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APOLLO II SECOND GENERATION

LUNAR EXPLORATION SYSTEM STUDIES

CFY 1963

(U)

October 1963

J. O. Matzenauer, Project Manager
Lunar & Planetary Systems

Approved by: M. W. J. Bell, Director
Lunar & Planetary Systems

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FOREWORD

The subject studies were accomplished under S.A. 6052 during Company fiscal year 1963 by a team consisting of Advanced Systems, Aerospace Sciences, and Information Systems personnel. The report investigates specific conceptual design configurations and, additionally, attempts to show in most cases the effects of major variables on these designs. Considerable background information is included so that the report can be used as a reference source in various areas of technology as well as a source of specific mission configuration data.

Acting as Project Engineers on portions of the study were W. H. Meier and L. W. Tucholski of Lunar & Planetary Systems. Mr. Tucholski has also been responsible for integrating the technical contributions into this report, with considerable assistance from Mr. Meier. Others who have contributed substantially to the concepts described herein are as follows:

D. Armsby, R. Barr, E. Beever, S. Bennis, A. Bialecki, N. Blitch, W. Blommer, D. Bradford, W. Brayton, R. Butler, M. Cantor, K. Cassatt, S. Coffman, D. Copping, W. Dailey, L. Drankham, R. Dycus, P. Fono, S. Gee, C. Gerber, H. Goodel, J. Grafton, D. Howard, D. Irwin, G. Johns, B. Karen, H. Kaysen, A. Kazanowski, B. Kilpatrick, R. Krause, J. Leffler, D. Lifton, J. Matzenauer, A. Nussberger, L. Pearce, D. Peebles, D. Pergrine, G. Pudonovich, M. Sanger, T. Seitz, C. Stover, J. Tracy, S. Turnbull, H. Woodlief.

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PART III. LUNAR SHELTERS

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CFY 1963 R&D STUDIES

SUMMARY

The Apollo II Second Generation Lunar Exploration System includes the direct landing spacecraft which consists of cargo command module, service module, and landing module. The landing module is also capable of being used as the Lunar Landing Vehicle (LLV) for landing unmanned cargos consisting of shelter modules such as the Lunar Occupancy Payload and other cargo in support of lunar surface operations. High energy cryogenic propellants are utilized to permit direct landing, manned, or logistic missions with use of a single Saturn V class booster.

In last year's studies, the LLV was configured for maximum payload and with consideration for the direct three-man landing and return mission. Light weight and low vehicle height above the lunar surface at touchdown were major objectives. Logistic cargos of more than 27,000 lb landed on the Moon were achieved within the single Saturn V boost capability.

For the manned mission, the lunar take-off weight was determined to be 28,000 lb ready for the return-to-Earth portion of the mission. The command module utilized was an advanced light-weight design weighing 10,000 lb, including supporting subsystems. Cryogenic oxygen/hydrogen propulsion was again used for maximum propulsion efficiency.

Study of the Lunar Occupancy Payload was also accomplished last year. This module was configured to serve as an early lunar shelter or outpost station or as a basic module of an integrated base module complex. Single and dual compartment versions, as well as special mission versions, were studied.

Elementary economic analysis quickly reveals the sensitivity of lunar exploration costs to the crew capacity of the transport vehicle supporting lunar operations beyond the original LEM landings. In this year's studies, it was determined that the present Apollo command module could be re-configured to contain five men for the direct lunar mission and that the Saturn V booster capability would need only minor uprating (or removal of margins) to accomplish the translunar injection of this heavier spacecraft. Likewise, a seven-man version was studied, and this configuration requires a 20 inch increase in command module base diameter. The resulting translunar injection weight of 110,000 lb would require a more substantial but still feasible uprating of the Saturn V booster.

For the lunar occupancy payload configurations, additional work was accomplished to investigate the results of increasing the unoccupied standby time on the lunar surface from the previous few weeks to as much as one year, and the occupancy time from the previous few weeks to as much as four months for several men. A shelter for five men can be configured for up to one year standby and four months occupancy for less than 30,000 lb.

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While the major objective of the Apollo II studies has been to accomplish the conceptual design of the maximum performance spacecraft, it was recognized that exigencies of timing and funding might make the accomplishment of its development some distance out into the future. In the meantime, there are several excellent adaptations and modifications of the present Apollo hardware and modules which can perform some of the tasks for extremely favorable development costs. For example, the Apollo command module can be configured as a temporary occupancy shelter for the LEM crew to extend their capabilities for mission accomplishment while on the lunar surface. This shelter version can be designed for less than 7,000 lb, which is the stated payload capability of the cargo version of LEM. An even more intriguing possibility is the use of an adapted Apollo service module for use as an early LLV which requires only the addition of remote guidance equipment, lunar landing gear, and throttling of the service module engine. This Early LLV, utilizing the existing 45,000 lb of Earth-storable propellants, can land a cargo of approximately 13,500 lb on the Moon, which is roughly twice the capability of the cargo LEM (in all probability, the development costs for this vehicle would be less than twice the development cost for the CLEM). By extending the service module tankage approximately 41 inches, the Early LLV can land on the order of 18,000 lb on the lunar surface without exceeding the Saturn V booster trans-lunar injection capability. Thus, with either version, the modified service module is capable of landing a shelter for the LEM crew plus a very substantial amount of extra supplies or other cargo. These configurations should have very considerable appeal by virtue of their potentially early development, maximum utilization of existing program hardware, and resulting minimum development costs.

INTRODUCTION

In the past two years, the Apollo mission has been studied at considerable length by Advanced Systems Division with various constraints and mission modes, including Earth orbital rendezvous, lunar orbit rendezvous, direct mission with Earth-storable propellants and NOVA booster, and direct mission (no rendezvous) with high-energy cryogenic propulsion employing a single Saturn V booster. These studies at one time constituted potential Apollo mission configurations, back-up possibilities, and alternate possibilities and were reported in References (1), (2), and (3).

After the resolution of the Apollo mission in the lunar orbital mode in mid-1962, the back-up and alternate mode studies were terminated and study emphasis was then re-directed toward the conceptual development of a true second generation, or Apollo II system. The objective sought in these studies was then to develop and analyze a concept which is not only compatible with the Saturn V booster system but, also, one which would provide maximum mission effectiveness per Saturn V launch. The first study resulted in an interim report, Reference (4), in late 1962 reflecting the conceptual design of a three-man direct mission spacecraft with gross weight at translunar injection well within the capability of the Saturn V booster system. This was accomplished by utilization of cryogenic oxygen/hydrogen propulsion in the lunar landing vehicle stage and in the lunar take-off and return service module. In keeping with the operational time period under consideration, early 1970's, the command module was similar to the current contract configuration but employed certain advanced concepts in structure and subsystems, permitting a substantial weight reduction.

These Apollo II Second Generation studies were further refined and, as a result of the then current interest in the lunar logistic system, the studies were summarized in two separate reports, References (5) and (6), one for the logistic support mission and one for the direct landing manned mission. In the logistic mission, the LLV or lander stage was configured to land a substantial cargo within one mile of a target on the Moon. For the manned mission, the same LLV stage was utilized with the addition of a return service module and propulsion stage combination, also employing oxygen/hydrogen propellants.

Meanwhile, in late summer of 1962, response was made to two NASA RFP's, one for study of the lunar logistic vehicle and one for the lunar occupancy payload, which was to be an LLV cargo module. These proposals were embodied in References (7) and (8). Although these proposals did not result in contracts, the effort in both directions continued. Studies of the lunar occupancy payload in several variations for the transporting of supplies and to serve as the initial outpost or module of a lunar base resulted in report of Reference (9).

Briefing material was generated for all these concepts and was contained

in References (10), (11), and (12). These briefings have been presented to NASA-Lewis, NASA Headquarters, NASA-MSC, and NASA-MSFC. In all cases, substantial interest has been shown in the concepts presented. All cognizant persons have recognized, either from these presentations or from their own independent research, that a critical problem in achieving these Apollo II missions is the long-term storage in space of the cryogenic propellants.

NASA-MSFC, who has nominal responsibility for the lunar logistic system, has developed and analyzed extensively a mission philosophy of their own within the framework of the lunar logistic system. This is the multi-mission module stage, or MMM, which is a central structural core with two RL-10 oxygen/hydrogen engines, necessary attitude control and ullage rockets, and associated control equipment. In this concept, the prime idea is flexibility of application so that the tanks are "rubberized" cylinders which can be lengthened or shortened, together with outside structural shell as necessary for a particular mission application. In keeping with this concept then, the cylindrical tanks are radiation-insulated with Linde super-insulation, and the tanks are supported on thin-walled tubular struts for low heat transfer by conduction. As a result of the emphasis on flexible application and the relatively uncomplicated insulation philosophy which resulted, together with the resulting low mass fraction of the MMM, it is necessary to utilize two of these modules in tandem to perform the lunar landing logistic mission. In contrast, our Apollo II configuration approach was to utilize the maximum possible performance and maximum efficiency design and to conceive a configuration which can take advantage of this best possible performance. (An additional constraint was minimum height above the lunar surface in touchdown condition.)

R&D studies for the Apollo II concepts during this calendar year have included expanding the previous studies in several respects. Studies of five- and seven-man command modules for the direct landing mission have been performed with the objective of achieving the maximum possible number of personnel landed on the Moon per Saturn V launch, and to do this with minimum hardware modifications. Simple economics quickly reveals that in supporting lunar exploration in the 1970's, the high cost of each Saturn V launch must be minimized by achieving the maximum pay-off in terms of usable cargo or crew. Likewise, the lunar occupancy payload, or early station module, has been further analyzed to determine the configurational changes involved in extending the period of unoccupied standby on the Moon from several weeks to several months and, also, increasing the period of useful occupancy from a few weeks to a few months.

References

1) SID 62-325, "Lunar Landing Mission Design Study - Interim Report," 1 March 1962

2) SID 62-383, "Interim Progress Report on Lunar Landing Module Studies," 1 April 1962

3) SID 62-672, "Progress Report on Apollo Lunar Landing Module Studies," 1 June 1962

4) SID 62-1189, "Lunar Mission Systems Studies - Interim Report," 30 October 1962

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6) SID 62-1467, "Lunar Mission Systems Studies," 31 December 1962

7) SID 62-1050-1, "Technical Proposal for a Study of a Spacecraft Vehicle for a Lunar Logistics System," 17 August 1962

8) SID 62-1051-1, "Technical Proposal for a Payload Performance Study for a Lunar Logistics System," 17 August 1962

9) SID 62-1473, "Lunar Occupancy Payload Study," 4 January 1963

10) SID 62-257, "Lunar Landing Mission Design Studies," (Briefing) February 1962

11) PS-42B, "Lunar Logistic System," (Lunar Occupancy Payload) (Briefing) December 1962

12) PS 63-41, "Apollo II Second Generation Lunar Exploration System," (Briefing) Revised November 18, 1963

RECOMMENDATIONS

The studies which were conducted during this study effort resulted in formulation of the preliminary designs for five- and seven-man transport vehicles, early exploratory lunar shelters and logistics modules, and the cryogenic tank insulation test program, which are described under Parts I through IV of this report. In addition to optimum design recommendations, the studies pointed out those technological areas requiring further investigation and analysis not possible during this reported effort because of time and funding limitations. Those technological areas which require further study effort to add depth to NAA's knowledge pertaining to the second-generation lunar exploration system development and, consequently, to enhance NAA's position during future contract competition are described below.

1. Further study effort should be expended in the refinement of the lunar shelter designs described in this report. The minimum and maximum allowable non-operating temperatures of the various equipments contained in the shelters must be further investigated and determined in order to realistically establish temperature limits for the shelter interior during unoccupied standby periods. Control of shelter temperature through surface coatings or covers should be investigated further. Other lunar shelter configurations using Apollo modules should be investigated also. Methods for lowering the shelter concepts to the lunar surface from their vertical landed positions should be studied to alleviate the ingress and egress problems associated with a shelter which has a considerable height above the lunar surface.

2. With a view toward establishing an early exploratory lunar base, feasible expedient methods and systems for interconnecting several lunar-landed shelters should be devised. The effects of friction, abrasion, and the general lunar environment on various materials, moving or flexible parts, seals, and lubricants which may be employed in a shelter interconnection system must be investigated to establish optimum material selection. Additional operations analysis should be performed to determine the requirements of early lunar exploration, optimum number of personnel, and, consequently, optimum number of shelter modules. Further effort should be devoted to the design of an unmanned guidance system for the lunar landing of shelter modules and cargo payloads so that control and separation of the landed modules can be established. Methods for moving and orienting the shelter and cargo modules must be conceived. Environmental control of the interconnected shelters forming the early exploratory base should be investigated further.

3. To increase the possible area of lunar exploration, lunar passenger and equipment-carrying reconnaissance vehicle concepts should be explored and developed. Operations analysis encompassing exploration requirements, range and payload capacity, and lunar-lander capability should be performed. Performance of various vehicles types must be investigated, power systems must be conceived and compared, and the effect of lunar environment on various materials resolved. Finally, preliminary design concepts of the most promising vehicle types must be conceived.

4. Further application of the Apollo spacecraft to other tasks and missions should be examined; for example, utilization of the CM as a lunar-orbiting reconnaissance vehicle is one application which is vitally needed and should be investigated.

5. Finally, continuations of the critical cryogenic tank insulation test program is mandatory to further NAA's technological knowledge in this important area of the lunar exploration program. In addition, the results will have widespread application on all manner of spacecraft missions which necessarily employ cryogenics for propulsion, life support, reactants, etc.

PART I. LUNAR BASE MISSION ANALYSIS

INTRODUCTION AND SUMMARY

To permit the design of feasible lunar transport and logistics systems and exploratory lunar base concepts, basic design criteria must first be established by mission analysis. This part of the Apollo studies report sets forth some basic lunar design criteria pertaining to lunar environment; postulates and appraises the lunar missions sequence, requirements, and logistics; discusses unmanned lunar landing guidance and control requirements; and presents a preliminary survey of lunar space suit requirements.

Lunar meteoroid, radiation, and temperature data recommended for the Apollo studies are presented herein. Based upon a thorough literature survey, comparison of specifications, and considerable research, a meteoroid environment is recommended for use in the Apollo II and subsequent studies; near-Earth, cis-lunar, and lunar surface meteoroid and micrometeoroid flux and velocity data are presented. Data pertaining to the galactic and solar radiation flux to which Apollo vehicles will be exposed is presented and a model recommended. Models for lunar surface temperatures, rates of cooling, and temperature relief effects of maria, craters, etc., are discussed, and lunar surface temperatures as a function of time are presented.

To compare the merits of alternative lunar vehicles, likely lunar operations and missions are first defined and described. Then a plausible, rational framework for lunar missions in terms of launch schedules, general task sequences, and requirements is defined prior to comparison of vehicle systems. Based upon cost, Saturn V availability, and anticipated size of base complement, the Apollo II LM/LLV approach to lunar exploration is far superior to use of the LEM/CLEM approach.

Finally, a preliminary survey of lunar space suit requirements and associated design criteria are presented.

LUNAR ENVIRONMENT

Several models for the lunar environment have been used for past studies. The more important ones have been specified by NASA, Houston (Reference f), Bell Comm for the lunar base (Reference g), and the Lunar Occupancy Payload report (Reference h).

To establish common criteria to be used during the Apollo II study, the various environments were compared by S&ID's Space Physics and Flight Sciences groups, and a basepoint environment is presented, discussed, and compared with earlier and other environment criteria.

METEOROID ENVIRONMENT

A meteoroid environment is recommended for use in the future Apollo II studies. These recommendations are based upon a thorough survey of existing data and conjecture, and upon considerable research in this area.

In this report, distinction will be made between the regimes of penetration-size meteoroids (10^{-4} gr and larger) and micrometeoroids (less than 10^{-4} gr). More is known about penetration-size meteoroids than about micrometeoroids, but lack of precise and reliable knowlege is prevalent over the entire meteoroid size spectrum. The degree of caution with which pronouncements by various authors must be viewed has been discussed in some detail in References (a) and (b). Determinations of mass, density, size, and composition of the meteoroids arriving in the outer atmosphere at approximately 120 km are far from consistent. There are almost as many flux interpretations, for example, as there are investigators, as can be seen in Figure 1. It must be emphasized that any determination of flux at a greater distance is, at this time, entirely dependent upon the dtermination of flux at an altitude of $\sim$ 120 km.

Recommended Near-Earth Flux

The flux at an altitude of $\backsim$ 120 km constitutes one of the fundamental building blocks for any extrapolation of flux outward from Earth. The exact determination of this flux is difficult. A thorough analysis of the data and literature have, however, led to the conclusion that the flux is not necessarily represented by a straight line log-log plot. Some of the considered data are shown in Figure 2, together with the recommended variable-slope near-Earth flux distribution.

This distribution is represented by the following equations for each curve segment:

Curve Segment	Equation (MKS)	Equation (English)
$0 < M_V < +12$	$\log n = -12.95 - \log m$	$\log n = -9.05 - \log m$

$+12 < M_V < +21$	$\log n = -14.50 - 1.35 \log m$	$\log n = -10.60 - 1.35 \log m$
$+21 < M_V < +30$	$\log n = -17.30 - 1.70 \log m$	$\log n = -13.40 - 1.70 \log m$

where m is the mass in grams and n the flux in meteoroids/m^2/sec or meteoroids/ft^2/day. The basis for this distribution is discussed in some detail in References (c) and (d).

The recommended mass assignment is 2.5 grams for a zero magnitude meteor. The mean velocity is assumed to be 30 km/sec for the entire mass range, although it is suspected the mean velocity may decrease for meteoroids of smaller masses. The following mean density assignments are also recommended.

Magnitude Range	Assumed Mean Density (gm/cm^3)
$-10 < M_V < 0$	3.5
$0 < M_V < +12$	1.0
$+20 < M_V < +30$	3.5

The Meteoroid Flux in Cis-Lunar Space

The Earth's gravitational field acts as a "meteoroid sink," therefore, flux convergence is realized in space near Earth. Determining the nature and extent of this convergence for micrometeoroid-size particles (also under the influence of magnetic, electrical, and drag forces) is practically impossible from the sparse data available today. It is recommended that the near-Earth micrometeoroid flux (particles of $M_V > +20$) be used for the entirety of cis-lunar space.

The effect of convergence on the flux of penetration-size meteoroids ($0 \quad M_V < +12$) has been tentatively solved by the use of the gravitational lens concept. The relative meteoroid flux is a function of distance from the center of the Earth and elongation from the Sun. Elongation from the Sun, λ, is defined as the angle measured eastward along the ecliptic from the Sun. In general, the flux increases to a maximum value before a distance of seven effective Earth radii is reached. After the maximum is reached, the flux slowly decays to a value of $\backsim 78$ percent of the near-Earth flux. This is illustrated in Figure 3 where the flux is shown as a function of distance from Earth for two representative values of λ.

The flux varies markedly with λ just above the sensible atmosphere as shown in Figure 4. As the distance from Earth increases, the presence of the Earth becomes increasingly unimportant. At a distance of 10 effective Earth radii, the flux varies only slightly with λ. The composite effect is best illustrated on the plot of iso-flux lines shown in Figure 5.

[redacted]

The Meteoroid Flux at the Lunar Surface

The Moon exposes on face to the Earth, and it is the sub-Earth point of this face which we shall call the lunar surface. The problems of interest to this study are the evaluation of any effect which the Earth's gravitational lens might have upon the flux at the lunar surface in terms of the near-Earth flux.

These problems have been tentatively solved and the results presented in Reference (a). It was found that the flux of penetration-size meteoroids at the lunar surface varies between 23 and 55 percent of the near-Earth flux. This flux variation is caused by the spatial anisotropy of the flux and is due primarily to the Moon's movement along its orbit.

The variation of the lunar surface flux is shown in Figure 6. Comparative fluxes are shown in Figure 7. The flux contribution of secondary ejecta, resulting from meteoroid impact on the lunar surface and reported to be possibly as high as 10^4 times the primary flux, Reference (e), has been omitted because of the lack of substantiating data.

Comparison of Near-Earth Flux Distributions

Figure 8 presents a comparison of the recommended near-Earth flux distributions. Three of the distributions agree remarkably well in the range of puncture-size meteoroids. The fourth distribution, from Reference (h), is the Whipple 1961 distribution which is more severe than the rest. Whipple has recently (1963) revised his flux distribution. His new distribution is shown in Figure 1 and agrees more closely with the other three distributions. The distribution recommended in this report for micrometeoroid-size particles is based upon satellite data (Figure 2) and is essentially the flux distribution recommended by McCracken and Alexander, Reference (i). The lower NASA flux of Reference (g) is probably more suitable to "undisturbed or free space" at 1 A.U. from the Sun. The higher flux seems well substantiated by various observational techniques, Reference (j); therefore, the NASA flux probably greatly underestimates the near-Earth flux.

Comparison of Densities

The recommended mean densities disagree among all three sources, but are within recognized limits. The table on the following page summarizes the differences.

Comparison of Mean Velocity Assignments

Reference (f) utilizes the Whipple mean velocity assignment. References (g) and (h) make no complete velocity assignments. Flight Sciences recommends a mean velocity of 30 km/sec over the entire mass range. While the general form of the Whipple velocity assignment might be intuitively inferred from physical arguments, no measurements exist over most of the mass range to confirm or deny any assignment. It is believed the assignment of 30 km/sec will serve as a reasonable maximum estimate until the velocity distribu-

[redacted]

[redacted]

SOURCE	PUNCTURE-SIZE METEOROIDS			MICROMETEOROIDS
	Type	% of Total	Density (gm/cm^3)	Density (gm/cm^3)
NASA, Ref. (f)	Cometary Asteroidal	90 10	.5 3.5	.5 and 3.5
NASA, Ref. (g)	Cometary Asteroidal	Dominate $M_V > +3$ Dominate $M_V < +3$	.05 3.-8.	.5
LOP, Ref. (h)	-	-	1.0*	1.1*
Flight Sciences	Asteroidal -	$M_V < 0$ $0 < M_V < +12$	3.5 1.0	3.5

*Varies as a function of mass

tion of micrometeoroids can be measured.

Cis-Lunar and Lunar Environment Comparisons

The NASA and LOP environments attempt to approximate the cis-Lunar and Lunar fluxes by applying the near-Earth flux over all regions. Flight Sciences has quantitatively analyzed the problem and has presented numerical values for the penetration-size flux reduction in cis-Lunar space and on the lunar surface.

Shower Data Comparisons

Reference (f) presents shower data, but assumptions must be made to convert it to flux values. Reference (g) presents some standard shower data. Meteoroid showers contribute possibly less than 10 percent of the total meteoroid flux, Reference (k). It is difficult to discriminate, however, between shower meteors and "sporadic" meteors. Most workers include all but the few pronounced showers in any "sporadic meteor" survey. It is believed that shower activity constitutes a minor consideration because most slight showers can be considered to be in the "sporadic meteor" background.

General Conclusions from the Comparisons

The following general conclusions are seen to be indicated from a study of the NASA and LOP environments:

1. The three environments present no recommendations which can be considered at complete variance with known facts.

2. All three proposed environments lack completeness.

3. Reference (g) appears to be a very studied opinion with no dis-

cernable technical mistakes. The LOP environment seems based on considered thought; any technical errors may be considered minor. Reference (f), while not wrong, utilizes some concepts which are highly questionable.

Application of the Recommended Meteoroid Environment

There are several comments which should be made in applying the recommended meteoroid environment to any calculations:

1. Flux as defined in this report is considered to eminate from all directions and pass through a unit area in unit time in both directions. The walls of any structure will receive flux from only a hemisphere of view; therefore, it is necessary to halve either the calculated flux or the exposed structure area.

2. The flux reductions in cis-Lunar space and on the Lunar surface result from physical causes and are, therefore, additional to the above reduction. Eighty percent of near-Earth flux can be adopted as an average cis-Lunar space value. An average value of forty percent near-Earth flux may be assumed for any point on the Lunar surface provided the exposure exceeds several months' duration.

References

(a) Bradford, D. C. and R. D. Dycus, "Some Aspects of the Problem of the Meteoroid in Motion," Paper presented at theAIAA Symposium, Space: The Environment and the Effects, Los Angeles, California, 25 April 1963.

(b) Bradford, D. C., "Particulate Environment in Solar Space - Part I," SID 62-1066-1, December 1962.

(c) Bradford, D. C. and R. D. Dycus, "Interim Meteoroid Environment Model for Apollo," FS/63-218, 16 July 1963.

(d) Dycus, R. D. and D. C. Bradford, "The Particulate Environment in Solar Space - Part IV," SID 62-1066-4, to be published.

(e) Gault, D. E., E. M. Shoemaker, and H. J. Moore, "Spray Ejecta From the Lunar Surface by Meteoroid Impact," NASA TN-D-1767, April 1963.

(f) "Apollo Environmental Criteria Specification," NASA; Houston, Texas, 25 March 1963.

(g) "The Meteoroid Environment of Project Apollo," contained in "Data for Lunar Base Studies," Study by Bell Comm, Inc., 31 January 1963.

(h) Lunar Occupancy Payload Study Report, SID 62-1473, 4 January 1963.

(i) McCracken, C. W. and W. M. Alexander, "The Distribution of Small Interplanetary Dust Particles in the Vicinity of Earth," NASA TN-D-1349, July 1962.

(j) Whipple, F. L., "The Dust Cloud About the Earth," Nature, 189, 127-128, 1961.

(k) Hawkins, G. S., "A Radio Echo Survey of Sporadic Meteor Radiants," Mon. Not. Roy. Astron. Soc., 116(1):92-104, 1956.

FIGURE 1

Some Suggested Near-Earth Flux Distributions

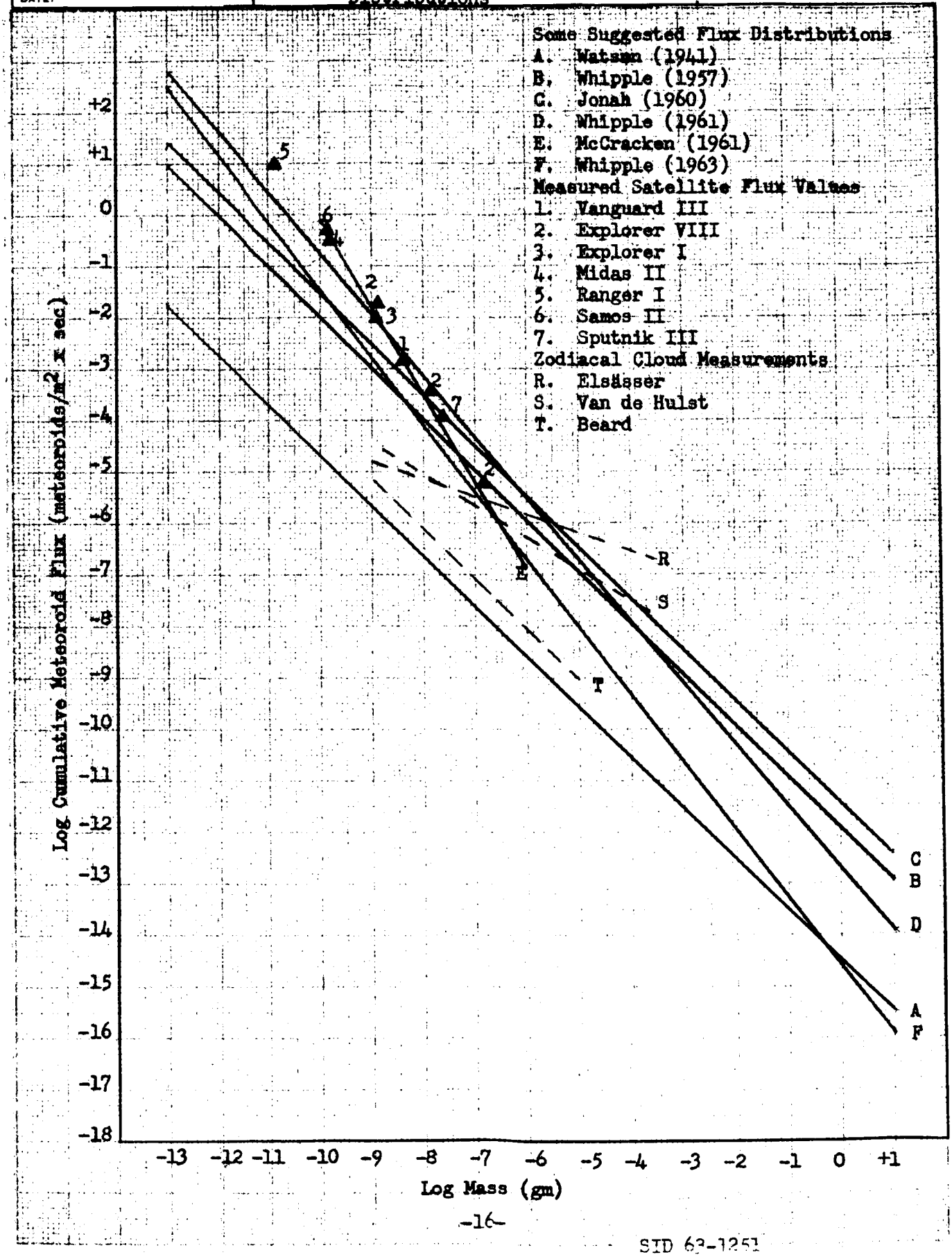
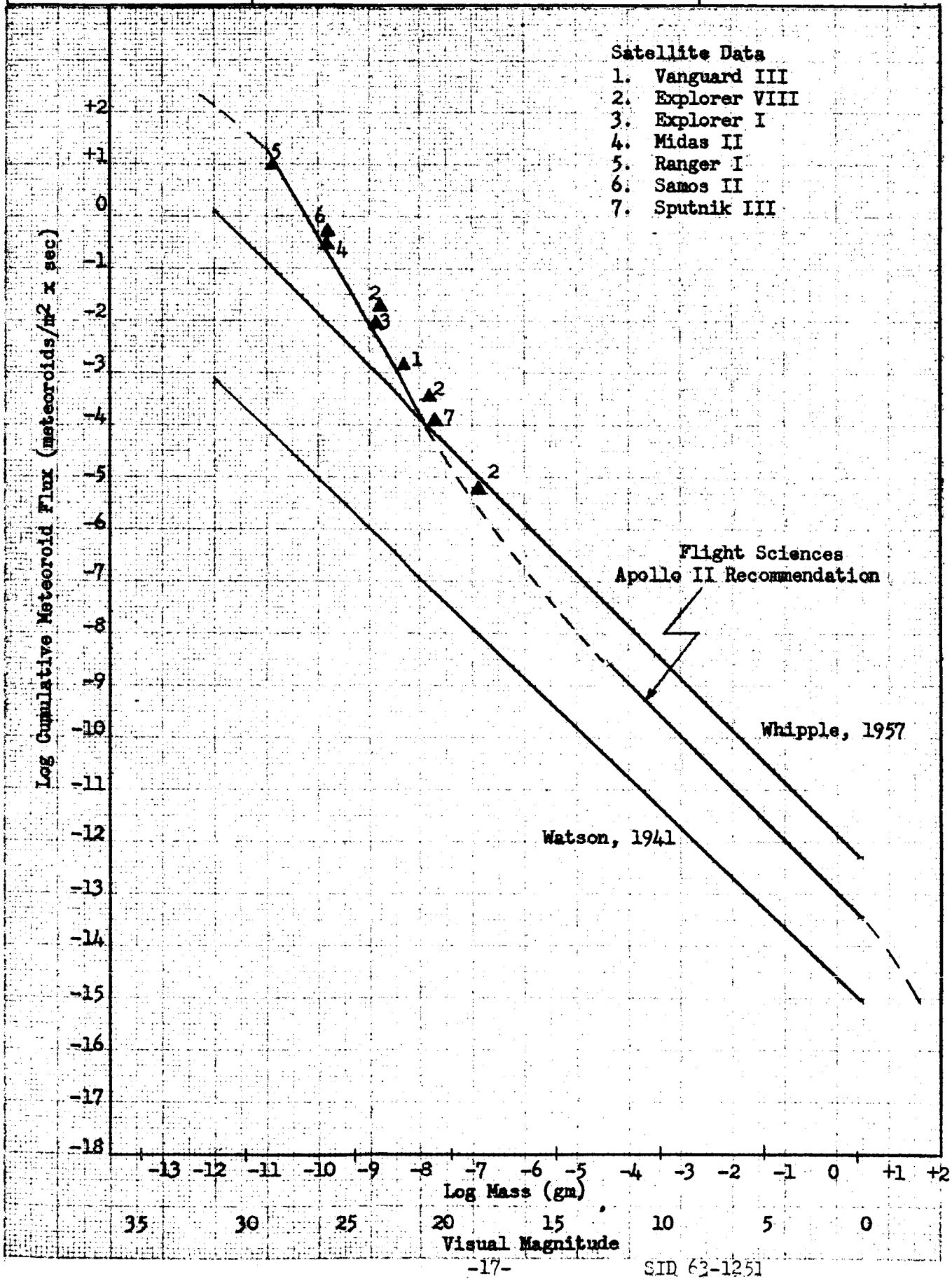


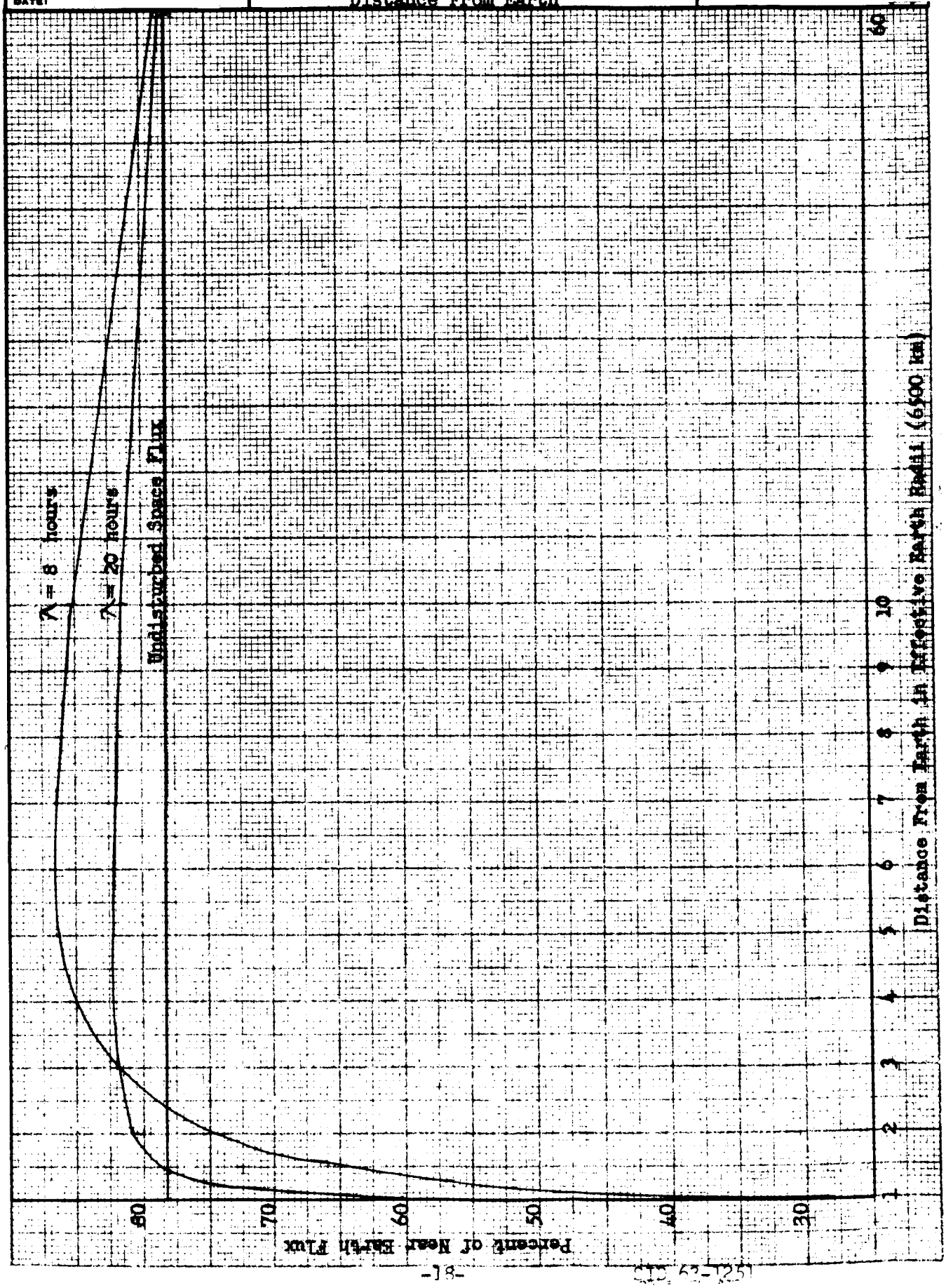
FIGURE 2

Near-Earth Flux Distribution



FORM M-113-N REV. 5-61

PREPARED BY:	NORTH AMERICAN AVIATION, INC. SPACE and INFORMATION SYSTEMS DIVISION	
CHECKED BY:	FIGURE 3	REPORT NO.
DATE:	The Variation of Flux as a Function of Distance From Earth	



SID 63-1251

FORM M-113-N2- (REV. 3-61)

PREPARED BY:	NORTH AMERICAN AVIATION, INC. SPACE and INFORMATION SYSTEMS DIVISION	PAGE NO.
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DATE:	The Variation of Flux as a Function of λ for Several Distances From Earth	MODEL NO.

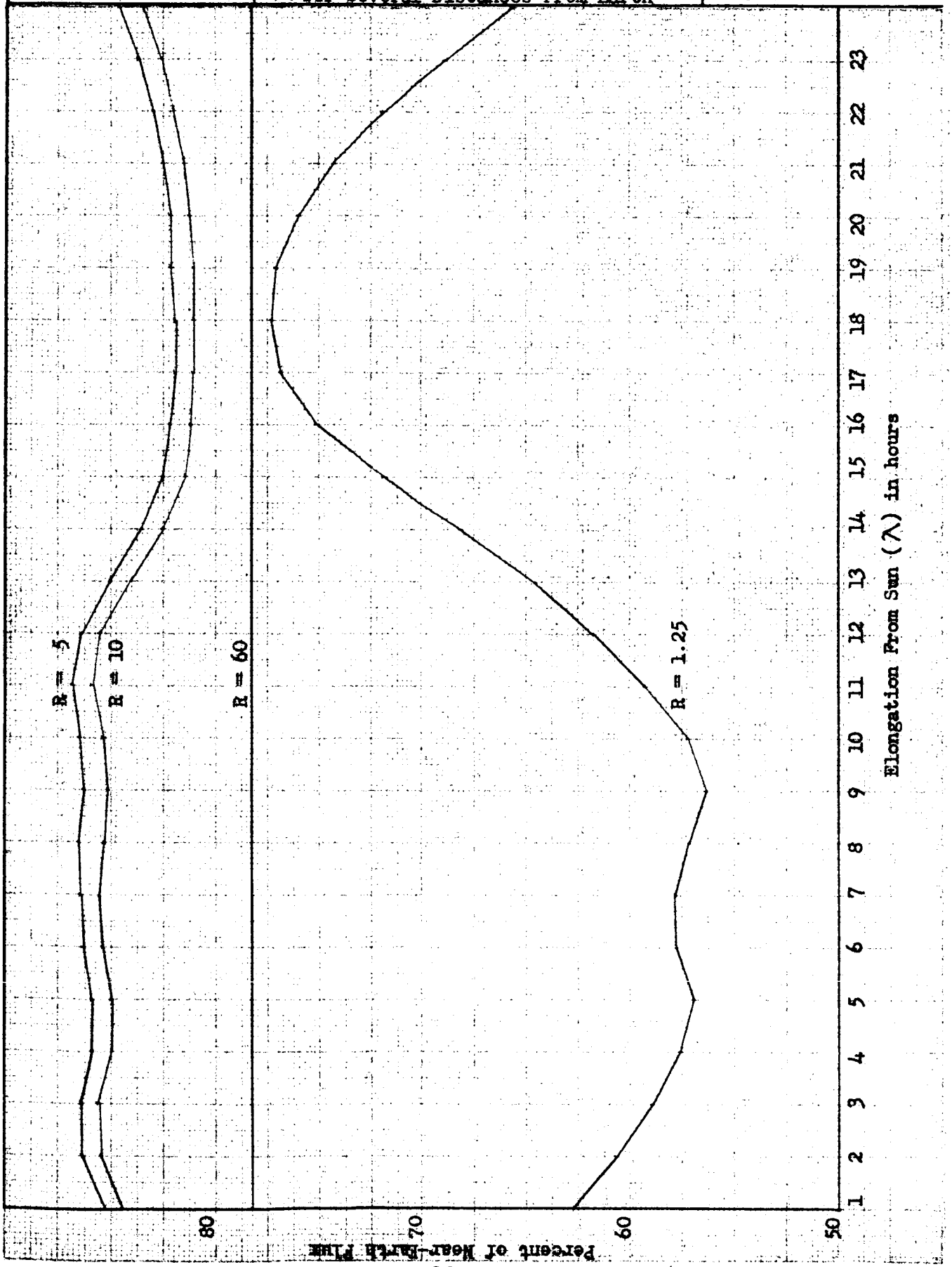
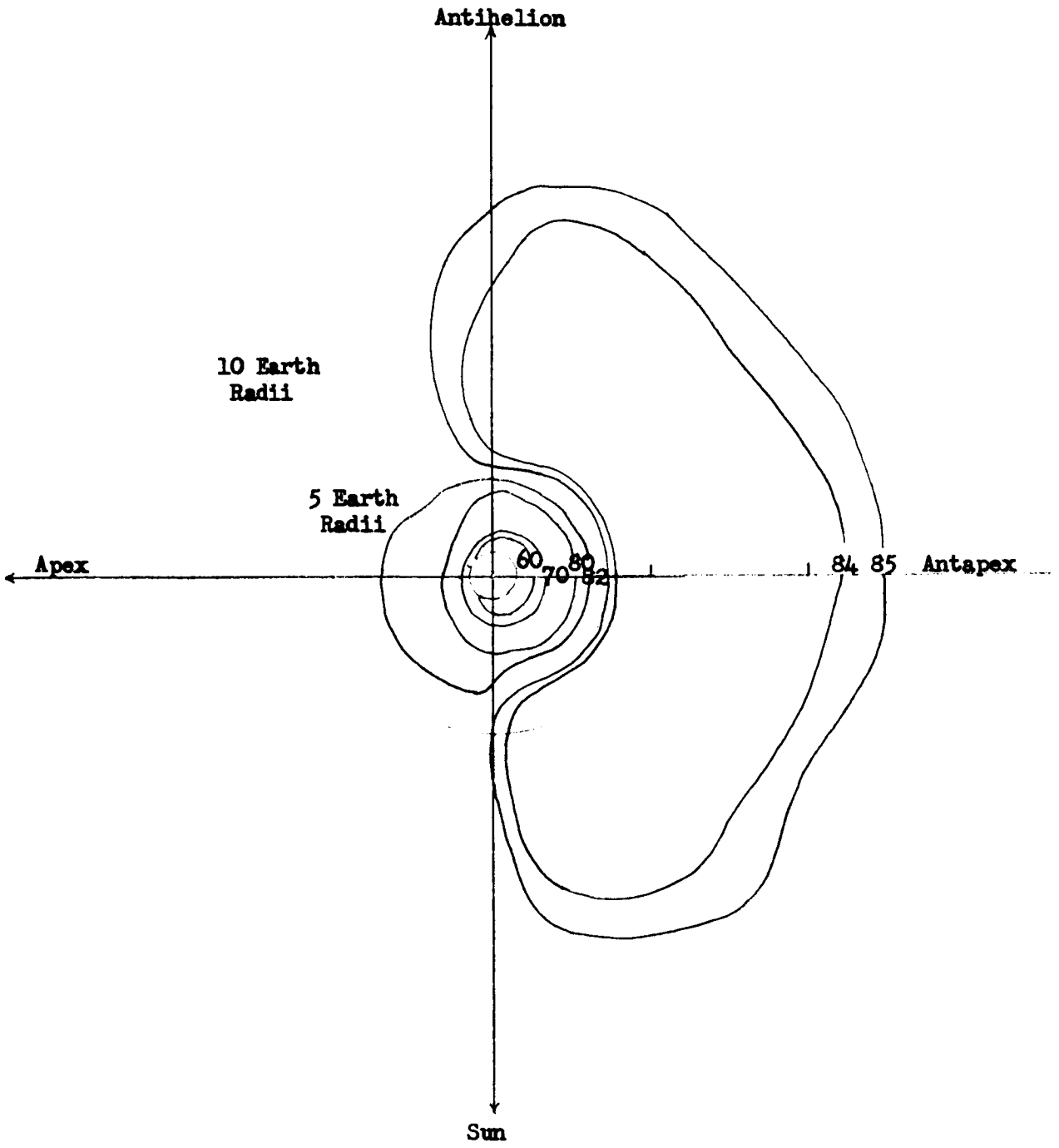


FIGURE 5

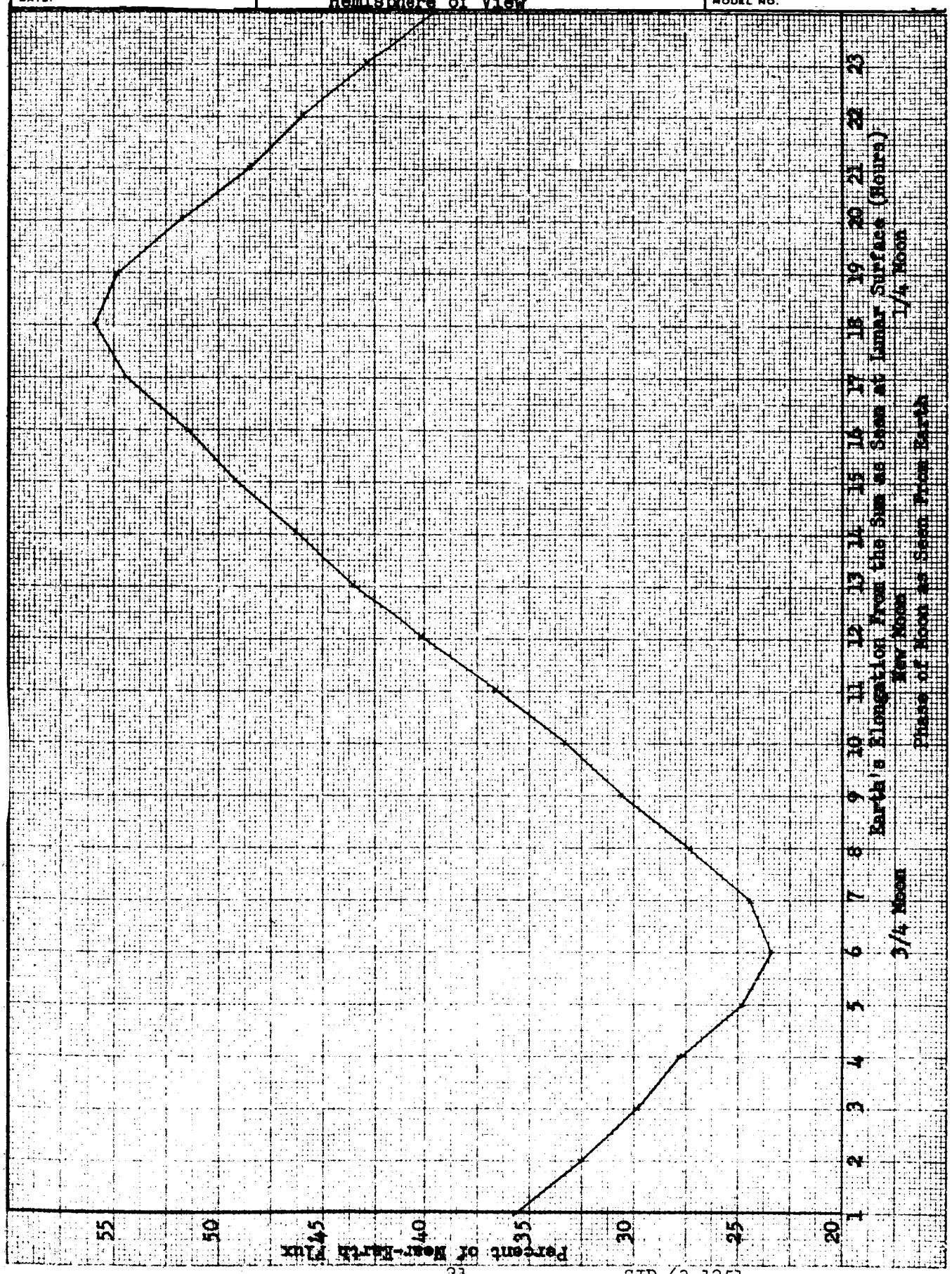
Polar Diagram, Iso-Flux Lines



Contours are in percent of near-Earth flux.

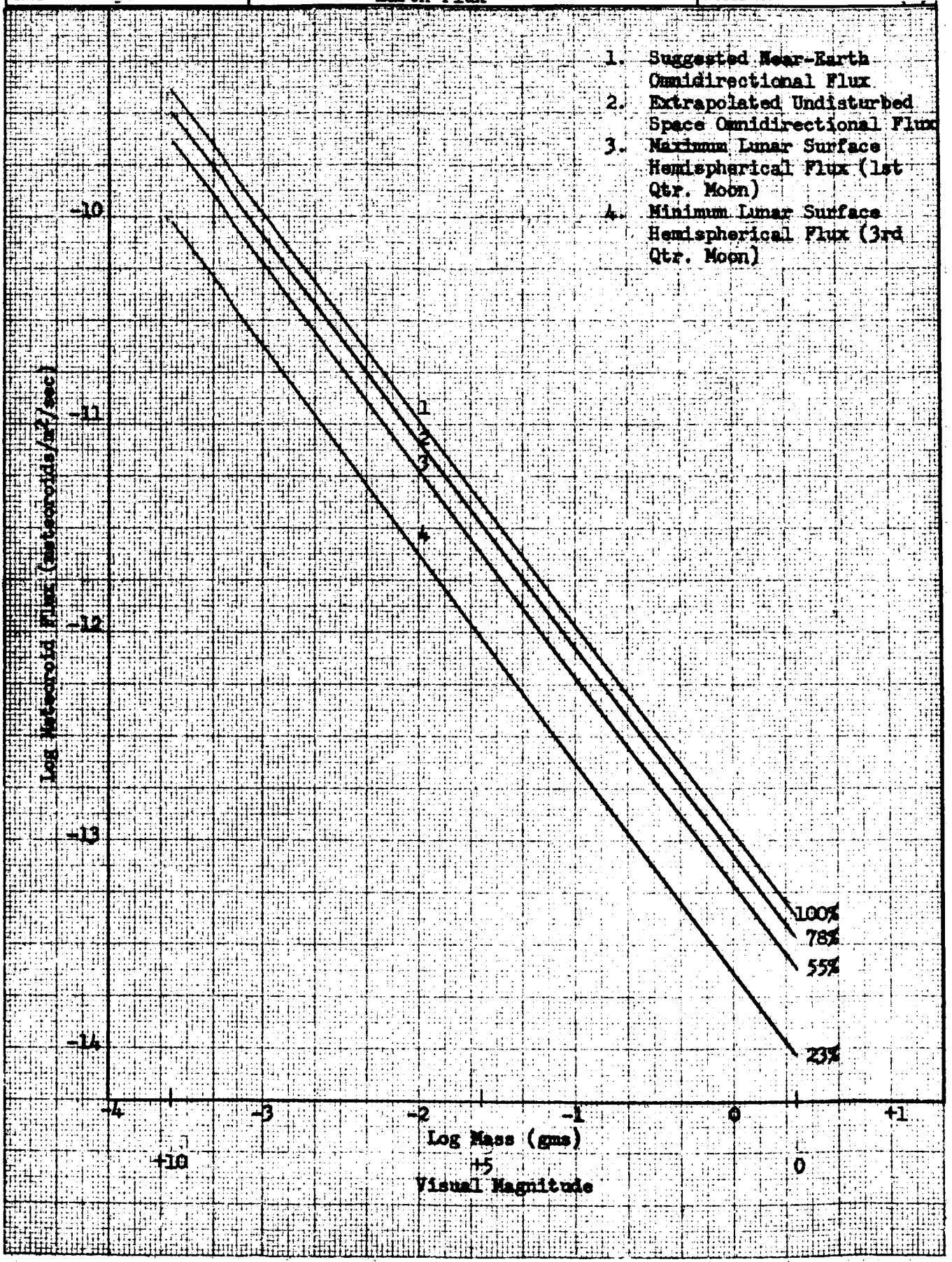
FORM M-113-N2- (REV. 3-61)

PREPARED BY:	NORTH AMERICAN AVIATION, INC. SPACE and INFORMATION SYSTEMS DIVISION	PAGE NO.
CHECKED BY:	FIGURE 6	REPORT NO.
DATE:	Variation of Lunar Surface Flux with Hemisphere of View	MODEL NO.



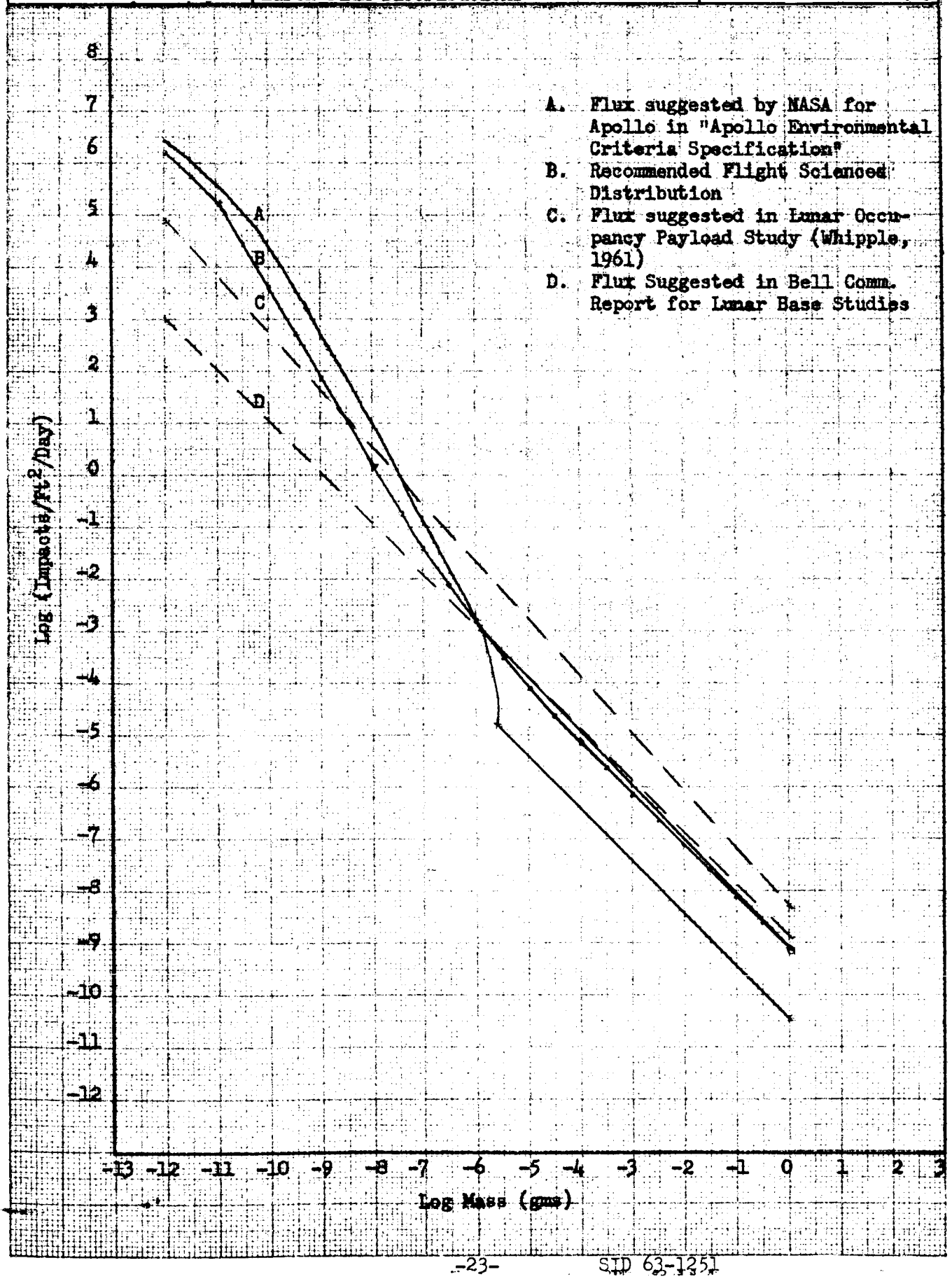
FORM M-113-N2- (REV. 3-61)

PREPARED BY:	NORTH AMERICAN AVIATION, INC. SPACE and INFORMATION SYSTEMS DIVISION	PAGE NO.
CHECKED BY:	FIGURE 7	REPORT NO.
DATE:	Lunar Vs. Undisturbed Space Vs. Near Earth Flux	MODEL NO.



FORM M-113-N2- (REV. 3-61)

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CHECKED BY:	FIGURE 8	REPORT NO.
DATE:	Comparison of Four Recommended Near Earth Flux Distributions	MODEL NO.



GALACTIC AND SOLAR RADIATION

Apollo II vehicles will encounter radiation from the Van Allen belts and solar plasma as well as solar and galactic corpuscular radiation. The radiation exposure to the Van Allen belts was discarded for the Apollo II studies, because of the short exposure time. The solar plasma exposure model was omitted pending more detailed definition of the trajectories and base design. However, no problems for the Apollo II vehicle due to Van Allen belts and solar plasmas are foreseen at this time.

Solar and Galactic Radiation Model

The primary radiations of concern to the Apollo II vehicle on the lunar surface are the galactic radiation and solar high energy particle emissions (called henceforth solar events), associated with certain major flares.

Figs. 9 and 10 present a comparison of the rigidity spectra for the galactic radiation and the solar events as published by various authors.

Of the spectra shown, it is recommended that for Apollo II:

a) the solar event maximum yearly time integrated flux vs rigidity be represented by

$$dN = 1.8 \times 10^{11} e^{-12.5P} dP \quad \text{protons-cm}^{-2}\text{-sr}^{-1}\text{-Br}^{-1} \qquad (1)$$

b) the maximum yearly time integrated galactic radiation flux vs rigidity be represented by

$$dN = 3.5 \times 10^{6} e^{-0.7P} dP \text{ protons-cm}^{-2} \text{-sr}^{-1} \text{-Bv}^{-1} \qquad (2)$$

This is 3.17×10^{-7} seconds times the lower dotted curve in Figure 1, which, in turn, may be approximated by

$$dN = 1.13 \times 10^{7} (1 + E)^{-2.5} dE \text{ P -cm}^{-2} \text{ -sr}^{-1} \text{ -Bev}^{-1}$$

The yearly time integrated flux of galactic corpuscular radiation varies inversely with respect to solar activity as shown in Table I. The minimum yearly time integrated flux of galactic radiation occurs near the maximum of the spot number cycle.

The maximum yearly time integrated flux from solar events, however, is expected about a year after the peak in the spot number cycle. The yearly time integrated flux values in Table I are recommended for use with Apollo II as normalizing factors for the corpuscular radiation integral spectra, Equations (1) and (2). Both environments are to be considered as isotropic over a half hemisphere on the lunar surface.

TABLE 1

Year Observed	Yearly Average Spot No.	Integrated Flux $p\text{-}cm^{-2}\text{-}year^{-1}$	Future Year*	Yearly time integrated solar event flux $(p - cm^2)$
1954	3	-	-	-
1955	23	-	-	-
1956	118	1×10^8	1965, 66, 67, 78	1×10^{10}
1957	181	7×10^7	1968, 1979	1.8×10^{10}
1958	196	6×10^7	1969, 1980	4.2×10^{10}
1959	168	6×10^7	1970, 1981	1.9×10^{11}
1960	119	8×10^7	1971, 1982	6.2×10^{10}
1961	64	1×10^8	1972, 73, 74, 83	-
1962	38	-	-	-

*An eleven-year solar activity cycle is assumed. This is an expedient but somewhat questionable assumption, since the spot number cycles have been observed to deviate by more than 35% from the eleven-year value.

Comparison of Model Solar High Energy Proton Event Spectra

During the past several years, the following solar proton event spectra have been used for shielding studies:

a) The 10 May 1959 event -constant flux from 1 hour after prime flare onset to 32.5 hours with a time squared decrease thereafter

$$\frac{dN}{dE} = 3.3 \times 10^{-16} E^{-4.8} \text{ protons-cm}^{-2}\text{-Mev}^{-1} \qquad (3)$$

where E is the kinetic proton energy and dN/dE is the differential time integrated number flux per Mev energy interval. This spectrum was used for the Apollo proposal effort.

b) The 10 May 1959 event with a hypothetical time dependent spectrum which, after time integration, gives:

$$\frac{dN}{dE} = \begin{cases} 3.14 \times 10^{12} E^{-2.07} & \text{for } 5 \leq E \leq 60 \text{ Mev} \\ 6.89 \times 10^{15} E^{-3.95} & \text{for } 60 \leq E \leq \end{cases} \qquad (4)$$

$$(\text{protons-cm}^{-2}\text{-Mev}^{-1})$$

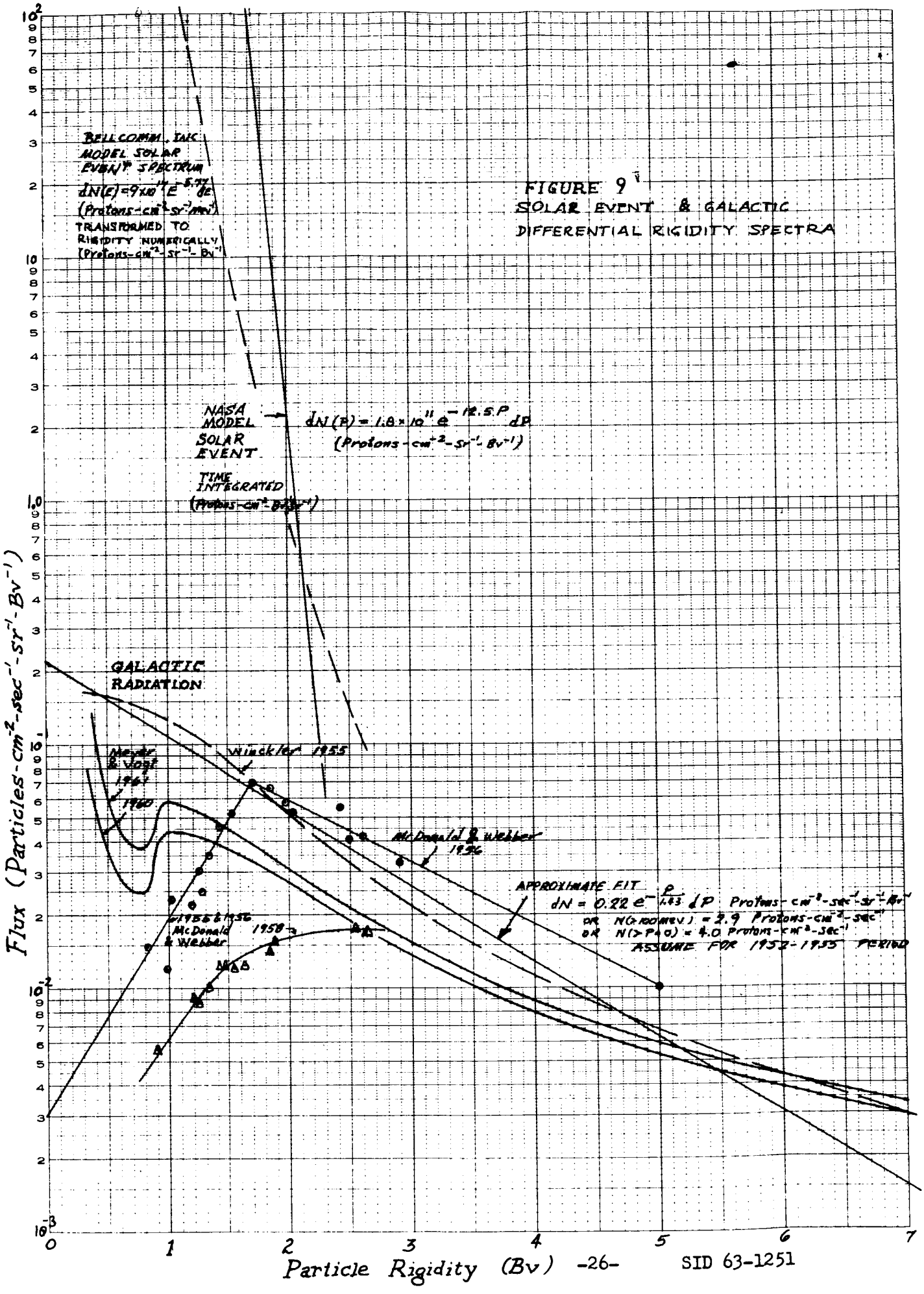
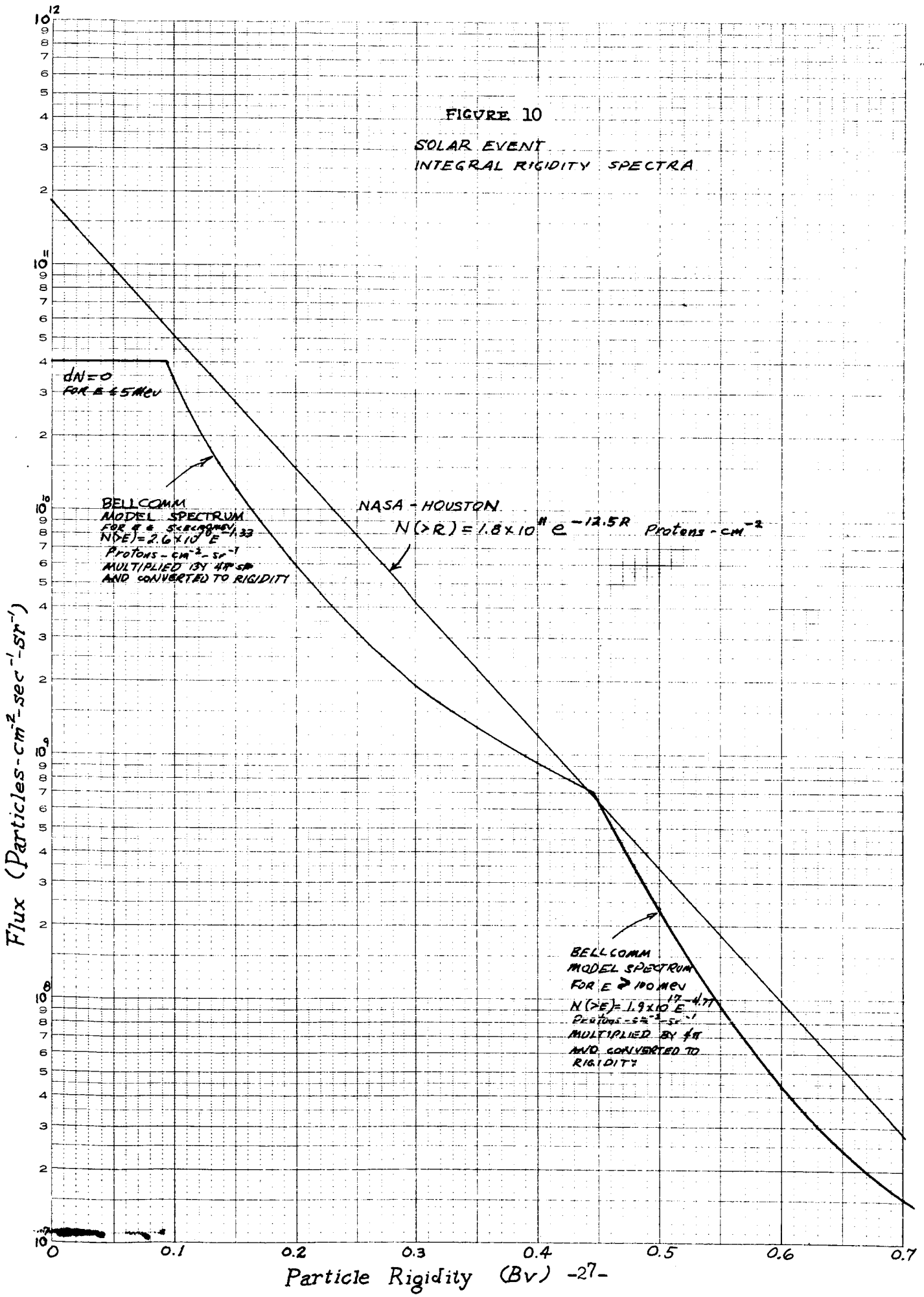


FIGURE 9
SOLAR EVENT & GALACTIC DIFFERENTIAL RIGIDITY SPECTRA


FIGURE 10
SOLAR EVENT
INTEGRAL RIGIDITY SPECTRA
dN=0
FOR E ≤ 5 Mev
BELLCOMM
MODEL SPECTRUM
FOR E ≥ 5 Mev
N(>E) = 2.6 × 10^8 E^-1.33
Protons - cm^-2 - sr^-1
MULTIPLIED BY 4π sr
AND CONVERTED TO RIGIDITY
NASA - HOUSTON
N(>R) = 1.8 × 10^11 e^-12.5R Protons - cm^-2
BELLCOMM
MODEL SPECTRUM
FOR E ≥ 100 Mev
N(>E) = 1.9 × 10^17 E^-4.77
Protons - cm^-2 - sr^-1
MULTIPLIED BY 4π
AND CONVERTED TO
RIGIDITY
Flux (Particles - cm^-2 - sec^-1 - sr^-1)
Particle Rigidity (Bv)
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7

This was the first spectrum used for Apollo contractural shielding studies. The specific time dependence and method of time integration resulting in Equation (2) were never presented by NASA-Houston to NAA.

c) The 23 February 1956 event prompt (1 hour after flare) and the 19 hour spectra obtained from sea level neutron monitors and extrapolated to the top of the atmosphere were also used for the Apollo proposal effort. The spectra were presented in graphical form. A close fit to these two spectra are

$$\frac{dN}{dE} = 5.72 \times 10^{2} \, (E+938) \, (E^2+1876E)^{-\frac{1}{2}} \, e^{-\frac{\sqrt{E^2+1876E}}{639}}$$

$$(\text{protons-cm}^{-2}\text{-sec}^{-1}\text{-Mev}^{-1})$$

for time = 1 hour (5)

$$\frac{dN}{dE} = 7.8 \times 10^{1} \, (E+938) \, (E^2+1876E)^{-\frac{1}{2}} \, e^{-\frac{\sqrt{E^2+1876E}}{128}}$$

for time = 19 hours after flare

Neither the time of the peak flux nor the detailed time distribution was determined accurately. A t^{-2} time decay of the integral flux from the first hour has been used for very rough calculations of the time integrated spectrum. Only recently has a method been developed for handling the time dependent change in spectral shape discussed below.

d) D. K. Bailey (J. Geo. Res. 67, 391, 1962) presented a family of synthetic solar proton time dependent integral energy spectra for varying times after a major flare. Bailey considered a number of spectra observed at different times after flares. He correlated the spectral shape with time after each prime flare. Assuming that all such events have about the same time dependence, he constructed a composite model. Although many of the larger events lasted longer than two days, the Bailey model is defined up to only 32 hours. The integral flux over all magnetic rigidity is 1.0×10^{9} protons-cm^{-2} according to calculations carried out by Pearce on Bailey's model (SID 63-1021).

Bailey's time integrated integral energy model spectrum represents an envelope spectrum made up of the largest values of the individual spectra. The 23 February 1956 event spectrum was dominant in Bailey's analysis.

Bailey made no distinction between the high energy-low flux and low energy-high flux events, nor was any consideration given to

differences in anisotropy and in flux rise times, which recently have been discussed in the literature by McCracken (J. Geo. Res. 67, 423, 1962). One of the largest events (16 July 1959) in Bailey's comparison taken by a factor of 15.5 too low by mistake when it would have dominated over the 23 February 1956 event total flux.

e) A study similar to Bailey's has been performed by NASA-Houston by Modisette with resulting time integrated spectrum

$$\frac{dN}{dP} = 2.25 \times 10^{12} \, e^{-12.5P \, (Bv)} \quad (6)$$

$$(\text{protons-cm}^{-2}\text{-Bv}^{-1})$$

or

$$\frac{dN}{dE} = 2.25 \times 10^{9} \, (E+938) \, (E^2+1876E)^{-\frac{1}{2}} \, e^{-\frac{\sqrt{E^2+1876E}}{80}}$$

$$(\text{protons-cm}^{-2}\text{-Mev}^{-1})$$

The NASA-Houston model represents an event expected with a frequency of 1% during an entire spot number cycle. That is, during the last cycle, only 1% of the events exceeded the total integral flux of the NASA model.

f) A theoretically derived time dependent model differential rigidity spectrum was presented by Pearce in SID 63-1021. This model was normalized to match Bailey's peak flux and terminal spectral slope:

$$\frac{dN}{dP} = 1.3 \times 10^{12} \, e^{-13.5P(Bv)} \quad (7)$$

$$(\text{protons-cm}^{-2}\text{-Bv}^{-1})$$

or

$$\frac{dN}{dE} = 1.3 \times 10^{9} \, (E+938) \, (E^2+1876E)^{-\frac{1}{2}} \, e^{-\frac{\sqrt{E^2+1876E}}{74}}$$

$$(\text{protons-cm}^{-2}\text{-Mev}^{-1})$$

g) The model integral spectrum presented in the Bellcomm Lunar Base Report, Appendix C, is an extrapolated version of Bailey's model flare. The spectrum is approximated by two power functions in energy:

$$N\,(>E) = 3.3 \times 10^{11} \, E^{-1.33} \text{ for } 5 < E < 10 \text{ Mev}$$

$$= 2.4 \times 10^{18} \, E^{-4.77} \text{ for } E > 100 \text{ Mev}$$

(protons-cm^{-2})

with a total flux of 6 x 10^9 protons-cm^{-2} with energy greater than 30 Mev. For a 10-day lunar mission, there would be a 1% chance of occurrence of a proton event with greater than 6 x 10^9 protons-cm^{-2}.

By differentiation, the differential energy spectra corresponding to (6) are

$$\frac{dN}{dE} = 4.3 \times 10^{11} E^{-2.33} \quad \text{for } E < E < 100 \text{ Mev}$$

$$= 1.15 \times 10^{19} E^{-5.77} \quad \text{for } E > 100 \text{ Mev} \qquad (9)$$

(protons-cm^{-2}-Mev^{-1})

Comparison of Model Spectra

Figure 11 presents a comparison of the spectra, Equations (6), (7), and (9). It can be seen that the Bellcomm spectrum represents a jump discontinuity at the 100 Mev point of transition. Such a jump could not be resolved by available sensing systems, and is not realistic. It is acceptable to fit a differential spectrum piecewise continuously by two power functions, but not an integral spectrum which should be smooth, curved, and only asymptotically related to power functions. Power function spectra do not have a finite integral over-all energy.

The Modisette and Pearce spectra are within a factor of 2 in integral flux and have nearly equal spectral shapes. The Pearce spectrum is to be used for the Ames Mars Study, but the Modisette curve for NASA-Houston efforts.

The Pearce spectra is defined analytically as a function of time, thus can be used as presented in SID 63-1021, p 21 for flux, or dose vs time analysis. The method of time dependence in SID 63-1021 may be utilized for the NASA-Houston spectrum very simply by changing the spectral constants by increasing the integral flux by 2 and by reducing their terminal slope from 13.5 to 12.5 billion volts in Equation (5) or in the equations on p 21 of SID 63-1021. The differences between the constants are physically small compared to the accuracy of the input data.

