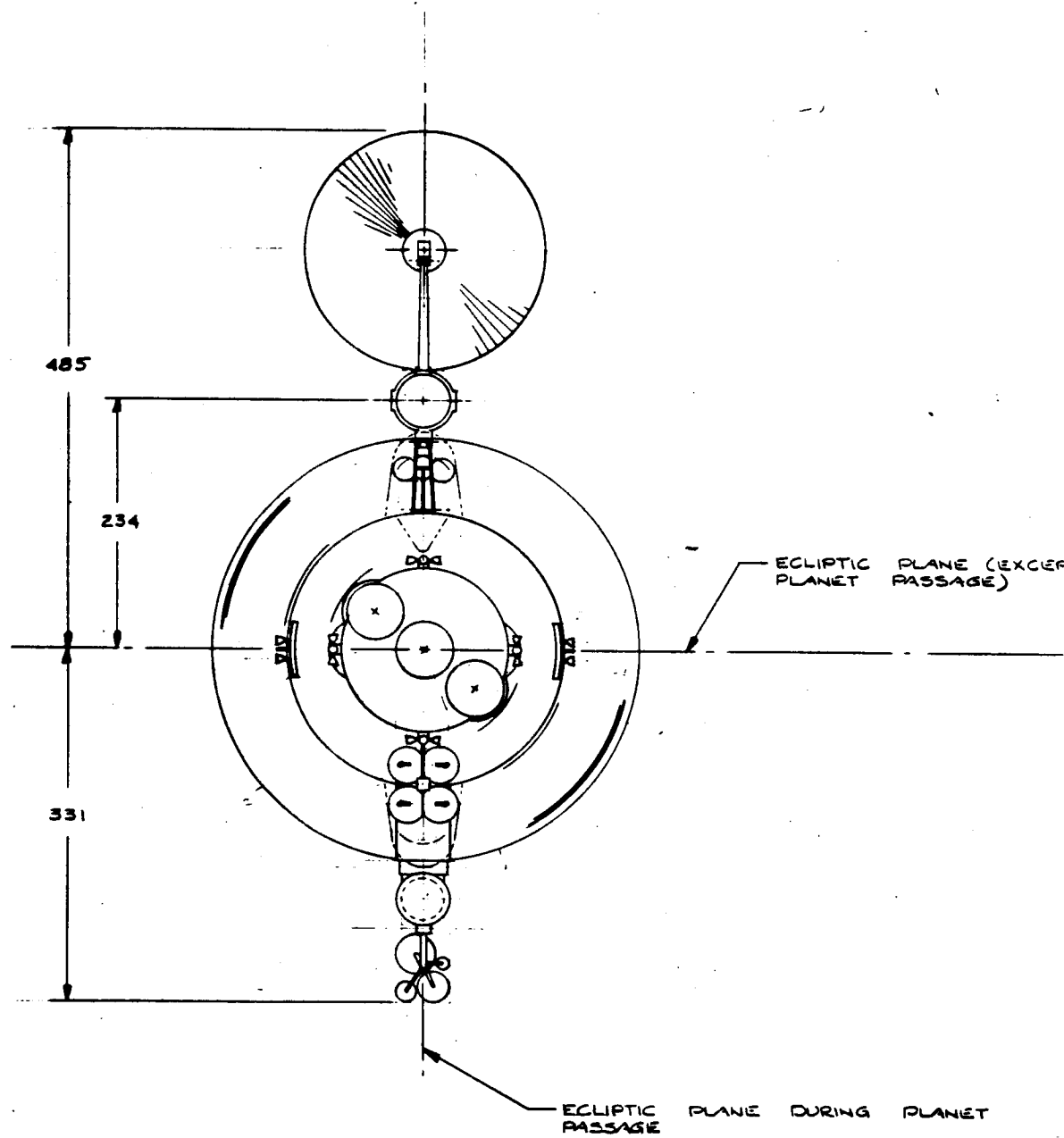
FOLDOUT FRAME

6

7

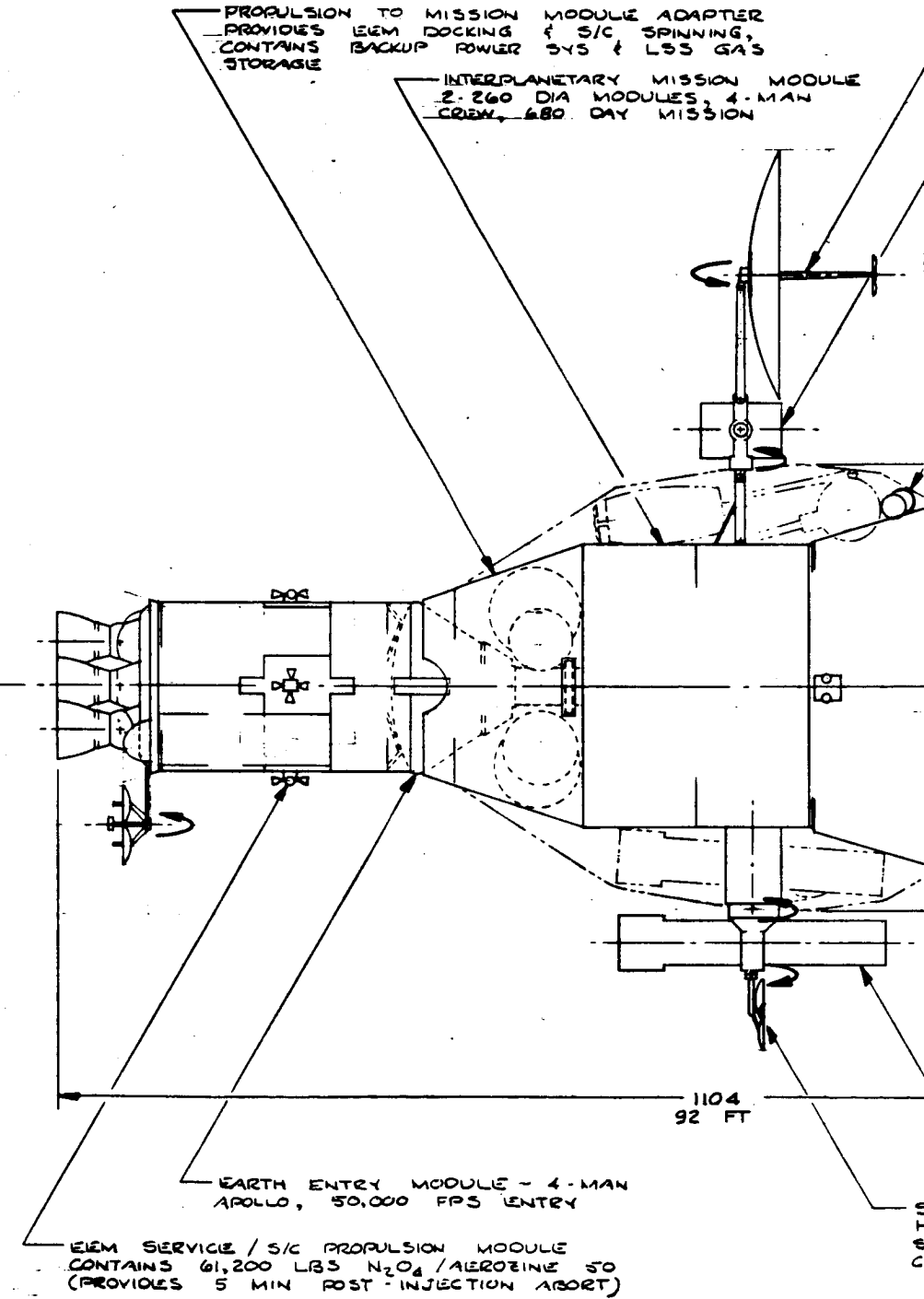
FOLDOUT FRAME

SID 67-478-1



FOLDOUT FRAME /

T DURING



PRIMARY MISSION ~ 1977 TRIPLE  
BACKUP MISSION ~ 1978 DUAL PLANET,  
SECONDARY MISSION ~ 1980 OR 1981 VEN

Figur

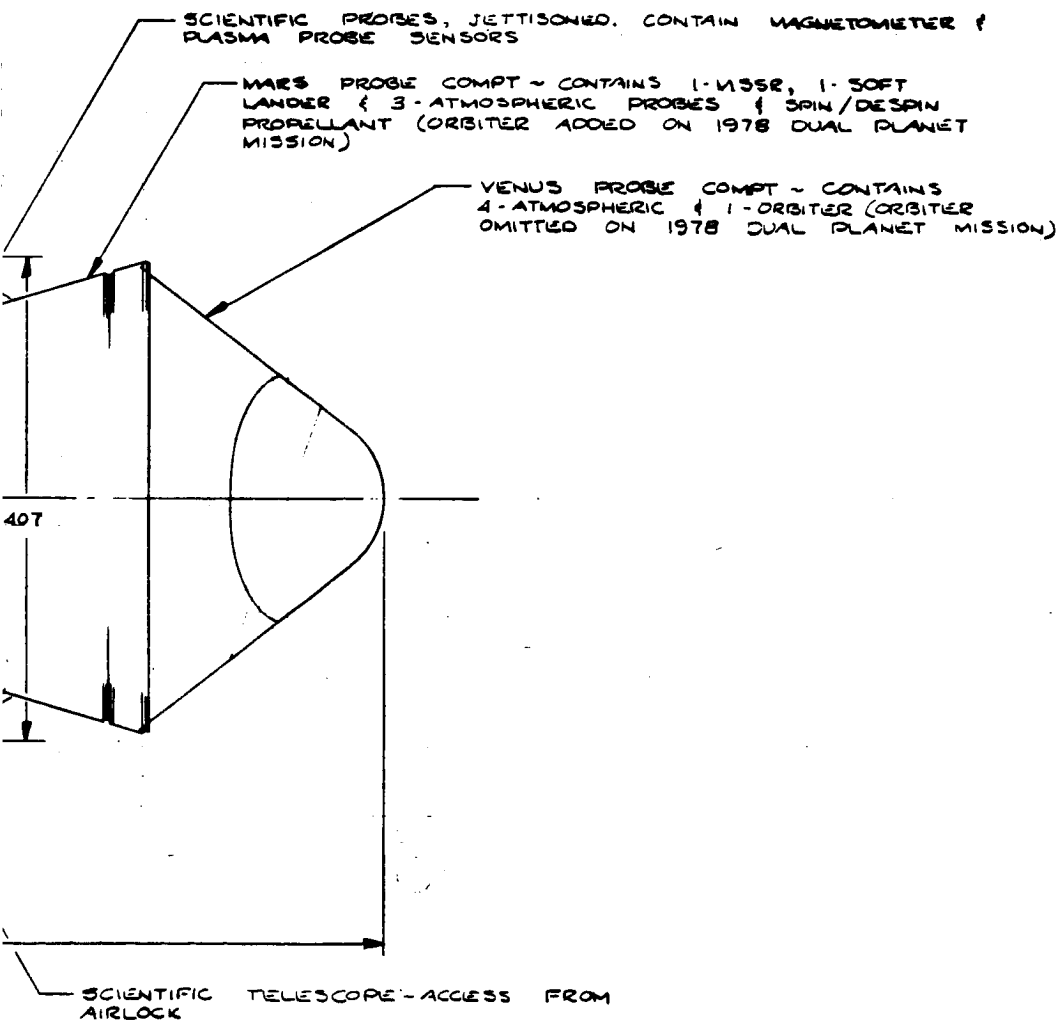
FOLDOUT FRAME

2



19 FT. DIA PARABOLIC ANTENNA ~ S/C TO EARTH & S/C TO PROBE COMMUNICATIONS

LASER EXPERIMENT ~ S/C TO EARTH COMMUNICATIONS



SCIENTIFIC SENSORS ~ RADAR & R.F. ANTENNAE, HIGH & LOW ENERGY PARTICLE & NEUTRON SPECTROMETER, SOLAR U.V. & X-RAY. ROTATE WITH TELESCOPE

PLANET  
UNPOWERED TURN  
TO LIGHTSIDE

2.4. Manned Mars/Venus Vehicle, Mission Module Concept

2-7, 2-8

SID 67-478-1

FOLDOUT FRAME 3

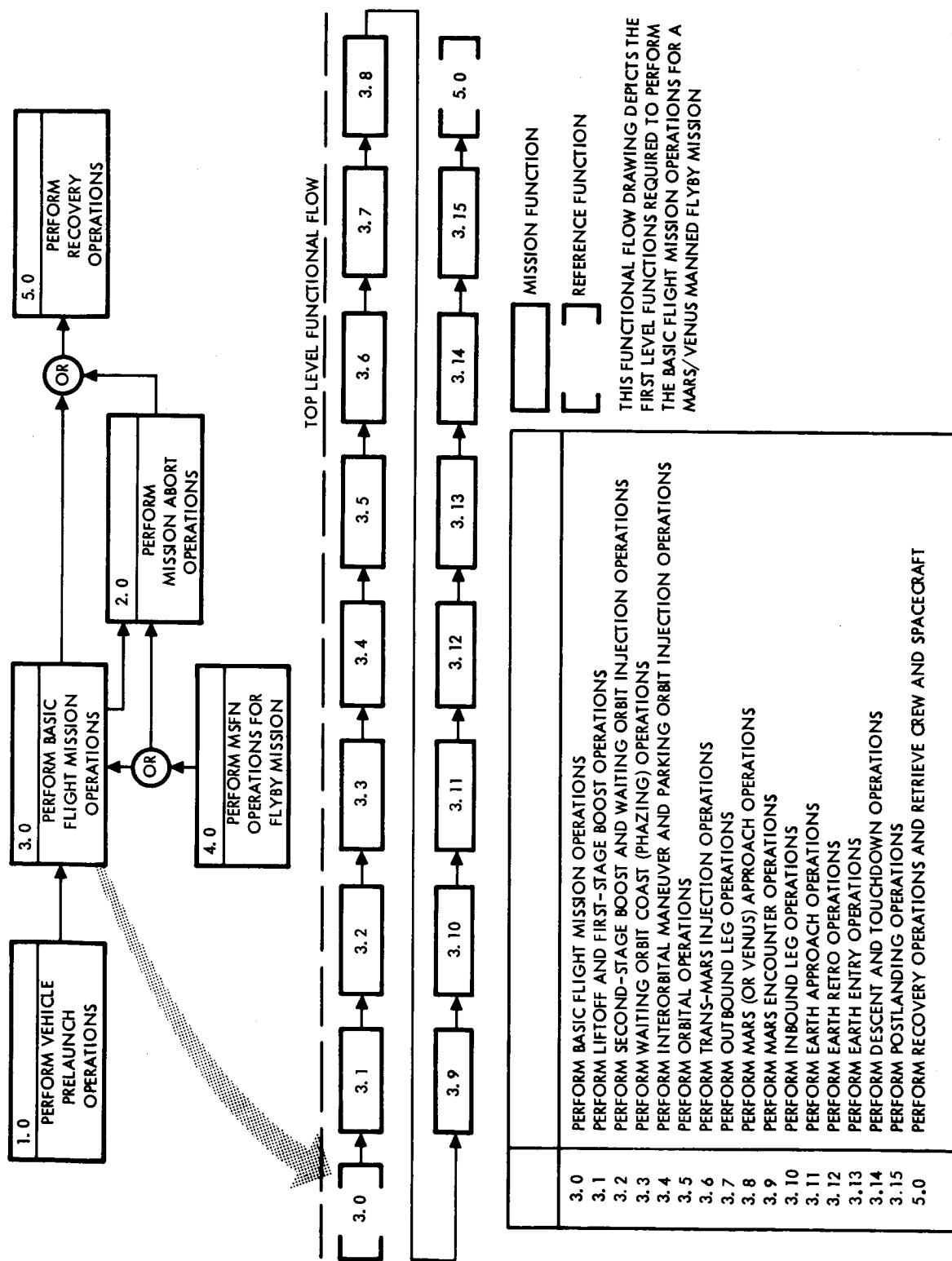
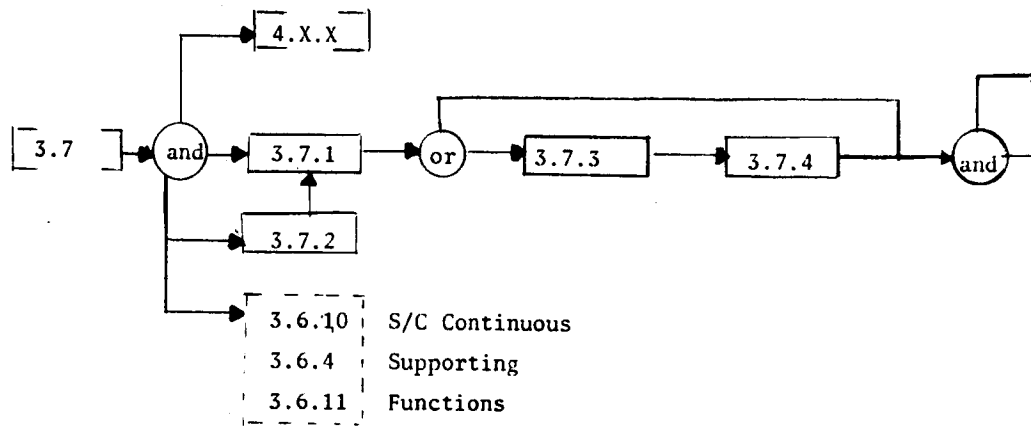


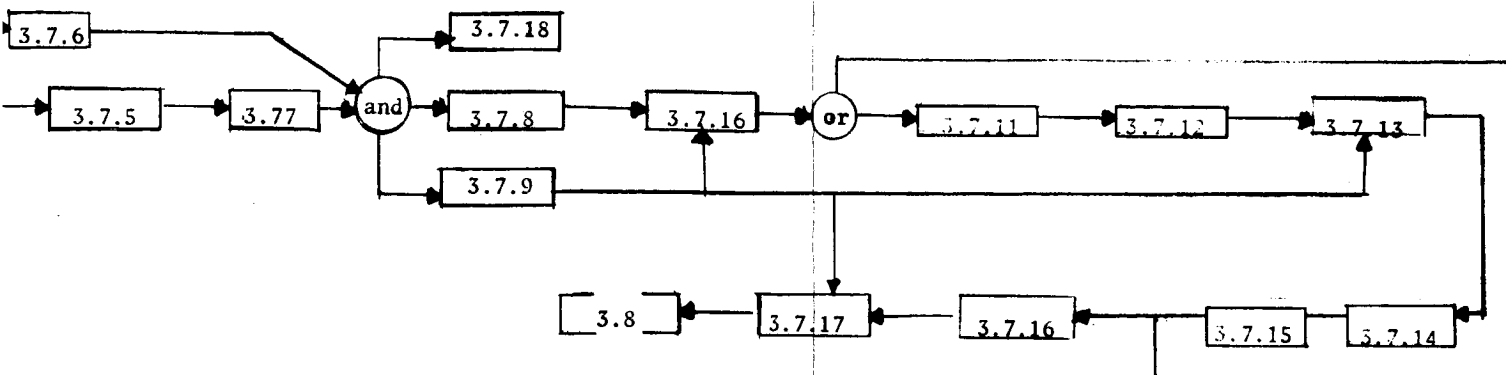
Figure 2.5. Top-Level Functional Flow Logic, Baseline Manned Mars Flyby

| Block No. | Function                                                    | Duration in House |
|-----------|-------------------------------------------------------------|-------------------|
| 3.6       | Perform Transplanetary Operations                           | - 0.70 -          |
| 3.6.1     | Perform Navigational Operations                             | - 0.30 -          |
| 3.6.2     | Perform Guidance Functions for Injection                    | 0.10              |
| 3.6.3     | Perform Countdown Operations for Injection                  | 0.05              |
| 3.6.4     | Perform Crew and System Status Monitoring                   | Continuous        |
| 3.6.5     | Apply Earth Control Guide. & Nav. Operation Correction Data | - -               |
| 3.6.6     | Initiate Injection Propulsion                               | 0.01              |
| 3.6.7     | Perform Injection Control and Monitoring Functions          | 0.25              |
| 3.6.8     | Terminate Injection Propulsion                              | 0.01              |
| 3.6.9     | Provide I.U. Trajectory Information from Booster            | Continuous        |
| 3.6.10    | Provide Earth/Spacecraft Communications                     | Continuous        |
| 3.6.11    | Provide Environment and Life Support Functions for Crews    | Continuous        |
| 4.x.x     | Provide MSFN Ground Tracking & Control Function             | Continuous        |
| 2.0       | Perform Abort Operations (as required)                      | - -               |
| 3.7       | Perform Transplanetary Leg Operations                       | - -               |

**Figure 2.6. Second-Level Functional Flow, 3.6 Preform Transplanetary Injection Operations**



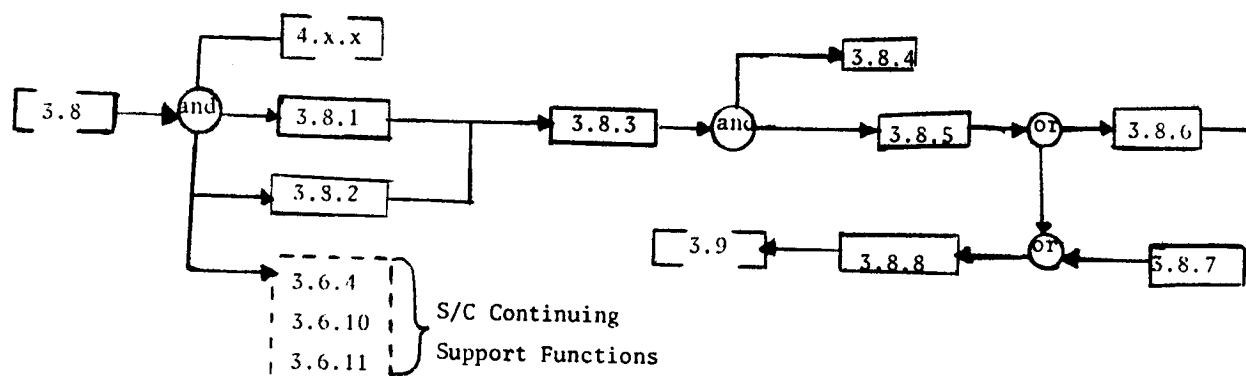
FOLDOUT FRAME /



| Block No. |                                                  | Duration in Hours |
|-----------|--------------------------------------------------|-------------------|
| 3.7       | Perform Transplanetary Leg Operations            | 3360.00           |
| 3.7.1     | Determine Trajectory Parameters                  |                   |
| 3.7.2     | Compare Trajectory Data with Earth Computed Data | 0.5               |
| 3.7.3     | Apply $\Delta V$ Correction                      |                   |
| 3.7.4     | Recalculate Trajectory Parameters                |                   |
| 3.7.5     | Perform a Complete S-C Checkout                  | 0.5               |
| 3.7.6     | Prepare for the Artificial Gravity Mode          |                   |
| 3.7.7     | Extend S/C, Achieve 1/3 Gravity and Stabilize    | 0.5               |
| 3.7.8     | Transplanetary Coast (1st Phase) *               | 1678              |
| 3.7.9     | Accept and Apply Earth Computed Trajectory Data  | 0.2               |
| 3.7.10    | Determine the need for Trajectory Correction     |                   |
| 3.7.11    | Retract S/C and return to 0 "G" Mode             | 0.5               |
| 3.7.12    | Apply $\Delta V$ for Midcourse Correction        | 0.5               |
| 3.7.13    | Recalculate Trajectory Parameters                |                   |
| 3.7.14    | Same as 3.7                                      | 0.5               |
| 3.7.15    | Same as 3.7                                      | 0.5               |
| 3.7.16    | Same as 3.8 (2nd Phase) *                        | 1678              |
| 3.7.17    | Same as 3.9                                      |                   |
| 3.7.18    | Perform Interplanetary Experiment Program        | $\approx 3300.00$ |
| 3.8       | Perform Planetary Approach Operations            |                   |

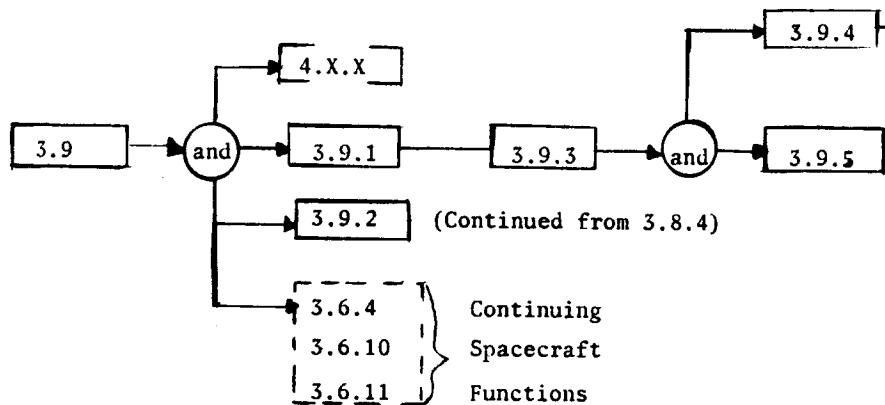
\* For details, see Third Level Functional Flow,  
Interplanetary Coast Phase

Figure 2.7. Second-Level Functional Flow, 3.7 Preform Transplanetary Leg Operations

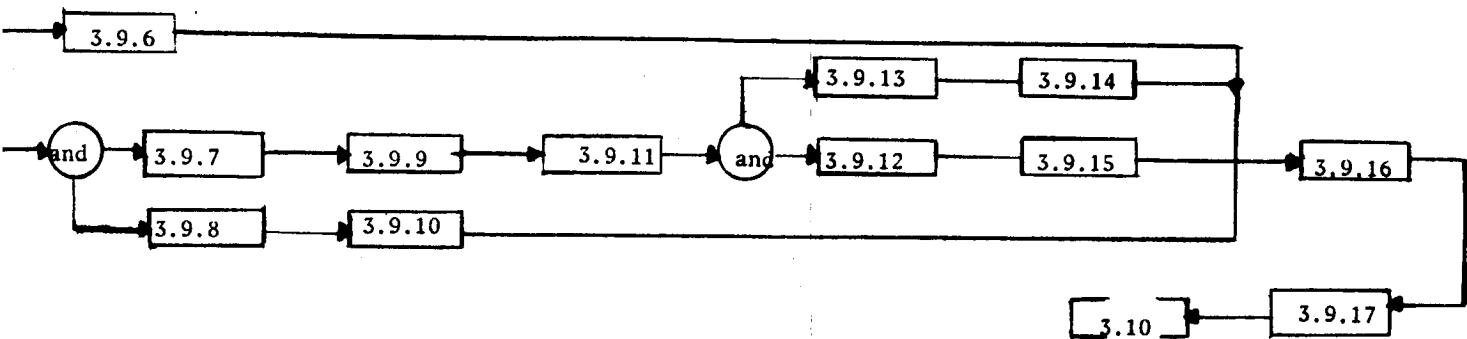


| Block No. | Function                                            | Duration in hours |
|-----------|-----------------------------------------------------|-------------------|
| 3.8       | Perform Planetary Approach Operations               | 20.00             |
| 3.8.1     | Prepare to Return to Zero "G" Mode                  | 0.5               |
| 3.8.2     | Accept & Insert Earth Computed Trajectory Data      | 0.5               |
| 3.8.3     | Return to Zero "G" Mode (as indicated by 3.8.2)     | 0.5               |
| 3.8.4     | Perform Spacecraft Stability Control Functions      | 20.00             |
| 3.8.5     | Calculate Trajectory Change Required and $\Delta V$ | 0.5               |
| 3.8.6     | Apply $\Delta V$ Correction (as required)           | 0.5               |
| 3.8.7     | Recalculate Trajectory                              | 18.5              |
| 3.8.8     | Continue Planetary Approach Coast                   |                   |
| 3.9       | Perform Planetary Encounter Operations              |                   |

Figure 2.8. Second-Level Functional Flow, 3.8 Preform Planetary Approach Operations