LUNAR SURFACE TEMPERATURE

Recommended Lunar Surface Temperature

As the following analysis will indicate, the recommended lunar surface temperatures are those shown in Figure 17, i.e., infrared temperature for the center of the disc ranges from 120°K at midnight to 394°K at noon. Latitude variations of these "center of disc" temperatures are distributed as $T_o \cos$

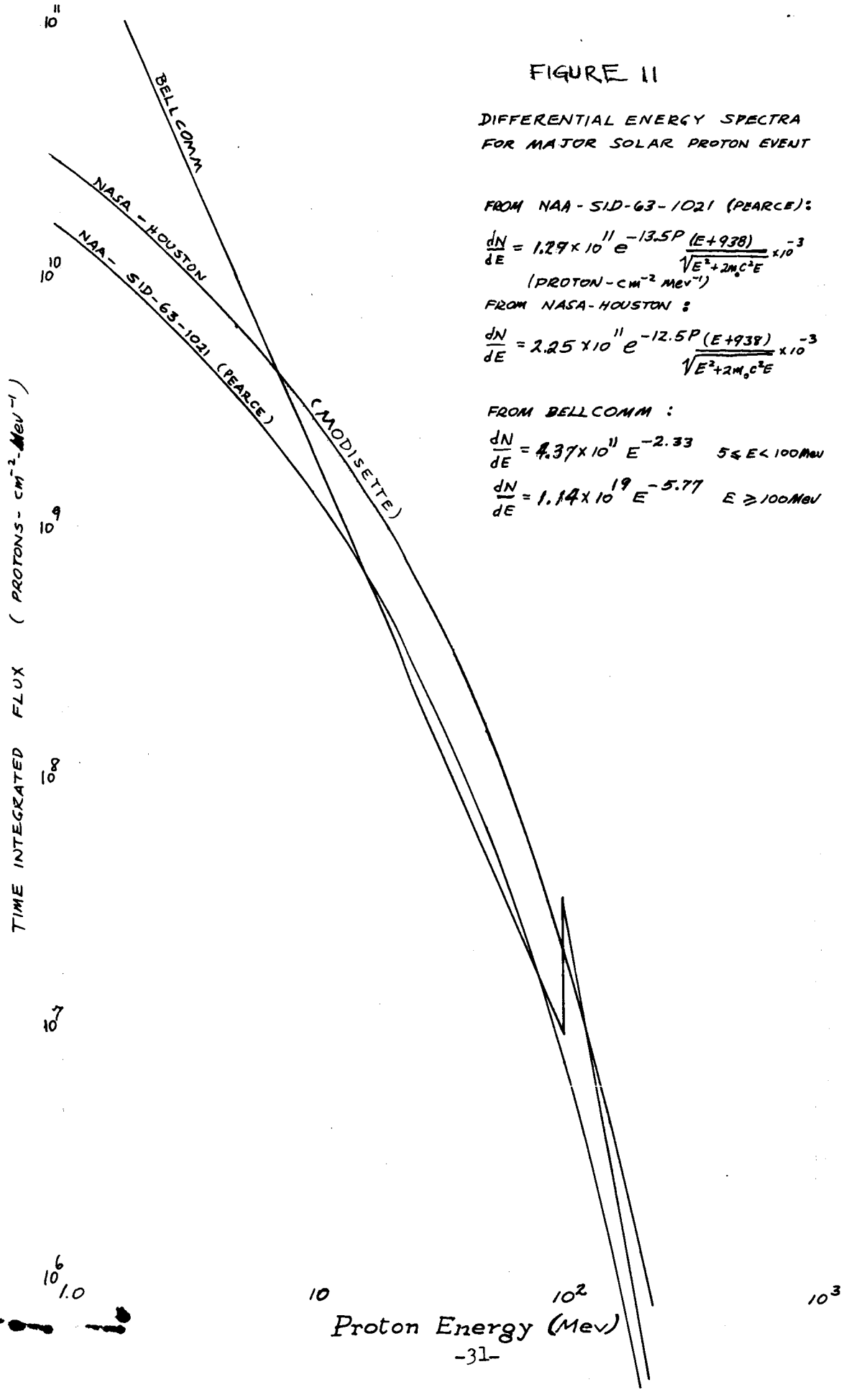


FIGURE 11

DIFFERENTIAL ENERGY SPECTRA FOR MAJOR SOLAR PROTON EVENT

FROM NAA - SID-63-1021 (PEARCE):

$$\frac{dN}{dE} = 1.29 \times 10^{11} e^{-13.5P} \frac{(E+938)}{\sqrt{E^2+2M_0c^2E}} \times 10^{-3}$$

(PROTON - CM^{-2} MEV^{-1})

FROM NASA - HOUSTON :

$$\frac{dN}{dE} = 2.25 \times 10^{11} e^{-12.5P} \frac{(E+938)}{\sqrt{E^2+2M_0c^2E}} \times 10^{-3}$$

FROM BELLCOMM :

$$\frac{dN}{dE} = 4.37 \times 10^{11} E^{-2.33} \qquad 5 \leq E < 100 \text{ MeV}$$

$$\frac{dN}{dE} = 1.14 \times 10^{19} E^{-5.77} \qquad E \geq 100 \text{ MeV}$$

1/6g.

Lunar Surface Temperature Determination

Lunar surface temperatures are determined from the relation:

$$\eta \sigma T^4 = E_e = E_R - E_r - E_c$$

where:

T = temperature (°K)

σ = Stefan-Boltzmann constant = 8.127×10^{-11} cal/cm^{2}min(°K)^4

η = emissivity (output coefficient) $\simeq 1$ (assumed)

E_e = energy radiated from Moon (cal/cm^{2}min)

E_R = energy received from Sun

E_r = energy reflected from Moon $\simeq 0.25$ cal/cm^{2}min at full Moon

E_c = energy conducted into Moon ≤ 0.1 cal/cm^{2}min

The energy received from the Sun is the "solar constant" at the distance from Sun and at the angle of incidence for the time of observation. Solar energy received by the center of the projected disk of Moon as seen from Earth is given by the relation:

$$E_R = E_o \cos\phi\cos(\omega t - \lambda)$$

where

E_o = "Solar constant" for the distance from Sun at time, t (cal/cm^{2}min)

λ = longitude of center of disk

ϕ = latitude of center of disk

$\omega = 360°/P$ (°/min) = 8.466×10^{-3} °/min

t = time (min)

$\lambda = 0$, $\phi = 0$, and $t = 0$ are the coordinates of the subsolar point at full Moon although, strictly speaking, $\lambda = 0$, $\phi = 0$ are the coordinates of the subsolar point at the coincident times of full Moon and ascending nodal passages of Earth's orbit about Sun and of Moon's orbit about Earth.

If Moon were a perfectly diffusing sphere, the distribution of energy

emitted over the disk at full Moon would follow Lambert's law:

$$E_e = E_{eo} \cos \theta$$

where

E_{eo} = energy emitted by subsolar point at full Moon

θ = selenocentric angle from subsolar point

Empirically, however, the energy emitted follows the relation:

$$E_e = E_{eo} \cos^{2/3} \theta$$

Visual and Infrared Observations

The corresponding temperatures for these relations are depicted in Figure 12. Curve A is from Pettit's and Nicholson's 1927 observations and Curve B is the Lambert law, based on E_{eo} = 2.25 cal/cm^{2}min (Reference 27). Curve C is from the Pulkova 1956 observations near full Moon (phase angle = 2°8), E_{eo} = 1.93 cal/cm^{2}min, and temperatures are averaged over a projected radius zone of 0.1 (Reference 44).

The large value of E_{eo} for curves A and B is due mainly to an overcorrection for atmospheric absorption in the 8-14 μ region (References 27 and 36). There is the possibility that part of the increase of the observed heat flux over the heat flux calculated from the Solar constant is due to bombardment by high energy primary particles from Sun (Reference 34).

Figure 13 depicts lunar temperatures as a function of time for the eclipses of 14 June 1927 and 27 October 1939, PST. The 1927 observations are reflected sunlight (cover-glass) from a point on the South limb. The 1939 observations are both reflected sunlight (cover-glass) and planetary heat (water-cell) for the center of the disk. The limits of the four eclipse phases, start of the eclipse, start of totality, finish of totality, and finish of the eclipse, are shown for both eclipses. The solid curve is the theoretical relation of Wesselink for the 1939 eclipse. See References (24), (26), (27), and (41).

Figure 14 shows Wesselink's theoretical curve for one lunation (Lunar day) as Curve A. Curve B is the theoretical eclipse curve from Figure 13 shown on the Lunar day scale, References (41) and (44). The point near the center is Pettit's and Nicholson's observed midnight temperature of 120°K, Reference (27).

Radio Wavelength Data

Temperatures derived from radio wavelengths show a different phase relation than do those derived from visual and infrared wavelengths. The

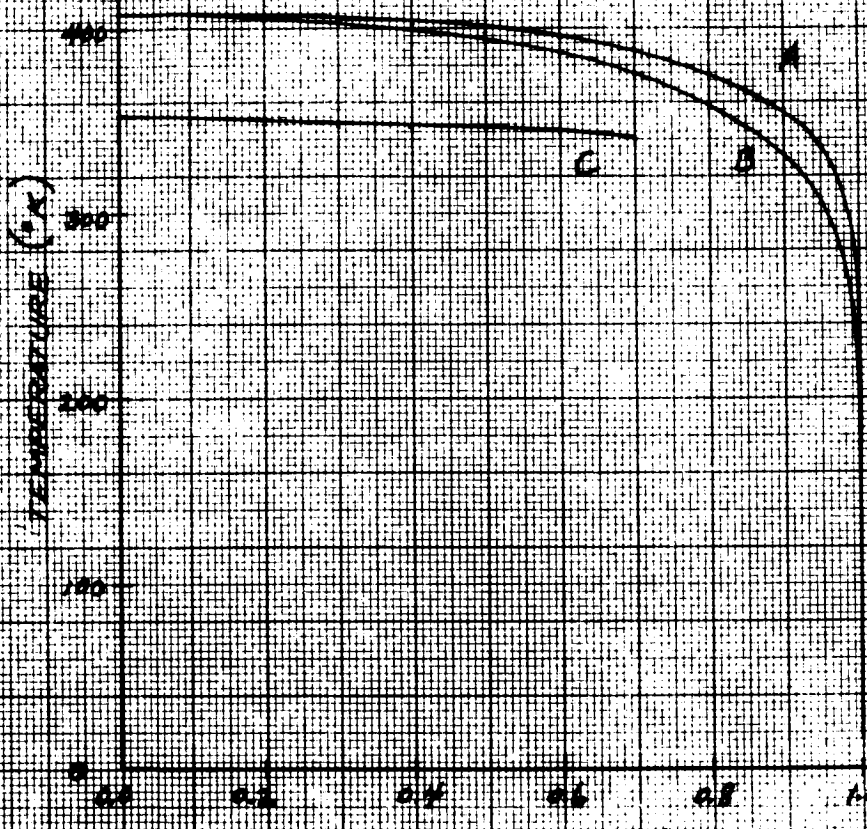


FIGURE 12 DISTRIBUTION OF TEMPERATURE OVER DISK AT FULL MOON AS A FUNCTION OF PROTECTED LUNAR RADII FROM THE SUBSOLAR POINT (REFERENCES 8, 19, 27, 44)

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DATE: 13 Sept 1963		MODEL NO.

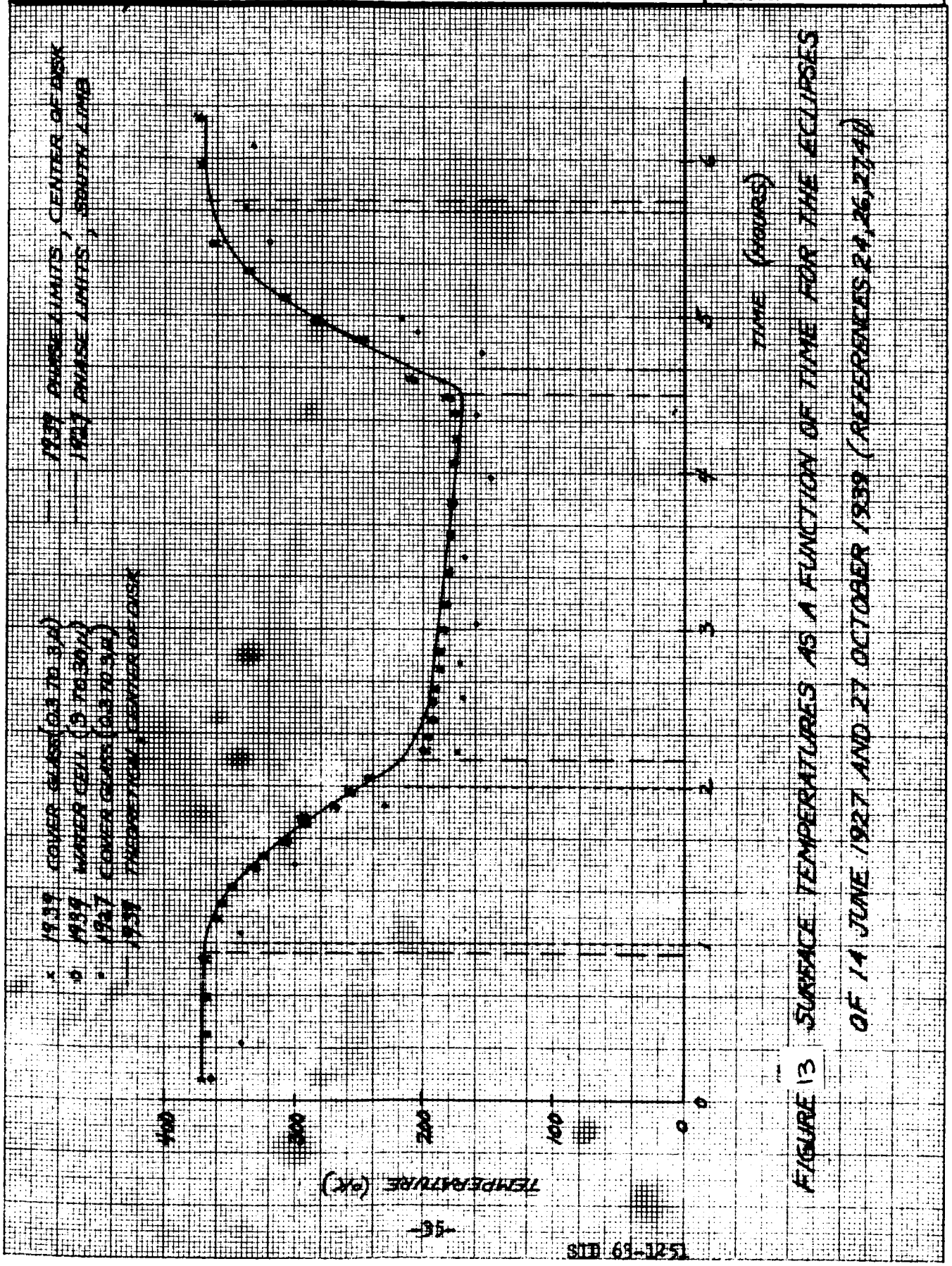


FIGURE 13 SURFACE TEMPERATURES AS A FUNCTION OF TIME FOR THE ECLIPSES OF 14 JUNE 1927 AND 27 OCTOBER 1939 (REFERENCES 24, 26, 27 & 41)

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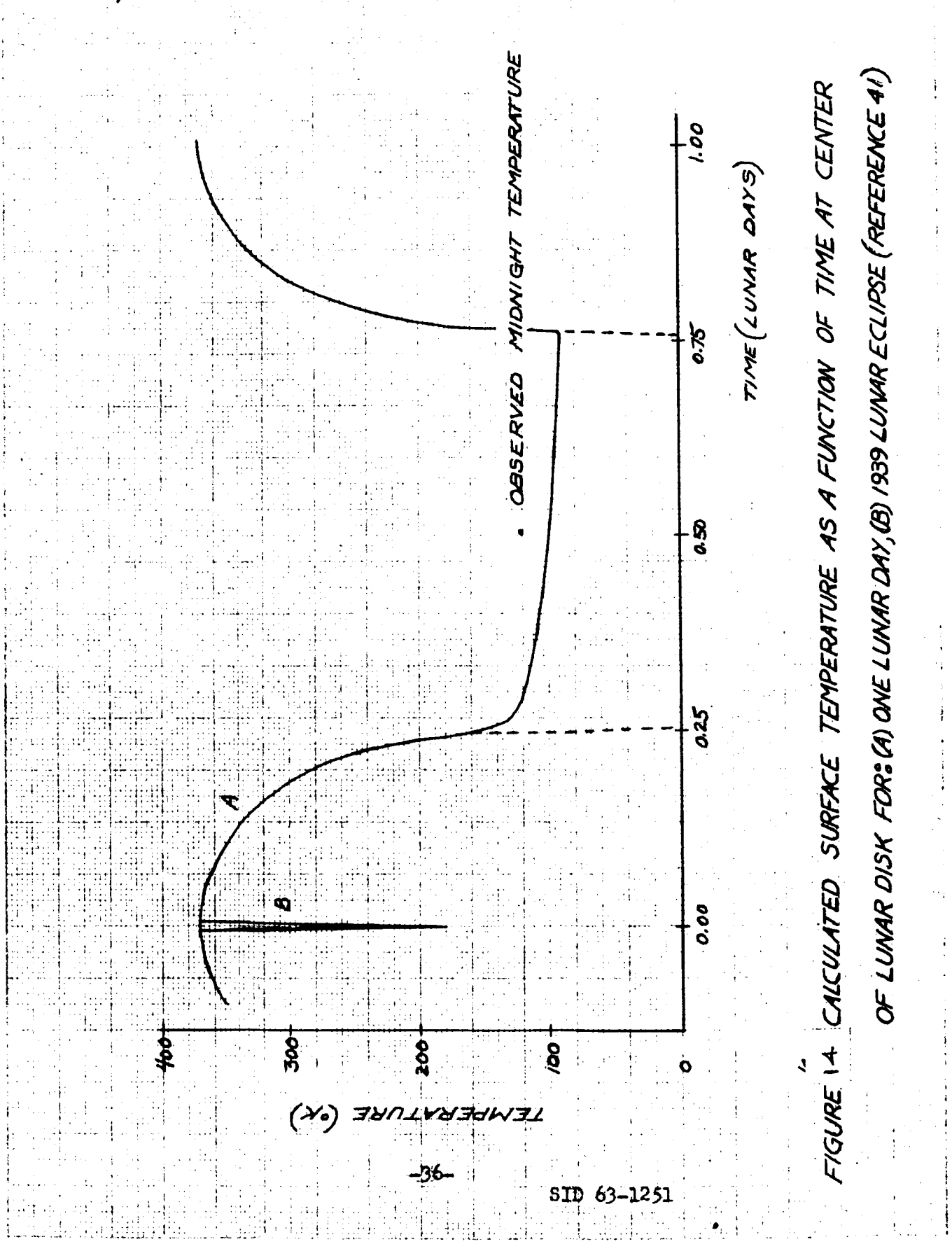


FIGURE 14 CALCULATED SURFACE TEMPERATURE AS A FUNCTION OF TIME AT CENTER OF LUNAR DISK FOR: (A) ONE LUNAR DAY, (B) 1939 LUNAR ECLIPSE (REFERENCE 41)

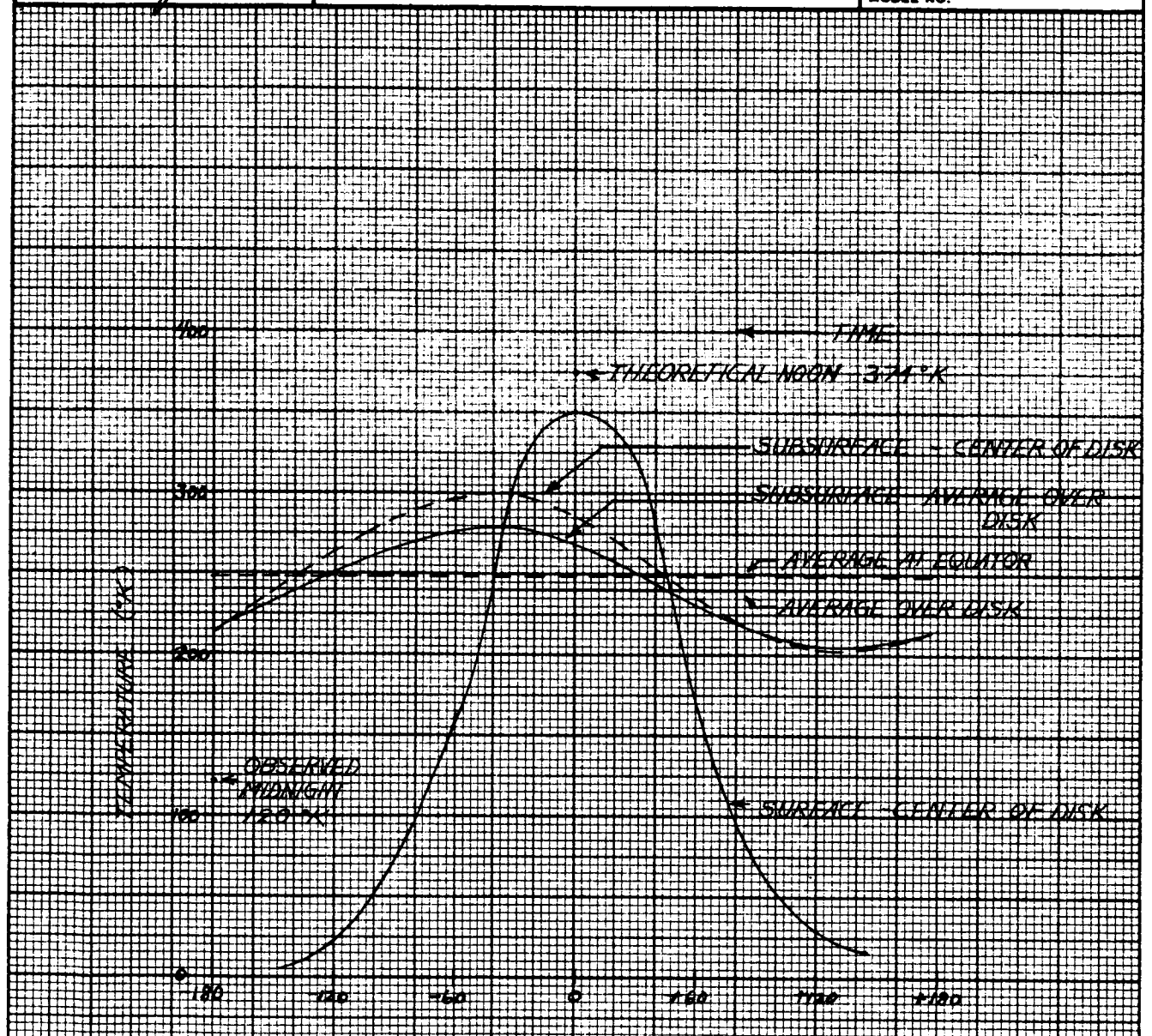


FIGURE 15 PHASE VARIATIONS OF SURFACE TEMPERATURE AT THE CENTER OF THE DISK (0.3 to 3.4) AND SUBSURFACE TEMPERATURES AT THE CENTER OF THE DISK AND AVERAGED OVER THE DISK (1.25 cm) (REFERENCES 8 AND 28)

radio temperature-phase angle curve is roughly sinusoidal with a phase lag with respect to full Moon.

Figure 15 shows the comparison among observed temperatures at the center of the disk and averaged temperatures at the equator and over the entire disk at 0.3 to 3 and observed tmperatures at the center of the disk and averaged over the disk for 1.25 cm as functions of phase angle, References (8) and (28).

The radio temperatures are labeled subsurface, since the explanation of the lesser amplitude of the radio temperatures as compared with the visual and infrared temperatures and the lag of the radio maximum 3.5 days behind the optical maximum is that the optical temperatures originate at the surface and the radio temperatures originate below the surface. On the basis of the decrease in amplitude and the phase lag, the radio radiation at 1.25 cm is from a mean depth of 40 cm, References (8) and (18).

Rates of Cooling

The rates of cooling during lunation and eclipse indicate that, in the absence of Solar radiation, surface temperature is determined primarily by conduction from beneath the surface. Thus the curve in Figure 14 is better described by:

$$\sigma T^4 = E_e = A + F_o$$

where:

A = heat energy absorbed by 1 cm^2 in 1 min

= 1.55 $\cos \omega t$ during lunar daytime

= 0 during lunar night-time

$\omega = 2\pi/P = 1.4776 \times 10^{-4}$/min

F_o = heat flow outward from a surface of 1 cm^2 in 1 min

$$= \left(\frac{\pi}{P}\right)^{\frac{1}{2}} (k\rho c)^{\frac{1}{2}} \sum_{n=0}^{\infty} n^{\frac{1}{2}} T \eta \cos\left(\frac{2\pi}{P} nt + \varepsilon_n + \frac{\pi}{4}\right)$$

k = heat conductivity (cal/cm^2min(°K))

ρ = density (gm/cm^3)

c = specific heat (cal/gm(°K))

ε_n = phase shift

During the lunar daytime, F_o is essentially negligible. At 2.5 hours before sunset (0.003 lunation) and at 0.5 hours after sunrise (0.0006 lunation), $A = F_o$. Complete rising and setting are at 0.0014 lunation from full Moon. Values for the coefficients of the first four terms of F_o are given in Table 2 . See References (36) and (41).

Steady-state conductivity is described by the equation:

$$\frac{\partial T}{\partial t} = \frac{k}{\rho c} \frac{\partial^2 T}{\partial x^2} = 0$$

where:

x = depth beneath the surface

A parametric, periodic solution is given by:

$$T(\xi, t) = \sum_{n=0}^{\infty} T_n e^{-2\pi \xi n^{\frac{1}{2}}} \cos\left(\frac{2\pi}{P} nt - 2\pi n^{\frac{1}{2}} \xi + \varepsilon_n\right)$$

where:

$$\xi = x/l$$

$$l = 2\pi P \, k/\rho c \text{ (diffusivity)}$$

At $\xi = 0$, i.e., at the surface, this equation simplifies to

$$T(0,t) = \sum_{n=0}^{\infty} T_n \cos\left(\frac{2\pi}{P} nt + \varepsilon_n\right)$$

From Pettit's and Nicholson's values for the surface temperature, $(k\rho c)^{-\frac{1}{2}} = 120$. From Wesselink's and Jaeger's models and from radio values, $(k\rho c)^{-\frac{1}{2}} = 1000$. See References (16), (17), and (41). The descripancy indicates that the thermal conductivity changes with depth and/or temperature.

The first four terms of the above series for the surface temperature at the center of disk for various values of $(k\rho c)^{-\frac{1}{2}}$ are given in Table 3, Reference (19).

Values of $(k\rho c)^{-\frac{1}{2}} \leq 1000$ imply that the surface material is highly porous and of low density. For wavelengths $\gtrsim 1$ centimeter, the depth of penetration is $\sim 40 \lambda$. See Reference (31).

Relief Effects of Temperature

Deviations of observed temperature distribution from the calculated distribution for a smooth sphere near the limb and at non-normal angles of incidence for the center of the disk are attributed to surface roughness.

Negative relief, i.e., maria, craters, and valleys should be warmer than surrounding positive relief, i.e., continental areas and peaks, near the center of the kisk near full Moon and cooler near the limb near full Moon and at low angles of incidence of Solar radiation. This is not only due to the shadowing effects on heating, but also due to the effect of the solid angle of radiation into space on cooling.

Observations confirm this. With the exception of Mare Imbrium, maria warm more rapidly toward noon and cool more rapidly after noon than the surrounding continental areas. For most craters near the terminator, the illuminated rim is warmer and the shaded interior is cooler than the surrounding plains. An exception to this is that, during the 13 March 1960 and 5 September 1960 eclipses, the rayed craters Tycho, Copernicus, Aristarchus, and Kepler cooled less rapidly and remained approximately 40°K warmer during totality than the surrounding regions. This is attributed to a thinner layer of dust on the crater floor because of the relatively small amount of micrometeorite influx duirng the relatively short life time of the "young" ray craters. See References (2), (3), (5), (6), (29), and (37).

Figure 16 depicts representative shadow variation curves as a function of the altitude of Sun for positive and negative relief, Reference (29).

The positive relief is represented by the mountain Piton, λ = -0.°00', ϕ = +41°00'. The shadow increases with decreasing altitude until it merges with the terminator with > 4X the projected base area of the mountain.

Negative relief is represented by the crater Mosting, λ = -06°00', ϕ = -00°40'. As Sun sets, the shadow passes the center, climbs the opposite wall, and just reaches the opposite rim as the altitude of Sun = 00°, since the entire rim is essentially the same relative height.

The respective maximum slopes for Pitron and Mosting are >34° and >37°.

Only craters can produce areas of permanent shade, and any crater away from the lunar equator that shows any area of noon shade is a permanently shaded area. A crater at ϕ = 52° will show approximately the same noontime shadow as a crater near the equator will show for a Solar altitude ~38°.

Because of the illumination of some peripheral regions and the inclination of the Lunar equator to the ecliptic, a correction to the measured shadow is needed. Argelander was selected as representative of the typical upland plor regions for estimating the correction factor. Most craters occur in the uplands.

Computed values of the percentage of permanently shaded areas for the various latitude zones are given in Table 4, Reference (29). The total percent of permanently shaded area may be regarded as a minimum value since:

1) Argelander appears to have fewer craters than the actual polar regions;

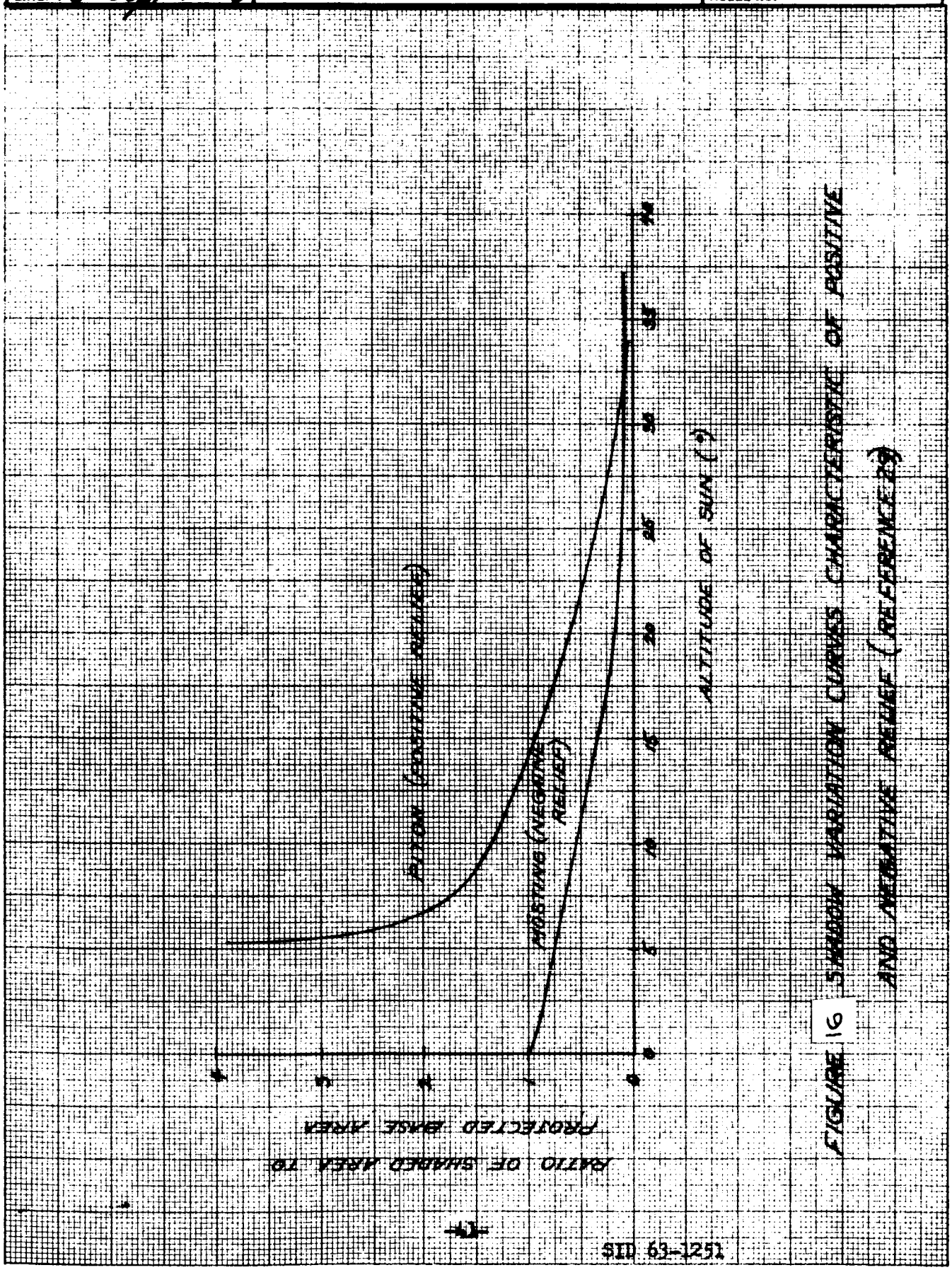


FIGURE 19 SHADOW VARIATION CURVES CHARACTERISTIC OF POSITIVE AND NEGATIVE RELIEF (REFERENCE 29)

2) Russian photographs indicate that the percentage of uplands is greater on the far side than on the near side;

3) Present photographs have insufficient resolution to show the many, small, steep-walled craters that may be expected to increase the total shaded area significantly, Reference (29).

Summary

In summary, the limits of Lunar surface temperature are 120°K to 394°K. The distribution of temperature over the daytime hemisphere is approximately 371°K x $\cos^{1/4}\phi$(selenocentric angle from the subsolar point at noon). The surface temperature of the center of the disk as seen from Earth can be calculated from the formulae in Table 3 for the appropriate values of thermal conductivity (5×10^{-6} to 10^{-5} cal/cm sec(°K)), volumetric specific heat (0.1 to 0.2 cal/cm^3(°K)), and low densities (0.1 to 2.0 gm/cm^3). Specific permanently shaded areas of craters can be estimated from conditions of latitude and interior slopes. Subsurface temperatures can be estimated from the general equations and a choice of corresponding parameters.

Further analysis of the reference material indicates that the most likely Lunar surface temperatures for the center of the Lunar disk, as seen from Earth, are those given by the theoretical curve that fits the recent observations of Sinton, et al. See Reference (36).

This curve is shown in Figure 17 for zero latitude ($\phi = 0°$) and is based on $(k\ c)^{\frac{1}{2}} = 0.0023$. The dotted curve is from the first four terms of the series given in Table 3 for $(k\ c)^{\frac{1}{2}} = 0.0025$.

Curves for other latitudes are shown in accordance with the empirical relation:

$$T_{\phi}^{4} \cos^{2/3} \phi$$

See Reference (27).

In applying the $\cos^{2/3}\phi$ relation, the question arises of whether or not the surface temperatures at higher latitudes during the Lunar night are lower than those for the center of the disk. This possibility appears to be confirmed by the eclipse observations shown in Figure 13 for the center of the disk and the limb and by the study of permanently shaded areas at higher latitudes shown in Table 4.

In summary, the recommended Lunar surface temperatures are those indicated by Figure 17, i.e., infrared temperatures for the center of the disk ranging from 120°K at midnight to 394°K at noon. Latitude variations of these center of the disk temperatures are distributed as $T_o \cos^{1/4}\phi$.

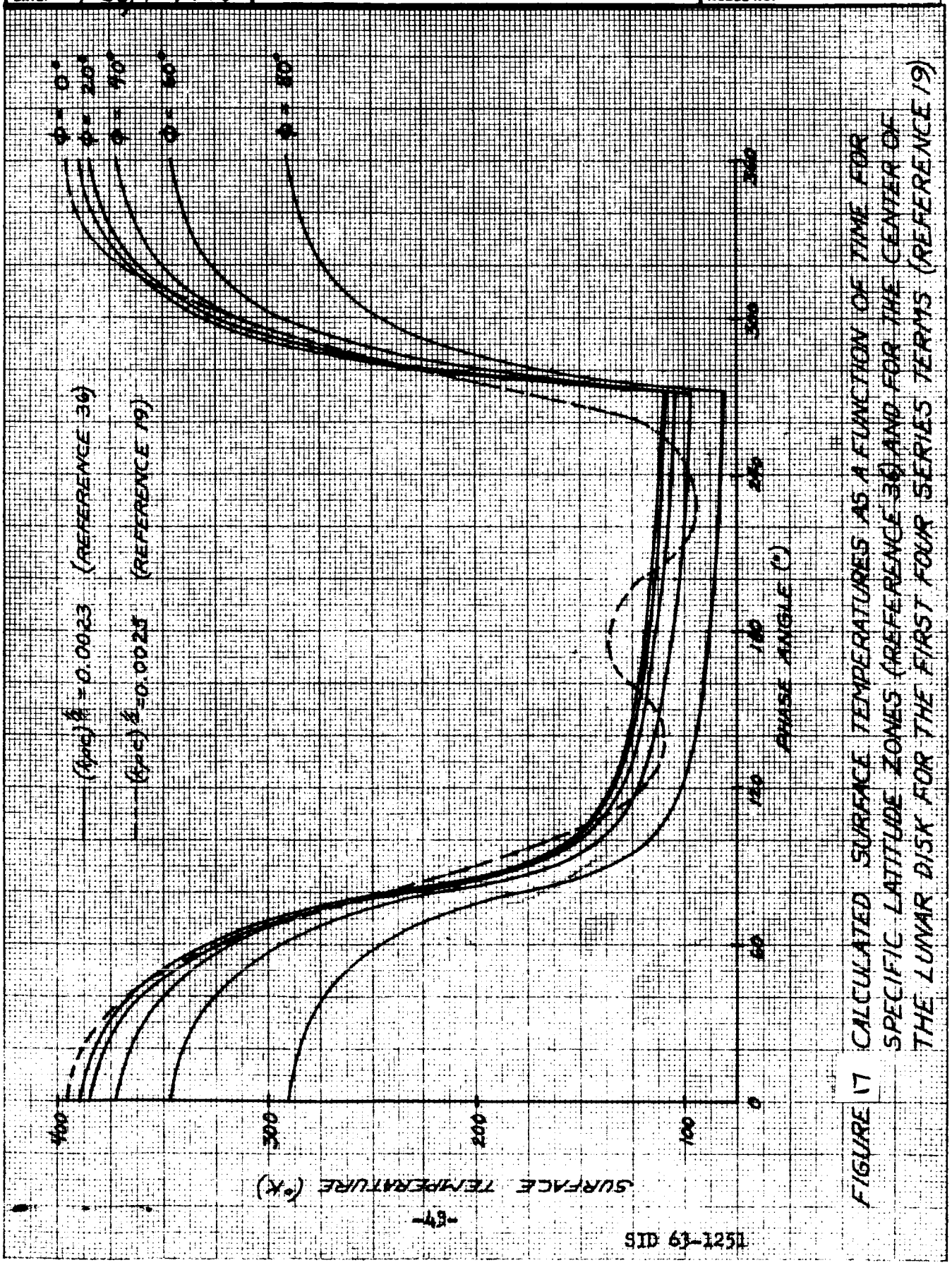


FIGURE 17 CALCULATED SURFACE TEMPERATURES AS A FUNCTION OF TIME FOR SPECIFIC LATITUDE ZONES (REFERENCE 30) AND FOR THE CENTER OF THE LUNAR DISK FOR THE FIRST FOUR SERIES TERMS (REFERENCE 19)

TABLE 2. COEFFICIENTS FOR SURFACE HEAT FLOW, Reference (36)

n	0	1	2	3
T_n	210°K	157°K	34°K	30°K
ε_n	0°	-6°	+6°	+159°

TABLE 3. SURFACE TEMPERATURES AT THE CENTER OF DISK, Reference (19)

$(k\rho c)^{-\frac{1}{2}}$	T (0,t) (°K)
20	$276 + 83 \cos(\omega t - 12°) + 26 \cos(2\omega t + 3°) - 6 \cos(3\omega t - 33°)$
125	$247 + 132 \cos(\omega t - 5°) + 34 \cos(2\omega t + 6°) - 19 \cos(3\omega t - 11°)$
250	$237 + 146 \cos(\omega t - 4°) + 35 \cos(2\omega t + 7°) - 23 \cos(3\omega t - 6°)$
400	$230 + 156 \cos(\omega t - 3°) + 36 \cos(2\omega t + 7°) - 26 \cos(3\omega t - 6°)$
500	$227 + 159 \cos(\omega t - 3°) + 36 \cos(2\omega t + 7°) - 28 \cos(3\omega t - 5°)$
700	$223 + 165 \cos(\omega t - 3°) + 36 \cos(2\omega t + 7°) - 30 \cos(3\omega t - 4°)$
1000	$219 + 170 \cos(\omega t - 2°) + 36 \cos(2\omega t + 6°) - 31 \cos(3\omega t - 3°)$
1200	$217 + 173 \cos(\omega t - 2°) + 36 \cos(2\omega t + 6°) - 32 \cos(3\omega t - 3°)$

TABLE 4.

PERMANENTLY SHADED AREA

Reference (29)

Latitude Zone (°)	Total Surface in Zone (%)	Percent Upland in Zone	Total Upland in Zone (%)	Percent Shaded Area at Noon Upland Only	Reduction Factor	Permanently Shaded Area (%)
90-80	1.53	100	1.53	21.5	0.85	0.278
80-70	4.54	100	4.54	5.7	0.55	0.141
70-60	7.41	94.5	7.00	2.5	0.33	0.057
60-50	10.04	78.5	7.88	1.5 (est.)	0.25	0.029
50-40	12.36	77.5	9.57	0.0	-	0.000
40-30	14.28	67.5	9.63	0.0	-	0.000
30-20	15.76	57.0	8.99	0.0	-	0.000
20-10	16.78	48.0	8.05	0.0	-	0.000
10-0	17.30	47.0	8.13	0.0	-	0.000
TOTAL	100		65.32			0.505

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LUNAR MISSIONS APPRAISAL

In beginning any complex system analysis, it is necessary to create a feasible model of the situation and to synthesize its operational characteristics, incorporating reasonable assumptions for the basic premises and the many individual variables. The objective of this section then is to define such a plausible, rational, framework for lunar missions in terms of launch schedules, general task sequences and requirements; and, then, within this framework, to compare the merits of alternative logistic vehicles. Specifically, an evaluation will be made of the comparative mertis of the LEM/CLEM approach and the Apollo II, LLM/LLV approach. Of primary importance are the lunar missions requirements of personnel and equipment and the abilities of these vehicles to satisfy the requirements within safety, cost, and time constraints.

APPROACH

In order to realistically appraise the merits of alternative approaches to lunar operations, it is necessary to first define and describe what those operations are likely to be. Admittedly, much uncertainty exists at the present time regarding details of lunar conditions and operations; however, once certain key assumptions are made, then the development of a basic, plausible, operations and missions sequence becomes feasible.

The approach utilized in the following discussion, then, divides the lunar operations into five basic phases ordered arbitrarily as follows:

1) Initial Landing Phase

2) Exploration Phase

3) Site Verification and Temporary (Modular) Base Establishment Phase

4) Permanent Base Construction Phase

5) Permanent Base Occupancy Phase

The dividing line between the third and fourth phases is particularly ill defined because some overlap will undoubtedly exist. The last two phases are not covered in detail because they are beyond the scope of this report. Within each of the first three phases, the specific missions and general tasks will be formulated so as to form a basis for the comparison of alternative approaches to their accomplishment, particularly with regard to vehicle requirements and characteristics.

In the analysis of each phase, major emphasis will be placed on developing a balanced concept. By this is meant that the constraints of safety (and possible rescue) requirements, tasks to be performed, likely space suit limitations, number of personnel, vehicle reliability, lunar stay time, costs,

etc, will all be taken into consideration simultaneously. By integrating the major requirements and limitations in this manner, a plausible, systematic, inter-related missions analysis can be generated.

The logistic requirements (personnel, equipment, and data) essential to the development of each phase in a balanced manner, will be determined and discussed. It is recognized that, in every phase, several key assumptions exert considerable influence on the requirements and schedules generated. In such cases, an effort will be made to formulate the requirements on a general level in keeping with the validity of the assumptions made.

A comparison will be made of the capabilities of the LEM/CLEM and the Apollo II LIM/LLV to fulfill the requirements developed. This analysis will be made for each of the phases previously analyzed. SID 62-1466 and SID 62-1467 were used to obtain performance data of the Apollo II LIM/LLV. Major emphasis will be placed on the costs of transporting payloads to the Moon and on the most effective use of launch vehicles (Saturn V's).

DISCUSSION

Manned Lunar Operations

Appendix A contains a possible sequential schedule for lunar exploration which places the Saturn V launches required for the operations discussed within the following phases, in chronological sequence. The Initial Phase of the schedule agrees with SID's proposed Apollo development schedule; the Exploration Phase sequence is strictly arbitrary.

1) Initial Landing Phase

Although the initial manned lunar landing is currently scheduled to be performed by two men in the LEM vehicle in late 1967, it is thought that this schedule may be optimistic. It assumes that, on the initial manned circumlunar flight, the CM/SM, upon being exposed to the hard vacuum of space (10^{-13} torr) for the first time, will retro into a lunar orbit and perform a photo-reconnaissance mission from possibly a relatively low altitude. In addition, at the time of this circumlunar flight, the lack of knowledge of lunar constraints (mass, shape, etc.,) may produce unanticipated orbital perturbations that also could jeopardize the mission. It appears that prudence might dictate that a more conservative schedule would actually be followed. Furthermore, a qualification flight of the LEM would appear to be highly desirable before a manned landing were attempted.

An automated LEM could be landed on the Moon while the entire operation was observed from an orbiting CM. The successful landing and subsequent observation of the LEM on the lunar surface would tend to remove questions regarding the LEM capability after exposure to deep space conditions. If the landing were successful, the automated LEM could provide some back-up to the manned LEM landing. After three days in transit to the Moon, maybe one day

in orbit, and two days on the lunar surface, for a total of six days, the LEM needs to have a reliability of better than 0.99. This means that the decay rate of the reliability has to be very slow over this period. Consequently, this fact forms the basis of the assumption that, after five times this length of time; and, after a complete temperature cycle, 0.7 to 0.8 reliability may remain. If this were the case, it would serve to justify much of the cost of an automated LEM mission.

It appears that, based on a number of considerations, NASA presently favors Sinus Medii for the initial landing location. In addition to being strategically located in the center of the Earth-oriented lunar disc, this region possesses a variety of significant geological features. Maria, highland, rilles, craters, and crater ejecta are all found within its confines. It appears that flat areas of sufficient size to present favorable landing targets can be found in close proximity to the above-mentioned features, so that significant information may be obtained on which to base subsequent missions and equipment requirements. Based on work done in France by M. Allais, Director of Research at the "Centre National de la Recherche Scientifique," on the economic prospects of mining exploration over large territories (Algerian Sahara Case Study), Sinus Medii (and surrounding highlands) is of sufficient size to present a good sample of lunar geology and minerals. Consequently, it may be assumed that suitable conditions will be found within the confines of Sinus Medii to warrant locating the permanent base in that area.

The cleft network radiating from Sinus Iridium includes the highlands bordering Sinus Medii. A dust-filled cleft may present advantageous base construction and protection features. In addition, the boundary may present a lunar stratigraphic analogy to the terrestrial latah formation which interleaves basaltic lava flows with sedimentary (on the Moon, dust) deposits. In such case, the area would be of basic scientific interest as a historical record of that lunar area.

The initial landing will probably be of short duration (a few hours rather than two days) for a variety of reasons. Among these are the regression of the CM/SM orbit if inclined, which could seriously complicate rendezvous, and the space suit reliability in the lunar environment.

2) Exploration Phase

a) Justification of Exploration: In view of the billions of dollars which a permanent base would cost, a program of careful exploration, preceding selection of a permanent site location for construction of a lunar base, would appear prudent. An expenditure of approximately 10% of the anticipated cost

of the base in site selection would not appear to be out of line, since this amount could readily be saved by locating a favorable site. The alternative would be to make an early commitment to a site selected virtually by chance. General Electric, in its Project Pilgrim study on the establishment of a lunar base, estimated the cost of a base at $8 billion; so that, on this basis, the expenditure of even $800 million on site selection would not be unreasonable.

Because of the high cost of transporting material to the Moon, the use of lunar resources will be mandatory. The one resource which numerous studies have indicated would be of most value is water. Water might be found in a variety of deposits:

1. Surface - possibly as ice in permanently shadowed zones

2. Subsurface blanket - as layers of ice or hydrated minerals near the lunar surface

3. Subsurface massive - the above-mentioned blanket could extend to considerable depth (lunar domes?)

4. Bare rock

Even if water were found on the initial manned landing, a number of landings to investigate other areas would be dictated by prudence. Other sites may possess more water, or other minerals such as basalt or sulphur, in addition to water. The possibility of locating usable minerals and favorable base construction features, i.e., lava tubes, rilles where excavation would not be required but only burying, etc., warrants exploratory missions. Because of temperature fluctuations and hazards from solar radiation and meteoroids, the major portion of a permanent base will most likely be buried except for hangars, observation tower, astronomical observatory, antennas, etc. Different sites would undoubtedly present different equipment, time, and cost requirements for base construction, so that exploratory missions to several different geological features could result in the location of a site that would significantly reduce the equipment, time, and cost requirements associated with the development of the lunar base.

It is thought that, even assuming the most favorable mineral discoveries, a minimum of three landing missions would be performed during this phase. A more realistic number would be five to ten. The Proposed Lunar Development Schedule (Appendix A) indicates seven. By scattering the exploration sites and locating them in areas where lunar features or orbital reconnaissance may indicate the presence of minerals, a sample

on which to base an estimate of the geology of the Moon can be obtained.

b) Exploration Constraints: In the examination of the exploration phase, restrictions imposed by the length of stay on the Moon, crew size, area of exploration, use of vehicles, space suit reliability, etc., need to be balanced in order to develop a realistic concept of this phase.

Even if a lunar transportation and exploration transporter were available, its radius of operation would most likely be restricted to the radius of manned space suit operations so that, in the event of failure of the transporter, the crew could journey back to the main vehicle. With a space suit life support limit of four hours, this radius of action may be assumed to be approximately four miles. The lunar transporter weight and fuel requirements would necessitate the use of a second Saturn V to transport it to the Moon. The use of solar-powered vehicles does not appear practical because of: the restriction of use to periods of sunlight only, need for auxilliary power when traveling in shadows, and limited power available. Assuming a 15% conversion efficiency, a solar collector would produce only about 8 horsepower.

c) Alternative Methods of Exploration: Several alternative schemes have been proposed for the conduct of lunar exploration.

1. LEM: The LEM, as it is designed, has a capability of carrying two men to the lunar surface for a stay time that is limited at the present time to two days. Even if the LEM could have its lunar stay time extended, the Command Module in lunar orbit would limit the stay time to a maximum of two week (without changes or lowered reliability).

2. LEM With CLEM or Lunar Occupancy Module: One approach that has been proposed to extend the lunar stay time is to provide a lunar shelter which would enable the men to stay on the Moon for thirty or more days. A major disadvantage of this approach is that the occupancy payload itself would require a precious Saturn V. Thus, every location which would be explored utilizing the occupancy payload would require two Saturn V's. The availability of these boosters is the most critical element and the pacing factor governing the exploration of the Moon. To use two to explore one limited area does not appear to present a desirable alternative. As mentioned previously, even with a surface vehicle, the maximum radius of exploration would be approximately four miles. Be-

cause of the limited life support capability of the LEM, the occupancy payload would be landed first and then the LEM. If the LEM stay time is longer than one week, it cannot return to Earth with the same CM/SM that brought it to the Moon, so that a relay system would need to be utilized. (However, it seems more likely that the Apollo CM/SM, already designed for one week vs the LEM's one day, would be relatively more capable than the LEM of extended stay times.)

After fourteen days of sunlight, during which exploration could be conducted, the fourteen-day lunar night would follow. During that time, the Earth phases will vary from half Earth at sunset to full Earth and half Earth again; it has been established that these lighting conditions are much better than full moonlight on Earth. Consequently, visibility should not restrict exploration during the long lunar night. It may very well be that the softer full Earth shine would create more favorable visibility conditions than the bright direct daylight with the associated glare and poor contrasts, Night travel would of course also require some adjustments and changes to the space or lunar exploration suit, but the temperature is not expected to present unsurmountable problems. Radiative shielding top covers (garments), or a poncho-like "overcoat", should be able to cope with the low temperature.

The LEM would also be subjected to extreme cold and high vacuum for an extended period with an inevitable loss of reliability. Present indications are that no large vacuum chamber on Earth will be able to simulate lunar conditions so as to enable the testing of complete systems under simulated lunar conditions.

After fourteen days of night, daylight exploration could once more be undertaken. However, because of the confined radius of exploration and limited equipment available, each succeeding day will be less productive of new data. Eight (continue on following page)

weeks later (based on assumed availability of the Saturn V of one every eight weeks), the return to Earth would require the launching of another Saturn V, probably with another exploration party or occupancy payload for a different site. In sixty days, after exposure to two complete cycles of +250°F and -250°F temperatures and a 10^{-12} torr vacuum, the probability that the LEM would be operational presents an assumption which is felt to be unacceptable. The alternative would be to use still another Saturn V with an automated LEM to make the pick-up. It is thought that use of Saturn V's in this manner is prohibitively uneconomic and other alternatives would have to be devised. If more than one LEM were landed to conduct the exploration, the point of diminishing returns would be reached that much sooner and the probability that a LEM would become inoperative would double. Finally, if a staggered relay system of LEM flights were adopted, the importance of having the new flights land at dawn for maximum light utilization, would mean that the LEM crew, to be picked up, would have to spend two weeks of lunar night in waiting, which means that life support weight for this period would be utilized with a very poor pay-off.

To provide a reasonable degree of flexibility with reliability and, also, a capability for rescue in conjunction with extended stay-time operations, it would be necessary to utilize a three-man LEM and a four-man Apollo. This would permit two-man translunar flight, one-man descent in a three-passenger LEM to pick up two stranded men, return to lunar orbit and rendezvous with Apollo, and trans-Earth return flight with four men aboard. This same vehicle could provide for rotation of the two-man station crew so that extended stay times for the LEM on the lunar surface would not be required. Of course, this might require an uprated Saturn V booster to accommodate the extra weight. It can be concluded (and is substantiated later in this report) that the capability for landing substantial cargo payloads considerably exceeds the anticipated capability for safe and reliable manned transport.

3. Apollo II LLM: A cryogenic Apollo could be direct-landed on the Moon with thre men shortly after dawn for a stay of seven days. The payload-carrying capability of the LLM might enable the carrying of a lunar scooter (83 lb and 12.5 cubic feet) with sufficient propellant (300 lb and approximately 6 cubic feet) for two round trips to points four miles away, including nine stops on the way out and nine more on the way back. Conse-

quently, virtually the same area could be explored as with the occupancy payload and, most significantly, with the use of only one Saturn V. No provisions would have to be made for operations in the -250°F night environment. In the event of a lunar launch failure, a rescue capability could be provided by having a Saturn V on a stand-by basis with an Apollo II LIM fitted out to hold five astronauts. A Saturn V has been allocated in the schedule as a stand-by rescue vehicle. Such a rescue capability would not be available in either the LEM or LEM with CLEM (or occupancy payload) exploration approches. This could be a most important operational consideration.

4. The use of an elaborate train of vehicles (life support, communications, power, fuel, etc.) does not appear to be a very desirable alternative for early exploration for the following reasons. Such a train would require many Saturn V's at a time when each Saturn V is precious. The knowledge of the lunar terrain and terrain characteristics would be incomplete to a point at which reliability of the resultant designs could not be assured. Much of the travel (at approximately 1 MPH) would be over relatively homogeneous terrain. In addition, if it is assumed that the effective sight distance is 100 feet on either side of the train, then in a 100-mile loop, only approximately $\frac{200}{5280} \times 100 = 3.8$ square miles are actually examined. Consequently, it appears that a more efficient approach would be to land missions at known (via astronomy, geology, or photo-reconnaissance) points of interest. After a lunar base has been established, then perhaps trains may be effectively used for the establishment of outposts.

3) Permanent Site Verification and Temporary (Modular) Base Establishment Phase

After key features of the Moon are surveyed during the exploration phase, a selection of the most promising site for a permanent base will be made. To qualify as "promising," a site would need to have water-bearing minerals, along with possibly other minerals, and possess features conducive to base building and utilization. To make certain that the minerals present are sufficient to warrant the establishment of a permanent base on the site, a more thorough survey will need to be made to ascertain the extent of the mineral deposits. If local minerals (i.e, basalt, sulphur, etc.) are to be used in base construction, pilot plants will need to be set up and the mineral extraction and utilization processes tested under conditions of the lunar environment. Even if local minerals were not going to be utilized for base construction, water extraction

techniques would need to be developed and tested in the lunar environment so that economic analyses of the merits of the site may be made.

The site verification phase will require that landings of substantial payloads be made at the site. Large core drilling equipment to be used to determine the extent of mineral deposits will need to be included in the early payloads. Also, power generating equipment (nuclear) and pilot plants for mineral conversion would be included. As favorable results from the site verification operations are received, additional commitments of material to the site would be made. The size of the "local" mineral deposits in comparison to use rates will constitute one of the limiting factors on the size of the base that will be built. During this phase, a modular base would be constructed. The schedule in Appendix A contains a proposed sequence of payloads (both men and equipment) which would be required for site verification and temporary (Modular) base construction.

The use of highly automated equipment for base construction has two significant disadvantages. "Highly automated" generally implies the use of complex electro-mechanical systems. Since these could be checked-out to only a limited extent on Earth, their reliability could impose high maintenance penalties. Furthermore, electronic systems would be susceptible to damage from solar flares. Personnel could take refuge in the centers of protective shelters, but to attempt to protect vehicular electronic systems (TV eyes, communications equipment, etc.), could pose serious design problems or significant equipment limitations. Nevertheless, vehicles for base construction will be required and prime emphasis in the design of these vehicles should be placed on reliability.

The actual location of the various modules on the lunar base would depend to a large extent upon the local surface features. Nuclear generators may be assumed to be located on the outskirts of the base. The possibility of blast and spilling damage during landing of the various modules and payloads would indicate that the landings of new modules occur at a distance from established modules. Conceivably, a lunar construction vehicle could have a lifting capability sufficient to move modules from a peripheral landing/launching pad to the base complex proper.

Because of the unavailability of sealants, gasketing material, etc., that are known to be capable of withstanding +250°F temperature cycles, radiation, and 10^{-12} torr vacuum, it may be assumed that the maximum effective life of an early module is no greater than five years and very likely closer to three years. Meteoroid puncture patches (or maybe patches upon patches) will probably deteriorate to the point that it is uneconomical to attempt to keep the modules operational. In addition, the high value placed on

a man hour on the Moon could make continuous maintenance of old modules prohibitively expensive. Of course, the various modules will have to be resupplied periodically within the three- to five-year life estimate.

Modules for regeneration of water (even if a large supply is found on the Moon), wastes, and oxygen will likely be landed during the 1973-1975 period. In reality, the various modules landed will likely contain small amounts of mixed cargo such as food, additional space suits, mail, psychological luxury items, medical supply replenishment, etc.

The CLEM, according to NASA sources, is reported to be capable of carrying 7,000 lb of payload to the Moon in the 1970 time period. It is assumed in this report that the 7,000 lb is net payload weight. However, by developing the Apollo II LLV, it could be used to land a logistics payload of approximately 28,000 lb net, or four times as much as the CLEM. The launching of either a CLEM or LLV would require the expenditure of a Saturn V. Consequently, on a cost-per-lb of delivered weight, the LLV exhibits a significant advantage over the CLEM.

4) Permanent Base Construction Phase

During this phase, large tonnages of material would continue to be deposited at the site selected if the mineral deposits justify the commitment. The quantities and constituents of the payloads would largely be dictated by the lunar characteristics and resources. Extensive exploration of the Moon will probably begin during this phase.

5) Permanent Base Occupancy Phase

At this time the missions for this phase have not been firmly established although they are being extensively studied. It can be reasonably assumed that growth versions of the Saturn V booster uprated as much as 50% payload will be utilized together with advanced Apollo spacecraft.

Utilization of Lunar Payload Landing Vehicles

During the initial landing and exploration phases, the personnel landing vehicles would be utilized to launch the personnel from the lunar surface, leaving behind the descent stages. As payloads were deposited, additional weights of equipment (descent rockets, tankage, guidance and stabilization equipment, etc.) would become available. An appraisal of the utility of this equipment indicates that three distinct stages are likely to occur. During the initial landing and exploration phase (with either the LEM or LLM), the sites will most likely be relatively far removed so that virtually no possibility exists for travel to other exploration sites

in order to utilize in any manner the stages left behind on previous landings. The main use envisioned for descent stages during the early phases is as a possible source of spares for the launch stage in the event of an emergency. However, the availability of such spares, even with a large degree of commonality between descent and launch stages, would not reduce the requirement for a stand-by rescue capability on Earth in the case of the Apollo II LIM. The space suit limitations could significantly hamper any complex repairs or replacements attempted. During the site verification phase, use could be made of tankage, insulation, piping, valves, motors, etc., specifically in the pilot plant operations. However, the equipment would need to be used almost "as is" since the cost of transporting elaborate equipment to extensively modify descent stage material would, in general, be prohibitive. Insulated tanks would be essential for the storage of water and cryogenic hydrogen and oxygen obtained from the water. During later stages of base construction and during routine occupancy, mixed payloads of personnel and supplies would probably be landed, with the descent stage possibly being refueled on the Moon and used to launch the Earth-returning vehicles. Other descent stages, specifically the accumulating LLV's as the complement of the base grows, could be refueled and used for lunar exploration and transportation. Additional descent stages could be disassembled and stored, thus furnishing a supply of spare parts for repairs.

Comparison of LEM/CLEM and Apollo II LIM/LLV Approaches to Lunar Base Development

1) Saturn V Costs

One of the major elements of cost associated with manned lunar activities is the cost of the Saturn V. It is estimated by NASA that the early boosters will cost approximately $50,000,000 each. (This figure includes no pro-rated R&D costs.) After about thirty have been built, it is anticipated that the cost will decrease to approximately $35,000,000. However, since the construction of the thirty will take approximately five years, during which it may be assumed that costs will rise, it is assumed, for purposes of cost comparisons, that the Saturn V will cost $50,000,000.

2) LEM/CLEM and Apollo II LIM/LLV Costs

a) Development Costs: The cost of development of the LEM may be assumed to be borne by the basic Apollo lunar landing program, so that an operational vehicle design is available as a free by-product of that program. The cost of the development of the CLEM has been estimated by Grumman to be about $100,000,000. (AW, 22 July 1963, p 163.) Compared to this is the rough estimated cost of development of the Apollo II LIM/LLV of approximately $450,000,000; man-rating of the LIM/LLV is estimated at an additional $50,000,000. The RL-10 engine used in the Apollo II (both LIM and LLV) is virtually an off-the-shelf item, so that the development cost of this item should be negligible. A new guidance system would need to be developed for the Apollo II LLV so that

it could be landed automatically, possibly utilizing beacons for guidance. The $350,000,000 difference in development costs between the CLEM and Apollo II LLV will need to be justified by possible savings in other areas. (Note that an early LLV can be adapted from Apollo hardware to provide two to three times the CLEM payload capability at approximately $190 million development cost; however, it would not necessarily provide for the needed improvement in personnel transport capability to accompany the cargo payload improvement.) Total Apollo II development cost is estimated at $825,000,000, including the LLV.

b) Production Costs: The components which comprise the LEM and LLM transport vehicles and the CLEM and LLV logistics vehicles are presented below. A rough estimate of the production cost of each of the components and the total vehicle configurations is also given.

LEM		CLEM (LEM TRUCK)		LLM		LLV	
Component	Cost	Component	Cost	Component	Cost	Component	Cost
LEM (2-man)	7.7 M	CLEM (7000 lb PL)	8.0 M	CM (5-man)	20.0 M	LLV and Equipment Adapter	16.9 M
CM	17.5 M	CM	17.6 M	Stage I	12.0 M		
SM	7.0 M	SM	7.4 M	Stage II	8.0 M		
TOTALS	32.2 M		33.0 M		40.0 M		16.9 M

3) Cost Comparison - Vehicle Hardware

The total cost, including development, production and Saturn V costs, of transporting cargo to the Moon by either CLEM or LLV is shown in Figure 18. The curves in Figure 18 are based on the fact that four Saturn V's (at $50,000,000 each), combined with CLEM cargo vehicles, are required to land the same payload weight as one Saturn V and an Apollo II LLV. For example, if it were assumed that only 140,000 lb of payload would ever be landed on the Moon, then either twenty CLEM's could be used or five LLV's. Considering development, production and Saturn V costs, 1100 M dollars are saved through utilization of the LLV as indicated in Figure 18. The differential $350,000,000 LLV development cost is over-shadowed by CLEM production and Saturn V costs for lunar-landed payloads greater than 10,000 lb. In addition, the 7,000 lb capacity of the CLEM limits the weight and volume of individual payload packages, thus limiting their utility.

The total cost, including development, production and Saturn V costs, of transporting personnel to the Moon by either LEM or LLM

is shown in Figure 19. These curves are based on the fact that five Saturn V's, combined with a two-man LEM, are required to land the same number of personnel as two Saturn V's combined with the LLM. For example, if it were assumed that thirty men would be landed on the Moon, then either fifteen two-man LEM's or six LLM's could be used. Considering development, production, and Saturn V costs, $280,000,000 are saved through utilization of the LLM, as indicated in Figure 19. The differential $475,000,000 LLM development cost is over-shadowed by LEM production and Saturn V costs for lunar-landed personnel in numbers greater than twenty using a two-man LEM.

4) Exploration Capability Comparison of Basic Vehicles

The initial landings will be made by LEM. If it is assumed that the productivity of exploration of a site (because of limited travel radius and limited equipment) decreases by one-half for every seven days, then the expenditure of two Saturn V's to enable crew men to stay on the Moon for forty-eight hours means that the direct cost is approximately

$$\frac{(50M + 32.2M)}{2 \text{ men x } 2/7 \text{ wks}} = \$144,000,000$$

(2-man LEM) per man-week-unit of exploration. One alternative is to land an occupancy payload to extend the time. However, because of the limited radius of exploration stemming from safety considerations, it is considered that lengthy stays yield continuously diminishing returns (with simultaneously increased probability of launch failure if the launch vehicle is standing by on the Moon during this period.)

The direct cost per man-week-unit of exploration of the two-man LEM plus CLEM with a CM Shelter ($15M) would be

$$\frac{2.50 + 32.2 + 33 + 7.2}{2(1 + \frac{1}{2})} = \$57,500,000$$

per man-week-unit of exploration. This assumes an two-week stay time (with 50% effectiveness assumed for the second week).

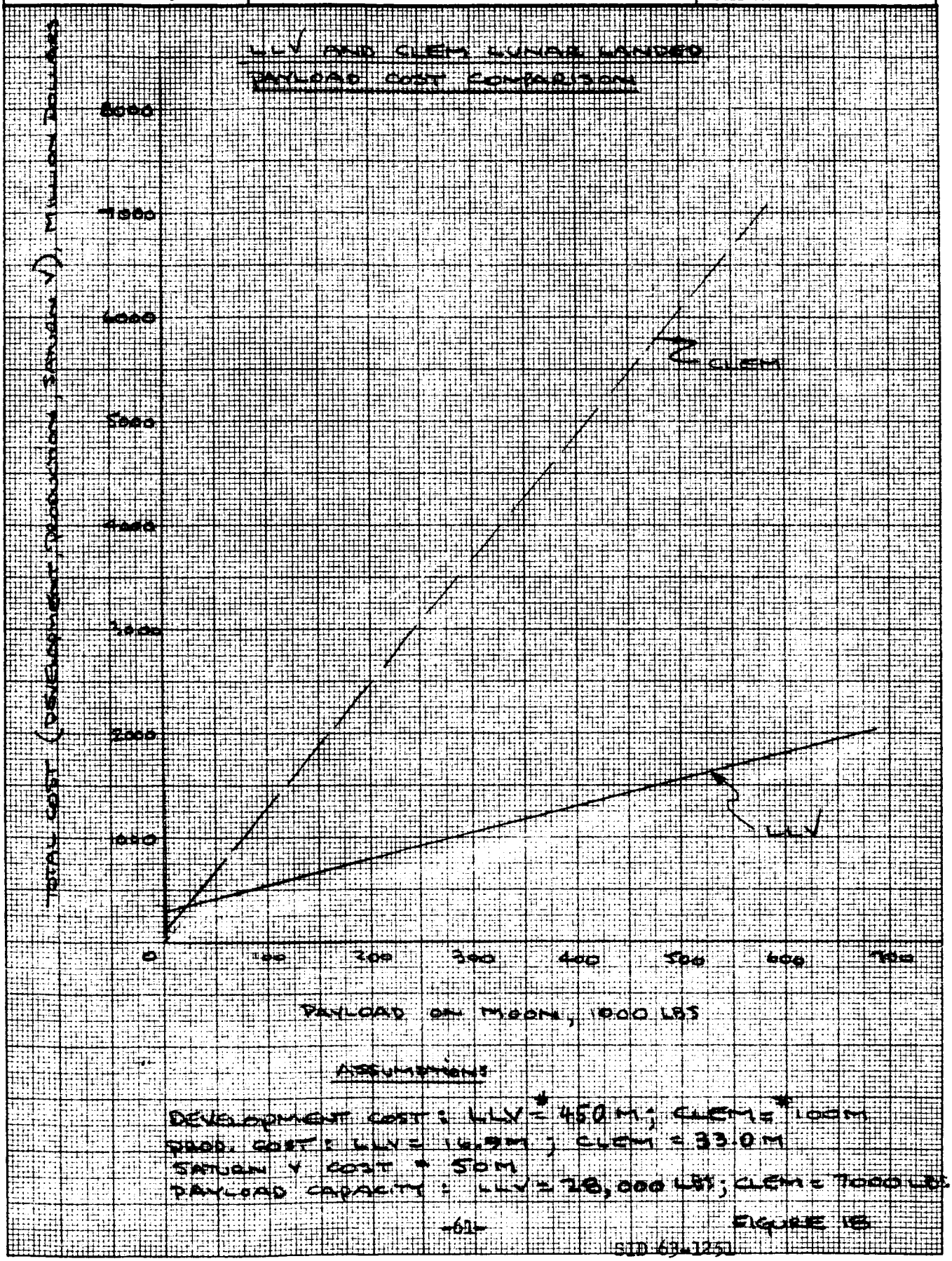
By going the Apollo II LLM route, only one Saturn V would be required. This approach would cost $\frac{50M + 40.0M}{5 \text{ man weeks}} = \$18,000,000$ per man-week-unit of exploration, or 68% less than the two-man LEM/CLEM approach.

5) Lunar Base Limitations

If the assumptions are made that:

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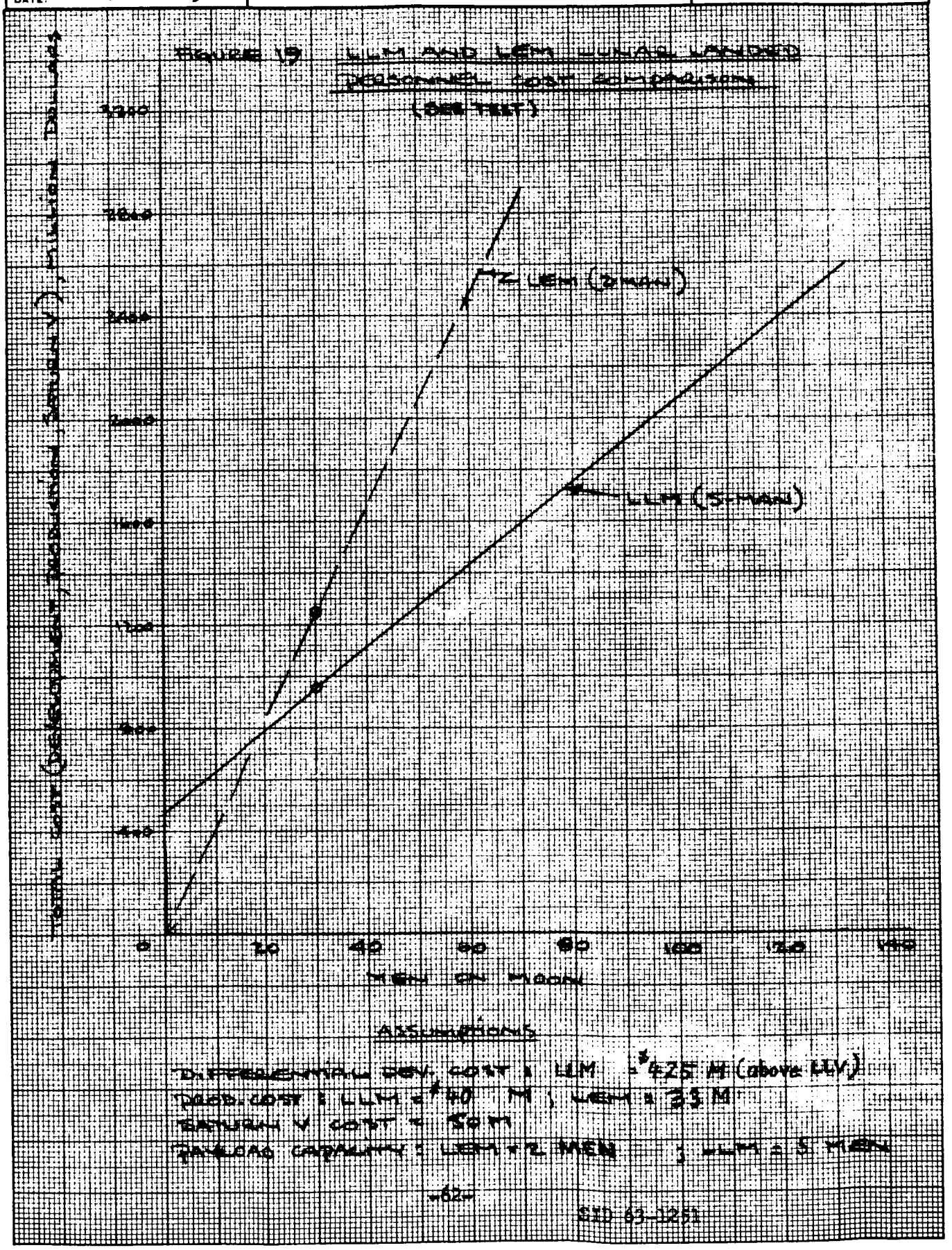
ASSUMPTIONS

DEVELOPMENT COST: LLV = 450 M; CLEM = 100 M
PROD. COST: LLV = 16.9 M; CLEM = 33.0 M
SATURN V COST = 50 M
PAYLOAD CAPACITY: LLV = 28,000 LBS; CLEM = 7000 LBS

FIGURE 16

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FIGURE 19 LLTV AND LEM HUMAN LANDED DEVELOPMENT COST COMPARISONS (DEC TEST)



ASSUMPTIONS

DIFFERENTIAL DEV. COST : LLTV = $425 M (above LLV)
PROP. COST : LLTV = 140 M ; LEM = 35 M
RETURN V COST = 50 M
PAYLOAD CAPABILITY : LEM = 2 MEN ; LLTV = 5 MEN

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a) The LEM will be utilized to transport people to the lunar base and that it can transport only two people.

b) Only one Saturn V is available every eight weeks.

c) One year is the maximum stay time for personnel on the Moon.

Then a total of only twelve people can be stationed on the Moon. At this level of manning, all Saturn V's are used to rotate personnel. If the lunar base complement was held at a maximum of six, then three Saturn V's would be available every year to land a CLEM payload (7,000 lb).