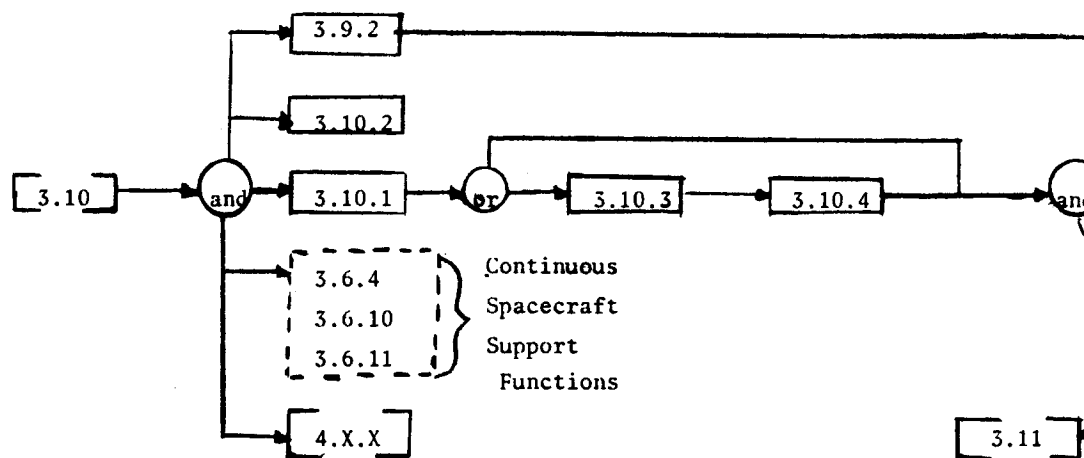
| Block No. | Function                                              |
|-----------|-------------------------------------------------------|
| 3.9       | Perform Planetary Encounter Operations                |
| 3.9.1     | Prepare Encounter Support Systems                     |
| 3.9.2     | Perform Spacecraft Attitude Control Function          |
| 3.9.3     | Prepare Planetary Scientific Equipment                |
| 3.9.4     | Initiate Optical Sensor Functions                     |
| 3.9.5     | Perform Lander Probe Deployment Functions             |
| 3.9.6     | Initiate Photographic Sensor Functions                |
| 3.9.7     | Confirm Lander Probe Trajectory                       |
| 3.9.8     | Establish Communications with Lander Probe            |
| 3.9.9     | Perform Lander Probe Tracking and Control Functions   |
| 3.9.10    | Perform Lander Probe Data Storage and Relay Function  |
| 3.9.11    | Perform Orbital Probe Insertion(s) Functions          |
| 3.9.12    | Perform Orbital Probe Tracking and Control Function   |
| 3.9.13    | Establish Orbital Probe to Spacecraft Communications  |
| 3.9.14    | Perform Orbital Probe Data Storage and Relay Function |
| 3.9.15    | Perform Encounter Scientific Experiments (others)     |
| 3.9.16    | Relay Data to Earth Control                           |
| 3.9.17    | Terminate Planetary Encounter Experiments             |
| 3.10      | Perform Transearth Operations                         |



|  | Duration<br>in Hours |
|--|----------------------|
|  | 460.00               |
|  | 0.5                  |
|  | 460.00               |
|  | —                    |
|  | —                    |
|  | 2.0                  |
|  | —                    |
|  | 1.0                  |
|  | 240.0                |
|  | 240.0                |
|  | 120.0                |
|  | 1.0                  |
|  | 6.0                  |
|  | 200                  |
|  | 200                  |
|  | 460                  |
|  | 460.0                |
|  | —                    |

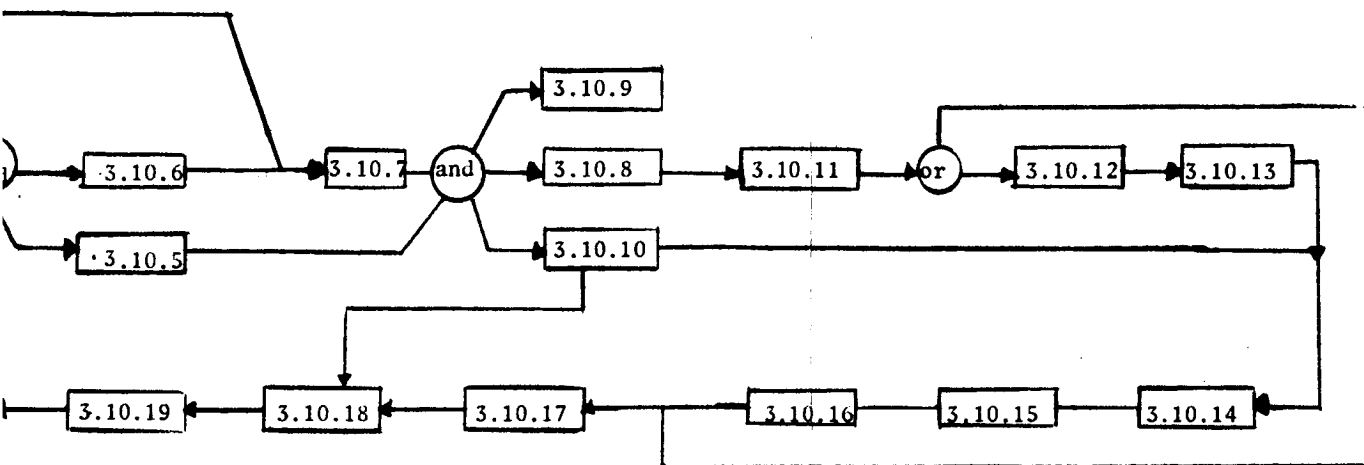
Figure 2.9. Second-Level Functional Flow, 3.9 Preform Planetary Encounter Operations

2-14, 2-15, 2-16



| Block No. | Function                                            |
|-----------|-----------------------------------------------------|
| 3.10      | Perform Transearth Operations                       |
| 3.10.1    | Determine Trajectory Parameters                     |
| 3.10.2    | Compare Trajectory Data with Earth Computed Data    |
| 3.10.3    | Apply $\Delta V$ Correction                         |
| 3.10.4    | Recalculate Trajectory Parameters                   |
| 3.10.5    | Perform a Complete Spacecraft Status Check          |
| 3.10.6    | Prepare for Artificial Gravity Mode (Includes Main) |
| 3.10.7    | Extend Spacecraft, Achieve 1/3 "G" and Stabilize    |
| 3.10.8    | Transearth Coast (1st Phase)*                       |
| 3.10.9    | Perform Transearth Experiment Program               |
| 3.10.10   | Accept and Apply Earth Computed Trajectory Data     |
| 3.10.11   | Determine Need for Trajectory Correction            |
| 3.10.12   | Retract Spacecraft and Return to Zero "G" Mode      |
| 3.10.13   | Apply $\Delta V$ for Midcourse Correction           |
| 3.10.14   | Recalculate Trajectory Parameters                   |
| 3.10.15   | Same as 3.10.6                                      |
| 3.10.16   | Same as 3.10.7                                      |
| 3.10.17   | Transearth Coast (2nd Phase)*                       |
| 3.10.18   | Same as 3.10.10                                     |
| 3.10.19   | Initiate Earth Approach Experiment Program          |
| 3.11      | Perform Earth Approach Operations                   |

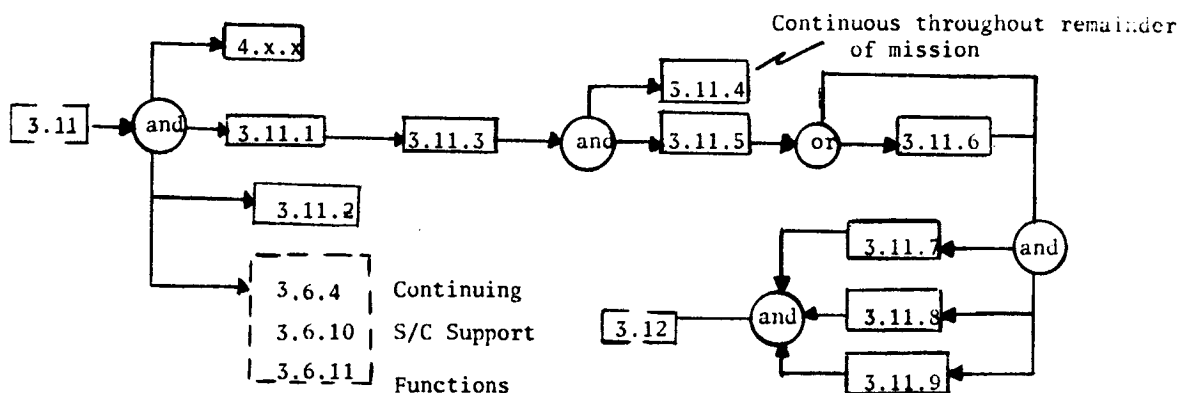
FOLDOUT FRAME



|             | Duration<br>in Hours |
|-------------|----------------------|
|             | 12,700.0             |
|             | 0.5                  |
|             | ≈ 0.5                |
| aintenance) | 0.5                  |
|             | 6,348.5              |
|             | 12,690.0             |
|             | 0.5                  |
|             | 0.5                  |
|             | 0.5                  |
|             | 0.5                  |
|             | 6,348.0              |
|             | as required          |
|             | ( unk )              |
|             | —                    |

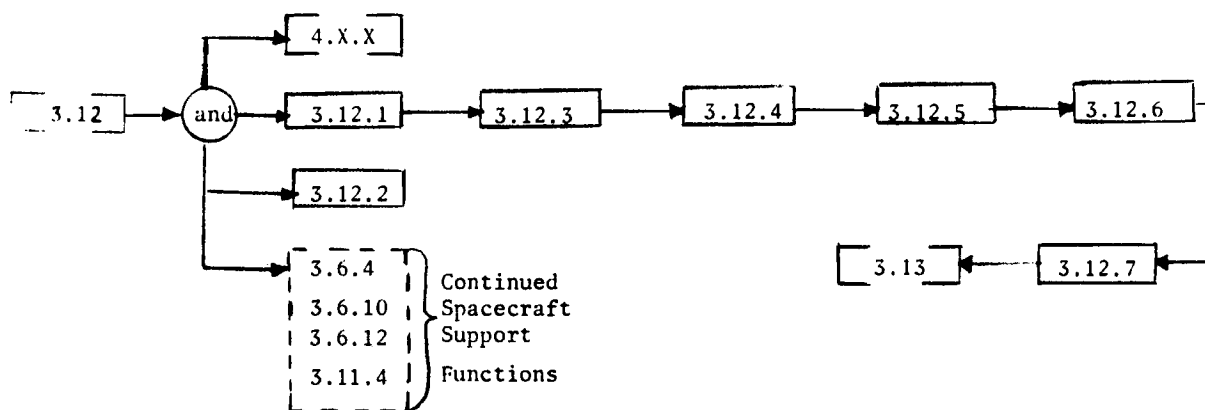
Figure 2.10. Second-Level Functional Flow, 3.10 Preform Transearth Operations

FOLDOUT FRAME 2



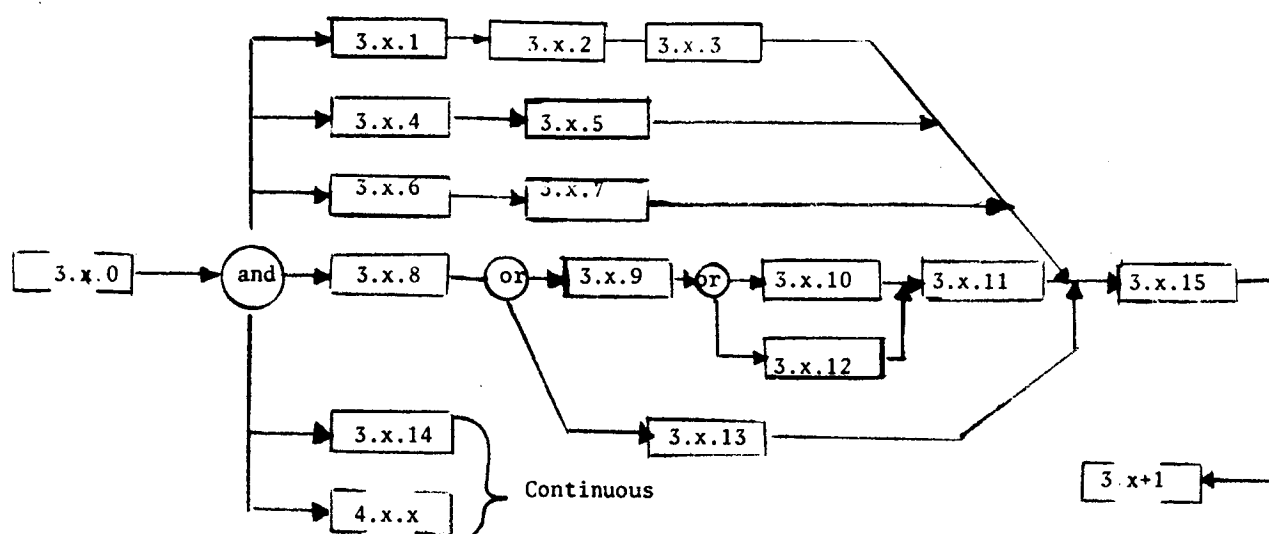
| Block No. | Function                                            | Duration in Hours |
|-----------|-----------------------------------------------------|-------------------|
| 3.11      | Perform Earth Approach Operations                   | ≈ 20.00           |
| 3.11.1    | Prepare to return to Zero "G" mode                  | 0.5               |
| 3.11.2    | Accept and Insert Earth Computed Trajectory Data    |                   |
| 3.11.3    | Return to Zero "G" Mode                             | 0.5               |
| 3.11.4    | Perform Spacecraft Attitude Control Functions       | 20.0              |
| 3.11.5    | Determine Trajectory Change and $\Delta V$ required | 0.5               |
| 3.11.6    | Apply $\Delta V$ Correction (as required)           |                   |
| 3.11.7    | Continue Earth Approach Coast                       | 18.5              |
| 3.11.8    | Terminate Experiment Program                        |                   |
| 3.11.9    | Check out and Maintain All Entry Critical Functions | unk               |
| 3.12      | Perform Earth Retro Operations                      |                   |

Figure 2.11. Second-Level Functional Flow, 3.11 Perform Earth Approach Operations



| Block No. | Function                                   | Duration in Hours |
|-----------|--------------------------------------------|-------------------|
| 3.12      | Perform Earth Retro Operations             | 0.5               |
| 3.12.1    | Determine Spacecraft Position and Velocity | "                 |
| 3.12.2    | Accept and Insert Earth Computed Data      | "                 |
| 3.12.3    | Orient Spacecraft along Earth Entry Vector | "                 |
| 3.12.4    | Apply Retro Power to Burn-out              | "                 |
| 3.12.5    | Reorient Spacecraft for Jettison Operation | "                 |
| 3.12.6    | Jettison Retro Vehicle                     | "                 |
| 3.12.7    | Continue Coast to Earth Entry              | "                 |
| 3.13      | Perform Earth Entry Operations             |                   |

Figure 2.12. Second-Level Functional Flow, 3.12 Perform Earth Retro Operations



\* See Fourth Level FF

| Block No. | Function                                                     | Duration - Hours |               |
|-----------|--------------------------------------------------------------|------------------|---------------|
|           |                                                              | Trans - Mars     | Trans - Earth |
| 3.x.x.    | Perform Transplanetary or Trans-Earth Functions              | 1680             | 6360          |
| 3.x.1     | Maintain Artificial "G"                                      | "                | "             |
| 3.x.2     | Maintain Wobble Damping                                      | "                | "             |
| 3.x.3     | Precess Spacecraft Spin Axis to maintain Desired Orientation | "                | "             |
| 3.x.4     | Provide Electrical Power for Experiments                     | "                | "             |
| 3.x.5     | Conduct Interplanetary Experiments                           | "                | "             |
| 3.x.6     | Maintain Communications with Earth                           | "                | "             |
| 3.x.7     | Store and Transmit Data to Earth                             | "                | "             |
| 3.x.8     | Monitor System Performance                                   | "                | "             |
| 3.x.9     | System Function Failure                                      | *                | *             |
| 3.x.10    | Perform Required Maintenance Action                          | *                | *             |
| 3.x.11    | Test and Restore Normal Operation                            | *                | *             |
| 3.x.12    | Switch to a Redundant Function                               | -                | -             |
| 3.x.13    | Continue Normal Operation                                    | -                | -             |
| 3.x.14    | Provide Habitable Environment conducive to Extended Missions | 1680             | 6360          |
| 3.x.15    | Prepare to Return to Zero "G" Mode                           | /                | /             |
| 3.x+1     | Conduct Next Phase Operations                                |                  |               |

Figure 2.13. Third-Level Functional Flow, 3.X.X. Preform Transplanetary or Transearth Functions, 1/2-Cycle



## 2.3 MISSION FUNCTIONAL REQUIREMENTS ANALYSIS

Since the advent of the systems engineering approach, functional flow logic has been used as a means of identifying system function requirements and interfaces. This technique has been applied to the flyby mission herein; however, details were only investigated for those mission phases starting with, and subsequent to Transplanetary Injection. Refer to Figure 2.5, the top-level functional flow diagram, and note that the phases of interest are those starting with step 3.6 and those following.

The mission phases prior to transplanetary injection are not pertinent to this study because of the inherent capability to abort from earth orbit at almost any time, and the very low likelihood of this event.

Figures 2.6 through 2.12 are the second-level functional flow diagrams for the baseline mission, but, in terms of time, they represent less than 5 percent of the total mission time. Figure 2.13 presents the third level for the long transearth and transplanetary phases where the major reliability problem will exist.

## 2.4 MISSION SYSTEM FUNCTION DUTY CYCLES

### 2.4.1 The Importance of Duty-Cycle Estimates

At best, duty cycle estimates are difficult to derive, even when there is a well-defined mission and implementing program. The intent is to determine the points during a mission, the system functions used, and the duration of these use periods. In essence, it requires conducting a simulated mission even before the mission system is designed.

Should the question arise as to why this is necessary, the requirement becomes self-evident from a reliability requirements analysis standpoint. Mission reliability (R) is expressed as:

$$R = e^{-\frac{t}{MTBF}}$$

where:

t = Duty cycle (time)

MTBF = Mean time before failure



It is evident from this that the more a system function is used (or required) the higher the MTBF must be to hold R constant and assure a successful mission. Further, a few simple calculations will reveal that the MTBF must be at least three times as great as its expected duty cycle, so as to provide even a reasonable probability of success. But even that is not good enough for missions where abort is impractical.

Available data indicates that many contemporary system components cannot meet the long MTBF/Life requirements imposed by the Flyby Mission; others are marginal. Therefore it is necessary to design a mission which minimizes the demands on a system, and further, to identify as accurately as possible what these demands will be. An error in the duty-cycle estimate of a factor of only 2 could increase the risk from five losses in one hundred (0.95) to ten in one hundred (0.9), thus doubling the risk. This must be reflected in a need for increased redundancy and/or spares and maintenance operations.

#### 2.4.2 Approach to Duty-Cycle Estimates

Since it has been established that the duty-cycle estimates are sensitive design criteria, assume that the method of derivation is also sensitive. A detailed time-line analysis was therefore performed to derive accurately the required values. The mission was simulated on a phase-by-phase basis, and each mission activity, as derived from the functional flow analysis, was evaluated for the individual function operational requirements. Available power and crew activities provided useful constraints to bound the probable values. The mission phase and selected mode of operation also provided useful boundary values. For example, the decision to use an artificial gravity mode for the long transplanetary/transearth phases eliminated the need for stability control, navigation sightings, and reaction control during this phase. Where there was some doubt as to a requirement, it was assumed to be required. For example, no midcourse correction may be required after the initial vector adjustment and before planetary approach. Therefore, the spin mode would continue, saving fuel and reducing the overall risk. However, the worse case was assumed.

#### 2.4.3 Manned Mars Flyby Duty-Cycle Estimates

The system function duty cycles were estimated in conjunction with the mission planners and subsystem designers. The results are presented in Table 2.1, Parts 1 through Part 3. The values are expressed in hours, as a function of the appropriate mission phase, except where noted. The total system function duty cycle is given in the last column.



Table 2.1. Top-Level, System Function Duty Cycle Estimates (Part I)

| Block Number                   | 1.0       | 3.1 to 3.4                | 3.5         | 3.6                    | 3.7                | 3.8             | 3.9              | 3.10       | 3.11           | 3.12        | 3.13        | 3.14     | 3.15           | Total System Usage |
|--------------------------------|-----------|---------------------------|-------------|------------------------|--------------------|-----------------|------------------|------------|----------------|-------------|-------------|----------|----------------|--------------------|
| Mission Phase                  | Prelaunch | Ascent to Orbit Injection | Earth Orbit | Trans-Planet Injection | Trans-Planet Coast | Planet Approach | Planet Encounter | Transearth | Earth Approach | Earth Retro | Earth Entry | Recovery |                |                    |
| Phase Duration                 | 96.0      | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0.10        | 0.36        | 72.0     | 16,801.2       |                    |
| <b>Communication</b>           |           |                           |             |                        |                    |                 |                  |            |                |             |             |          |                |                    |
| HF Transceiver                 | 0.5       | 0                         | 1.0         | 0                      | 0                  | 0               | 0                | 0          | 0              | 0           | 0           | 12       | 13.05          |                    |
| VHF/AM Transceiver             | 0.5       | 12.0                      | 24.0        | 0.70                   | 33.6               | 2.0             | 460              | 12.7       | 8.0            | 0.10        | 0.36        | -        | 553.96         |                    |
| VHF Beacon                     | 0.5       | 0                         | 1.0         | 0.70                   | 8.0                | 0               | 0                | 0          | 8.0            | 0.10        | 0.36        | 72       | 90.66          |                    |
| "5" Band Tran and Amp          | 0.5       | 0                         | 1.0         | 0                      | 336                | 10.0            | 230              | 1270       | 10.0           | 0           | -           | -        | 1,847.5        |                    |
| Unified "S" Band Rec.          | 0.5       | 0                         | 1.0         | 0                      | 3360               | 20.0            | 460              | 12,700     | 0              | 0           | -           | -        | 16,551.5       |                    |
| "C" Band Transponder           | 0.5       | 24                        | 48.0        | 0                      | 0                  | 0               | 0                | 0          | 20.0           | 0.10        | 0.36        | -        | 82.96          |                    |
| <b>Antenna</b>                 |           |                           |             |                        |                    |                 |                  |            |                |             |             |          |                |                    |
| DSIF, High-Gain                | 0         | 0                         | 0.30        | 0                      | 3360               | 20.0            | 460              | 12,700     | 0              | 0           | -           | -        | 16,551.5       |                    |
| UHF Multiplexer                | 1.0       | 12                        | 24          | 0.70                   | 0                  | 10.0            | 460              | 48.0       | 8.0            | 0.10        | 0.36        | -        | 564.16         |                    |
| VHF, Omnidirectional           | 1.0       | 24.0                      | 24          | 0.70                   | 0                  | 10.0            | 460              | 48.0       | 8.0            | 0.10        | 0.36        | -        | 576.16         |                    |
| VHF Switch (Cycles)            | 10 cycles | 0                         | 4 cycles    | 0                      | 0                  | 30 cycles       | 1380 cycles      | 144 cycles | 4 cycles       | 0           | 0           | -        | 1,572.0 cycles |                    |
| <b>Data Handling</b>           |           |                           |             |                        |                    |                 |                  |            |                |             |             |          |                |                    |
| Audio Center                   | 5.0       | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460.0            | 12,700     | 20.0           | 0.10        | 0.36        | 12.0     | 16,710.2       |                    |
| Premod. Processor              | 3.0       | 12.0                      | 24.0        | 0.70                   | 1120               | 20.0            | 460              | 4,250      | 10.0           | 0           | -           | -        | 5,899.70       |                    |
| Data Storage                   | 3.0       | 0                         | 24.0        | 0                      | 1680               | 20              | 460              | 6,350      | 8.0            | 0           | -           | -        | 8,545.0        |                    |
| PCM Telemetry                  | 3.0       | 24.0                      | 24.0        | 0.70                   | 0                  | 10              | 460              | 48.0       | 10.0           | -           | -           | -        | 565.70         |                    |
| TV Camera                      | 1.0       | 0                         | 12.0        | 0                      | 8.0                | 0               | 24               | 8.0        | 1.0            | -           | -           | -        | 54.0           |                    |
| Central Timer                  | 5.0       | 24.0                      | 48.0        | 0.70                   | 3360               | 20              | 460              | 12,700     | 20.0           | 0.10        | 0.36        | -        | 17,560.46      |                    |
| Up Data Link                   | 3.0       | 24.0                      | 24.0        | 0.70                   | 0                  | 10              | 460              | 8.0        | 10.0           | 0           | 0.36        | -        | 530.16         |                    |
| Teleprinter                    | 1.0       | 0                         | 0.5         | 0                      | 33.6               | 0               | 1.0              | 127        | 0              | 0           | -           | -        | 163.1          |                    |
| Signal Conditioner             | 3.0       | 24.0                      | 24.0        | 0.70                   | 1120               | 20              | 460              | 4,250      | 10.0           | 0.10        | 0.10        | -        | 5,911.90       |                    |
| <b>Guidance and Navigation</b> |           |                           |             |                        |                    |                 |                  |            |                |             |             |          |                |                    |
| Guidance Computer              | 4.0       | 24                        | 12          | 0.70                   | 168                | 18.0            | 400              | 168        | 18.0           | 0.10        | 0.36        | -        | 813.16         |                    |
| Inertial Measure               | 4.0       | 24                        | 24          | 0.70                   | 168                | 18.0            | 400              | 168        | 18.0           | 0.10        | 0.36        | -        | 825.16         |                    |
| Optical Alignment              | 0.1       | 2                         | 2           | 0                      | 1.0                | 0.30            | 1.0              | 1.0        | 0.30           | 0           | 0           | -        | 7.70           |                    |
| Displays and Control           | 4.0       | 24                        | 24          | 0.70                   | 168                | 18.0            | 400              | 168        | 18.0           | 0.10        | 0.36        | -        | 825.16         |                    |