Similar assumptions applied to the Apollo II LLM (5-man) indicate that the maximum complement of the lunar base would be thirty, at which point all Saturn V's would be expended in personnel rotation.

Figure 20 illustrates the relationships between lunar-landed cargo and personnel for the alternative approaches to landing a mix of cargo and personnel.

The basic logistics model that indicates the relationships between the pertinent variables is as follows:

Let L = number of launches per year
M = number of months stay-time allowed
P = lunar base complement
C = number of cargo launches per year
W = cargo payload weight
A = number of personnel transportable in one lunar landing vehicle

Then the number of cargo launches possible per year may be obtained from the following equation:

$$C = L - \frac{12P}{AM} \qquad (1)$$

The pounds of cargo payload weight per man per year is equal to:

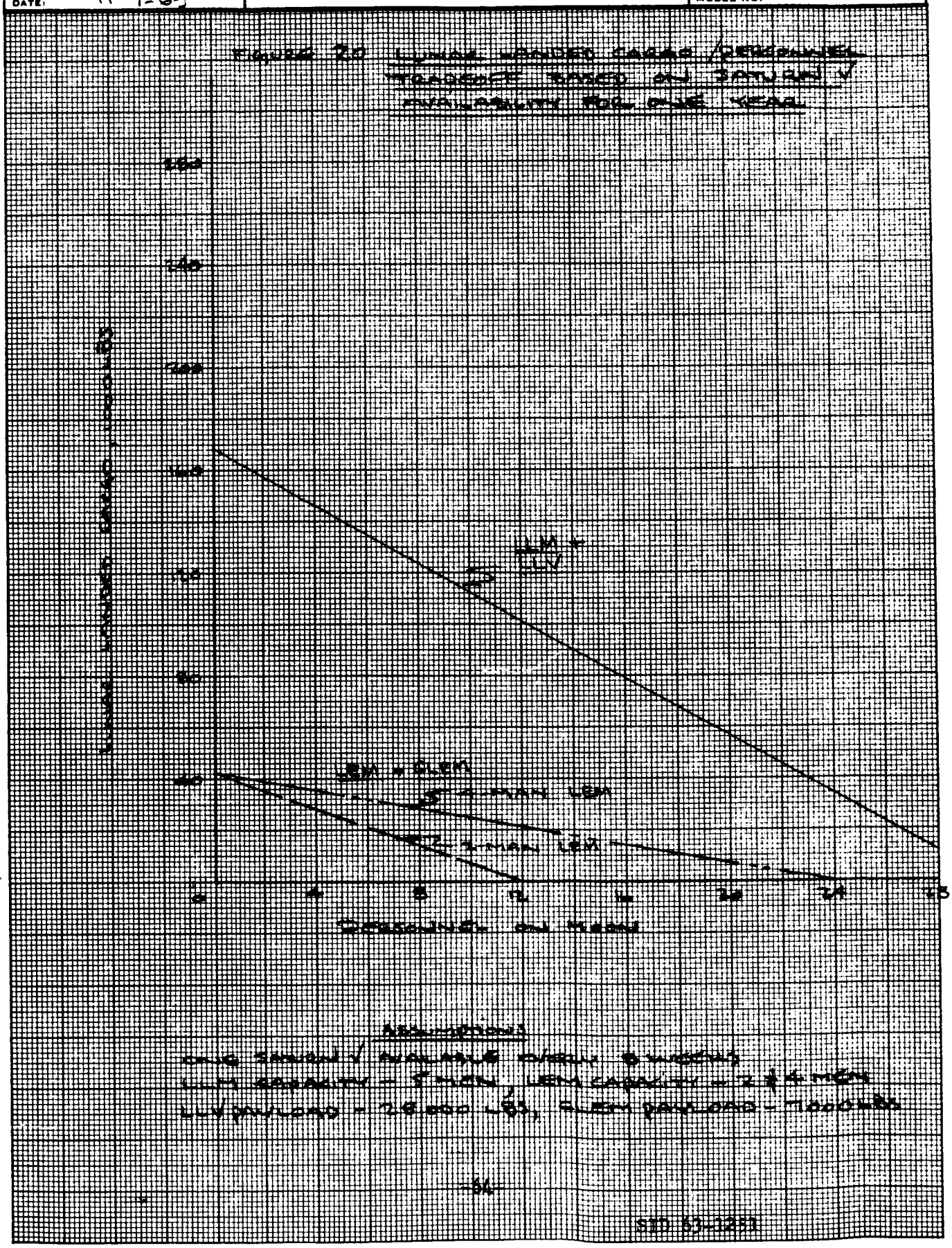
$$\frac{CW}{P}$$

For example, if the base complement is eighteen men, (P = 18), and if a three-man vehicle is used for rotation (A = 3) and eighteen Saturn V boosters are available per year for lunar base development (L = 18); then, if six months' stay time is the maximum permissible (M = 6), the number of cargo launches (of the eighteen launches) that can be landed is:

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FIGURE 20 LUNAR LANDED CARGO / PERSONNEL TRADEOFF BASED ON 3 MTU/MO V AVAILABILITY FOR ONE YEAR



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$$C = L - \frac{12\,P}{AM}$$

$$= 18 - \frac{12 \times 18}{3 \times 6}$$

$$= 18 - 12 = 6$$

On the other hand, if the stay time were extended to eight months, then:

$$C = 18 - \frac{12 \times 18}{3 \times 8}$$

$$= 18 - 9 = 9$$

Or, for a 33% increase in lunar stay time (from six to eight months), 50% more cargo (nine instead of six) launches could be made. By the use of Equation (1), numerous trade-offs between the logistics variables can be developed. Figure 21 is a nomograph of Equation (1).

Figures 22 and 23 show the impact of a two-, three-, five-, or seven-man transport vehicle on lunar base buildup with 5000 lbs/man/year and 10,000 lbs/man/year of expendables. The basic ground rules assumed are:

a) six months stay time, and

b) 7,000 lb lunar payload capability (CLEM) associated with the two-man (LEM 2) transport vehicle,

c) 28,000 lbs with the LEM 3,(3 Man LEM)

d) 30,000 lbs with Apollo 5, and

e) 35,000 lbs with the Apollo 7.

NOTE: * With an uprating of LEM to 3-man capacity, it is assumed that the cryogenic Apollo-type LLV with 28,000 lb landed payload capacity is available. Alternately, a 3-man direct mission Apollo II transport could be assumed with same result.

LUNAR LOGISTICS NOMOGRAPH

EQUATION: $(L-C) = \frac{12P}{AM}$

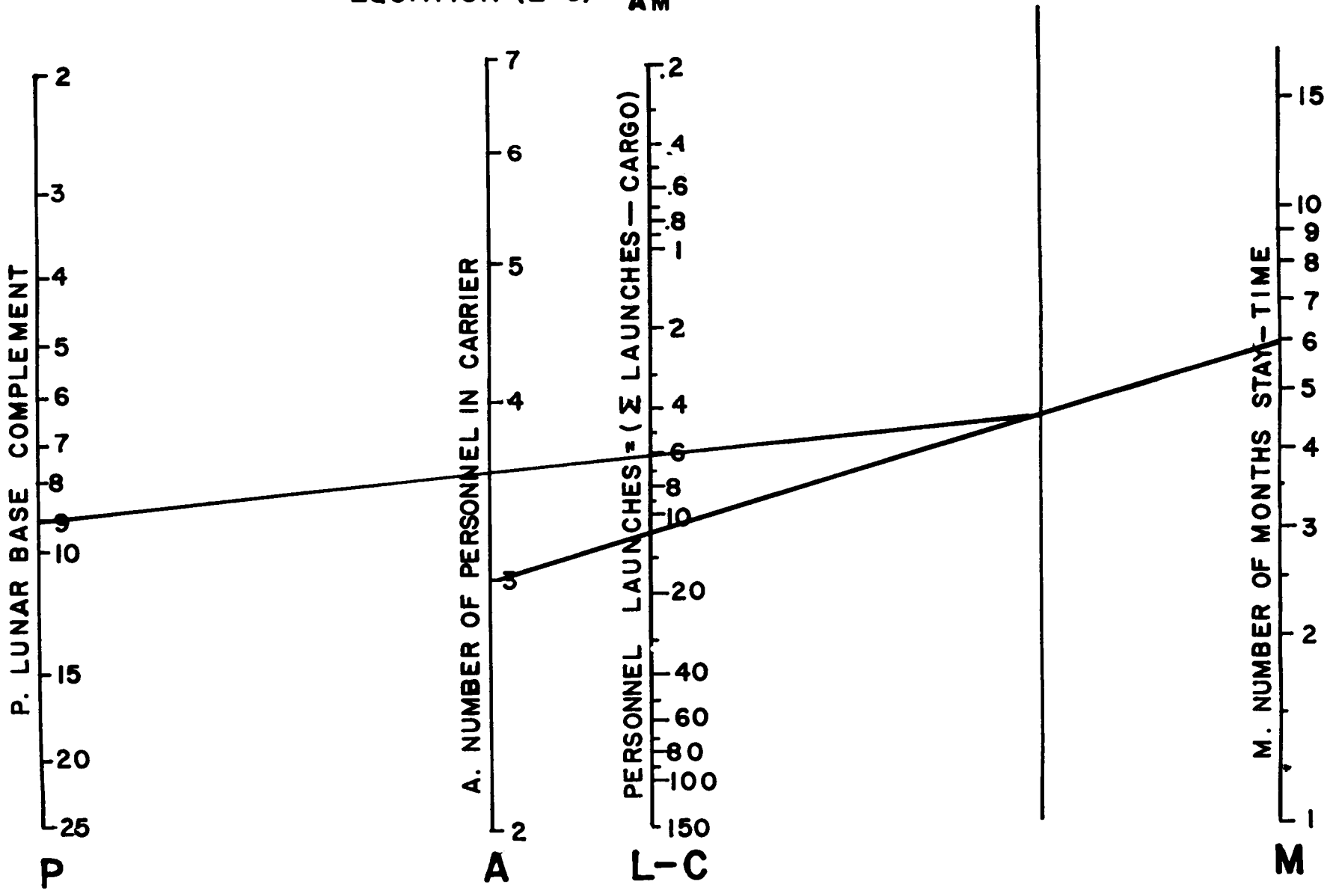


FIGURE 21

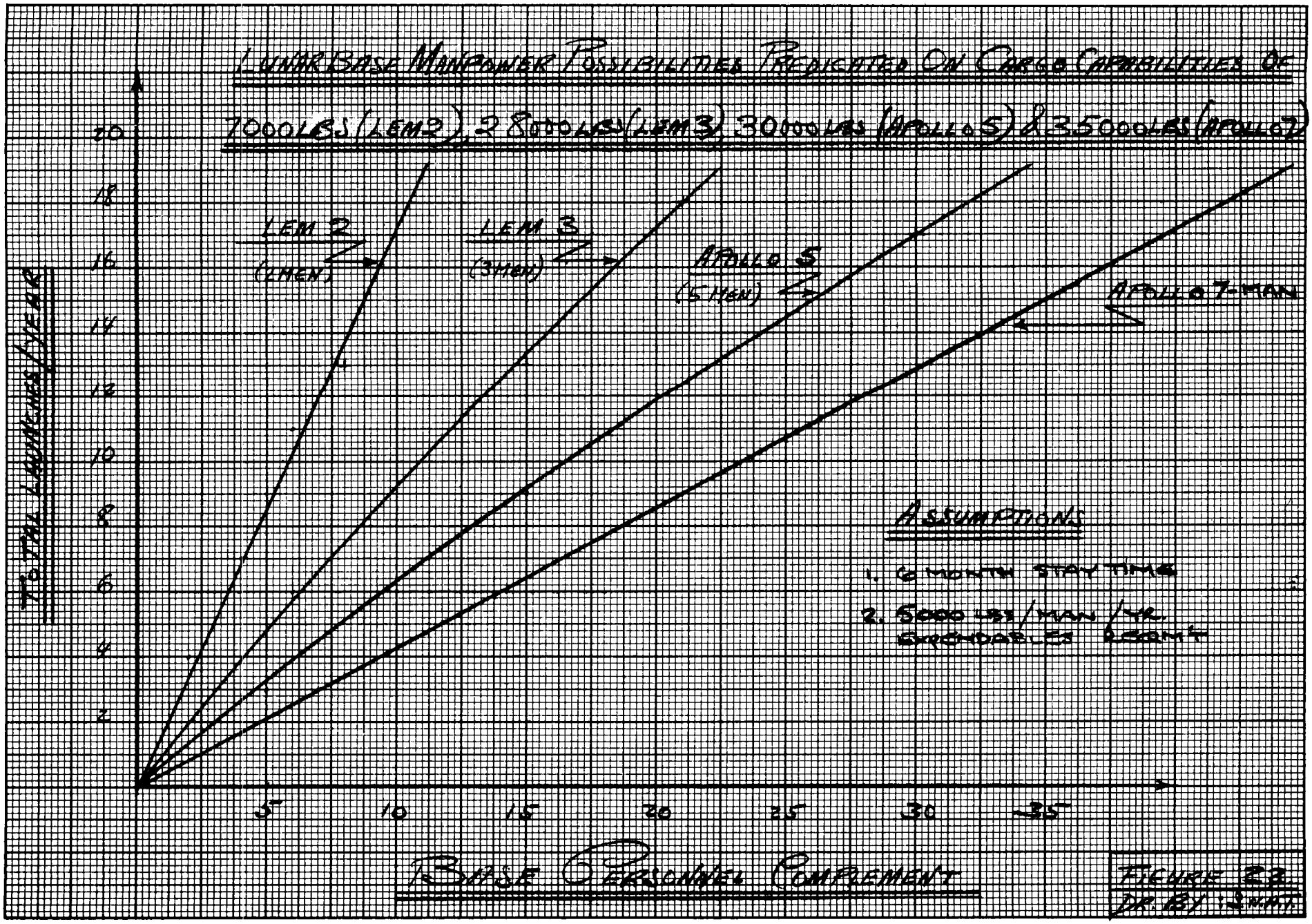


FIGURE 23
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Eight cases were considered. For each of the vehicles contemplated (LEM 2-man, LEM 3-man, Apollo II 5-man, and Apollo II 7-man), two different expendable cargo requirements per man per year were assumed (5,000 and 10,000 lb per man per year).

The 5,000 lbs/man/year is based on 13.72 lbs of expendable per man/day. The self-imposed 13 lbs/man/day of expendables requirement compensates adequately for the expected leakage and air-lock losses in view of the basic life support need which is approximately 6 lbs per man per day.

In the case of the 5,000 lb/man/year expendable cargo, Figure 22 indicates that the Apollo 5 and Apollo 7 possess definite advantages over either the LEM 2 or LEM 3 for increasing base manpower. This can be easily observed by finding the base manpower differentials between the Apollo 5 or 7 vehicle and the LEM 3 or 2. For example, the Apollo 7 will sustain a 42 man base with 18 launches per year, and the Apollo 5 a 30 man base with 17 launches/year, whereas the LEM 3 can only support only a 21 man base with 18 launches/year and the LEM 2 a 14 man at the high launch rate of 24/year.

When the requirement for expendables is made 10,000/man/year, the base manpower buildup capabilities, with the different vehicles, are seen in Figure 23 to be almost as significant as those for the 5000 lbs/man/year case (Figure 22). The increase in base personnel achieved by using the Apollo 5 or the Apollo 7 instead of the LEM 3 or the LEM 2 is still appreciable, although not as considerable as in the case of the 5000 lbs/man/year. Table 5 contains all the data used to draw Figures 22 and 23.

TABLE 5
LUNAR BASE LOGISTICS (SIX-MONTHS PERSONNEL STAY TIME)

LEM 2-MAN (5000 Lb/Man/Yr Expendable Requirement)					LEM 3-MAN (5000 Lb/Man/Yr Expendable Requirement)				
Lch/Yr	Base Pers Comp	Cargo Lch/Yr LEM = 7000 Lbs	Available Expendables Lbs/Man/Yr	Total Launches Per Yr	Pers Lch/Yr	Base Pers Comp.	Cargo Lch/Yr LLV = 28,000 Lbs	Available Expendables Lbs/Man/Yr	Total Launches Per Yr
2	2	2	7000	4	2	3	1	9330	3
4	4	3	5250	7	4	6	2	9990	6
6	6	5	5820	11	6	9	2	6670	8
8	7	5	5000	13	8	12	3	7000	11
8	8	6	5250	14	10	15	3	5600	13
10	10	8	5600	18	12	18	4	5900	16
12	12	9	5200	21	*14	21	4	5060	18
*14	14	10	5000	24	16	24	5	5820	21
					18	27	5	5180	23
(10,000 Lb/Man/Yr Expendable Requirement)					(10,000 Lb/Man/Yr Expendable Requirement)				
2	2	3	10,500	5	2	3	2	18,000	4
4	4	6	10,500	10	4	6	3	14,000	7
6	6	9	10,500	15	6	9	4	12,450	10
8	7	10	10,000	17	8	12	5	11,650	13
8	8	12	10,500	20	10	14	5	10,000	15
					10	15	6	11,200	16
10	10	15	10,500	25	12	18	7	10,900	19
12	12	18	10,500	30	14	21	8	10,650	22
*14	14	20	10,000	34	16	24	9	10,500	25
					18	27	10	10,370	28
					20	30	11	10,270	31

*Indicates optimum utilization of vehicle transport and logistic capability.

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TABLE 5 (Cont'd)
LUNAR BASE LOGISTICS (SIX-MONTHS PERSONNEL STAY TIME)

Apollo 5-Man (5000 Lb/Man/Yr Expendable Requirement)					Apollo 7-Man (5000 Lb/Man/Yr Expendable Requirement)				
Pers Lch/Yr	Base Pers Comp	Cargo Lch/Yr 30,000 Lbs	Available Expendables Lbs/Man/Yr	Total Launches Per Yr	Pers Lch/Yr	Base Pers Comp	Cargo Lch/Yr LLV = 35,000 Lbs	Available Expendables Lbs/Man/Yr	Total Launches Per Yr
2	5	1	6000	3	*2	7	1	5000	3
4	6	1	5000	5	*4	14	2	5000	6
4	10	2	6000	6	*6	21	3	5000	9
6	12	2	5000	8	*8	28	4	5000	12
6	15	3	6000	9	*10	35	5	5000	15
8	18	3	5000	11	*12	42	6	5000	18
8	20	4	6000	12	*14	49	9	5000	21
10	24	4	5000	14					
10	25	5	6000	15					
*12	30	5	5000	17					
14	35	6	5140	20					
16	36	6	5000	22					
(10,000 Lb/Man/Yr Expendable Requirement)					(10,000 Lb/Man/Yr Expendable Requirement)				
2	3	1	10,000	3	*2	7	2	10,000	4
2	5	2	12,000	4	*4	14	4	10,000	8
4	6	2	10,000	6	*6	21	6	10,000	12
4	10	4	12,000	8	*8	28	8	10,000	16
6	12	4	10,000	10	*10	35	10	10,000	20
*6	15	5	10,000	11	*12	42	12	10,000	24
8	18	6	10,000	14					
8	20	7	10,500	15					
10	24	8	10,000	18					
10	25	9	10,800	19					
*12	30	10	10,000	22					
14	35	12	10,300	26					
16	36	12	10,000	28					

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The sensitivity of base personnel complement to vehicle and launch rate can be seen from the following table based on the 5000 lbs/man/year expendables requirement, and on 7000 lbs cargo capability for the cargo LEM 2, and on cargo capabilities of 28,000, 30,000, and 35,000 lbs respectively for the LEM 3, Apollo 5, and Apollo 7.

BASE COMPLEMENT SENSITIVITY TO VEHICLE CAPABILITY

Launches	Vehicles	Base Pers. Complement	% Increase in Pers.
6	LEM 2	2	---
	LEM 3	6	200
	Apollo 5	10	67
	Apollo 7	14	40
9	LEM 2	4	---
	LEM 3	9	125
	Apollo 5	15	67
	Apollo 7	21	40
12	LEM 2	6	---
	LEM 3	12	100
	Apollo 5	20	67
	Apollo 7	28	40
18	LEM 2	10	---
	LEM 3	21	110
	Apollo 5	30	43
	Apollo 7	42	40

The table above points out the following trends:

a) If only six launches per year are available and 5000 lb/man/year of expendable support is required, then the LEM 3 would enable the establishment of a six-man base, the Apollo 5 of a 10 man base, and the Apollo 7 a 14 man base.

b) When nine launches per year and 5000 lbs/man/year are available, the Apollo 5 would enable a lunar base personnel increase of 67% over the LEM 3, and the Apollo 7 a 133% increase over LEM 3.

c) If twelve launches per year are available and 5000 lb/man/year of expendable support is required, then the Apollo 5 would enable the establishment of a twenty-man base, and Apollo 7 a 28-man base.

d) If eighteen launches per year are available and 5000 lb/man/year of expendable support is required, then the Apollo 7 offers an appreciable increase in base personnel; however, the development cost and additional production cost would need to be weighed against the advantages of having 32 more men on the Moon than LEM 2 could support.

e) Finally, it is significant to note that a twenty-one man base

utilizing the LEM 3 would require eighteen launches per year, while a twenty-man base, utilizing the Apollo 5, would require only twelve launches per year. The $600 million thus saved would need to be compared with the R&D and increased production costs inherent in the use of the Apollo 5.

Launch Facility Analysis

The approach used in this analysis of launch facilities in the 1970-75 time period was to use currently available NASA planning information for Apollo and Saturn in the 1967-68 period and then make certain judicious but conservative estimates of probably conditions which would exist in 1970-75 for Apollo II. The analysis considers: (1) a NASA-suggested launch rate for lunar base planning studies of one successful Saturn V launch every eight weeks (NASA HQ RFP #10-1132, dated May 3, 1963), (2) the maximum launch rate possible with presently planned facilities, and (3) the launch rate that would be possible if certain expansion of facilities took place.

Lunar base missions will require use of the Saturn V booster and must, therefore, be made from Launch Complex 39 facilities. Launch Complex 39, according to current information, is assumed to include the facilities listed below (additional facilities which would be available under a possible expansion program will be discussed later):

1) Four high bay areas in the Vertical Assembly Building (VAB) for S-1C and integrated booster/spacecraft checkout.

2) Four low bay areas in the VAB for S-II assembly and checkout.

3) Four low bay areas in the VAB for S-IVB assembly and checkout.

4) Three launch pads.

Currently planned assembly and checkout times (working days) for facilities in the 1967-68 time period are as follows (see Figure 25):

	Low Bays	High Bay	Launch Pad
S-IC Stage	-	45	13
S-II Stage	26	45	13
S-IVB Stage	40	45	13
Payload	-	15	13

From the above and Figure 25, it is apparent that low-bay assembly and checkout of the upper stages is not critical but that the high by plus launch pad time of fifty-eight days will be governing (the checkout and control facilities are utilized until the vehicle is launched so that physically moving to the launch pad does not permit use of the vacated high-bay area by another

vehicle). Pad refurbishment time of one month (26 working days) has been assumed and will easily permit three launch pads to handle the vehicles at the rate they are processed through the four high-bay areas. Prior to being mated with the Saturn V, the payload must be processed through the operations and checkout (O&C) facilities on Merritt Island.

At a briefing on September 26, 1963, NASA representatives from AMR indicated that O&C area facilities would be adequate to support the launch capabilities of Complexes 34, 37, and 39. It should be noted that current Apollo launch schedules allow about 128 working days at AMR for manned flights. However, the experience to be gained on Apollo prior to the assumed time period of 1970-75, the probable mix of lunar mission payloads which would include unmanned Lunar Logistic Vehicle (LLV) modules requiring less checkout time, and the utilization of all available assembly and checkout facilities in the O&C area are expected to permit attainment of projected launch capabilities. A typical booster/space vehicle flow chart is depicted in Figure 20.

With the fifty-eight-day flow time constraint due to high bay and launch pad requirements, the launch rate per year is determined by the number of high bays (4) and the number of working days per year (assumed as 305, based on a six-day work week with eight holidays).

$$\text{No. Launches} = 4 \times \frac{305}{58} = 21 \text{ per year}$$

Obviously, then, presently planned facilities can easily support the austere launch rate of one each eight weeks suggested by NASA RFP #10-1132.

At the NASA briefing noted in a preceding paragraph expansion of Complex 39 facilities, to include two additional high bay areas and a fourth launch pad, was noted as a possibility. O&C area facilities would be similarly expanded. With seven days per week around-the-clock operation, it was indicated a launch rate from Complex 39 as high as forty-eight per year could be attained.

Turning now to the feasibility of "bunched" launches which might be desired in the build-up phase; particularly, of establishing a lunar base, there should be no operational problem in launching three vehicles at four-day intervals with the three launch pads assumed available. If the first module arrived at the beginning of lunar day, there would be approximately six Earth days available after the third module arrived at the lunar base before the onset of lunar night. The four-Earth-day interval between launches requires that only one vehicle be tracked at a time and, of course, permits verification of arrival of each module before committing the next. A fourth launch could be made at the onset of the next lunar day, assuming that four modules are needed to complete a temporary base. Thereafter, aside from economic limitations, further launches would be limited by the launch capabilities of Complex 39 discussed previously. Within these limitations, additional three shot "bunches" could be scheduled after a suitable build-up period. Such "bunching," of course, causes uneven work loads in all operational and logistic support areas as compared to evenly spaced launches.

The above discussion does not provide an emergency rescue standby vehicle which, if required, would allow only two shot "bunches" if only three launch pads are available.

Throughout the foregoing discussion, it has been assumed that the Deep Space Instrumentation Facility (DSIF) could be available for Apollo II missions as needed, including the twelve-day period for "bunched" launches. This facility is required on lunar missions since the GOSS capability extends only to 5,000 miles from the Earth. There are other proposed missions which would require the DSIF in the 1970-75 period. In addition to possible space station requirements and a possible Venus fly-by, there are three favorable times for Mars missions with possible dates as follows (dates of launch and Earth return if required):

Launch	Earth Return
May 1971	June 1972
August 1973	September 1974
September 1975	November 1976

It is reasonable to assume that, during at least one of these periods and perhaps during all three, there would be missions to soft-land instruments, at least on the Martian surface, with manned flights also a possibility. It would be necessary to schedule lunar missions, if Martian missions were planned during the above periods, for those times during the Martian missions when the DSIF was needed only on an intermittent basis.

CONCLUSIONS

Because of the high cost of a Saturn V, and in view of their limited availability in the time period 1965-1973, the Apollo II LLM/LLV approach to lunar exploration and base establishment appears to be far superior in terms of both time and cost than the LEM/CLEM approach because of the lesser number of Saturn V's required to perform the various missions.

Using the criterion of cost in dollars per man-week-unit of exploration, the Apollo II LLM offers a superior approach to lunar exploration to the LEM/CLEM approach.

The limitations of maximum lunar base complement inherent in the LEM approach indicates that, if a base of significant size is to be established on the Moon, then a vehicle with superior capabilities will be required.

If the importance of locating water on the Moon is (in addition to life support requirements) to enable lunar refueling and, if the Saturn V is assumed to be the only booster available for major lunar missions, then a cryogenic vehicle such as the Apollo II will need to be developed.

Unless the Saturn V availability for lunar missions increases beyond

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SATURN V ASSEMBLY, CHECKOUT & LAUNCH - COMPLEX 39

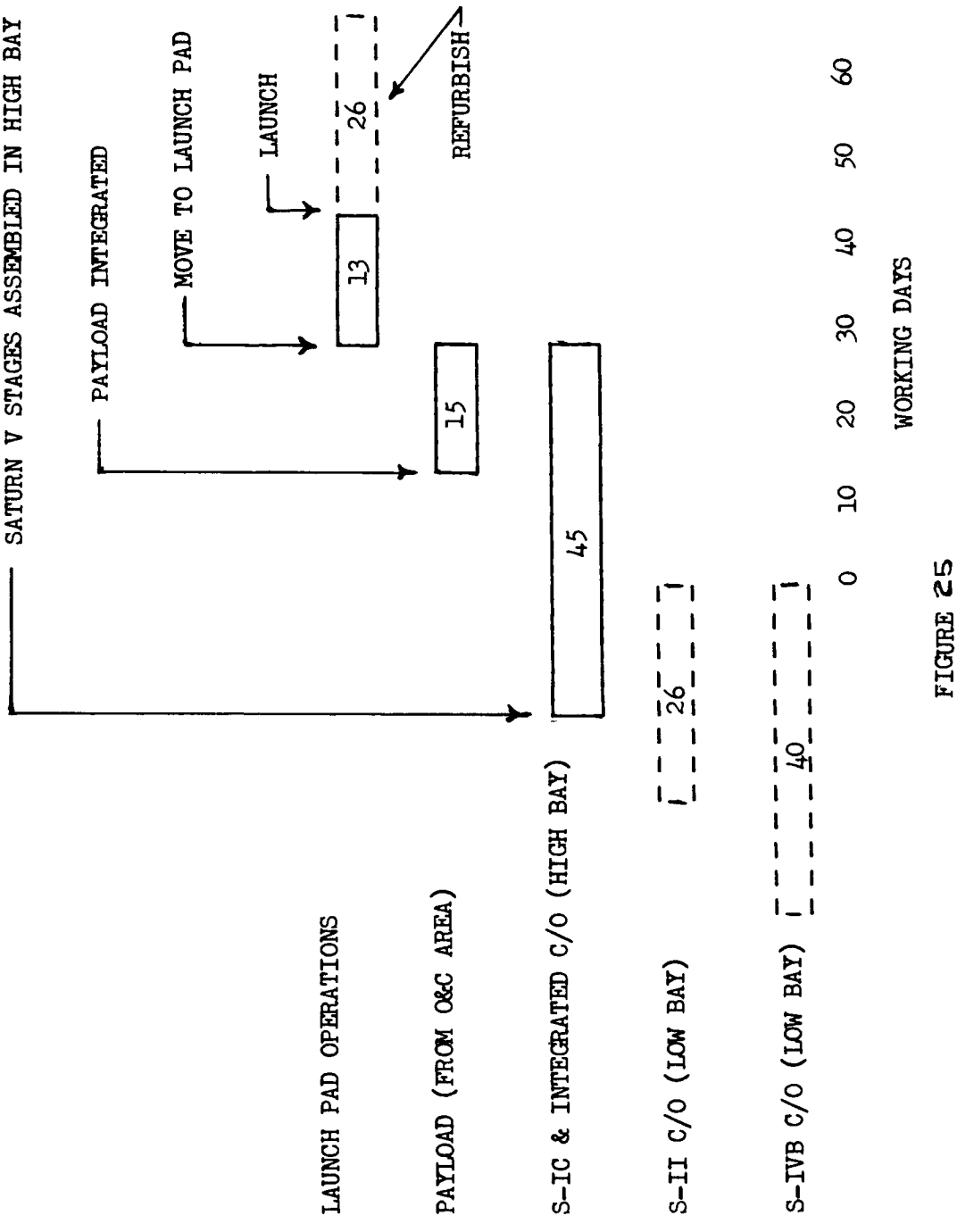


FIGURE 25

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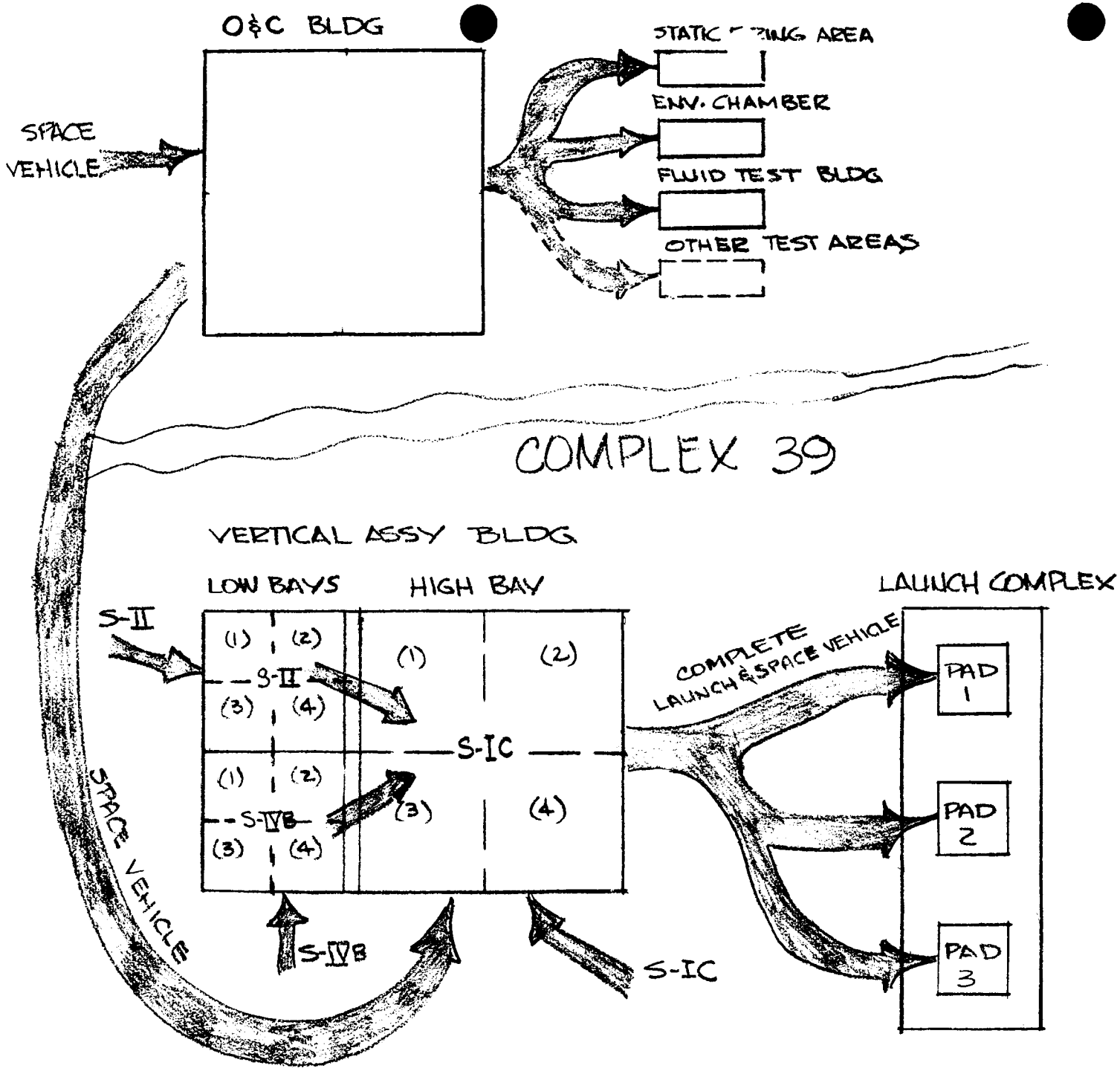


FIGURE 26

BOOSTER/SPACE VEHICLE FLOW CHART

one every eight weeks, or unless a larger or more efficient booster becomes available, it appears that a lunar base complement of approximately fifteen is near the optimum size (assuming the use of the Apollo II LLM/LLV). With LEM/CLEM, the base would be smaller.

LUNAR SPACE SUITS

PRELIMINARY REQUIREMENTS

The ability of man to perform outside lunar facilities will have an important impact on the design of equipment and bases for lunar expeditions. Facilities, tools, power-generating equipment; and, actually, all devices to be used for lunar operations and travel, must be designed for efficient utilization by man encased in a spacesuit or other contrivance to protect him from the Moon's environment. Stated another way, environmental protection for man on the Moon must be based on two categories of design criteria: (1) The equipment must protect man from the lunar environment and supply him an atmospheric envelope in which he can survive; and (2) the equipment must permit man to perform as necessary to maintain himself, conduct scientific activities, explore, prospect, and construct and maintain lunar facilities.

Pressure suits used in the transporting spacecraft will also be used during initial lunar exploration. These suits will have been designed for both space and lunar applications.

Shortly after the early landings, lunar exploration will be initiated and base construction will be started after a favorable site is found. There will be increasing logistics support for these later expeditions and lunar resources will be used as they are discovered and means to take advantage of them are devised. To take full advantage of man's capabilities, it is important to develop, by this time, special suits for lunar use. In addition, it would not seem wise to risk destroying or incapacitating the suits used for space travel in hazardous lunar activities. As more supplies become available, it may well be found desirable to furnish a variety of special-purpose lunar suits.

There is also sound economic grounds for unique lunar suits to increase man's efficiency. Without quoting dollar costs, it is clearly expensive to transport men and materials to the Moon. The cost of labor has been estimated in tens of thousands per man-hour. Any increase in man's efficiency will speed up the accomplishment of lunar objectives and result in earlier establishment of scientific facilities. Current pressure suits greatly increase the effort required to perform any activity. While the reduced gravity on the Moon will make work easier, it has been estimated that this advantage is approximately off-set by the restrictions imposed by the pressure suit being designed for use in the Apollo spacecraft.

Current suits and those which are for use in first generation spacecraft are designed to keep volume at a minimum so as to be compatible with the space restrictions and access provisions. Increasing the volume of

lunar suits would allow a design more closely approximating a constant pressure suit but would result in a heavier, bulkier suit complicating spacecraft design. In order to permit easier movement and reduce the metabolic requirement of the suited man, design and development effort must be expended in achieving a space suit of minimum volume but less constraining than those currently proposed. The increased efficiency of man and savings in life support requirements will more than offset the relatively small development cost.

There is little to be gained at this time in belaboring the lunar suit problem with hypothetical and controversial conjecture. This survey stipulates some criteria for suits which will be required for lunar exploration, facility construction, and base maintenance. Additional study will be necessary to raise the level of confidence with which one can regard the information on the following pages. However, it is hoped that these early estimates will be usable for preliminary design of equipment and facilities for lunar exploration and base construction. A continuing effort will be required to assure the compatibility of lunar base design and man's performance capability.

LUNAR ENVIRONMENTAL FACTORS

Environmental factors are discussed here only to point out their relationship to lunar suit design. Any one of the parameters could be treated at great length, but it is neither feasible nor desirable to do so at this time.

SURFACE TEMPERATURE (-240°F to +240°F)

Early expeditions will be planned to take advantage of relatively temperate periods. Specific exploratory trips could be scheduled for either of two thirty-hour periods during the lunar day when the surface temperature ranges from 0°F to 120°F. Current anthropomorphic pressure suits could be used in this range. A different design suit may be required for lunar night use.

VACUUM (About 10^{-13} torr)

Practically an absolute vacuum. A highly reliable protection will be required since man cannot survive more than momentarily if exposed.

RADIATION

Still a subject of considerable debate. More study is needed to estimate expected types and levels and to specify the protection which will be required. Some protection is certainly needed.

METEOROIDS

No really accurate data on how serious a danger meteoroids and lunar particles splashed up from hits will be. Some protection must be planned until realistic requirements can be specified from observed conditions.

TERRAIN

Not generally regarded as a serious hindrance to man's travel. Rills may require special equipment for bridging, and pockets of dust should be considered. High and low surface temperature make special boots necessary. Design to protect against the possibility of tearing suits may be unnecessary because of the provisions to protect against meteoroids. Climbing (including the possibility of falls) and crevasses and cave exploration should be taken into account in suit design and provisioning of accessories.

GRAVITY

Reduced gravity will alleviate problems associated with suit weight and bulk but must also be considered as a limiting factor. The presence of gravity will ease problems associated with liquid transfer which plague the designers of suits for use in space.

Lunar Performance Requirements

Mans' performance while suited on the Moon will fall into three general categories:

1) Operations outside lunar facilities:

   a. Construction and repair of facilities.

   b. Movement from facility to facility.

   c. Exploration and prospecting.

2) Operations in unpressurized facilities:

   a. Maintenance, servicing, troubleshooting, and repair of equipment.

   b. Packing and unpacking equipment and supplies.

3) Emergency operations:

   a. Back-up measure in pressurized facilities.

   b. Walking to base from stalled vehicle.

   c. Rescure of disabled persons.

These few activities are representative of the sort of activities which will be required during lunar expeditions. From this sample, it can be deduced that lunar crews will be required to perform a wide variety of tasks while suited. To list a few, he must be able to:

1) Walk

2) Climb

3) Dig

4) Crawl

5) Use a variety of tools

6) Manipulate controls

7) Lift and carry heavy objects

RECOMMENDATIONS FOR LUNAR SUITS

In view of the lunar environment and operational requirements discussed briefly above, it is possible to make some recommendations for lunar suit design at this time. It is quite possible that these may change with further study and certainly more precise criteria will result from lunar expeditions.

1) An anthorpomorphic suit especially designed for lunar use may be desirable.

2) The development of a less constraining (by present standards) suit should be considered.

3) The suit should be modular. There should be special separate outer garments for protection against various conditions of heat, cold, terrain, radiation, meteoroids, and other variables.

4) A transparent (bubble type) helmet would be desirable if not precluded by thermal and radiation factors.

5) Biomedical and environmental sensors and warning systems should be incorporated.

6) Communications and telemetry equipments should be integrated.

7) A thorough study of accessories should be made to determine the desirability of such things as inflatable sunshades, shelters, lights, climbing equipment, and special emergency supplies.

8) A small basic environmental control back-pack should be designed

to ease mobility in cramped quarters. Provisions for the addition of several supplementary, throw-away units should be made.

9) To preserve the back-pack expendables, there should be provision for umbilical hookups to vehicle and work-station environmental control systems whenever possible.

10) Vehicle and back-pack environmental control units should be interchangeable for use in walk-back and rescue operations.

SPACE SUIT EFFECTS ON LUNAR BASE DESIGN

Lunar suits will have many primary and secondary effects on lunar mission planning and base design. Some of the more important are:

1) Earth - Moon Logistics

   a. Lunar suit weight and volumetric data will be basic considerations in planning Earth-lunar flights and logistical support.

   b. All lunar operations must be analyzed to determine the relative economic, logistical feasibility and merit of using manpower or mechanical methods for transporting, fabrication, and operation of all lunar base components.

2) Facility Construction

   a. Lunar suit size will dictate minimum hatch dimensions for lunar structures.

   b. Performance capability of suited personnel must be considered in design and planning construction equipment, tools, methods, and schedules.

3) Equipment Design and Installation

   a. Suited performance must be evaluated in the transporting and installation plans for each separate piece of equipment. Both expected and emergency environmental conditions should be considered.

   b. Operating capability of suited personnel must be used as criteria for control design and in the planning, servicing, maintenance, and repair concepts.

Suit Provisioning

As soon as possible after logistics support is available, anthropomorphic lunar suits and accessories should be provisioned on the following basis:

[redacted]

Original Requirements

1 suit per man

1 spare suit per four men or fraction thereof

2 rechargeable portable life support systems (PLSS) back-packs per suit

2 accessory kits per three suits or fraction thereof

1 spare parts kit (PLSS) per back-pack

1 spare parts kit (suit) per back-pack

4 decontamination kits per suit

Replacement Requirements

1 suit per each two lunar man-years

1 PLSS back-pack per lunar man-year

1 accessory kit per each four lunar man-years

1 spare parts kit (suit) per lunar man-year

1 spare parts kit (PLSS) per lunar man-year

1 decontamination kit per each six lunar man-months

Shipping Weight

Lunar Suit	-	80 lb*
PLSS Back-pack (uncharged)	-	30 lb
Accessory Kit	-	12 lb
Spare Parts Kit (suit)	-	15 lb
Spare Parts Kit (PLSS)	-	11 lb
Decontamination Kit	-	2 lb

*Based on two pairs of gloves and boots and two over-garments to provide capability for operation in temperate and extreme thermal conditions. Net weight of suit for temperate conditions is 48.3 lb; for extreme, 54.0 lb.

Shipping and Storage Volume

	Shipping Volume (Cubic Inches)	Lunar Storage Volume (Cubic Inches)
Lunar Suit*	33,373	38,733
PLSS Back-pack	1,560	1,560
Accessory Kit	1,728	1,728
Spare Parts Kit (suit)	144	144
Spare Parts Kit (PLSS	384	384
Decontamination Kit	216	216

Structural Considerations

Minimum hatch and tunnel size for all lunar construction should be 36" in diameter. This is based on:

Shoulder measurement Apollo suit (Note: Apollo tunnel diameter is 27.562")	27 in
4" added to lunar suit by larger convolute joints	4 in
1" added for extreme thermal protective garment	1 in
4" added to allow for tracks and sled or crawling	4 in
Total	36 in

The above assumes round tunnels and hatches. However, arch-shaped tunnels and hatches would save material and ease tunnel sled transportation. The relative sealing problems should be examined. An arch 34 inches wide by 30 inches high would allow adequate mobility and access. The height dimension could be reduced by 6 inches if the back-pack could be removed during hatch and tunnel operations (see Figure 27).

*Shipping volume based on three 24" x 24" x 16" boxes for a pressure and two thermal protective garments. Nested storage (68" x 27" x 18") is assumed for hanging the three garments during lunar storage.

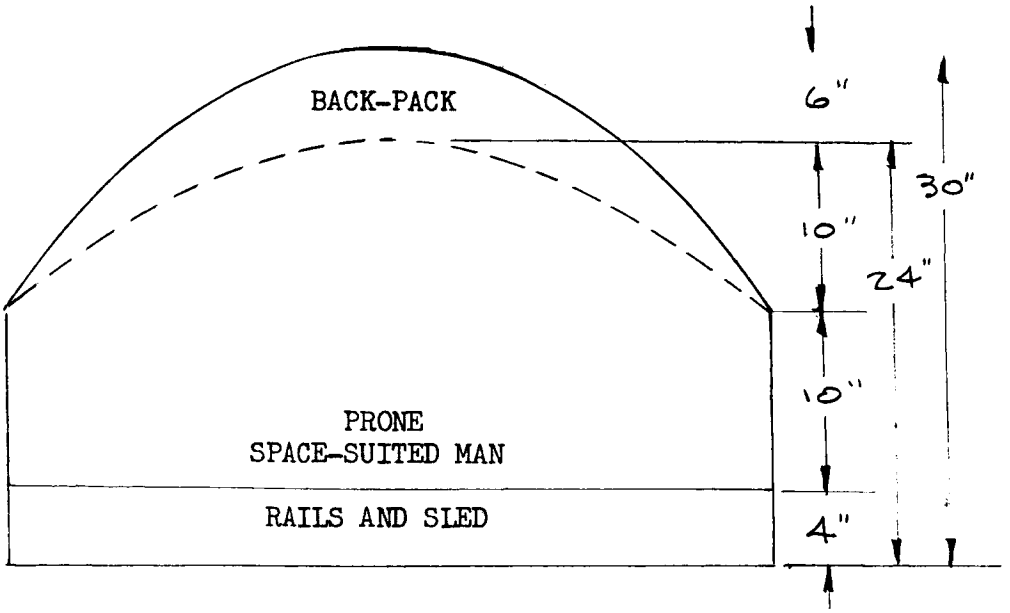


FIGURE 27

ARCH TUNNEL CONFIGURATION FOR LUNAR FACILITIES

(The same external dimensions would pertain for hatches.)

It should be remembered that the suited man is only one criterion for hatch and tunnel size. As an example, the Apollo hatch is considerably bigger than the tunnel because larger access was required to install the couches.

The hatch dimensions of 36 inches diameter are for personnel in lunar suits. Thus the hatch from a pressure lock to the lunar exterior, should conform to these dimensions. The hatches from the pressure locks to the interior of pressurized chambers should be a minimum of 28 inches in diameter, which would allow a shirtsleeved man (or one in an Apollo suit) to enter the chamber. This arrangement would simplify the sealing problem somewhat and would permit storage of lunar suits in the pressure lock and thus reduce the contamination hazard for pressurized modules.

1) Equipment (possibly a vacuum hose) for decontaminating lunar suits in the air-locks.

2) Storage for lunar suits in the air-locks.

3) Provisions to maintain air-lock pressure during suit donning and checkout; and doffing and decontamination (about 10 minutes each).

4) Storage of spacecraft pressure suits for emergency use in pressurized modules.

Before structural modular designs are finalized, careful analyses should be completed to determine where manpower and machine techniques should be utilized in base construction. Only comprehensive studies will develop reliable standards for mans lunar performance. However, a few generalizations may be of some help at this time.

1) Suited or in shirt sleeves, man's capability to lift will be at least tripled in the lunar environment. (No problem with objects up to 300 lb.)

2) Endurance will probably be limited in spacesuit operations. Such activity as carrying objects for long distances or excavating should be figured at about normal Earth capability.

3) In environmentally conditioned areas where man can work in shirt sleeves, his endurance should be as on Earth with ability to lift as indicated in (1).

4) The combination of power and hand implements (such as a vehicular cran with handpowered pulleys and hoists) should be considered.

Analysis of construction methods should consider:

1) Logistics feasibility of transporting tools and construction equipment to the Moon in the time period required vs manhours available and hand tools and job aids required to perform the tasks.

2) The long-term requirement for equipment transported to the Moon or its possible conversion for another use.

3) The relative support costs of using manual or automated methods. The requirement to return men to Earth according to the established rotation schedule must be taken into account.

Equipment Design and Installation

Plans to install lunar equipment should be governed by the same general parameters discussed under "STRUCTURAL CONSIDERATIONS."

Lunar equipment design must take into account both normal and emergency operation, servicing, maintenance, and repair. Specific human factors design for lunar equipment will depend on man-machine task allocations, the types of lunar suits developed, and the specification of maintenance concepts. Until lunar specific data is available, information developed for Mercury, Gemini, Apollo, and LEM spacecraft; and for space stations should be used as minimum standards.

APPENDIX A

A POSSIBLE SEQUENCE

FOR

LUNAR EXPLORATION

A POSSIBLE SEQUENCE FOR LUNAR EXPLORATION

Although there may be many variations of lunar development schedules, it is of interest to set down one possible sequence as a guide to lunar development thinking and to aid in focusing interest on the more important variables. The initial phase agrees with S&ID's proposed Apollo development schedule; the launch and mission sequence during the Exploration Phase is strictly arbitrary.

The following schedule version is based on a variety of assumptions; which, if granted, result in its evolution. For example, no allowance is made for Saturn V failures. This may be rationalized by saying that the assumed availability schedule of one Saturn V every three months until the end of 1969, rather than one Saturn V every eight weeks (4 vs 6 per year) makes an allowance for two failures. Also, an availability rate of one every eight weeks means that 6.5 will be available each year. Only six are shown.

This schedule should be recognized as being highly tentative, and periodic revisions to it are planned as additional knowledge is obtained.

Saturn V Serial No.	Launch Date	Mission
Initial Phase		
505	Sept. 1967	Circumlunar flight - retro into lunar orbit, perform photo-reconnaissance.
506	Nov. 1968	Initial manned LEM landing.
Exploration Phase		
507	Jan. 1969	Saturn V assigned to rescue reserve with 5-man CM.
508	March 1969	Earth-orbiting space station launch while evaluating lunar data and making required modifications.
---	---- ----	Rescue reserve Saturn V equipped with Apollo II LIM.
509	May 1969	Apollo II with 3-man LIM - landing at Sinus Medii. Seven days - explore highland, maria, local craters.
510	July 1969	Rescue reserve Saturn V rotated.

507	July	1969	Apollo II with 3-man LIM - landing on lunar dome. Seven days - explore rilles.
511	Sept.	1969	Apollo II with 3-man LIM - landing on lunar dome. Seven days of exploration and seismic tests.
512	Nov	1969	Apollo II with 3-man LIM - landing at Copernicus or Triesnec ker. Seven days - explore crater and walls. (lunar scooter.)
513	Jan.	1969	Apollo II with 3-man LIM - landing at Alphonsus. Four days - two-man "stripped-down" version for landing at approximately 13°S latitude.

Henceforth, one Saturn V available every eight weeks (indicated by NASA)

514	March	1970	Apollo II with 3-man LIM landing at photo reconnaissance selected site. Seven days - search for minerals.
515	May	1970	Apollo II with 3-man LIM landing at another photo reconnaissance selected site. Seven days search for minerals.

Site Verification & Temporary Base Establishment Phase

516	July	1970	Accumulate Saturn V's while evaluating exploration data and selecting verification site.
517	Sept.	1970	
518	Nov.	1970	
519	Jan.	1970	Launch accumulated site-verification modules utilizing all launch pads except one for rescue reserve Saturn V.

Payloads:

516 Apollo II LIM with 5 men

517 Apollo II LLV 5-man Personnel Shelter Module

518 Apollo II LLV Payload - Equipment Module with construction

			vehicle and core-drilling equipment.
			519 Apollo II LLV Power Module (nuclear)
			Tasks are to activate modules, (bury reactor) take core samples, construct landing pad, verify vehicles reliability.
520	March	1971	Apollo II LLV with payload - Equipment Module mineral extracting pilot plant and vehicle.
	May	1971	Some Saturn V components used for Mars mission.
521	May	1971	Rescue reserve Saturn V rotated.
510	May	1971	Apollo II LLV with Personnel Shelter Module
522	July	1971	Apollo II with 5-man LIM - Additional base personnel - total complement now 10.
523	Sept.	1971	Apollo II LLV with Communication Module
524	Nov.	1971	Apollo II LLV with Life Support Module
525	Jan.	1972	Apollo II LLV with Maintenance Module
526	March	1972	Apollo II with 5-man LIM - Rotate initial base crew.
527	May	1972	Apollo II LLV with Power Module
528	July	1972	Rescue reserve Saturn V rotated.
531	July	1972	Apollo II with 5-man LIM rotate base crew.
529	Sept.	1972	Apollo II LLV with Personnel Shelter Module
530	Nov.	1972	Apollo II with 5-man LIM. Additional base personnel - total base complement now 15.

531	Jan. 1973	Apollo II LLV with Payload - Equipment Module base building and mineral processing equipment.
532	March 1973	Apollo II LLV with Fuel Module

Permanent Base Construction Phase

Dependent on site characteristics, minerals available, and environment; difficult to predict at this time.

[redacted]

PART II. LUNAR PERSONNEL TRANSPORT VEHICLES

INTRODUCTION

During 1962, S&ID conducted extensive studies leading to the preliminary design of a direct lunar landing and return spacecraft configuration, whose lunar landing stage could serve as a lunar logistics vehicle as well. The nominal 3-man, 14-day mission was made compatible with the Saturn V lunar injection capability by employing cryogenic high-energy propulsion.

As Part I of this report indicates, it is mandatory to ferry as many men to the lunar base as possible per Saturn V launch. Therefore, 5 and 7 men vehicles were also considered. These vehicles exceed the nominal injection capability of the Saturn V (90,000 lb to escape) and require moderate to intensive uprating. However, with regard to the increased transport capability of a 5 and 7 man vehicle, the development of an uprated C-5 will very well pay off for the support of lunar base operations.

The primary purpose of these two vehicles is to serve as shuttle vehicles rather than for exploration independent of lunar base operations. Therefore, no scientific equipment is provided, and extremely severe restrictions are imposed upon the crew. It is felt, however, that these restrictions can be accepted for the nominal 3-day outbound and inbound trips.

It should be noted that this part of the report concerning the 5 and 7 men vehicles represents essentially a feasibility study. Development of the cryogenic lunar landing and take-off stages needed for these vehicles, though highly desirable because of their **increased payload capability, has not yet** been initiated.

The discussion presented herein does not cover all aspects of the two vehicle configurations. Only those areas that differ considerably from the configuration described in SID 62-1467 (Lunar Mission System Studies; three man vehicle) are discussed, and a detailed weight breakdown is presented for both the five and seven man missions. **It should be noted that the direct mission vehicles are inherently designed for a 7-day lunar stay time; thus, each vehicle could essentially perform the function of serving as a complete short-term lunar outpost. However, for this mission the vehicle crew complement would probably be limited to two or three because of cramped quarters.**

DESIGN REQUIREMENTS

The mission objective for the two vehicles considered in this part of the report is basically crew rotation for lunar base operations, and very little compromise was made with this goal. The mission profile is the same as the one originally specified for the Apollo program; namely, a total duration of 14 days, direct landing on the Moon, and nominal 66-72 hour trajectories out - and inbound. The vehicle has to be compatible with an up-rated Saturn V.

For a direct landing vehicle with an Apollo-shaped command module, visibility at touch-down on the Moon has been felt by some to be a problem. It is believed that this problem has been solved satisfactorily, even if the pilot cannot visually observe the exact landing point, which may be obstructed in any event by dust clouds generated by engine exhaust. **(See Figures 32-35.)**

The total mission duration could be cut to 7 days in case of a "touch-and-go" crew rotation mission with corresponding reduction in weight of expendables and propellant boil-off. No reduction in the reserve Δ V allocations (5%) have been made, though this margin may be reduced once the re-supply flights become routine.

SELECTED CONFIGURATION

The same conceptual design philosophy presented for the 3 men direct lunar landing mission in Report SID 62-1467 was applied to the 5- and 7-man mission vehicle. A drawing of the complete 3 man vehicle is presented in Figure 28 . The five and seven-man vehicles will be boosted to escape by an up-rated Saturn V.

The lunar landing stage, described in detail in SID 62-1466, is cylindrical, with a diameter of 260 inches. Two throttlable (10/1) RL-10A engines are mounted side-by-side and parallel in the lower center part of the vehicle. They will be canted outboard 15° for the hover phase prior to landing on the moon in order to align the thrust centerlines through the center of gravity, thus avoiding immediate and violent overturning of the spacecraft in case of engine failure and providing for abort capability. The engine arrangement implies the use of toroidal LOX and LH_2 tanks to achieve minimum total height; the tanks are suspended with low heat conduction wires from the structure and insulated with super-insulation.

The lunar take-off stage is propelled by one RL-10 engine. This engine extends down into the upper center part of the landing stage and will not be used for lunar landing. The liquid oxygen is contained in a central ellipsoidal tank; the liquid hydrogen is contained in a toroidal tank surrounding the LOX tank. As in the landing stage, the propellant tanks are suspended from wires and insulated with super-insulation. The lunar take-off stage, its diameter tapering down into the command module outline, serves as service module as well. It contains the reaction control system, the electrical power supply system, and environmental control system in a manner similar to the

present Apollo I service module.

The five-man configuration will be very much like the three-man configuration shown in Figure 28 ; only the propellant tanks need to be slightly enlarged.

The seven-man configuration uses an enlarged command module, thus resulting in a slight change of the outline of the lunar take-off stage as shown in Figures 39 and 39A.

The landing gear arrangement will be basically the same for both vehicles as the one outlined in SID 62-1467.

Egress and ingress into the command modules will be accomplished through the airlock on top. In an emergency, the C/M could be decompressed and egress and ingress is feasible through the side hatch.

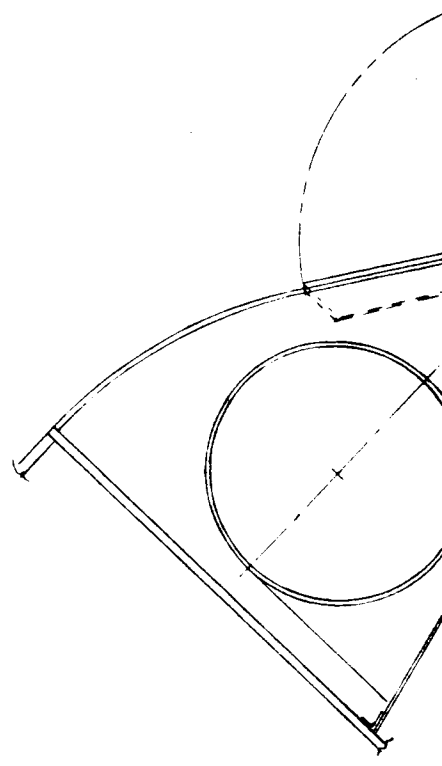
The slightly enlarged launch escape tower will be used for pad and low-altitude abort.

SYSTEM REQUIREMENTS

Reliability, human factors and propulsion requirements for the 5- and 7-man transport vehicles are presented herein. Since the weight of both the 5- and 7-man vehicles exceed the nominal injection capability of the Saturn V, uprating of the Saturn V through modifications of the S-IVB stage is discussed. Reliability decay considerations for mission durations exceeding the nominal 14 days, and general human factors criteria for accommodation of astronauts in cramped quarters are also discussed in this section.

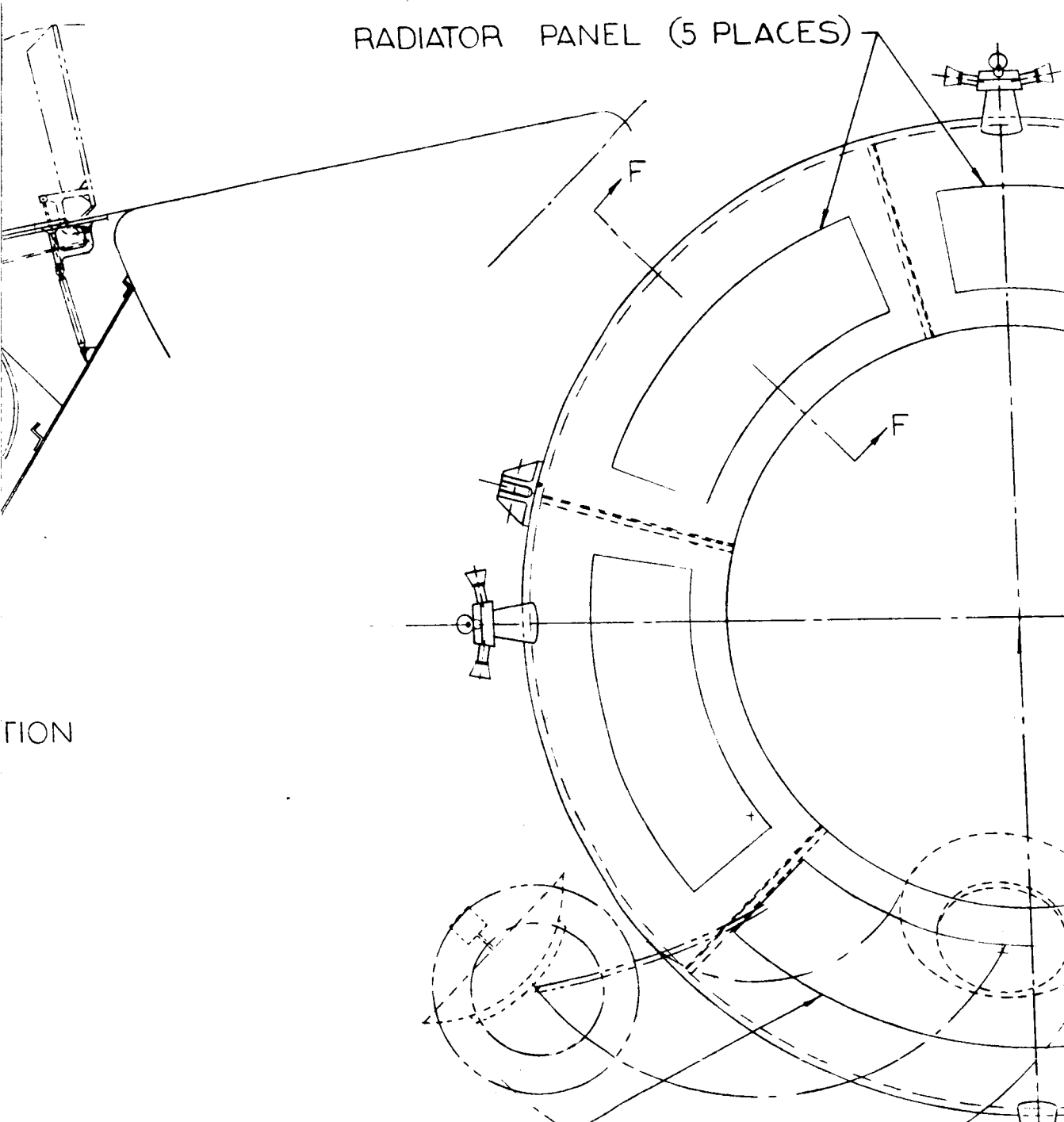
Reliability Considerations for Ferry Spacecraft During Extended Stay on the Moon

The three-man direct lunar landing mission discussed in SID 62-1467 as well as the five- and seven-man lunar base crew ferry vehicles outlined in this report, are based on using the same mission profile; namely, the nominal fourteen-day mission, with travel times of sixty-six to seventy-two hours out- and inbound. The total mission time can be cut without any problems by reducing the stay time on the Moon; however, is some cases, it may be desirable to extend the stay-time on the Moon and, therefore, the total mission duration. A requirement for temporary increase of the lunar base crew complement, or the requirement for an additional stand-by emergency crew evacuation vehicle during critical phases of the base build-up, are the most likely reasons for missions exceeding the nominal fourteen days, with corresponding increase in stay-time on the Moon and exposure to the lunar environment. Any extension of the mission will result in some degrading effect on reliability. The reliability decay that may be experienced in the Apollo II subsystems as a result of standing by on the lunar surface for a period of one week or longer is a function of the environmental and functional stresses which will be



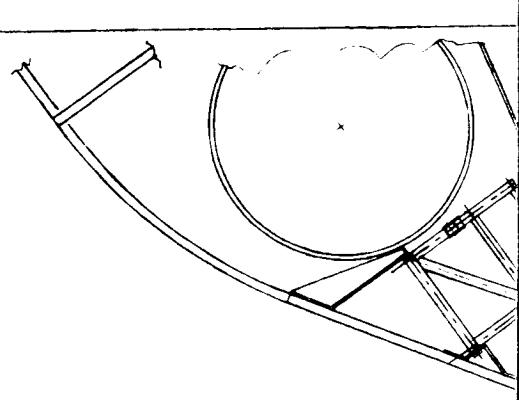
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RADIATOR INSTALLA



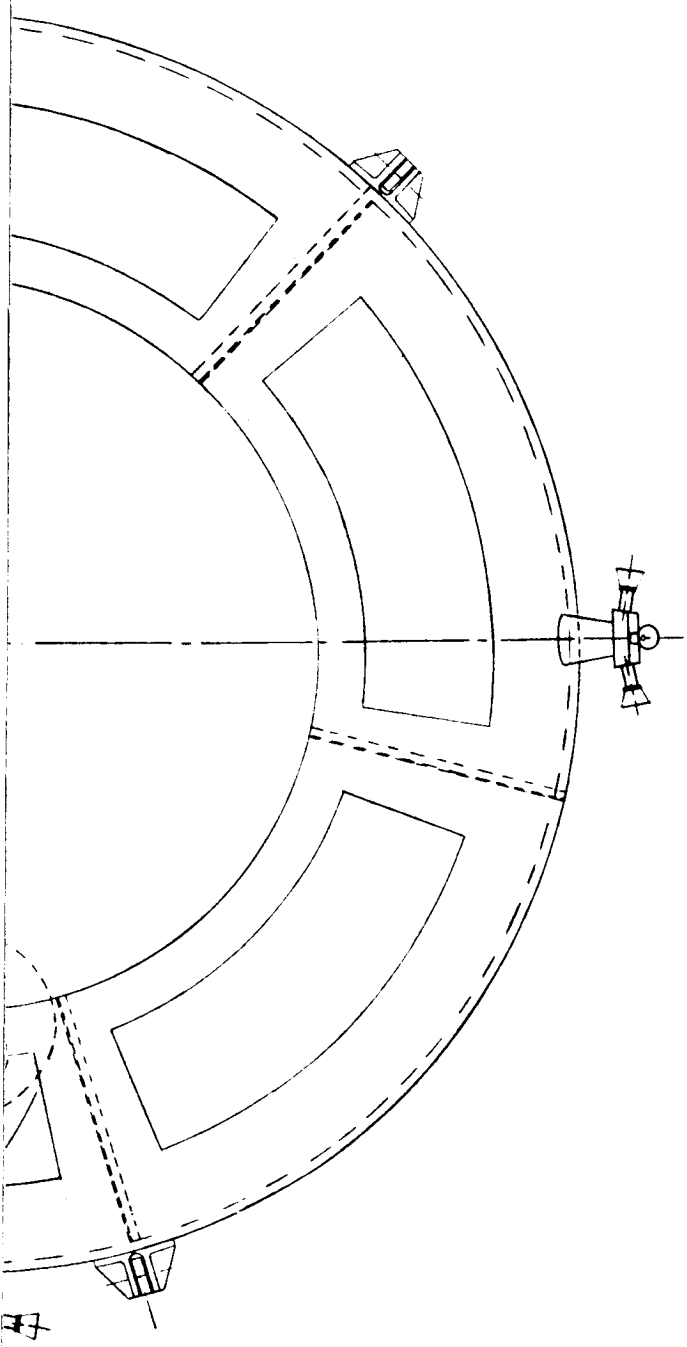
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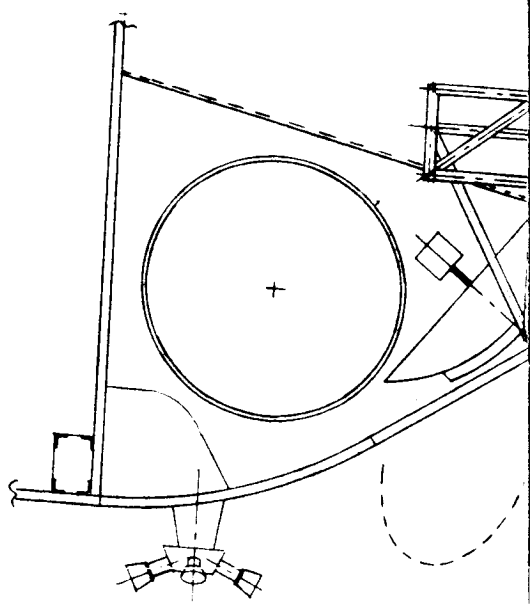
PLAN V



VIEW E-E

LH_2 CONTAINER

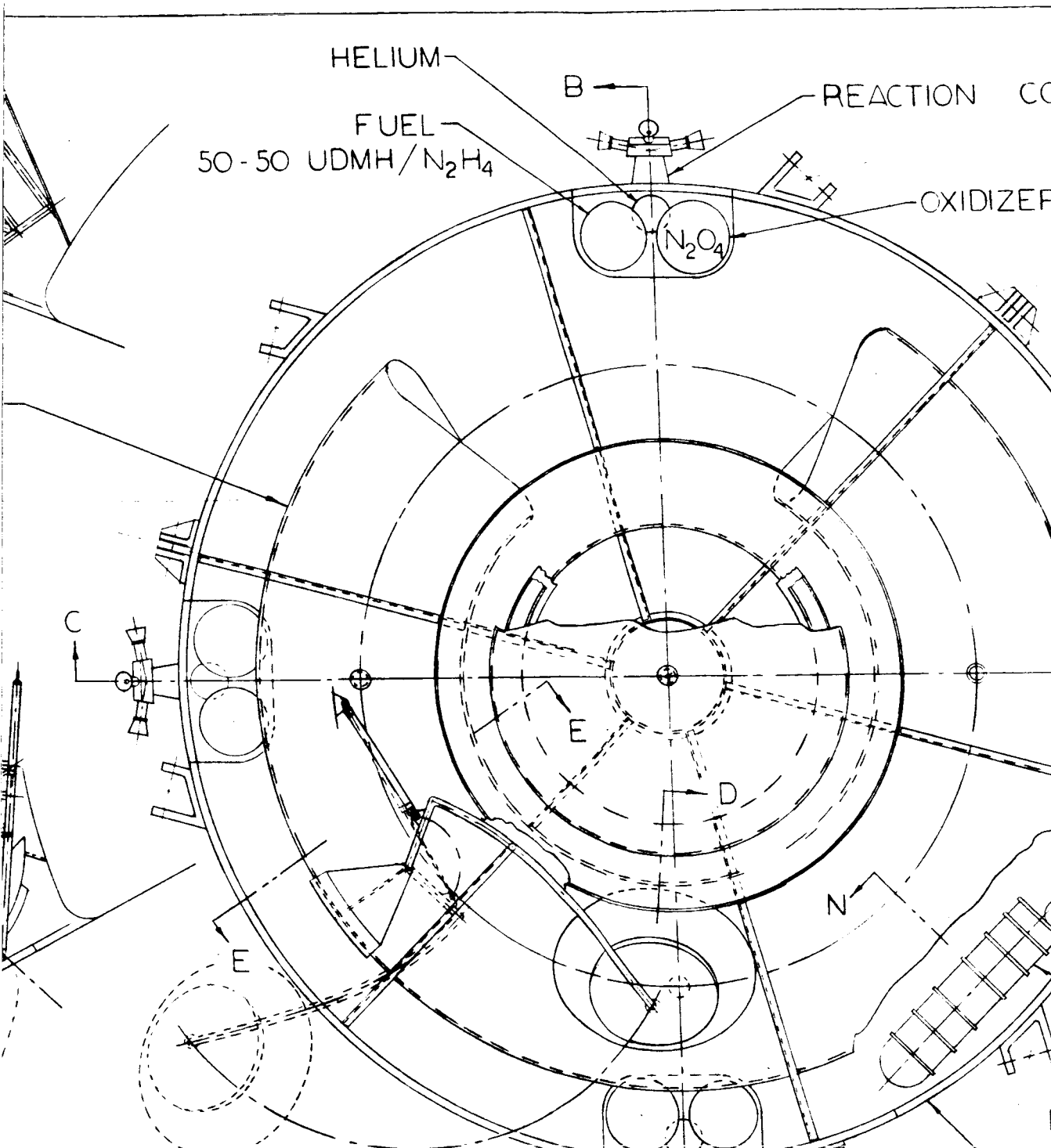




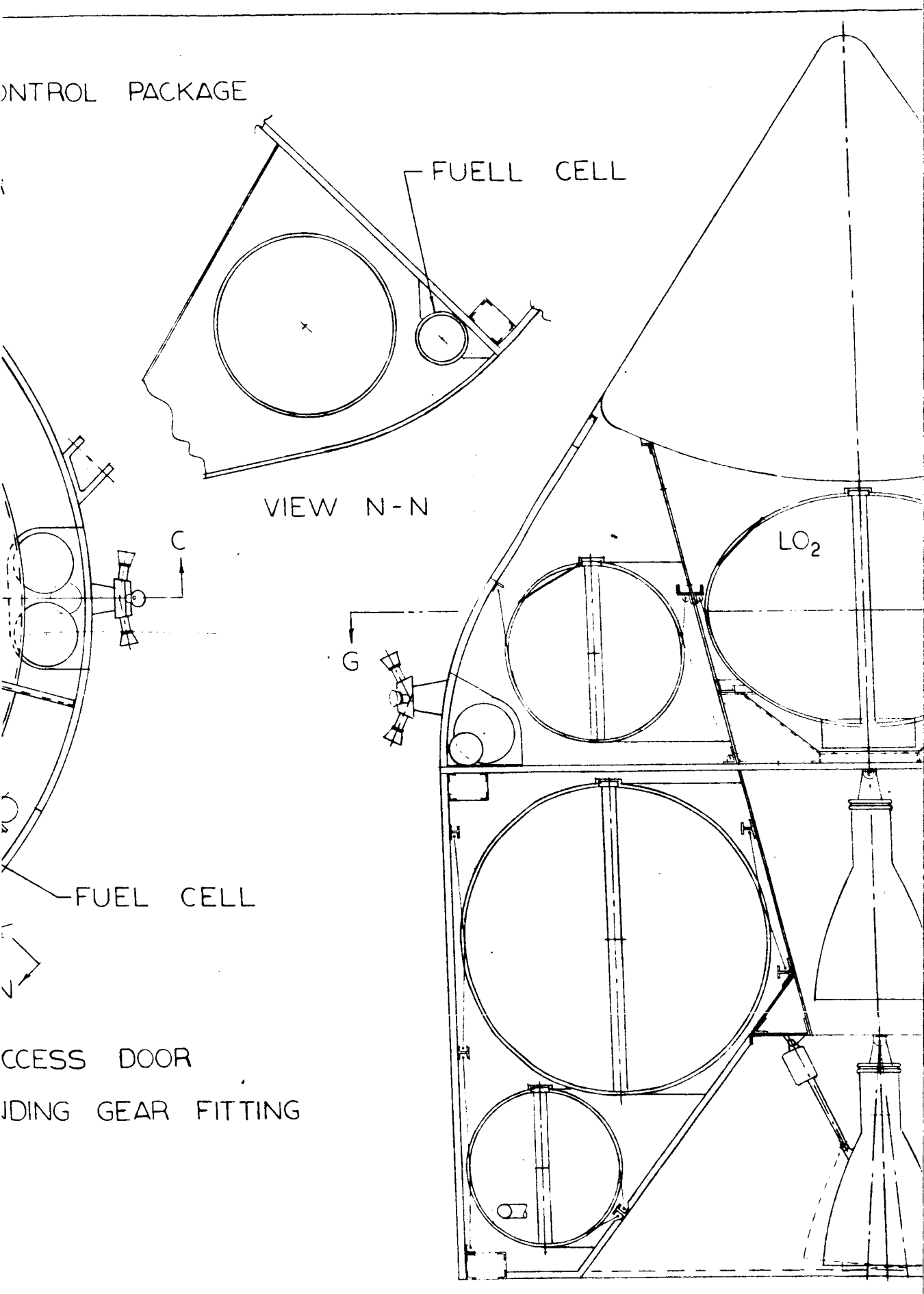
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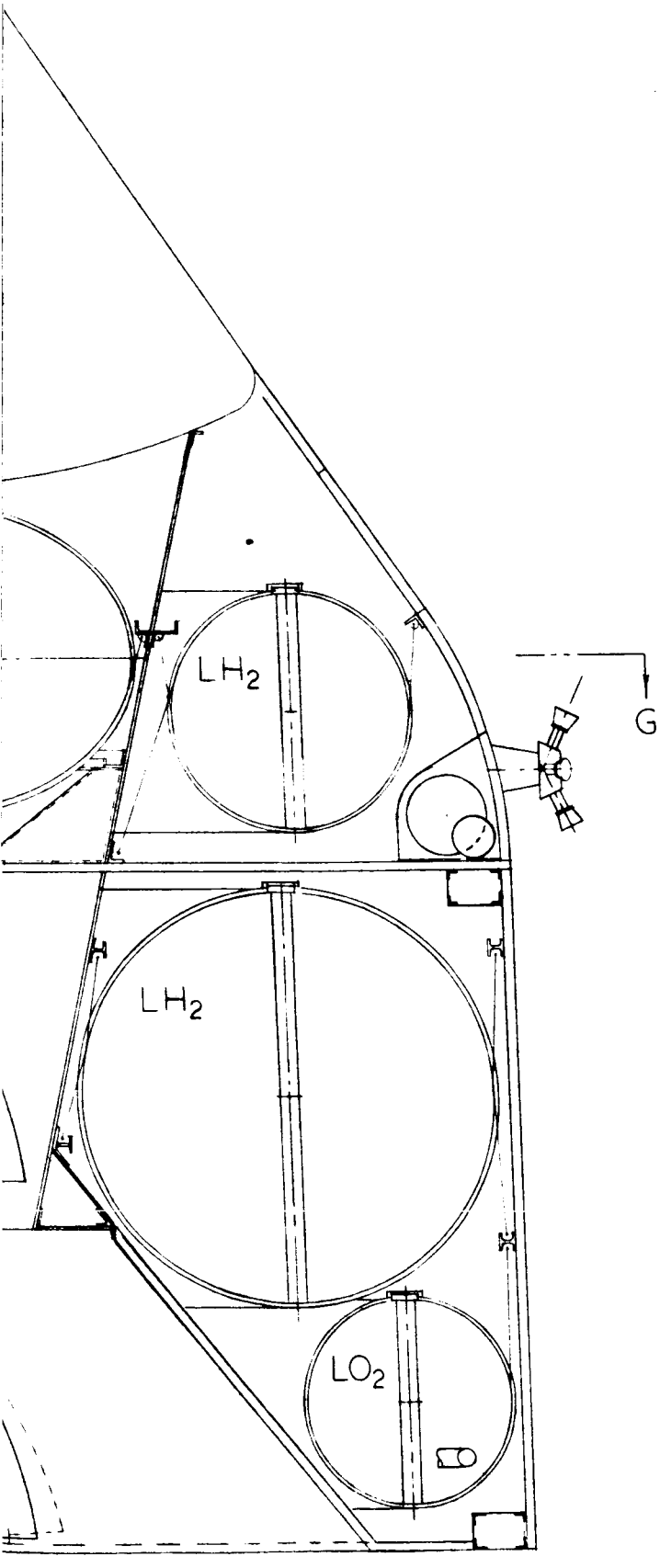
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IEW