Table 2.1. Top-Level, System Function Duty Cycle Estimates (Part 2)

| Block Number                | 1.0       | 3.1 to 3.4                | 3.5         | 3.6                    | 3.7                | 3.8             | 3.9              | 3.10       | 3.11           | 3.12        | 3.13        | 3.14     | 3.15 | Total System Usage |
|-----------------------------|-----------|---------------------------|-------------|------------------------|--------------------|-----------------|------------------|------------|----------------|-------------|-------------|----------|------|--------------------|
| Mission Phase               | Prelaunch | Ascent to Orbit Injection | Earth Orbit | Trans-Planet Injection | Trans-Planet Coast | Planet Approach | Planet Encounter | Transearth | Earth Approach | Earth Retro | Earth Entry | Recovery |      |                    |
| Phase Duration              | 96.0      | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0.10        | 0.36        | 72.0     |      | 16,801.2           |
| <u>Spin - Desin Control</u> |           |                           |             |                        |                    |                 |                  |            |                |             |             |          |      |                    |
| Electronics                 | 1.5       | 0                         | 0.30        | 0                      | 3190               | 1.2             | 0                | 12,532     | 1.2            | -           | -           | -        | -    | 15,727.2           |
| Propellant Storage          | 18        | 24.0                      | 0.70        | 0.70                   | 3190               | 20.0            | 460              | 12,532     | 20.0           | -           | -           | -        | -    | 16,312.7           |
| Propellant Management       | 1.5       | 0                         | 0.30        | 0                      | 3190               | 1.2             | 0                | 12,532     | 1.2            | -           | -           | -        | -    | 15,726.2           |
| Engine - Cycles (Sp.)       | 1         | 0                         | 1           | 0                      | 7                  | 1               | 0                | 7          | 1              | -           | -           | -        | -    | 16 cycles          |
| - Burn                      |           | 0                         |             | 0                      | 4090               | 1250            | 0                | 3,615      | 1000           | -           | -           | -        | -    | 9,995 sec          |
| Wobble Damper               | 1.5       | 0                         | 0.30        | 0                      | 3350               | 1.2             | 0                | 12,532     | 1.2            | -           | -           | -        | -    | 15,886.2           |
| Process Engines (cy)        | 1 cy.     | 0                         | 1 cy.       | 0                      | 8 cy.              | 0               | 0                | 10 cy.     | 0              | -           | -           | -        | -    | 20 cycles          |
| - Burn                      | 2 sec     | 0                         | 2 sec       | 0                      | 175 sec            | 0               | 0                | 200 sec    | 0              | -           | -           | -        | -    | 379 sec            |
| <u>Mechanical Systems</u>   |           |                           |             |                        |                    |                 |                  |            |                |             |             |          |      |                    |
| S/C Structure (MM)          | 0         | 0                         | 46.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0.10        | -           | -        | -    | 16,606.8           |
| Centrifuge                  | 0.2       | 0                         | 0           | 0                      | 560                | 0               | 0                | 2,120      | 0              | -           | -           | -        | -    | 2,680.2            |
| Ext. - Ret. Cont.           | 0.5       | 0                         | 0.30        | 0                      | 3350               | 1.2             | 0                | 12,690     | 1.2            | -           | -           | -        | -    | 16,043.2           |
| Ext. Ret. Mech.             | 0         | 0                         | 0.30        | 0                      | 3350               | 1.2             | 0                | 12,690     | 1.2            | -           | -           | -        | -    | 16,042.7           |
| Docking Mech.               | 0         | 0                         | 0           | 0                      | 10.0               | 18.8            | 460              | 10.0       | 20.0           | 0.10        | -           | -        | -    | 518.9              |
| Reentry Module              | 0         | 24.0                      | 2.0         | 0                      | 0                  | 0               | 0                | 0          | 0              | 0.10        | 0.36        | 72       | 98.4 |                    |
| Storm Cellar                | 0         | 0                         | 0           | 0                      | (3.4)              | 0               | 0                | (127)      | -              | -           | -           | -        | -    | (130 Max)          |
| Performance Monitor         | 3.0       | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 20.0           | -           | -           | -        | -    | 16,635.7           |
| <u>Diagnostic Systems</u>   |           |                           |             |                        |                    |                 |                  |            |                |             |             |          |      |                    |
| S/C Propulsion              | 0.5       | 0                         | 2.0         | 0                      | 80                 | 1.0             | 10.2             | 302        | 1.0            | -           | -           | -        | -    | 396.7              |
| Propellant                  | 18.0      | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0.1         | -           | -        | -    | 16,602.8           |
| Propellant Management       | 1.5       | 2.0                       | 2.0         | 0                      | 0                  | 2.6             | 0                | 2.0        | 0              | 0.1         | -           | -        | -    | 9.6                |
| Engine - cycles             | 0         | 0                         | 1 cy.       | 0                      | 1 cy.              | 2 cy.           | 0                | 2 cy.      | 0              | 1 cy.       | -           | -        | -    | 7 cycles max       |
| - burn                      | 0         | 0                         | 0.5         | 0                      | 25                 | 50              | 0                | 50         | 0              | 25          | -           | -        | -    | 155 seconds        |



Table 2.1. Top-Level, System Function Duty Cycle Estimates (Part 3)

| Block Number                      | 1.0       | 3.1 to 3.4                | 3.5         | 3.6                    | 3.7                | 3.8             | 3.9              | 3.10       | 3.11           | 3.12        | 3.13        | 3.14     | 3.15       | Total System Usage |
|-----------------------------------|-----------|---------------------------|-------------|------------------------|--------------------|-----------------|------------------|------------|----------------|-------------|-------------|----------|------------|--------------------|
| Mission Phase                     | Prelaunch | Ascent to Orbit Injection | Earth Orbit | Trans-Planet Injection | Trans-Planet Coast | Planet Approach | Planet Encounter | Transearth | Earth Approach | Earth Retro | Earth Entry | Recovery |            |                    |
| Phase Duration                    | 96.0      | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0.10        | 0.36        | 72.0     | 16,801.2   |                    |
| <u>Electrical Power</u>           |           |                           |             |                        |                    |                 |                  |            |                |             |             |          |            |                    |
| Isotope Source                    | 5.0       | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0.10        | 0.36        | -        | 16,638.2   |                    |
| Energy Conversion                 | 5.0       | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0.10        | 0.36        | -        | 16,638.2   |                    |
| Processing and Distribution       | 5.0       | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0.10        | 0.36        | -        | 16,638.2   |                    |
| Storage - cycles                  | 1 cy.     | 0                         | 5 cy.       | 0                      | 336 cy.            | 10 cy.          | 230 cy.          | 1,270 cy.  | 2 cy.          | 0           | 0           | 0        | 1,854 cy.  |                    |
| - duration                        | 5.0       | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0.10        | 0.36        | 72.0     | 16,710.2   |                    |
| <u>Stability Control</u>          |           |                           |             |                        |                    |                 |                  |            |                |             |             |          |            |                    |
| Electronics                       | 3.0       | 24.0                      | 48.0        | 0.70                   | 168                | 18.8            | 460              | 168        | 18.8           | 0.10        | -           | -        | 909.4      |                    |
| Fuel Storage                      | 18        | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 18.8           | 0.10        | -           | -        | 16,649.6   |                    |
| Fuel Feed (cycles)                | 1 cy.     | 100 cy.                   | 150 cy.     | 2 cy.                  | 602 cy.            | 400 cy.         | 8,000 cy.        | 602 cy.    | 200 cy.        | 2 cy.       | -           | -        | 10,059 cy. |                    |
| Reaction Eng. (cycles)            | 1 cy.     | 100 cy.                   | 150 cy.     | 2 cy.                  | 602 cy.            | 400 cy.         | 8,000 cy.        | 602 cy.    | 200 cy.        | 2 cy.       | -           | -        | 10,059 cy. |                    |
| Manual Control                    | 0.2       | 0                         | 0.5         | 0                      | 0                  | 2.0             | 46.0             | 0          | 2.0            | 0.10        | -           | -        | 50.8       |                    |
| <u>Life Support (L.S.)</u>        |           |                           |             |                        |                    |                 |                  |            |                |             |             |          |            |                    |
| Cabin Temperature                 | 5.0       | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0.10        | 0.36        | -        | 16,638.2   |                    |
| O <sub>2</sub> Supply and Control | 5.0       | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0.10        | 0.36        | -        | 16,638.2   |                    |
| N <sub>2</sub> Supply and Control | 5.0       | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0.10        | 0.36        | -        | 16,638.2   |                    |
| Waste Management                  | 0         | 24.0                      | 48.0        | 0                      | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0           | -           | -        | 16,632.0   |                    |
| CO <sub>2</sub> Removal           | 0.5       | 24.0                      | 48.0        | 0                      | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0           | -           | -        | 16,632.0   |                    |
| Trace Cont. Removal               | 0.5       | 24.0                      | 48.0        | 0                      | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0           | -           | -        | 16,632.0   |                    |
| Water Management                  | 0.2       | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0           | -           | -        | 16,632.0   |                    |
| Personnel Hygiene                 | 0.2       | 0                         | 4.0         | 0                      | 786                | 2.0             | 19               | 2,180      | 4.0            | 0           | -           | -        | 16,633.0   |                    |
| Food Management                   | 0.2       | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0           | -           | -        | 16,638.2   |                    |
| Personal L.S. (Portable)          | 0         | 0                         | 4.0         | 0                      | 0                  | 4.0             | 0                | 2.0        | 2.0            | 0           | -           | -        | 12.0 (Max) |                    |
| <u>Environment Control</u>        |           |                           |             |                        |                    |                 |                  |            |                |             |             |          |            |                    |
| Coolant (Equip) Temp              | 5.0       | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0.10        | 0.36        | -        | 16,638.2   |                    |
| Atmospheric Pressure              | 3.0       | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0.10        | 0.36        | -        | 16,636.2   |                    |
| Humidity Cont.                    | 3.0       | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0.10        | 0.36        | -        | 16,636.2   |                    |
| Space Suits                       | 3.0       | 24.0                      | 24.0        | 0                      | 0                  | 4.0             | 0                | 2.0        | 2.0            | 0.10        | 0.36        | 0        | 59.6 (Max) |                    |
| Isotope Cooling                   | 5.0       | 24.0                      | 48.0        | 0.70                   | 3360               | 20.0            | 460              | 12,700     | 20.0           | 0.10        | 0.36        | -        | 16,638.2   |                    |



Note that these estimates are normally the maximum expected requirements. Any out-of-service time resulting from maintenance can reduce many of these totals without undue penalties on the crew or mission.

The block numbers at the column heads correspond to the associated function on the functional flow diagrams. To facilitate a more accurate estimate of the transearth/transplanet phase requirements, function 3.7 and 3.10, these were expanded on Tables 2.2 and 2.3, respectively.

## 2.5 MISSION SUCCESS CRITERIA AND CREW SAFETY ASSURANCE

### 2.5.1 Providing Crew Safety Assurance

Crew safety assurance is an intangible result of a complete program which has taken into account all of the contributing factors. Through a logical process of design, development, and tests, the program has produced data which, when presented in concert, demonstrated a reasonable probability of safe return for the crew. In this context, it is concerned with the crew only. Five basic alternatives, open to assure space mission crew survival, follow below:

1. Provide high reliability
2. Provide for local action in the event of failure
3. Provide for a means of safe abort
4. Provide for a means of escape
5. Provide for a rescue mission

Each of these alternatives can be used under certain conditions; but none are expected to fit all situations. To determine the usefulness of these alternatives for the Mars Flyby Mission, each was analyzed and the results presented in the following. In the report, those alternatives which show promise are investigated in detail.

A high reliability design is one in which no failures will occur throughout the mission resulting in immediate or eventual loss of the crew. Former studies<sup>(1)</sup> have shown this goal to be impractical for the next decade. State-of-the-art reliabilities average between 0.1 and 5.0 times  $10^4$  hours, MTBF, even using optimum redundancy. The average Mars Mission requirement is between 20 and 1000 times  $10^4$  hours, MTBF. Obviously then, the deficit is too high to risk the mission without preparation for failure.



Table 2.2. Second-Level Duty Cycle Estimates, 3.10 Transearth Phase

| Block Number                   | 3.10.1 - 6   | 3.10.7             | 3.10.8 - 11 | 3.10.12           | 3.10.13 - 15          | 3.10.16            | 3.10.17 - 19 | Total System Usage |
|--------------------------------|--------------|--------------------|-------------|-------------------|-----------------------|--------------------|--------------|--------------------|
| Mission Phases                 | Zero-g Coast | Spin-up Spacecraft | Spin Coast  | Despin Spacecraft | Mid-Course Correction | Spin-up Spacecraft | Spin Coast   |                    |
| Sub Phase Duration             | 168          | 1.2                | 6,185.2     | 1.2               | 168                   | 1.2                | 6,185.2      | 12,700             |
| <u>Stability Control</u>       |              |                    |             |                   |                       |                    |              |                    |
| Electronics                    | 168          | 0                  | 0           | 0                 | 168                   | 0                  | 0            | 336                |
| Fuel Storage                   | 168          | 1.2                | 6,185.2     | 1.2               | 168                   | 1.2                | 6,185.2      | 12,700             |
| Reaction Eng. -cycles          | 301 cycles   | 0                  | 0           | 0                 | 301 cycles            | 0                  | 0            | 602 cycles         |
| Fuel Feed - cycles             | 301 cycles   | 0                  | 0           | 0                 | 301 cycles            | 0                  | 0            | 602 cycles         |
| <u>S/C Propulsion</u>          |              |                    |             |                   |                       |                    |              |                    |
| Propellant Storage             | 168          | 1.2                | 6,185.2     | 1.2               | 168                   | 1.2                | 6,185.2      | 12,700             |
| Propellant Management          | 2.0*         | 0                  | 0           | 0                 | 20*                   | 0                  | 0            | 22                 |
| Engine - cycles                | 2 cycles     | 0                  | 0           | 0                 | 1 cycle*              | 0                  | 0            | 3 cycles           |
| - burn                         | 25 sec*      | 0                  | 0           | 0                 | 25 sec*               | 0                  | 0            | 50 seconds         |
| <u>Spin - Despin Control</u>   |              |                    |             |                   |                       |                    |              |                    |
| Electronics                    | 0            | 1.2                | 6,185.2     | 1.2               | 0                     | 1.2                | 6,185.2      | 12,374             |
| Propellant Storage             | 168          | 1.2                | 6,185.2     | 1.2               | 168                   | 1.2                | 6,185.2      | 12,700             |
| Propellant Management          | 0            | 1.2                | 6,185.2     | 1.2               | 0                     | 1.2                | 6,185.2      | 12,374             |
| Spin Eng. -cycles              | 0            | 1 cycle            | 2 cycles    | 1 cycle           | 0                     | 1 cycle            | 2 cycles     | 7 cycles           |
| - burn                         | 0            | 1200 sec           | 85 sec      | 1150 sec          | 0                     | 1100 sec           | 80 sec       | 3,615 seconds      |
| Precess Eng. -cycles           | 0            | 0                  | 5 cycles    | 0                 | 0                     | 0                  | 5 cycles     | 10 cycles          |
| - burn                         | 0            | 0                  | 100 sec     | 0                 | 0                     | 0                  | 100 sec      | 200 seconds        |
| Wobble Damper                  | 0            | 1.2                | 6,185.2     | 1.2               | 0                     | 1.2                | 6,185.2      | 12,374             |
| <u>Guidance and Navigation</u> |              |                    |             |                   |                       |                    |              |                    |
| Guidance Computer              | 84           | 0                  | 0           | 0                 | 84                    | 0                  | 0            | 168                |
| Inertial Measure               | 84           | 0                  | 0           | 0                 | 84                    | 0                  | 0            | 168                |
| Displays and Cont.             | 84           | 0                  | 0           | 0                 | 84                    | 0                  | 0            | 168                |
| *Worst Case Assumed            |              |                    |             |                   |                       |                    |              |                    |



Table 2.2. Second-Level Duty Cycle Estimates, 3.10 Transearth Phase (Cont)

| Block Number                 | 3.7.1 - 6    | 3.7.7              | 3.7.8 - 10 | 3.7.11            | 3.7.12 - 14           | 3.7.15             | 3.7.16 - 18 | Total System Usage |
|------------------------------|--------------|--------------------|------------|-------------------|-----------------------|--------------------|-------------|--------------------|
| Mission Phases               | Zero-g Coast | Spin-up Spacecraft | Spin Coast | Despin Spacecraft | Mid-Course Correction | Spin-up Spacecraft | Spin Coast  |                    |
| Sub Phase Duration           | 168          | 1.2                | 1,515.7    | 1.2               | 168                   | 1.2                | 1,515.7     | 3,360              |
| <u>Stability Control</u>     |              |                    |            |                   |                       |                    |             |                    |
| Electronics                  | 168          | 0                  | 0          | 0                 | 168                   | 0                  | 0           | 336                |
| Fuel Storage                 | 168          | 1.2                | 1,515.2    | 1.2               | 168                   | 1.2                | 1,515.7     | 3,360              |
| Reaction Eng. -cycle         | 301 cycles   | 0                  | 0          | 0                 | 301 cycles            | 0                  | 0           | 602 cycles         |
| Fuel Feed-cycle              | 301 cycles   | 0                  | 0          | 0                 | 301 cycles            | 0                  | 0           | 602 cycles         |
| <u>S/C Propulsion</u>        |              |                    |            |                   |                       |                    |             |                    |
| Propellant Storage           | 168          | 1.2                | 1,515.2    | 1.2               | 168                   | 1.2                | 1,515.2     | 3,360              |
| Propellant Management        | 20*          | 0                  | 0          | 0                 | 20*                   | 0                  | 0           | 40                 |
| Engine - cycles              | 2*           | 0                  | 0          | 0                 | 1*                    | 0                  | 0           | 3 cycles           |
| - burn                       | 25 sec*      | 0                  | 0          | 0                 | 25 sec*               | 0                  | 0           | 50 seconds         |
| <u>Spin - Despin Control</u> |              |                    |            |                   |                       |                    |             |                    |
| Electronics                  | 0            | 1.2                | 1,515.2    | 1.2               | 0                     | 1.2                | 1,515.2     | 2,034              |
| Propellant Storage           | 168          | 1.2                | 1,515.2    | 1.2               | 168                   | 1.2                | 1,515.2     | 3,360              |
| Propellant Management        | 0            | 1.2                | 1,515.2    | 1.2               | 0                     | 1.2                | 1,515.2     | 2,034              |
| Spin Eng. -cycles            | 0            | 1 cycle            | 2 cycles   | 1 cycle           | 0                     | 1 cycle            | 2 cycles    | 7 cycles           |
| - burn                       | 0            | 1400 sec           | 20 sec     | 1350 sec          | 0                     | 1300 sec           | 20 sec      | 4,090 seconds      |
| Precess Eng. -cycle          | 0            | 0                  | 4 cycles   | 0                 | 0                     | 0                  | 4 cycles    | 8 cycles           |
| - burn                       | 0            | 0                  | 75 sec     | 0                 | 0                     | 0                  | 75 sec      | 150 seconds        |
| Wobble Damper                | 0            | 1.2                | 1,515.2    | 1.2               | 0                     | 1.2                | 1,515.2     | 2,034              |
| <u>Guide and Navigation</u>  |              |                    |            |                   |                       |                    |             |                    |
| Guidance Computer            | 84           | 0                  | 0          | 0                 | 84                    | 0                  | 0           | 168                |
| Inertial Measure             | 84           | 0                  | 0          | 0                 | 84                    | 0                  | 0           | 168                |
| Display and Cont.            | 84           | 0                  | 0          | 0                 | 84                    | 0                  | 0           | 168                |
| *Worst Case Assumed          |              |                    |            |                   |                       |                    |             |                    |



These same studies demonstrated that, although failures will occur, there may not be very many and a modest maintenance plan will probably solve the problem.

Local Actions (Maintenance and Repair) is the basic subject of this study and has shown promise as a solution to the crew safety assurance problems.

Escape is a form of self-help, where the crew is provided a means of leaving the parent vehicle and returning to earth via some secondary vehicle. This technique is impractical because of the severe design requirements, if used for any planetary mission, after injection into the transplanetary. Such action would require a completely redundant spacecraft, as will become evident from the abort analysis.

Rescue, as an alternative for enhancing interplanetary mission crew survival, may at first seem a possibility. But, after a careful review of the timing problem, it must also be discarded. If a relief vehicle were on the pad and ready for launch, the elapsed time to rendezvous could easily exceed the time for the mission vehicle to return to earth. This does not consider launch windows or the relative velocities of either the vehicles or earth at return.

A special case of this approach, in terms of a simultaneous launch, does hold promise, but it may be impractical. Here, two vehicles travel together, each with the capability of supporting all crew members. This mission concept is bound to be wasteful because it would complicate launch operations and extend the earth orbital support activity by up to seventy (70) days. (2)

Mission abort capability implies that, in the event of a critical failure, the mission would be terminated and the crew returned home by the most expedient course. In the Mercury, Gemini, and Apollo programs, substantial gains in safety can be realized through this course of action - it amounts to over an order-of-magnitude improvement for the Apollo. However, such a course of action depends on the ability of the spacecraft to perform the required maneuver which is known to vary drastically with mission phase. The mission abort capability must be assessed on a phase-by-phase basis.

Abort performed at any time from pad operations through the earth orbital phases can proceed under the same ground rules applied to the Apollo mission, and with approximately the same effect on crew safety.



But abort performed after transplanetary injection is quite another story. During the injection phase, and, as long as the vehicle velocity (V) is less than the escape Velocity (Ve), a ballistic reentry from orbit is possible. However, when  $V > V_e$ , then abort propulsion is required to reduce V to a point where the vehicle may be recaptured by the earth's gravitational field. The abort problem geometry is given in Figure 2.14 from which it may be seen that, as the departing velocity vector  $V_o$  increases, the required abort velocity increment increases until the point is rapidly reached where abort becomes impractical. See Reference 2.1 for a detailed analysis.

Figure 2.15 presents the abort capability requirement for the first hour or more of the mission after injection, and Figure 2.16 shows the picture for the remainder of the mission. These data indicate that abort is impractical after approximately the first hour and until about 60 days after injection. Even after 60 days, a significant amount of fuel is required to initiate abort. And further, the return-leg duration represents a significant proportion of the total mission duration; it amounts to between 60 to 80 percent of the balance. In all probability it is assumed that if they can survive then, they might as well finish the remainder of the mission.

#### 2.5.2 Emergency Nature and Causations

Some planetary design concepts have included great penalties to provide an abort capability for those first few minutes, and/or despite the long return trips associated with the latter aborts. The question of concern here is, "are the capabilities worth the weight/cost penalties involved?"

To establish value, it must be shown that the best abort capability is present when it is most likely to be required or, at the least, the requirement and capability should be proportional. An examination of the emergency causations indicate that systems failures will probably create most of the emergencies requiring a compensating action. Since the likelihood of a system failure (emergency) is time dependent, i. e.,

$$\text{probability of a failure} = 1 - e^{-\frac{t}{\text{MTBF}}}$$

the further into the mission, the more likely it is that a failure will arise. This indicates that an abort capability during the first few minutes is far less necessary than later on. However, since the return leg of the later aborts is so long, abort is considered impractical for the Mars Flyby missions.

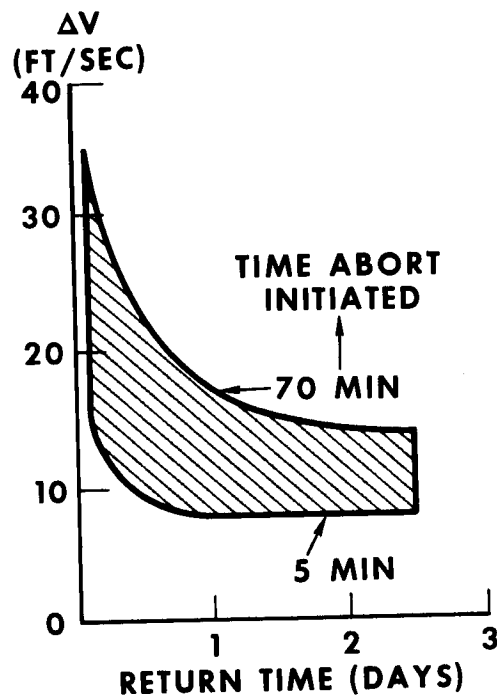


Figure 2.14. Abort Constraints and Return Time as a Function of Time Initiated

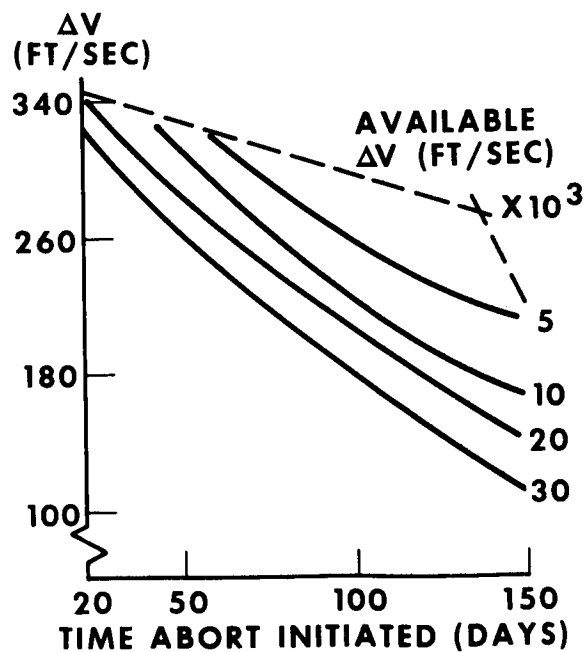


Figure 2.15. Sample Abort Profile Duration as a Function of Principle Elements

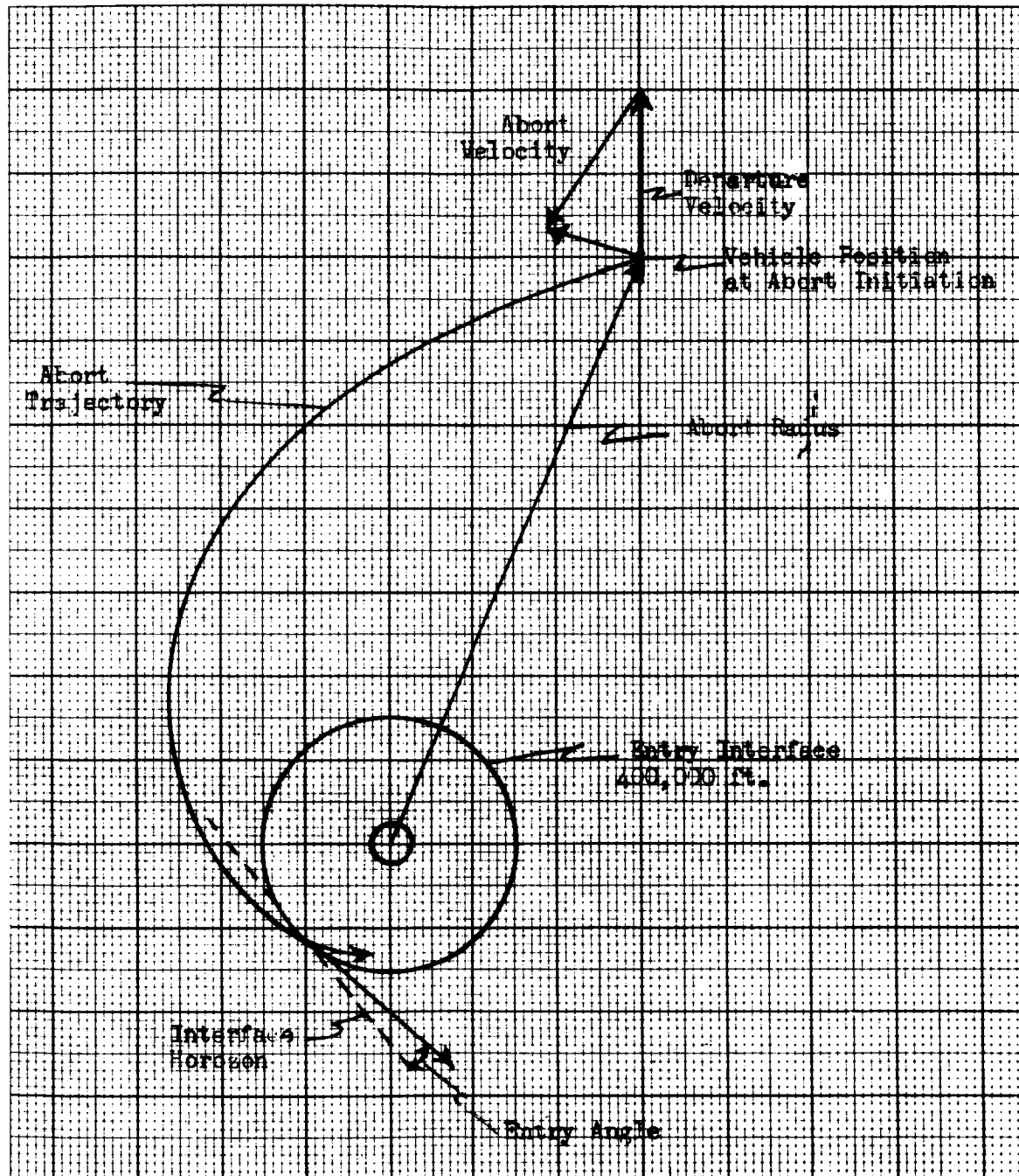


Figure 2.16. Post-Planetary Injection, Abort Problem Geometry



## 2.6 MISSION SUCCESS VERSUS CREW SAFETY FOR THE PLANETARY MISSIONS

### 2.6.1 The Problem Description

Historically, mission success is a descriptor used to indicate the probability of a mission meeting a set of prescribed objectives without failure. Equally historical is the long list of space "failures" listed as successful. For example, all of the Mercury and Gemini flights encountered failures of some form. Thus, by definition, mission success was not achieved, yet, no one would call any of these flights an actual failure. Why the paradox? The answer is in the definition of "mission success." To be classified as a success, a mission must have completed the mission plan without compromising any of the individual objectives. Loosely interpreted, this means no system failures are allowed.

Of course, this is a rather impractical way of viewing the results and leads to a lack of understanding of reliability terminology. To clarify this situation and to assure a reasonable level of safety in the manned planetary missions, a different approach is proposed for this study.

### 2.6.2 Definition of Terms

Three terms are proposed, the Probability of Safe Return ( $P_R$ ), Missions Systems Availability ( $A$ ) and Mission Success ( $P_S$ ):

The Probability of Safe Return ( $P_R$ ) expresses a crew's chance to return safely to earth after injection into the transplanetary phase. It does not necessarily imply that all the systems supporting the crew or the mission functioned throughout the mission. It is possible that many systems and functions can be out of order and yet return the crew safely to earth. For example, even though artificial gravity may be desirable and indeed mandatory for good physical health, the mission can return safely without that mode in an emergency. The same is true of such functions as communications, for it may be desirable for good mental health and to provide data transferred channels, but the crew could return safely without it.

Mission systems availability ( $A$ ) expresses the chance the mission will progress as planned and the mission systems will meet the planned profile commitments. The definition excludes only those functions specifically associated with the scientific investigations. Availability includes the effects of maintenance and repair provided that these activities are accomplished within any governing constraints. The availability concept allows for failures and takes them into account in the design activities. To meet the availability goal, all mission systems must function properly for the expressed percentage of mission time.



Mission success ( $P_s$ ) is used in a semi-conventional sense, because it is associated with all the mission goals and expresses the percentage of scientific objectives accomplished with respect to those planned. Using the term in this manner provides for a more meaningful expression of mission accomplishment. Applications of this latter class is beyond the scope of this study and, in fact, would be meaningless apart from a specifically defined mission wherein all of the scientific objectives were clearly defined.

### 2.6.3 Applying the Definitions - A Criticality Concept

Definitions of applicable mission system objectives have been provided, each expressed in terms of a probability of accomplishment and/or a percentage. These definitions provide a framework for generating the associated reliability logic diagram. These, in turn, permit an assessment of the problems associated with meeting the respective objectives. To facilitate the selection of those functions and associated components which are required to meet these mission objectives, a system of classification has been devised that will be used to identify the functions and components with the specific mission objectives. The classification is based on the criticality of the function. Therefore, it follows that,

Criticality I applies to those functions and components associated with meeting the safe return objective ( $P_R$ )

Criticality II applies to those functions and components not required to achieve  $P_R$ , but are required to achieve the mission systems availability goal (A)

Criticality III applies to those functions associated with obtaining and processing the scientific data, and are not required for the first two classes. (Note that this class will not be considered in this study.)

To facilitate an understanding of the application of criticality classes, Table 2.4 is provided at the system level. However, it should be noted that not all functions within the system belong within any one class.

### 2.6.4 Selecting Mission Goals

Establishing a set of mission goals for each of the identified success objectives has always been an arbitrary operation where the criteria used was clouded with emotion, and perhaps somewhat divorced from logic. During this study, an investigation of the risk picture for various occupations is summarized. The insurance company actuarialists were consulted and U.S. Air Force data was researched to determine what might be considered a "reasonable risk" for programs of this type. (See references 2.2 and 2.3.)



Figure 2.17 presents estimates of mortality rates for three careers that might be considered analogous to the astronaut. In summary, it indicates that the mortality rate or occupational hazard for these classes of careers is in excess of 10 per thousand, per year. In short, there is one chance in one hundred that they will not survive the year. In these figures, age was not considered a factor.

Table 2.4. Mars Flyby System Criticality Assignments

| System                         | Criticality Class | Associated Objective | Application Remarks               |
|--------------------------------|-------------------|----------------------|-----------------------------------|
| 1. Environmental control       | I                 | P <sub>R</sub>       | Some equipment cooling excluded   |
| 2. Life (habitability support) | I                 | P <sub>R</sub>       | Personnel hygiene items excluded  |
| 3. Stability control           | I                 | P <sub>R</sub>       | Applies during zero-g phase only  |
| 4. Guidance                    | I                 | P <sub>R</sub>       | Some subsystems may be in II      |
| 5. Navigation                  | II*               | A                    | Earth backup possible and primary |
| 6. Communications              | II*               | A                    | Data systems excluded             |
| 7. Propulsion                  | I                 | P <sub>R</sub>       | - - -                             |
| 8. Spin control                | II                | A                    | Required for artificial g only    |
| 9. Data systems                | III               | P <sub>s</sub>       | Required for scientific data only |
| 10. Space suits                | II                | A                    | May be required for maintenance   |
| 11. Electrical power           | I                 | P <sub>R</sub>       | Only part of the system           |

\* One or the other required for P<sub>R</sub>

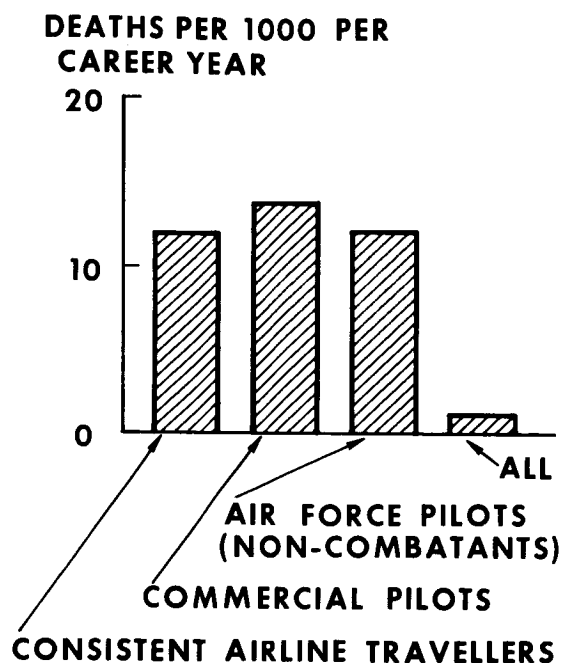


Figure 2.17. Mortality Rates for Applicable Careers in Estimating Reasonable Risks for Planetary Astronauts

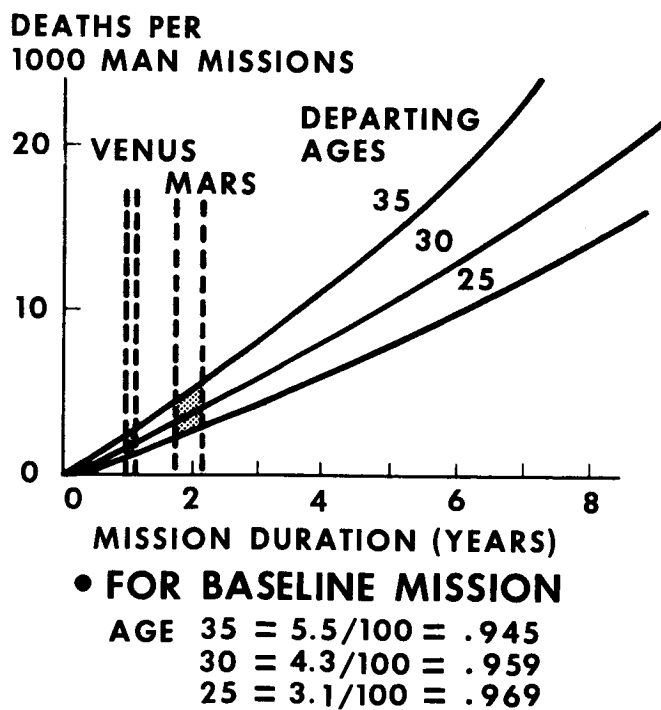


Figure 2.18. Estimate of Crew Mortality as a Function of Mission Duration, Based on Actuarial Data on Deaths in the USA From All Causes



Now, refer to Figure 2.18 where the mortality rates are plotted as a function of mission duration and age, at the time a mission was to start. These were plotted using data from deaths from all causes and career types. The data indicates that the hazard length to the Mars Flyby mission would be for a departing age of:

1. 35 yrs old, 55 per 1000 Missions.
2. 30 yrs old, 43 per 1000 Missions.
3. 25 yrs old, 31 per 1000 Missions.

The higher risk factors associated with such "safe" occupations as the airline pilot or even the flying business man raises the average risk levels to between 31 and 55 deaths per thousand missions, depending on the departing age. Since most astronauts are in the 30- to 35-year ranges, use of a risk near the upper limit would not be incongruous. One additional consideration is pertinent - crew size. If a four-man crew is involved, these figures must be multiplied by a factor of four; or, if the crew is larger, a factor equal to the crew size must be used to express the total probability of a death during a period equivalent to the mission duration. Here it is assumed that the normal occupation of the crewmen was that of an Air Force or airline pilot.

By applying these actuarial data to determine survival probability of the four-man crew for a period equivalent to the Mars Flyby mission, it is estimated that this risk could be as high as 2 out of 10. This amounts to a probability of safe return for the total four-man crew of as low as  $P_R = 0.8$ .

Thus, with these data as a background, the basic premise is that an astronaut should be willing to accept a mission risk not greater than that associated with his normal occupation, but should not be asked to accept a risk greater than the individual crew member. In this manner, a realistic goal can be set for the various mission objectives, and proposed as follows:

1. Probability of Safe Return ( $P_R$ )  $\geq 0.99$
2. Mission System Availability  $\geq 0.95$
3. Mission Success  $\geq 0.90$

It should also be noted that these goals are compatible with those applied to contemporary manned space programs. Henceforth, these will form the basis for the requirements analysis. The mission systems will be analyzed and defined to the extent necessary to assure at least these levels of risk.



## 2.7 MISSION REQUIREMENTS REFERENCES

| <u>Ref. No.</u> | <u>Title and Source</u>                                                                                                                                              |
|-----------------|----------------------------------------------------------------------------------------------------------------------------------------------------------------------|
| 2.1             | Manned Planetary Flyby Mission Based on Saturn/Apollo Systems, NAS8-18025, NAA/S&ID, SID 67-110 <u>et al.</u>                                                        |
| 2.2             | Accident Facts, National Safety Council, 1962-1963 Edition.                                                                                                          |
| 2.3             | Transactions of the Society of Actuaries, Vol. 1963, No. 2, pages 60 and 65.                                                                                         |
| 2.4             | Mortality by Industry and Cause of Death Among Men, 20 to 64 Years of Age. USA, 1950. Department of Health, Education, and Welfare, Vol. 53, No. 4, September, 1963. |
| 2.5             | Accident Rates and Fatalities, U.S. Certified Route and Supplemental Air Carriers, Total Operations 1949 - 1963, CAB Bureau of Safety, August, 1964.                 |



### III. MAINTENANCE AND REPAIR (M&R) CONSTRAINTS

#### 3.1 CONSTRAINTS CAUSATIONS

Given that failures and other types of emergencies will occur during long space missions, and that the mission duration is significantly long enough to require restoration of the lost function, all constraining factors must be properly identified and bounded. The constraints stem from an analysis of the functional flow logic required to accomplish the mission and perform the given task. Therefore, the resulting constraints may be classified by these sources.

The first class of constraints are imposed by the need for the function to assure mission success and/or to maintain life. These are termed down-time constraints, or maintenance time constraints (MTC). An example is the CO<sub>2</sub> removal function which may be said to be out of service on a typical planetary spacecraft for no more than 36.7 hours. Therefore, its MTC is 36.7 hours. This means simply that if the complete function fails, it must be returned to service within that time period to avoid disaster. In this case, loss of crew.

The second class of constraints are imposed by man's inability to cope with the emergency (by M&R action) as they occur and under the prevailing circumstances. For example, if the spacecraft is tumbling at a fast rate, the crew could do very little until it was stopped or slowed down. These are termed crew action constraints (CAC) since they describe the man-imposed constraints. These factors are evaluated in Section 4.0.

Each of these classes are explored in detail in the following paragraphs.

#### 3.2 DEFINITIONS

The most critical problem created by a failure is the loss of the associated function it provides to the mission. Maintenance or repair action is effective only so long as the mission can continue safely without the function, or if a backup function (or system) is available for use during the repair action. It is logical to assume that the function is required during some specific part of the mission, or a percentage of the total duration, randomly distributed throughout the mission. It is also just as logical to assume that



the mission can proceed in a degraded mode and perhaps only for a very short period of time. If this is true, that period can be used for both periodic maintenance and the necessary repair or replacement actions. The downtime constraint is therefore defined as:

A restriction imposed on the total allowable elapsed time a mission system function can be out of service before a situation is created that would result in ultimate loss of the mission spacecraft and/or crew.

It should be recognized that downtime constraints are not always described by a single value defining an all-black or all-white situation. Crew or function degradation may be gradual as in the case of CO<sub>2</sub> buildup, or, almost instantaneous, as would be the case at rapid decompression.

System functional downtime constraints are imposed by two separable mission design factors; the need to provide a habitable environment, and the need to meet the mission profile commitments. There is an obvious interface between some of these factors, but where they exist, the most stringent factor will be assessed and subsequently applied to the mission system design.

### 3.3 SPACECRAFT STABILITY OR AN ATTITUDE CONTROL

A profile constraint is created by both mission commitments because a stable and orientable platform is needed for navigation, guidance, and conduct of experiments; and because crew personnel are limited in their ability to tolerate random and repetitive motion.

The attitude control systems can fail in these modes:

#### A. Under Zero-g Phases:

1. Loss of the ability to activate the system or any part thereof, resulting in random drift.
2. Failure of engine control - resulting in a runaway engine and an accelerating spin creating motion in at least one plane.
3. Failure of the engine nozzle (burnthrough), resulting in vector misalignment.



### B. Under Artificial Gravity Phases:

1. Loss of ability to activate spin system results in loss of ability to change the spacecraft state; i. e. , go from artificial gravity to zero  $g$  , or from zero  $g$  to the artificial gravity mode.
2. Failure of engine control resulting in a higher  $g$ -level than that desired for the artificial  $g$ -mode - probably not excessive due to fuel restraints.
3. Runaway precession engine is most likely only during the artificial gravity mode. This will result in changing the spacecraft spin-axis vector by up to two degrees, as a maximum, prior to running out of fuel.

The failure modes have been evaluated to determine the potential down-time constraints associated with these function failures. These are given in Table 3.1, but some explanation as to the meaning and application is required. Since the spacecraft is expected to be operated in two basic mocks with respect to control system problems under zero  $g$  and with artificial gravity, the associated control system emergencies will be quite different.

#### 3.3.1 Under zero $g$

The spacecraft must maintain a given attitude through use of a stable platform, as a reference, and as a reaction control system for repositioning or stopping motion. The system must work constantly throughout the mission to prevent random drift. The tighter the "dead band" the more it works. Loss of the system function leaves the spacecraft with an unchanged course vector, but uncontrolled in attitude. It could remain in this condition without serious effects for a considerable portion of a mission, but communications, the heat shield, and reentry or aerobraking heat rejection capabilities may eventually be affected. It is considered probable that the spacecraft would eventually "weather-vane" into the solar winds and remain semistable.

If one of the reaction control jets failed, the roll rate would build up as indicated in Figure 3.1, the rate varying according to the axis involved; the axis of symmetry being the "worst case." If it continued unimpeded, it would eventually run out of fuel, as indicated by the "tankage limits." This would occur after no more than 900 seconds of burning time, and much sooner if the failure occurred late in the mission. If all of the possible resultants were plotted on the crew tolerance flat plot taken from Reference 3.1, the results would be as expressed in Figure 3.2. This analysis indicates that the crew can tolerate for an indefinite period any potential rotational rate, possibly resulting from an open reaction jet on the baseline spacecraft. This has been substantiated by considerable test data from Reference 3.2.



Table 3.1. Downtime Constraints, Spacecraft Attitude, and Stability Control Operations

| System Function/Mode                 | Maximum Allowable Downtime (hours)                      | Constraining Factors                                                                                                       |
|--------------------------------------|---------------------------------------------------------|----------------------------------------------------------------------------------------------------------------------------|
| A-1. Zero g, no stability control    | over 24                                                 | Loss of communications and heating balance. Random drift with astronaut movement and/or alignment with solar winds         |
| A-2. Zero g, Runaway reaction engine | At least 0.2 hours to neutralize; over 24 hours for M&R | Physiological limits and his subsequent inability to perform useful work                                                   |
| A-3. Zero g, no spin control         | Over 24 hours                                           | Uses more fuel and creates a use hazard                                                                                    |
| B-1. Zero g, no spin control         | Over 24 hours - probably none                           | Unable to establish artificial gravity mode, a personnel-imposed constraint                                                |
| B-1. Spin mode no de-spin control    | Over 24 hours, normally                                 | Time remaining to the required de-spin action, i. e., planetary or earth-approach phase                                    |
| B-2. Runaway spin engine             | 3.6 to over 24 hours                                    | Extension system design strength could be exceeded if it occurred early in the mission and the full fuel load was expended |
| B-3. Runaway precession motor        | None                                                    | Full fuel load will permit only a very small attitude change                                                               |

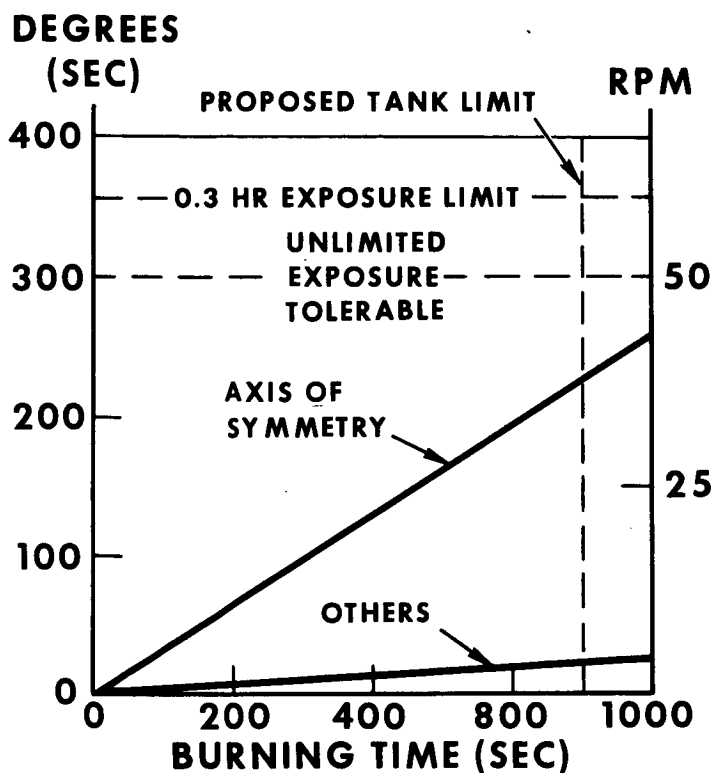


Figure 3.1. Spacecraft Rotation Rate as a Function of Reaction Engine on Time, With the Resulting Time Constraints

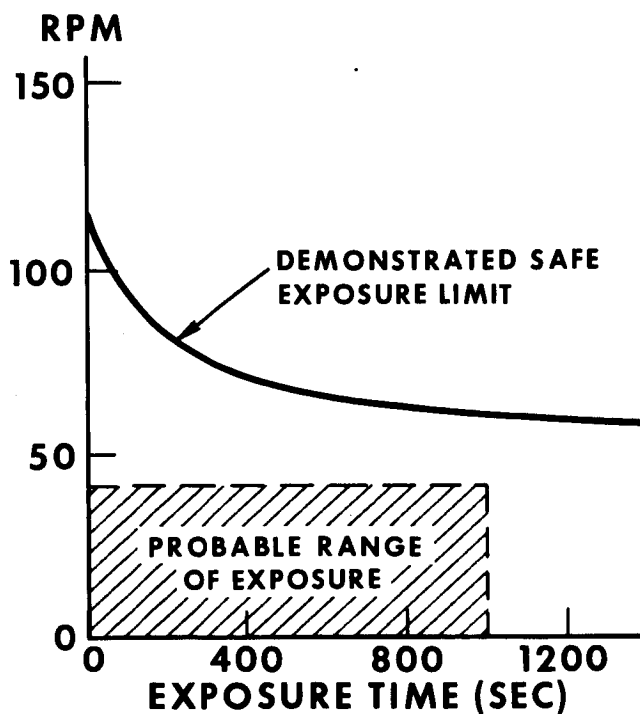


Figure 3.2. Crew Tolerance to Rotations and Potential Exposure for the Baseline Spacecraft



### 3.3.2 Under Artificial Gravity Mode

In this condition, the spacecraft will be extended and rotating, producing about one-third of the force of the earth's gravitational field. Under these conditions and after the rotation has been established, complete loss of the function will have little to no affect on the mission or personnel for the remaining coast phase. However, as either the earth or Mars is approached, the system is required to return to the zero-gravity mode because inability to do so would be disastrous. A runaway jet would only increase the simulated gravitational force. However, the larger gravitational forces could exceed the design margin for the cables connecting the two spacecraft modules, but since it takes 3.6 hours to build up a force equivalent to the earth's gravitational force, there is enough time to take whatever compensating actions are necessary. Further, during the final phases of the mission, the fuel reserve will be reduced considerably resulting in expenditure prior to overstressing the extension system.

### 3.4 SPACECRAFT VELOCITY CONTROL - A PROFILE CONSTRAINT

Midcourse Corrections are normally required at some time after transplanetary injection, and prior to arrival at periplanet. Loss of subsystem functions which preclude the course correction at a specific point in time may not be very critical if it can be returned to operational status. The data resulting from several studies and the Mariner program indicate that it could be deferred for up to 100 days. (See data in Figure 3.3.) These curves are representative of many of the flyby missions. The fuel penalties are expected to be less than 150 feet per second.

Periplanet Corrections are those made while approaching the maximum influence of the planet (point of closest proximity). The timing of this activity is most critical. As shown in Figure 3.4, this  $\Delta V$  correction must be made within about 7 hours to take maximum advantage of the velocity vector provided by the planet's gravitational field. Failure to make the approach correction will result in probable loss of the mission since the chance of a rendezvous with earth will be considerably lessened. Making it too early, prior to  $T - 10$ , may result in too large an error and required a second  $\Delta V$  application.

Earth Approach Velocity Corrections are those made as though spacecraft enters the earth's sphere of influence, before establishing its position relevant to the reentry corridor. This is an activity which is also critical to a degree in that it must be performed; however, there are about fifteen hours during which the task could be performed without appreciable penalty. Failure to make the correction could easily result in passing earth or burning up on reentry.

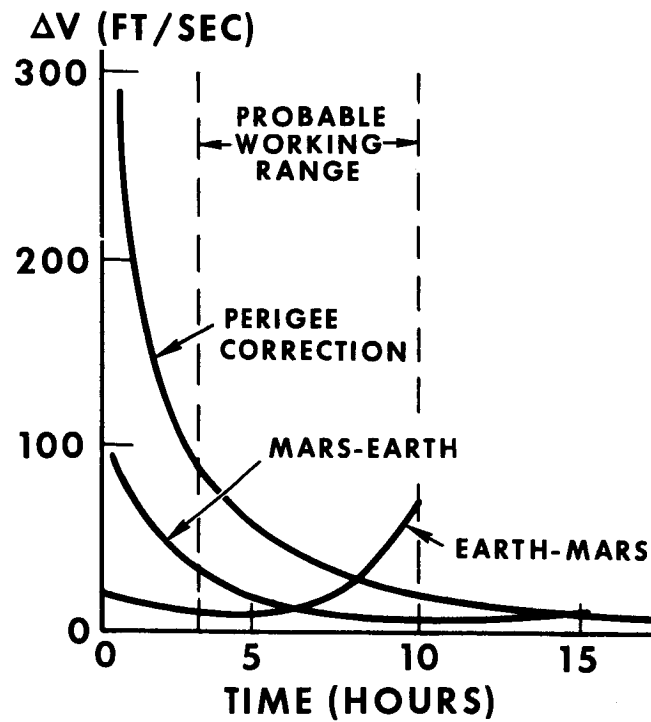


Figure 3.3. Approach Correction Sensitivities to Time of Application

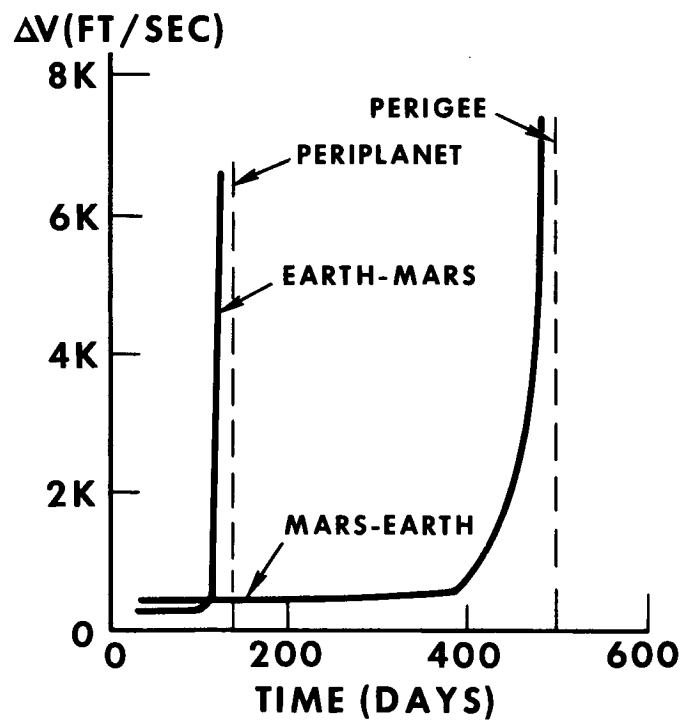


Figure 3.4. Effect of Delays in the Application of Midcourse Velocity Corrections



Table 3.2 Velocity Control Downtime Constraints Summary

| Mission Phase            | Downtime Constraint (Hours) | Constraining Factor                                                                       |
|--------------------------|-----------------------------|-------------------------------------------------------------------------------------------|
| 1. Midcourse             | (Duration)                  | Function not required                                                                     |
| 2. Midcourse corrections | 2400                        | Application of a Midcourse $\Delta V$ and associated fuel                                 |
| 3. Planet approach       | 7.0                         | Application of periplanet $\Delta V$ correction for proper return vector                  |
| 4. Earth approach        | 15.0                        | Application of approach $\Delta V$ correction to establish correct arrival vector         |
| 5. Earth retro maneuver  | 0.25                        | Retro must be initiated as close to a point 21 minutes prior to 400,000 feet as possible. |

Earth Retro Maneuver is required about 400,000 feet above the surface as the spacecraft approaches the earth's atmosphere. The retro force is most efficient if it is made close to the start of sensible atmosphere. But enough time must be allowed to jettison the propulsion module and reorient the reentry Command Module for the reentry phase. It is estimated that seven to ten minutes are required to establish the desired attitude with respect to the entry corridor. Thus, fifteen minutes has been allowed for this operation. In addition, the retro burn will take from five to six minutes. As a result of these requirements, the retro should start at atmospheric entry minus twenty-one minutes. For safety sake it could be initiated earlier and delayed somewhat later with some associated penalties in fuel and risk. Therefore, it is estimated that a 15-minute period exists where the retro could be initiated.

Earth Entry phase requires the uninterrupted operation of all associated system function; thus, a very narrow corridor must be acquired and maintained by constant vernier control. Any downtime on the part of any associated function could easily result in disaster.



### 3.5 ATMOSPHERIC PRESSURE CONSTRAINTS

Atmospheric pressure is required to make the spacecraft habitable to man. Pressure below about 3.0 psia will eventually result in death because body fluids vaporize at pressures below about 1.5 psia. The minimum safe boundary is therefore about 3.0 psia; however, at this pressure, man must be provided pure oxygen for breathing. Limited exposure to vacuum may be possible, according to U.S. Air Force tests.