PLAN VIEW
LUNAR LAUNCH STAGE


ONTROL PACKAGE
FUELL CELL
VIEW N-N
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G
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FUEL CELL
CCESS DOOR
JDING GEAR FITTING
VIEW



LU

COMMAND MODULE

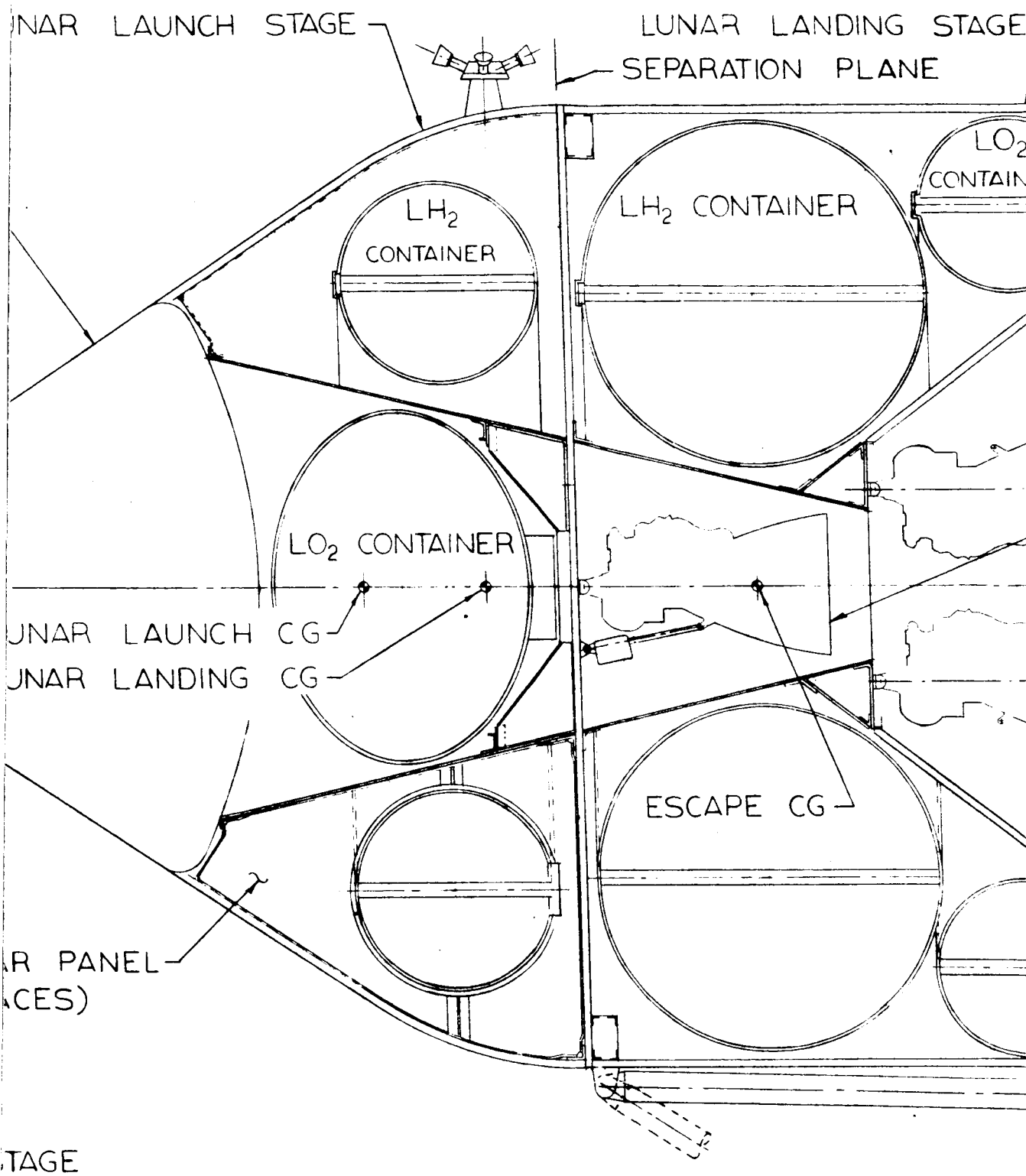
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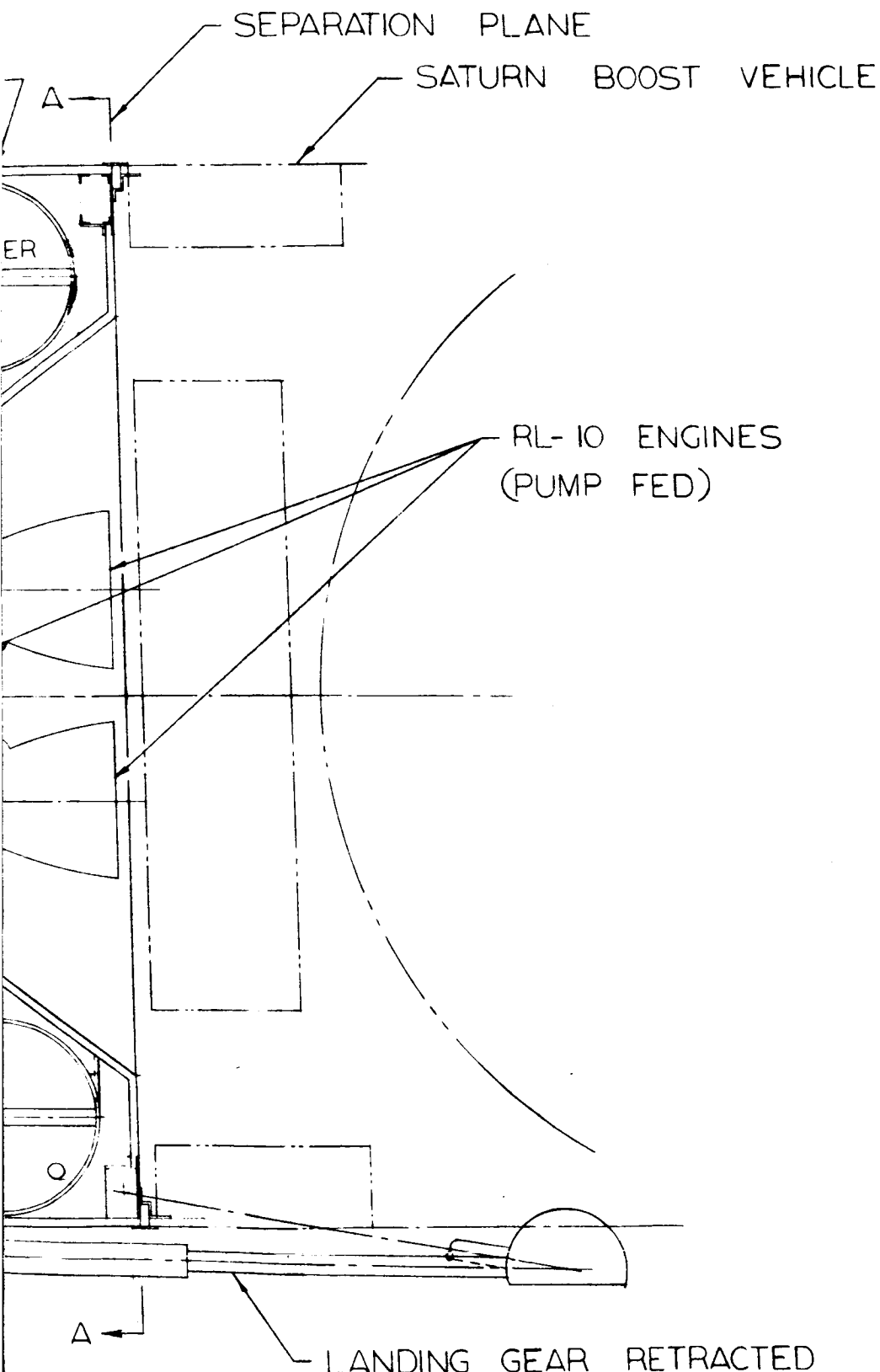
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PROPELLANT CONTAINER
SEE INBOARD PROFILE -
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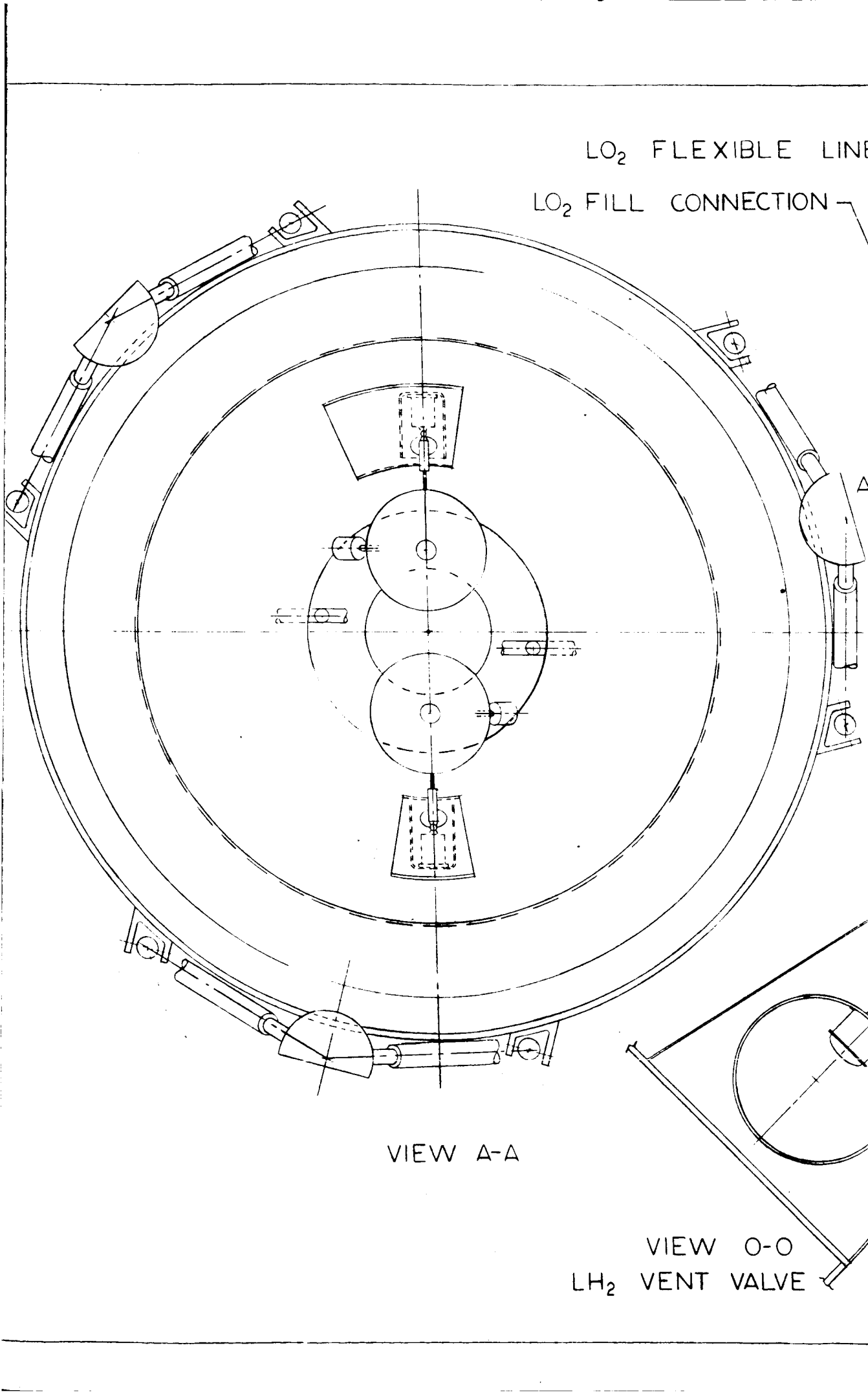
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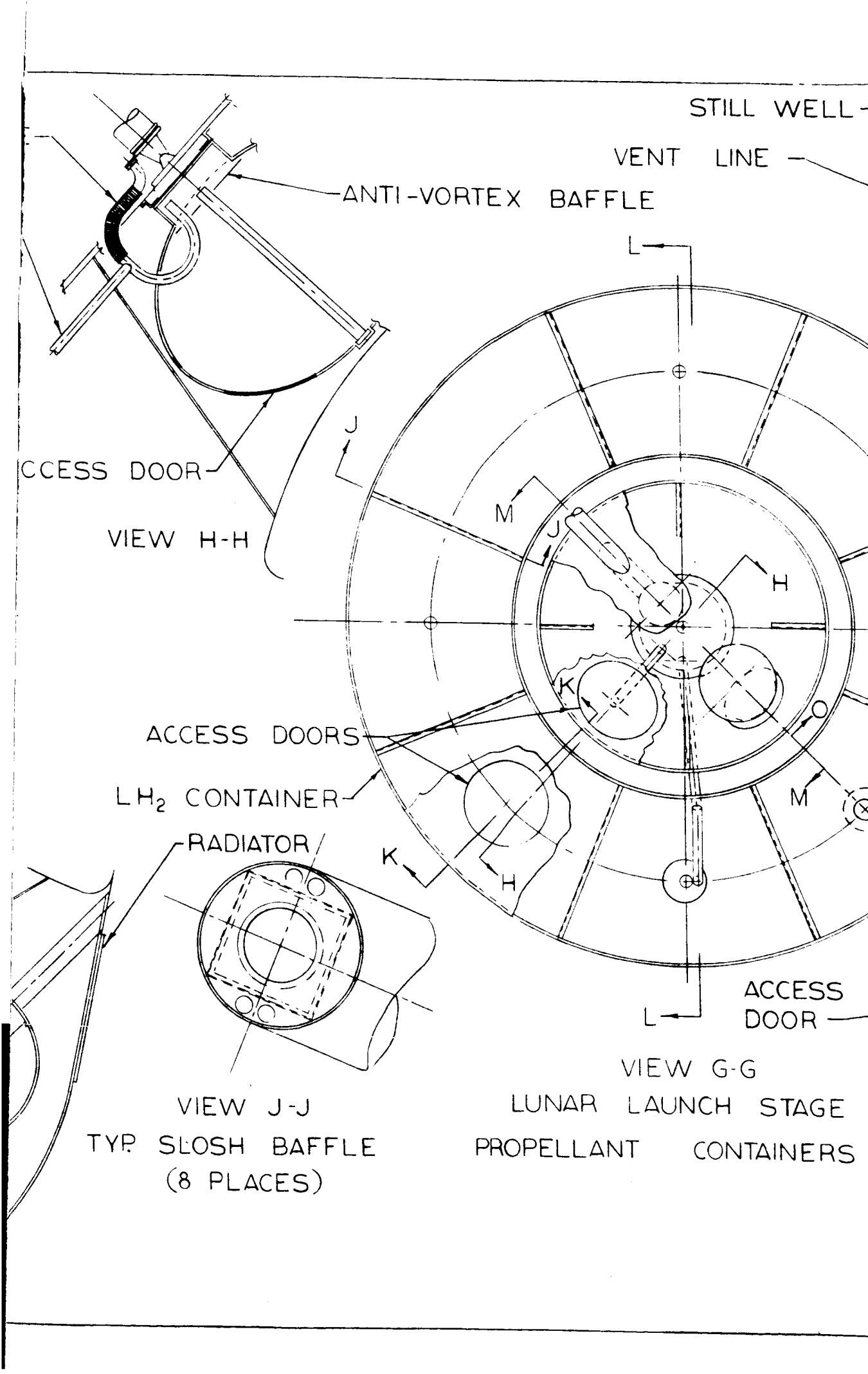


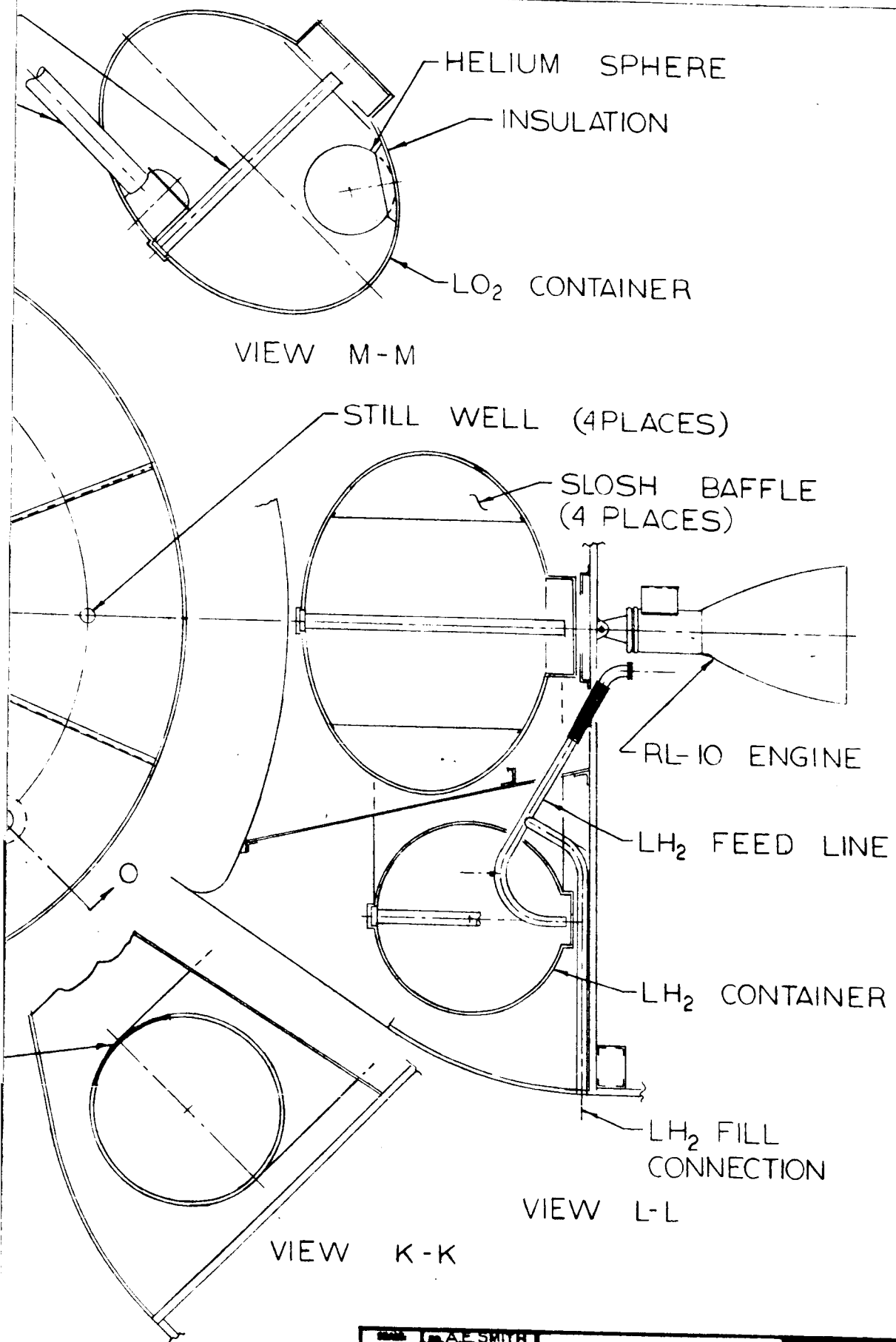
TAGE
DETAILS
- LUNAR

VIEW B-B


SEPARATION PLANE
SATURN BOOST VEHICLE
A
ER
RL-10 ENGINES
(PUMP FED)
Q
A
LANDING GEAR RETRACTED


LO_2 FLEXIBLE LINE
LO_2 FILL CONNECTION
A
VIEW A-A
VIEW O-O
LH_2 VENT VALVE


STILL WELL
VENT LINE
ANTI-VORTEX BAFFLE
CCESS DOOR
VIEW H-H
L
J
M
J
H
K
O
ACCESS DOORS
LH2 CONTAINER
RADIATOR
K
H
M
L
ACCESS DOOR
VIEW J-J
TYP. SLOSH BAFFLE
(8 PLACES)
VIEW G-G
LUNAR LAUNCH STAGE
PROPELLANT CONTAINERS



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INBOARD PROFILE - LUNAR DIRECT MISSION - MANNED VEHICLE		3586-26

encountered. These include radiation, meteorites, temperature, time, and the functional stresses imposed by equipment operation.

Radiation is not considered to be a critical factor inasmuch as human tolerance is much less than equipment tolerance and will be the governing factor in total permissable exposure.

Complete information regarding the meteorite flux on and near the lunar surface is not available. Estimates of meteoroidal impact probabilities have been developed for the 14 day Lunar Apollo mission and other related studies. Based on data derived from these studies, largely hypothetical, the degrading effects on equipment caused by extending the lunar stay do not appear significantly greater than those predicted for the basic 14 day Lunar Apollo mission.

The factor of temperature can be a reliability problem. The equipment housed within the command module will be exposed to the moderate temperatures of a maintained habitable environment and will be well within the temperature values which adversely affect hardware failure rates. Equipment housed within the service module will be exposed to a greater degree to the extreme temperature variations expected on the lunar surface. Precautions along the line of heating or insulation provisions should be considered to avoid deleterious effects of these temperature extremes on seals, wiring insulation, and the expansion and contraction of mechanical parts. Cryogenic storage for extended periods, particularly when ambient temperatures reach high peak values is a problem. For purposes of assessing reliability degradation it is assumed that adequate cryogenic storage for the conditions to be encountered will be provided.

The most critical factors from the reliability aspect in the Apollo II consist of functional stresses and time. The effects of time, including wear-out, are the most critical and will degrade equipment to varying degrees, depending on whether the equipment is quiescent or operating. Figure 29 shows a general relationship for electronic equipment under conditions to be encountered in space or on the lunar surface.

Complex systems failure characteristics, in general, follow the exponential distribution. Therefore, if lunar Apollo subsystem reliabilities are assumed, the degrading effects of lunar storage time on operating systems may be predicted with reasonable accuracy. The collective inherent reliability of the Apollo electronics systems is less than 0.80, redundancy and spares being utilized to attain the required goal. Figure 29 shows the effect of time on the reliability of an operating system. The reliability degradation of a quiescent system would be considerably less.

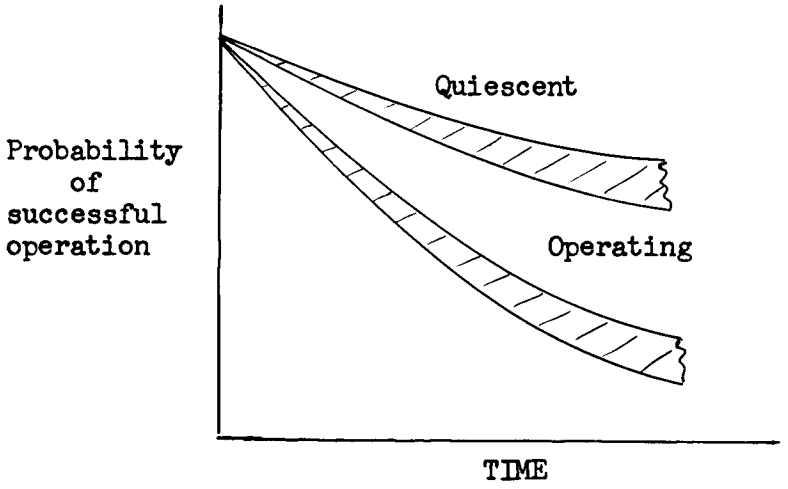


FIGURE 29

To achieve the reliability goals for longer than the basic 14 day mission will require an increasing capability for maintenance and replacement of parts.

In the five- and seven-man vehicles, the life support system will be complicated by the addition of water recovery provisions and molecular sieves for CO_2 removal. Reliability goals for this vital equipment will be achieved by redundancy (stand-by units) and in-flight (or on the Moon) maintenance. The water recovery system is even in normal operation working intermittent and inherently offers the possibility for maintenance.

The redundant features of the fuel cell-electrical power system appear to be sufficient to provide reliable operation for the extended mission time. Isotopic-dynamic power systems, as employed in the seven-man vehicle, will have redundant conversion equipment; they are not suitable for maintenance or repair during the missions.

Precise predictions for Apollo mission success probability involve consideration of full and part time subsystem operation, partial operation of a subsystem as with communications, and quiescent states. Utilization of a computer simulation program such as used in the basic Apollo program would be necessary to develop meaningful predictions which would take these variables

into consideration. The exponential failure distribution does not apply because of the redundant features and maintenance capability. It is probable that the degradation would not be as severe as that expected from the exponential, particularly with a number of quiescent systems. However, to attain approximately the same degree of mission success probability as the basic Apollo, provision must be made for additional redundancy and/or maintenance capability. Attainment of the required goals in this manner appears quite feasible.

HUMAN FACTORS CONSIDERATIONS

Physiological and Psychological Considerations

Confinement Effects

One syndrome in sensory deprivation suggests that man's normal mental state is a function of his perceptual contact with his normal environment. When deprived of these contacts for periods of time, he may find himself seeking to fill the void by reminiscing or occupying his mind with such diversions as counting his pulse, the minutes, or his breathing rate. Eventually, if this fails, disorganized thinking, oppression and depression, increased suggestibility and, in some cases, delusions, hallucinations, and panic follow. Much experimental research in sensory deprivation has been done with both individuals and with groups. The experiments, however, were designed to produce an extreme case of sensory deprivation, under conditions that man is not likely to encounter either in space or on earth.

The most dramatic of these experiments were made at McGill University by D. C. Hebb and his associates. They produced violent hallucinations and delusions in test subjects who were deprived of sensory stimuli and for periods of time ranging between 24 and 72 hours. The subjects wore translucent goggles, gloves, and cardboard cylinders over their arms and lay on beds inside a partially soundproofed room. After two to three days many of the subjects could tolerate the sensory deprivation no longer and had to be released. Those who remained had various complaints ranging from nausea and headaches, fuzzy vision, to delusions that their bodies were floating and that heads were detached from bodies. Several suffered from violent hallucinations.

There are certain factors in such experiments that make them in some ways inconclusive. For example, it is extremely hard to duplicate the isolation of space on earth since the subjects know full well that they are not in space and that they can end the experiment at any time. Also, the mechanics of the test tend to lower realism.

In summary, the full significance of sensory deprivation on the astronaut is hard to visualize. It is probable that the astronaut will never be faced with a condition of absolute sensory deprivation since his tasks and movements within the vehicle (although very limited on the 5 man Apollo) will always afford some sensory stimuli. However, the seriousness of the situation

must not be overlooked. Therefore, it seems logical to perform a test, as do the Soviets, in selecting prospective astronauts (or in our case Lunar Base personnel) to see how well they can withstand an environment of limited sensory inputs.

Radiation Hazards

Due to the relatively abbreviated nature of the 5 and 7 man Apollo mission, the danger of exposure to large cumulative radiation doses is greatly reduced. The problems remaining are those of short term exposures to such phenomena as solar flares, normal emissions and secondary particles of collision. Shielding methods, utilizing the spacecraft's hull or the astronaut's spacesuit, and chemical shielding agents can be utilized to reduce the hazard to tolerable limits. No change from the 3 man Apollo mission requirements is contemplated. For radiation dosage limitations, see Table 6.

Training

Because of the crowded conditions and short mission duration, any onboard training period is unnecessary. Training to perform the functions required by crewmen on the flight will present some special problems. Simulated missions should be run to accustom the crew to the special confinement and general living problems of spending 72 hours in the Apollo capsule. Special training should be provided in a "Direct Lunar Landing" mission simulator.

Motivation

Motivation (Reference 1) is known to be a highly significant factor in achievement of any task. Under stressful conditions which exist in a mission such as this, motivation will be an even more significant factor.

One must assume, of course, that professional astronauts are highly motivated men. It would seem that any other technical crew member taken to a lunar base in a passenger status would also be highly motivated even though in all respects his outlook might not parallel that of the "real" astronauts. Training programs should certainly be oriented toward making mission success in every phase of a personal goal of each crewman.

Some motivations which might prompt men to volunteer for a mission such as this are:

1. Drive for prestige
2. Drive for escape from an unpleasant situation
3. Feelings of inadequacy which prompt one to "prove himself".
4. Desire to acquire money
5. Intellectual curiosity

TABLE 6 - RADIATION EFFECTS

DOSAGE	EFFECT	COMMENTS
300 Millirems or less per week	None	
300 millirems per 13 weeks	None	
5000 millirems per year	None	
0-50 rems single exposure	Minor blood changes	No noticeable effect
80-120 rems single exposure	Vomiting and nausea in 5 to 10% of personnel exposed	Some fatigue, but no lasting effect
130-170 rems single exposure	Vomiting and nausea in 25% of exposed personnel	No deaths expected
180-220 rems single exposure	Vomiting and nausea in 50% of exposed personnel	No deaths expected
270-330 rems single exposure	Vomiting and nausea in all personnel. Other symptoms of radiation sickness 20% death within 2-6 weeks	Survivors convalescent for about 3 months
400-500 rems single exposure	Vomiting and nausea in all personnel. Other symptoms of radiation sickness 50% deaths within 1 month	Survivors convalescent for about 6 months
550-750 rems single exposure	Vomiting and nausea in all personnel. Other symptoms of radiation sickness up to 100% deaths	Few survivors convalescent for about 6 months
Over 1000 rems single exposure	All symptoms of radiation sickness 100% deaths within relatively short period	

Obviously, any or several of these factors could well apply to any astronaut or lunar base technician. Similarly, any of these motivations could be sufficient to produce an efficient astronaut. One cannot single out "good" and "bad" motivations per se. It seems that the most significant factor is the intellectual and emotional makeup of the man in respect to his motivations. This would dictate his reactions to stressful or dangerous situations. Obviously the man who can be expected to react intellectually rather than emotionally in these situations stands a much better chance of surviving.

Work/Rest Cycles

Although much time has been spent investigating work/rest cycles, there is still considerable doubt regarding the effects of various work/rest schedules on performance. Work/rest cycles involve many variables such as the ratio of work to rest; the length of sleep and work periods; and the number, length and spacings of rest periods. These, along with such experimental variables as task characteristics, degree of confinement and environmental stress, individual differences, and uncontrolled sources of variability, make it difficult to specify optimum cycles. As a consequence, a work/rest cycle must be compatible with the particular work load, task characteristics, crew characteristics, and other variables specific to a given system. As applied to evaluation of work/rest cycles in the two configurations, they must be tested against the alternative conditions specific to each.

Conventional Vs. Non-Conventional Cycles

Although man is accustomed to an eight-hour sleep period, there is some experimental evidence to indicate that consideration can be given to use of four-hour sleep periods as well. The experimental evidence supporting the shorter sleep periods is meager and sleep duration effects are confounded with the effects of other variables such as tasks and the time of day of the sleep period within the studies available.

Adams and Childs* found that although there were no significant performance differences between 2-hour, 4-hour, 6-hour, and 8-hour work/rest schedules, there were differences in adjustment to and satisfaction with the schedules and the confinement period. In all measures of satisfaction with the mission, an inverse relationship was indicated between the number of favorable responses and the length of the work and rest periods. Only one of the sixteen subjects preferred the eight-hour work/rest cycle.

Recent simulation studies conducted at Langley Research Center also cast some doubt upon the advisability of using an eight-hour sleep period. A series of three missions were simulated with the first of these using an 8-hour sleep period. It was found that the pilots were unable to sleep the full eight hours and became very tired after two nights of very little sleep. The average uninterrupted sleep was four hours with the remaining four hours spent catnapping or trying to sleep. Because of this unsatisfactory sleeping

*Ref. 2

[redacted]

situation, a second duty cycle was adapted for Mission II. This second duty cycle consisted of splitting the eight-hour sleep period into two 4-hour sleep periods. The pilots were in unanimous agreement that the second duty cycle was very good and could be used over long periods of time. The pilots were able to obtain more sleep and they did not experience the degree of boredom associated with duty cycle I.

One criticism of these studies is that the crew members, when sleeping the eight-hour period, did not sleep the same eight hours every day. There is other evidence that indicates this to be undesirable in itself; therefore, it cannot be concluded that the length of the eight-hour sleep period was the source of dissatisfaction.

For this study, a 4-hours on and 4-off schedule for pilot personnel is recommended.

Personal Hygiene

Personal hygiene would become a problem in such a confined environment for a prolonged trip. For a two to three day trip, however, it will be somewhat less of a problem.

Fecal material may not be a problem on such a short flight. A fecal cannister and some storage arrangements must be allowed for, but with a low residue diet may be possible to avoid defecation for a 3-day mission. The present 3-man Apollo fecal cannister can be used with the center couch folded back out of the way. Urine will be disposed in the same manner in the five as the present 3 man Apollo system except for adding provisions for taking care of the additional personnel; a water recovery system is installed in the seven man vehicle.

Bodily cleansing should be allowed for at least once on the trip. Wet-pack type chemical pads will be used for this. The center lower couch area would be the best location for this duty. Body odors and perspiration will be removed by the Apollo environmental control system through suit ventilation. Oral hygiene will be accomplished by means of dentrifice tablets satisfactorily.

Change of clothing should not be necessary during the trans-lunar flight. It would be advisable for the flight crew before returning to earth.

Personal Equipment

Personal articles and clothing of crew members must be transported to the lunar base. Flight crew members who will return the vehicle to earth will require only a change of underclothing. Space restrictions will limit the quantity of clothing and articles that can be taken, but allowance should be made for several changes of undergarments and socks, personal hygiene gear

such as razor, toothbrush, comb, etc., and a few personal items such as a photograph, pocket chess set, etc.

Bio-Medical Instrumentation

Taking Apollo program development time into consideration and advances in state-of-the-art knowledge concerned with the physiological well being of astronauts involved in earth orbital and lunar missions, second generation Apollo spacecraft operations of such short duration as planned for the 5 and 7 man Apollo should not require biomedical instrumentation of its occupants. However, if it is contemplated that this same logistics vehicle be used for extended missions, then provisions for biomedical instrumentation should be included if not used on some missions.

Biomedical Specialists are generally of the opinion that the astronauts can provide at regularly scheduled intervals the basic medical information required via voice communications. This would require a minor amount of training for each astronaut and/or spacecraft passenger in methods of monitoring and recording personal medical data. Also the specialist can directly interrogate each crewman if necessary.

Psychopharmacology and Hypnosis

Psychopharmacology (Reference 1) is a relatively new branch of science that investigates the way certain drugs produce psychological effects. Much work has been done in this field, though little of it with application to space travel. Perhaps the most familiar examples are the tranquilizer in its many forms and the drugs derived from mushrooms that produce very dramatic psychological effects. But there are other drugs that can alter the mental state of man. These may come in handy on long space voyages or during the first trying days of colonization on either the Moon or some planet. Following the excitement of the initial exploration and once the space explorer's task becomes fixed and routine, such drugs could be used to produce a euphoria in order to insure the success of the mission.

The use of drugs for prophylactic purposes among soldiers during war or stress is an accepted practice. During World War II and the Korean War, malaria suppressants were used to keep down debilitating symptoms and enable soldiers to perform their duties. Modern jet pilots often take sedatives before, and stimulants during, long flights.

Analeptic or restorative drugs have been used in controlled tests on subjects at the U.S. Air Force School of Aerospace Medicine. Results showed that certain of these drugs have a potential for space travel and produce no incapacitating side effects. More important, the investigations show that these drugs produce no sudden letdown after they wear off, a reaction that could endanger the outcome of a space mission.

One difficulty associated with the use of these drugs is that there is no way of knowing what their effects will be upon an astronaut in space conditions. Another factor to be considered is the problem of addiction after long-term use.

Further advances in psychopharmacology include drugs which make an individual more susceptible to hypnosis. As these drugs become reliable, it is possible that the psychopharmacological and hypnotical approaches to problems of extended space travel may be simplified. Hypnosis, a legitimate and useful tool of therapeutic medicine, may also have applications in space medicine. At least eight such applications are already suggested:

(1) In selecting candidates for astronautical training.
(2) In creating realism within space simulators or training devices.
(3) In planting ideas within the minds of astronauts that will reduce fear and anxiety involving the unknowns of long space flights.
(4) In reducing boredom by compressing off-duty time and creating the illusion of stimuli to occupy time.
(5) In helping focus attention on specific tasks that will be performed under stress.
(6) In reducing the metabolic rate and therefore the oxygen and food requirements of space travelers.
(7) In helping the astronaut maintain an uncomfortable physical position over long periods of time.
(8) In training astronauts to induce self-hypnosis in order that they may put themselves to sleep and awaken on schedule refreshened and alert. (As Major Titov did in his orbital flight of August 6, 1961.)

The usefulness of hypnosis as an aid in overcoming some of the stresses of space flight lies in the fact that a person under hypnosis can endure a great deal of physical stress without reactions that might endanger the success of a mission.

References

1. Ordway, Gardner, and Sharpe, "Basic Astronautics," 1962

2. Adams, Oscars, and Chiles, W. Dean, WADD TR-60-248, "Human Performance as a Function of the Work/Rest Cycle," March 1960

FIVE-MAN APOLLO II

Human Factors

The accommodation of as many astronauts as possible in the Apollo II command module presents a variety of problems. The human aspects of four-, five-, and six-man configurations are discussed in this section.

Number of Crew and Placement

There are several factors that are common for the direct lunar landing Apollo for different configurations of crew and passengers that are under consideration in this study. These factors are:

1) The requirement for additional controls and displays for the lunar landing.

2) Internal volume available for additional couches.

3) Problems of ingress and egress with additional couches. The six-man configuration is not acceptable when considering crew safety in egress from the capsule under emergency conditions. Also, the guidance and navigation equipment cannot be operated because there is not enough room to get to the lower equipment bay.

   Placement of the couches and shock attenuation equipment will be difficult for all configurations. The six-man configuration is out because of work space limitations and the four-man configuration does not provide enough additional space to outweigh the advantages of taking the additional man as in the five-man configuration.

   Although space is at a premium, it is felt that all necessary functions can be accomplished safely and within reasonable time limits by accepting the five-man configuration. Placing the second pilot in the right position rather than in the center location would give him access to the communication and environmental control equipment and provide him with exterior visual reference through the right-hand forward flight hatch.

   Using this configuration would require little modification of existing display panels, except for the addition of the necessary displays and controls for lunar orbit retro, lunar descent, and lunar launch.

Control and Display Arrangements

Actual flight control sequences are of short duration and only required during launch, injection from one orbit to another, during lunar descent and launch, and, finally, during the critical Earth re-entry. Present Apollo flight displays and controls are concentrated near the left seat and should

[redacted]

remain this way as nearly as possible. Should the pilot commander become incapacitated for any reason, he could be removed and placed in one of the other couches in either the four- or five-man configuration but not in the six-man configuration. Since the Apollo spacecraft does not require constant flight control as does an airplane, there should be ample time to shift personnel prior to the critical flight events. Shifting personnel in an emergency because of incapacitation is a calculated risk that should be considered at this time, especially when taking into account the cost of modifications and the weight penalties of control and display duplications.

Maintenance

Performing maintenance functions in the four-man configuration will be difficult at best because of space restrictions. In the five-man configuration, it will be extremely difficult; and, in the six-man configuration, it will be, for all practical purposes, impossible. There just isn't room to move.

Physiological and Psychological Problems

The greatest psychological and physiological hazard to be encountered in the direct lunar landing Apollo is the extreme restriction of physical movement imposed on all personnel in the spacecraft. Space and time requirements for donning the space suit are prohibitive, making it necessary to wear the space suit at all times during the mission. Except for emergency use, the space suit need only be ventilated to maintain normal comfort level.

Confinement stress should not be too serious a problem for well-trained and highly-motivated personnel. However, all personnel must be thoroughly screened for claustrophobic tendencies because of spacecraft limitations.

Conclusions and Recommendations

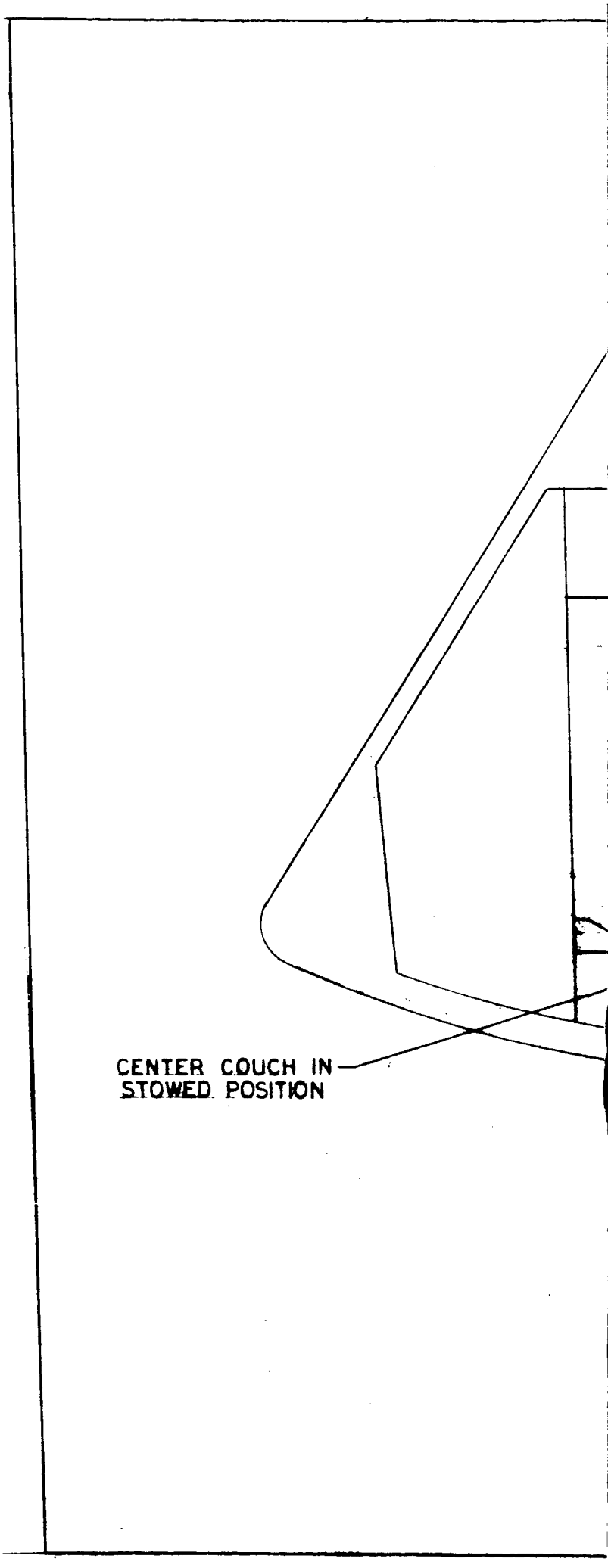
After consideration of the three concepts and the various human factors concerned with each, the five-man concept is recommended. It will provide the most reasonable approach from the standpoint of modifications to the Apollo and in delivering a crew that is physically and mentally capable of performing their assigned duties on arrival at their destination.

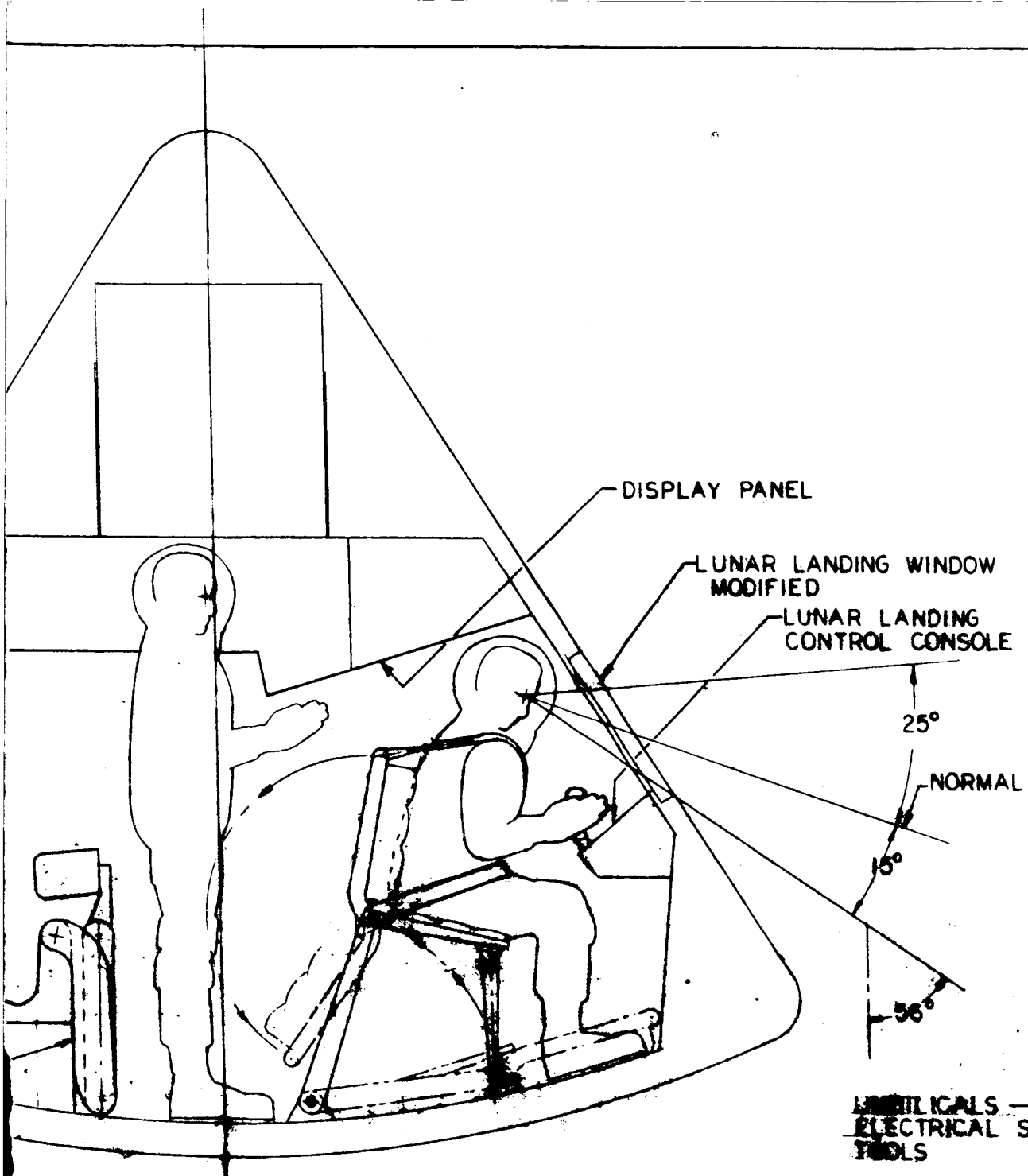
Preliminary Design

The five-man Apollo II Command Module (CM) has the same shape, basic design, and geometrical configuration as the present lunar Apollo CM. In its detail design, however, it is more closely related to the three-man Apollo II vehicle outlined in SID 62-1467; advanced technology, systems, equipment, and fabrication methods are employed. A drawing of the five-man CM is shown in Figure 31.

The primary modifications from the present Apollo are as follows:

1) In the outer structural shell - replacement of the steel honey-


CENTER COUCH IN
STOWED POSITION

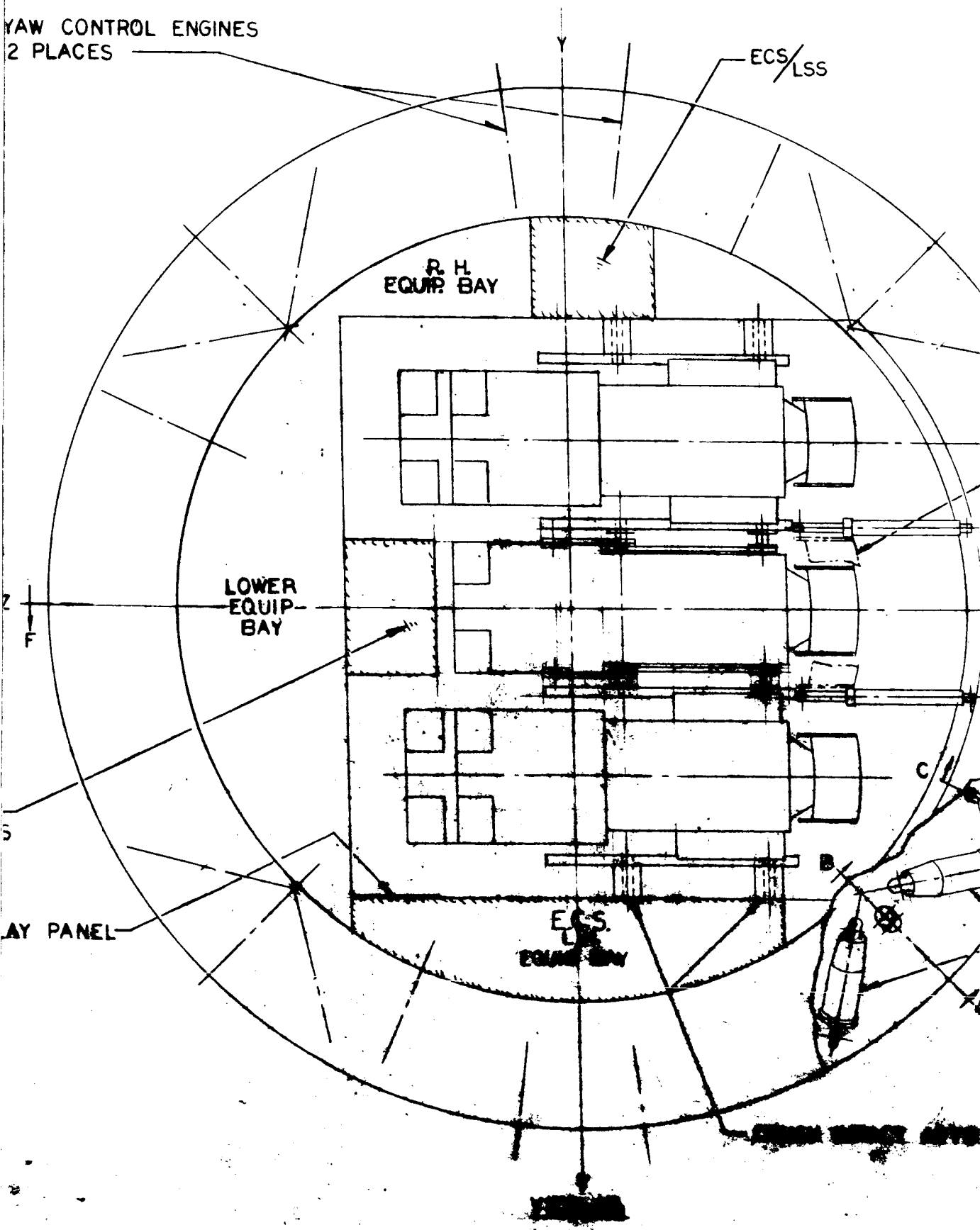


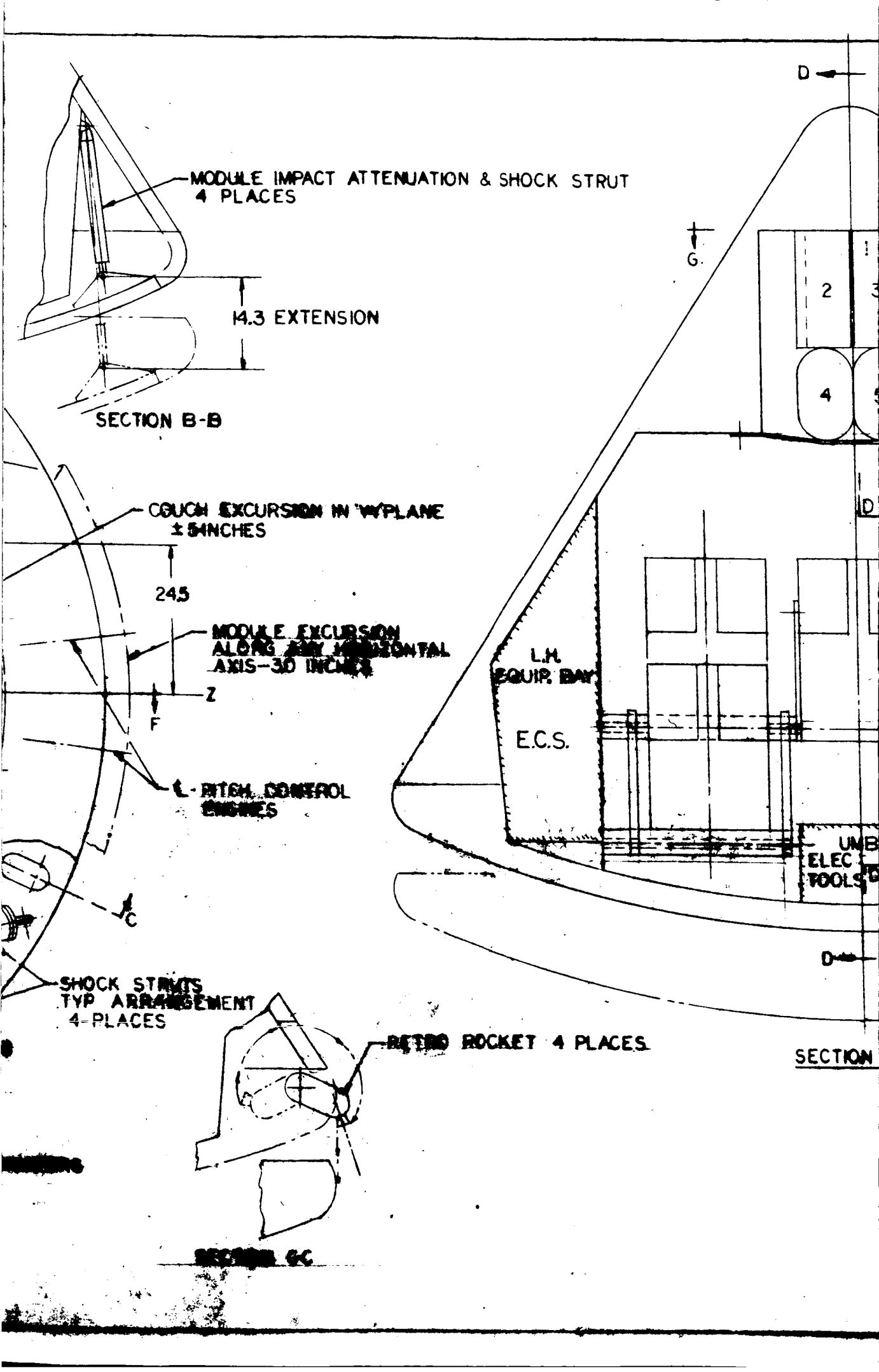
ELECTRICAL SPARE

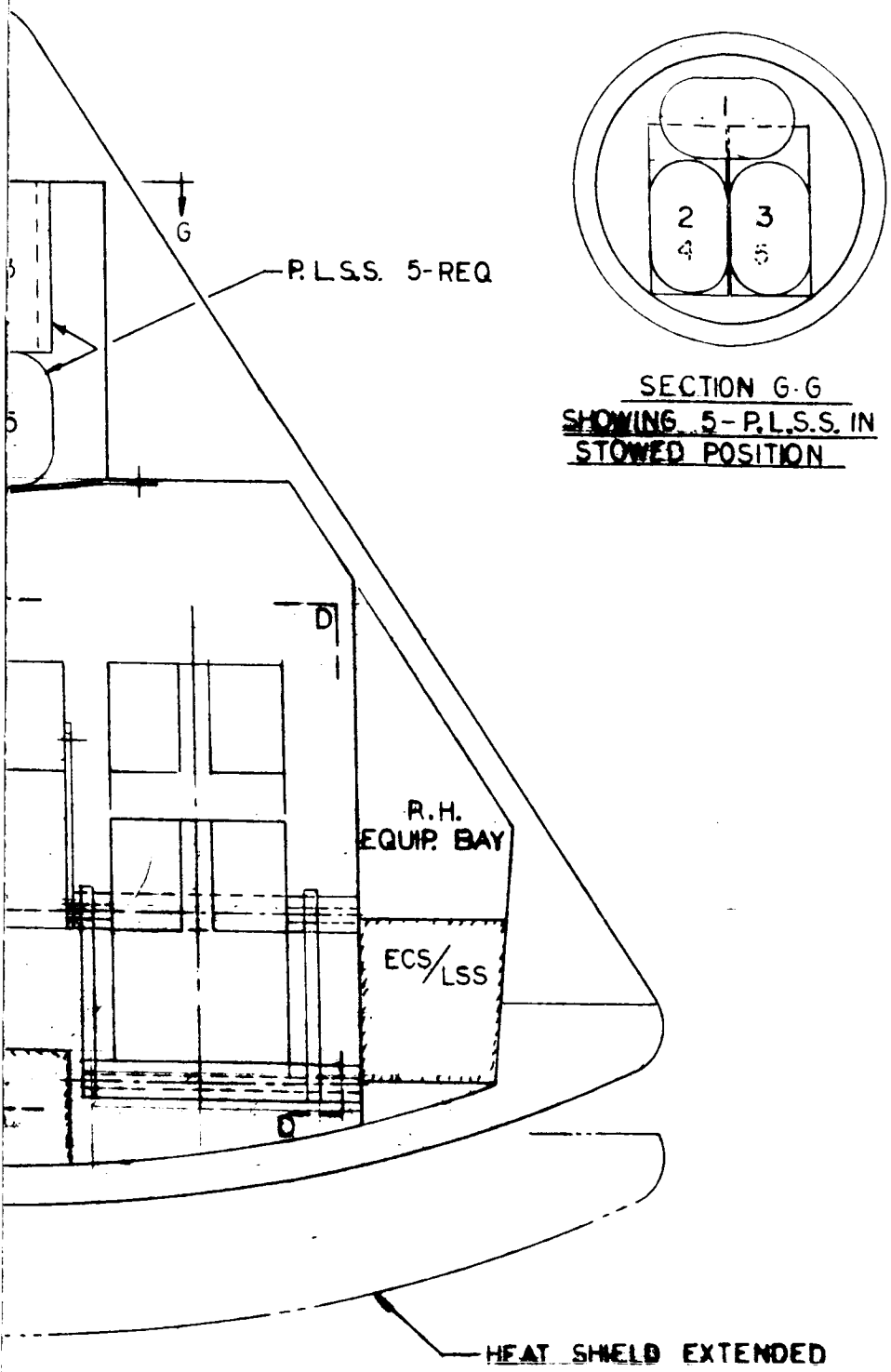
TOOLS

VIEW SHOWING LUNAR LANDING CONTROL SEAT IN POSITION

ECS DISPL


YAW CONTROL ENGINES
2 PLACES
ECS/LSS
Y
R. H.
EQUIP. BAY
LOWER
EQUIP-
BAY
Z
F
C
B
AY PANEL

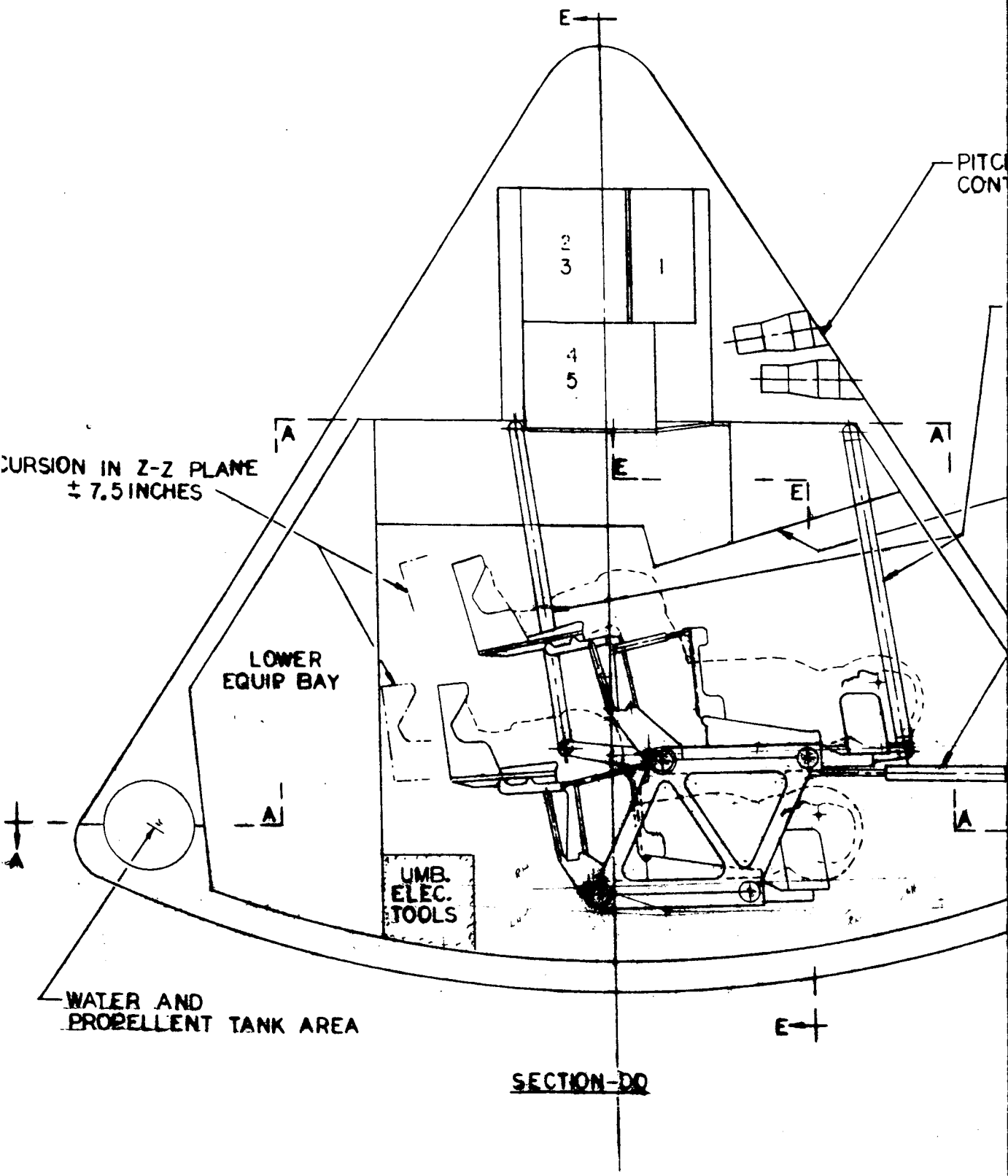

MODULE IMPACT ATTENUATION & SHOCK STRUT
4 PLACES
14.3 EXTENSION
SECTION B-B
COUCH EXCURSION IN W PLANE
±5 INCHES
24.5
MODULE EXCURSION
ALONG ANY HORIZONTAL
AXIS-30 INCHES
Z
F
PITCH CONTROL
ENGINES
C
SHOCK STRUTS
TYP ARRANGEMENT
4-PLACES
RETRO ROCKET 4 PLACES
SECTION C-C
D
G
2
4
L.H.
EQUIP. BAY
E.C.S.
UMB
ELEC
TOOLS
SECTION



E-E

SECTION G-G
SHOWING 5-P.L.S.S. IN
STOWED POSITION

COUCH EX


E
PITC
CONT
2
3
1
4
5
A
A
E
E
A
A
CURSION IN Z-Z PLANE
± 7.5 INCHES
LOWER
EQUIP BAY
A
A
UMB.
ELEC.
TOOLS
WATER AND
PROPELLENT TANK AREA
E

SECTION-DD

H REACTION
TROL ENGINES

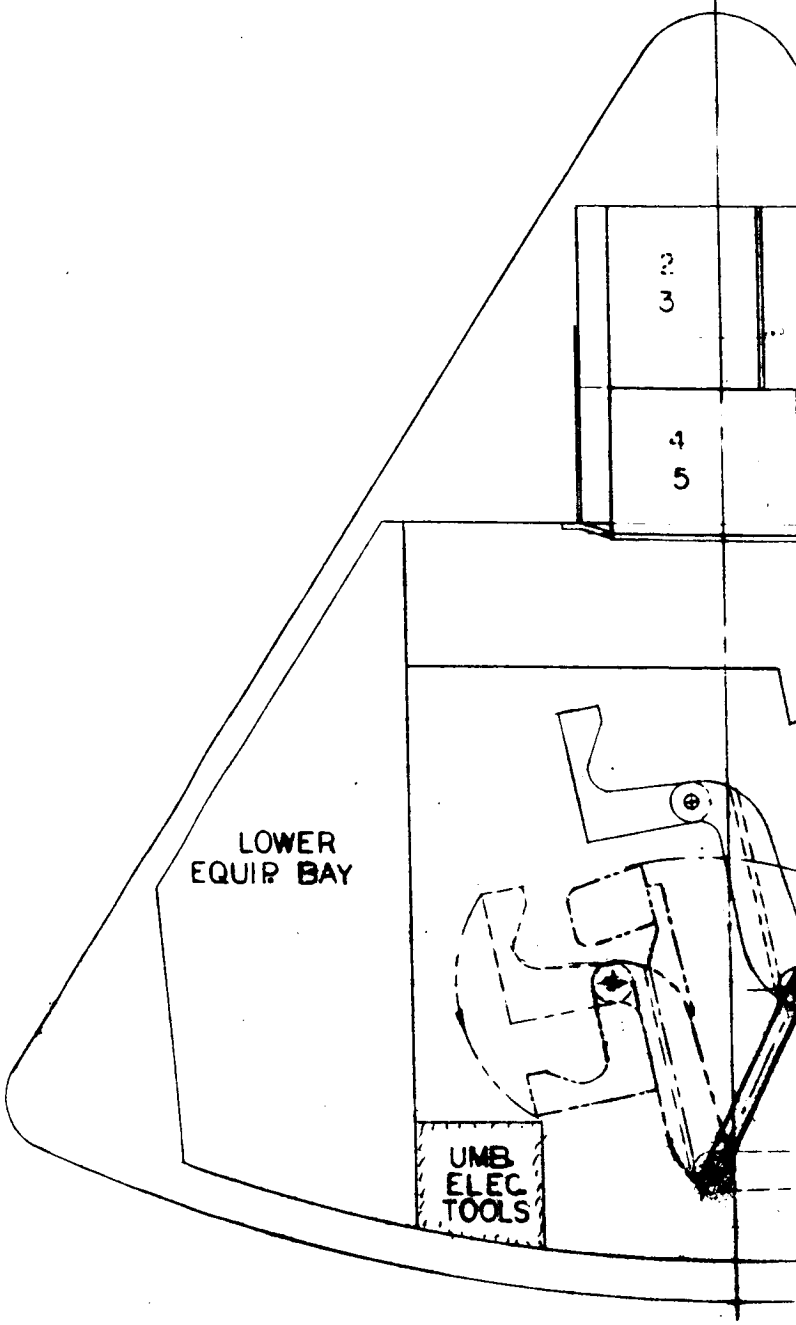
RIGID COUCH SUPPORT LINKS

DISPLAY PANEL

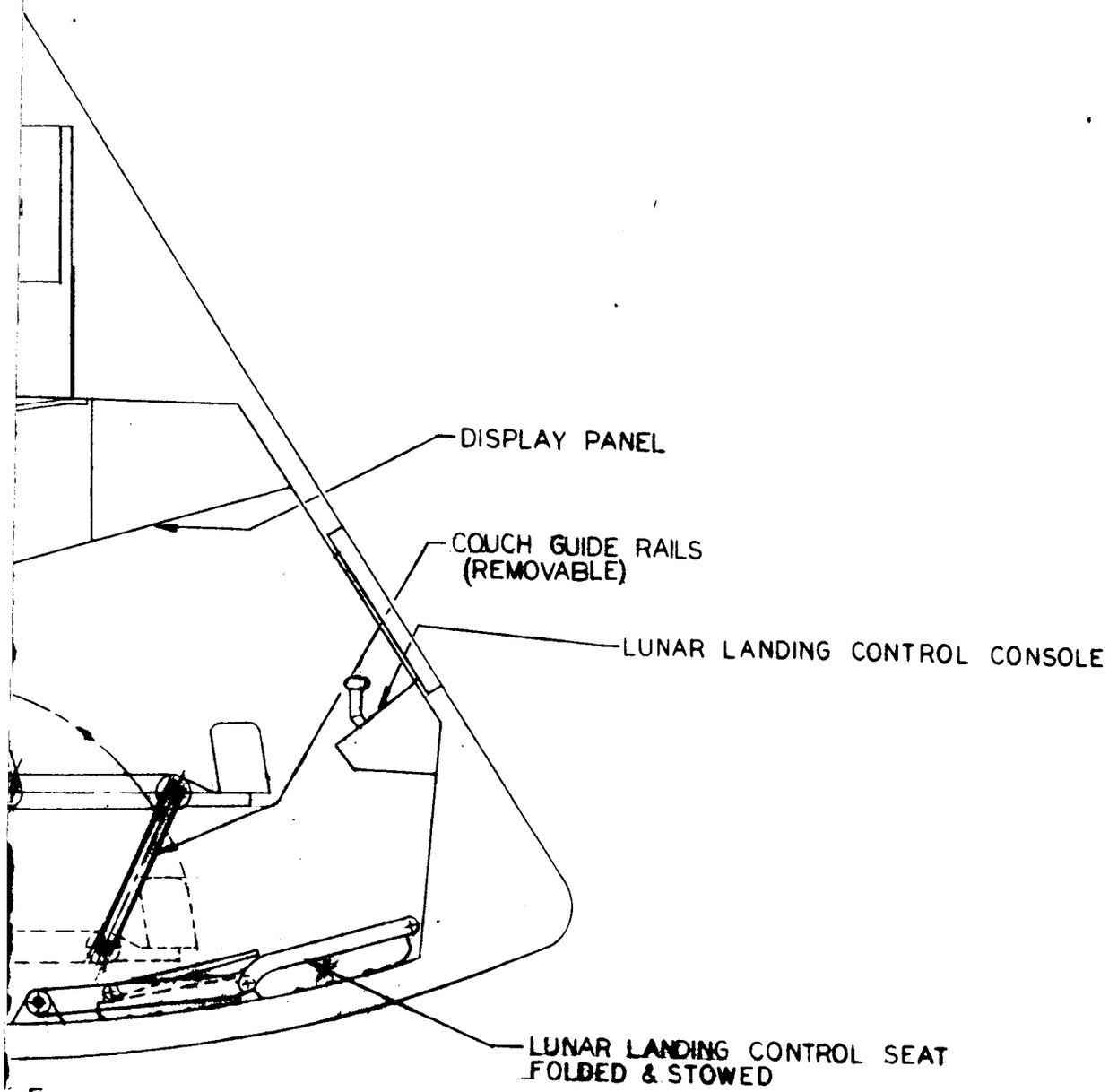
COUCH IMPACT ATTENUATOR

PITCH REACTION
CONTROL ENGINES

A



SECTION
SHOWING CENTER FOL
IN LOWERED POSITION



-F
NG. COUCH
FOLDED

comb by brazed titanium honeycomb.

2) Between the two structural shells - replacement of the microfiber insulation by ADL 16/17 (high vacuum insulation).

3) Secondary structure - a slight weight reduction is achieved due to the removal of the numerous LiOH canisters and employment of light-weight equipment.

4) Replacement of the Avcoat 5026-22 wet blend ablation material with Avcoat 5026-39 dry blend.

5) Removal of docking mechanism and docking windows.

6) Weight savings in systems and equipment.

The interior layout of the CM is dictated by the crew couch arrangement. Since the present Apollo display panel is designed for convenient monitoring by three crew members, it was decided to place three men abreast in the front row and let them perform their tasks in very much the same way as on the present Apollo. Therefore, for powered flight phases, three Astronauts will be seated in the front row and two passengers in the back row behind the left- and right-hand couches.