Failures in, or loss of the atmospheric control system are covered under either the N<sub>2</sub> or O<sub>2</sub> subfunctions. However, an additional failure mode constraint is created by the potential rapid decompression hazard. This hazard is created by a large meteoroid impingement or by the blow-out of some form of access hatch or relief system which results in inadvertent release of cabin pressure.

Under any of these failure modes, the time to catastrophe cannot be rigidly set at any specific value since it is a direct function of hole size. This relationship is demonstrated in Figure 3.5, where the time relationship for hole size and decompression time is estimated. However, since the Mars mission spacecraft cabin volume has been estimated from present and former studies as between 2400 and 4500 cubic feet, one boundary can thus be estimated. In addition, the upper boundary (0.999 probability of one, or less than a one-inch hole) can be estimated. From these data, it can be stated with reasonable assurance that the safe time constraint is not less than about 720 seconds for a 400 ft<sup>3</sup> cabin.

These data are replotted in Figure 3.6 to express the time and hole-size relationship for the probable Mars Cabins.

### 3.6 OXYGEN SYSTEM CONSTRAINTS

The oxygen system is designed to provide for the metabolic needs of the astronauts. In addition, the system provides a portion of the atmospheric pressure. For the flyby mission, O<sub>2</sub> will probably be mixed on a 50/50 basis with nitrogen which is to be used as a diluent. The total pressure is normally expected to be 7 psia, and the oxygen partial pressure about 3.5 psia.

Loss of the oxygen system functions can result in loss of the balance control between O<sub>2</sub> and N<sub>2</sub>, or, it can result in the inability to feed O<sub>2</sub> to the cabin and/or crew. Both of these factors impose potential constraints on the allowable time the functions can be inoperative.

Loss of O<sub>2</sub> balance or regulation can cause the O<sub>2</sub> partial pressure to either increase or decrease with respect to the total cabin content. Man is very flexible in his tolerance to wide variations in O<sub>2</sub> partial pressure as

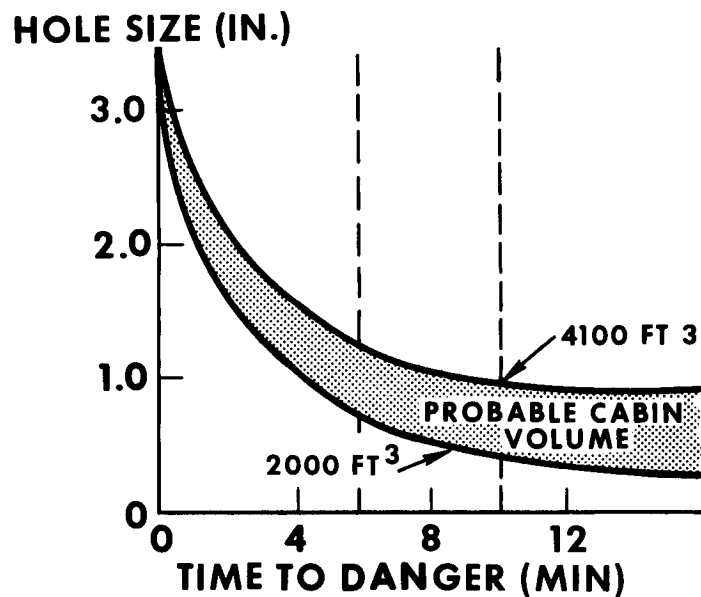


Figure 3.5. Rapid Decompression Time as a Function of Cabin Size and Hole Size, Without Flood-Control Compensation

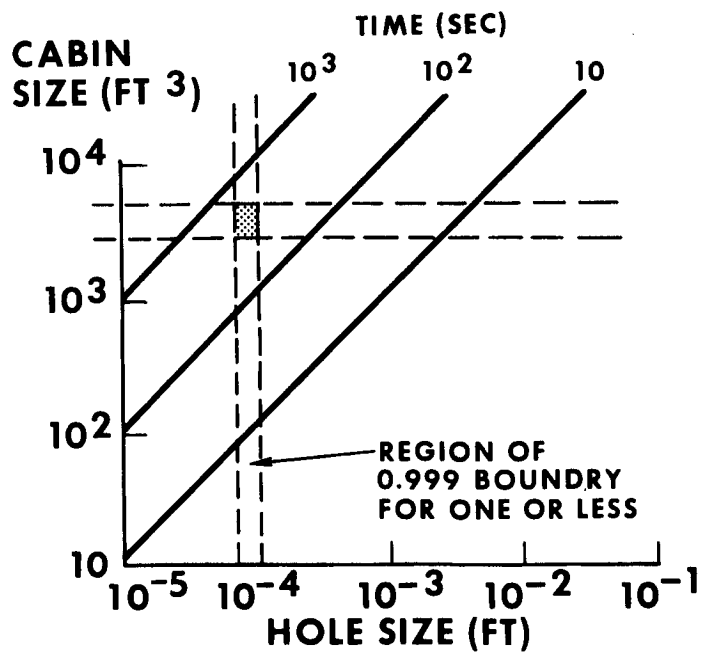


Figure 3.6. Time Relationship of Hole Size and Decompression Time for Astronaut Safety



shown by the data in Figure 3.7. Therefore, if the pressure builds up within the cabin, the crew is safe for at least 24 hours at a pressure nearly 40 psia and a 50/50 mixture, before performance degrades due to oxygen toxicity. If the partial pressure (or mixture ratio) changes, it could increase to 100-percent  $O_2$  and remain there indefinitely if the total pressure is held below about 7 psia.

Since a failure in the mixture control will not affect the volume of  $N_2$  present, the  $O_2$  partial pressure would probably progress as shown by the  $O_2$ -overpressure curve. Under these conditions, a form of  $O_2$  flooding, the concentration could be raised by a factor of over 5 to a pressure of over 21 psia (85 percent  $O_2$  by volume) and the crew would survive for over 24 hours without noticeable degradation in normal performance. Over-exercise may result in mild toxicity earlier.

Loss of the  $O_2$  feed system where the  $O_2$  is not replenished would seem more hazardous, but the data does not support this position. If it is assumed that the cabin air consisted of about 50-percent  $O_2$  and 50-percent  $N_2$  at the time the loss occurred; and further, that the cabin volume was between 700 cubic feet per man (2800  $ft^3$  total) and the 4510 cubic feet free volume projected for the Mars Flyby Spacecraft (Ref. 1.1), the situation would progress as depicted in Figure 3.8. These data were based on the following assumptions as projected for the Mars Flyby Mission:

1. The leakage ratio is about 1.1 cubic feet per hour at 7.0 psia, or about 3.55 pounds per day.
2. Metabolic consumption is about 0.74 cubic feet per hour at 7.0 psia for each crew member.
3. There were four crew members working at a normal rate and under normal temperature.
4. Hypoxia sets in at an  $O_2$  partial pressure, below about 2.8 psia, or after a 20-percent reduction in the available  $O_2$  under the 50/50 mixture at 7 psia.

These data indicate that the  $O_2$  feed system could be inoperative and down for repair for a period between 150 and 225 hours, depending on the cabin size. This could be reduced by such factors as higher leak rates, increased work activity, and high temperatures. But, it could be increased if the  $CO_2$  removal system continued to operate and/or the emergency supplies were used.

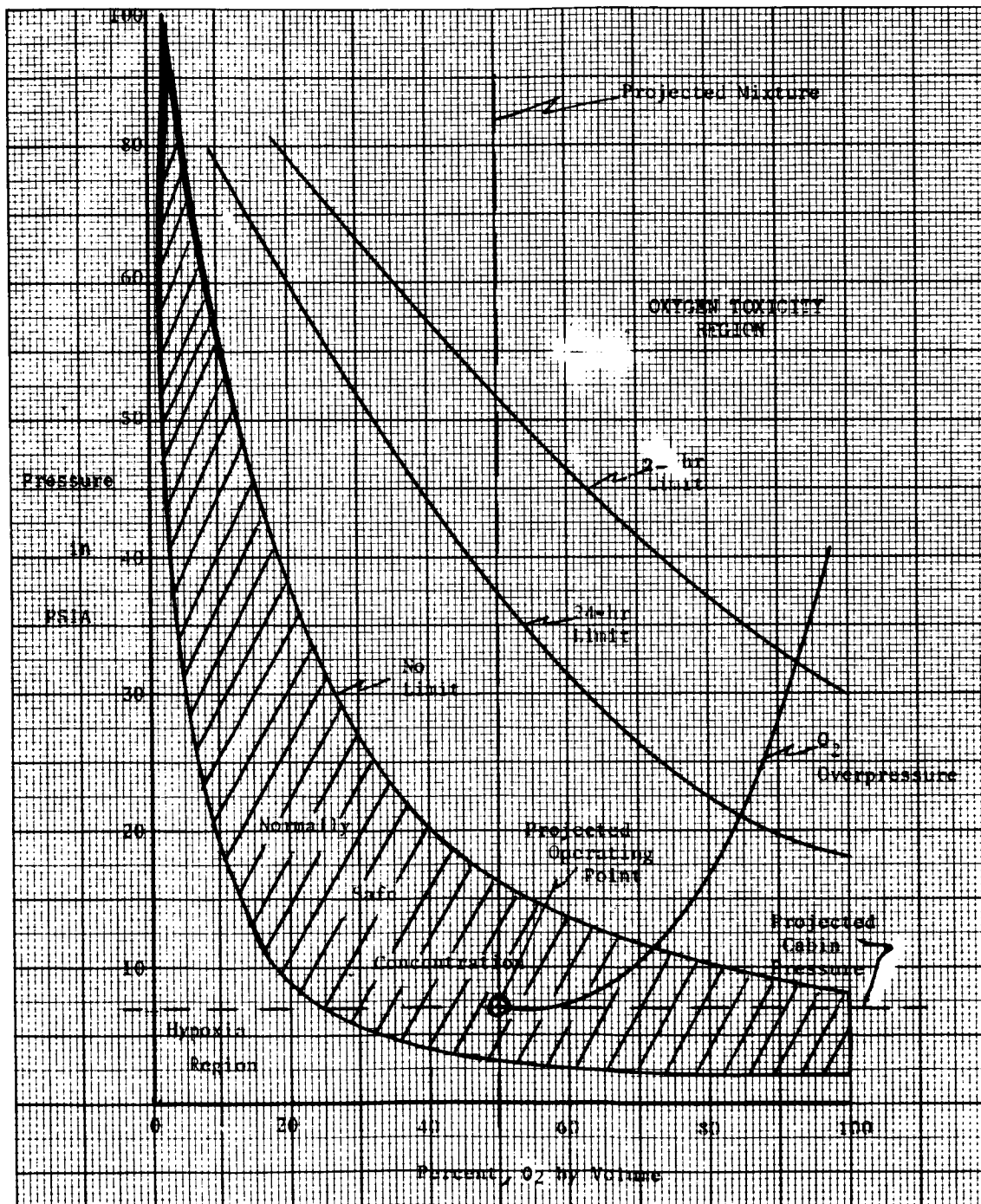


Figure 3.7. Man's Sensitivity to Oxygen Pressure and Diluent Ratio

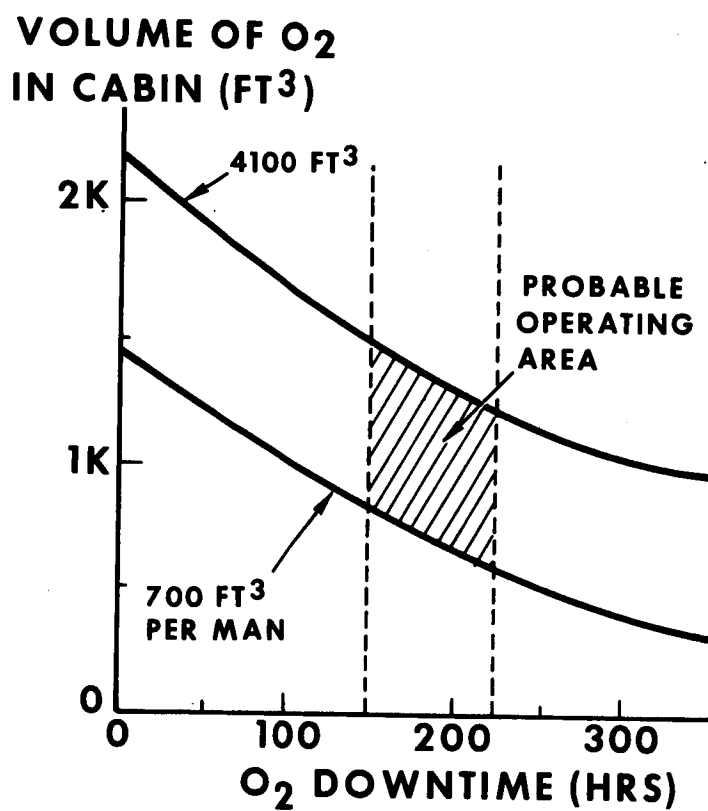


Figure 3.8. Effects of Loss of Oxygen System on Remaining Partial Pressure and Crew Safety



### 3.7 NITROGEN SUPPLY CONSTRAINTS

The nitrogen gas is used as an atmospheric diluent. In the systems proposed for planetary flyby missions, nitrogen is mixed at about 50/50 with the O<sub>2</sub> supply at a pressure of 7.0 pounds. If a failure occurs in the N<sub>2</sub> system where it will no longer provide the diluent, the gas will eventually leak out at a rate estimated to be about 4.29 pounds per day.

The question of tolerance to the N<sub>2</sub> system outage is not clear cut. Without replenishing, it will gradually be lost through leaks, at a rate approximated by Figure 3.9. It could also be made up by building up the O<sub>2</sub> partial pressure at an equivalent rate as shown. A second alternative is to allow the total pressure to drip as N<sub>2</sub> leaks out. If it all leaked out, the resulting pressure (constant temperature) would be 3.5 psia-the lower safe limit. However, a better alternative is to plan starting the O<sub>2</sub> makeup at an absolute pressure of about 5 psia, as depicted in the center curve to minimize the total O<sub>2</sub> loss.

Under any of these circumstances, the allowable downtime would be no less than about 1360 hours, constrained by the allowable O<sub>2</sub> loss only.

### 3.8 CARBON DIOXIDE SYSTEM CONSTRAINTS

The carbon dioxide (CO<sub>2</sub>) system functions to remove the CO<sub>2</sub> from the cabin atmosphere and to keep it below a partial pressure of 4 millimeters of mercury or 0.11 percent by volume. The normal atmosphere contains only about 0.03 percent. Man's average daily output is between 1.76 pounds or 0.61 cubic feet per hour, and 2.25 pounds per hour, due to normal metabolic functions. Under stress and high activity, this will increase considerably.

Extensive tests and operational situations have provided data which relates man's performance under various conditions and CO<sub>2</sub> concentrations. There seems to be no sharp line between the safe zone and a fatal dose. Rather, performance is gradually degraded until the crewman becomes unconscious and eventually dies, perhaps due to other allied causes.

The problems associated with loss of the CO<sub>2</sub> reduction system functions and the subsequent buildup are related in Figure 3.10 where the buildup rate and the critical points are estimated for both the 700-cubic-foot-per-man spacecraft and the Mars flyby spacecraft (4510 cubic feet free volume). These data are based on the higher buildup rate (2.25 pounds per hour), an estimated cabin leak rate of 1.0 pounds per day, and an ambient temperature of 75 F. In addition, the cabin atmospheric content was about 50/50 of O<sub>2</sub> and N<sub>2</sub>, with between 3.5 and 4.0 mm of Hg of CO<sub>2</sub>.

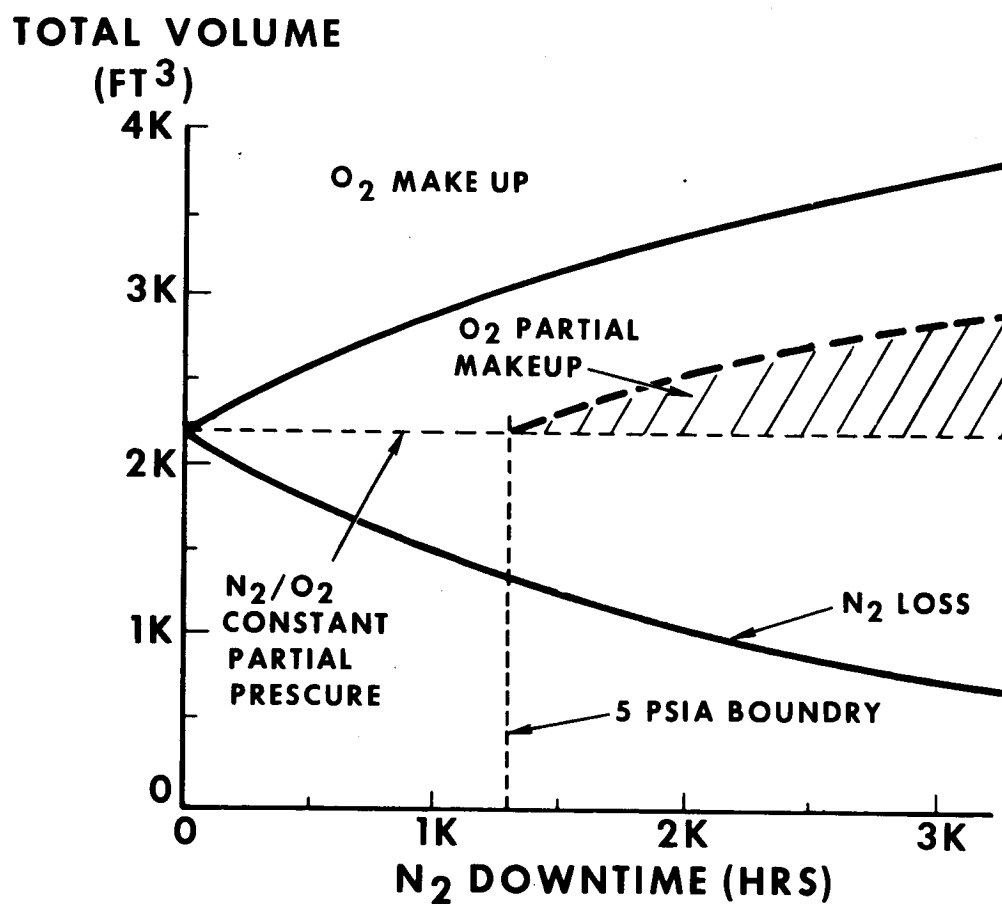


Figure 3.9. Loss of Nitrogen System on Cabin (Pressure) Volume and Crew Safety



The data indicate that the complete CO<sub>2</sub> removal function can be out of service for up to 136 hours with little perceptible degradation in performance, even in the smaller 700-cubic-foot-per-man cabin. After this, the crew members would probably develop headaches and slowly become less and less efficient. After about 300 hours they may become unconscious and eventually die. Therefore, it can be said that a reasonable downtime constraint for the CO<sub>2</sub> removal function need not be any less than the 136 hours, and could possibly be 220 hours, depending on the final free volume of the mission module and its associated leak rate.

### 3.9 TRACE CONTAMINANT CONTROL CONSTRAINTS

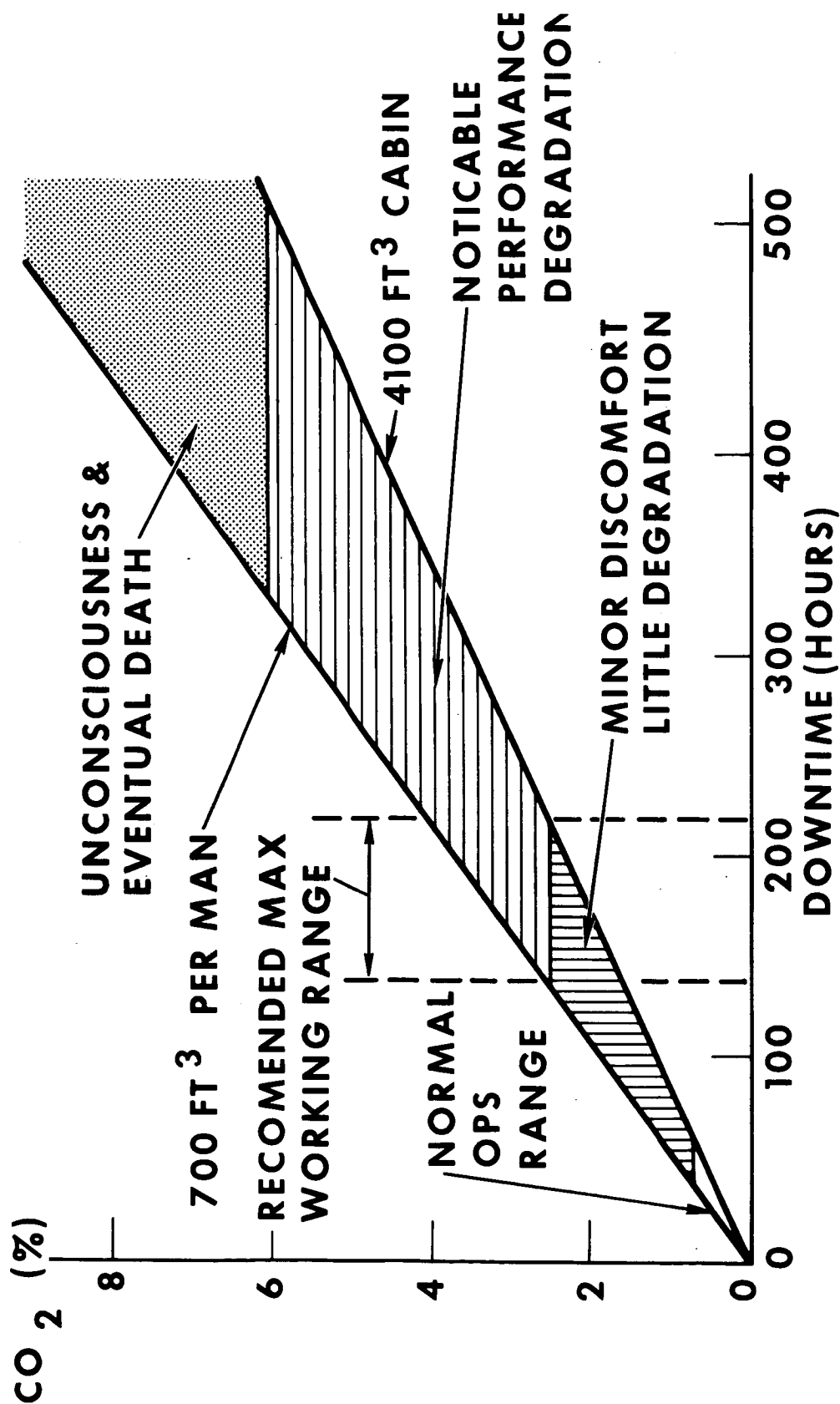
Trace contaminants are manifest in the form of gases, vapors, aerosols and particulates, and they stem from two major sources — the equipment within the spacecraft, and the human body. The body is considered the greatest source of these contaminants via the metabolic process. Under actual tests, approximately 100 contaminants have been isolated within the spacecraft cabin, but neither the buildup rate or the toxicity of the individual contributors have been identified.

If the trace contaminant removal function fails and is down for any length of time, they will probably reach a point of equilibrium, probably above the toxic level. However, the buildup rate is known to be so low that, with caution in the material selection, it will take days for the levels to reach a toxic state. It is most certainly expected to be less than that associated with CO<sub>2</sub> removal, which can therefore provide an estimate of the lower boundary, approximately 136 hours.

### 3.10 TEMPERATURE CONTROL CONSTRAINTS - EQUIPMENT

The equipment temperature control function is required to provide coolant through equipment heat sinks in the form of cold plates, and to remove the excess heat produced by a nuclear or radioisotope electrical power source. Of the two functions, the power-source cooling is most critical and will probably form the limiting case. Since most designs presently considered provide a separate cooling loop for the isotope power source, this will be assumed to be true for this analysis, with both treated as two loops, and therefore, as two separate constraints.

3.10.1 The isotope coolant loop is most sensitive to cooling system outages. Since the power conversion systems used in conjunction with the isotopic sources are not very efficient and vary between 18 to 27 percent, the remaining heat generated must be dissipated. All extended mission plans project the use of a space radiator and an associated active coolant loop for this purpose. In addition, and because the active cooling is lost prior to the

Figure 3.10. Crew Performance as a Function of CO<sub>2</sub> Concentration



reentry phase, a large heat sink is provided to absorb the heat during this phase and up to splash down. At that time, the compartment is flooded with sea water for cooling, until it can be deactivated.

Two or three basic conversions with widely varying efficiencies are under consideration. Each system design will have a heat sink designed to limit the isotope capsule to about 2200 F, 66 minutes after loss of the normal cooling loop. The capsule time - temperature history of the Brayton and Mercury Rankine systems are shown along with a typical heat sink (beryllium) in Figure 3.11. Most isotope heat sinks will probably be sized to reach the 2200 F in no less than 66 minutes. However, because of the inherent design margin, the temperature could continue to rise for about another 14 minutes for a total of 80 minutes. The heat sink will melt within the next 10 minutes if the coolant loop is not restored to operation. The maximum downtime for the isotope cooling loop lies therefore between 66 and 80 minutes.

### 3.10.2 Temperature and Humidity Constraints - Personnel Imposed

The cabin temperature and humidity are controlled to assure a habitable and comfortable environment for the crew. Further, since crew performance is affected by these factors, they are regulated to achieve an optimum environment conducive to peak crew efficiency. Factors which affect the temperature and therefore the relative humidity are:

1. Each crew member produces an average of 11,200 Btu's per day.
2. Heat lost through the wall of the spacecraft, as estimated in a former study, (Ref 1.1) was about 5,320 Btu per hour at 1.52 au (Mars vicinity) and 2,870 per hour at 0.58 au.
3. Heat may be gained from radiant energy of the sun which varies drastically with spacecraft orientation and distance from the sun. Data in (2.) assumes a ratio of 3 to 1 of the exterior surface exposed to space versus that exposed to direct solar radiation. In emergencies, this may be changed to a degree by reorienting the spacecraft.

The data indicates that cabin temperature is quite sensitive to the loss of the air-control function. A total of about 9551 Btu's is being introduced into the cabin every hour by the mission system equipment alone. This, added to the 1870 Btu produced by the crew, would elevate the cabin air by 37 F per hour at 1.52 au and up to 51 F per hour at 0.58 au without the cooling function. These represent the extreme cases resulting from being in farthest or closest proximity with respect to the sun at the time the loss

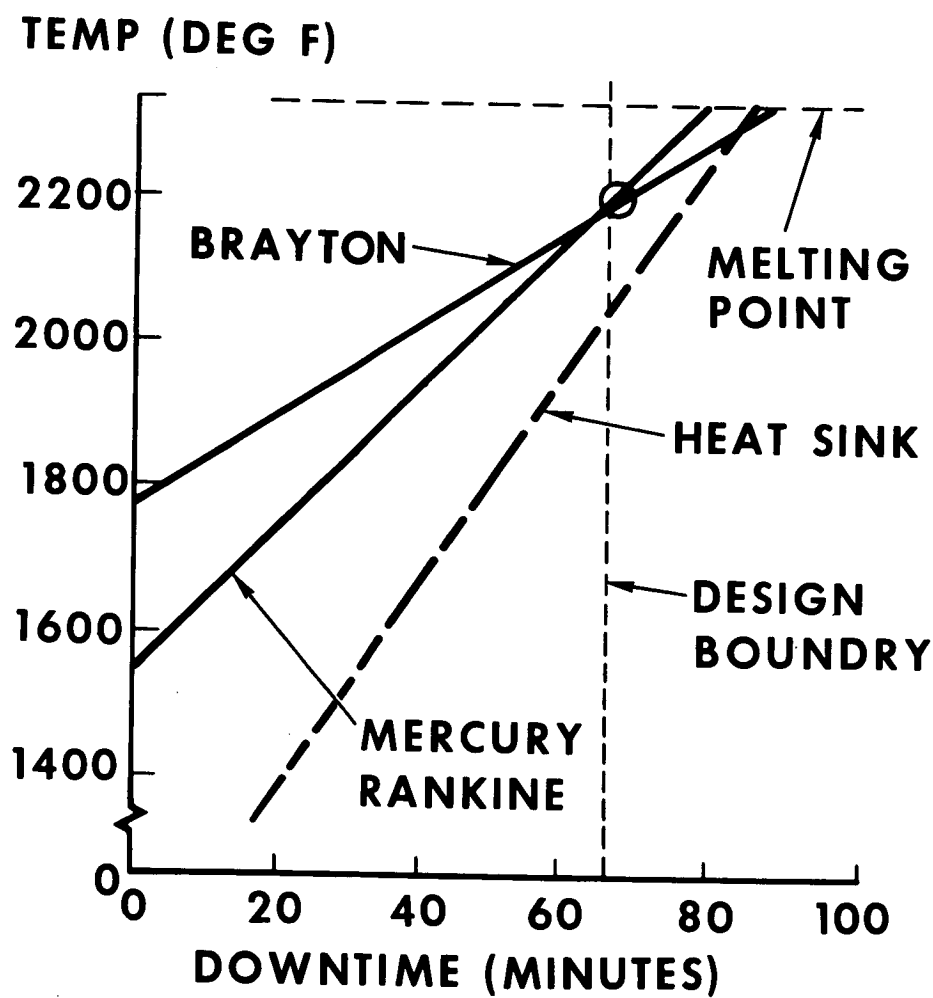


Figure 3.11. Temperature History for Isotopic Power Sources After Loss of Cooling Loop Function



occurs. The most likely value is therefore somewhere between both extremes. These boundaries, along with the results of various other intermediate conditions, are plotted as a function of time in Figure 3.12.