In order to provide more space and a limited freedom for movement, the upper center couch can be lowered on rails into the back row. The Astronaut (during mid-course) may then get up, fold and remove the center seat, and store it in the center bay below the guidance and navigation equipment and instruments. It is felt that, with this arrangement, the minimum requirements for freedom of motion for personal hygiene, eating, and changing clothes can be satisfied.

To perform the lunar landing, a big window (on the "cool" side at reentry) in the egress hatch is provided. The Astronaut in command is seated right behind this window and has in front of him a small console with the necessary controls and displays to safely accomplish the landing.

This scheme is the same as the one originally worked out for the present Apollo. A mock-up of this arrangement was built at S&ID to check visibility; the tests were discontinued after the LEM decision. Figures 32 through 35 show photographs of the mock-up, indicating the good visibility.

The question of impact attenuation is of major importance. Almost no vertical stroke inside the cabin for the couches can be provided. It was therefore decided in this preliminary design concept to install the same type of external impact attenuation system originally proposed and built (in boilerplate version) for the present Apollo. It consists of a deployable heat shield, oleo-pneumatic shock struts, and the necessary release provisions. Also indicated on the drawing are four small retro-rockets, swinging out after heat shield deployment, and firing just prior to impact. The value of these retros is questionable because of their high weight

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LANDING CONTROLS AND VIEWING WINDOW (VIEW 1)
FIGURE 32

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LANDING CONTROLS AND VIEWING WINDOW (VIEW 2)
FIGURE 33

CONFIDENTIAL



LANDING CONTROLS AND VIEWING WINDOW (VIEW 3)
FIGURE 34

CONFIDENTIAL

CONFIDENTIAL



LANDING CONTROLS AND VIEWING WINDOW (VIEW 4)

FIGURE 35

CONFIDENTIAL

and low reliability; they may be deleted. The lateral stroke of the couches in both y and z direction is in accordance with the present Apollo.

It is assumed that "water landing only" can not be guaranteed with sufficient certainty to delete the external impact attenuation system in the five-man command module even if primary water landing is specified.

Accommodation of systems and equipment may not be a severe problem. Replacing LiOH with a molecular sieve for CO_2 removal saves weight and, especially, volume. Furthermore, no scientific equipment (10 ft^3) is carried; and all systems, especially guidance and stabilization, make use of miniaturized equipment. The bulky personnel life support systems (PLSS; back-packs) are stored in the airlock tunnel. It was assumed that the Astronauts, at least while in their couches, wear their space suits most of the time.

The reaction control system will not be changed. The parachute system will have to be modified. Due to the higher CM weight, the chute diameter will be enlarged accordingly.

Electrical Power Supply and Water Management

Two different types of electrical power supply systems merit consideration for the five-man direct lunar Apollo mission:

1. Fuel cells, which provide water for the crew.

2. Isotopic - dynamic power systems, with no water production.

Both power systems, as well as the life support system, will be discussed in more detail in the following sections. The integration of power supply and water management to achieve minimum weight is outlined herein.

Table 12 summarizes the power requirements for a typical mission.

Table 13 indicates the crew water demand for the same mission.

Table 14 summarizes the weights of several system combinations. Column 1 is for a fuel cell system which is sized to provide enough water for the crew (11 lb/man-day); no water recovery is intended, but the fuel cells produce excess electric power.

Column 2 reflects a fuel cell - total water recovery system combination. The fuel cells provide power for the mission requirements and to operate the water recovery system. It is evident that a considerable savings in weight can be achieved over a fuel cell system sized for the water requirements. However, the fuel cell and total water recovery system together produce excess clean water.

Column 3 is representative for a partial water recovery system in combination with fuel cells. The water requirements of the crew average 55 lb/day; the fuel cells produce roughly 40.2 lb/day on the average over the

mission; 14.8 lb/day or 2.96 lb/man-day will have to be made up by the water recovery system from any combination of urine, washwater, and from the air conditioning system. The water recovery rates per man and day are listed in Table 13.

It is evident that water recovery from the air-conditioning system alone can cover the difference of 2.96 lb/man-day, but marginal. Differences and anomalies in the water consumption or production could upset the balance. It is, therefore, necessary to carry more fresh water supply at launch.

Column 4 is representative of a more complete water recovery-fuel cell-system combination. Water recovery from either urine or washwater can be supplemented to air-conditioner water recovery, which is easiest. It can be assumed that reclaimed water from the air conditioner and from urine will be available. The availability of wash water is less certain, especially during the early part of the mission. Therefore, water recovery from the air conditioning system and urine was assumed (the reclamation rate is somewhat higher than actually needed). This combination turns out to be somewhat lighter than total recovery but heavier than recovery from air conditioner only.

In Column 5, an isotopic-dynamic power supply, in combination with a total water recovery system is presented. This combination has the lowest over-all weight and would therefore be highly desirable. However, the isotopic -dynamic system is in an early stage of development, whereas the fuel cells are readily available.

Column 6, for reasons of comparison, list the weights for a three-man, fourteen-day mission as studied in SID 62-1467.

Among the fuel cell system combinations (Columns 1 - 4), it appears that Column 3 yields the lightest over-all system. The more complete water recovery systems turn out to be heavier because of hardware weight and the fuel penalty to operate it.

The systems combination summarized in Column 3 will be chosen for the five-man Apollo II: Fuel cells and water recovery from the air conditioning system. This choice is based on lowest weight, availability of the fuel cells, and comparative ease of water recovery from the cabin atmosphere with high reliability; survival of the crew can still be assumed even if the recovery system fails.

Fuel Cell Power System for the Five-Man Apollo II

The fuel cell system to be employed for the five-man Apollo II is essentially the same as the one described for the three-man direct lunar landing mission in SID 62-1467. The power output required for the five-man version is only slightly (0.1 KW) higher than for the three-man version; no problems are anticipated in adapting the same fuel cell to the slightly higher output. The system is described herein and a weight summary is presented.

TABLE 12 - POWER REQUIREMENTS FOR A TYPICAL MISSION; PRODUCTION OF WATER

		Ascent	Earth Orbit	Trans-Lunar Inject	Trans-Lunar	Moon Orbit	Moon	Moon Ascent	Trans-Earth Injection	Trans-Earth Coast
		9 Min.	21 Hrs.	6 Min.	70 Hrs.	12 Hrs.	168 Hrs	6 Min	2 Min.	65 Hrs.
Illumination		150	150	150	100	100/150	20	150	100	100
Propulsion				(150)	(150)			(150)	(150)	(150)
In-Flight Test			(100)		(100)	(100)	(100)			(100)
Guidance & Navigation		210	274	210	50	210	50	210	274	50
Stabilization & Control		202	202	202	100	100	50	202	202	100
Telecommunication		182	182	112		112	40	112	112	112
		(202)	(202)	(192)		(192)	(242)	(192)	(192)	(192)
Displays		100	100	100	100	100	70	100	100	100
Environmental Control	5 men	690	690	690	690	690	690	690	690	690
	7 men	746	746	746	746	746	746	746	746	746
Auxiliary		210	210	210	210	210	190	190	210	210
Total	5 men	1744	1808	1674	1250	1522	1110	1654	1748	1362
	7 men	1800	1864	1730	1306	1578	1166	1710	1804	1418
Scientific Instruments on Pad for Design Variations		200	200	200	400	200	400	200	200	200
Total - incl. 6% Distribution Losses	5 men	2060	2070	2000	1750	1830	1600	1965	2070	1650
	7 men	2120	2130	2060	1810	1890	1660	2025	2130	1710
Total - KW Hours	5 men		43.5		122.5	22.0	268.8			107.2
	7 men		44.7		126.7	22.7	280.0			111.1
Production of Water, lb	5 men		43.5		123.0	22.0	267			107
	7 men		44.7		127	22.7	280			111
TOTAL WATER PRODUCTION	5 men	562								
	7 men	585								

TABLE 13

BASIC DAILY WATER DATA

ITEM	NUMBER OF MEN IN CREW						
	1	2	3	4	5	6	7
Water Requirements							
Food & Drink, lbs	6.00	12.00	18.00	24.00	30.00	36.00	42.00
Wash Water, lbs	5.00	10.00	15.00	20.00	25.00	30.00	35.00
TOTAL, LBS	11.00	22.00	33.00	44.00	55.00	66.00	77.00
Water for Recovery							
Urine, lbs	3.34	6.68	10.12	13.36	16.70	20.04	23.38
Perspiration & Respiration, lbs	3.38	6.76	10.14	13.52	16.90	20.28	23.66
Wash Water, lbs	5.00	10.00	15.00	20.00	25.00	30.00	35.00
TOTAL, LBS	11.72	23.44	35.36	46.92	58.60	70.32	82.04

TABLE 14 - WEIGHT COMPARISON, ELECTRIC POWER, & WATER SYSTEMS, FIVE-MAN APOLLO II

	Fuel Cell Supplies all Water	Fuel Cell + Total Water Recovery	Fuel Cell + Air Cond. Water Rcvy.	Fuel Cell + Urine & Air Cond. Rcvy.	Isotopic-Dynamic + Total Rcvy.	3-Man* Fuel Cell, no Water Rcvy.	Remarks
Fuel Cell	450	380	365	380	-	365	
Isotopic-Dynamic Power Source	-	-			860		Shielded for 50 rem/yr PM-147, 5 KW
Pwr Conv. & Distribution	358	424	424	424	424	424	
Reactants-H_2+O_2	903	762	678	757	-	370	Includes 10% reserve
Tanks & Support	133	80	75	90	-	75	
Rad. & Hardware	75	75	75	75	**	75	
Stored Water	50	50	150	50	250		Reserve supply
Cooling Water	10	10	10	10	10		
Water Leakage	28	28	28	28	28	28	7 psi cabin pressure
Water Recovery System Weight	-	60	3	45	60	-	Dual redundant intermittent operation
Tankage & Plumb.		40	10	25	40	-	
Backpack Water	120	120	120	120	120	-	
Power Output(KW)	(2.3 KW)	(2.1 KW)	(1.85 KW)	(2.1 KW)	(2.1 KW)		
TOTAL SYSTEMS WT	2127	2029	1938	2004	1792	-	

* SID 62-1467
** Included above

Recommended Fuel Cell System Design

The fuel cell power plant will be designed for partial stand-by redundancy. In contrast with the present Apollo design, where several fuel cell systems operate in parallel at the same time, only one fuel cell system operates. For redundancy and high reliability, all essential subassemblies are duplicated. As explained later, some of the duplicated subassemblies are continuously operating; some of them are on stand-by. This arrangement will have comparable reliability with the present Apollo fuel cell system and will have less weight and requires less space. See Figure 36.

The basic fuel cell system is based on the present Apollo fuel cell design using the latest version of cells. This arrangement will use thirty cells in series connection and will maintain 28V $\pm$ 2V under a load range of 1 KW to 1.8 KW. See Figure 37.

The minimum load on the system under normal conditions is 1.0 KW with 29.3V output. Under maximum load condition of 1.8 KW, the voltage will drop to 26.0 volts. If the load would increase beyond 1.8 KW, the storage batteries would be connected automatically and would share the load with the fuel cells until the load current decreases to a value corresponding to 1.5 KW load. This arrangement will maintain relatively constant voltage on the D.C. line and will operate the fuel cells at high efficiency.

Arrangement of the Cell System

The fuel cells are arranged into five sections, with ten cells in each section. The five sections are in one assembly but have individual connections to both the electrical system and to the fuel system. Under normal operation, three sections are connected in series while two additional sections are on stand-by. Only three groups generate voltage. Under any failure of one section, the stand-by cells are connected into the system and the faulty section is isolated. The active cells provide enough heat for the stand-by cells to maintain stand-by redundancy instead of the present parallel redundancy, increasing the system over-all reliability.

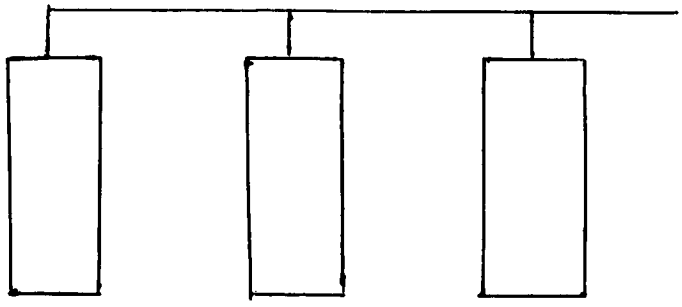
The hydrogen regenerator, reactant regulators, and coolant system are designed to be redundant.

There will be a complete stand-by hydrogen regenerator assembly which is switched on as soon as the power to the separator pump motor is disconnected.

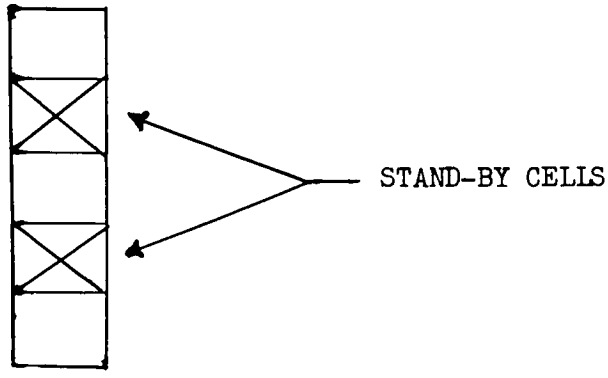
The reactant regulators are connected in series and the main regulator operates in tendem with the stand-by regulators. If one regulator fails, the other still will maintain the pressure within range of specifications.

There is a complete stand-by coolant circuit which, under normal conditions, is not in use. Only when fuel cell temperature exceeds a certain value would this circuit be activated.

FIGURE 36



PRESENT APOLLO FUEL CELL ARRANGEMENT
3 FUEL CELL SYSTEMS ARE OPERATING
ALL THE TIME IN PARALLEL CONNECTION



PROPOSED FUEL CELL ARRANGEMENT WITH
STAND-BY CELLS INSERTED BETWEEN
OPERATING CELL TO PROVIDE EASY START-UP

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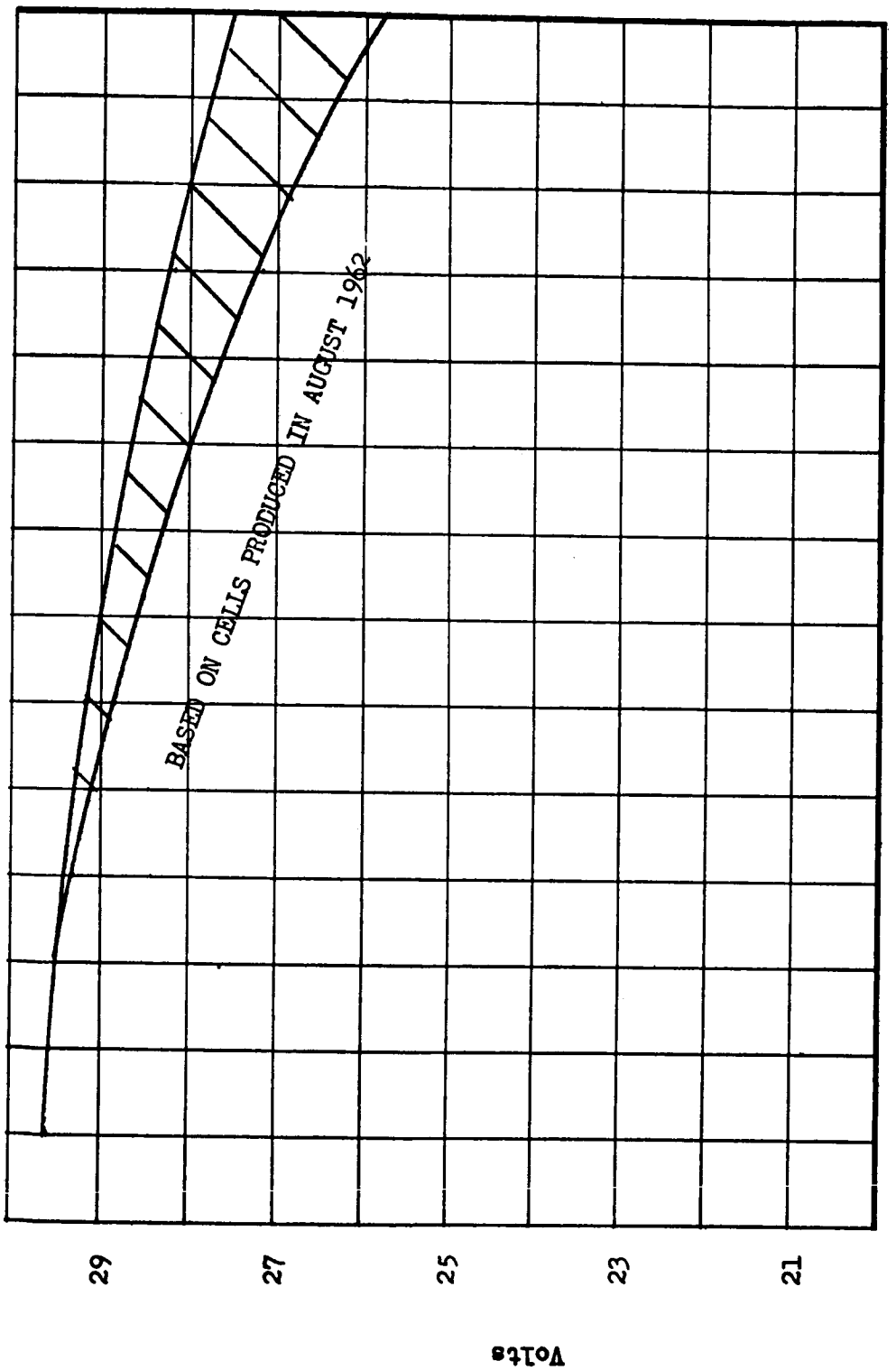


FIGURE 37

PERFORMANCE OF A 1.35 KW POWERPLANT USING 30 P&W FUEL CELLS IN SERIES

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The fuel cells considered are identical to the cells presently in production and the design is based on the test data obtained from Pratt & Whitney Aircraft. It is, however, expected that the fuel cell efficiency actually will be improved over the present data.

In Figure 38, the losses of the fuel cells are plotted. These are the losses of the electro-chemical conversion only, and do not include the losses of necessary coolant pumps and water separation. Considering these losses to be 100 watts for each fuel cell system, the break-even point for best efficiency can be calculated. Presently, the break-even point between the use of one or two parallel operating fuel cell power plants is 1300 watts. If less power is consumed, one cell alone is more efficient. For more power, two parallel operating fuel cells should be selected. On the graph, the difference of losses is plotted using one and two fuel cell systems. With advances in fuel cell technology, the use of one fuel cell power plant only instead of two in parallel is more and more advantageous and, considering an average load of 1200 watts during the mission, some weight saving is obtained.

The use of stand-by redundant sections requires start-up of cells during the mission. Stand-by cells are filled with inert gas (nitrogen or equivalent). To activate these cells, the inert gases have to be flushed out. During flushing of the cells, some extra fuel is consumed. It also takes a short time to start up the cells. During start-up of the stand-by cells, two methods can be used:

1) Switching to the storage batteries which can be activated when the voltage drops below 25V.

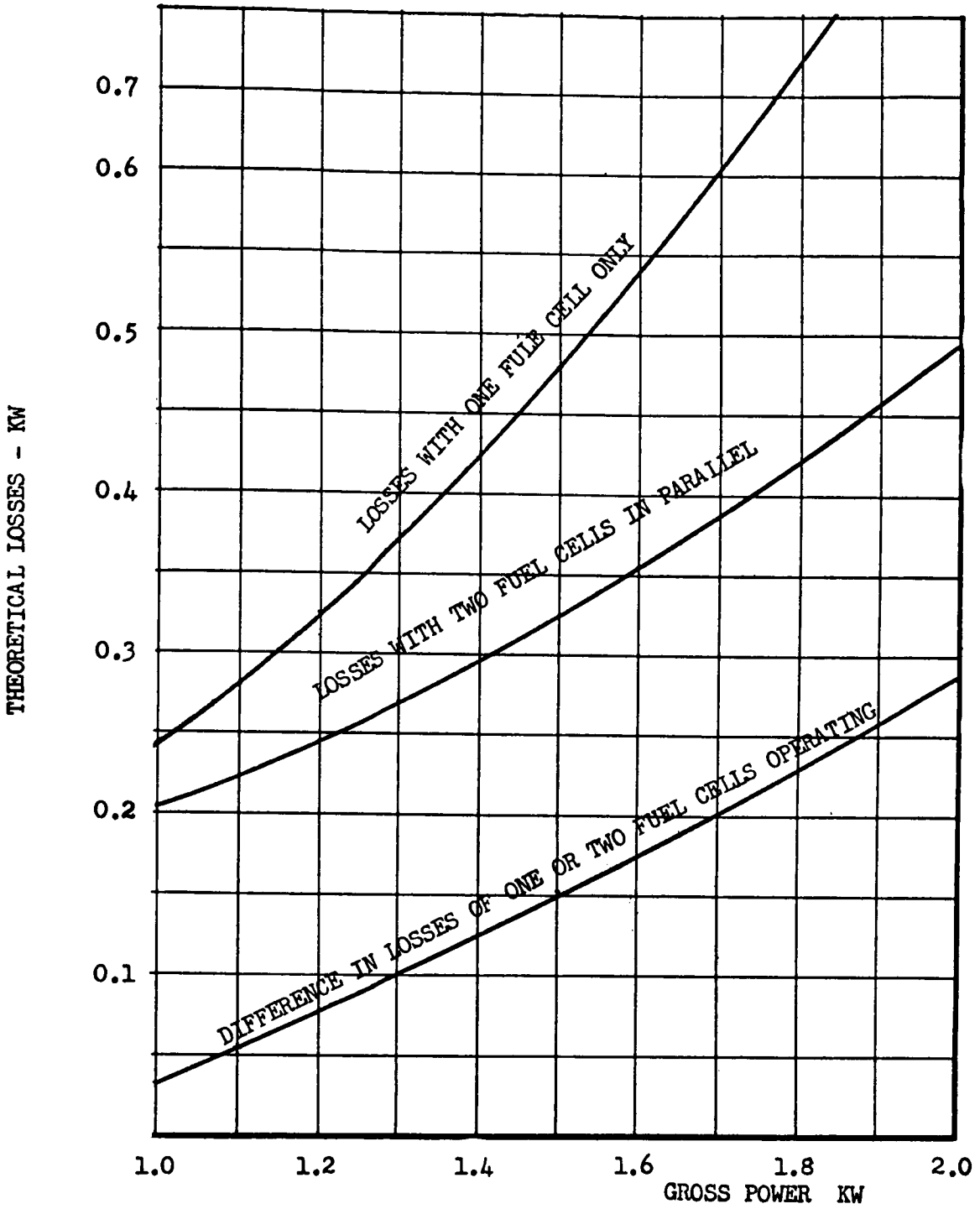
2) Reducing the power requirements to emergency level and operate using the faulty cells.

The chances are practically nil that a fuel cell would fail open circuited. If a cell is short circuited, the power supply voltage drops approximately by 1 volt, but it is still operative. A simple check of voltage output of each section would reveal the location of the fault.

Weight of the Fuel Cell Power System

The weight of the fuel cell power system used for the Apollo was originally estimated to be 245 pounds. Three redundant power systems are used in parallel. Since the original estimate, improvements in technology and weight reduction of components have brought the weight down to 215 pounds. An additional weight saving is expected by reducing the size of the fuel cells f for the parallel operating systems. The smaller cells would still have the same efficiency as the worst cells now in test. This reduction would bring the weight of the individual power system to 185 pounds as presently used for the Apollo.

Weight of the proposed power system would be more than the weight of one of the presently used power systems but less than two and considerably



NOTE: Data presented here are pessimistic and based on poorest fuel cell tested to date.

FUEL CELL LOSSES

FIGURE 38

less than the presently used power system package of three parallel redundant fuel cells.

Estimated weight of the new system with 5/3 redundancy using thirty cells in series and twenty stand-by cells, all using the present electrode area, is calculated in Table 15.

TABLE 15

FUEL CELL SYSTEM WEIGHT

Service Module	Fuel Cell Assembly	365 lb
	Power Conversion	78 lb
	Reactants, $H_2 + O_2$	678 lb
	Tankage + Supports	75 lb
	Radiator + Hardware	75 lb
	SM SUB-TOTAL	1271 lb
Command Module	Power Distribution, Conversion, Batteries	346 lb
TOTAL POWER SYSTEM WEIGHT		1617 lb

Environmental Control and Life Support Systems

The environmental control and life support system of the Apollo II CM is subject to several constraints. With respect to the basic Apollo mission profile, a 5 psia pure oxygen atmosphere is adequate. If the vehicles are tied in with lunar base operations, or, if an extension of the mission is considered, a mixed atmosphere (N_2/O_2) at higher internal pressures (7 psia) may offer operational advantages or even be necessary. For the purpose of this study, a 5 psia pure oxygen atmosphere has been tentatively selected, but a mixed atmosphere has also been considered. The requirements for the ECS/LSS systems are listed in Table 16.

The anticipated losses due to gas leakage amount to approximately 70 lb over the mission.

Table 17 presents the atmosphere losses due to airlock operations for both pure oxygen and mixed atmospheres. For a typical mission, thirty excursions, resulting in sixty airlock operations, have been assumed. Table 18 lists the corresponding weights for the back-pack charges, and Table 19 summarizes the total weight losses to be expected for extravehicular opera-

PAGES 127 - 134

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TABLE 16

FIVE-MAN APOLLO II VEHICLE - ECS/LSS REQUIREMENTS

Mission - 14 Days
Crew - 5 Men

Compartment Atmosphere	Selected	Alternative
Total pressure	5.00 psia	7.00 psia
Oxygen partial pressure	5.00 psia	4.50 psia
Carbon dioxide partial pressure (max.)	0.14 psia	0.14 psia
Diluent	None	Nitrogen
Temperature	60-70°F	60-70°F
Relative humidity	40-70%	40-70%
Leakage rate (assumed)	5.00 lb/day	5.00 lb/day
Compartment volume	210 ft^3	210 ft^3
Airlock volume	30 ft^3	30 ft^3

Crew Requirements

Metabolic inputs: 11,500 BTU/man-day (2900 Kcal)

Oxygen	1.80 lb/man-day
Water for food and drink	6.00 lb/man-day
Water of oxidation	0.79 lb/man-day
Food (freeze-dried, high calorie)	1.33 lb/man-day

Metabolic outputs:

Carbon dioxide	2.11 lb/man-day
Water:	
Perspiration and respiration	3.38 lb/man-day
Urine	3.34 lb/man-day
Fecal water	0.22 lb/man-day
Feces (solid)	0.08 lb/man-day

[redacted]

TABLE 17

LOSSES OF ATMOSPHERE DUE

TO

AIRLOCK OPERATIONS

Airlock Operations	ATMOSPHERE			
	100% O_2 @ 5 psia lbs	Mixed O_2-N_2 (O_2 Part. Press. = 3.5 psia) Total Pressure = 7 psia		
		O_2 lbs	N_2 lbs	H_2O lbs
5	4.215	2.956	2.580	0.120
10	8.430	5.911	5.159	0.240
15	12.645	8.867	7.739	0.360
20	16.860	11.822	10.318	0.480
25	21.075	14.778	12.898	0.600
30	25.290	17.733	15.478	0.720
42	35.406	24.826	21.668	1.008
45	37.935	26.600	23.217	1.080
60	50.580	35.466	30.956	1.440
90	75.870	53.199	46.434	2.160
100	84.300	59.110	51.590	2.400
120	101.160	70.932	61.908	2.880
140	118.020	82.754	72.226	3.360

TABLE 18

RESUPPLY WEIGHTS FOR BACK-PACK OPERATIONS

(Based on NASA Space Suit Weights)

Number of 4-Hour Operations	RESUPPLY WEIGHTS			
	O_2 Recharge from ECS lbs	O_2 Stored in Individual Bottles lbs	Lithium Hydroxide Cartridge lbs	Water for Cooling lbs
1	0.90	4.15	3.80	4.00
2	1.80	8.30	7.60	8.00
3	2.70	12.45	11.40	12.00
4	3.60	16.60	15.20	16.00
5	4.50	20.75	19.00	20.00
6	5.40	24.90	22.80	24.00
7	6.30	29.05	26.60	28.00
8	7.20	33.20	30.40	32.00
9	8.10	37.35	34.20	36.00
10	9.00	41.50	38.00	40.00
20	18.00	83.00	76.00	80.00
30	27.00	124.50	114.00	120.00
60	54.00	249.00	228.00	240.00
90	81.00	373.50	342.00	360.00

TABLE 19

WEIGHT OF MATERIALS EXPENDED DUE TO EXTRAVEHICULAR EXCURSIONS

Number of 2-man Excursions	Back-pack Expendables - lbs			Airlock Losses - lbs			
					Mixed Atmosphere - 7 psia		
	O_2	LiOH Cartridges	H_2O	100% O_2 5 psia	O_2	N_2	H_2O
1	1.80	7.60	8.00	2.529	1.773	1.548	0.072
4	7.20	30.40	32.00	10.116	7.093	6.191	0.288
10	18.00	76.00	80.00	25.290	17.733	15.477	0.720
14	25.20	106.40	112.00	35.406	24.826	21.668	1.008
21	37.80	157.60	336.00	53.109	37.239	32.502	1.512
28	50.40	212.80	224.00	70.812	49.652	43.336	2.016
35	63.00	266.00	448.00	88.515	62.065	54.170	2.520
42	75.60	319.20	672.00	106.218	74.478	65.004	3.024

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tions.

The atmospheric life support requirements are summarized in Table 20, both for pure oxygen and mixed atmospheres. The weight for back-pack charges and airlock operations are based on thirty excursions, with a total of sixty airlock operations. One puncture and two repressurizations have been allowed.

Systems Consideration

The two major modifications from the present lunar Apollo are the incorporation of a molecular sieve unit for removal of carbon dioxide and the employment of a partial water recovery system (from the cabin atmosphere conditioner). Total ECS/LSS weight is shown in Table 21.

Structures and Weights

The basic structure of the five-man Apollo II is the same as for the three-man vehicle outlined in SID 62-1467 (see Figure 28). Only very minor modifications are necessary.

For the lunar launch stage ($S_t$age II) of the three-man vehicle, a conventional double-wall sheet and stringer construction was found to be optimal and was, therefore, selected (material: Rene 41 for boost temperatures). The face sheet thickness was determined by the meteorite probability of no penetration of 0.99 (14 days); this structure was adequate to carry the impending loads with a comfortable margin. The slightly higher payload weight of the five-man version results in stresses still below the allowable level. Therefore, based on the same meteorite probability of no penetration, no change or weight increase in the Stage II structure is necessary.

The lunar landing stage for the three-man mission (outlined in SID 62-1467, "Lunar Logistic Vehicle Preliminary Study") was designed as a logistic vehicle able to land a variety of payload shapes and sizes on the Moon. The entire load is carried through the outer cylindrical shell (see Figure 28). This shell consists of an aluminum honeycomb (foam-filled for meteorite protection) sandwich construction; it was designed for high-allowable stress levels because of payloads unknown at the time of its design. Again, no modifications to the structure of the landing stage (Stage I; LLV) is necessary.

The landing gear of the lunar logistic vehicle (Stage I) was designed to the requirements for unmanned landings which require a higher stability parameter k (ratio of center of gravity height above ground to landing gear spread radius) than manned landings. Therefore, a considerable reduction of landing gear size and weight can be achieved by designing for manned landings only. This modification was incorporated for the five-man Apollo II mission. (If unmanned missions are also desired, it may be necessary to beef up the gear for the higher stability criteria.)

The gage thickness of the propellant tank walls were determined by internal tank pressure in the cryogenic vehicles studied earlier. Due to the

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TABLE 20

BASIC ATMOSPHERIC REQUIREMENTS

		Mixed Atmosphere	
	100% O_2	O_2	N_2
Metabolic @ 1.8 lb/man-day	126.00 lb	126.00 lb	-
Leakage	70.00 lb	37.38 lb	32.62
Subtotal metabolic and leakage	196.00 lb	163.38 lb	32.62
50% margin	98.00 lb	81.69 lb	-
Total metabolic and leakage	294.00 lb	245.07 lb	32.62
Puncture ($\frac{1}{2}$ dia hole - 5 min)	5.73	3.60	3.14
2 Repressurizations	5.90	11.70	7.22
Total req'd (less airlock & back-pack)	305.63 lb	260.37	42.98
15% margin			6.45
Residual	30.56	26.04	4.94
Vent Losses	3.06	2.61	0.53
Fill Weight	339.25	289.02 lb	54.90
Volume $\frac{W_f}{68.8} = \frac{377}{68.8}$	5.48 ft^3	4.20 ft^3	
Volume $\frac{V}{49.4}$			1.11 ft^3
Storage vessel weight	66.00 lb	59.20 lb	12.08
Hardware and plumbing	13.20 lb	11.84 lb	2.42
Total stored weight (less airlock and back-pack)	418.45 lb	360.06 lb	70.40

Airlock and Extravehicular Operations (30)
(Pure Oxygen Atmosphere)

Airlock losses, O_2	38
Backpack charges, O_2	124
LIOH	114
Water	120
TOTAL	396 lb

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[redacted]

TABLE 21

TOTAL ECS/LSS WEIGHT

COMMAND MODULE	
Back-pack Charges & Supplies	358 lb
Pressure suit circuit	90 lb
Water-glycol cirucuit	59 lb
Pressure and temperature control	19 lb
Oxygen supply system	25 lb
Water supply system	45 lb
Plumbing, wiring, instrument	35 lb
Supports, controls, electrical	38 lb
Molecular sieve unit	60 lb
Air conditioning water recovery	5 lb
Charcoal	5 lb
Freon	10 lb
	391 lb
SERVICE MODULE	
Water-glycol circuit	94 lb
Water supply system	8 lb
Oxygen supply system	3 lb
Oxygen	377 lb
Oxygen storage vessel	79 lb
Water stored at launch	188 lb
TOTAL ECS/LSS	1498 lb

[redacted]

slightly larger propellant quantities (the tanks still fit into the same envelope), the radii of curvature of the toroidal and ellipsoidal tanks will increase, resulting in higher stress. The necessary gage increase (percent) over the gages given in SID 62-1466 and 1467 are listed below in Table 22. They result in only a slight over-all weight increase. A preliminary weight estimate of the five-man Apollo II is shown in Table 23.

TABLE 22

Module	Designation	Propellent Weight (lbs)	Tank	Shape	Percent Gage Increase
Lunar Take-off	SM II-5	17,064	LH_2	Torus	1.6%
			LO_2	Ellipsoid	4.5%
Trans-Lunar Landing	LIM/LLV	54,145	LH_2	Torus	1.2%
			LO_2	Torus	.5%

SEVEN-MAN APOLLO II

Human Factors

The purpose of this study is to evaluate the life support requirements for a seven-man CM with a configuration similar to Apollo I but enlarged by a factor of 1.14.

The areas considered are interior dimensions (couch arrangements), life support requirements, egress, habitability, and metabolic requirements.

Interior Arrangement

The most applicable positioning of couches in a seven-man Apollo II would be three above and four below, with a central aisle availability and ability to stow the top center couch. Some of the problems for consideration in this arrangement are:

1) Shock attenuation for each couch.

2) Ease of control and navigation from the top left and right couches.

3) Ability and area to stow top center couch.

4) Shifting of personnel in the event of incapacitation of crew members.

5) Umbilical connections from couches to environmental control and life support systems.

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TABLE 23

PRELIMINARY WEIGHT ESTIMATE

APOLLO II - 5-MAN DIRECT MISSION

STAGE II			
Structural Section		1,850	
Propulsion System		(1,355)	
Propellant Tanks		640	
LO_2	285		
LH_2	355		
Pressurization System		240	
Propellant Feed System		35	
Engine Installation		440	
Trapped Propellant		250	
Residual Vapor		85	
Booster Burn-out Weight		3,540	
Propellant (Total ΔV = 10,074 fps, T/W_o = 0.46)		17,064	
LO_2	14,200		
LH_2	2,864		
Gross Weight, Booster Stage II			20,604 lb
Payload			
Command Module		(8,712)	
Structure		3,600	
Crew Systems		2,000	
Communications & Instrumentation		760	
Guidance & Navigation		250	
Stability and Control		152	
Reaction Control System		355	
Electrical Power		346	
Environmental Control System		749	
Earth (Water) Landing System		500	
Service Module Equipment		(3,059)	
Electronics		139	
Electrical		1,271	
Reaction Control		900	
Environmental Control		749	
Contingency		(100)	
Propellant Boil-off		(145)	

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[redacted]

Total Payload and Boil-off			12,016 lb
LUNAR LAUNCH WEIGHT			32,620 lb
STAGE I			
Structural Section		2,450	
Propulsion System		4,445	
Propellant Tanks, Insulation, Support			
LO_2 Tank	735		
LH_2 Tank	1,890		
Pressurization System	790		
Propellant Feed System	120		
Engine Installation	910		
Landing Gear		1,540	
Trapped Propellant		900	
Residual Vapor		250	
Propellant (Total Δ V = 11,261 fps, T/W_o = 0.43)		54,145	
LO_2	45,105		
LH_2	9,040		
Propellant Boil-off		655	
Stage I Gross Weight			64,385 lb
ESCAPE WEIGHT			97,005 lb

6) Stowage and ability to don helmets and boots from couch position.

Egress

Egress from the module under normal conditions would require the removal of the top center couch and the top crew members could egress from the center aisle. The lower inboard crew members could egress in the same manner, but the outboard lower crew members would have to roll or work over the lower inboard couches into the center aisle.

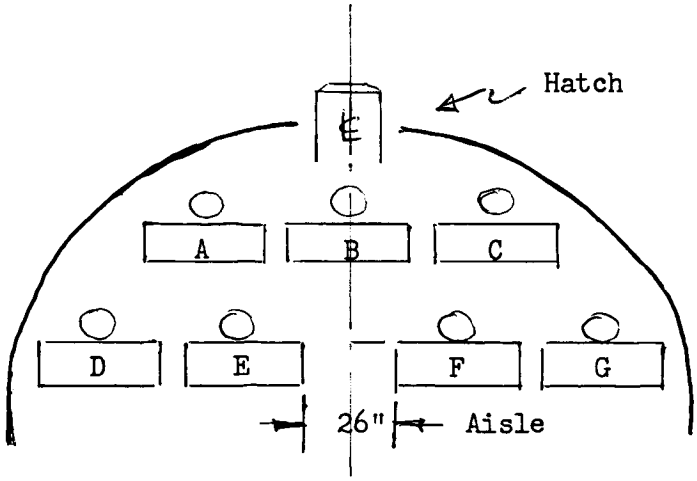
Egress training and test simulation with simple mock-ups would disclose the areas that created the greatest difficulty in body movement to egress.

Habitability

It is essential to consider habitability factors because they have an important effect on man's physiological and psychological well being. These factors are (1) environmental control, (2) nutrition and personal hygiene, (3) gravitational conditions, (4) living space, and (5) crew work-rest cycles.

Hand and face cleansing provisions are necessary, but body cleansing should not be required during the short mission.

Because of the confined conditions, special training in a simulator of the seven-man Apollo configuration should be provided.



Requirements

1) Clearance between lower and upper tier of couches equivalent to a 90 percentile crew member fully equipped (less back-pack). Enough clearance in couches for food and water sustenance.

2) Ability for crewman B to egress from couch and stow couch. Overhead swing bar or hand holds (typical for all couches).

3) Couch E and F slide left and right to create a 26" wide aisle. Ability for crew members E and F to swing into center aisle; ability for crew members D and G to transfer to empty couches and

to aisle.

4) Design of shock attenuation system compatible to seven-couch arrangement.

5) Ability to transfer and minister aid to any incapacitated crew member.

6) Ability for crew members A and C to have accessibility to controls and displays for mission requirements, lunar landing visibility and lunar landing capability.

Conclusions

The Apollo II-7 human factors requirements parallel the five-man direct lunar landing study, with the added factor of an enlarged module to accommodate seven crew members. The areas for study and simulation to establish applicable design criteria, operational capabilities, and habitability conditions from a couch-position are:

1) Egress - ingress from couches.

2) Ability to don helmets and pressurize suits.

3) Removal of top center couch and ability to shift crew members.

4) Controls - accessibility and checking possibility of inadvertently activating.

5) Communications.

6) Shock attenuation.

7) Maintenance - capabilities with couch arrangement.

8) Safety - radiation, emergency conditions - egress with incapacitated crew member.

9) Personal psychological factor - confinement, waste disposal, eating.

Preliminary Design

To accommodate seven astronauts, the base diameter of the command module had to be enlarged by eighteen inches. Otherwise, the principal design remains very much the same.

To be compatible with the up-rated Saturn V, weight is at a premium. Therefore, the same lightweight structural design will be employed as in the five-man CM: double-sandwich wall construction (titanium and aluminum) with ADL 16/17 insulation in between, lightweight ablation material; miniaturized

and lightweight equipment.

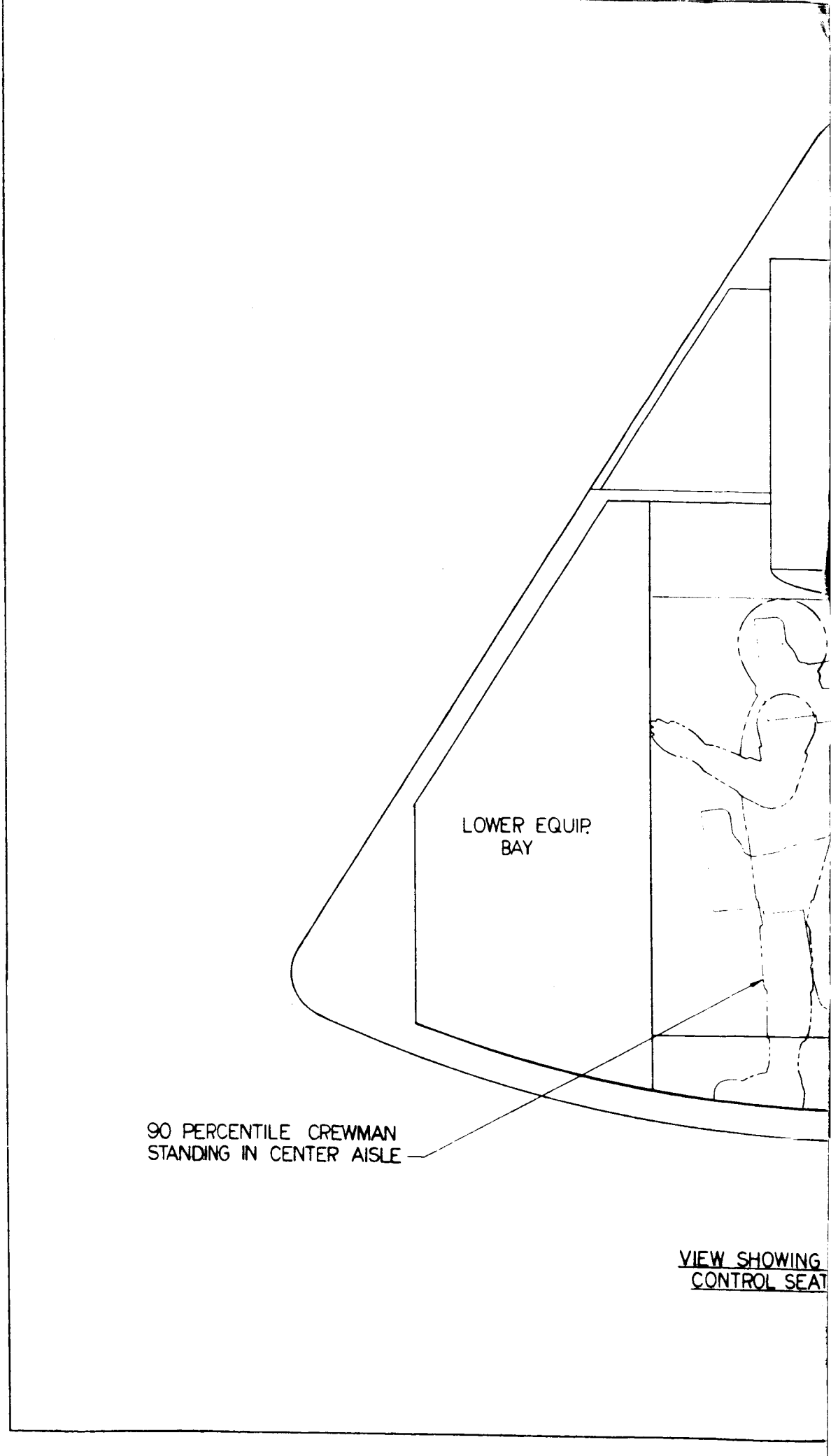
The logical crew arrangement is obviously three men in the front and four in the back row. The couches, suspended from the cabin ceiling, are arranged that way for launch and Earth re-entry. During mid-course and for the lunar landing, the couch assembly will be split up. The three left-hand couches (one in the front and two in the back row), will be moved five inches to the left, and the three right-hand couches will be moved in a similar manner to the right to form an aisle twenty-six inches wide; the forward center couch will be removed and folded. Thus, the command module will be operated during launch and entry by a full complement of three astronauts as in the basic Apollo. The internal arrangement showing the couches in various positions is shown in Figure 39. The astronaut in command will perform the lunar landing from a seated position in front of a window built into the egress hatch; the controls and displays necessary for the landing are arranged on a retractable control panel below the window. The lunar landing position arrangement using the upper center couch less footrest as a seat is indicated in Figure 39.

The airlock is installed on top of the pressurized compartment; during launch, midcourse and Earth re-entry, the personnel life support systems (PLSS) are stored in the airlock.

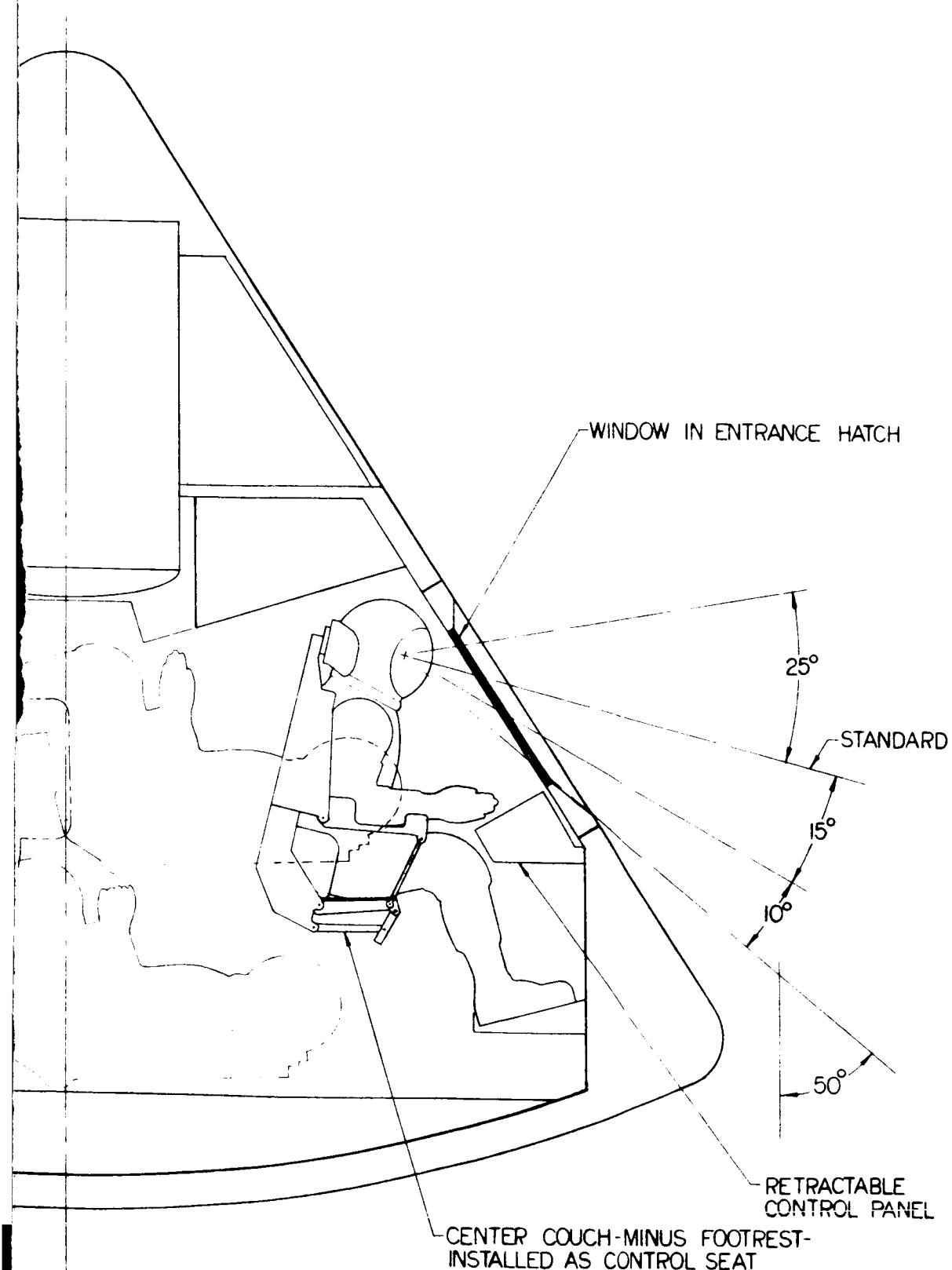
The systems, equipment, and supplies are located in the same manner as on the present Apollo; in equipment bays. It is realized, however, that the cramped accommodation will make in-flight repair and maintenance extremely difficult if not impossible.

Due to the two-row couch arrangement, no vertical stroke for impact shock attenuation can be provided; the lateral stroke (y and z axis), however, is adequate. It was therefore assumed that an external impact attenuation system is installed, consisting of a deployable heat shield, oleopneumatic shock struts, and the necessary release mechanisms. This attenuation system should keep the decelerations to an acceptable level (eyeballs in, 20 g normal, 35 g emergency landing condition).

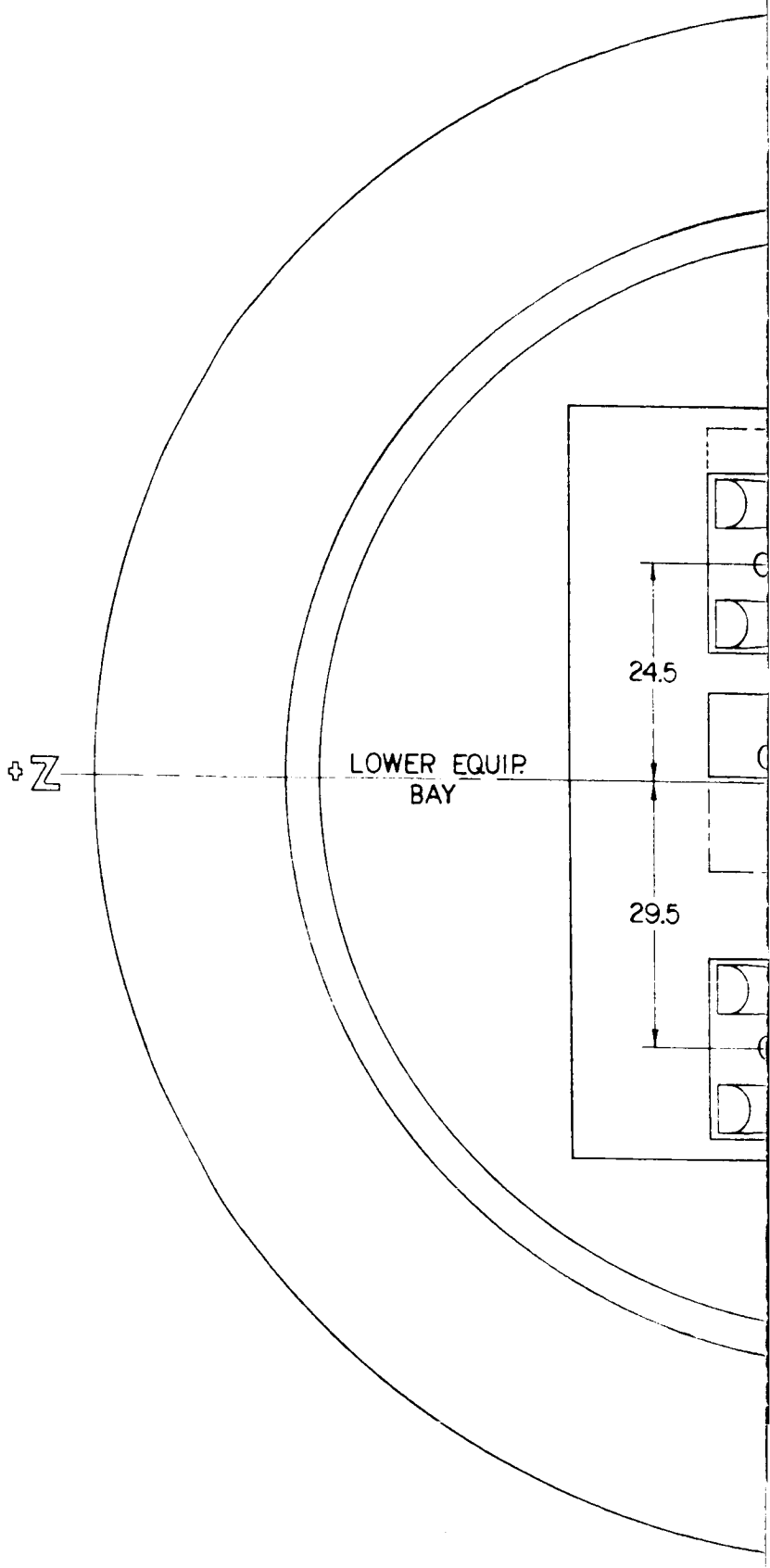
An alternate internal arrangement of the crew was given a cursory examination to determine if it would be possible to reduce the base diameter of the command module. This arrangement consisted of four crewmen in the lower layer, positioned in two rows with a fixed twenty-four inch space between the rwo rows. In each row, two crewmen were positioned in line, head to foot. The remaining three crewmen were positioned in an upper row three abreast directly above two of the lower crew. This provided three crewmen positioned in front of the display panels similar to Apollo. An over-all evaluation of this arrangement did not permit as small a base diameter as the configuration shown and, in addition, it eliminated any possibility of having a lower equipment bay, a necessity for obtaining both the c.g. offset in the z direction, and an aid in providing the relatively low vertical c.g. position. No further work was done on this concept.



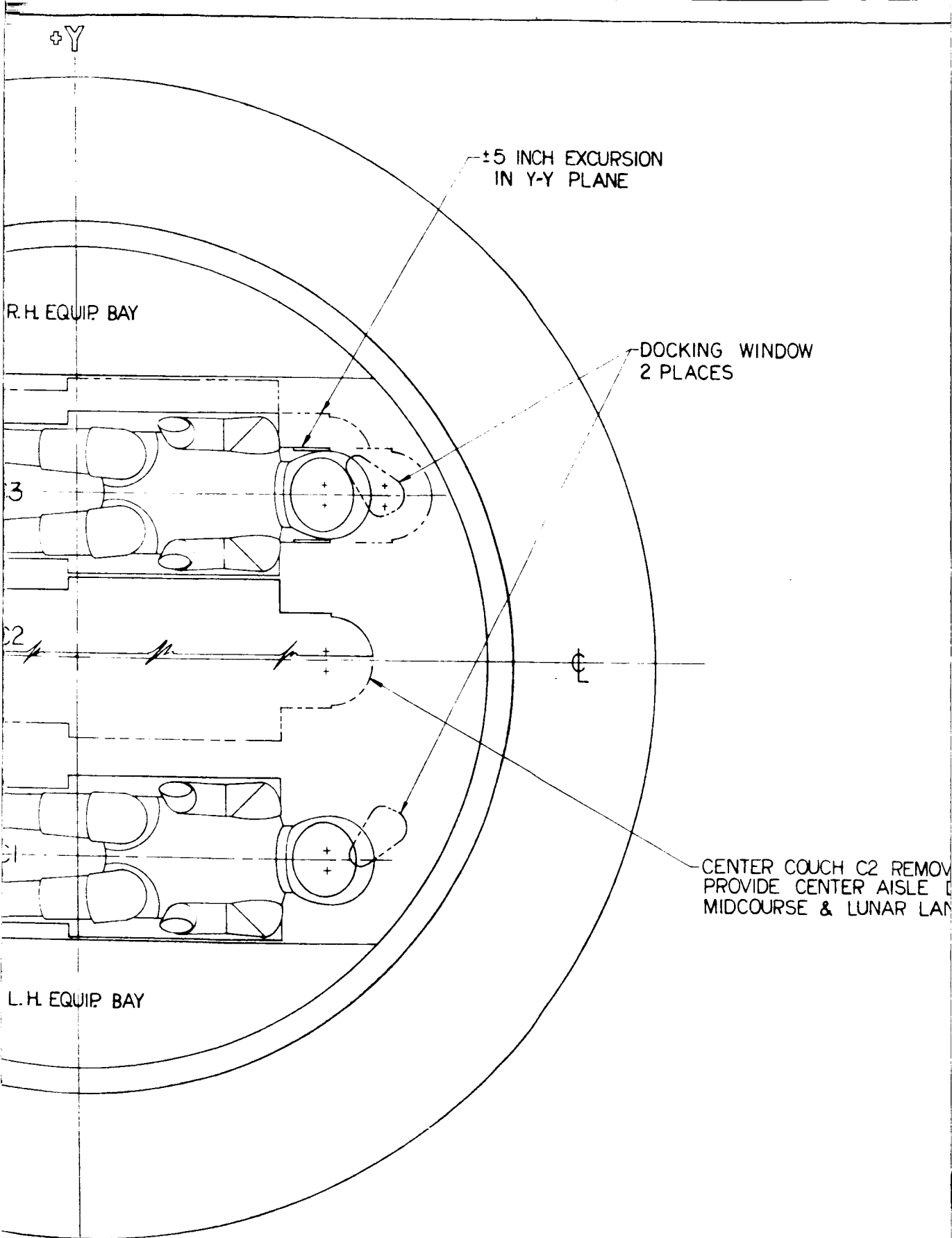
VIEW SHOWING
CONTROL SEAT



LUNAR LANDING
IN POSITION

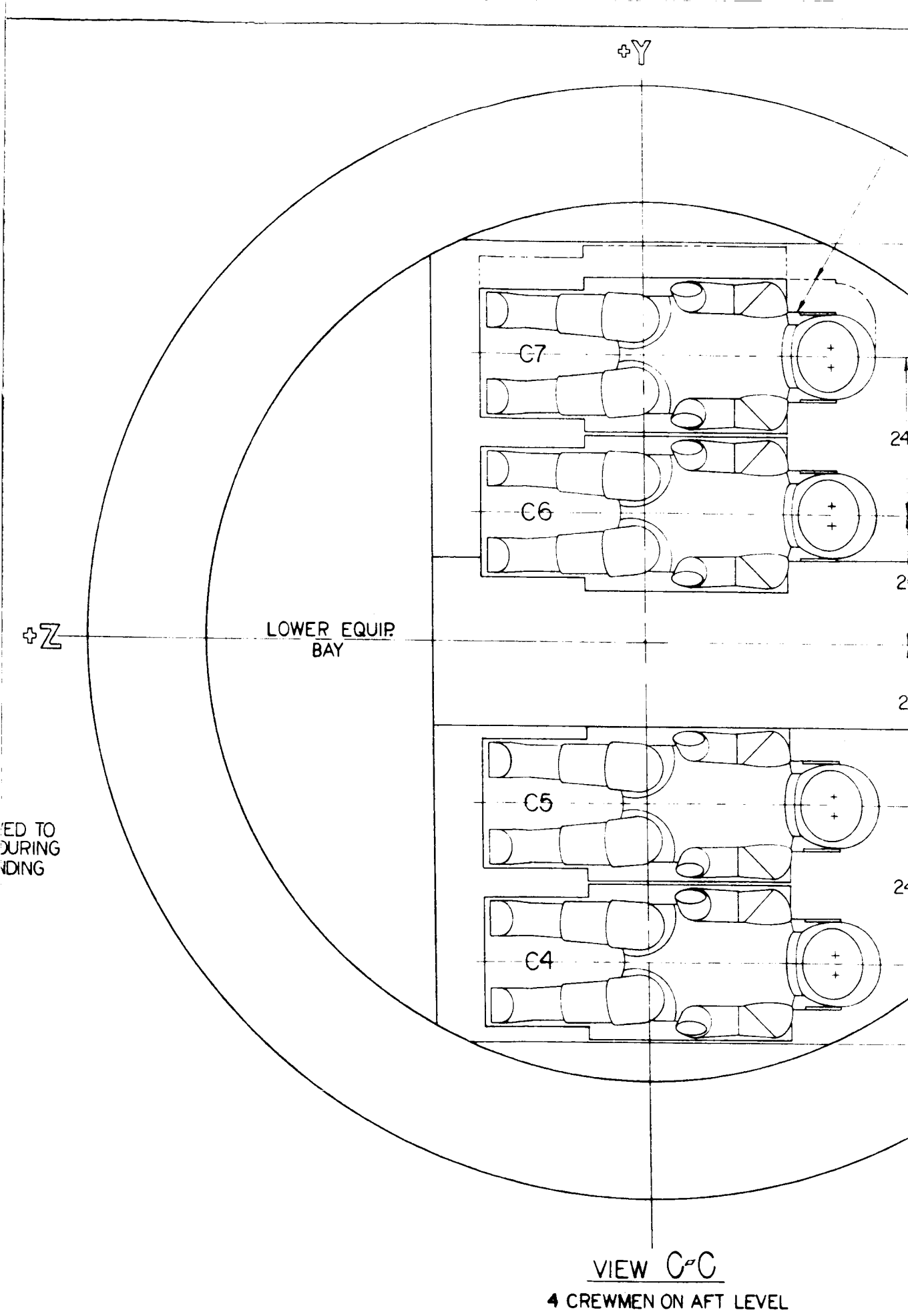


3 CRE



VIEW B-B

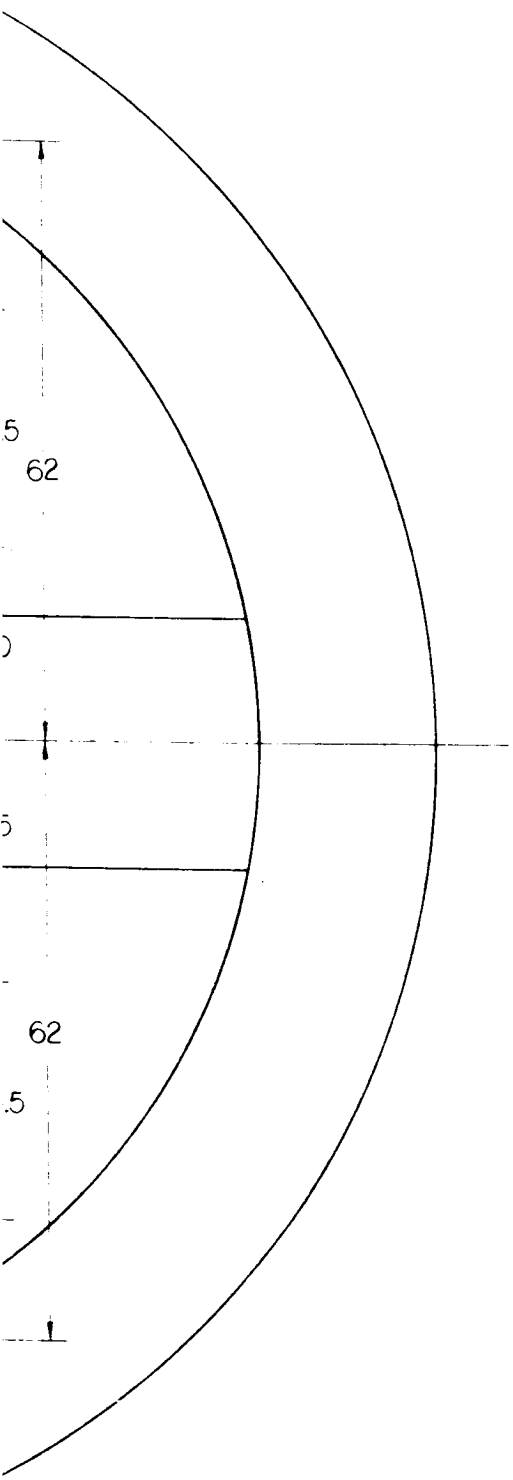
MMEN ON FORWARD LEVEL



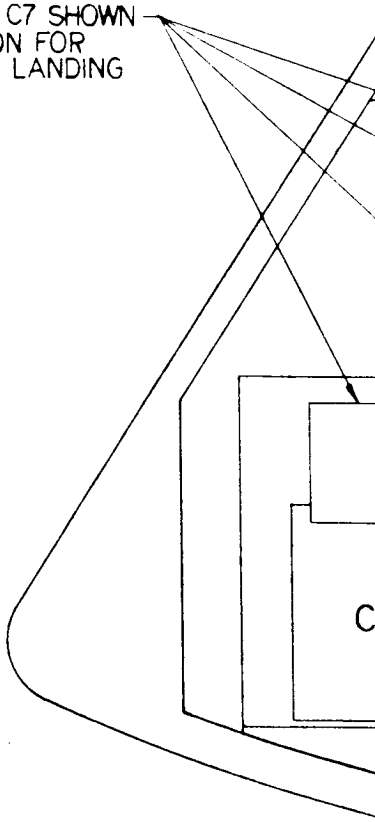
VIEW C-C

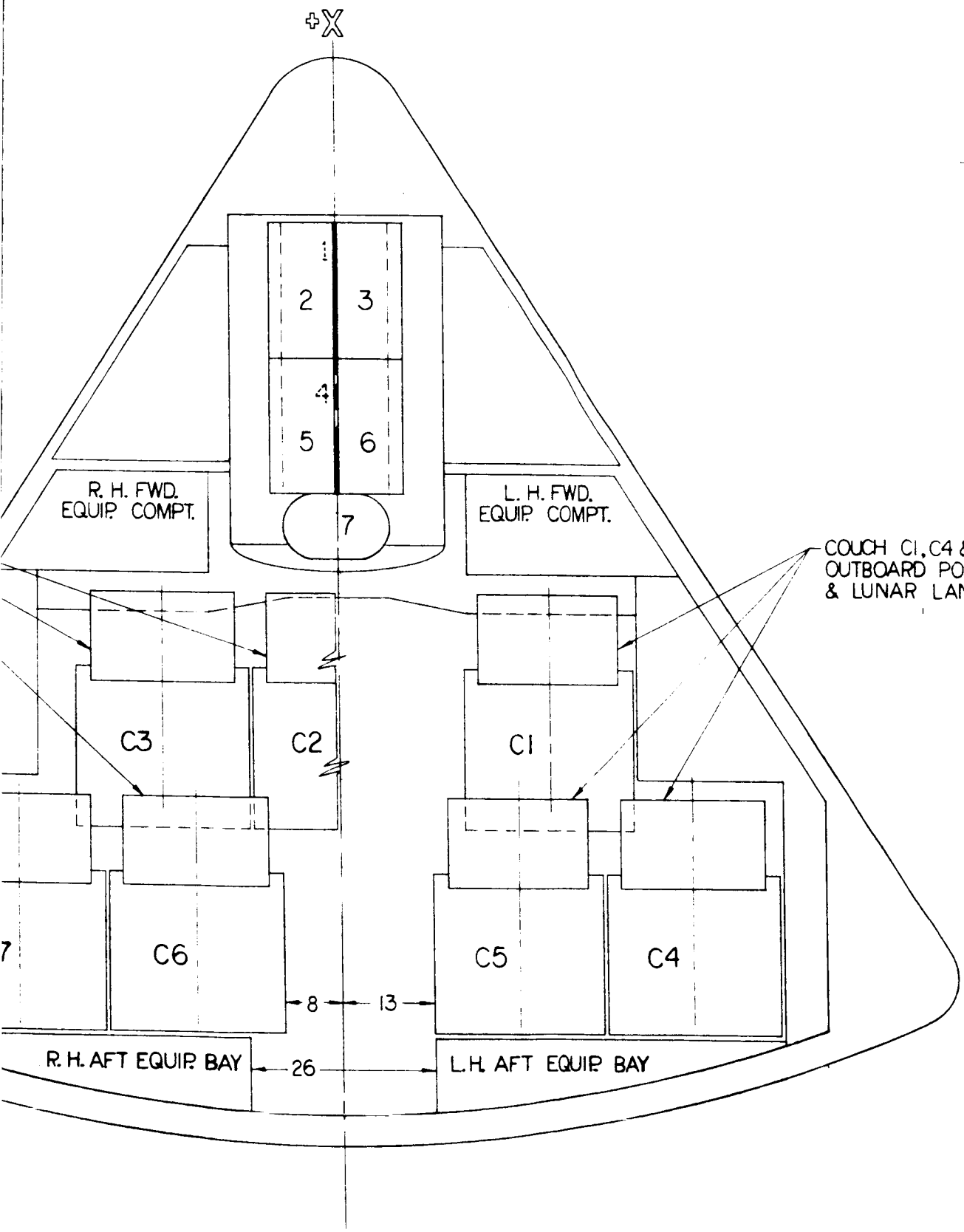
4 CREWMEN ON AFT LEVEL

± 5 INCH EXCURSION
IN Y-Y PLANE

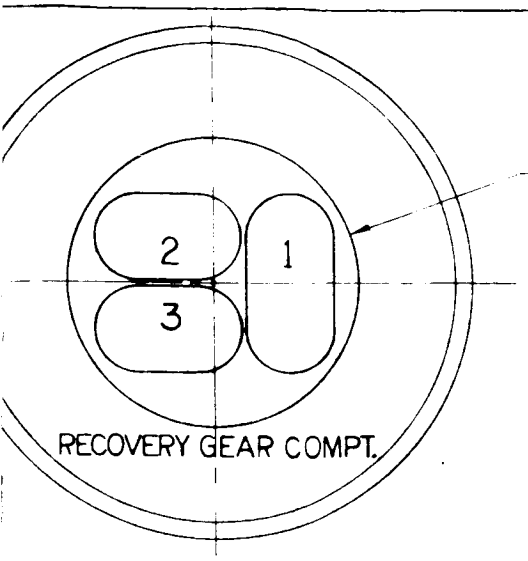


COUCH C2, C3, C6 & C7 SHOWN
IN NORMAL POSITION FOR
LAUNCH & EARTH LANDING




+X
1
2
3
4
5
6
7
R. H. FWD. EQUIP. COMPT.
L. H. FWD. EQUIP. COMPT.
COUCH C1, C4 & C5 OUTBOARD POSITION & LUNAR LANDING
C3
C2
C1
C6
C5
C4
8
13
26
R. H. AFT EQUIP. BAY
L. H. AFT EQUIP. BAY

VIEW A-A



FWD. ACCESS HATCH

SECTION D-D

SHOWN IN
N FOR MIDCOURSE

A

10.3 RAD.

152.0

P. L.S.S. STOWED POSITION
7 REQ'D.

30.0 DIA.

130.0

125.5

17°

23°

D

D

RECOVERY
GEAR
COMPT.

94.0

95.5

INNER M
OUTER M
± 7.5 INCH EXCURSION
IN Z-Z PLANE

84.0

80.0

FWD. DOCKING

POSITION FOR
DOCKING VISI

3.0

B

C

48.6

LOWER
EQUIP. BAY

196.5 RAD.

201.0 RAD.

6.6 RAD.

48.0

4.2

19.6

S.E.P.

+38

95° TYP.

2° TYP.

15.0

4.5

0.0

33

68.0 RAD.

86.35 RAD.

172.7 DIA. STRUCTURE

178.2 DIA. ABLATION M

SCALED-UP APOLLO SHAPE
STRUCTURE SHOWN ONLY

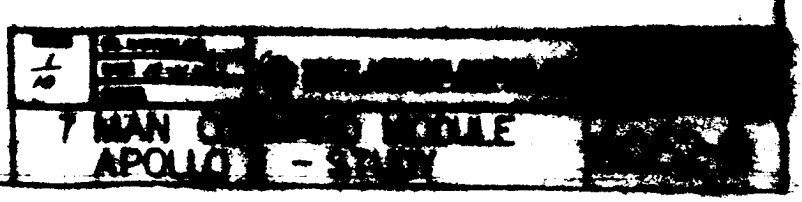
WINDOW

CI & C3
ON

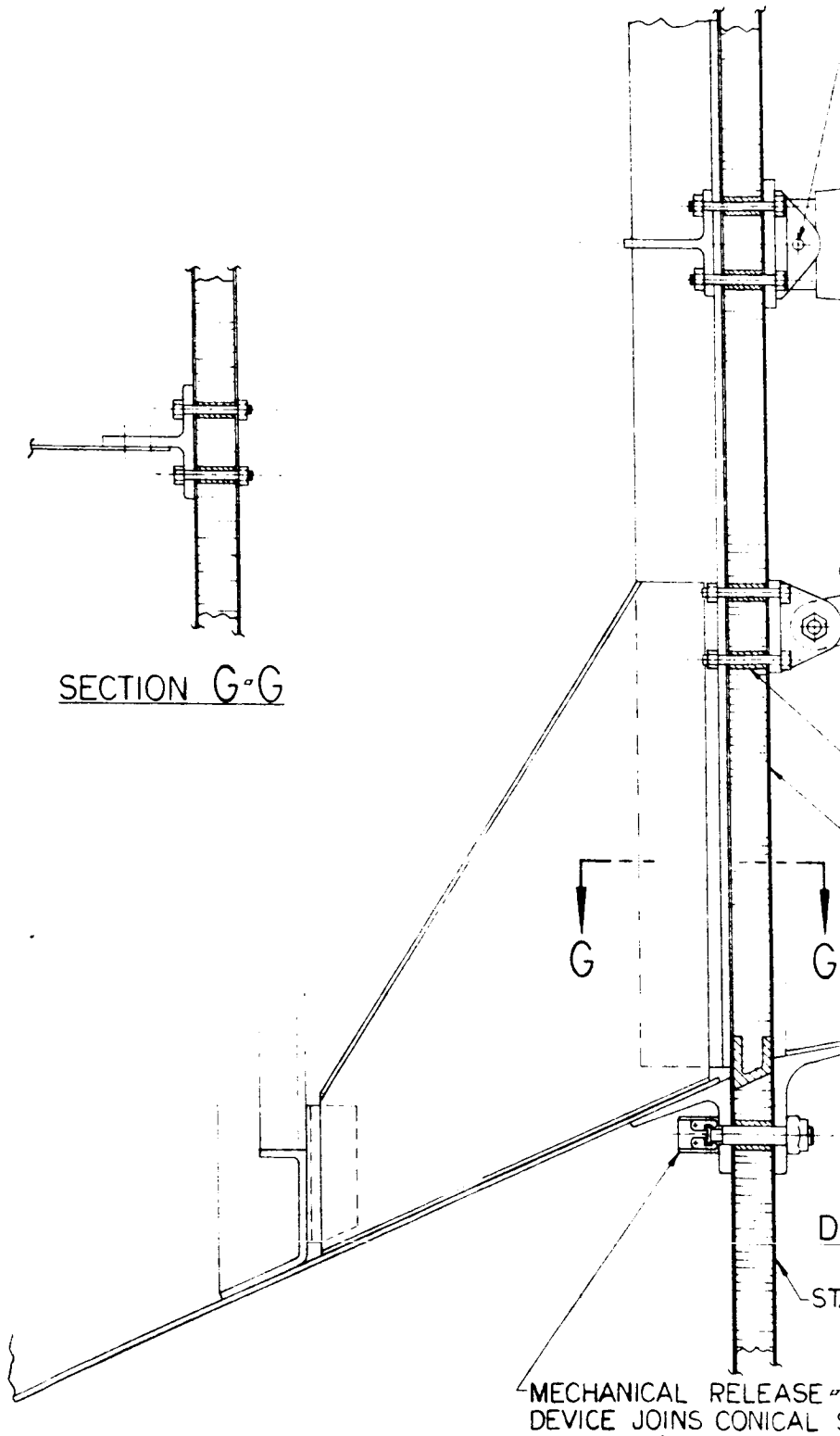
NOTE:

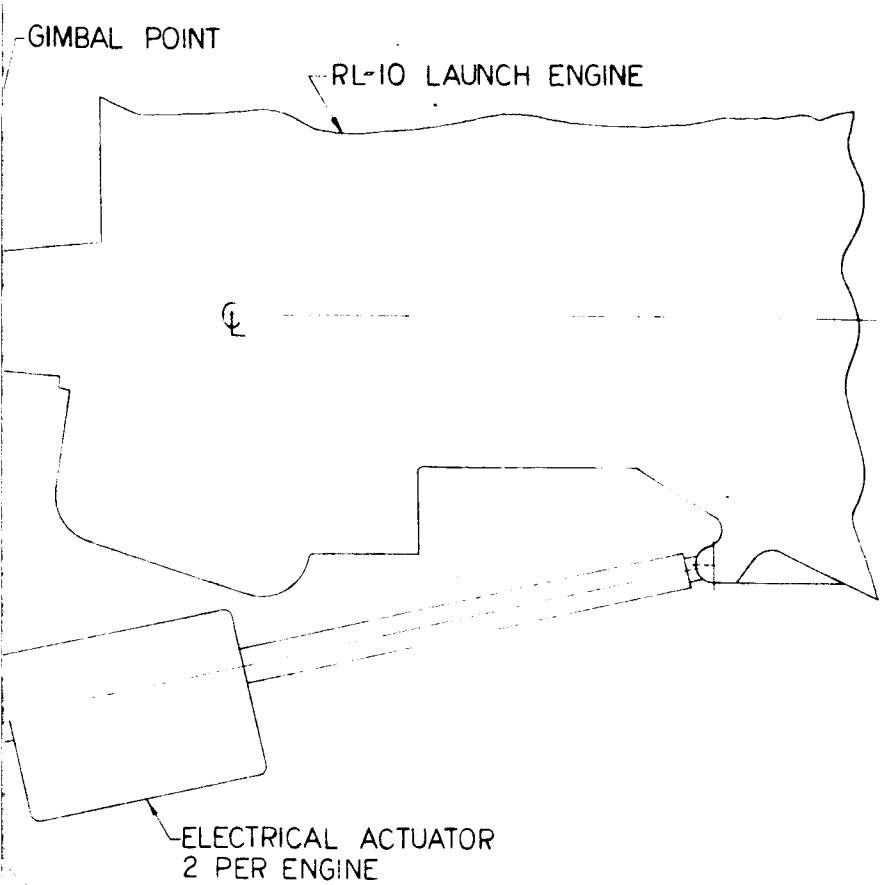
NO VERTICAL SEAT STROKING USED. HEAT SHIELD
EXTENSION OR OTHER VERTICAL IMPACT
ATTENUATION SYSTEM ASSUMED.

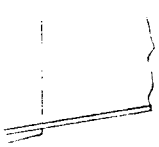
90 PERCENTILE CREWMAN
IN PRESSURIZED SUIT (TYP.)

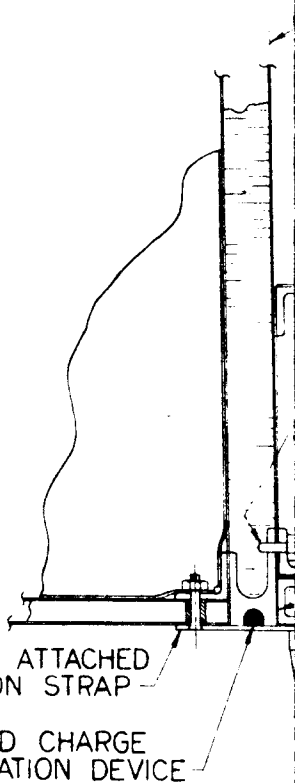


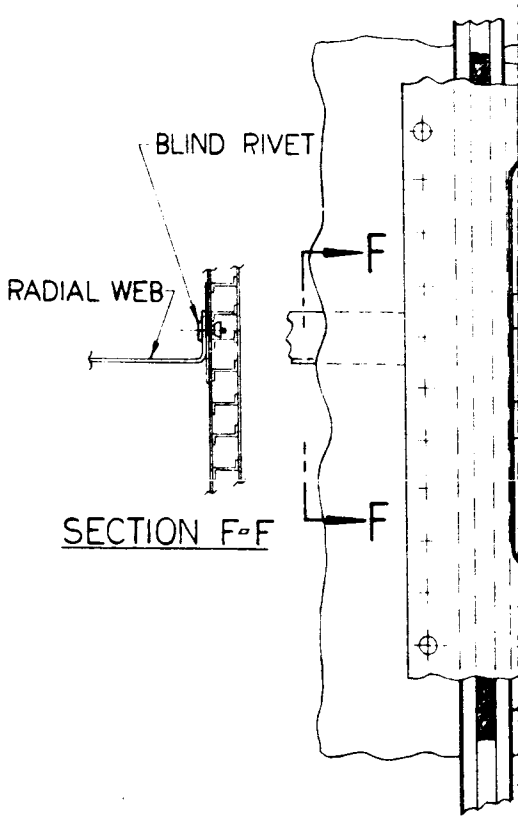
SECTION G-G


G
G
D
ST
MECHANICAL RELEASE
DEVICE JOINS CONICAL
LANDING & LAUNCH ST
BOOST AND MAY BE REL
AFTER END BOOST







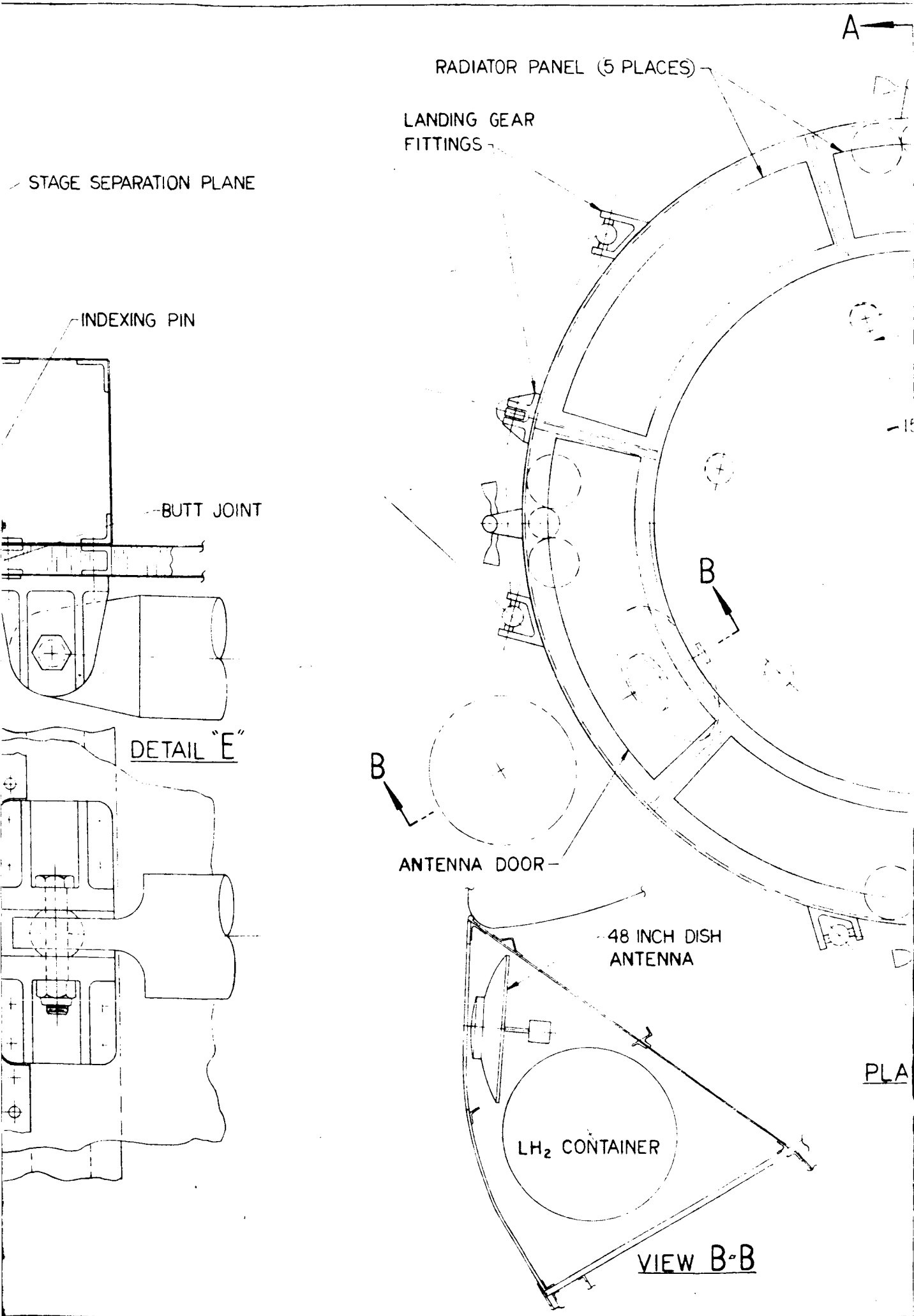


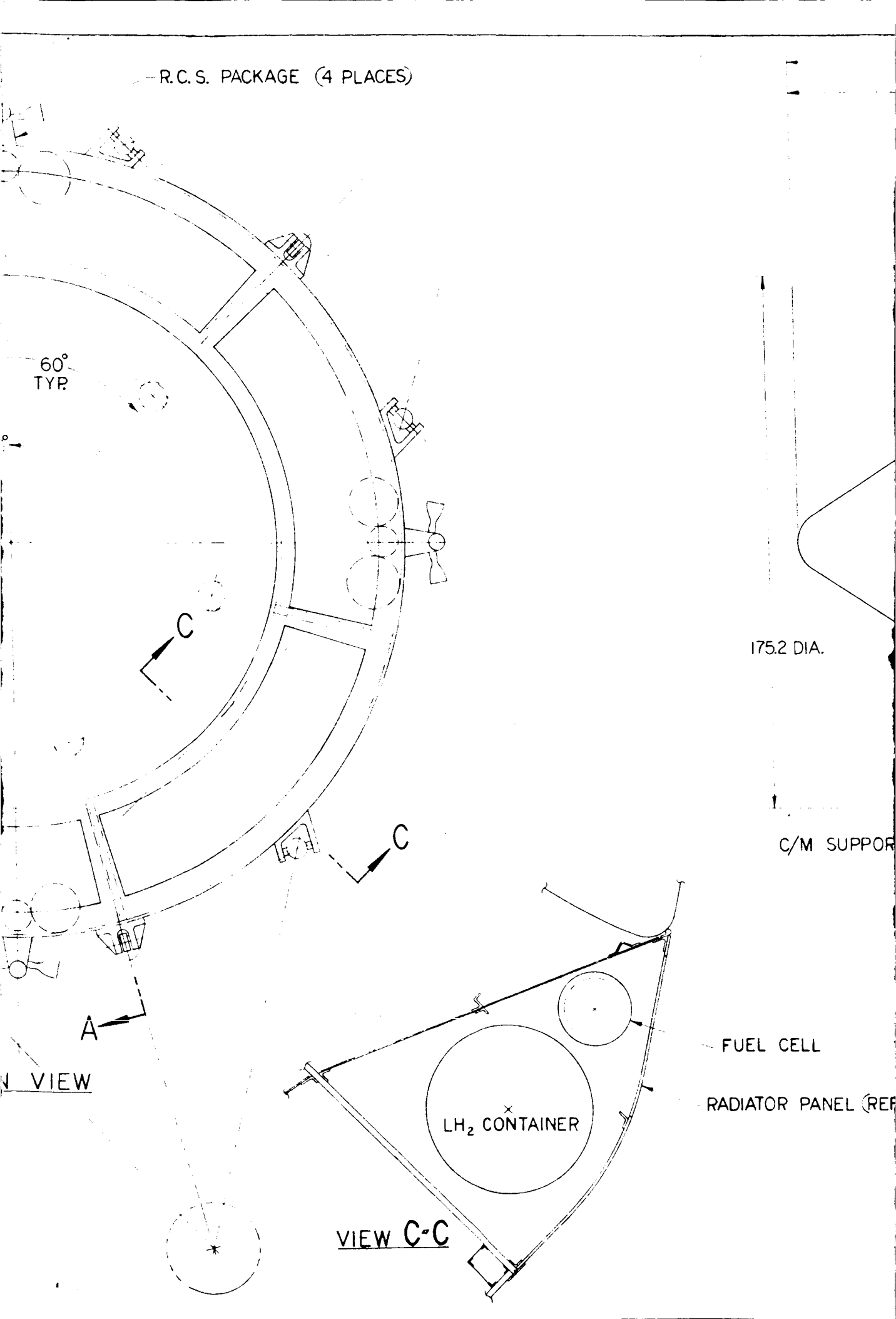
SECTION F-F

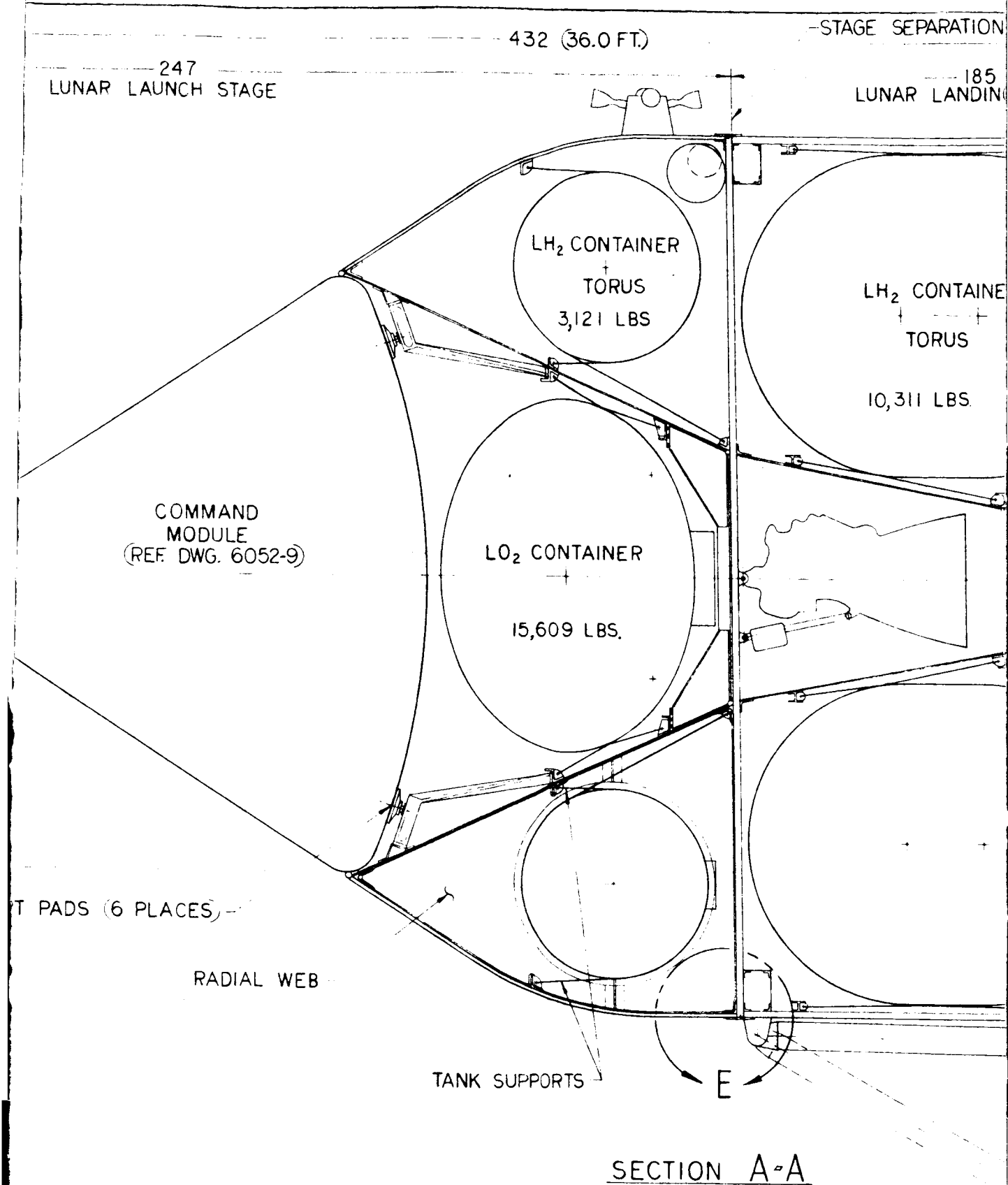
ETAIL OF STAGE INTERFACE

AGE SEPARATION PLANE

6 PLACES
ECTIONS OF
AGES DURING
EASED ANYTIME


A
RADIATOR PANEL (5 PLACES)
LANDING GEAR FITTINGS
STAGE SEPARATION PLANE
INDEXING PIN
BUTT JOINT
B
B
ANTENNA DOOR
48 INCH DISH ANTENNA
DETAIL "E"
LH2 CONTAINER
VIEW B-B


R.C.S. PACKAGE (4 PLACES)
60°
TYP.
C
C
A
175.2 DIA.
C/M SUPPOR
FUEL CELL
RADIATOR PANEL (REF
LH2 CONTAINER
VIEW C-C
VIEW


432 (36.0 FT.)
STAGE SEPARATION
247
LUNAR LAUNCH STAGE
185
LUNAR LANDIN
LH2 CONTAINER
TORUS
3,121 LBS
LH2 CONTAINE
TORUS
10,311 LBS.
COMMAND
MODULE
(REF. DWG. 6052-9)
LO2 CONTAINER
15,609 LBS.
T PADS (6 PLACES)
RADIAL WEB
TANK SUPPORTS
E

SECTION A-A

PLANE

G STAGE

SATURN BOOST VEHICLE

D

LO_2 CONTAINER
TORUS
51,554 LBS.

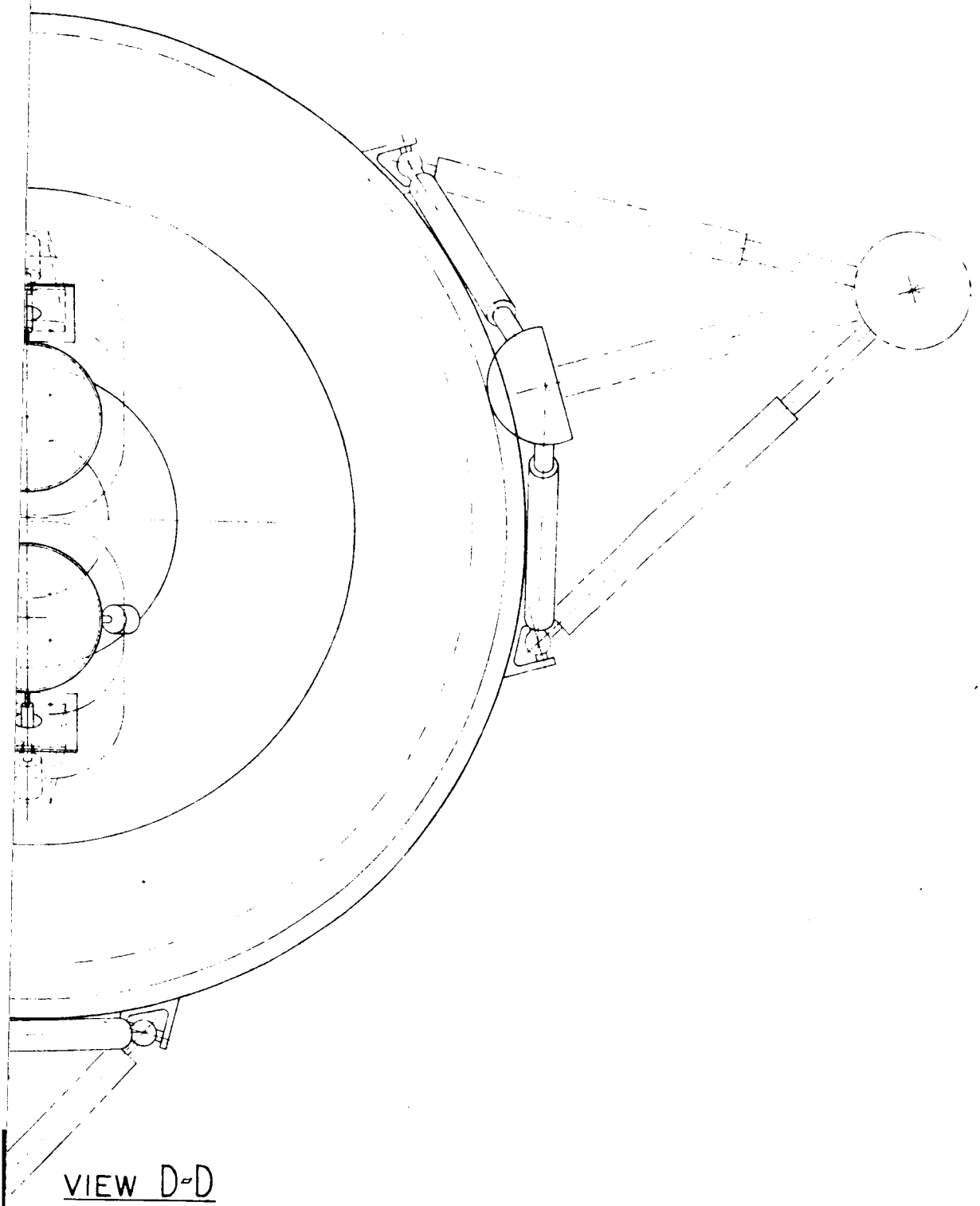
R

260 DIA.

RL-10 ENGINE
(3 PLACES)

CIRCULAR PANEL EJECTED BY
LAUNCH ENGINE BLAST

D



VIEW D-D

SCALE 1/20	BL WOODLIEF	NORTH AMERICAN AVIATION, INC. SPACE and INFORMATION SYSTEMS DIVISION 12214 LAKEWOOD BLVD. DOWNEY, CALIFORNIA
	DATE	
	MODEL	

SERVICE & LUNAR LANDING MODULES
APOLLO II – 7 MAN DIRECT MISSION

6052-10

[redacted]

Electrical Power and Water Management

Summary

Two different types of electrical power systems may be suitable for the seven-man direct lunar Apollo II mission: fuel cells, which provide water for the crew; isotopic-dynamic power systems with no water production. Both systems are compared herein to establish the minimum weight combination for water and power systems. Table 12 summarizes the power requirement profile for a typical mission. Table 13 shows the crew metabolic rates for the same mission.

Table 24 summarizes the weights of several combinations of systems. Column 1 is for a fuel cell system. The fuel cells are sized to provide the quantity of fresh water needed by the crew (11 lb/man-day). The electric power output (approximately 3.2 KW cont.) is, therefore, in excess of the power requirements and wasted.

The total water requirements over the mission amount to 1078 lb; the fuel cells generate 585 lb over the same time. The deficit of 493 lb, or 5.04 lb/man-day, will have to be reclaimed from waste water. Since waste wash water may not be available during the early part of the mission, recovery from the air conditioner and urine will be provided. Such a systems combination is presented in Column 2; it shows clearly the weight advantage over a fuel cell system which is sized for water instead of power demand.

In Column 3, an isotopic-dynamic power system in combination with total water recovery is summarized. This is clearly the lightest combination; it was chosen for the seven-man Apollo II mission. Although isotopic-dynamic power supply is in an early stage of development (fuel cells are readily available), it has to be mentioned that the seven-man vehicle uses an enlarged command module and is also farther away in the future. Furthermore, the seven-man vehicle requires considerable uprating of the Saturn V and weight is at a premium. For these reasons, the choice of the isotopic-dynamic system appears justified and consistent with circumstances present when it will be operational.

Isotopic-Dynamic Power Supply

The lightest suitable power source for the seven-man Apollo II is an isotopic-dynamic system as shown in Table 24. The preliminary design of such a system has actually been undertaken by S&ID's Power Group; additional data was generated in parametric form. Only current technology is employed in the system outlined herein; the components involved have proven high reliability and performance. The use of a single-phase inert gas as the working fluid eliminates many of the problems associated with other fluids; particularly, material problems in turbine erosion and operational problems of zero g. The weight summary is considered conservative, with considerable potential for reduction.

Table 25 lists the system requirements, and Table 26 shows the most im-

TABLE 24

ELECTRIC POWER AND WATER SYSTEMS WEIGHT COMPARISON - 7-MAN APOLLO II

	Fuel Cell Providing All Water for Crew	Fuel Cells Water Recovery from Urine & Air Conditioner	Isotopic-Dynamic Power Source, 5.7 KW Water Total Recovery System
Fuel Cell	640	420	-
Isotopic-Dynamic Power Source	-		950*
Power Conversion & Distribution	432	432	432
Reactants, $H_2 + O_2$	1,260	814	-
Tankage and Support	173	110	-
Radiator and Hardware	74	74	(included above)
Water Stored at Launch	70	140	140
Launch & Entry Cooling	10	10	10
Leakage Water Loss	30	30	30
Water Recovery System + Tanks and Plumbing	-	53	70
Water for Back-packs (42)	168	168	168
Power Output KW	(3.2 KW)	(2.25 KW)	(2.3 KW)
TOTAL SYSTEMS WEIGHT, LB	2,857	2,251	1,800

*PM 147, shielded for 50 Rems/year

portant isotopes applicable for spaceflight power supply.

TABLE 25- APOLLO II POWER SUPPLY REQUIREMENTS

Power Level	2.3 KW continuous
Design Life	
Standby	1 week - 1 year
Operational	14 days (or more)
Reliability	Unspecified
Radiation Level	Unspecified (assumed 50 rems/yr)
Output	28 Volts D.C. 3200 CPS, 3 Phase A.C.
Availability	Before 1972

TABLE 26- ISOTOPES TO BE CONSIDERED

Isotope	Decay Product	Half-life (yrs)	Thermal Power		Useful Mission Life (yrs)
			$\frac{BTU}{CURIE}$ x 10^{-3}	$\frac{BTU}{gm}$ x 10^{-3}	
SR-90	BETA	28	22.2	682	10
PU-238	ALPHA	90	116	1640	10
PM-147	BETA	2.6	1.37	615	2.5
CE-144	BETA-GAMMA	.78	26.7	7,850	1.0
PO-210	ALPHA	.38	109	478,000	.5

System Description

The system consists of the heat source, a combined rotating unit operating in a closed Brayton cycle, taking advantage of a recuperator. The working fluid, based upon results obtained in Reference 2, is selected as argon. Both argon and neon may be considered. Turbines and compressors are several points more efficient when operating on argon rather than neon. This reflects higher cycle efficiencies and a reduced heat load requirement for the isotope; consequently, a reduced isotope supply. However, neon is a better heat transfer fluid than is argon. The temperature limitations of materials in the turnbine determine the maximum cycle temperature. This is taken as 2000 R representing a conservative limit since turbine inlet temperatures of 2100 R in the presence of corrosive combustion products are

routine. A large fraction of the reject heat is recovered by the recuperator leading to a high thermal efficiency. In the operation of the system, the power output of the turbine can be regulated by varying the mass flow through the recuperator by-pass. This varies the efficiency of the cycle and, consequently, the power output.

Start-up

To start the turboelectric power system, it is necessary to bring the turbine up to 30 - 40% of the design speed. With the addition and removal of heat to and from the closed cycle, the unit will become self-sustaining and can accelerate to the design speed of 64,000 rpm.

Throughout the launch and stand-by phases (on the Moon), heat rejection may be accomplished through small high-temperature radiators. This stand-by condition may be integrated with the ECS demands and, also, thermoelectric elements may be incorporated into the high-temperature radiator to provide any stand-by electrical power demand. Other details concerning system design start-up, pre-launch ascent, stand-by, shut-down and/or change-over, safety devices, and control design will be left to a future time owing to the limits of this study and the extent of additional effort required in this area.

System Weight Calculations

To arrive at system efficiency, it is necessary to determine the effective heat sink conditions. A conservative estimate may be yielded by considering a flat panel parallel to the lunar surface radiating to space with maximum incident at lunar noon receiving the full solar constant, i.e.,

$$T_s^4 = \frac{\alpha_s S}{\sigma E} = \frac{0.25\ (443)}{0.1713 \times 10^{-8}\ (0.85)}$$

$$T_s = (760 \times 10^8)^{\frac{1}{4}} = 530\ R$$

Calculational methods are taken from Reference 1. Reference T-S diagram of the cycle, Figures 40 and 41.

Sample Calculation:

$T_1 = 2000R$

$T_4 = 575R$

$$\frac{T_5}{T_4} = \left(\frac{P_2}{P_1}\right)^{\gamma} \quad (2.89)^{0.4} = 1.53 \qquad (1)$$

$T_5 = 880$

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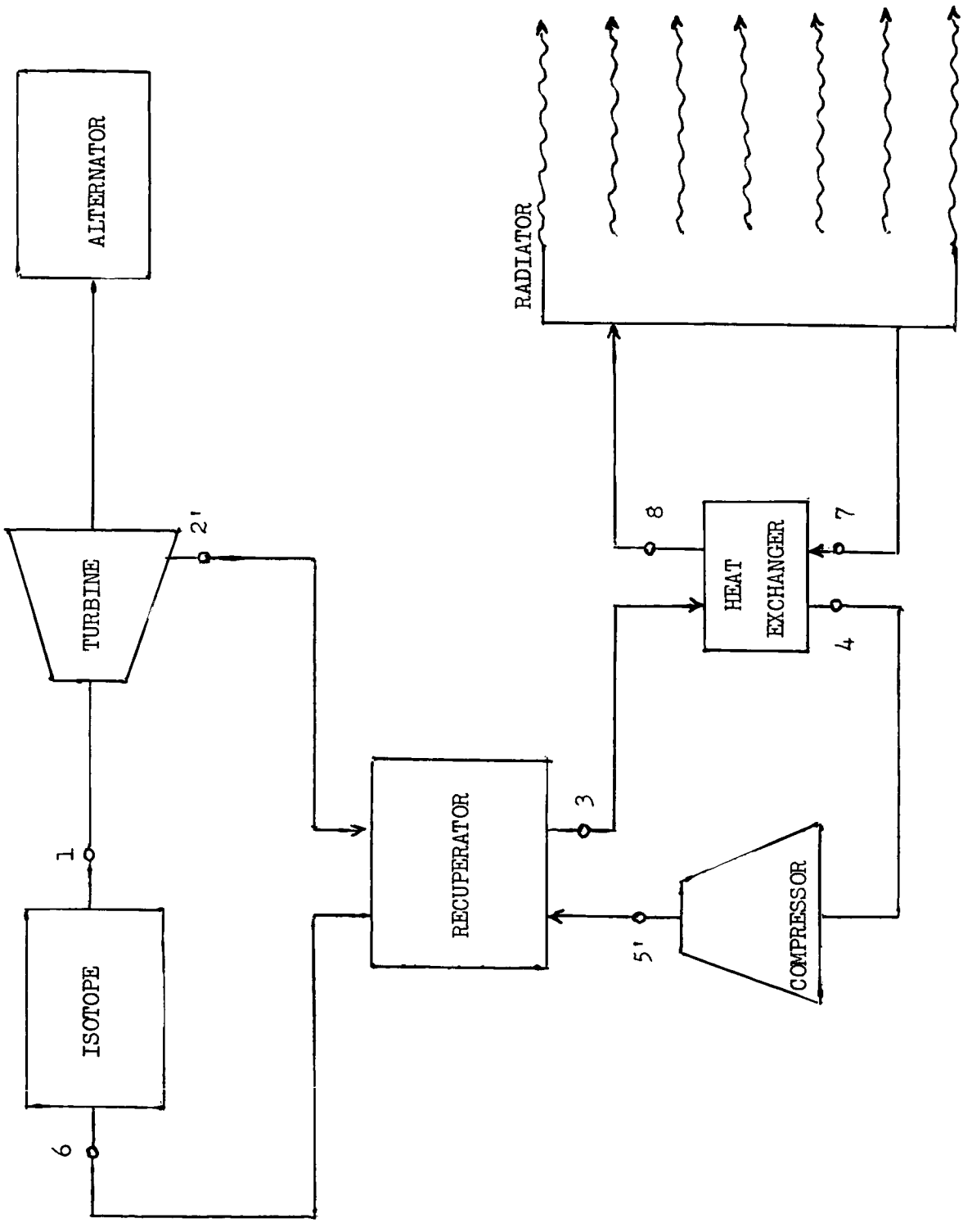


FIGURE 40 - ISOTOPIC-DYNAMIC SCHEMATIC

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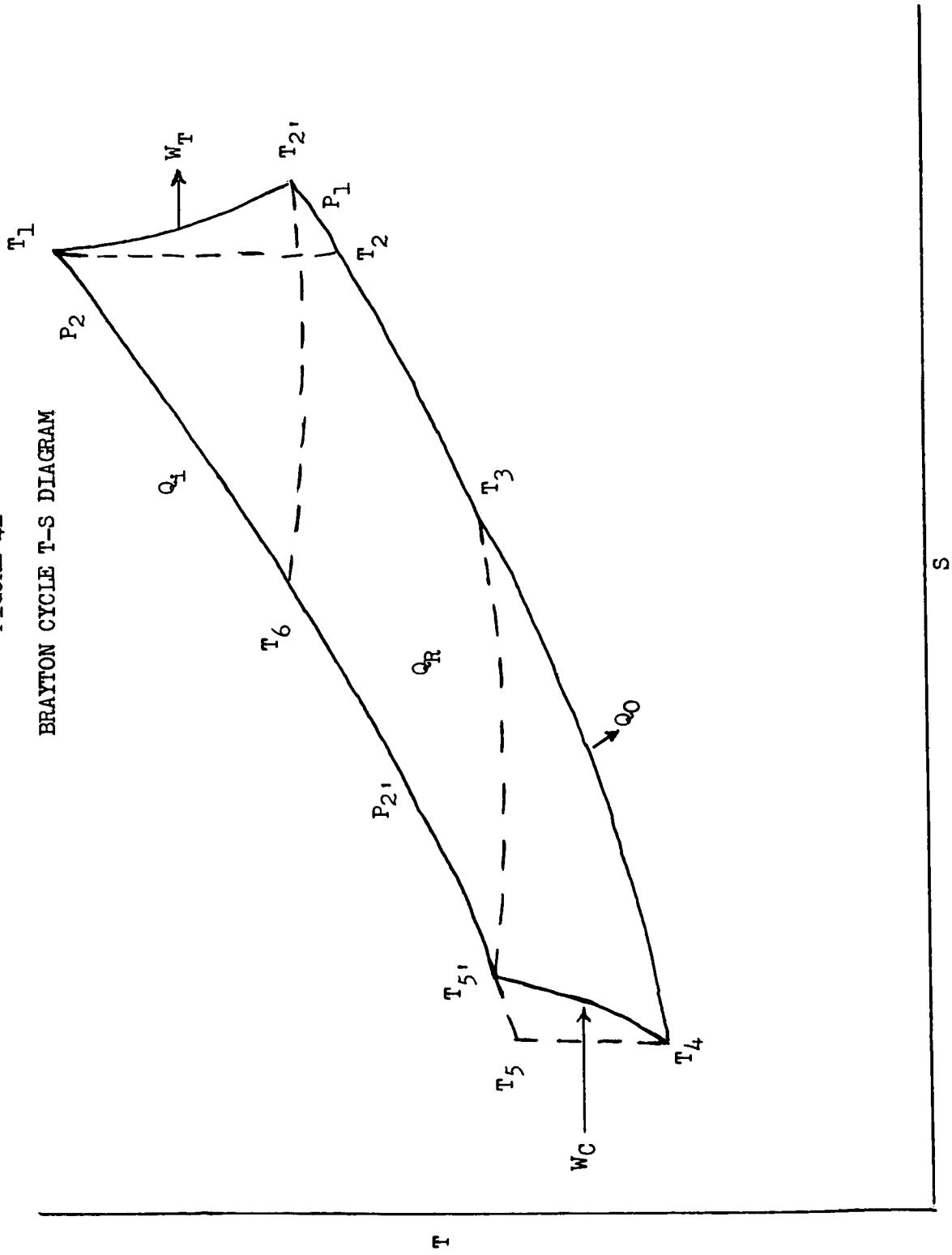


FIGURE 41

BRAYTON CYCLE T-S DIAGRAM

[redacted]

$$T_5^1 - T_4 = \frac{T_5 - T_4}{E_c} = \frac{305}{0.8} = 381 \quad (2)$$

$$T_5^1 = 956$$

$$\frac{T1}{T2} = \left(\frac{P1}{P2}\right)^{\alpha} = (2.5)^{0.4} = 1.44 \quad (3)$$

$$T2 = 1385$$

$$(T_1 - T_2^1) = E_t - (T_1 - T_2) = 0.8\ (615) = 492 \quad (4)$$

$$T_2^1 = 1508$$

$$T_2^1 - T_3 = E_r^1\ (T_2^1 - T_5^1) = 0.9(1508 - 956) = 496 \quad (5)$$

$$T_3 = 1012$$

$$T_6 - T_5^1 = E_r^1(T2^1 - T5^1) = 496 \quad (6)$$

$$T_6 = 1452$$

$$W_t - W_c = \frac{1.0 \times 3415}{0.68} = 3880 \quad (7)$$

$$\eta = \frac{(T1 - T2^1) - (T_5^1 - T_4)}{T_1 - T_6} = 0.203 \quad (8)$$

$$Qi = \frac{Wt - Wc}{\eta} = \frac{3880}{0.203} = 19{,}100 \quad (9)$$

$$Qo = Qi\ (1 - \eta) = 15{,}200 \quad (10)$$

$$Qr = Qo\ \frac{T2^1 - T3}{T_3 - T_4} = 17{,}300 \quad (11)$$

$$m = \frac{Wt - Wc}{Cp\left[(T_1 - T_2^1) - (T5^1 - T_4)\right]} = 281 \quad (12)$$

$$T_3 - T_7 = \frac{Qo}{m\ Cp\ Ex} \quad \frac{15200}{280\ (.24)\ (.90)} = 484 \quad (13)$$

$$T_7 = 528$$

$$\Delta T = 350$$

$$T_8 = 878$$

[redacted]

$$Y = \frac{3(T_8 - T_7)}{\left[\frac{1}{T_7^3} - \frac{1}{T_8^3}\right]} = \frac{3(350)}{\left[\frac{1}{(528)^3} - \frac{1}{(878)^3}\right]} = 1985 \times 10^8 \quad (14)$$

$$A^1 = \frac{Q_o}{\epsilon \sigma Y} = \frac{15{,}200}{(.85)\ (.171 \times 10^{-8})\ (1985 \times 10^8)} \quad (15)$$

$$A^1 = 52.7$$

Radiator

The required radiator area is calculated by determining the effective area required, assuming entire surface is at the fin base temperature, and then calculating the radiator effectiveness under the actual fin temperature distribution and sink temperature conditions. Upon the basis of previous study, a T of 350 R (across the radiator) is selected, permitting temperature limits to be met, yet maintaining low vapor pressure and low viscosity. For the radiator calculations, an existing computer program was modified to incorporate calculational procedures and fluids as outlined in References 1 and 2. This program calculates and optimizes the design parameters for a single-phase fluid. The results of the calculations are summarized in Figures 42, 43, and 44.

TABLE 27

RADIATOR FLUIDS (Non-flammable Fluids Only)

Fluid	Selected Range	Limits and Explanations
Coolanol 45	-7F to +438F	Lower limit due to increased viscosity, i.e., = 0.07 lb/ft sec at 0F
Freon 21	-91F to +259F	Upper limit due to high vapor pressure, i.e., in excess of 300 psia at 275F
FC 75	-58F to +292F	Lower limit due to increased viscosity, i.e., .10 lb/ft sec at -70F Upper limit due to a low thermal conductivity, i.e., K = 0.0112 $\frac{BTU}{hr(ft)F}$

Power Conversion

The generator is designed for a nominal 5 - 6 volts D.C. and, consequently, a D.C. to D.C. boost converter is designed and manufactured to provide the required voltage. This device inverts the D.C. current to A.C., puts it through a step-up transformer, and rectifies back to D.C. The normal tolerance permitted is $\pm$ 5% with a maximum of 1% ripple. The maxi-

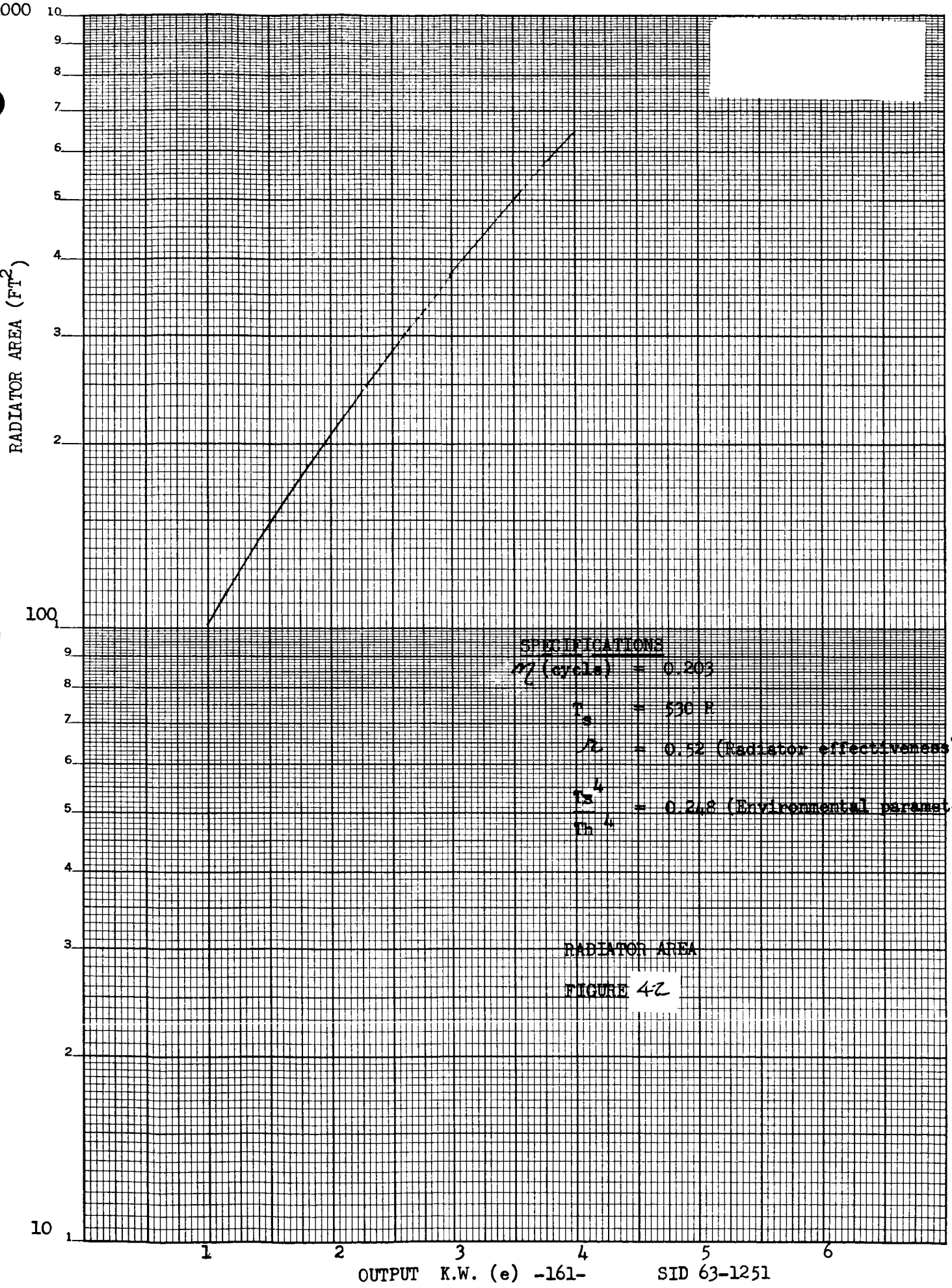
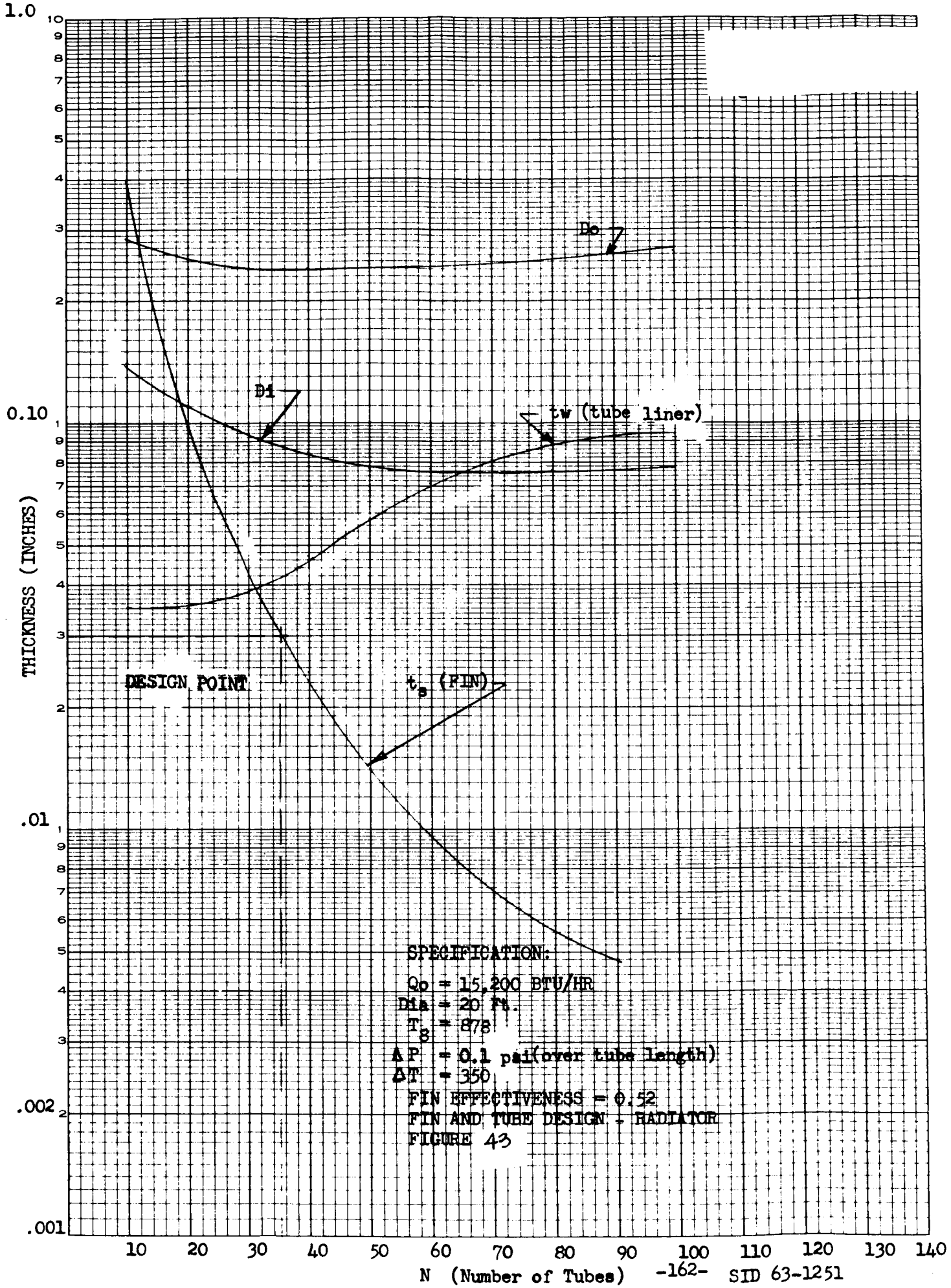
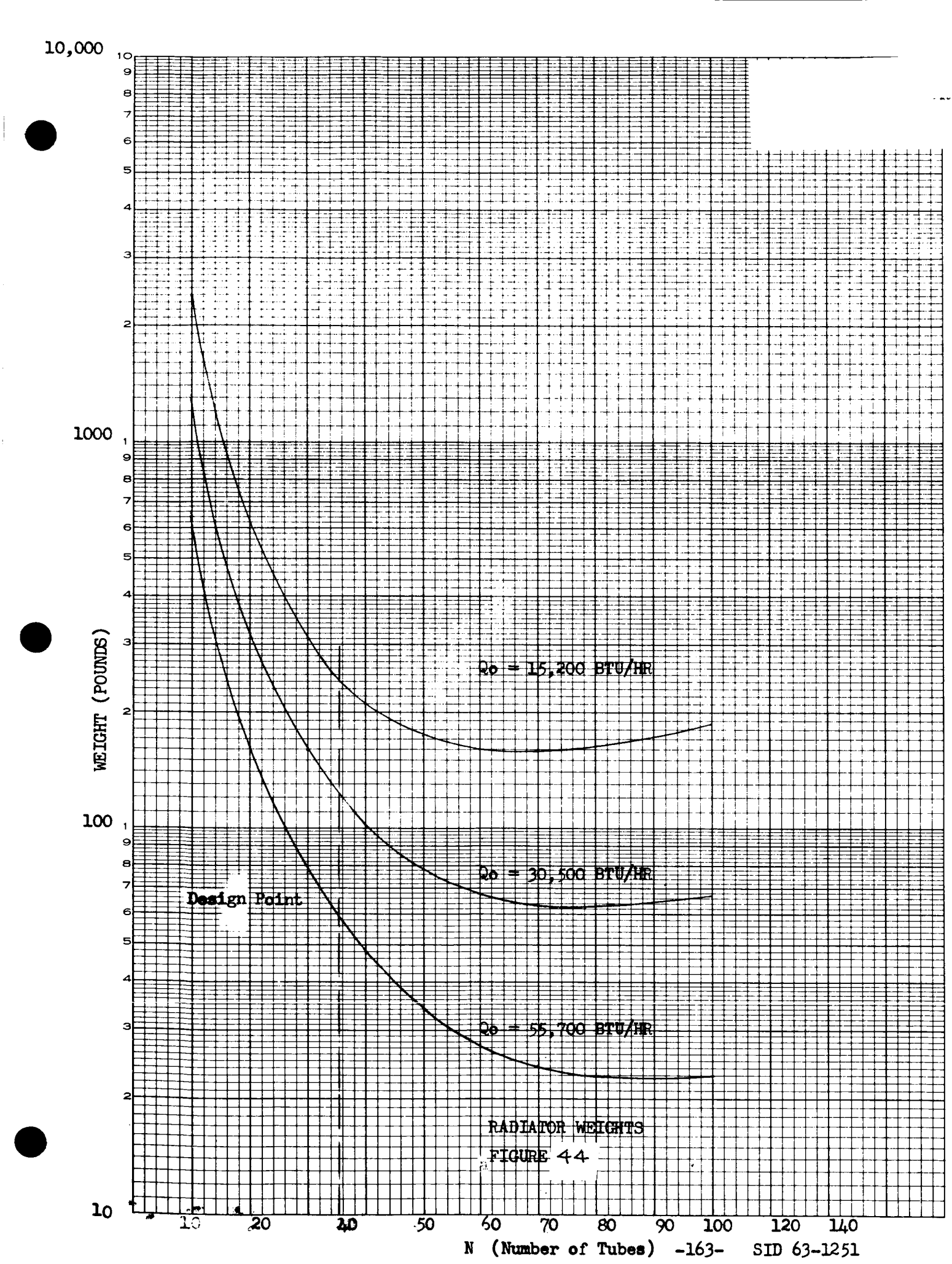

1000
100
10
1
RADIATOR AREA (FT^2)
1
2
3
4
5
6
OUTPUT K.W. (e)
SPECIFICATIONS
η_c (cycle) = 0.203
T_s = 530 R
$\mathcal{R}$ = 0.52 (Radiator effectiveness)
T_s^4 / T_h^4 = 0.248 (Environmental parameter)
RADIATOR AREA

FIGURE 42


1.0
0.10
.01
.002
.001
THICKNESS (INCHES)
Do
Di
tw (tube liner)
DESIGN POINT
t_s (FIN)
SPECIFICATION:
Qo = 15,200 BTU/HR
Dia = 20 Ft.
T_g = 878
ΔP = 0.1 psi(over tube length)
ΔT = 350
FIN EFFECTIVENESS = 0.52
FIN AND TUBE DESIGN - RADIATOR
FIGURE 43
10 20 30 40 50 60 70 80 90 100 110 120 130 140
N (Number of Tubes)



RADIATOR WEIGHTS

FIGURE 44

mum operational efficiency of these converters is presently ~ 80%; however, it is anticipated that efficiencies of 90% or more could be attained. This generator will be integrated with a secondary battery to handle both the peak loads and the continuous loads.

The three potential secondary battery systems used in space applications are: (1) nickel-cadmium, (2) silver-cadmium, and (3) silver-zinc.

Nickel-Cadmium: The nickel-cadmium battery has been the most widely used battery for space applications, and is the only secondary battery that has a proven hermetical seal for vacuum operation. Although the nickel-cadmium battery possesses the lowest initial energy-to-weight ratio (watt-hr/lb), it has, by far, proven its reusability for thousands of cycles.

Using a 1.0 volt end point cutoff voltage decreasing temperature will lower capacity to 80% at 0F and to 40% at -40F. At temperatures higher than 100F, the separators presently used will tend to dissolve in the potassium hydroxide electrolyte.

At high discharge rates for short durations, the energy-to-weight ratios will decrease to approximately 12 watt-hr/lb at the one-hour rate (80F) and to 9 watt-hr/lb at the one-half hour rate (80F). Cycle lives for nickel-cadmium batteries greater than 10,000 cycles have been reported at a 10% depth of discharge (80F) and lives of up to 6000 cycles at a 35% depth.

The charging voltage need only be in the order of 33-34 volts, with overcharge rates varying from C/6 to C/10 (C is the rated capacity of the battery) being reported.

Silver-Cadmium: The recent development of silver-cadmium batteries for space applications has increased the initial energy-to-weight ratio of a secondary system to approximately 20 watt-hr/lb at the one-hour rate (80F) and to 15 watt-hr/lb at the one-half hour rate. The effect of temperature is comparable to that of the nickel-cadmium.

At present, a well-proven seal has not been developed for the Ag Cd cell and, thereby, has reduced the cyclic life of Ag Cd batteries to 2,500 cycles at a discharge depth of 20% and an overcharge of C/20. The Ag Cd system would require in the range of 38-40 volts for recharge.

Silver-Zinc: Secondary silver-zinc batteries possess high energy-to-weight ratios of 30 to 50 watt-hr/lb for relatively short cyclic lives (below 150 cycle and 10% depth). Temperatures will reduce the capacity of a Ag Zn battery to 40% at 0F (1 hour rate) and to 5% at -40F.

At present, a hermetic seal has not been developed for the secondary Ag Zn cell, thereby, hindering its application for long space applications.

As in the case of the Ag Cd cell, the Ag Zn battery would require a boost regulator to meet a recharge potential of approximately 38-40 volts.

TABLE 28 - WEIGHT SUMMARY OF COMPONENTS

η	Output KW (e)	Recuperator			Heat Exchanger			Turbo-Comp-Alt (2)	Controls (2)	Battery (2)	Structure	Insulation, Ducting
		Qr (BTU/Hr)	Wt (lbs)	Vol (ft^3)	Qo (BTU/Hr)	Wt (lbs)	Vol (ft^3)					
0.203	1	17,300	35	0.3	15,200	20	0.21	50	32	70	25	30
	2	34,760	110	1.3	30,500	35	0.35	50	32	70	25	30
	3	63,400	315	2.8	55,700	80	0.81	50	32	70	25	30

TABLE 29 - WEIGHT SUMMARY HEAT SOURCE

η	Output KW (e)	Sr-90		Pu-238		Po-210		Pm-147	
		5 rem/yr (lbs)	50 rem/yr (lbs)	5 rem/yr (lbs)	50 rem/yr (lbs)	5 rem/yr (lbs)	50 rem/yr (lbs)	5 rem/yr (lbs)	50 rem/yr (lbs)
0.203	1	2000	1250	1300	390	210	205	225	170
	2	3500	2360	2050	710	460	305	460	300
	3	4700	3350	2800	1020	740	350	730	415

[redacted]

TABLE 30

WEIGHT SUMMARY-RADIATOR

Specifications

Tube and Fin type radiating from one side (aluminum)

$T_g = 878$

$T = 350$

$P = 0.1$ psi

$= 0.52$

$D = 20$ ft.

$= 0.85$

$t_s = 0.30$ (thickness of fin)

Fluid = OS 45

P(0) one year = 0.99 (meteoroid protection)

Qo (BTU/hr)	L (ft.)	W (#/hr)	T_s	$\frac{T_s^4}{Th^4}$	A^1 (ft^2)	A (ft^2)	Wt (lbs)	Header Ducts (lbs)
15,200	1.617	82.97	530	0.248	52.7	101.5	59.7	19.0
30,500	3.24	166.3	530	0.248	106	204	120	33
55,700	5.92	303.9	530	0.248	193.5	372	240	39

Radiation Level

Although the life span of the vehicle may be up to one year, duty times for the men will probably not exceed a period of four months. A dose rate of about 7.5 m rem/hr (5 rem/mo) has been taken as a design value for the manned spacecraft (Reference 9). The isotopic shielding weights are an appreciable portion of the total system weight and the selection of a reasonable radiation allowable level must be made. Advantage must also be taken of attenuation by structures and equipment and proximity to the isotope.

[redacted]

The application of normal industrial worker standards would make the shielding weight prohibitive; other limits must be established. A philosophical decision by NASA authorities as to maximum levels to be tolerated with hope of mission success and allotment to nuclear systems of this tolerated level is required before a positive evaluation of the isotope system can be made.

Summary

The Isotopic Dynamic System appears very attractive in many ways. Of the long-life isotopes, both Pu-238 and Pm-147 result in extremely low weight systems. Of these two isotopes, Pm-147 is a fission product, relatively inexpensive; however, it is required in large quantities since it has a very low power density. Ultimately, it will be one of the most available isotopes; but, at the present time, only Sr-90 has assured availability in large quantities. The use of Sr-90 results in a heavy weight penalty owing to its shield requirements. The level of allowable radiation has not been settled. For illustrative purposes, 50 rem/yr was selected in order to present representative weights (Figure 45). This level will assure <7 mrem/hr over the expected mission (manned) periods.

In addition to the problem of isotope availability, it is estimated that three years' final development time would be required for the Brayton cycle conversion equipment.

The power output for the Pm-147 system to be employed is 2.3 KW continuous. The weight of the complete power source is 950 lb (see Figure 45); the weight of the electrical system, including power conversion and distribution, is 1382 lb (see Tables 28 and 29).

Environmental Control and Life Support Systems

The same 5 psia pure oxygen atmosphere as for the five-man Apollo II was selected for the seven-man vehicle. Carbon dioxide removal will be accomplished by a molecular sieve. A typical schematic of such a unit is shown in Figure 46. It consists of two parallel circuits, each containing a molecular sieve and a silica gel bed. Contaminated air from the compartment enters the unit, flows across the silica gel bed where water vapor is removed prior to its passing through the molecular sieve where the CO_2 is removed. In the parallel circuit within the unit, the molecular sieve which has become saturated with CO_2 is regenerated by exposing it to the vacuum of space. This will cause a small loss of compartment air. The silica gel bed in the parallel circuit is regenerated by driving of the water absorbed with an electric heater. The water vapor re-enters the airstream downstream of the molecular sieve to be removed later in the humidity control system.

Water Recovery

Tables 30, 31, and 32 list the water recovery data on a daily basis for a total water recovery system such as to be employed in the seven-man Apollo II.

[redacted]

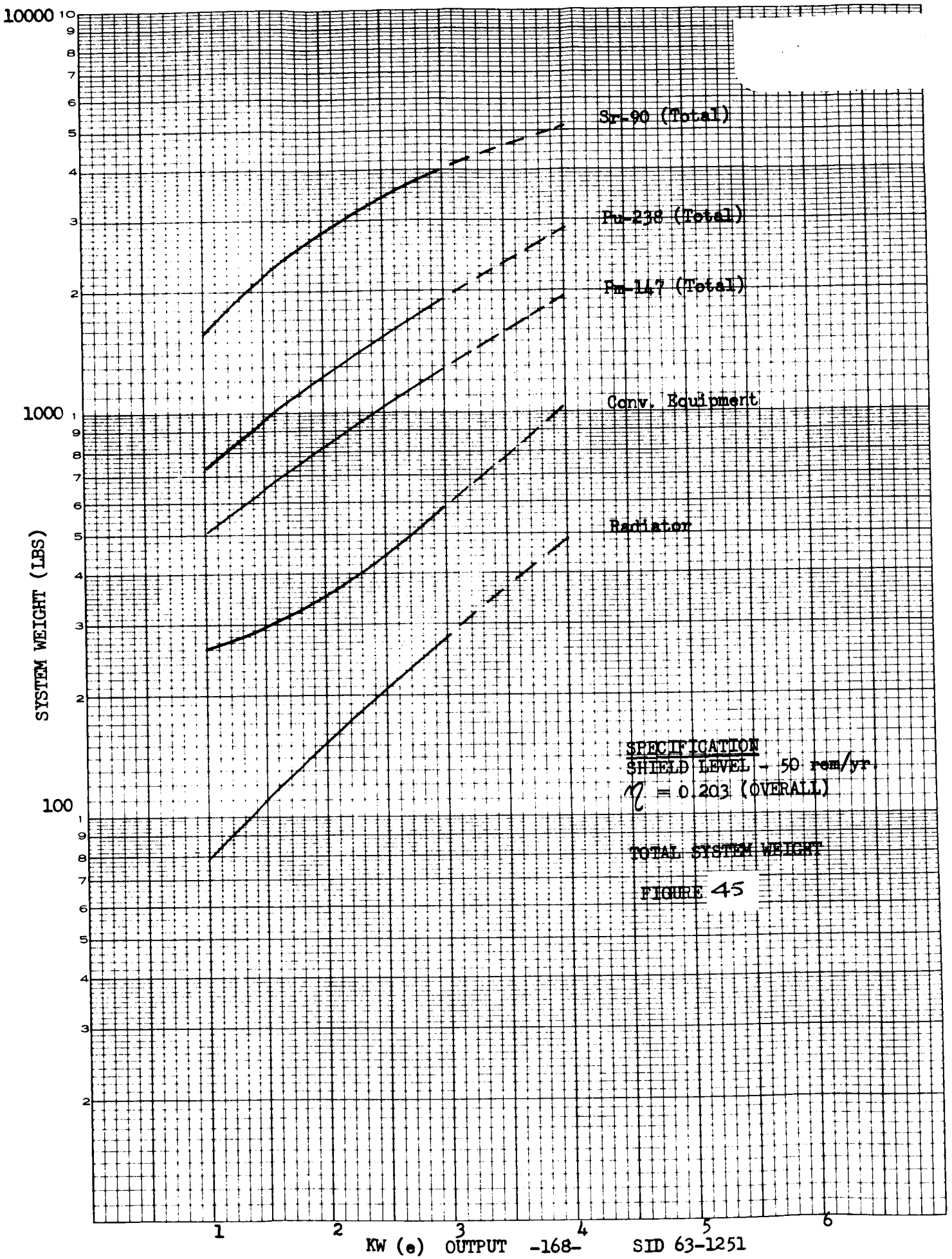


FIGURE 45

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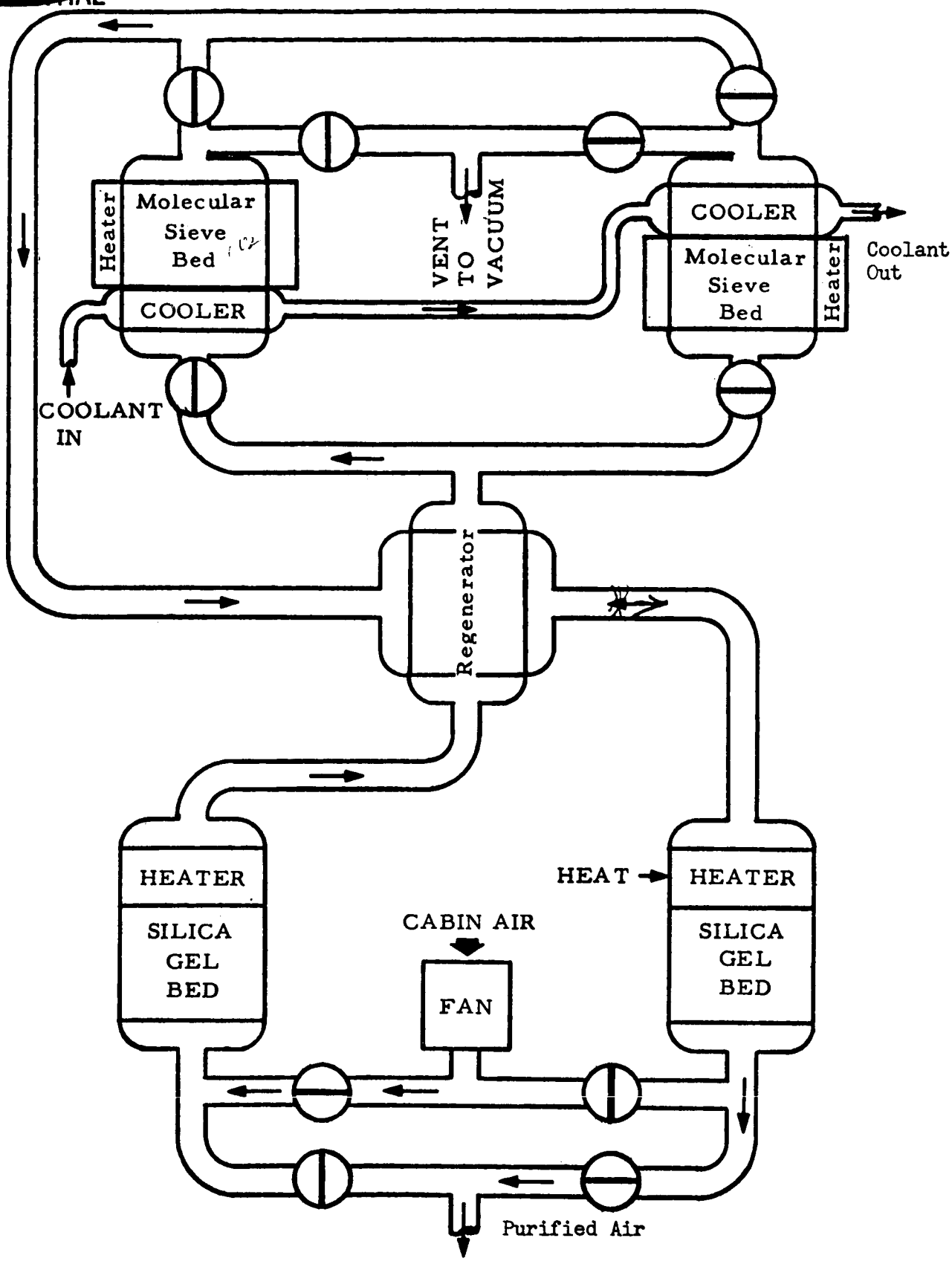


Figure 46 Typical Molecular Sieve Regenerative CO_2 Removal System

Figure 47 shows a schematic of the water reclamation system.

Table 33 summarizes the weights, volumes, operating cycles, and power consumption on a daily basis for the seven-man crew.

Several processes to reclaim the water are feasible; the one considered here for urine and washwater is known as electrodialysis. The membrane electrodialysis process uses synthetic ion-exchange membranes. These membranes are thin sheets of either cation- or anion-exchange resins and are, therefore, respectively selective as to the migration of cations or anions. They exhibit high electrical conductivity and low permeability to the passage of water. Upon the application of an electromotive force, the positive ions (such as Na^+) pass through the cation-permeable membrane, whereas the negative ions (such as Cl^-) move in the opposite direction and pass the anion membrane. Thus, the water passing between alternate membrane pairs is depleted of salt, while that passing through the intervening pairs is enriched (see Figure 49).

Electrodialysis will remove only ions and not urea. For this reason, urine has to be first treated with a complexing agent converting all of the urea into ionic ammonium carbonate which can then be removed by the electrodialysis. The conversion of urea by urease requires from a half of an hour to ten hours, depending on the amount of the enzyme used. A supply of 120 grams of that enzyme will be sufficient for seven astronauts for two weeks. The power requirement is very small - only a few watts per man.

Electrodialysis adapts itself very well to weightless conditions; if the amount of residual effluent can be made small and the desalted water can be made potable over a long period, then this method would have considerable merit.

The electrodialysis unit delivers roughly 60% potable water and 40% concentrated urea. The latter will be further processed in a permeable membrane.

Membrane permiation is a separation techniuque allied to distillation. Membranes that permit water but not urea to pass are available. Specially treated forms of "cellophane," acrylic sheets, or copolymers of polyvinyl alcohol have been used for similar distillations. Ionics Incorporated proposed to force urine (Figure48) under slight pressure into a "boiler" where the urine is heated to a suitable temperature. At this temperature, water permeates the membrane, while urea and other solutions do not. On the other side of the membrane is a condenser compartment filled with sponges and kept under a vacuum. Hot water evaporates from the membrane into this compartment and condenses on the cooling surfaces. For zero gravity condition, liquid water is collected in sponges.

The relatively high temperature in the "boiler" sterilizes the feed. The high temperature assures that NaCl will not precipitate until about 95% of the water is removed.

This system has not yet been developed. It is hoped that it will re-

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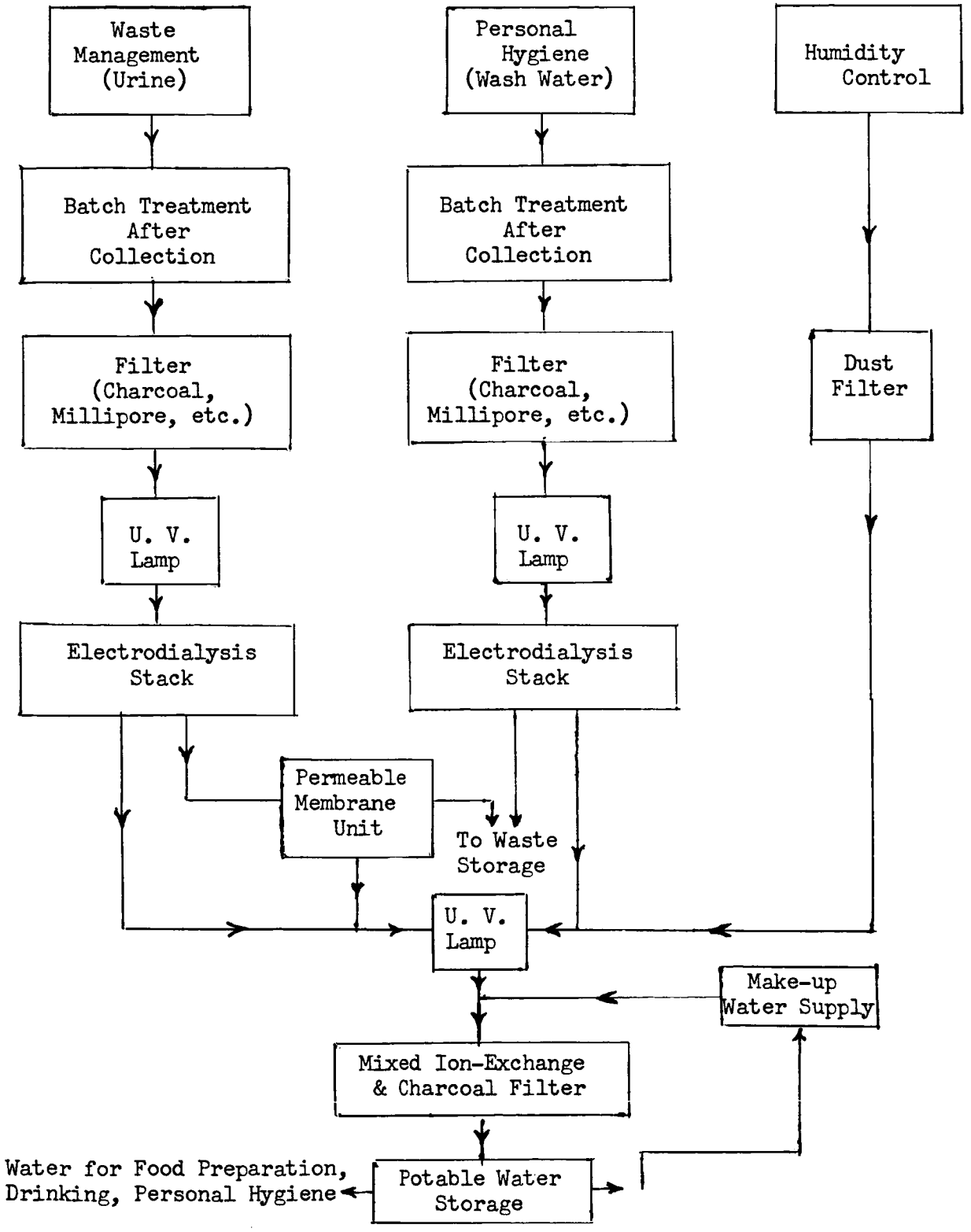


FIGURE 47 - BLOCK DIAGRAM-WATER RECLAMATION SUBSYSTEM

Item	Number of Men in Crew					
	2	3	4	5	6	7
Urine Processed, lbs	6.68	10.12	13.36	16.70	20.04	23.38
Operating Time, hrs	3.65	5.52	7.30	9.13	10.95	12.75
Equipment Weight, lbs	25.00	25.00	25.00	25.00	25.00	25.00
Equipment Volume, ft^3	1.15	1.15	1.15	1.15	1.15	1.15
Activated Charcoal, lbs	0.435	0.653	0.870	1.085	1.305	1.520
Complexing Agent, lbs	0.044	0.065	0.087	0.109	0.131	0.152
Waste Containers, lbs	0.021	0.031	0.042	0.052	0.062	0.073
Charcoal Storage, ft^3	0.017	0.026	0.035	0.044	0.052	0.061
Agent Storage, ft^3	0.039	0.058	0.077	0.097	0.116	0.135
Power Consumption, watt-hrs	143.50	215.00	287.00	359.00	431.00	503.00

TABLE 31

WATER RECOVERY BY ELECTRODIALYSIS - DATA FOR ONE DAY OPERATION - URINE = 3.34 lb/man-day

Item	Number of Men in Crew					
	2	3	4	5	6	7
Wash Water Processed, lbs	10.00	15.00	20.00	25.00	30.00	35.00
Operating Time, hrs	1.00	1.50	2.00	2.50	3.00	3.50
Equipment Weight, lbs	10.00	10.00	10.00	10.00	10.00	10.00
Equipment Volume, ft^3	0.55	0.55	0.55	0.55	0.55	0.55
Activated Charcoal, lbs	0.125	0.188	0.250	0.313	0.375	0.438
Complexing Agent, lbs	0.001	0.001	0.001	0.001	0.001	0.001
Waste Containers, lbs	0.007	0.010	0.014	0.017	0.021	0.024
Charcoal Storage, ft^3	0.005	0.008	0.011	0.013	0.016	0.018
Agent Storage, ft^3	0.001	0.001	0.001	0.001	0.001	0.001
Power Consumption, watt-hrs	10.00	15.00	20.00	25.00	30.00	35.00

TABLE 31

WATER RECOVERY BY ELECTRODIALYSIS - DATA FOR

ONE DAY OPERATION - WASH WATER = 5.00 lb/man-day

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SPACE and INFORMATION SYSTEMS DIVISION

Item	Number of Men in Crew					
	2	3	4	5	6	7
Water Recovered						
Urine, lbs	6.46	9.80	12.90	16.15	19.35	22.55
Perspiration & Respiration, lbs	6.74	10.11	13.48	16.85	20.22	23.59
Wash Water, lbs	9.90	14.85	19.80	24.75	29.70	34.65
Total, lbs	23.10	34.76	46.18	57.75	69.27	80.79
Water Losses						
Urine, lbs	0.22	0.32	0.46	0.55	0.69	0.83
Perspiration & Respiration, lbs	0.02	0.03	0.04	0.05	0.06	0.07
Wash Water, lbs	0.10	0.15	0.20	0.25	0.30	0.35
Total, lbs	0.34	0.50	0.70	0.85	1.05	1.25

TABLE 32

BASIC DAILY WATER RECOVERY DATA

TABLE 33 - WEIGHTS, VOLUMES, AND POWER CONSUMPTION OF TOTAL WATER RECOVERY SYSTEM

	7 Men
Urine Recovery Subsystem	
Hardware weight	25.00
Expendables weight	24.43
Total	49.43 lb
Hardware volume	1.15
Expendables volume	2.75
Total	3.90 ft^3
Operating cycle	12.75 hrs/day
Power Consumption	7.05 kw-hrs
Wash Water Recovery Subsystem	
Hardware weight	10.00
Expendables weight	6.48
Total	16.48 lb
Hardware volume	0.55
Expendables volume	0.27
Total	0.82 ft^3
Operating cycle	3.50 hrs/day
Power consumption	0.49 kw-hrs
Humidity Water Recovery Subsystem	
Hardware weight	2.00
Expendables weight	0.25
Total	2.25 lb
Hardware volume	0.25
Expendables	0.17
Total	0.42 ft^3
Operating cycle	Continuous
Power consumption	1.35 kw-hr

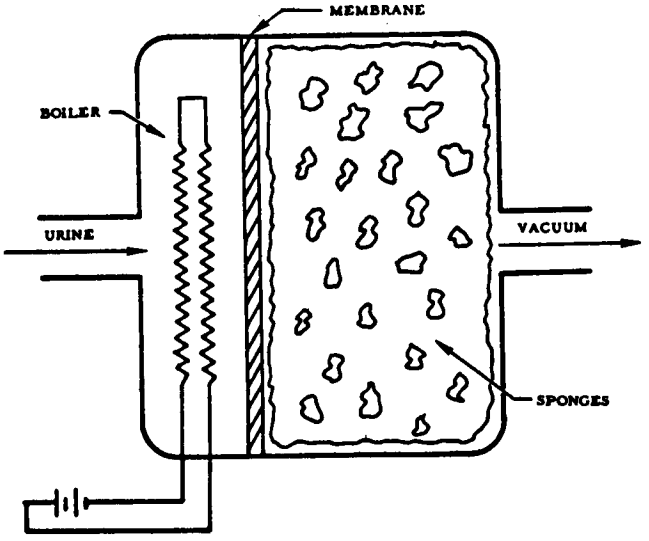


Figure 46 Membrane Permeation

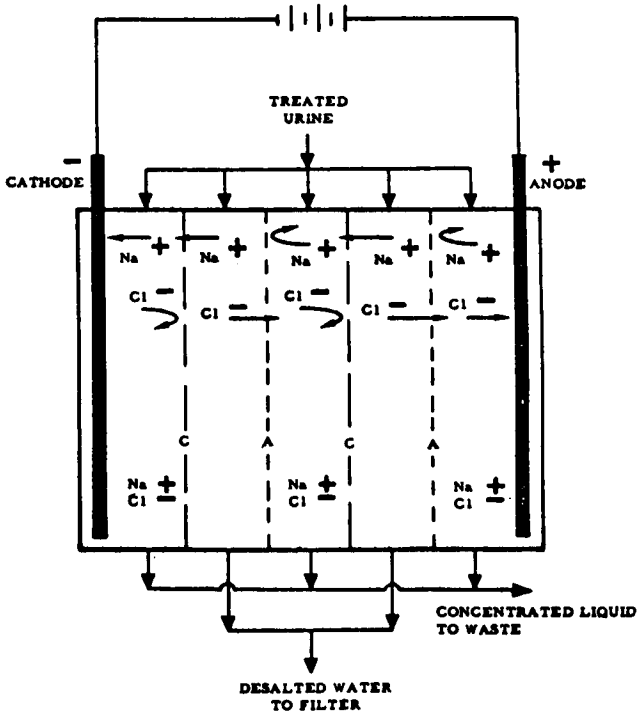


Figure 49 Electrodialysis Cell

TABLE 34

TOTAL ECS/LSS WEIGHT

Command Module	
Back-pack charges and supplies (restricted to 28 sorties)	335 lb
Pressure suit circuit	110
Water-glycol circuit	65
Pressure & temperature control	20
Oxygen supply system	25
Water supply system	45
Plumbing, wiring, instrumentation	35
Supports, controls, electrical	38
Molecular seive unit	60
Water recovery system (+ chemicals)	70
Charcoal	8
Freon	14
	825 lb
Service Module	
Water-glycol circuit	100
Water supply system	10
Oxygen supply system	5
Oxygen	528
Oxygen storage vessel	112
Water stored at launch	180
	935 lb
TOTAL ECS/LSS	1760 lb

quire comparatively small volume and little weight. It will, however, require some power (290 watts for a daily recovery of water from three men).