The analysis indicates that a minimum of only one hour is available for repairs that affect cabin cooling if, and only if, all other equipment is operated normally. About three hours is available if only the environmental and life support system functions are operated. Notice that thermal equilibrium can be maintained at the point closest to the sun, for a nearly indefinite period of time by just operating the two cabin air fans. Manual control can be achieved by operating the required equipments in a cyclic fashion. Therefore, because of the wide latitude of manual control available to the crew, the resulting downtime constraint is not less than one hour and could be extended to almost any reasonable length of time - certainly over 24 hours.

Because of a potential humidity condensate problem, a major consideration is the need to sustain in operation the temperature and the humidity control function.

### 3.11 HUMIDITY CONTROL CONSTRAINTS

Humidity control and atmospheric temperatures are directly related. Humidity condensate in the air, walls, or equipment, could be very deleterious to total mission. For that reason, close control is required. The design goal is to maintain humidity between 35 and 75 percent. Since each man exhausts 3.75 pounds per day into the air, the concentration will build up rapidly if the dehumidifier is out of service.

Figure 3.13 presents the cabin atmospheric humidity condition as a function of time for the 700 ft<sup>3</sup>-per-man-volume and the proposed 4510 ft<sup>3</sup> Mars flyby cabin. Since the humidity can be anywhere between 35 and 75 percent at the start, the potential available downtime is a variable resulting from these two extremes. The dehumidifier may work in cycles starting up at 75 percent, and stopping at 35 percent. If this is the case, initial conditions at failure can be anywhere, in between, on a random basis. Under no conditions should the atmosphere be allowed to achieve 100 percent humidity. The temperature can be assumed to hold comparatively constant at 70 F and the cabin pressure at about the 7 psia.

Given the preceding conditions, the allowable downtime for the 700 ft<sup>3</sup> could be as low as 1.6 hours, but no more than 3.8 hours. For the 4-man Mars flyby cabin this would be not less than 2.7 hours, or more than 6.1 hours. By elevating the ambient temperature from 10 degrees to 80 degrees F, an additional 2.7 hours can be added to any of the stated values. However, since the starting point is randomly distributed, this is

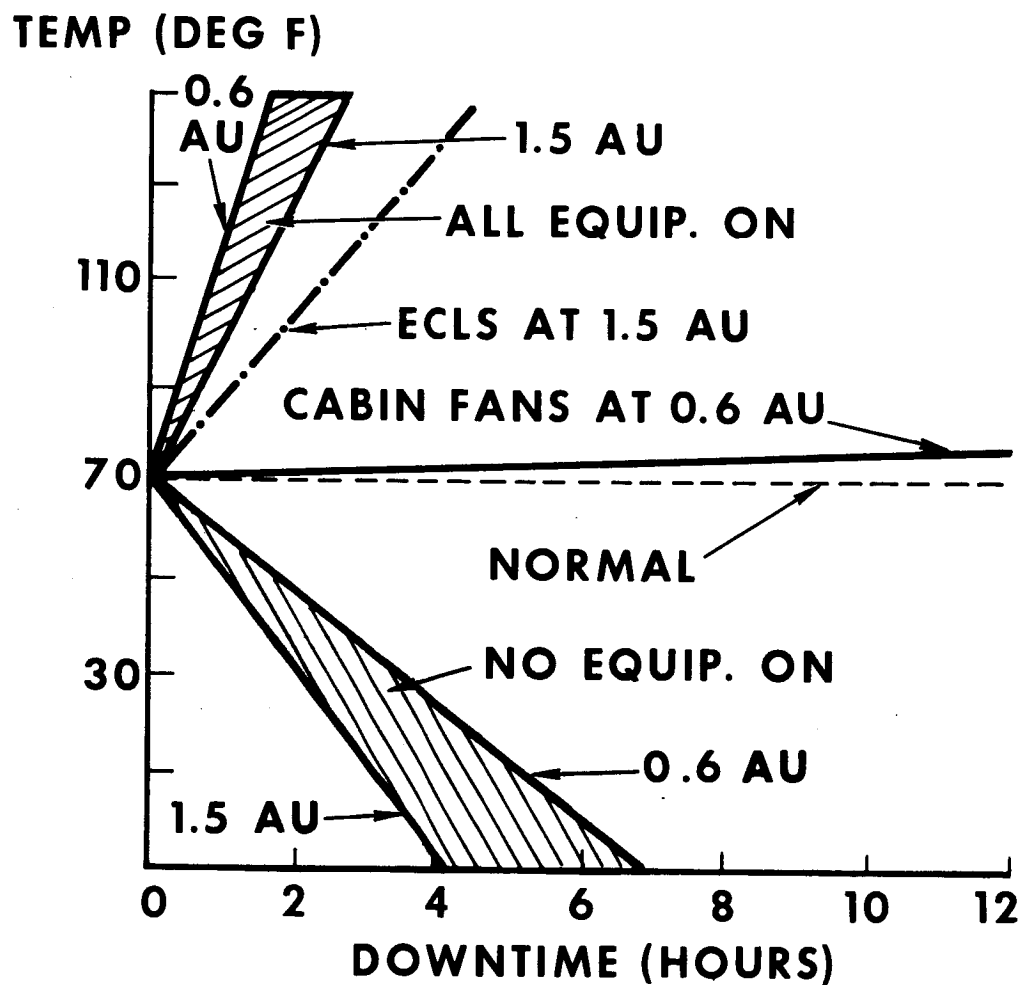


Figure 3.12. Cabin Air Temperature History as a Function of Spacecraft Position and Systems Operating, After a Loss of Temperature Control Function

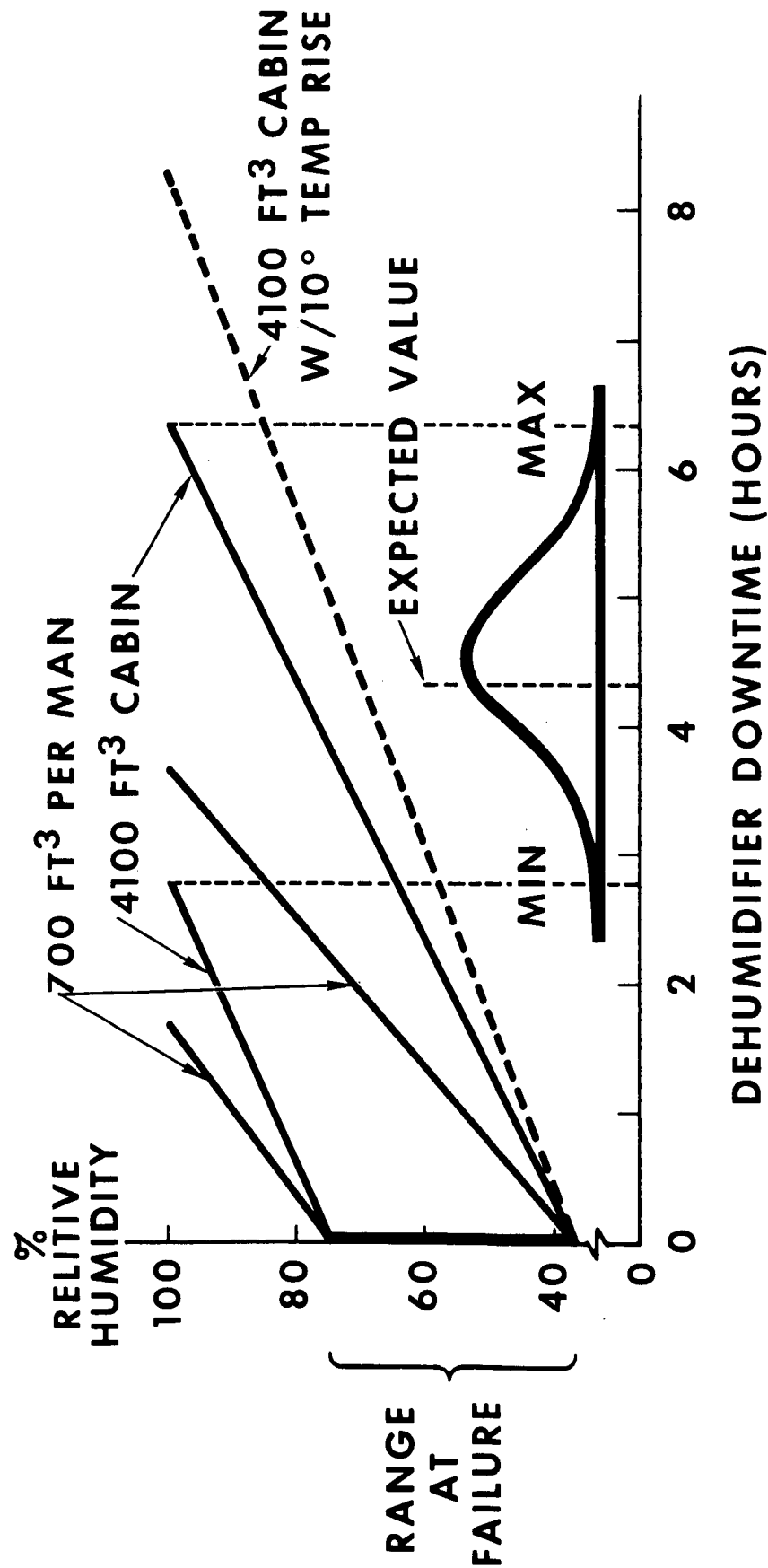


Figure 3.13. Mission Module Humidity as a Function of Dehumidifier Downtime



also true of the actual downtime constraint. Using the normal distribution, the most likely value would be about 4.5 hours, or about 4.0 hours for 90-percent of the potential failure incidents.

By virtue of the foregoing reasoning, 4.0 hours is taken as the potential downtime constraint for the humidity control function.

### 3.12 REFERENCES FOR DOWNTIME CONSTRAINTS

#### Ref. No.

- 3.1 Bioastronautics Data Book, NASA SP3006, 1964.
- 3.2 Preliminary Definition Phase Study Environmental Control System Apollo Extension System, AiResearch Mfg., SS-3942, November 1965.
- 3.3 Analytical Methods for Space Vehicle Atmospheric Control Processes, ASD Technical Report 61-162, Contract (AF 33 6616) -8323, July 1962



#### IV. CREWMAN CAPABILITIES AND CONSTRAINTS

##### 4.1 DATA, SCOPE, AND APPLICATION

For the crew to perform a given maintenance or repair task, it must be shown that the associated subtasks are within his performance envelope, in terms of the conditions under which the task must be accomplished. To assure that these contingent conditions can be accommodated, the crewman's performance envelope must be clearly defined for any potential condition a M&R task may be expected to impose. This section is devoted to defining that envelope as far as the available data will permit.

A survey of data available on human strength capabilities disclose a conspicuous lack of sound data which adequately define the effects of the space environment on man's ability to produce useful work. The force and dexterity decrements resulting from the need to wear a full pressure suit (pressurized or unpressurized), together with the reduced or zero-gravitational forces can be significant. Considerable data does exist on the force producing capability of various segments of the population under normal conditions; that is, 1-g and shirtsleeve environments. These data employ body configurations and orientations which are typical for the performance of tasks which were normal for the 1-g condition, but may not prove optimum for similar tasks in zero or reduced gravitational fields and/or in a pressure suit.

These data do provide a performance index which, through careful application of the available space environment data, can provide boundaries for the crew performance envelope under the expected working conditions. An optimized restraint system, which will satisfy Newton's third law, can facilitate application of nearly the same force vectors in reduced gravity as under the 1-g situation.

The data used in this section is the result of an extensive literature search and studies conducted by the Life Sciences Group of S&ID. Much of the data was derived from maintenance task simulation accomplished through use of S&ID's six-degrees-of-freedom simulator, the lightest one in existence at the time of this report (Figure 4.1). Much of the S&ID data has not yet been reduced to report form.



#### 4.2 FORCE PRODUCTION IN ONE G VERSUS ZERO G

The following data was derived through tests on man's capability to perform work in the normal earth environment and under various positional constraints. These values represent the upper limits of his work envelope. They are nearly applicable to the zero-g spaceflight situation when:

1. The worker is restrained in such a manner as to apply a reactive forces vector equivalent to that of the earth's gravitational field.
2. The work is to be performed within a pressure shell where sufficient atmospheric pressure can be supplied to permit working in the equivalent of a shirtsleeve environment.

The data presented in Figures 4.2 through 4.7 are applicable to any maintenance and/or repair operation that can be accomplished from a seated or prone position without respect to gravitational force so long as the worker is restrained by a simple belt across his loins. (Ref. 4.1).

Figures 4.2 and 4.3 present the measured mean arm strength  $\pm$  one standard deviation, at varying degrees of elbow flexion for a seated worker in 1-g or reduced gravity, with a conventional seat belt.

Figures 4.4 through 4.7 present the resulting mean arm strength,  $\pm$  one standard deviation, at varying degrees of elbow flexing for an operator in a prone position. The data G is for the one-g case, or, where the worker is or restrained at the approximate c.g. by a simple strap.

Figures 4.8 through 4.10 present data derived from several S&ID and Boeing studies (Ref. 4.2). Figure 4.8 presents the derived workman capability to exert a torquing force on a control wheel and/or valve control as a function of the working conditions, and the actuator wheel size. Note that when the work surface is horizontal with respect to the worker, and at the knees or below, he cannot produce any significantly useful work. In addition, under the zero-g situation, it is impractical for the worker to attempt an activity of this nature while the work surface is horizontal with respect to his position, since he cannot apply an isometric force. This would change with proper restraints, but his area of capability would be severely limited. With the work surface vertical, or parallel with his body, he can be very effective, although under zero g, one-half of the force would be required to satisfy Newton's third law. The values are 10-percent less for counter-clockwise motion.

Figure 9 presents the worker's torque producing capability through a six-inch moment arm, equivalent to that of a six-inch ratchet wrench. Where two hands can be used, the values may be increased by a factor of 2.