Structures and Weights

The seven-man command module has the same basic Apollo shape, but its diameter was increased by eighteen inches. The structural system, however, will not be changed - double-sandwich wall construction with insulation in between, four longerons leading tup to the parachute and escape tower attachment, airlock on top, unpressurized aft compartment, and membrane-type spherical bottom. The crew couches will again be suspended from the ceiling. It is anticipated that a small increase in the gage thickness of the inner pressure shell may be required.

The lunar take-off stage would consist of conventional sheet and stringer construction. Since the structure (in the three-man version) was determined by meteorite probability of no penetration, it is not expected that a considerable weight increase results from the heavier CM and the different outline.

The lunar landing stage may have to be modified to accommodate the larger propellant tanks. However, the mass ratio of this stage should improve or, at least, remain the same. The landing gear weight was determined by extrapolation from data developed for the lunar logistic vehicle (SID 62-1466) and other vehicles.

The expected propellant tank wall gage increases are listed below:

TABLE 35

Module	Designation	Propellant Weight (lbs)	Tank	Shape	Percent Gage Increase
Lunar Take-off	SMII-7	19,700	LH_2	Torus	3.5%
			LO_2	Ellipsoid	10.3%
Trans-Lunar Landing	LLM/LLV	66,400	LH_2	Torus	6.5%
			LO_2	Torus	1.0%

A weight statement for the seven-man Apollo II is shown in Table 36.

TABLE 36

PRELIMINARY WEIGHT ESTIMATE APOLLO II - 7-MAN DIRECT MISSION

STAGE II		
Structural Section		1,950
Propulsion System		(1,420)
Propellant Tanks		700
LO_2	327	
LH_2	373	
Pressurization System		245
Propellant Feed System		35
Engine Installation		440
Trapped Propellant		260
Residual Vapor		90
Booster Burn-Out Weight		3,720
Propellant		18,470
LO_2	15,392	
LH_2	3,078	
Gross Weight, Booster Stage II		22,190
Payload		
Command Module		(10,463)
Structure		4,460
Crew Systems		2,700
Communication & Instrumentation		760
Guidance & Navigation		250
Stability and Control		150
Reaction Control System		360
Electrical Power		346
Environmental Control System		825
Earth (water) Landing System		612
Service Module Equipment		(3,112)
Electronics		140
Electrical		1,037
Reaction Control		1,000
Environmental Control		935
Contingency		(140)
Propellant Boil-off		150
Lunar Launch Weight		36,055
STAGE I		
Structural Section		2,550
Propulsion System		(4,775)
Propellant Tanks, Insulation, Support		2,915

LO_2 Tank	785	
LH_2 Tank	2,130	
Pressurization System		820
Propellant Feed System		130
Engine Installation		910
Landing Gear		1,630
Trapped Propellant		950
Residual Vapor		260
Lunar Landing Weight		46,220
Propellant		60,915
LO_2	50,762	
LH_2	10,153	
Propellant Boil-off		680
Escape Weight		107,815

[redacted]

PART III - LUNAR SHELTERS

INTRODUCTION

After the initial landing of astronauts on the moon by the Apollo/LEM systems the next steps in the overall lunar program will be the exploration and site verification and temporary base establishment phases. To accomplish these phases lunar shelter must be available for use as emergency housing should transport vehicle malfunction occur, for extended exploration time, and as bases for early exploratory missions.

This part of the report describes several lunar shelter concepts which are logical, practical out-growths of the current Apollo program and are unique in that they utilize, to the fullest extent, current Apollo hardware, thus minimizing the engineering, development, tooling, and procurement cost and time requirements.

SUMMARY

The objective of this part of the study was to develop conceptual designs of early shelter and logistic modules which utilize, to a maximum degree, current Apollo hardware. The extended use of Apollo hardware is viewed as a practical extension of the current Apollo program and the economic advantage is almost intuitively evident.

The following sections describe the requirements, analysis, design, and evolution of four early lunar shelter, or logistics module, concepts. These are:

1) CLEM-shelter (Concept 1)

2) SM-shelter/logistics module (Concept 2)

3) LLV-shelter (Concept 3)

4) Extended Lunar Occupancy Payload

Concept 1 (Figure 50) the CLEM-shelter, is the natural evolution of the Apollo CM into an early lunar shelter. The shelter employs the basic Apollo CM structure, a slightly modified outer shell to withstand lunar environment, an airlock to allow ingress and egress on the lunar surface, and a redesigned interior eliminating crew couches, guidance, controls, and incorporating two bunks and other gear required for personnel occupancy. The Earth landing system is eliminated and a short service module adaptor is added to provide packaging volume for environmental control, life support, and electrical power system components which are similar to those employed in the Apollo SM. The shelter-service module assembly is mounted on the "CLEM", the cargo-LEM proposed by Grumman, and is landed (unmanned) on the

lunar surface by the CLEM. The gross weight of the CLEM shelter is 6765 lbs and provides lunar shelter for two crewmen for a period of three weeks.

Concept 2 (Figure 51), the CM-shelter/Early Logistics Vehicle (CMS/ELV), is the next practical step in shelter development. The shelter is similar to Concept 1 except that the exterior shell is modified to endure boost loads, atmospheric environment, and the long term (6 months) exposure to the lunar environment. The shelter is mounted on the standard Apollo SM which is modified for a direct unmanned landing lunar mission. These modifications include a throttleable propulsion engine, the addition of a 4-legged landing gear, and an unmanned guidance system.

At an injection weight of 67,150 lbs, the modified standard 154" diameter SM is capable of landing a useful payload of 14,854 lbs which provides for unmanned lunar standby of the shelter for 3 months and occupancy, after this standby period, of 90 days by two crewmen. If the standard SM, and its propellant tanks, are lengthened 44 inches, the full injection capability (90,000 lbs) of the Saturn V is utilized. Consequently, a payload of 20,970 lbs may be landed in the moon, which will allow, in addition to the standby and occupancy times described above, the landing of 6116 lbs of extra cargo.

Concept 3 (Figure 53), the LLV-shelter, integrates the CM-shelter utilized in Concepts 1 and 2 within a larger occupancy module. This shelter concept is a version of the Lunar Occupancy Payload studied earlier (SID 62-1473) and is mounted on a 260" diameter Lunar Logistics Vehicle, an unmanned direct lunar-landing vehicle also studied earlier (SID 62-1466). The electrical power is supplied by a dynamic isotopic source depending on occupancy crew size from 2 to 5 men, the lunar landed weight of this occupancy shelter varies from 23,840 lbs to 33,645 lbs for a standby time of one year and an occupancy time of 4 months. Current (90,000 lbs) injection capability of the SV will permit a landed weight of 27,500 lbs; therefore, uprating of the Saturn-V will be required to provide an occupancy period of 4 months for 5 men.

Concept 4 describes means for extension of the standby and occupancy periods of the Lunar Occupancy Payload (SID 62-1473) from 30 days standby and 30 days occupancy to one year and four months, respectively. With a dynamic isotopic prime electrical power source, Concept 4 has a landed weight of 22,089 lbs for 2-man occupancy and 25,384 lbs for 3-man occupancy, well under the allowable landed weight capability of 27,500 lbs.

SYSTEMS REQUIREMENTS

The general requirements for all shelter concepts are as follows:

1. Capable of providing shelter to the occupants from the lunar temperature, radiation, and meteoroid environment.

2. Provide for unoccupied standby periods as well as multi-man occupancy.

3. Allow maximum utilization of Apollo hardware.

4. Require minimum design and development.

WEIGHT CONSTRAINTS

- Concept 1 (CLEM-shelter) shall have a gross weight of 7000 lbs or less.

- Concept 2 (SM-shelter) shall have a gross weight, including expendables, of 15,700 lbs, or less, for the standard-length SM. Landed weight shall not exceed 23,900 lbs.

- Concept 2 (SM-shelter) shall have a gross weight, including expendables, of 20,970 lbs, or less, for the extended length SM. Landed weight shall not exceed 30,780 lbs.

- Concept 3 (LLV-shelter) and the extended LOP shall have a gross weight, including expendables of 27,485 lbs for the current Saturn-V injection capability of 90,000 lbs.

- Concept 3 and the extended LOP shall have a gross weight, including expendables, of less than 35,000 lbs for an uprated Saturn-V.

OCCUPANCY REQUIREMENTS

Shelter Concepts 1 and 2 shall provide environmental and life support requirements for two crewmen; Concept 3 and the extended LOP shall provide shelter for three or more crewmen.

TIME REQUIREMENTS

- Concept 1 shall be capable of an occupancy time of at least two weeks after, preferably, a one week standby period.

- Concept 2 shall be capable of an occupancy time of at least 60 days after standby times ranging from 60 to 90 days.

- Concept 3 shall be capable of standby and occupancy times approaching that of the extended mission LOP, e.g. 120 day occupancy after one year standby.

- The extended mission LOP shall increase its standby and occupancy times from 30 days and 30 days respectively, as reported in SID 62-1473, to one year standby and 120 day occupancy.

OPERATIONAL REQUIREMENTS

- Concept 1 shall be an Apollo CM shelter capable of being mounted on the Grumman proposed CLEM in its launch position in the SM adapter. Concept 1, because of its location during launch from Earth, will not "see" aerodynamic heating or dynamic pressure loads. The shelter will be landed, unmanned, on the lunar surface by the CLEM vehicle and will remain mounted on the CLEM during its lunar stay. During the unmanned period, or standby, on the lunar surface, the shelter communications and telemetry equipment shall be operable remotely to permit telemetry interrogation, but the ECLSS and occupancy power system will not be operated until the shelter is occupied by the astronauts.

- Concept 2 shall be on Apollo CM shelter mounted on the Apollo SM in a manner similar to the mounting of the current Apollo CM. The shelter shall be capable of enduring the aerodynamic, thermal, and dynamic loads imposed upon it during launch from Earth, in addition to the lunar environmental loads. The shelter shall be landed, unmanned, on the lunar surface by the SM in a direct landing mission and shall remain mounted on the SM throughout its lunar stay. To accomplish the direct unmanned lunar landing mission the Apollo SM shall be modified to include a landing gear, a throttleable engine, and an unmanned guidance system. During the unmanned, or standby period on the lunar surface the shelter telemetry and communications equipment shall be operable, but the ECLSS and occupancy power systems will not be operated until the shelter is occupied by the astronauts.

- Concept 3 shall be an Apollo CM shelter integrated with an exterior 260" diameter shelter with a configuration similar to the LOP (SID 62-1473) and mounted on the LLV (SID 62-1466 and 1467). Because of its location the Apollo CM shelter will not be exposed to aerodynamic heating and dynamic loadings, but the exterior shell of the configuration must be designed to withstand these loads in addition to the lunar environment. The shelter shall be landed, unmanned, on the lunar surface by the LLV in a direct landing mission and shall remain mounted on the LLV throughout its lunar stay. The LLV will utilize an unmanned guidance system as in Concept 2. During the unmanned standby period only the communications and telemetry equipment shall be operating; upon occupancy the ECLSS and occupancy power system operation shall be initiated.

- The extended mission LOP concept shall be similar to the LOP described in SID 62-1473 with the exception that its standby time capability shall be increased from 30 days to one year and its occupancy time capability increased from 30 days to 120 days. All other operational features are the same.

SYSTEMS ANALYSIS AND DESIGN

The systems analyses which were performed and culminated in the shelter concepts described herein are presented in this section of Part III of the report. Those systems which are most affected by mission duration are treated in detail; these are thermo-structural, environmental control and life support, and electrical power. In addition, discussions pertaining to throttling the Apollo SM engine, lunar landing gear, and unmanned guidance requirements and methods are given.

In the initial portion of this section, the preliminary design of each of the three early lunar shelter concepts is described along with a method of lowering the Concept 2 shelter from its vertical lunar-landed position to a horizontal attitude on the lunar surface; then the analyses of each of the pertinent technological areas mentioned above are presented and, finally, a weight statement for each of the lunar shelter concepts is tabulated.

LUNAR SHELTER DESIGN CONCEPTS

Concept 1

Concept 1, the first shelter design concept investigated during this study, was designed primarily to meet the system requirements specified earlier (e.g. 2 man shelter) and to be landed on the lunar surface by the cargo LEM descent stage proposed by Grumman. The conceptual arrangement and interior details of this shelter are shown on Drawing No. 6052-4, Fig. 50.

A basic objective in the design of this concept was the determination of the modifications necessary to transpose the Apollo command module into a habitable 2-man shelter so that engineering, development, tooling, and procurement time might be reduced to a minimum.

The primary structural change to the Apollo CM consisted of the elimination of the ablative material covering the entire vehicle. Interior structural changes involved removal of the display panels and upper equipment bays and the addition of a cylindrical airlock for ingress and egress on the lunar surface.

The three basic Apollo CM lower and side equipment bay locations have been retained, although it is assumed that considerable changes would be made in the equipment contained in those areas. The lower equipment bay would be used for food storage and scientific equipment. Food preparation facilities and the ECS equipment are located in the right hand equipment bay and electrical power system and telecommunications with associated display panels are located in the left hand equipment bay.

Crew couches have been removed and two floor level bunks are located on the left and right hand sides of the habitable area with a 24 inch aisle between the bunks. A folding work table and two swivel chairs are positioned at the foot of the bunks in what was formerly designated the forward equipment bay of the Apollo CM.

Life support consumables, in spherical tanks, 13 in number, are located in the annular toroidal space between the pressure shell and outer structural shell at the maximum capsule diameter. Consumables for the fuel cell electrical power system consists of three 25" diameter LH_2 spheres and two 22" diameter LOX spheres located in the interstage area between the CM and the LEM descent stage.

All Earth landing systems (e.g. parachute systems, shock attenuation devices) were removed. The parachute landing system compartment, in the apex of the command module, was converted into an equipment and supply storage area; total volume available in this compartment is approximately 22 cubic feet.

Shelter Concept 1, mounted on the LEM descent stage, is completely compatible with the LEM adapter interstage as shown on Drawing 6052-4 (Fig. 50) which depicts the current full length conical adapter.

Concept 2

The Concept 2 shelter was designed to be landed on the lunar surface by the current Apollo SM and to form, with the SM, a shelter/logistics module. To achieve this objective the shelter design developed for Concept 1 is mated with the Apollo SM. A cursory design examination was made into the general exterior arrangement of the modified SM as shown on Figure 51. This drawing shows an Apollo SM, lengthened 44" to provide additional propellant capacity and fully utilize the 90,000 lb injection capability of the Saturn-V and the addition of an extendable four-legged landing gear. The landing gear has a maximum half tread measured from the centerline of the vehicle of approximately 300". This provides approximately a 1:1 ratio of landing gear half tread to vertical C.G. above groundline. In the launch configuration the landing gear are folded alongside of the SM with the lower extremeties of the landing gear resting on the interstage adapter which would serve as a structural transition between the modified SM and the S-IV B stage of the Saturn-V launch vehicle. The overall height of the combined shelter and modified S/M after landing gear stroking is 41.75 feet. Access to and from the shelter, in view of this height, requires further investigation. This concept also employs an unmanned guidance system and a modified throttleable SM engine, both of which are discussed in later portions of this section of the report. This extended SM version of Concept 2 permits the lunar landing of a long duration shelter together with a cargo of more than 3 tons.

The standard length SM version of the Concept 2 shelter/logistics module is the same as that described above for the extended SM version except that the landing gear will have a narrower tread and very little, if any, cargo capacity will be available. Therefore, a very preliminary design investigation was made into a possible mechanical technique for lowering the Apollo C/M shelter to the lunar surface. This very first order concept is illustrated on Figure 52. The concept requires the addition of a single telescoping strut system, located between two of the other 4 man gear struts, hinged from the top of the S/M and capable of being extended so that it contacts the ground after landing. The two main landing gear systems on the same side as the single strut are physically disconnected from the service module after the single strut has been extended and released the load on the two landing gear struts. The single strut is allowed to move outward away from the S/M by a roller attached to the end. As the end of the strut rotates outward, it is controlled by a snubbing cable system. The single strut retracts as it moves outward, permitting the vehicle to slowly lay over on one side. This concept, or some variation may offer attractive possibilities for providing greater and safer access to the shelter. It would still be required to reposition the shelter so that the floor was parallel to the lunar surface. No design concept for this particular problem was investigated.

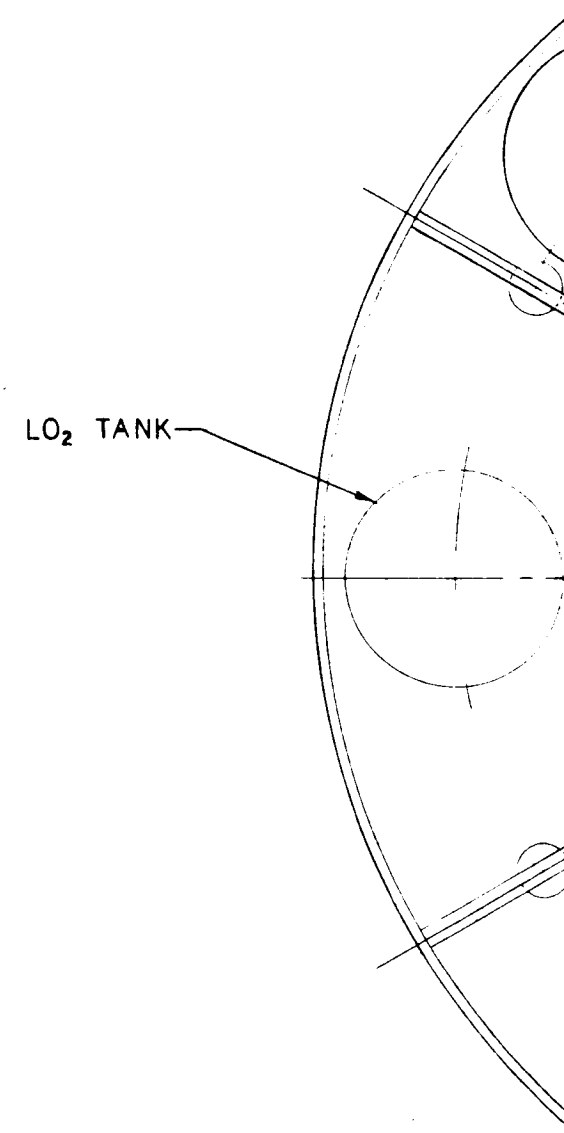
Concept 3

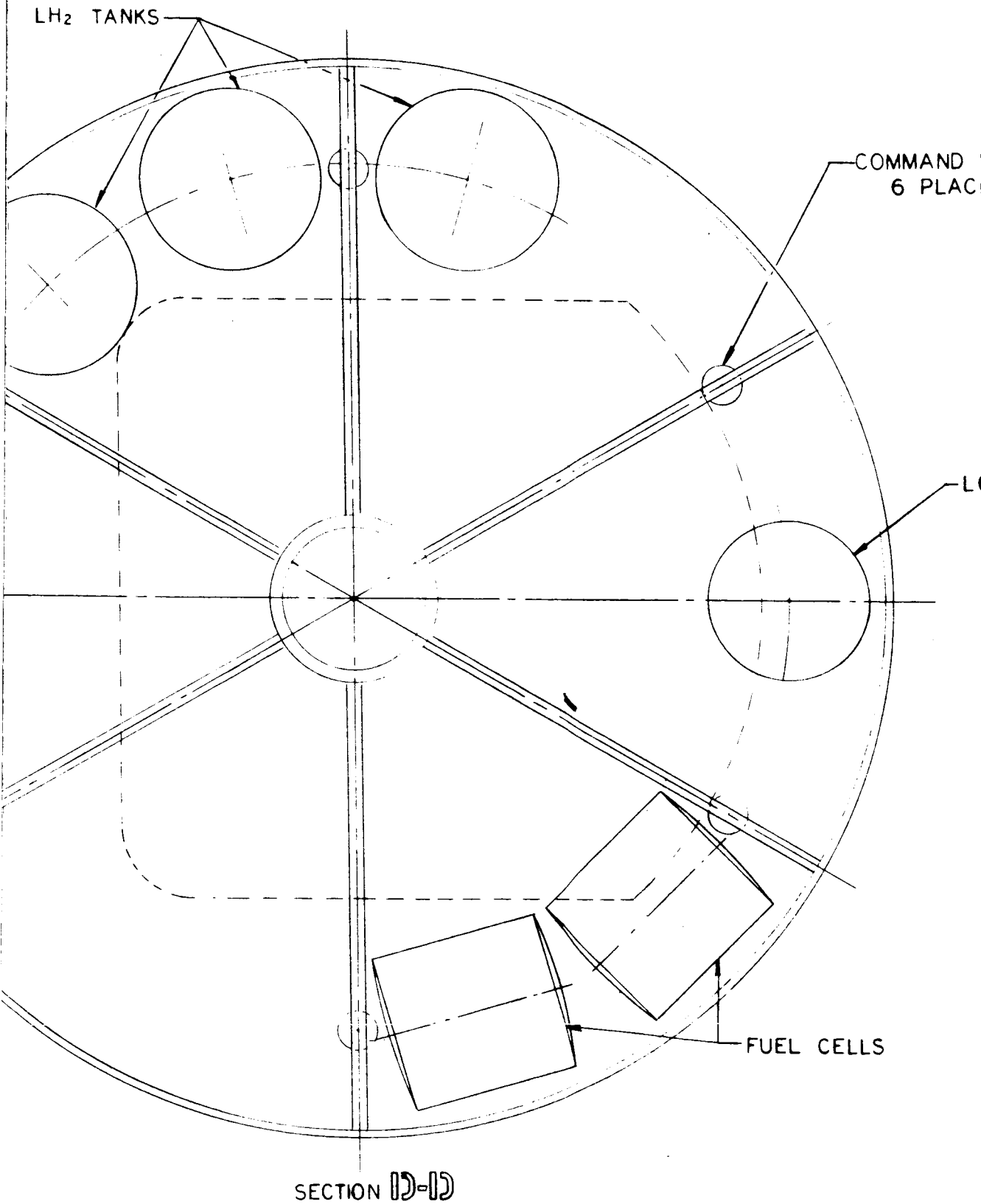
The Concept 3 shelter was designed to be landed by the cryogenic lunar landing vehicle studied and reported in SID 62-1466. The increased payload capability of this lunar lander permitted a greater sophistication in shelter design and mission duration.

The basic concept, shown in Figure 53, features modified Apollo C/M, providing the bulk of the life support systems, contained within the large structural shroud similar to the lunar occupancy payload vehicle developed and discussed in the Lunar Occupancy Payload Study Report SID 62-1473.

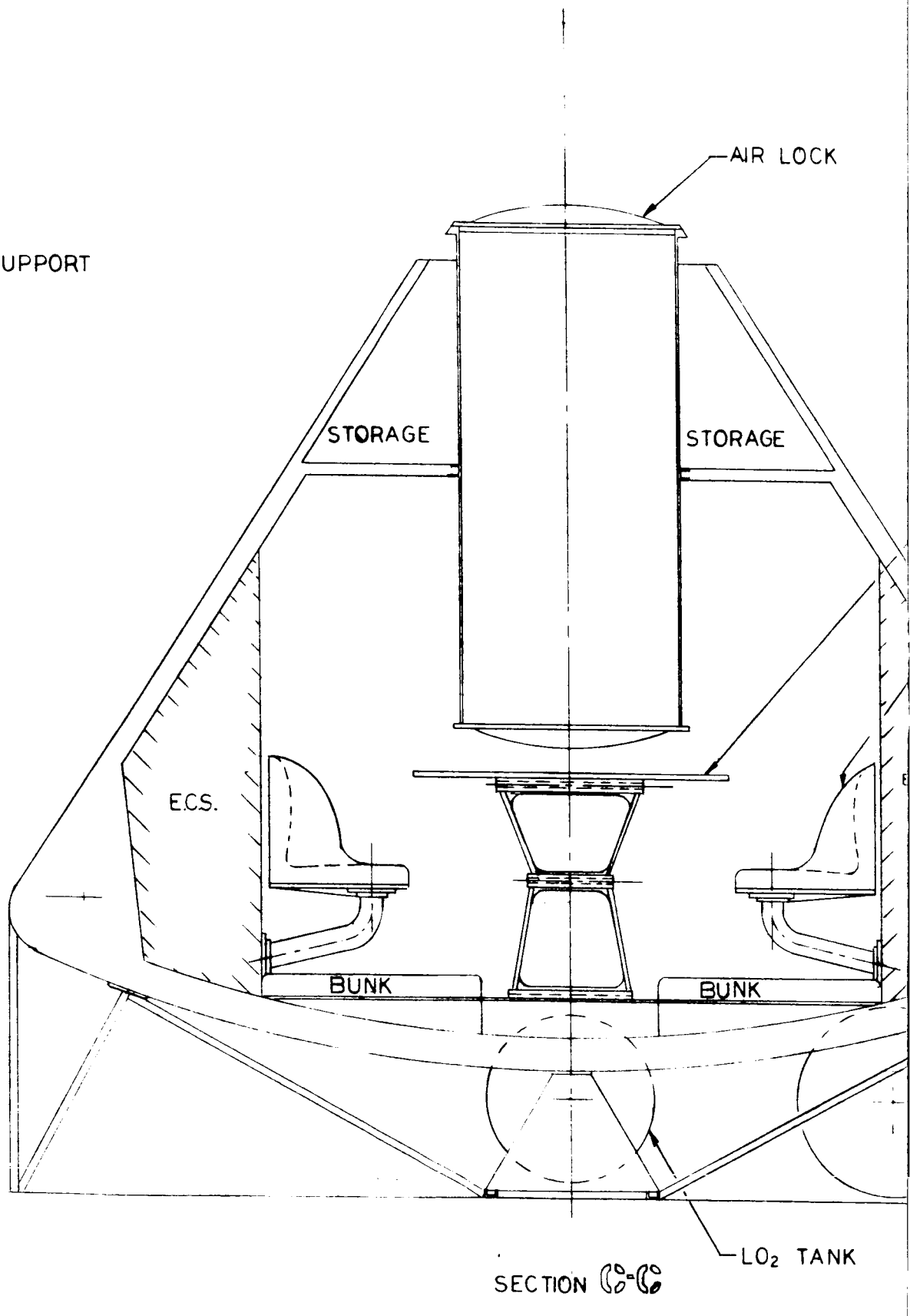
The stripped and modified C/M shelter provides the space for the bulk of the environmental control systems, bunks for the 2 to 3 occupants and a pressurized habitable area basically redundant to the large exterior pressure shell. The annular space between the Apollo shelter and the outer shell is used for operational equipment and work space. Specifically, within the annular volume is located the scientific equipment, electrical power system, communications and data equipment, telecommunications and associated instrument panel, and general storage area. The life support consumables and electrical power consumables are located in spherical tanks positioned below the primary floor in the annular habitable area.

All antennas and T. V. cameras are stowed in an unpressurized compartment in the forward nose section of the payload.


LO_2 TANK



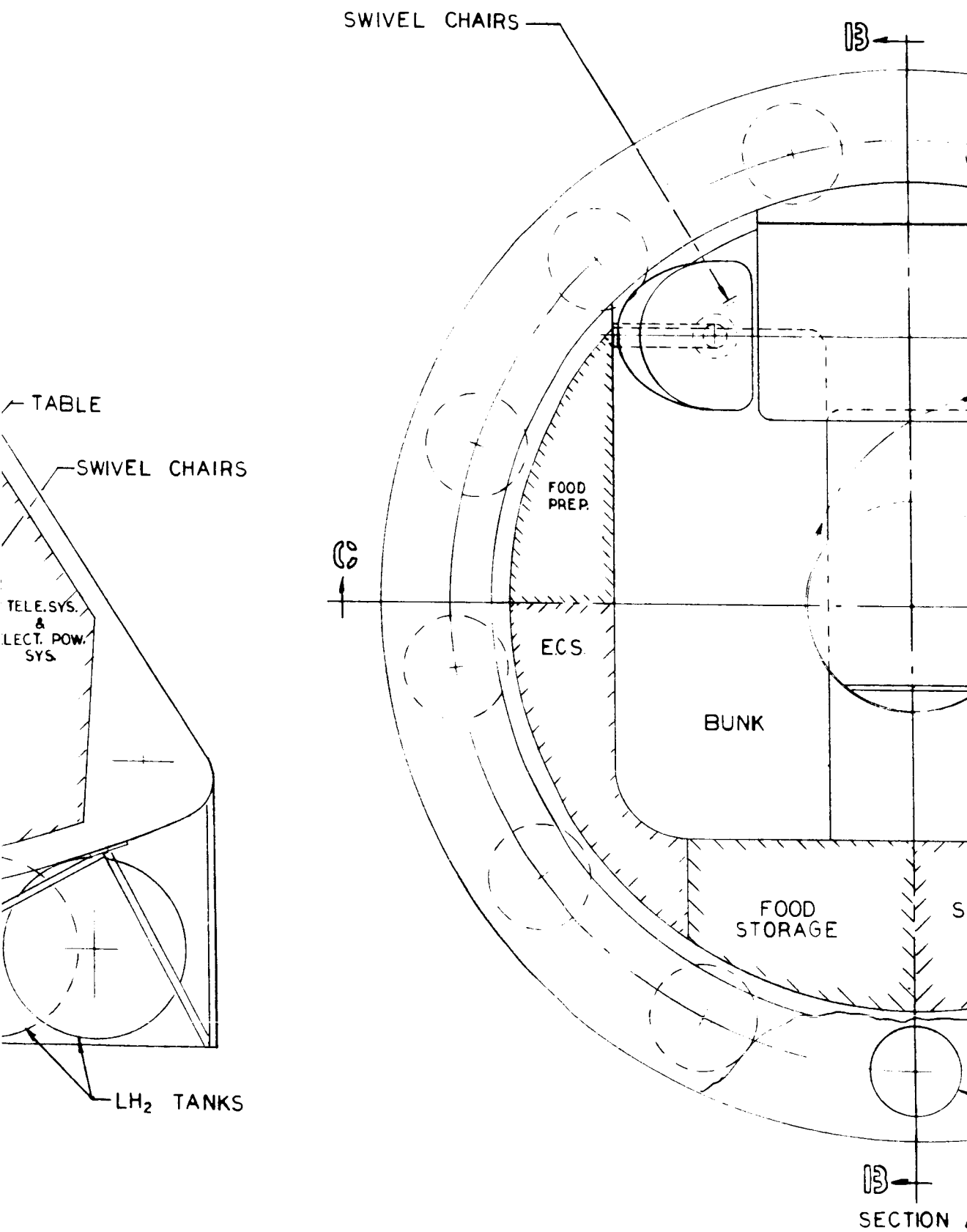
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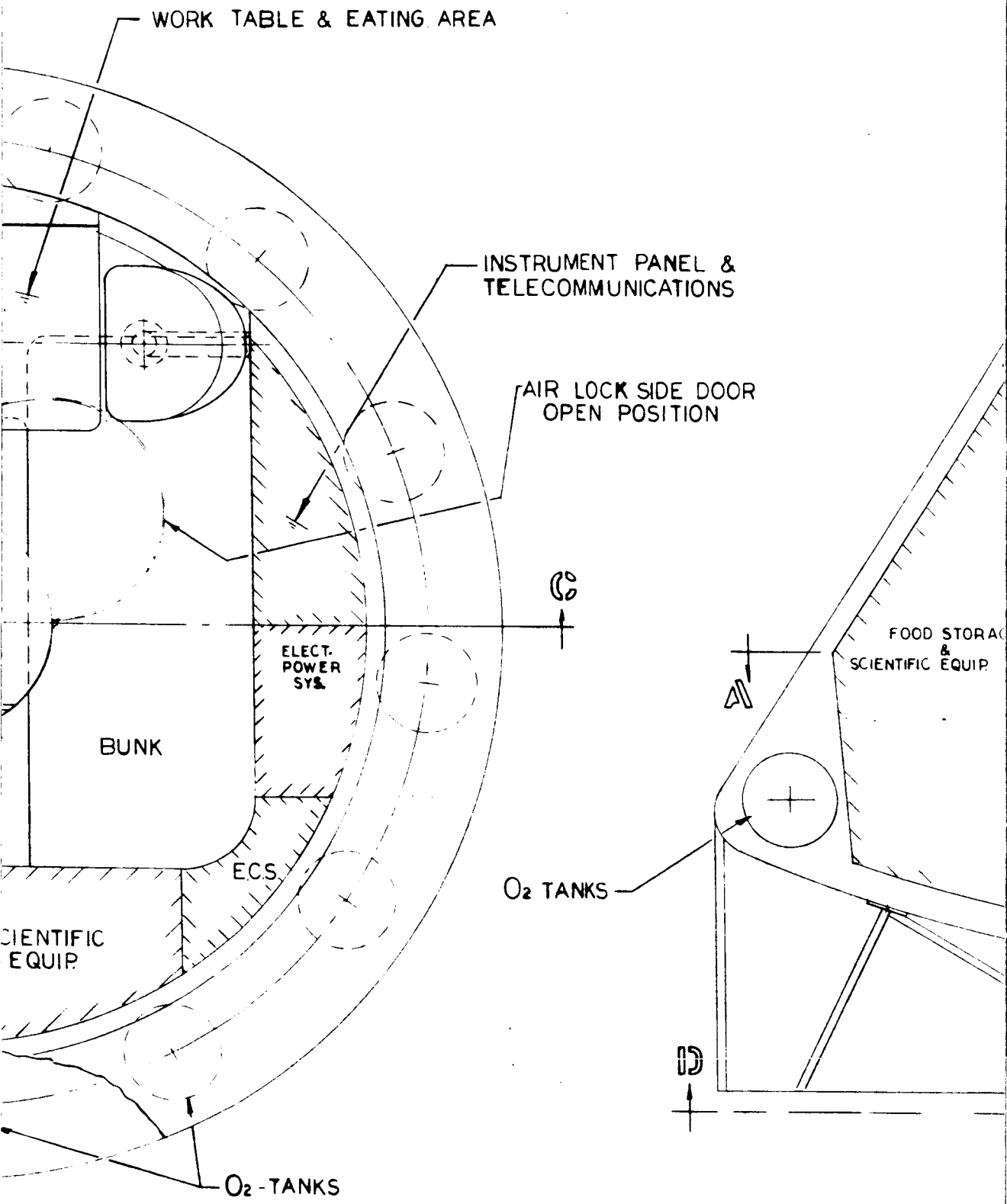


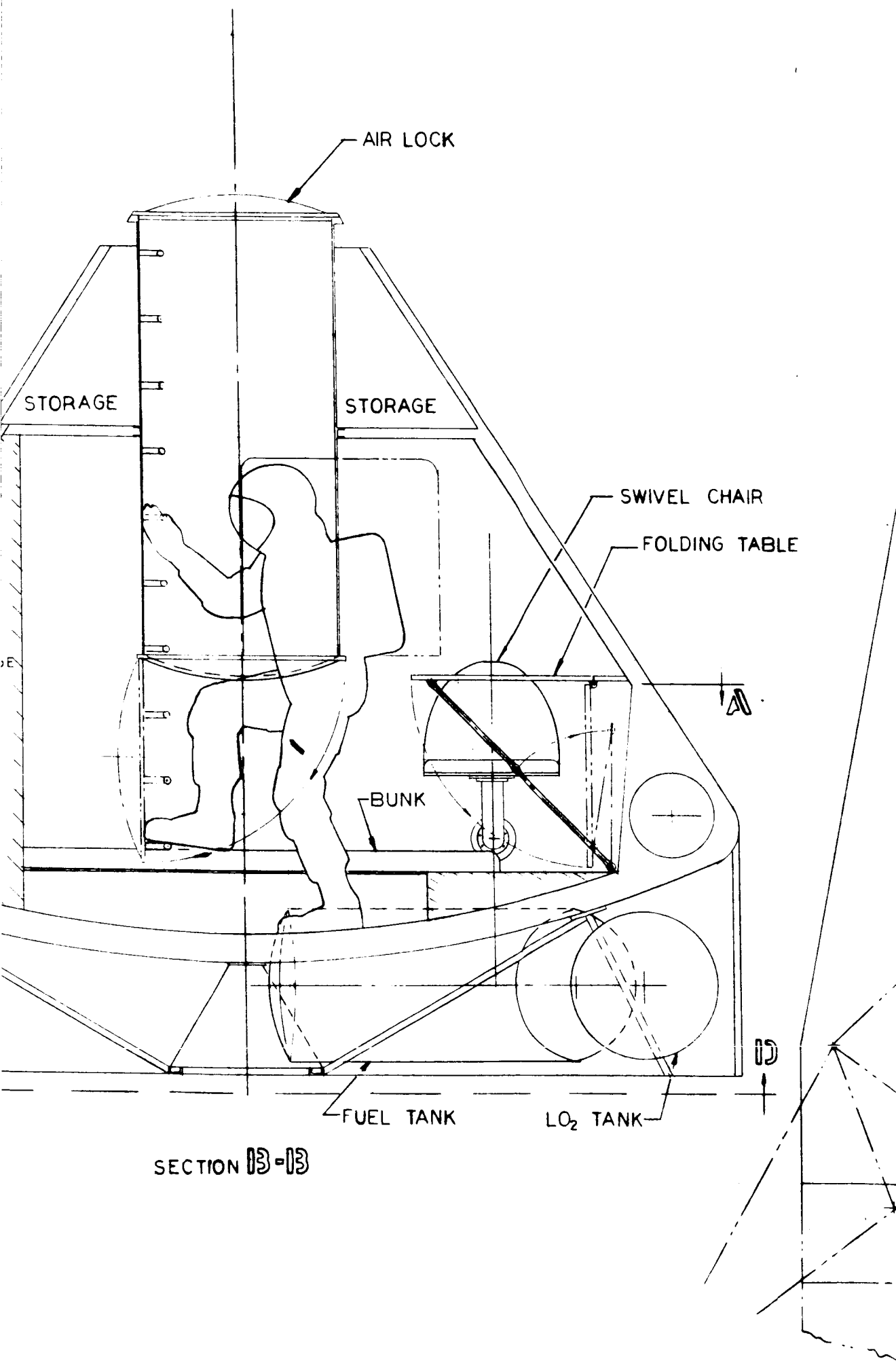
MODULE SUPPORT
ES

O_2 TANK

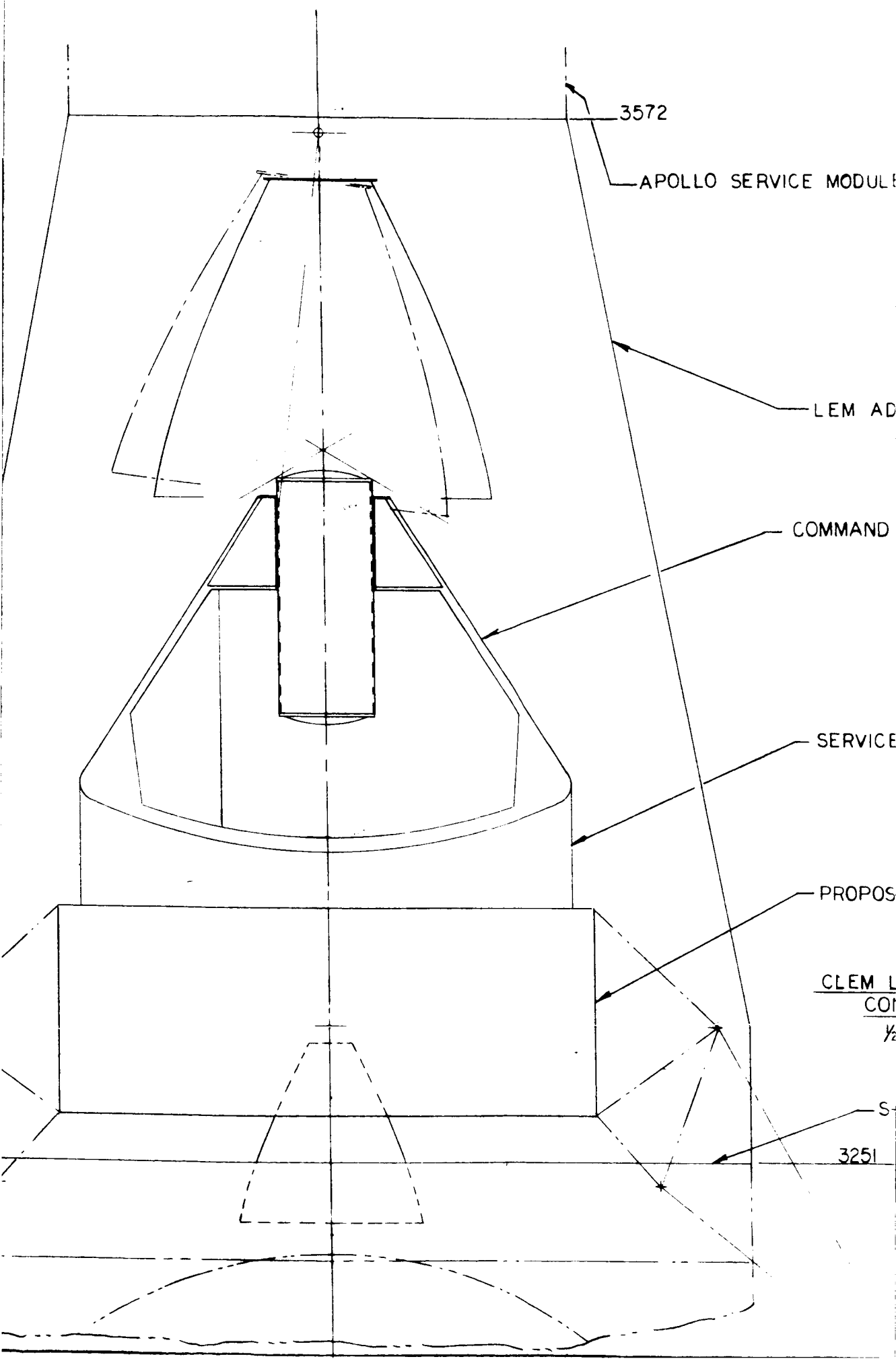
SECTION C-C

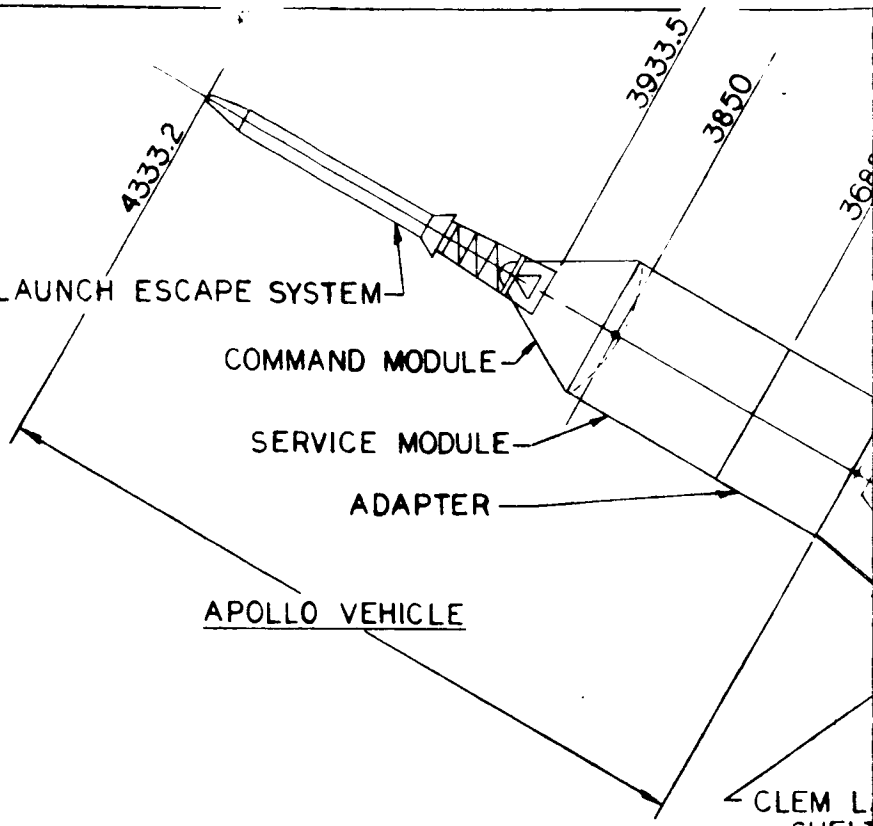

SWIVEL CHAIRS
B
TABLE
SWIVEL CHAIRS
TELE. SYS.
&
LECT. POW.
SYS.
FOOD
PREP.
C
ECS
BUNK
FOOD
STORAGE
S
LH2 TANKS
B
SECTION


WORK TABLE & EATING AREA
INSTRUMENT PANEL & TELECOMMUNICATIONS
AIR LOCK SIDE DOOR OPEN POSITION
C
ELECT. POWER SYS.
BUNK
E.C.S.
CIENTIFIC EQUIP.
O2 - TANKS
FOOD STORA
&
SCIENTIFIC EQUIP.
A
O2 TANKS
D
A A


AIR LOCK
STORAGE
STORAGE
SWIVEL CHAIR
FOLDING TABLE
A
BUNK
D
FUEL TANK
LO_2 TANK

SECTION D-D


3572
APOLLO SERVICE MODULE
LEM AD
COMMAND
SERVICE
PROPOS
CLEM L
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S
3251



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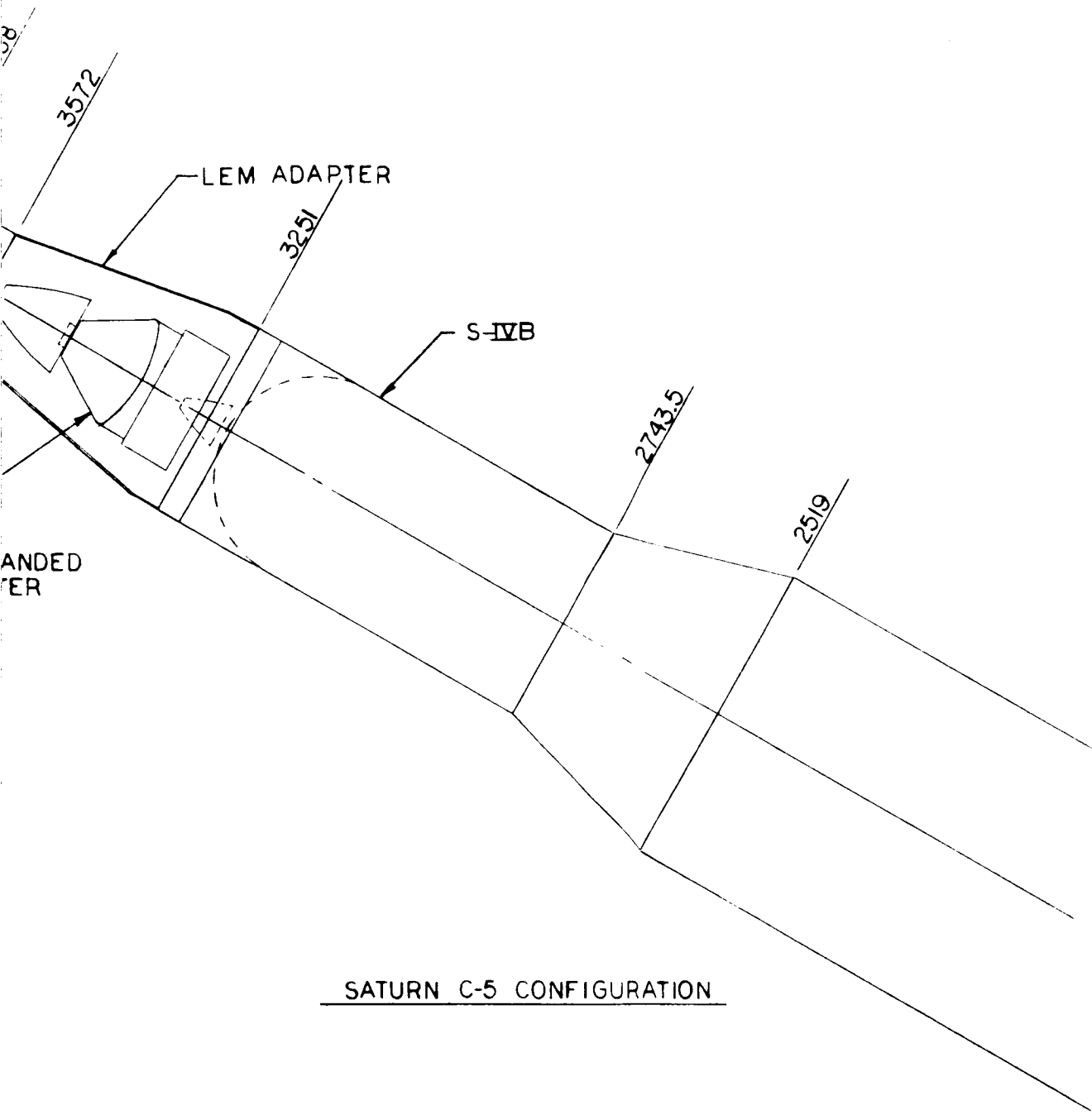
MODULE SHELTER

MODULE

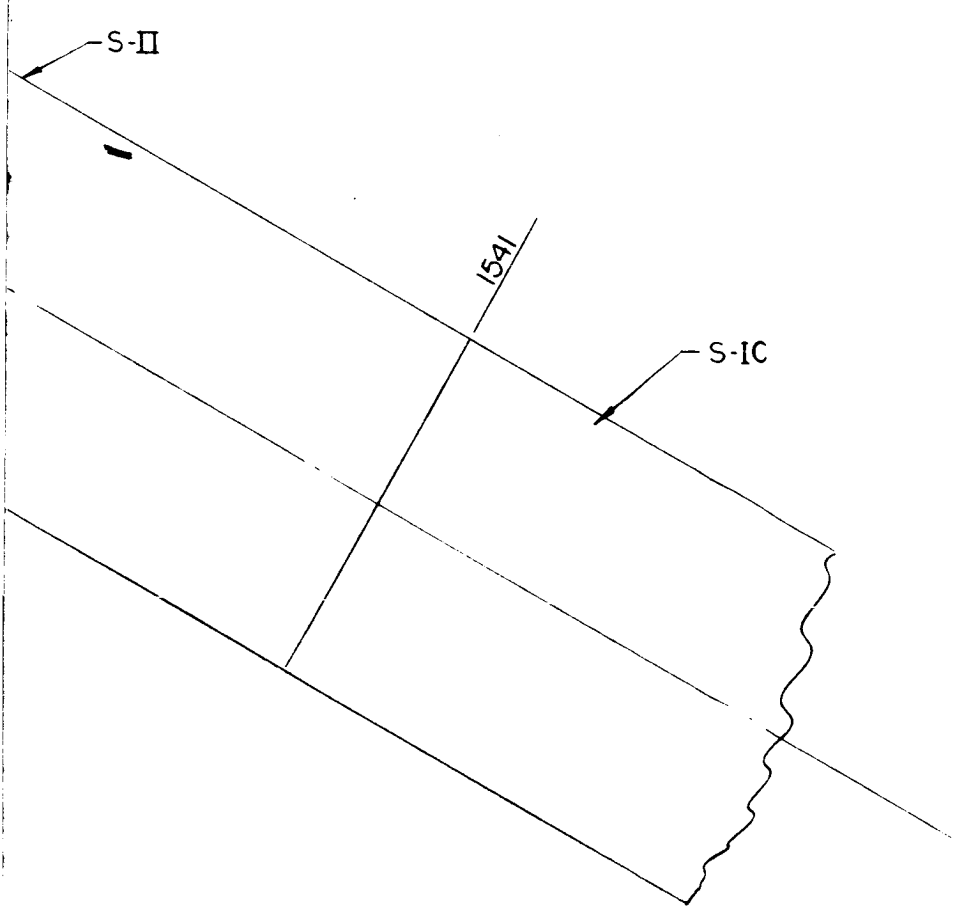
ED CLEM

ANDED SHELTER
NFIGURATION
o SCALE

IVB



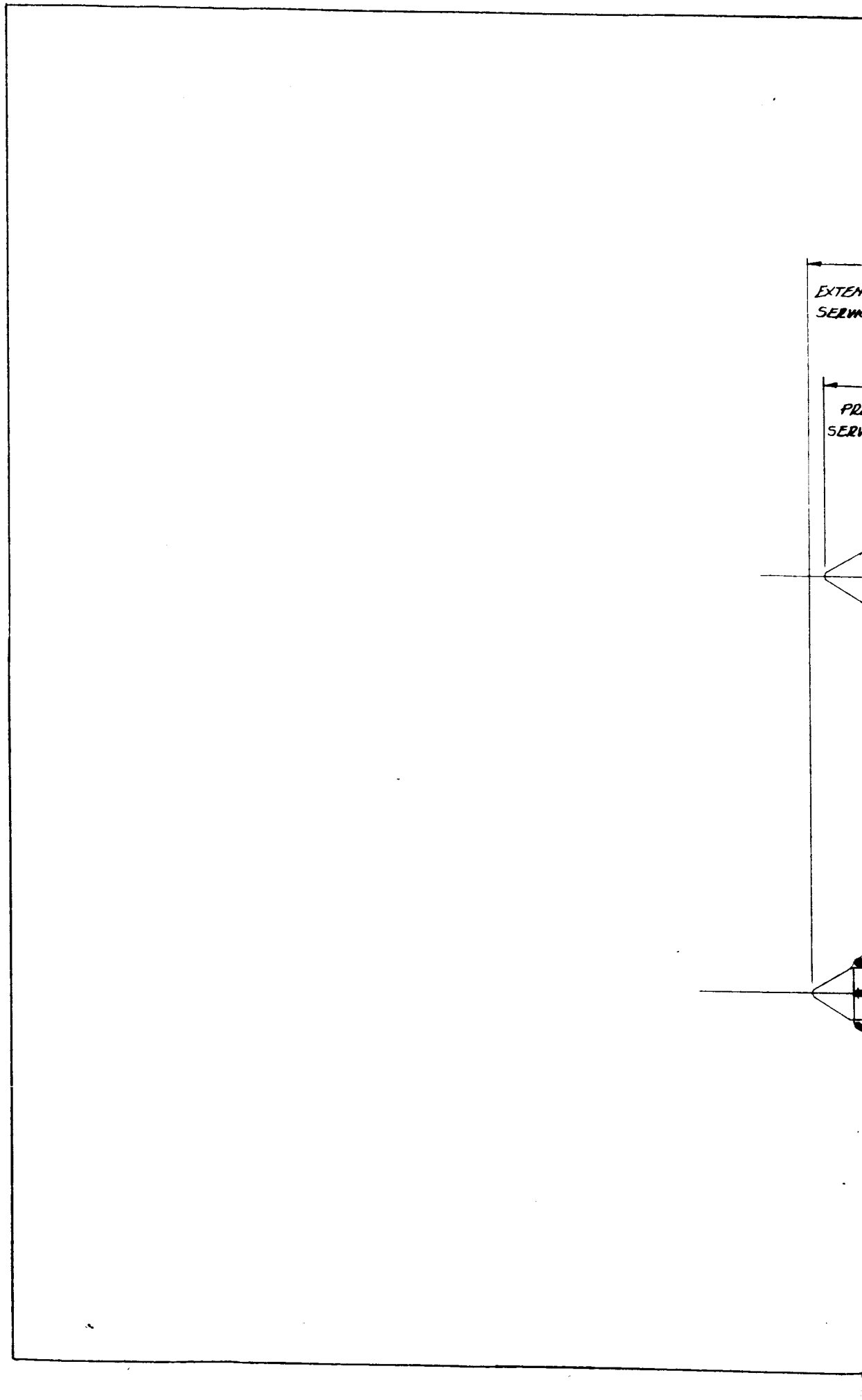
SATURN C-5 CONFIGURATION

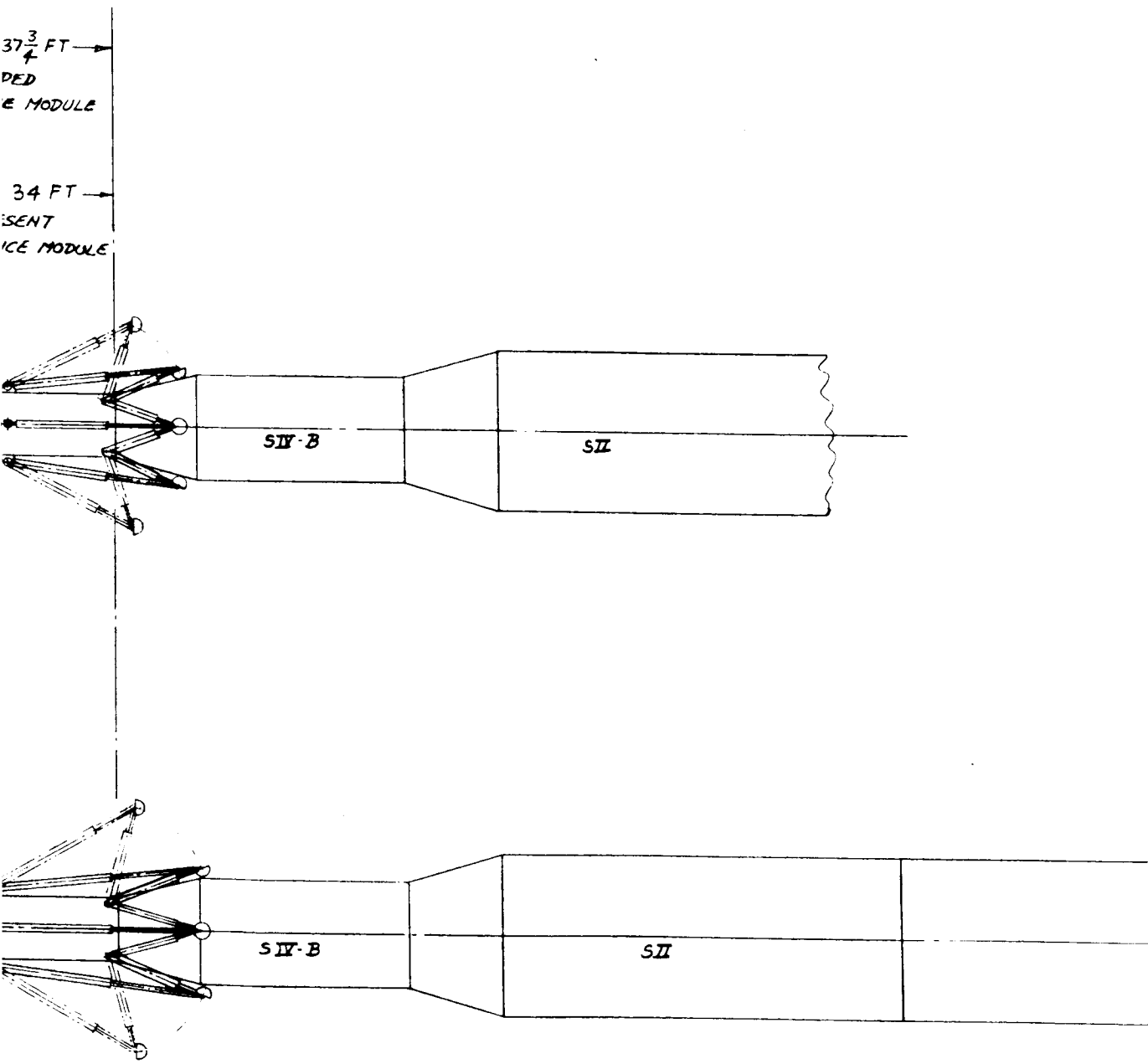

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SCALE 1/10 AND NOTED	BY RUSS DARR DATE 9-12-63 MODEL	NORTH AMERICAN AVIATION, INC. SPACE and INFORMATION SYSTEMS DIVISION 12214 LAKEWOOD BLVD. DOWNEY, CALIFORNIA

2 MAN COMMAND MODULE SHELTER
APOLLO VEHICLE CLEM-LANDED

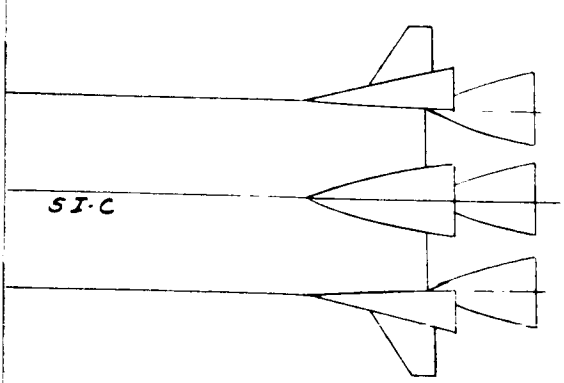
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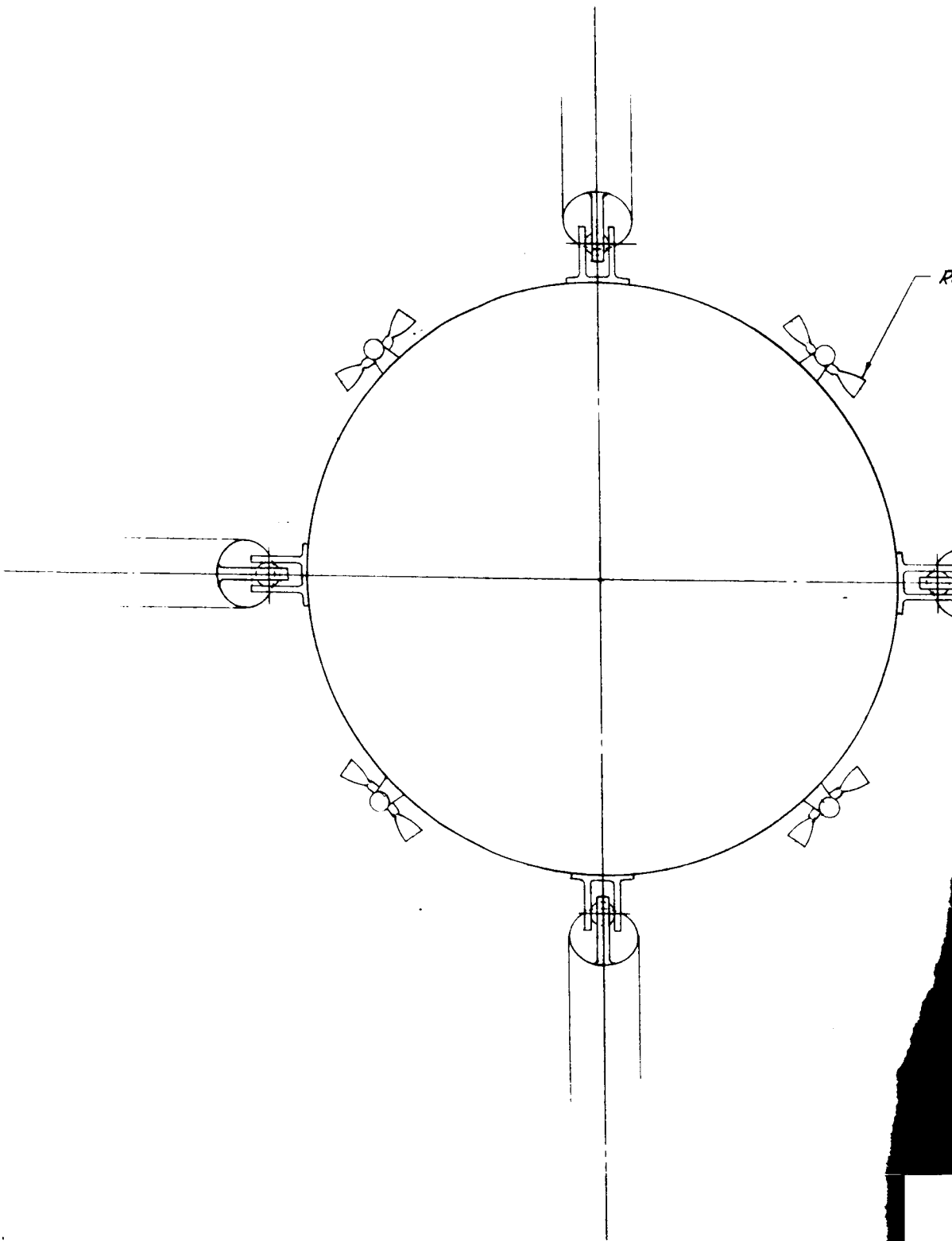

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SATURN V CONFIGURATION

SCALE - 1/200


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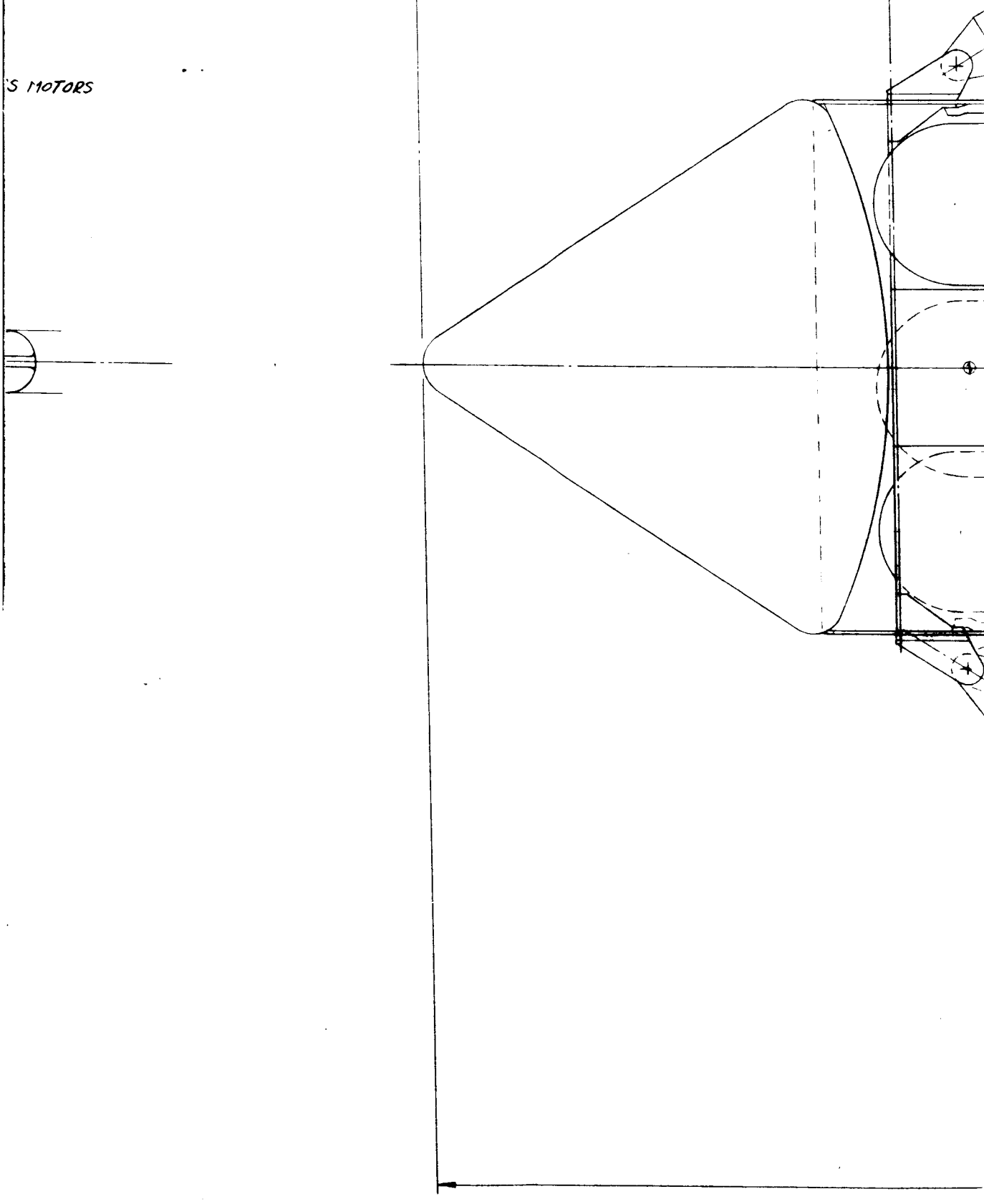