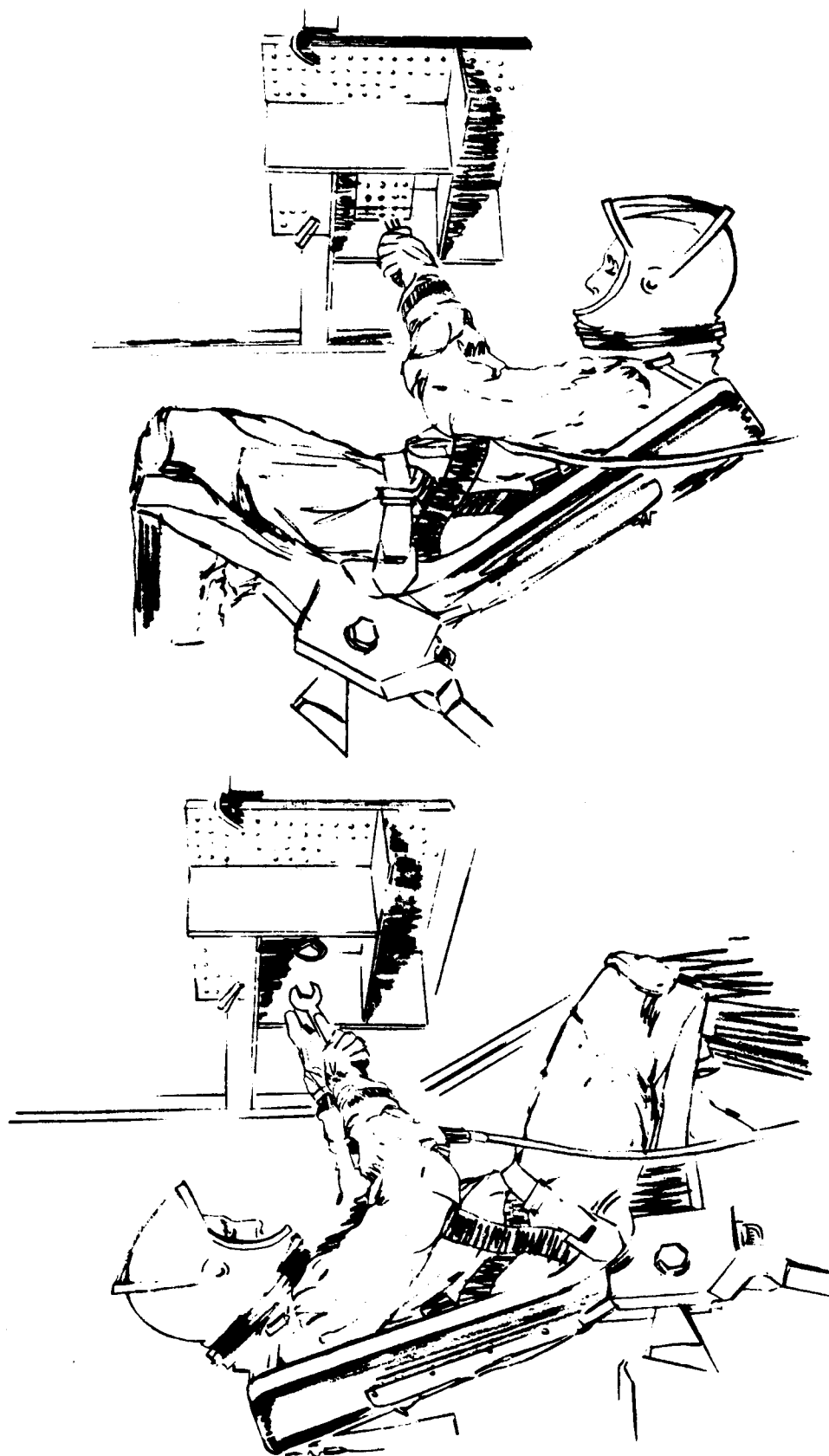


Figure 4.1. Applications of Six-Degrees-of-Freedom Simulator

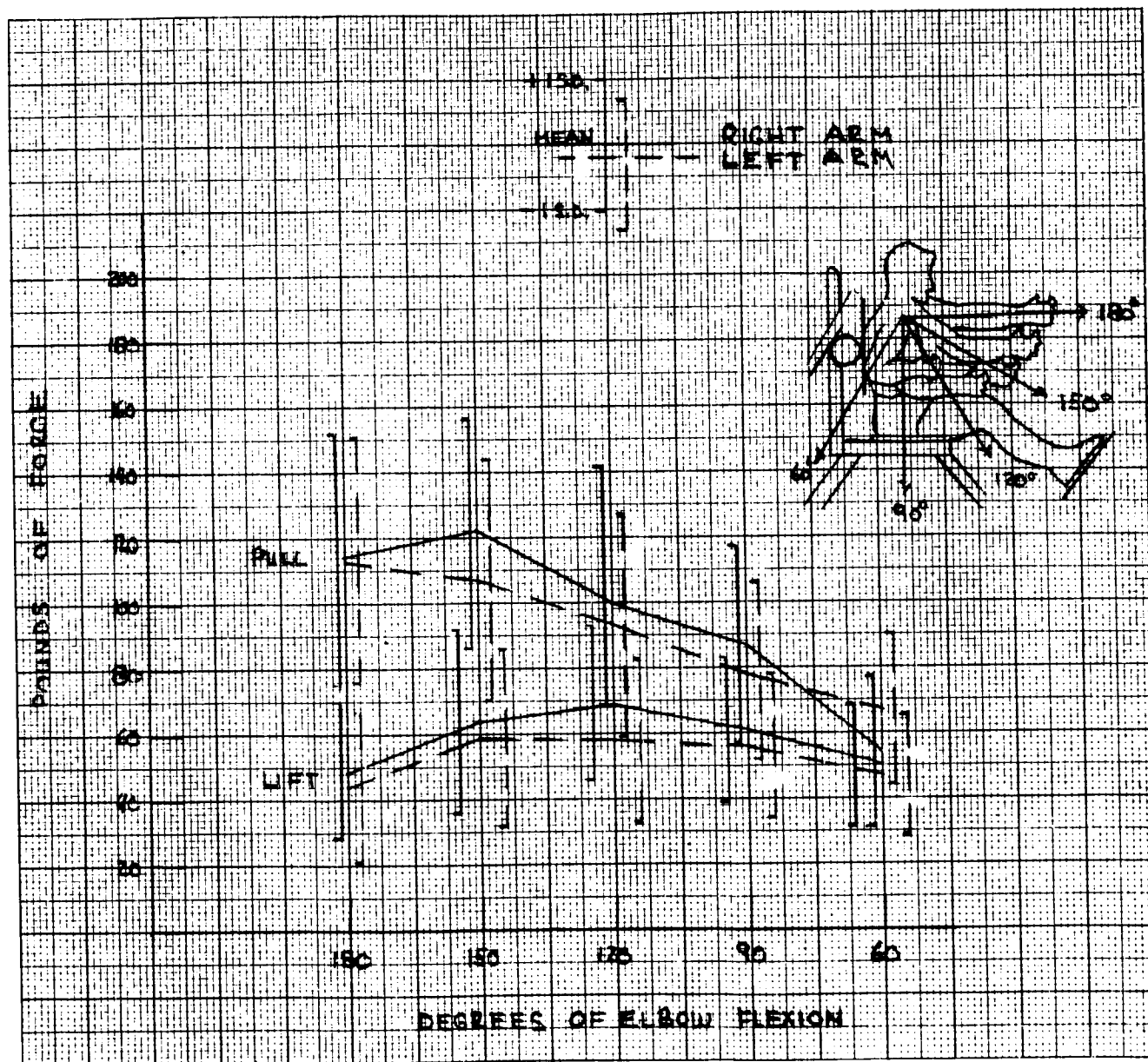


Figure 4.2. Arm Strength at 1 G

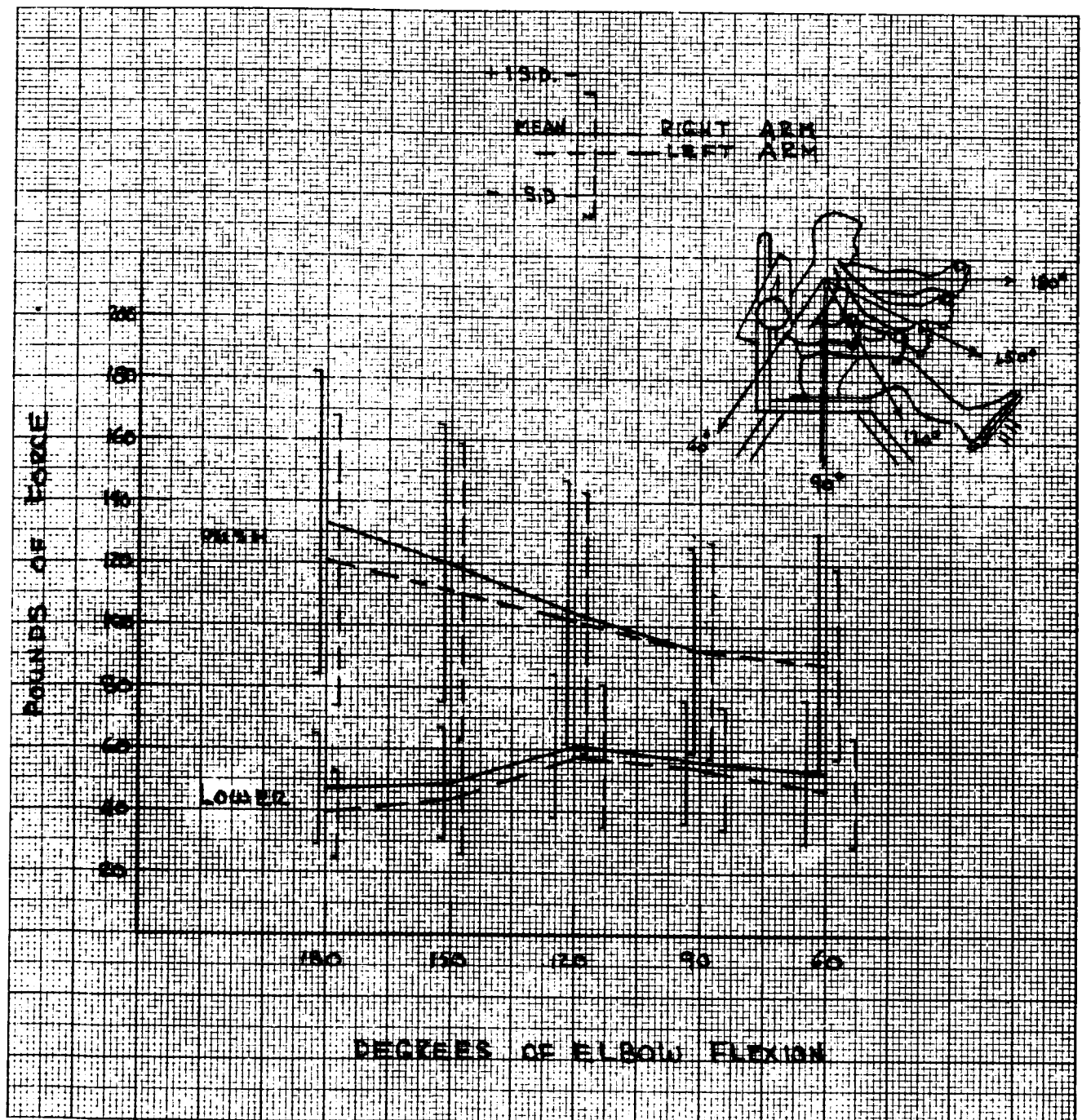


Figure 4.3. Arm Strength at 1 G

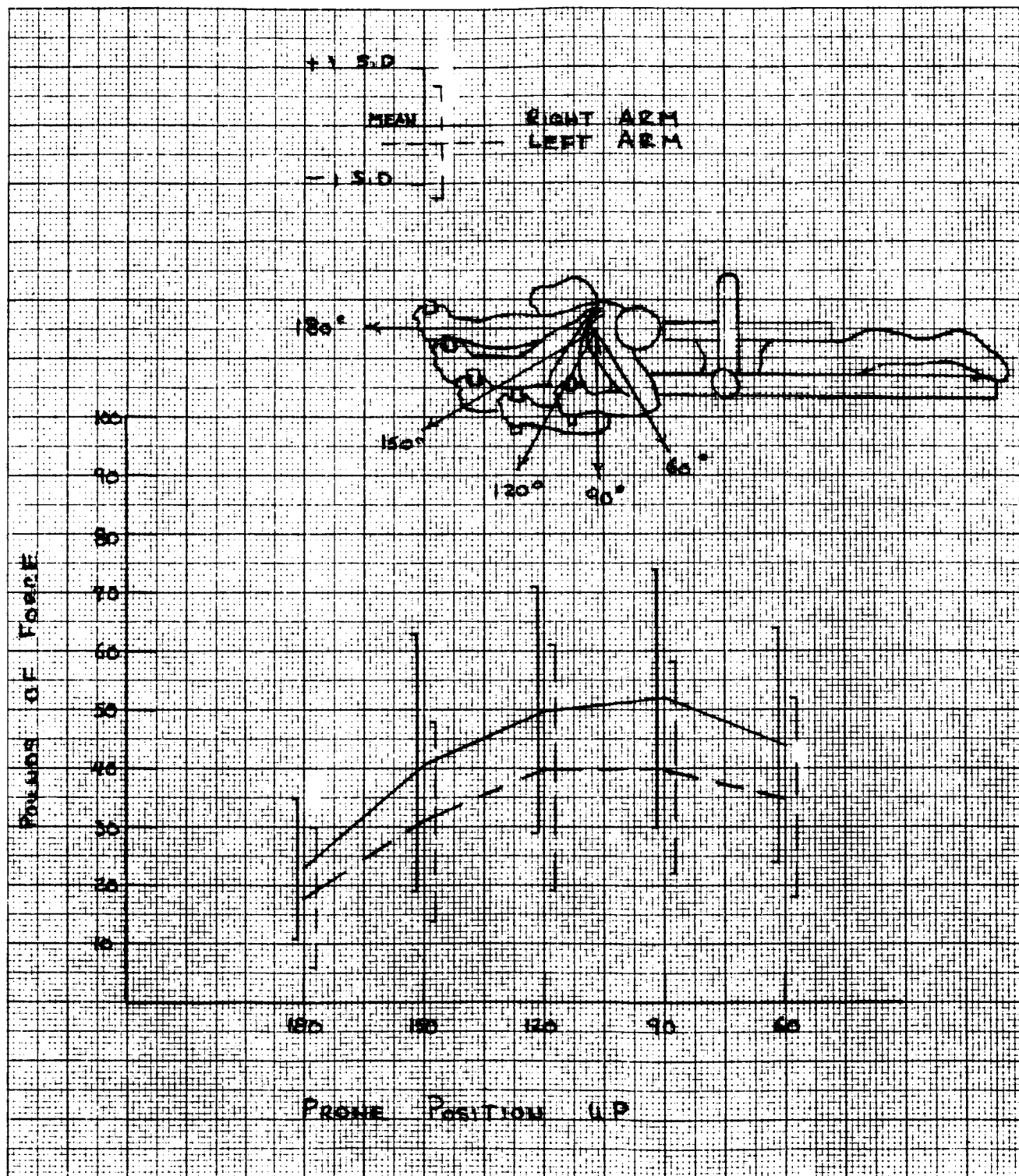


Figure 4.4. Arm Strength at 1 G

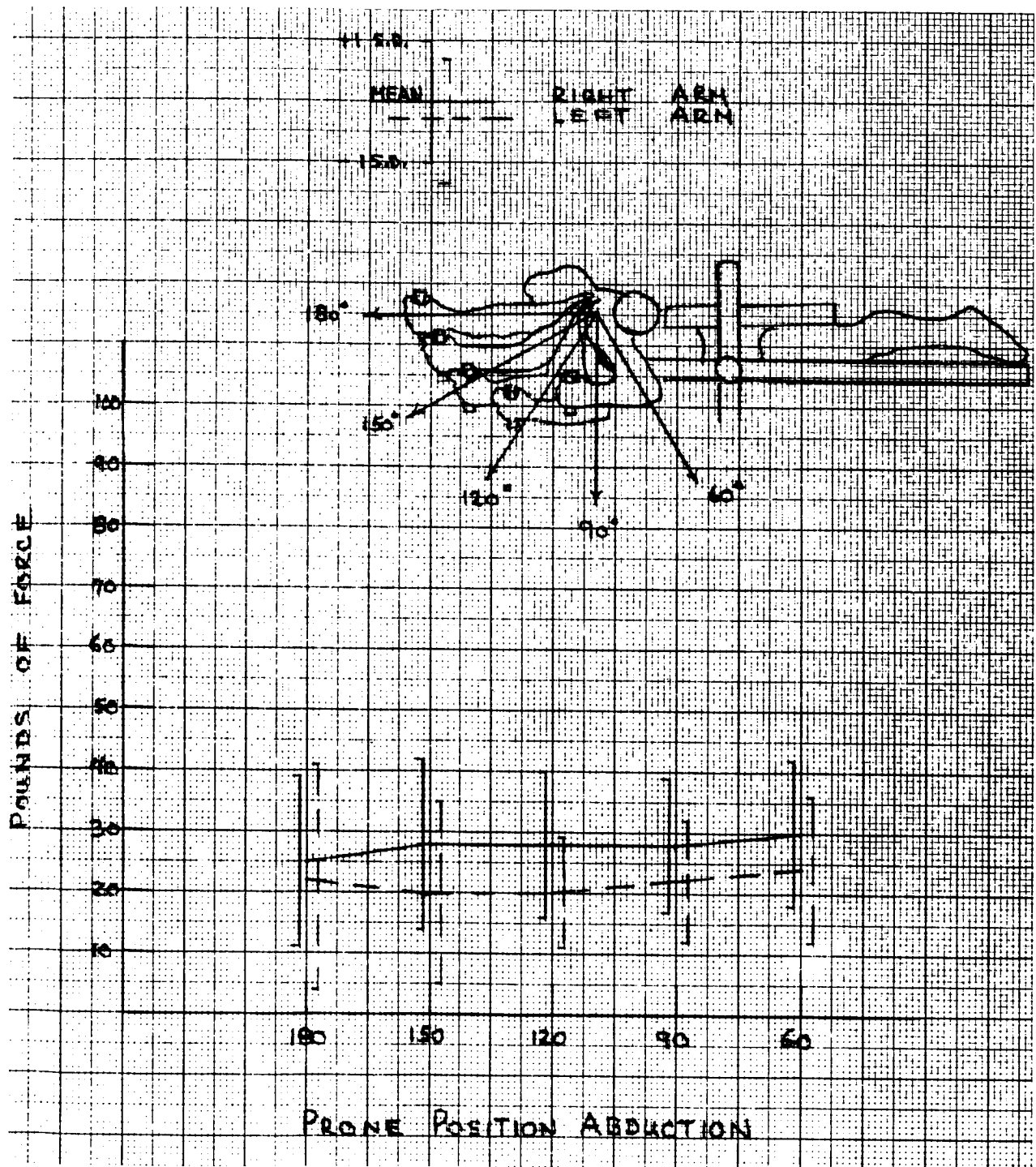


Figure 4.5. Arm Strength at 1 G

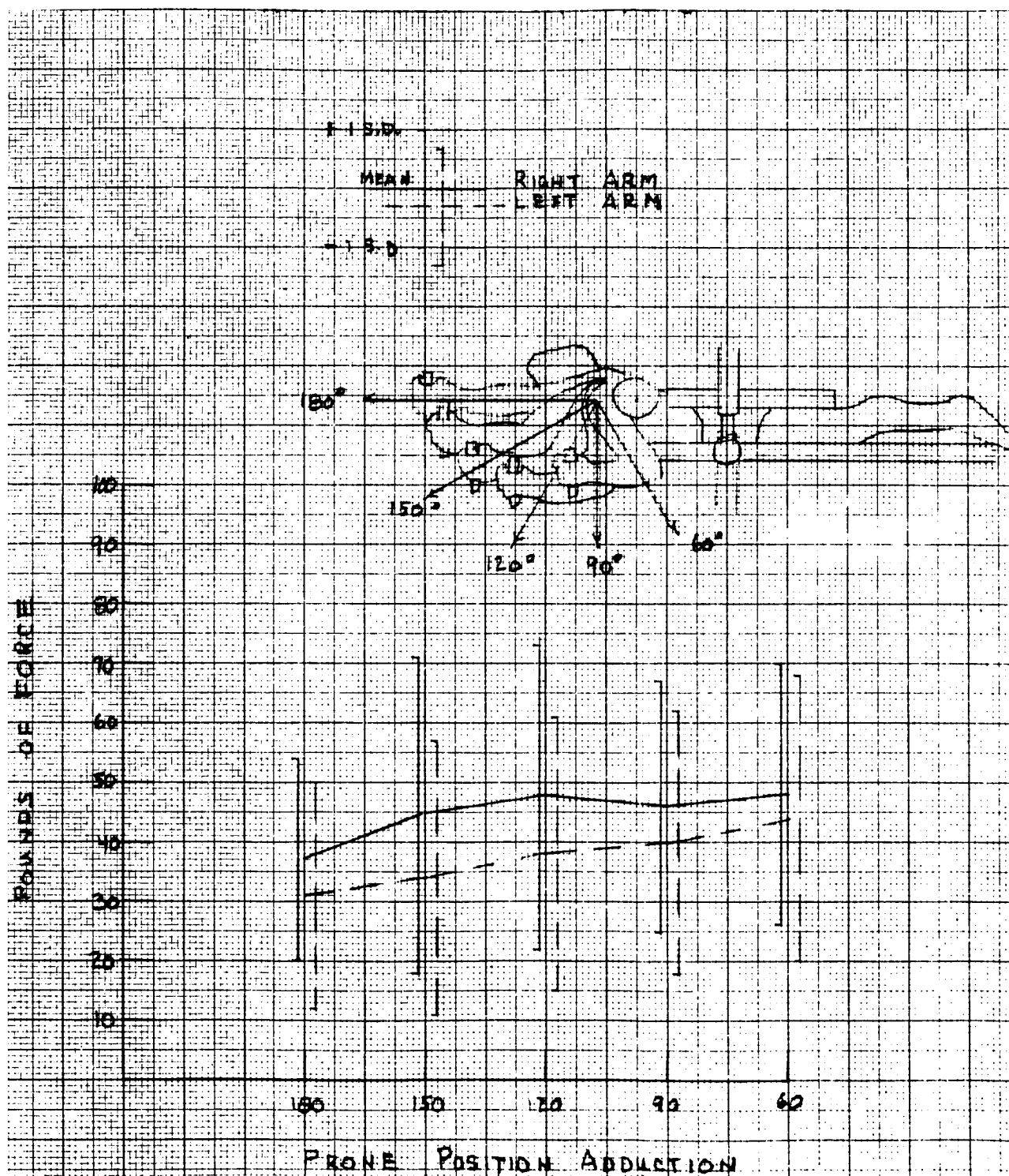


Figure 4.6. Arm Strength at 1 G

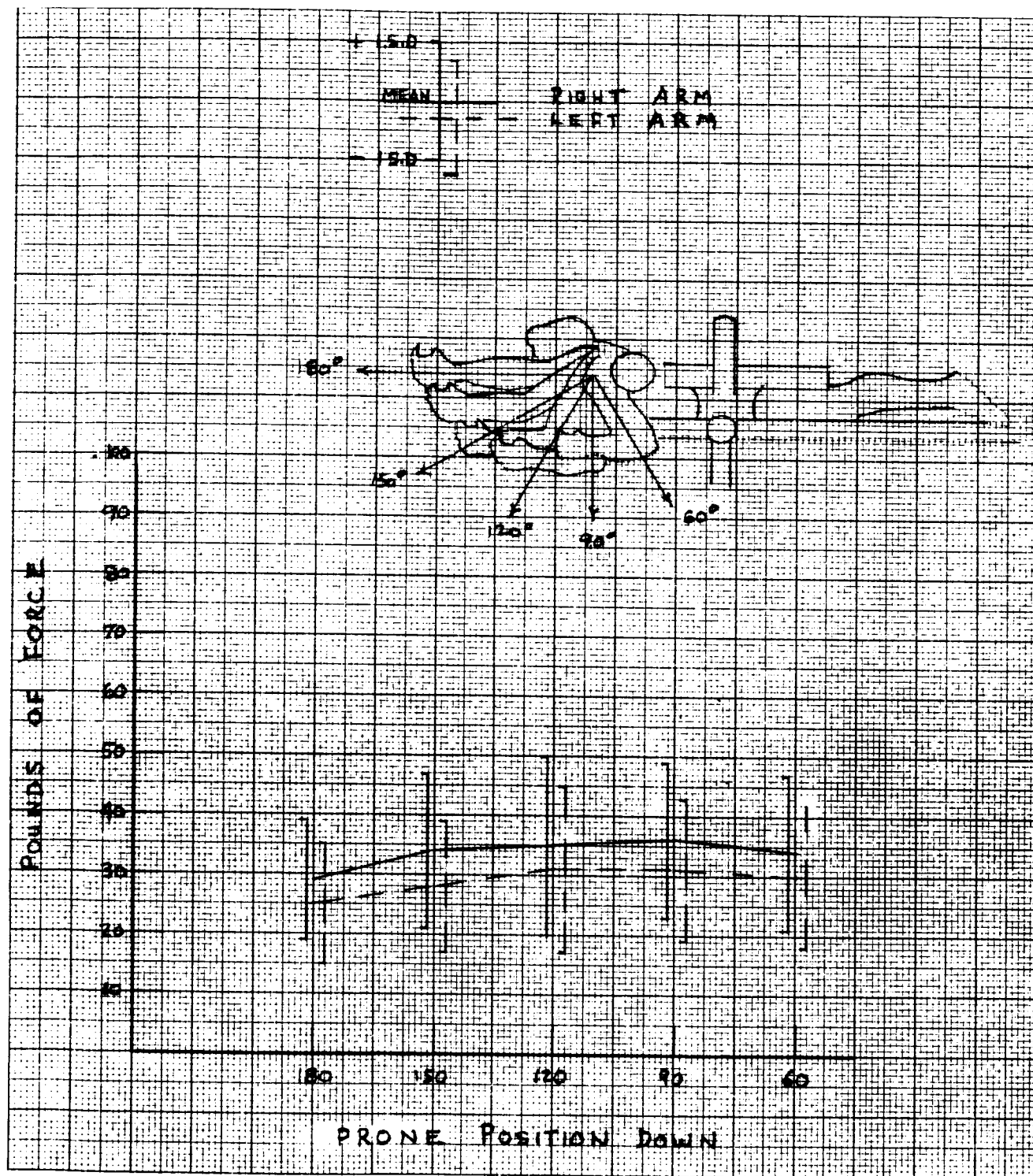


Figure 4.7. Arm Strength at 1 G

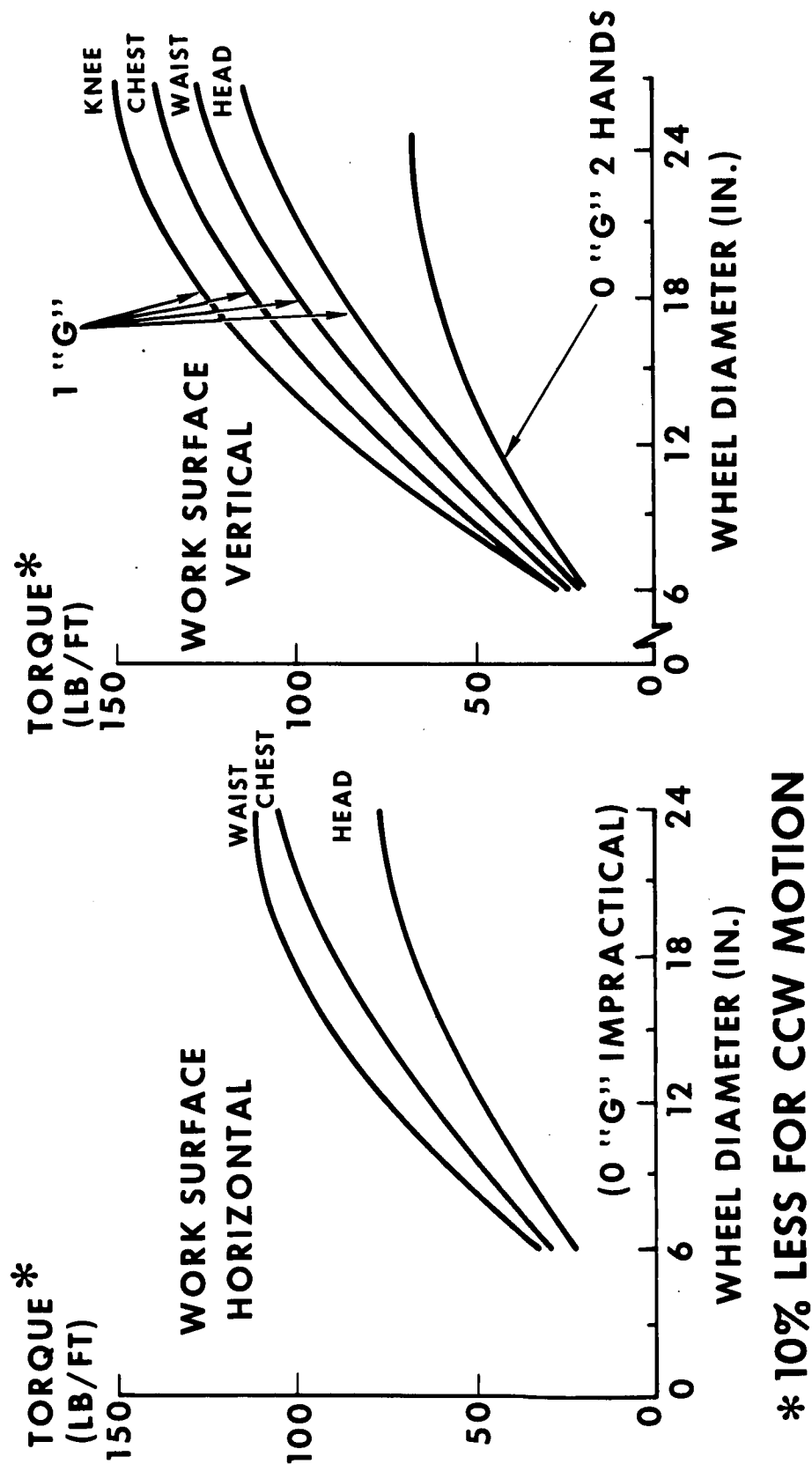


Figure 4.8. Torque Producing Capability as a Function of Wheel Size and Relative Position of the Work

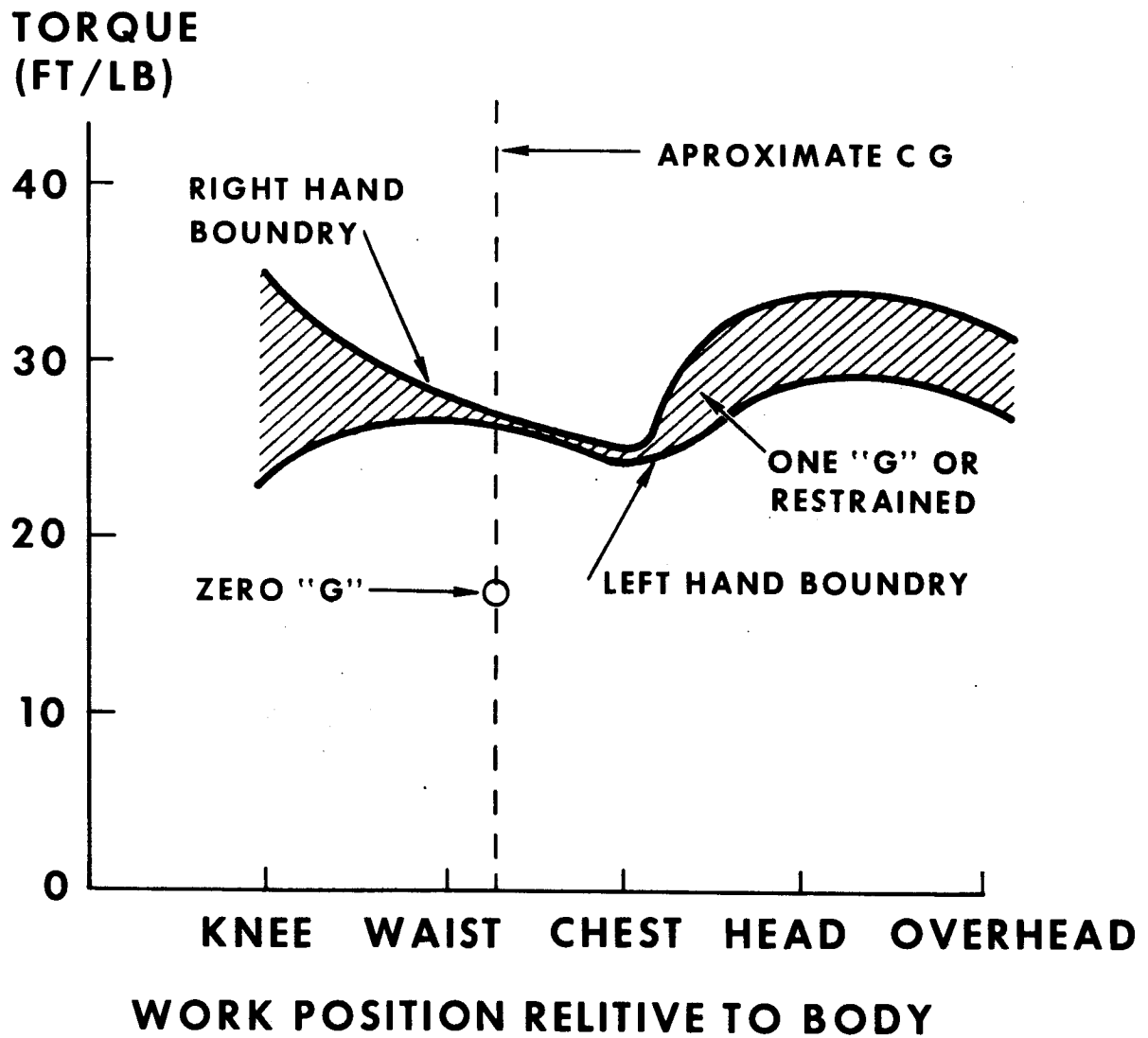


Figure 4.9. Torque Producing Capability Through a Six-Inch Moment Arm, as a Function of Work Relative Position

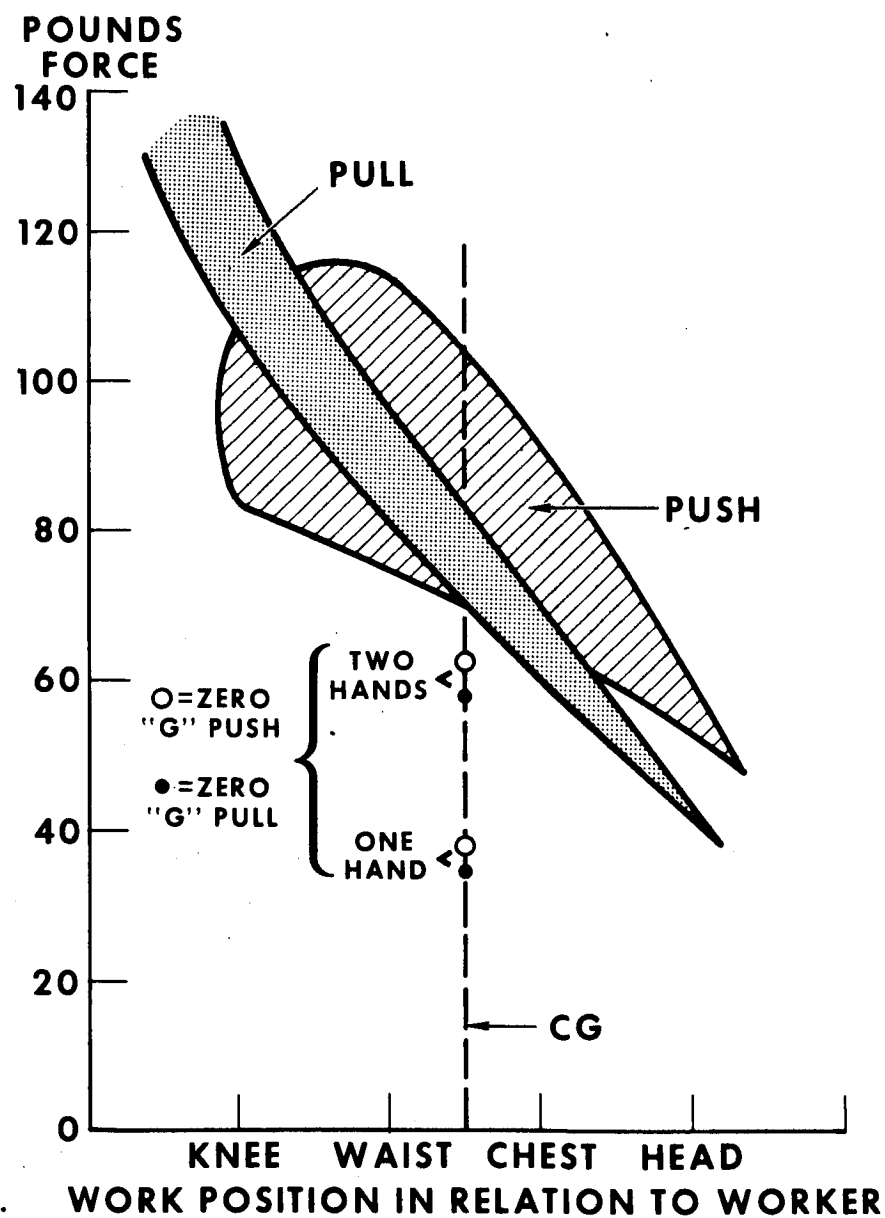


Figure 4.10. Force Producing Capability in Translation as a Function of Working Conditions



The zero-g case is unrestrained and is the result of the need for an isometric force. The resulting recommended design values are:

- a. Restrained in zero  $g < 24$  foot pounds
- b. Unrestrained in zero  $G < 18$  foot pounds

Note: Hand-holds must be available in the appropriate positions.

- c. Two hands restrained  $< 45$  foot pounds

Figure 4.10 presents the measured capability of a worker to produce a translatory force as a function of various working conditions. In this case, the lower boundary represents the mean one-hand value, and the upper boundary is the mean two-hand value. In zero  $g$ , but restrained at the c.g., the force producing capability is reduced, but is higher than the chest or head values. The wide variation displayed as a function of relative position is attributed to the moment arm created by the distance from the anchor point (the feet). The worker can push harder than he can pull. The resulting values are:

- a. Restrained in zero  $g < 60$  pounds - push or pull
- b. Unrestrained in zero  $g < 35$  pounds - isometric

NOTE: Hand-hold or parallel surface required.

#### 4.3 FORCE PRODUCTION IN REDUCED GRAVITY

During a major portion of the planetary missions, and perhaps the longer earth orbital missions, the spacecraft will probably be operating in an artificial gravity mode. Since this is true, and most of the maintenance and repair actions will be performed under these conditions, it is necessary to identify workman's capability, unaided, under these conditions. Figures 4.11 and 4.12 presents some applicable data from reference 4.3 as correlated with S&ID studies. The gravitational force simulated was about  $1/6$  of earth's force.

Figure 4.11 provides estimates of force producing capabilities under these conditions but for the work approximately parallel with the worker's c.g. These data, when used as a correction factor in conjunction with that of Section 4.2, can provide values for any relative position.



Figure 4.12 presents the results of tests involving typical maintenance and repair activities where gravity plays an important role in neutralizing the reactive force vector. It is evident from these data that the worker capability for all operations is substantially less in the reduced gravity with the exception of lifting. Note that in contrast with the equivalent one-g data, his capabilities to perform these tasks are reduced by over a factor of three.

#### 4.4 THE TRACTIONLESS ENVIRONMENT (ZERO G)

The tractionless environment is that situation where the worker is not under a direct or simulated gravitational force - he is essentially weightless. In this mode his useful work output depends on his ability to apply an isometric force, or take advantage of his moment of inertia (Ref. 4.2). Application of the isometric force satisfies Newton's third law, but, in so doing, the useful work producing capability is reduced to less than one-half ( $1/2$ ) of the normal environment. This is made clear through the contrasting data of section 4.2.

Where the worker's moment of inertia is the only neutralizing reactant for the force vector, little to no useful work can be performed even with a non-isometric handhold. This is demonstrated by the S&ID data and that taken from references 4.2 and 4.4. A sample test result is presented in Figure 4.13. The conclusions to be drawn from these data are summarized by the word impractical. Work in the tractionless state is impractical without the use of appropriate restraints which effectively neutralize the reactive forces.

#### 4.5 DEXTERITY IMPEDIMENTS AND WORKING CONDITIONS

Dexterity impediments are a secondary effect created by the need to work in the hostile environment known as "space." The need for maintenance and repair activity, external to the spacecraft, is known to be low; that is, less than five percent of the total expected load. This will probably be less than one in any 90-day period; however, when required, it will probably be a very essential operation to crew safety. For this reason a careful definition of the restraints resulting from the suited mode of operation is mandatory. The dexterity impediments are created by the need for a pressurized suit. Wearing a suit imposes decrements in:

- a. Manual dexterity
- b. Force values

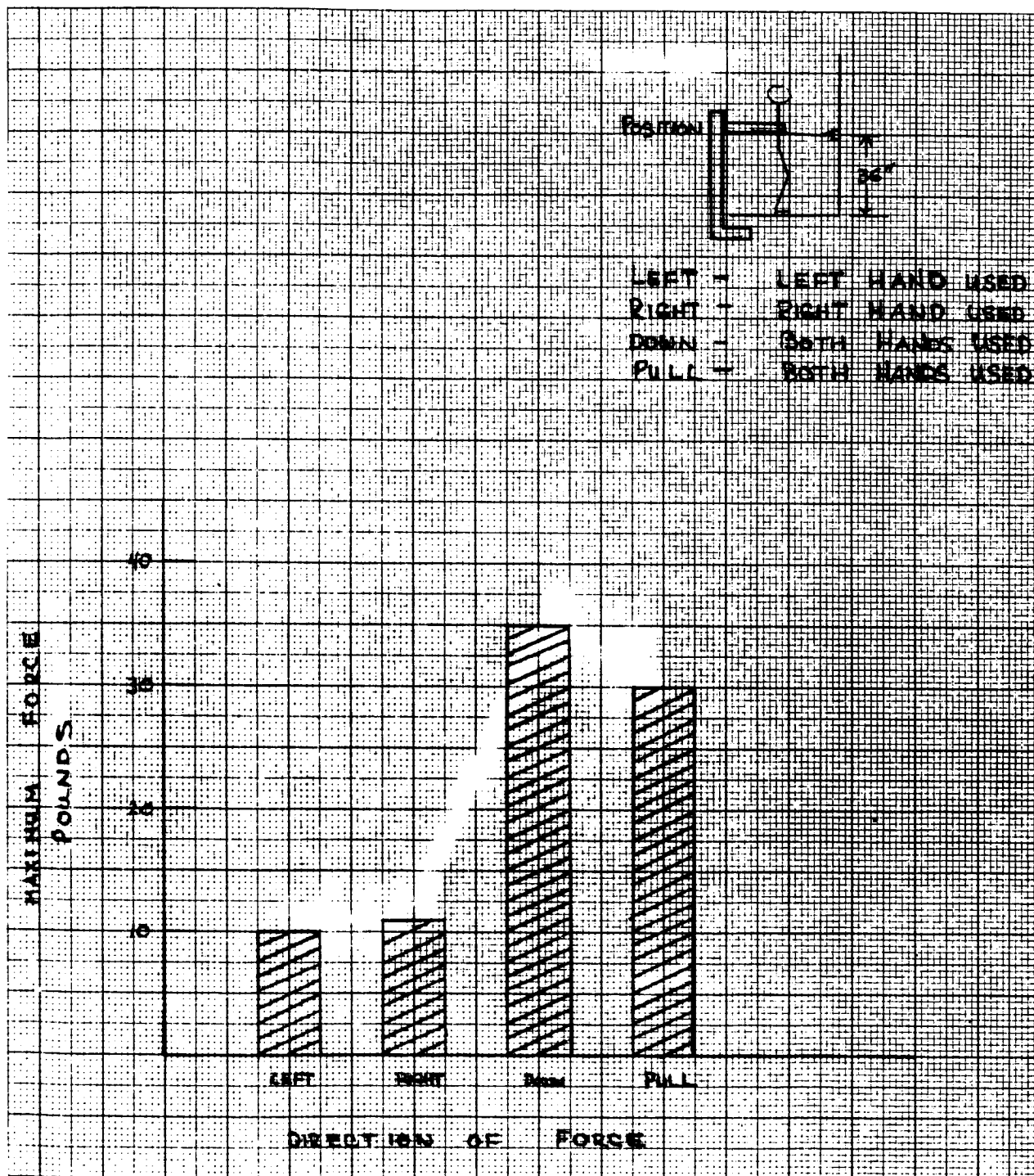


Figure 4.11. Maximum Forces Exerted at 1/6-G Without Loss of Footing

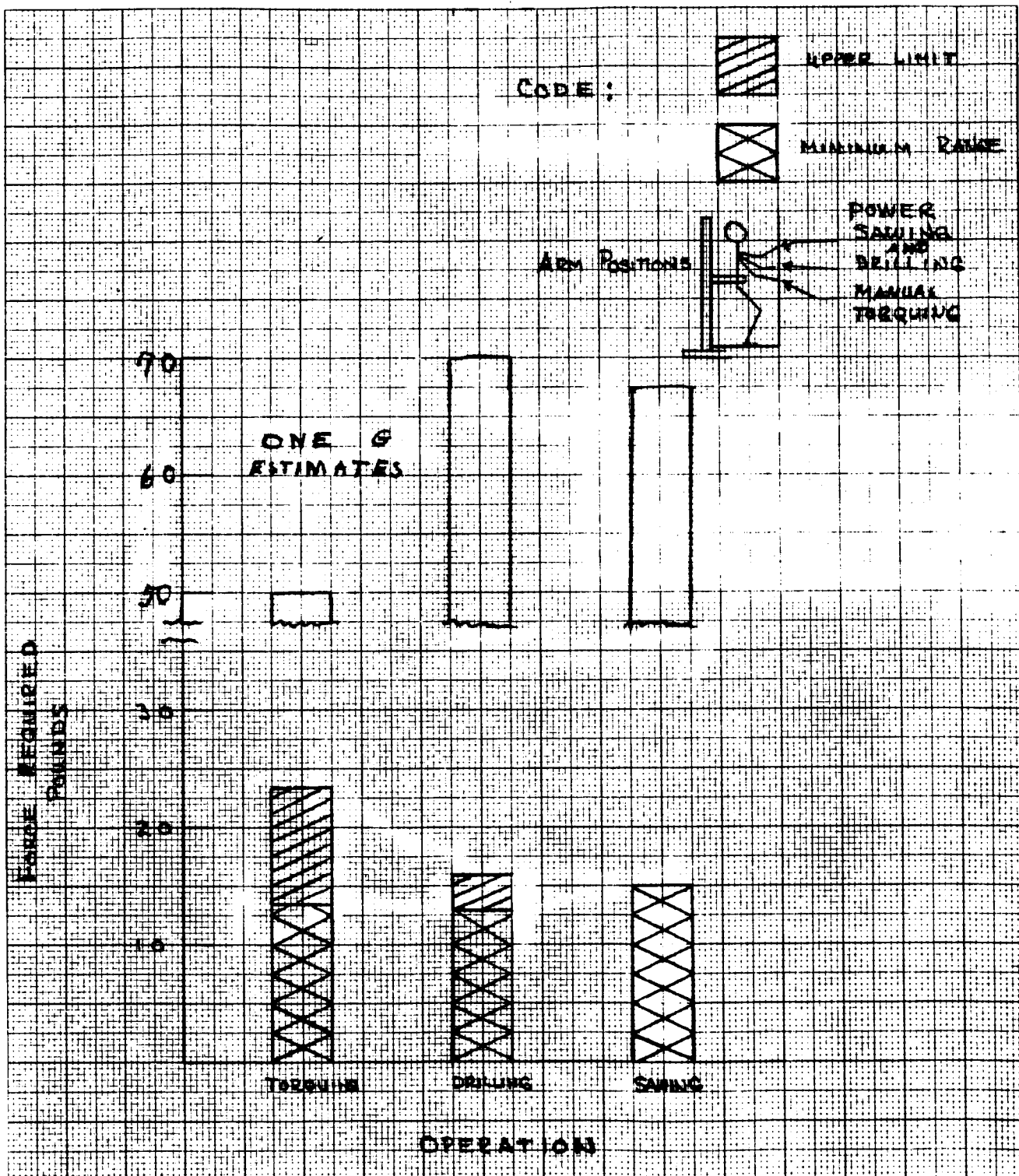


Figure 4.12. Force Requirements for Maintenance Operations at 1/6-G

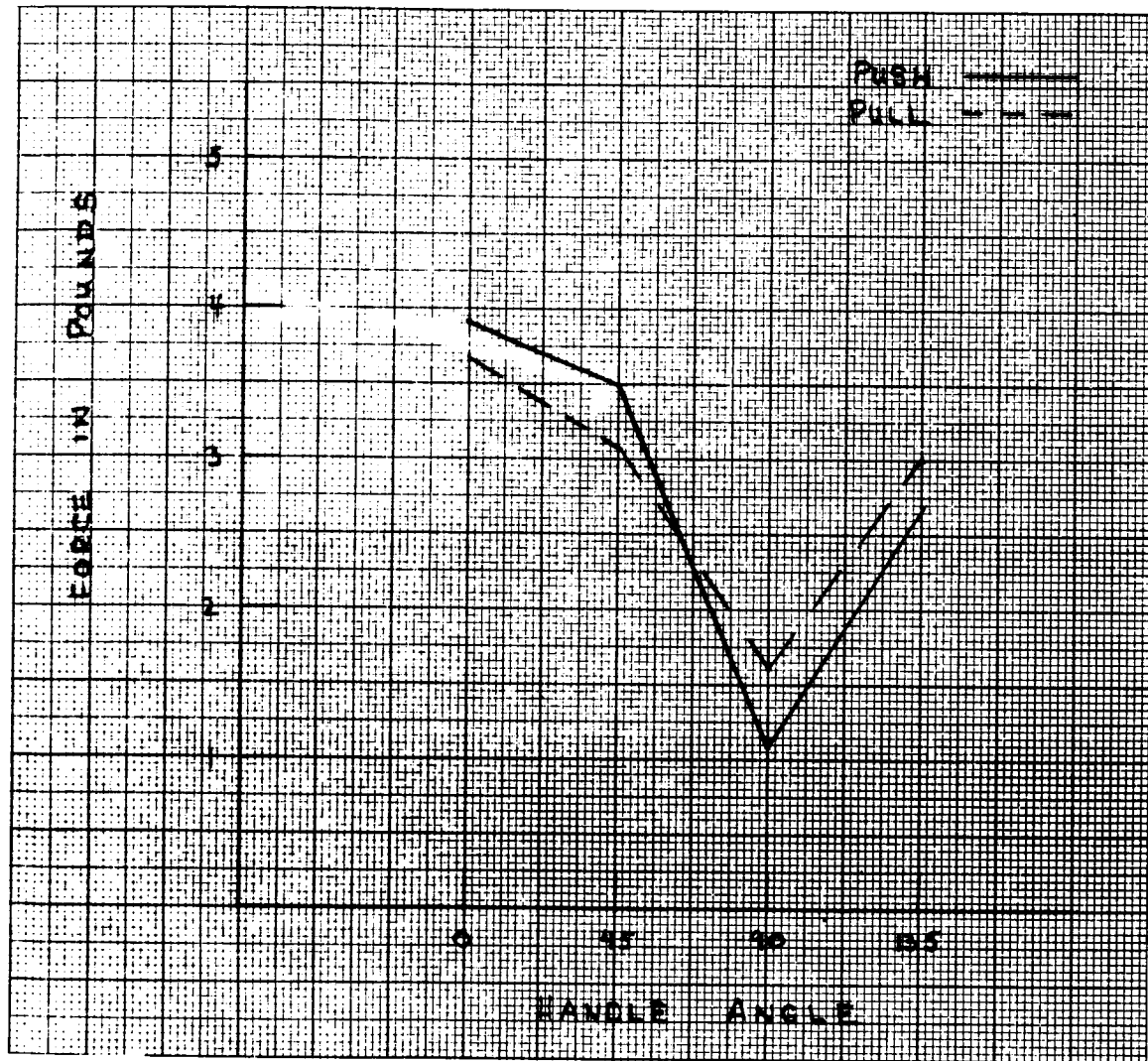


Figure 4.13. Mean Resultant Forces for Six Subjects on a Frictionless Scooter Corrected for the Scooter's Friction



- c. Total work capability
- d. Time required to accomplish a given task.

Table 4.1 presents some of the result from Reference 4.2, an evaluation of man's capability to use specific tools, as a function of suit conditions. In essence, it shows that in applying a torque through a screwdriver, it is known the pressure suit reduces capability by only about 20 percent, yet, his ability to grip pliers is reduced to almost 50 percent. Both values assume that the task size is within the dexterity envelope of the suited workman.

Figure 4.14 presents some of the results from Reference 4.5 and demonstrates the decrements introduced on force capability by the pressure suit at 3 psig. A comparison of these data and those of Section 4.2, the unsuited case, indicate a suit decrement in the order of about 35 to 40 percent for a given set of conditions.

Dexterity is a measure of the worker's ability to use his hands and fingers in accomplishing relatively delicate operations. The worker's manual dexterity is adversely affected by the relative inflexibility of the existing pressure suits. This relative effect can be seen from Figure 4.15, which was prepared from data in Reference 4.6. It plots the number of fins which can be placed in the Purdue pegboard with either hand, and both within a 30-second time interval. The lower line denotes the worker's ability to complete the indicated number of assembly operations within a sixty-second interval. The data presented indicates a considerable decrement in refined finger and hand dexterity results from use of a pressurized suit. The test can be roughly construed as being analogous to activities requiring the handling of small piece parts, such as nuts and bolts associated with typical assembly and disassembly tasks. These data indicate that the parts handled must be much larger than normal, and more clearance between each part/operation is required. Further, more time must be allowed for each discrete task.

#### 4.6 TASK TIME DECREMENTS

Figure 4.16 presents a composite of some results derived from S&ID tests and those in Reference 4.7. The results of these sources seem to correlate within acceptable experimental error. These data indicate that the impediments imposed by suited activities impose a very noticeable decrement in the time required to accomplish a given task. A full-pressure suit will probably increase a given task time by a factor of 2.6, or to as much as 260 percent of the normal time.



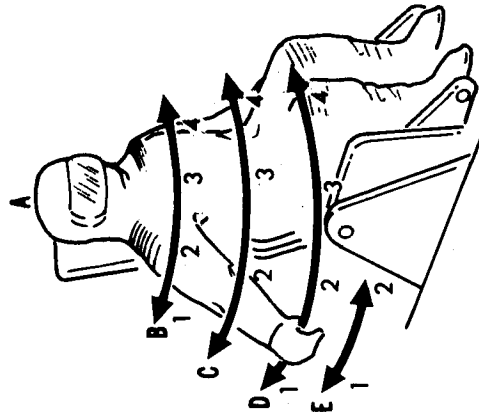
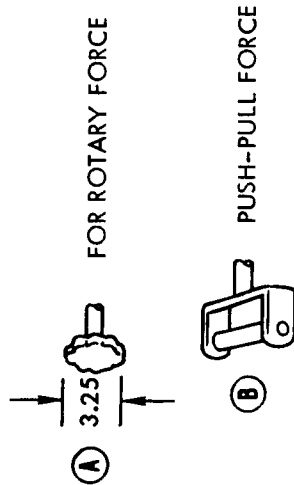
Table 4.1. Force Producing Capability Through Some Typical Tools

| Tool                          | Working Environment |                     |                   |
|-------------------------------|---------------------|---------------------|-------------------|
|                               | Shirt Sleeve*       | Unpressurized Suit* | Pressurized Suit* |
| Knob<br>CW<br>CCW             | 118                 | 118                 | 105               |
|                               | 117                 | 117                 | 105               |
| Screwdriver<br>CW<br>CCW      | 69                  | 62                  | 55                |
|                               | 69                  | 62                  | 55                |
|                               | 57                  | 45                  | 47                |
| Ball Shape<br>CW<br>CCW       | 83                  | 74                  | 60                |
|                               | 73                  | 71                  | 50                |
| Pliers (Grip)<br>R.H.<br>L.H. | 112                 | 81                  | 60                |
|                               | 110                 | 78                  | 60                |

\*Torque in inch-pounds except grip which is in pounds



| MEASURING POINTS | DISTANCE FROM SRP |           |                | STIRRUP (B) |      | KNOB (A)  |                   |                   |
|------------------|-------------------|-----------|----------------|-------------|------|-----------|-------------------|-------------------|
|                  | VERTICALLY        | LATERALLY | LONGITUDINALLY | PUSH        | PULL | CLOCKWISE | COUNTER-CLOCKWISE | COUNTER-CLOCKWISE |
| A-1              | 34                | 12        | 11             | 15          | 60   | 110       | 100               | 45                |
| B-1              | 28                | 30        | 5              | 50          | 80   | 125       | 140               | 75                |
| -2               | 28                | 24        | 15             | 45          | 85   | 105       | 105               | 55                |
| -3               | 28                | 16        | 20             | 35          | 85   | 145       | 140               | 70                |
| -4               | 28                | 9         | 21             | 30          | 80   | 80        | 100               | 65                |
| C-1              | 18                | 31        | 4              | 45          | 70   | 125       | 145               | 65                |
| -2               | 18                | 28        | 12             | 55          | 90   | 85        | 120               | 60                |
| -3               | 18                | 21        | 19             | 65          | 80   | 115       | 135               | 60                |
| -4               | 18                | 6         | 22             | 50          | 80   | 125       | 140               | 55                |
| D-1              | 8                 | 28        | 4              | 45          | 55   | 130       | 150               | 55                |
| -2               | 8                 | 25        | 12             | 50          | 75   | 105       | 140               | 45                |
| -3               | 8                 | 19        | 18             | 50          | 90   | 120       | 140               | 65                |
| -4               | 8                 | 10        | 20             | 45          | 80   | 105       | 150               | 65                |
| E-1              | 2                 | 20        | 5              | 60          | 85   | 100       | 125               | 50                |
| -2               | 2                 | 12        | 14             | 60          | 80   | 95        | 140               | 60                |



20TH PERCENTILE MAN IN MARK IV FULL PRESSURE SUIT PRESSURIZED TO 3.5 PSIG SECURED BY SEAT BELT. MEASURING POINTS ARE IN INCHES - PUSH/PULL FORCES IN POUNDS AND ROTATIONAL FORCES IN INCH-POUNDS

Figure 4.14. Manual Forces in a Pressure Suit

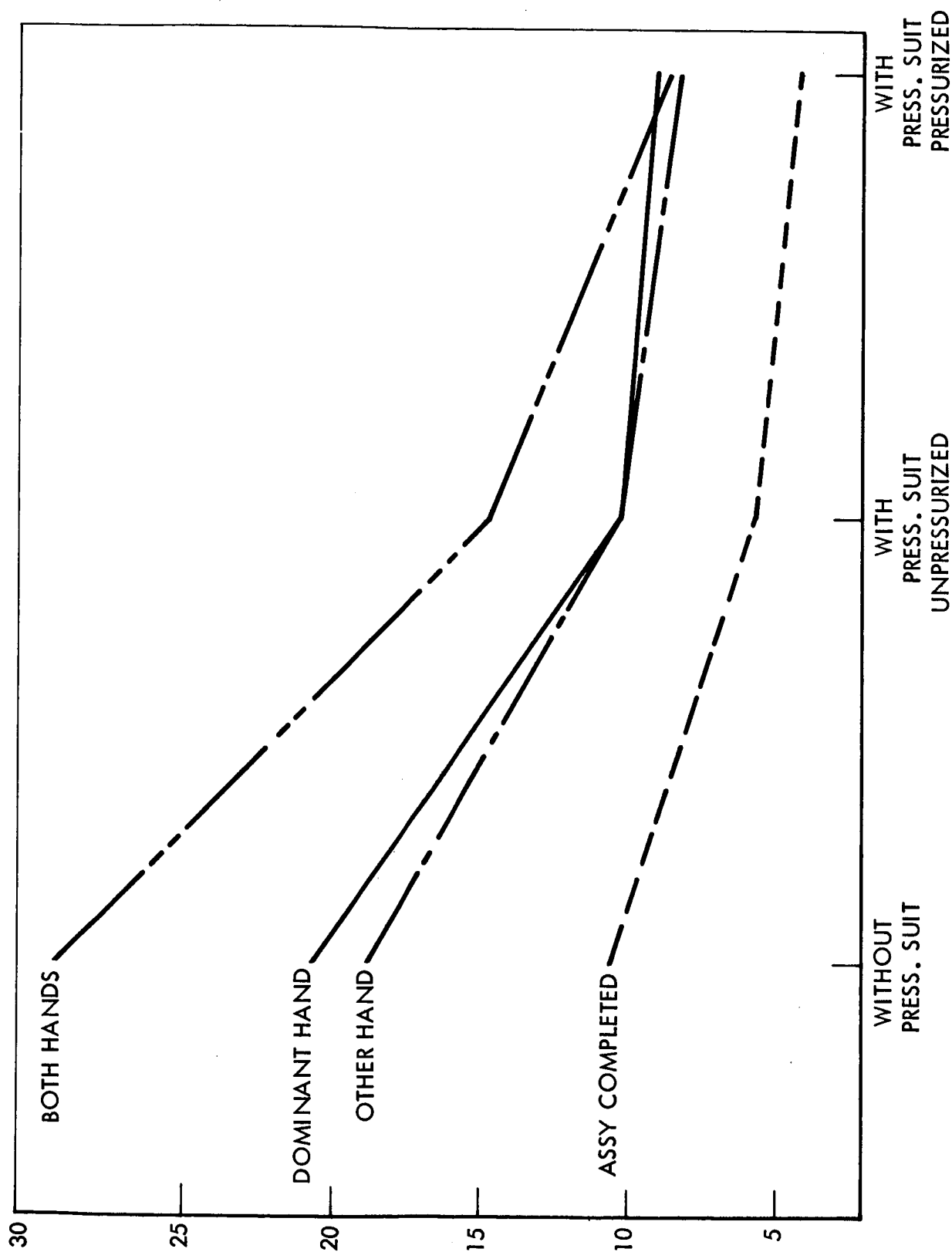


Figure 4.15. Refined Finger Dexterity—Number of Pins Placed in Holes (Purdue)

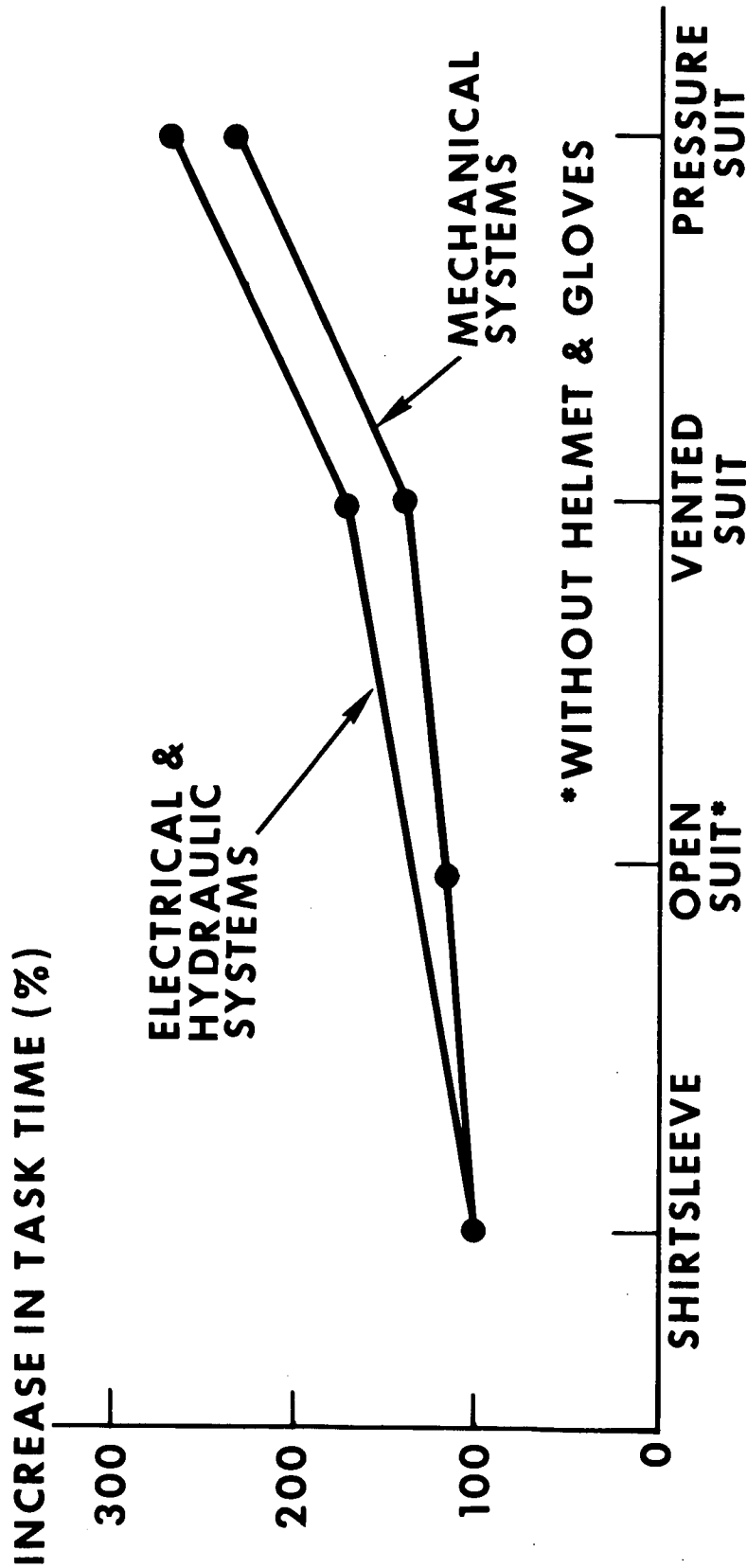


Figure 4.16. Percent Increase in Task Time as a Function of Working Condition Impediments



Zero g is not expected to adversely affect task time after adequate training. These same data sources indicate that after between 15 to 20 repetitions of a given task in zero g, the time required to perform the operation levels out at nearly the same value required on earth, all other conditions being the same.

#### 4.7 REFERENCES FOR CREWMAN CAPABILITIES AND CONSTRAINTS

- 4.1 Hunsicker, P.A. "Arm Strength at Selected Degrees of Elbow Flexion," WADC TR 54-548, WADC, WPAFB, Ohio (August, 1955).
- 4.2 Springer, W.E., and Streimer, I. "Work and Force Producing Capabilities of Man," Document No. DZ-90245, Boeing Airplane Company, Seattle, Washington (August, 1962).
- 4.3 Holmes, A.E. "Space Tool Kit Survey, Development and Evaluation Program," Martin Company, Baltimore. Final Report ER 13942, prepared under contract No. NAS 9-3161.
- 4.4 Dzendolet, E., and Rievley, J.F. "Man's Ability to Apply Certain Torques While Weightless," WADC TR 59-94 WADC, WPAFB, Ohio (April, 1959).
- 4.5 Pierce, B.F. "Manned Force Capabilities of a Pilot in a Full-Pressure Suit: Techniques of Measurement and Data Presentation," Engineering and Industrial Psychology, 1960. pp. 2, 27-33.
- 4.6 Pierce, B.G. "Effects of Wearing a Full-Pressure Suit on Manual Dexterity and Tool Manipulation," The Journal of the Human Factors Society, Vol. 5, No. 5 (October, 1963).
- 4.7 Seeman, J.S., and Smith, F.H. "A Technique to Investigate Space Maintenance Tasks," AMRL-TR-66-32 AMRL, WPAFB, Ohio (April, 1966). (Also Ref. (A)).