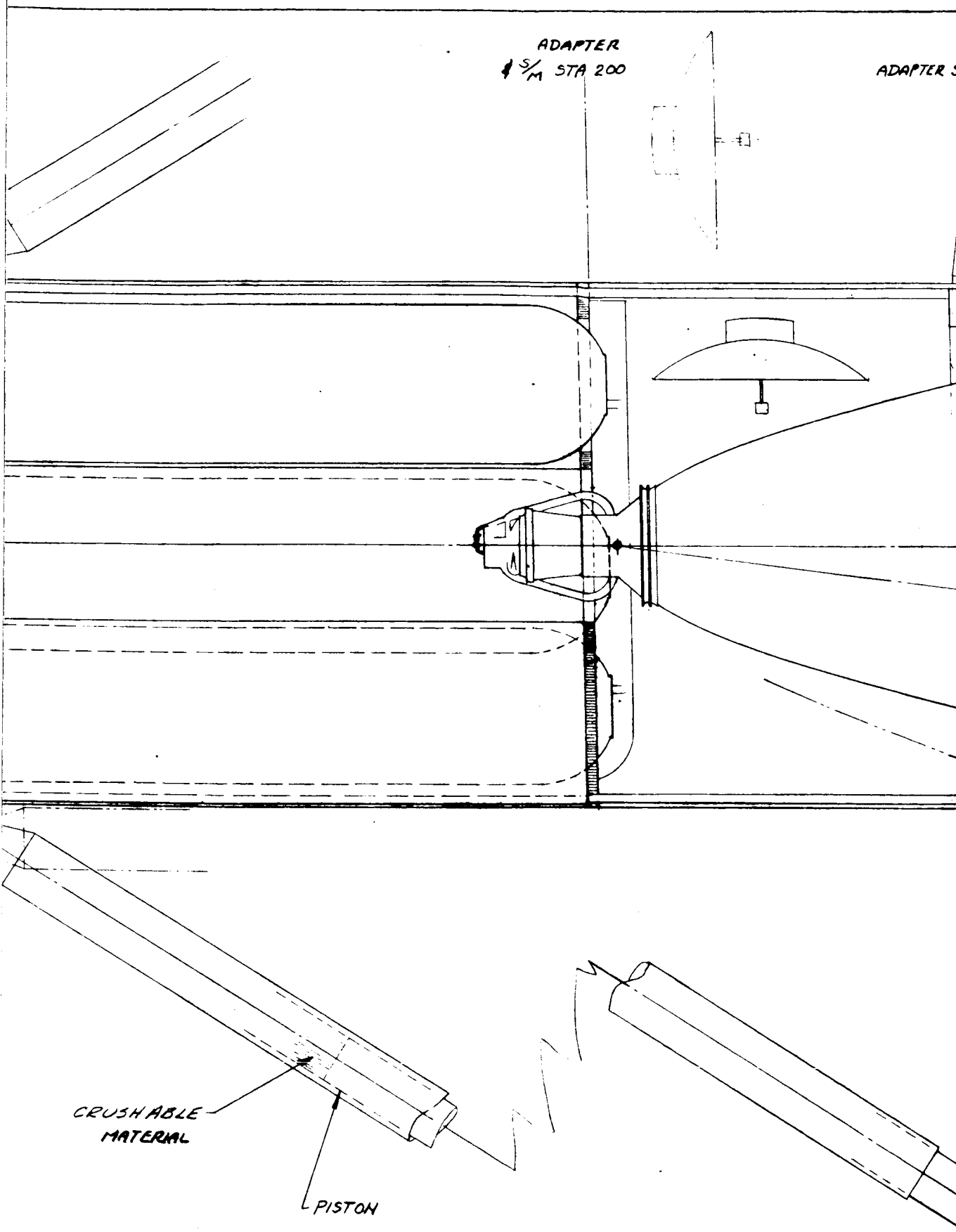
TOP VIEW

S/M STA 535

S/M STA 399

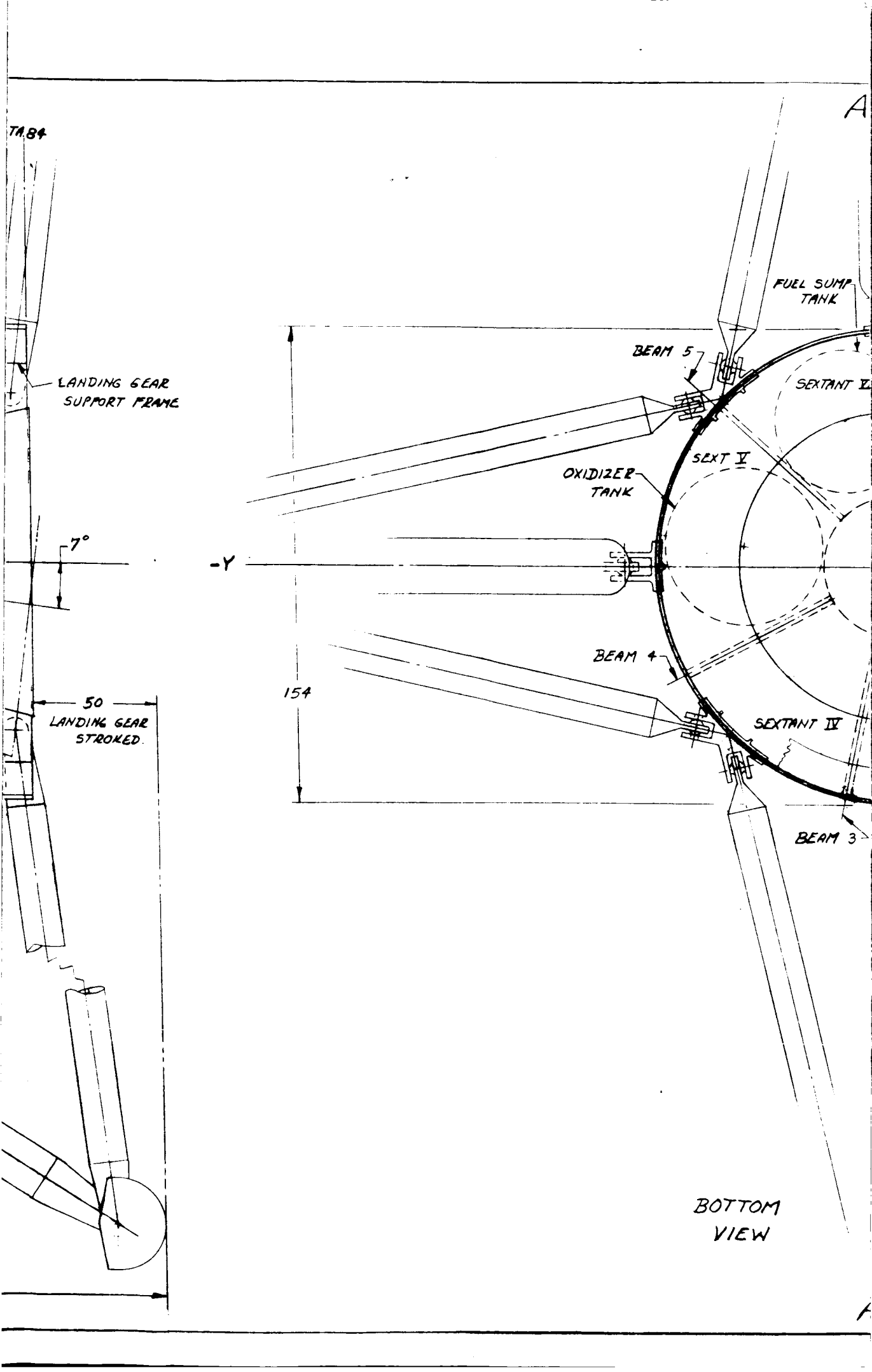
S MOTORS

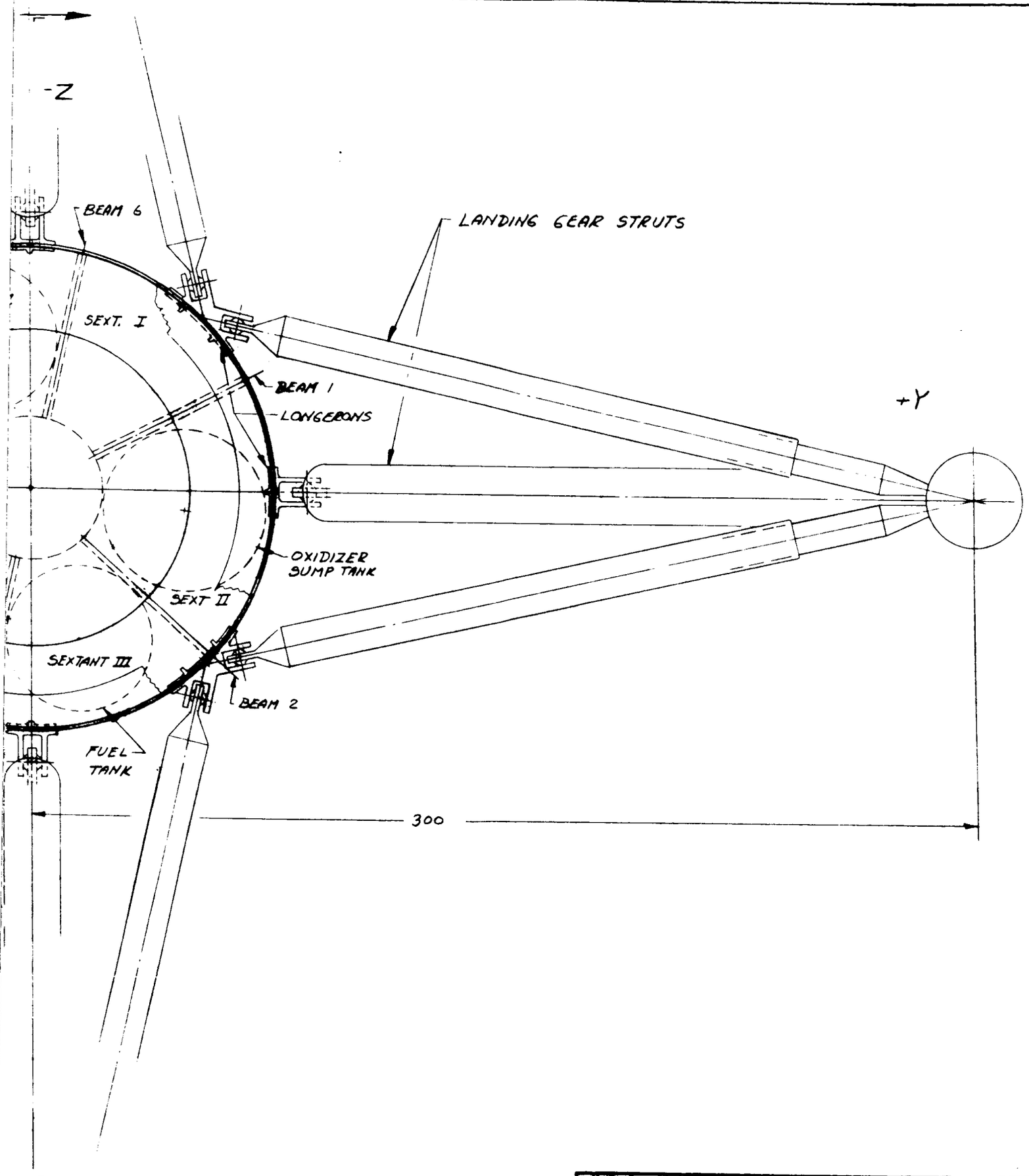




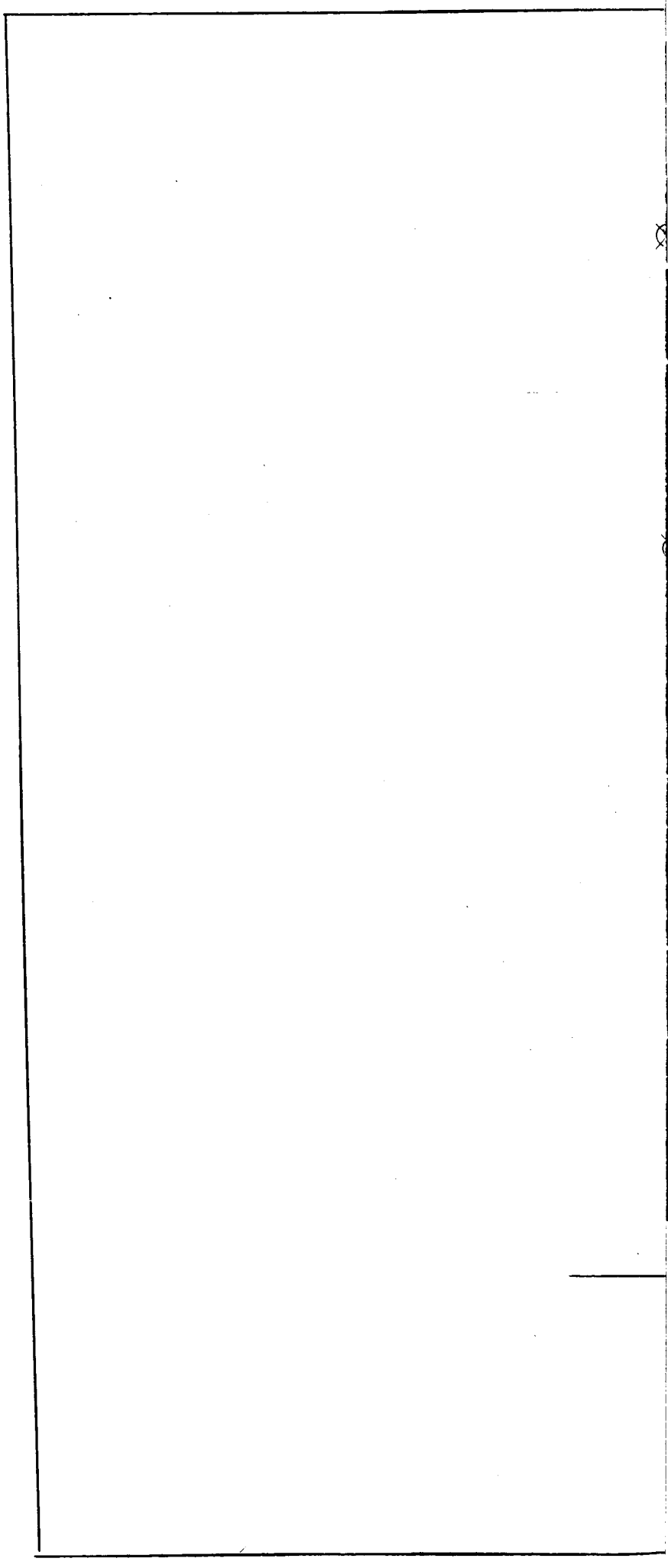
SECTION AA

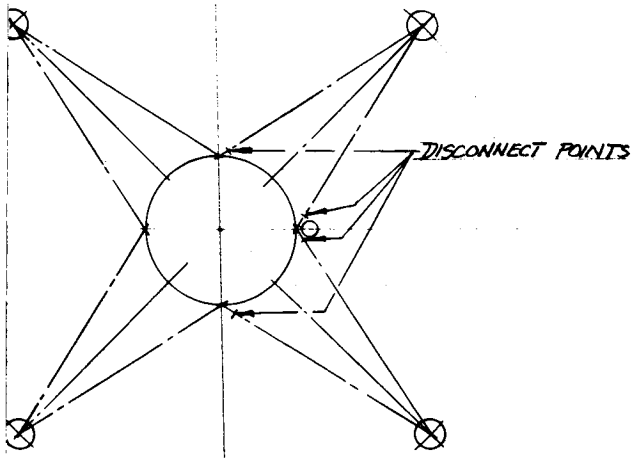
501 (41.75 FT)

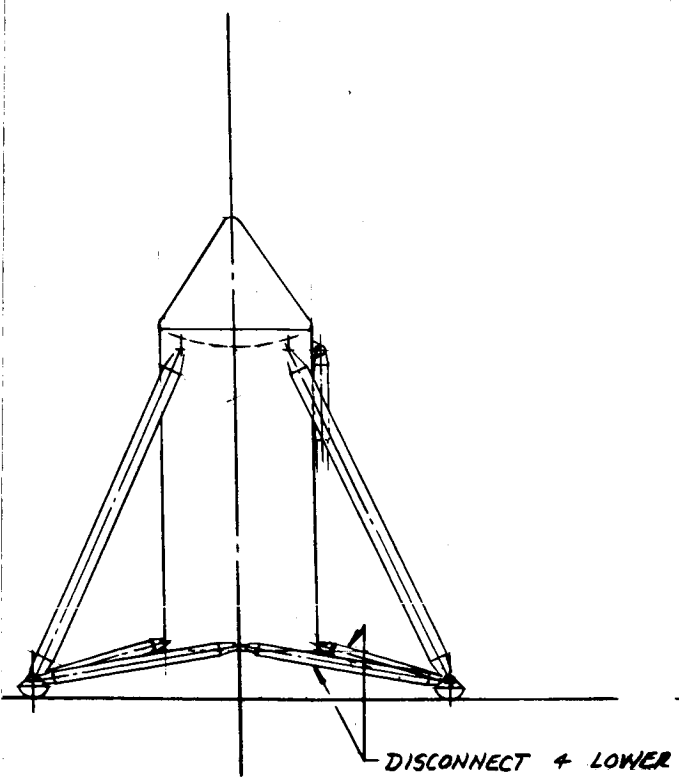

TA 84
LANDING GEAR SUPPORT FRAME
7°
-Y
50
LANDING GEAR STROKED
154
FUEL SUMP TANK
BEAM 5
SEXTANT V
SEXT V
OXIDIZER TANK
BEAM 4
SEXTANT IV
BEAM 3
BOTTOM VIEW



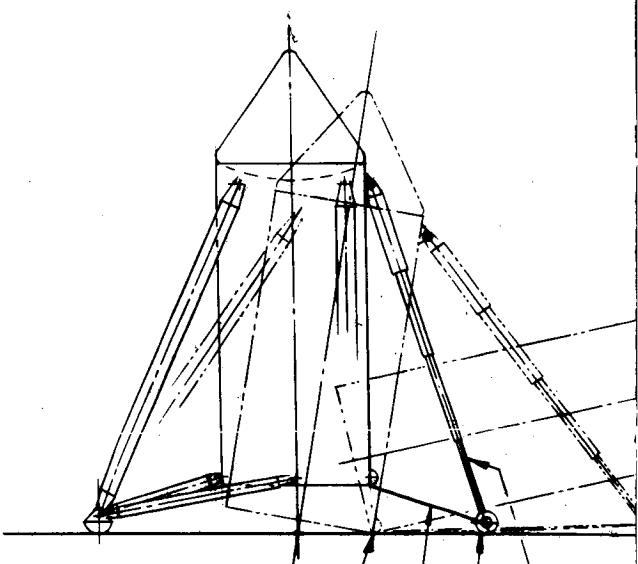
K20 & MODS	R. DAILEY	NORTH AMERICAN AVIATION, INC. SPACE and INFORMATION SYSTEMS DIVISION 12214 LAKEWOOD BLVD., DOWNEY, CALIFORNIA
APOLLO II STUDIES - MODIFIED APOLLO C/M & S/M AS LUNAR OCCUPANCY SHELTER		6052-6

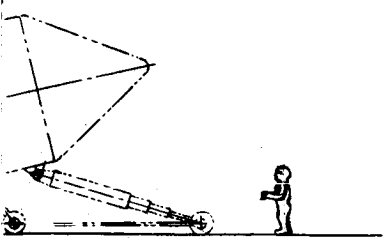






VEHICLE IN LANDED POSITION
$\frac{1}{100}$ SCALE





:OPING STRUT

STRUT)

S
NDING STRUTS.

NORTH AMERICAN AVIATION, INC.
12214 LAKEWOOD BLVD., DOWNEY, CALIFORNIA

APOLLO II STUDY - LUNAR SHELTER
LAY DOWN CONCEPT

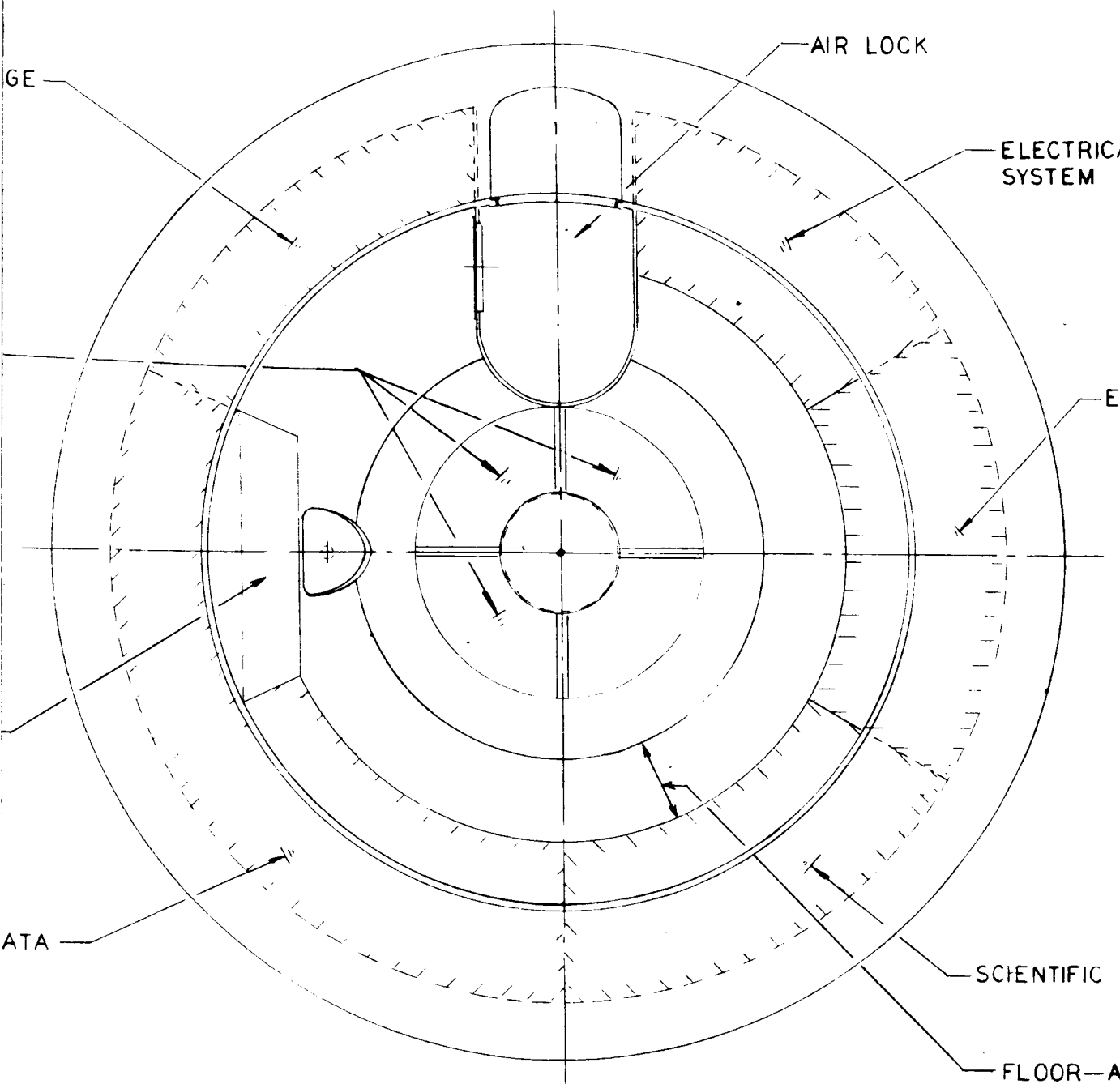
6052-7

STORA

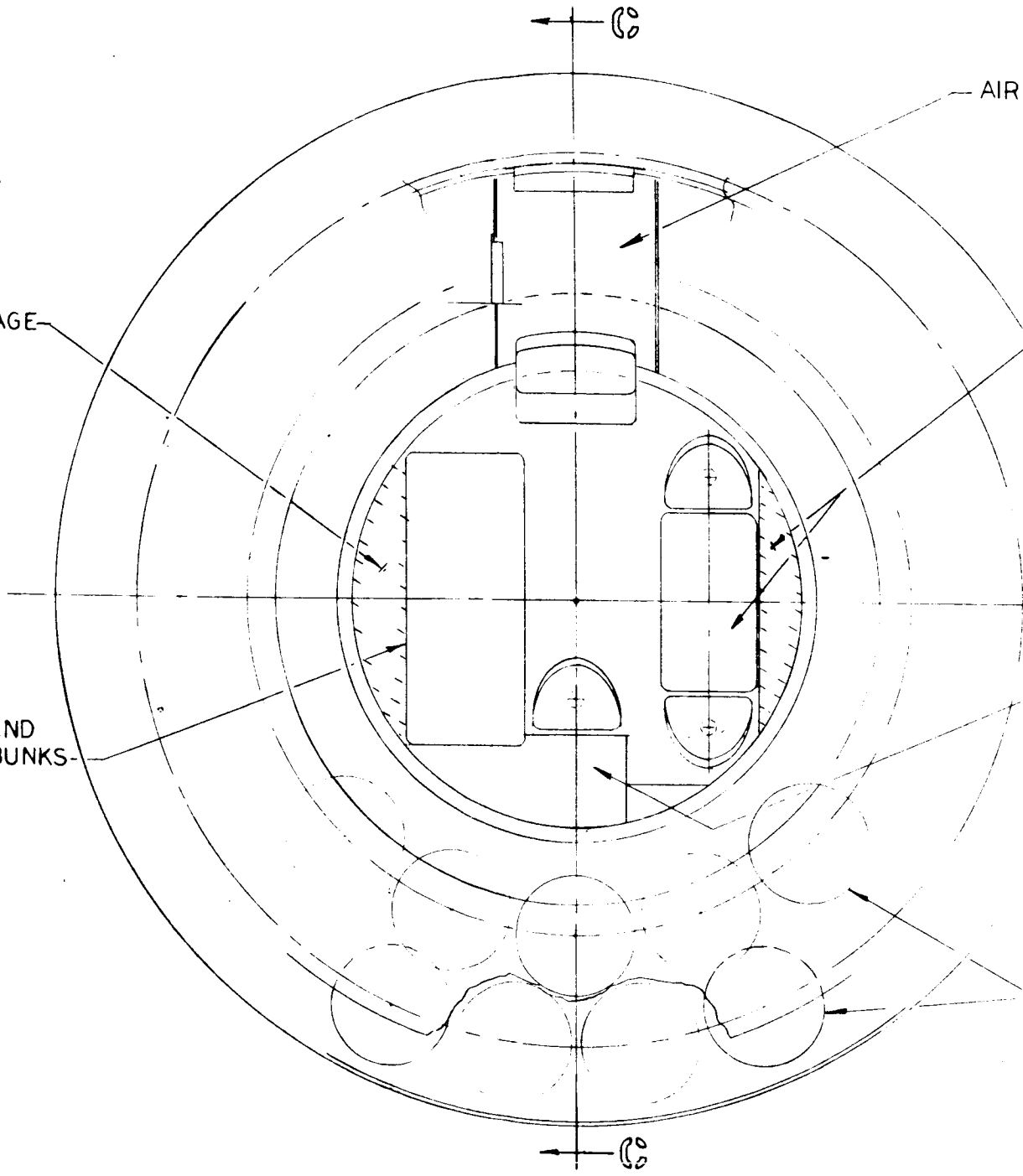
SPACE FOR
MISC EQUIP.

INSTRUMENT PANEL &
TELECOMMUNICATION CENTER-

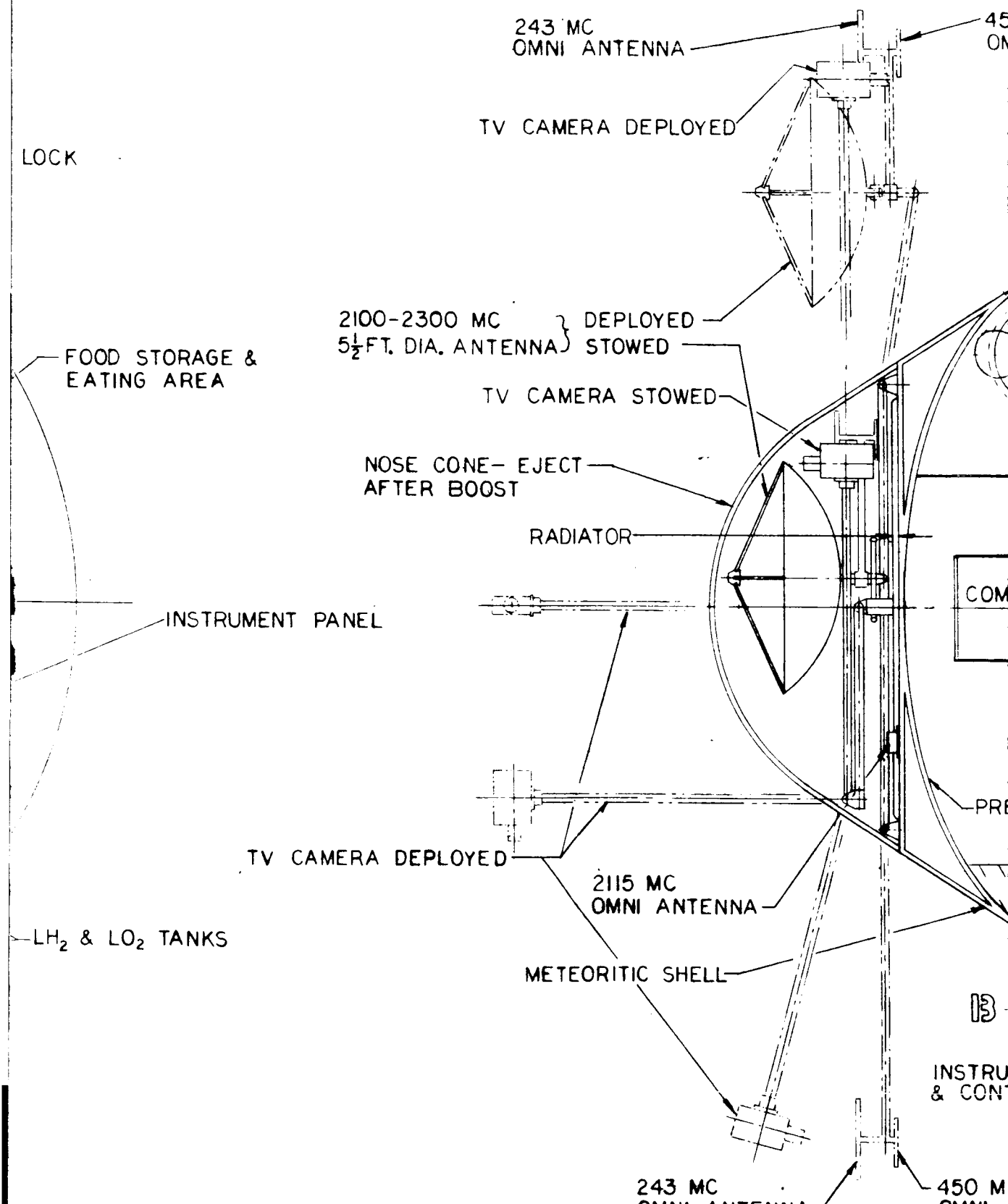
COMMUICATION & D
EQUIP.

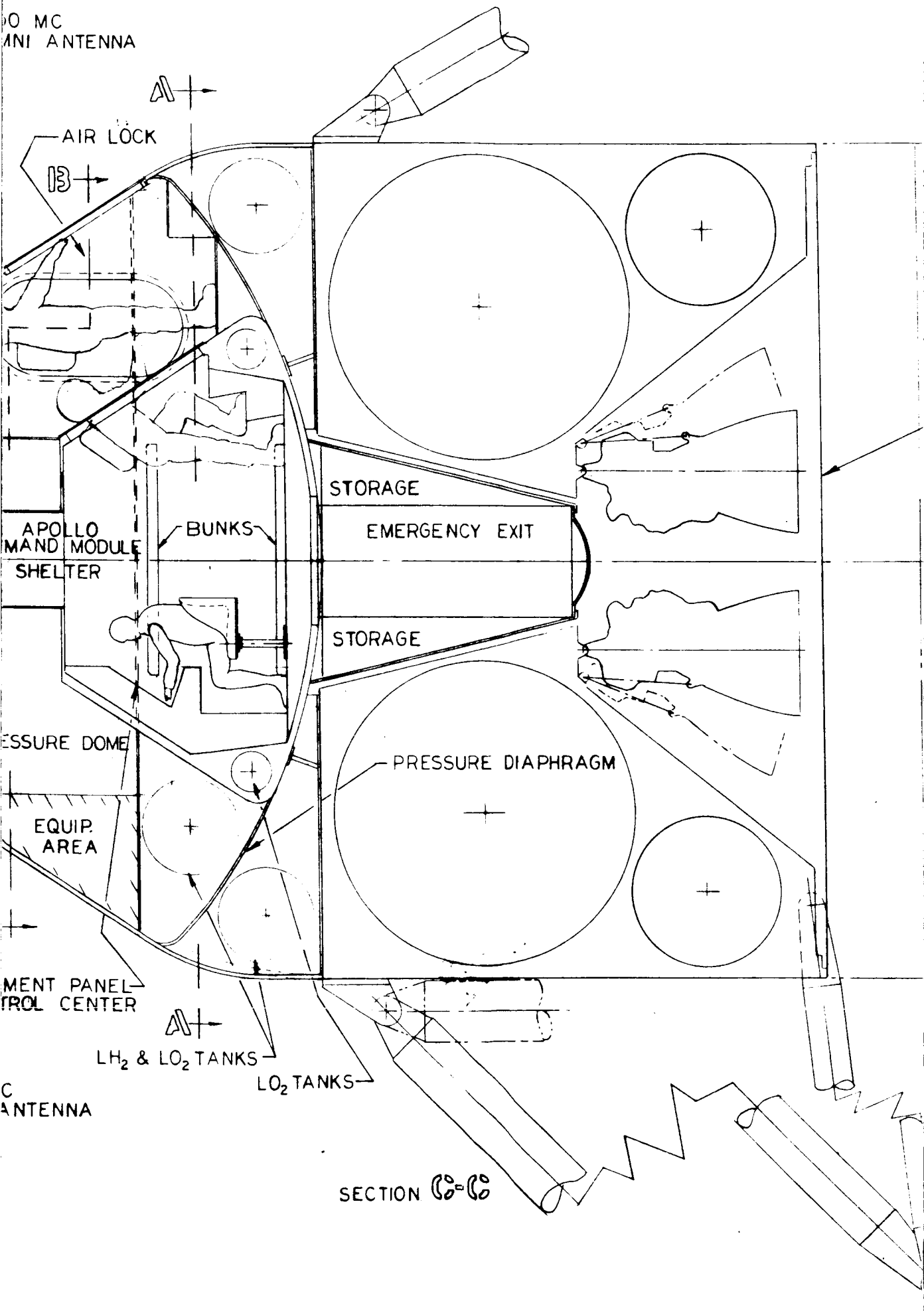


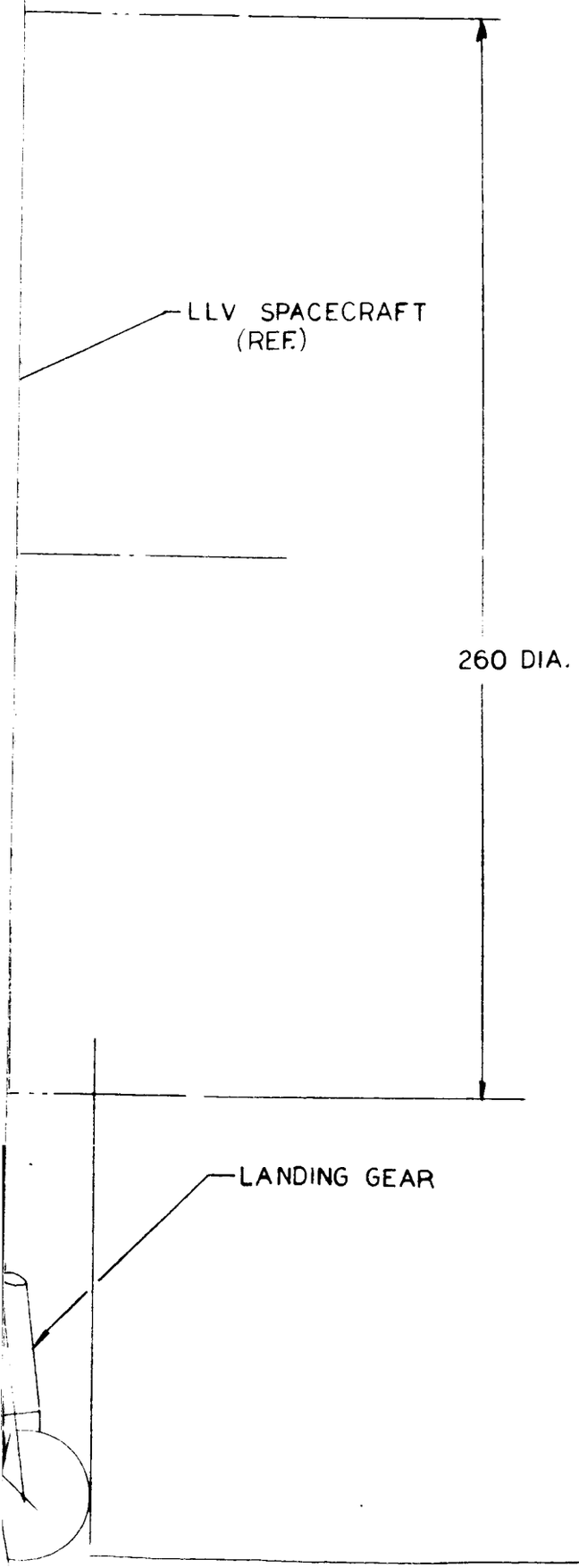
SECTION 13-13


AIR
AL POWER
STORAGE
CS
UPPER AND
LOWER BUNKS
EQUIP
CCESS TO
EMS EQUIP

SECTION A-A


243 MC
OMNI ANTENNA
TV CAMERA DEPLOYED
LOCK
2100-2300 MC
5½ FT. DIA. ANTENNA
DEPLOYED
STOWED
FOOD STORAGE &
EATING AREA
TV CAMERA STOWED
NOSE CONE- EJECT
AFTER BOOST
RADIATOR
INSTRUMENT PANEL
TV CAMERA DEPLOYED
2115 MC
OMNI ANTENNA
LH2 & LO2 TANKS
METEORITIC SHELL
243 MC
OMNI ANTENNA


O MC
NI ANTENNA
A
AIR LOCK
B
STORAGE
EMERGENCY EXIT
APOLLO
MAND MODULE
SHELTER
BUNKS
STORAGE
ESSURE DOME
PRESSURE DIAPHRAGM
EQUIP.
AREA
MENT PANEL
TROL CENTER
A
LH2 & LO2 TANKS
LO2 TANKS
C
ANTENNA
SECTION C-C



SCALE 1/20	DR. Ross Barr	NORTH AMERICAN AVIATION, INC.
	DATE 9-28-63	
	CHECK	

2-3 MAN COMMAND MODULE SHELTER
APOLLO VEHICLE- G5 - LLV LANDED

6052-8

Primary ingress and egress from the shelter is through an airlock located in the side of the shelter. This permits access to the annular habitable area of the shelter. In addition, a separate door is located in the side of the Apollo interior shelter to provide redundancy to the habitable area in case of loss of pressure in the annular space. Emergency ingress and egress capability is provided by an airlock located below the floor level in the LLV spacecraft.

STRUCTURES

The structural analysis and design of lunar shelter shell construction, considering the lunar criteria of meteoroid penetration, radiation, temperature variations and vacuum atmosphere, are presented in the following sections for each of the three early shelter concepts. In addition, dynamic and structural analysis of 3, 4 and 5 legged SM landing gear configurations are discussed.

Shelter Concepts

Shelter Concept 1

The structural design of this configuration consists of modification of Apollo I to provide for prolonged lunar stay time. As a consequence, the lunar environment criteria of meteoroid penetration, radiation, thermal control and pressurization for lunar vacuum become critical areas for structural investigation. The shelter is enclosed within the launch vehicle interstage, thus protected from airload pressures, and high temperatures due to aerodynamic heating. The only boost loads applied to the shelter structure will be the inertia loads of the shelter. This shelter is permanently attached to the CLEM for lunar alightment. Preliminary analysis indicate that lunar landing conditions are not critical structural criteria.

The type of shell construction considered for this shelter, as shown in Figure 54 for a maximum of twenty-one days lunar stay time, consists of multiwall construction to provide an efficient meteoroid shield. Due to the anticipated long duration soak at the extreme temperature of 260°F, Rene '41 sandwich construction was selected for the outer wall. "Zee" section stringers and frames, spot welded to the skins provide adequate shell stability without reducing the ballistic limit efficiency of the meteoroid shield. A parametric analysis of this type of construction was performed in order to determine the optimum structural arrangement of frame and stringer combination in terms of the allowable axial load intensity. Results of this study shown in Figure 55 indicate a minimum .008 manufacturing gage is feasible for boost loads. The inner structural wall forms the pressure vessel, as used on Apollo I, and consists of an aluminum honeycomb construction. Gage sizes of the facing sheets had to be increased to provide

adequate meteoroid protection. Minor modifications of equipment support structure and attachments will also be required on the inner facing sheets. Super insulation is provided between the inner and outer wall forming the thermal barrier to provide suitable living and operational environment inside the shelter. Both the inner and outer walls are held together by a low modulus of elasticity material (fiberglass) to allow the outer shell to expand and contract without inducing appreciable load interaction upon the shell. Figure 56 shows the required face sheet gages as a function of lunar stay time.

The structural design criteria need for this concept is presented below.

1) Safety Factors applied to:

   a. primary structure: 1.5 times limit load equal ultimate load.
   b. pressure vessels: 2.0 times limit pressure equals proof pressure.

2) Maximum cabin differential pressure:

   p = 7 psi (limit); 14 psi (proof)

3) Boost Loads:

   a. Max $q\alpha$ condition:

      N_x = 2.0 g (limit axial)
      N_y = .4 g (limit lateral)

   b. End of Stage I condition

      N_x = 6.0 g (limit axial

4) Lunar Landing Conditions:

      N_x = 3.0 earth gravities (limit axial)

5) Lunar Environment:

   a. The extreme temperature environment anticipated is approximately -250°F to +260°F

   b. The meteoroid penetration model, including secondary lunar ejecta per NASA RFP No. 16-1132 was used.

      The required equivalent single skin thickness was determined from the following relationship:

$$t^3 N = 4 \times 10^{-17}$$

$$N = \frac{1 - P_o}{AT}$$

where: t = equivalent single skin thickness (CM)
N = number of penetrations
P_o = probability factor (.999 during occupancy) (.9 during standby)
A = surface area (square meters)
T = time (seconds)

Figure 57 shows the required equivalent single skin thickness for aluminum and steel material. The ballistic limit efficiency factor used was 8.8 considering a void between the steel sandwich and the insulation used as a filler between the inner and outer wall. The aluminum honeycomb panel was considered as a single sheet with no contribution toward increasing the ballistic efficiency.

Shelter Concept 2

Structural design of this configuration utilizes the structural concept developed in Concept 1. Similar multi-wall construction is used for meteoroid protection for the maximum of 150-day lunar stay time, shown in Figure 58. The outer structural wall constructed of steel sandwich with the reinforced stringers and frames are designed for boost loads. Critical conditions for shell stability consists of external collapsing airload pressures combined with axial load intensity. Due to the longer standby and occupancy desired, the required face sheet gages were plotted as a function of lunar stay time (Figure 59).

The structural criteria for this shelter is similar to that used in the Concept 1 shelter except for the boost loading conditions and lunar alightment. Safety factors, cabin differential pressure and the lunar environment criteria is identical to the CLEM-Landed Concept I Shelter.

Boost loads:

a. Max. $q\alpha$ condition (limit)

N_x = 2.0 g (axial)
N_y = .4 g (lateral)
q = 800 ##/ft^2
$q\alpha$ = 120 lb-rad/ft^2
$c_{n\alpha}$ = .03
c_d = .75

b. End of Stage I condition

N_x = 6.0 g (limit axial)

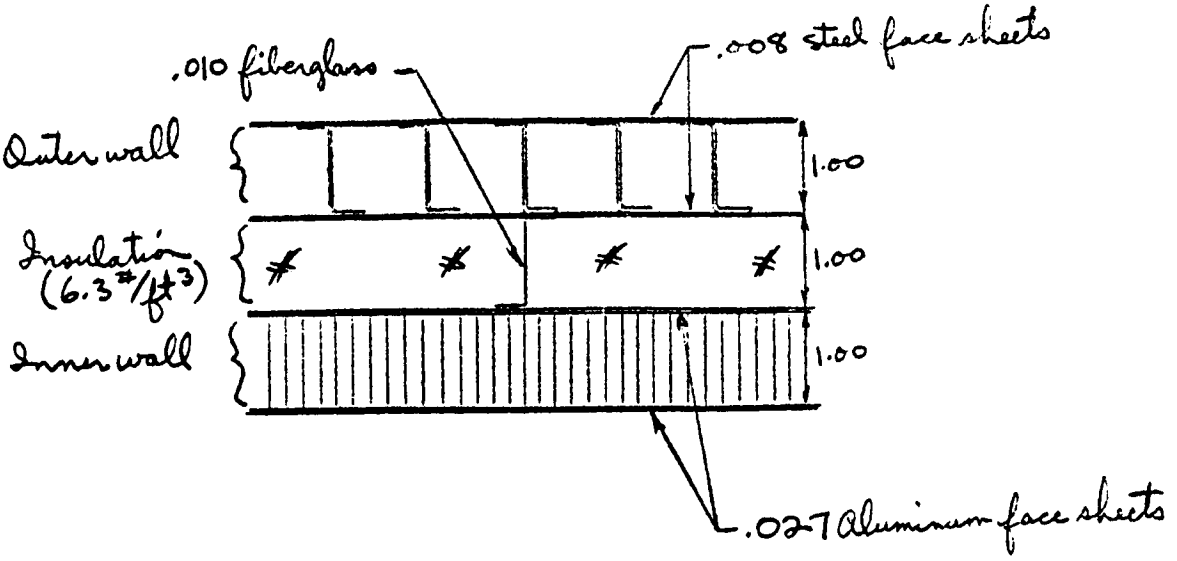


Figure 54 . Shell Construction for 21-Day Lunar Stay

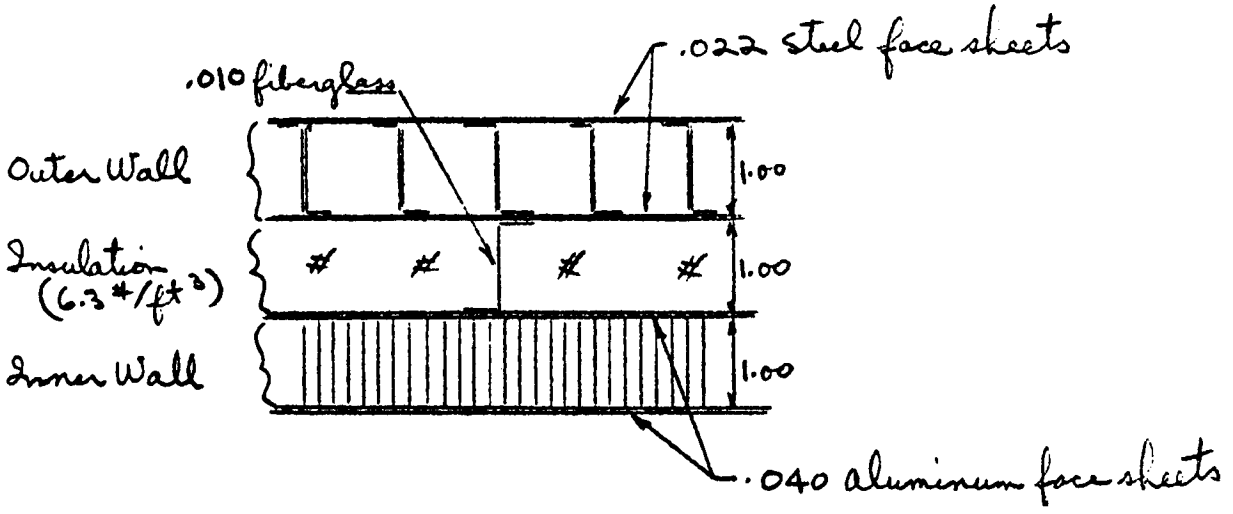
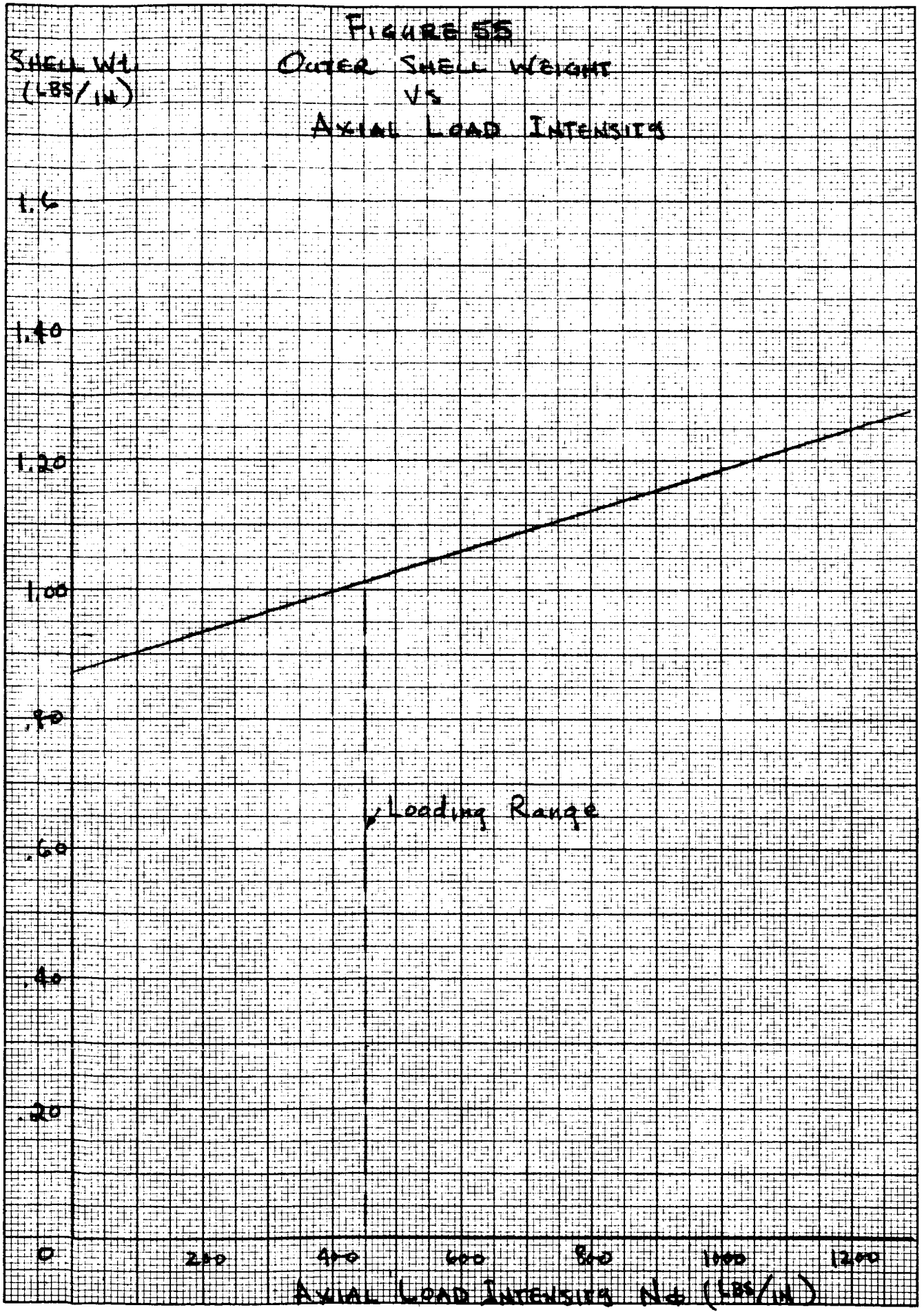
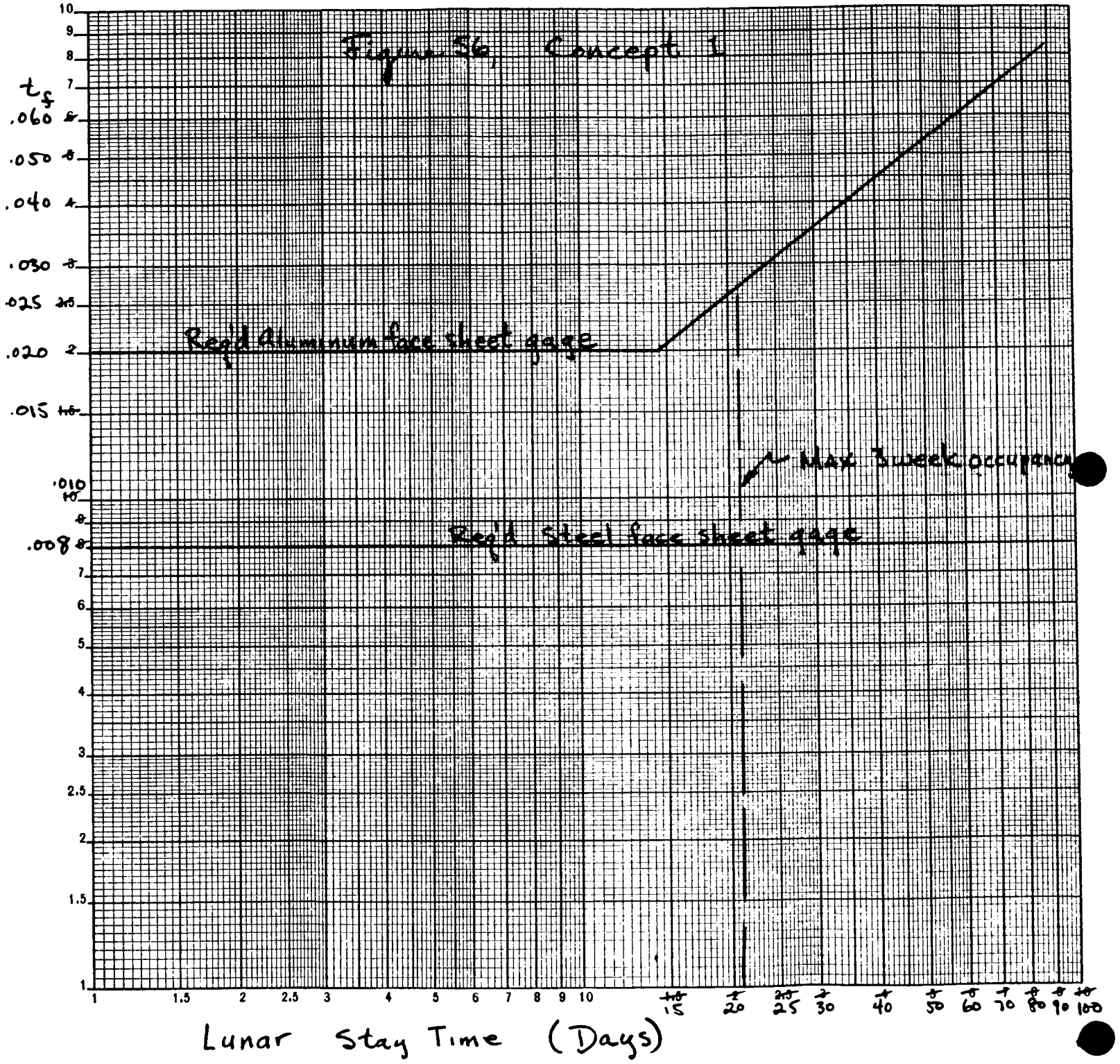
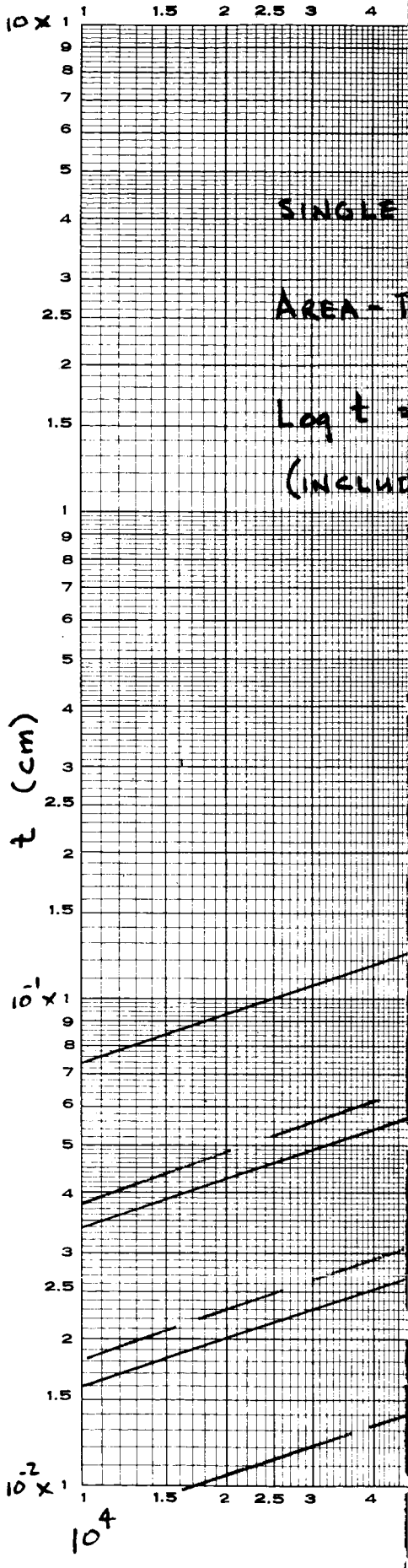
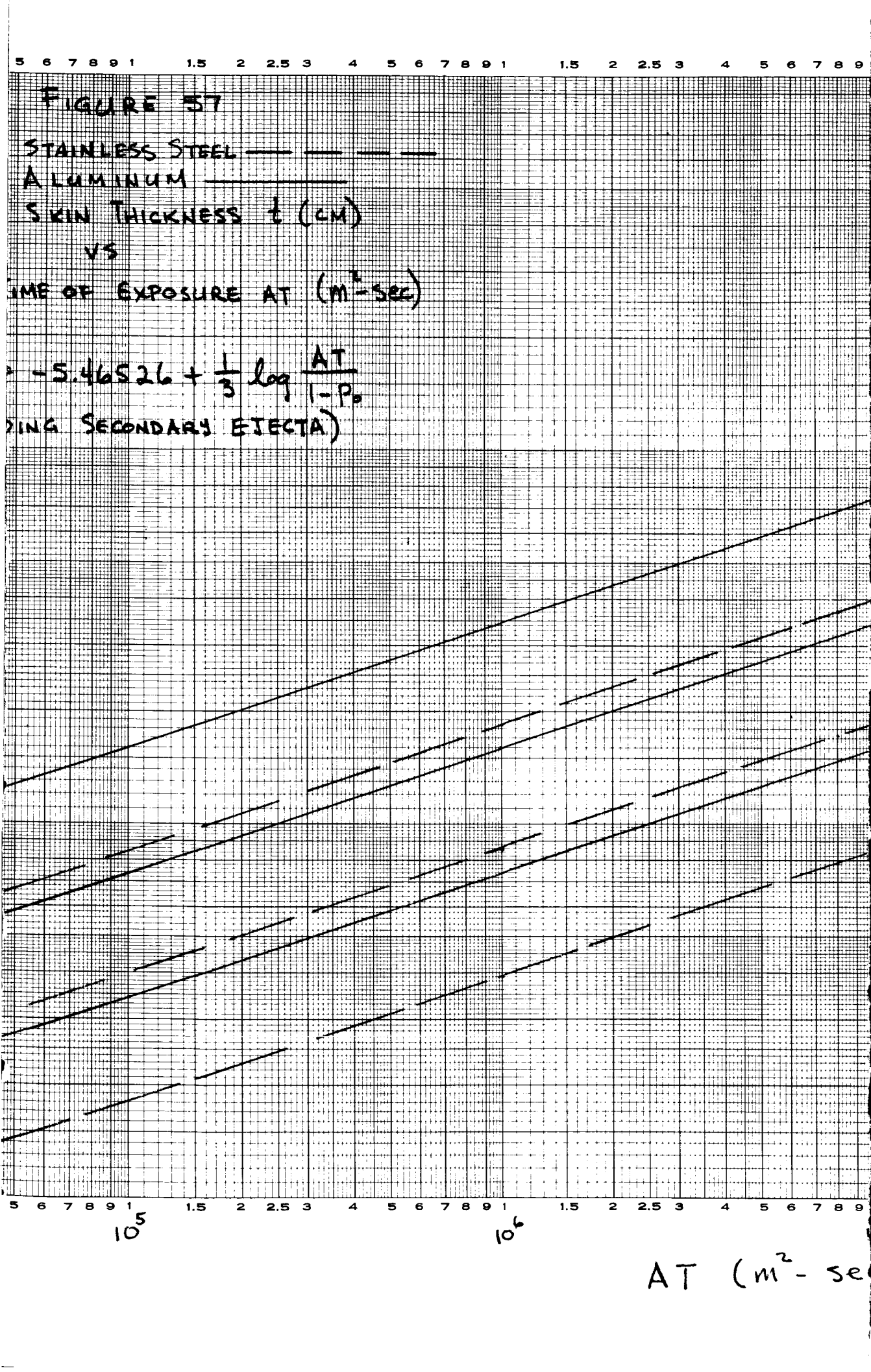


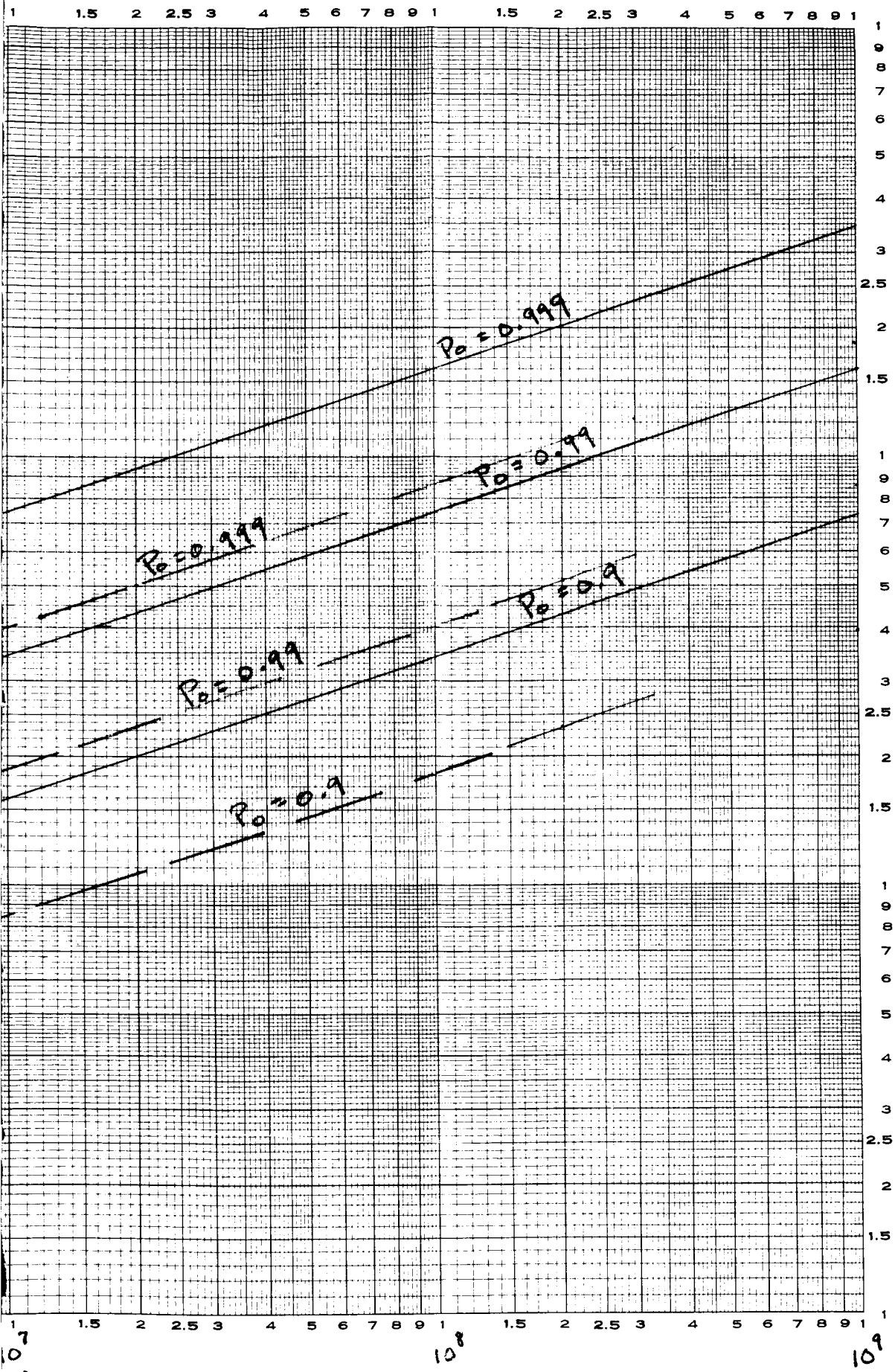
Figure 58 . Shell Construction for 150-Day Lunar Stay


FIGURE 55
OUTER SHELL WEIGHT
VS
AXIAL LOAD INTENSITY
SHELL WT. (LBS/IN)
1.6
1.40
1.20
1.00
.80
.60
.40
.20
0
200
400
600
800
1000
1200
Loading Range
AXIAL LOAD INTENSITY Nφ (LBS/IN)

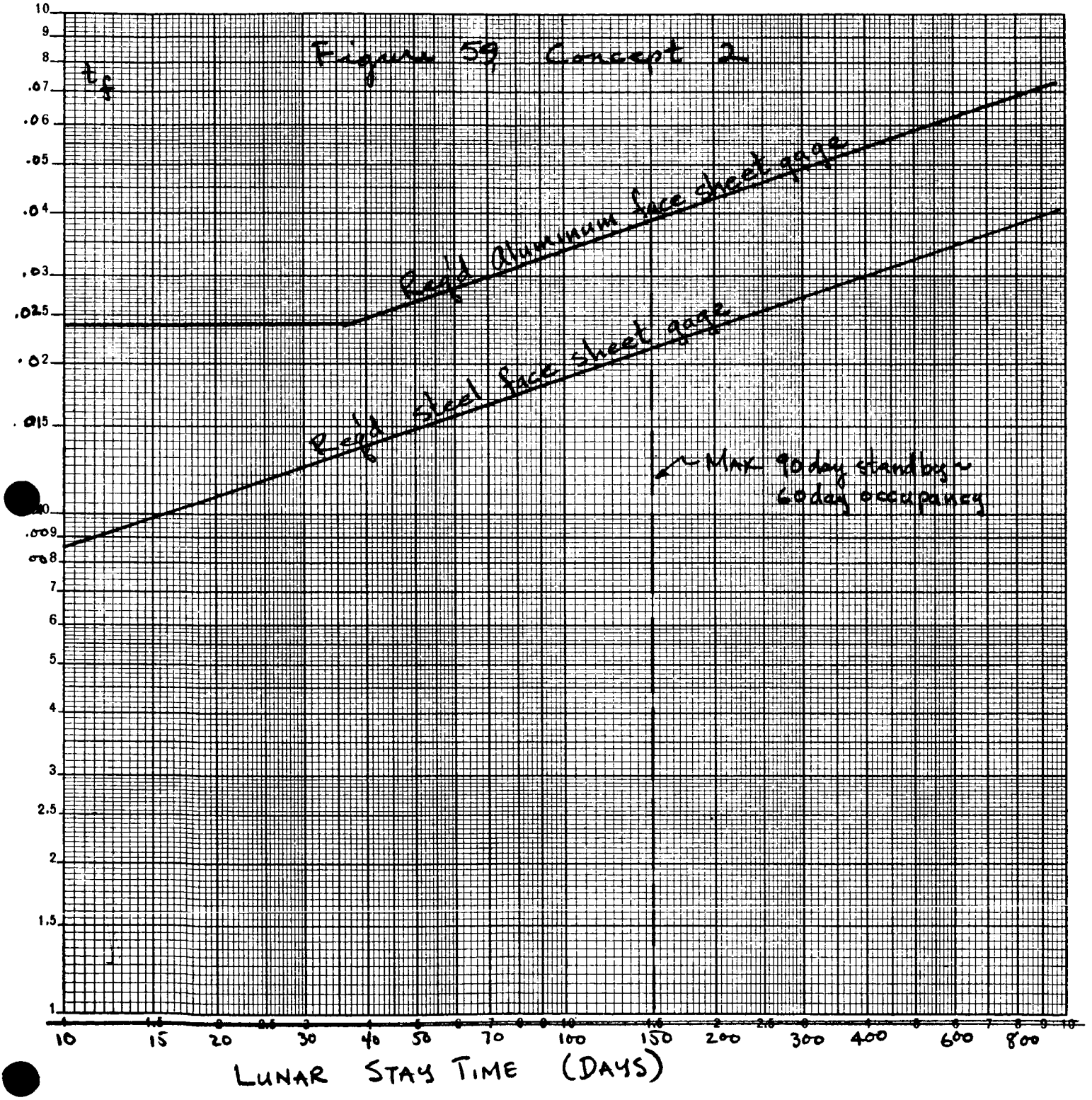

Figure 56, Concept 1
t_f
.060
.050
.040
.030
.025
.020
.015
.010
.008
Reqd Aluminum face sheet gage
Reqd Steel face sheet gage
Max 3 week occupancy
1
1.5
2
2.5
3
4
5
6
7
8
9
10
15
20
25
30
40
50
60
70
80
90
100
Lunar Stay Time (Days)


SINGLE
AREA - T
Log t =
(INCLUD
t (cm)
10 X
10^{-1} X
10^{-2} X
1 1.5 2 2.5 3 4
10^4


FIGURE 57
STAINLESS STEEL
ALUMINUM
SKIN THICKNESS t (CM)
VS
IME OF EXPOSURE AT (M^2-SEC)
$= -5.46526 + \frac{1}{3} \log \frac{AT}{1-P_0}$
)ING SECONDARY EJECTA)
5 6 7 8 9 1 1.5 2 2.5 3 4 5 6 7 8 9 1 1.5 2 2.5 3 4 5 6 7 8 9
10^5
10^6
AT (m^2- se



-203- -204- SID 63-1251


Figure 59 Concept 2
t_f
Weld Aluminum face sheet gage
Weld Steel face sheet gage
Max 90 day standby
60 day occupancy
10
9
8
.07
.06
.05
.04
.03
.025
.02
.015
.009
.008
7
6
5
4
3
2.5
2
1.5
1
10
15
20
30
40
50
70
100
150
200
300
400
600
800
LUNAR STAY TIME (DAYS)

Lunar landing conditions

N_{χ} = 3.0 Earth gravities (limit)

Coefficient of friction μ = 1.00

Shelter Concept 3

This configuration considers a modified command module shelter mated with the proposed LLV/LLM. A shroud enclosing the command module was pressurized and thermally controlled to contain additional storage and equipment. From the previous concepts it was noted that for extended long term standby and occupancy time, the meteoroid shield requirements become critical structural design criteria. The external shell of this concept, as in Concept 2, was found to be critical for the meteoroid penetration requirements.

Lunar Shelters LOP

The Lunar Occupancy Payload has been updated to provide for critical meteoroid protection requirements using the same external shell construction as developed for the Apollo I SM-Landed Concept 2 shelter.

Landing Gear

Dynamic Analysis

The following paragraphs discuss the parameters found to primarily affect landing stability and, therefore, landing gear design. Also, significant results of past studies are related and general conclusions stated.

1. $\bar{r}$/R Ratio

In past studies it has been noted that by either decreasing a vehicle's radius of gyration, $\bar{r}$, or by increasing the vehicle's leg spread, R, the critical velocity-stability profile is generally enhanced. Treating the critical two-dimensional landing in a simplified fashion as is illustrated in Figure 1, the ratio of $\bar{r}/R$ can be seen to play an important role in vehicle stability. The analytical relationship illustrated in Figure 1 was based on the following assumptions:

(1) The vehicle does not change significantly in geometry during the impact.

(2) Second impact does not occur during the initial impact.

(3) The ground force may be treated as a net resultant force with a constant lever arm about the C.G.

In general, it will be found that the resultant ground force will have a moment arm R about the C.G. larger than the vehicle radius of gyration $\bar{r}$. In addition it has been observed from computer predicted angular acceleration histories that the first impact imparts the overturning energy (for critical landings) and the second impact resists overturn. Therefore, if the ratio of $\bar{r}/R$ during the first impact is normally smaller than 1.0 for most vehicles, then increases in $\bar{r}$ or decreases in R will degrade stability by allowing a larger percentage of the initial impact energy to be converted to overturn rotation as is illustrated in Figure 60. Figure 61 presents computer predicted results for varying $\bar{r}$. It can be seen that stability is degraded with increasing value of $\bar{r}$. The addition of a landing leg to a vehicle will add stability and allow the gear spread to be decreased somewhat. The decrease in R will, however, cause an increase in the $\bar{r}/R$ ratio and will tend to offset a portion of the stability gained from the added leg.

2. Deceleration Magnitude

It has been observed that varying the magnitude of the leg decelerator causes a shift in the stability profile which appears to be related to the energy level directed normal to the impact surface. It can be shown that for an impact, assuming non-changing geometry and a constant effective line of application of deceleration force, proportional amounts of the original energy directed normal to the surface will be transformed to rotation, remain in translation, and be absorbed, see Figure 60 It has also been noted that for landing velocities near parallel with the surface overturn is apparently independent of the deceleration magnitude. Based on these observations the following critical velocity approximations were derived to correct a given stability profile for a change in decelerator magnitude.

$$V_{V_2} = \frac{V_{V_1}\left(\cos\theta\sqrt{\frac{F_2}{F_1}} + \sin\theta\tan\theta\right) + V_{L_1}\left(\cos\theta\tan\theta - \sin\theta\sqrt{\frac{F_2}{F_1}}\right)}{\cos\theta + \sin\theta\tan\theta}$$

$$V_{L_2} = \frac{V_{V_1}\sin\theta\left(1-\sqrt{\frac{F_1}{F_2}}\right) + V_{L_1}\left(\cos\theta + \sin\theta\tan\theta\sqrt{\frac{F_2}{F_1}}\right)}{\cos\theta + \sin\theta\tan\theta}$$

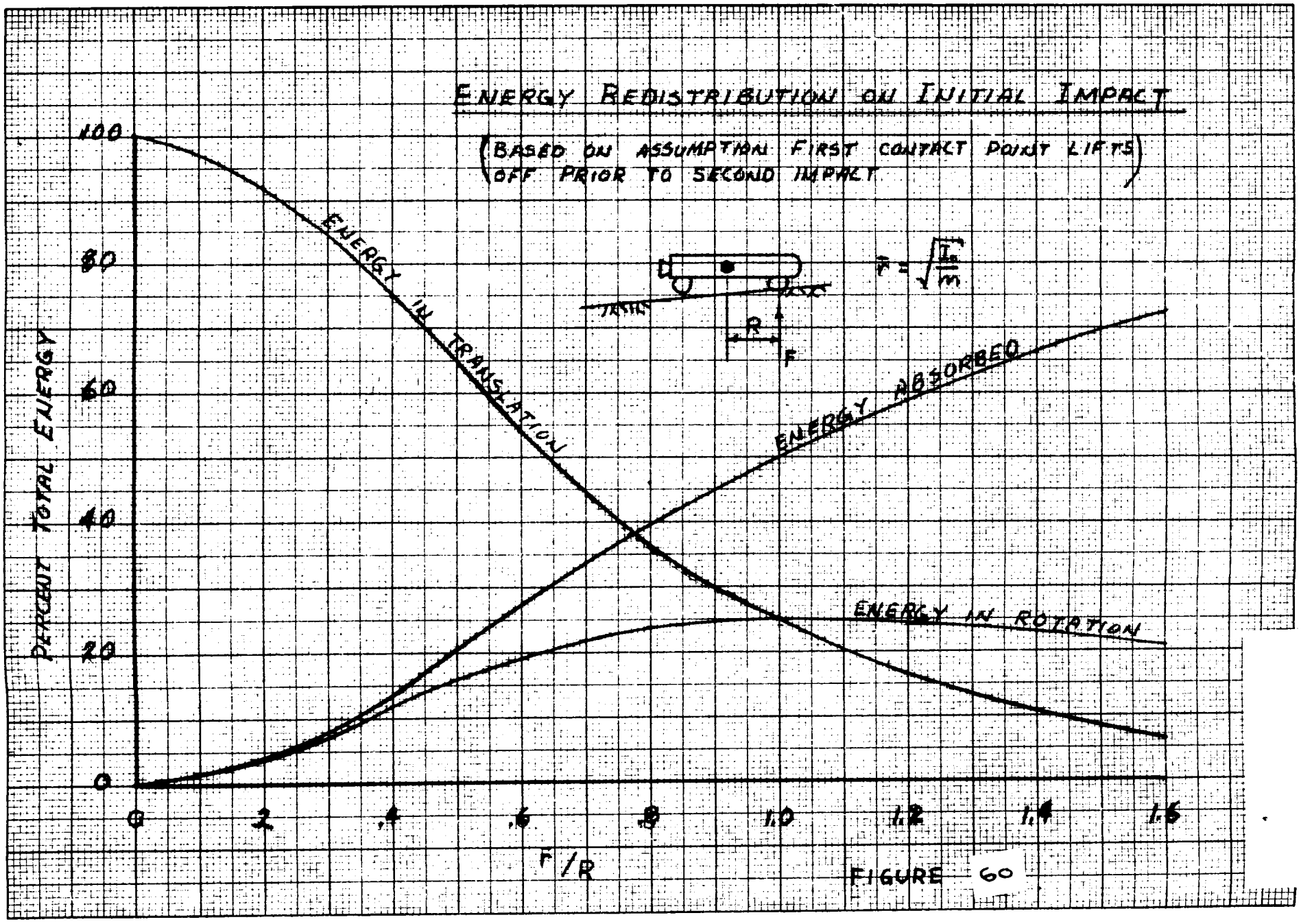
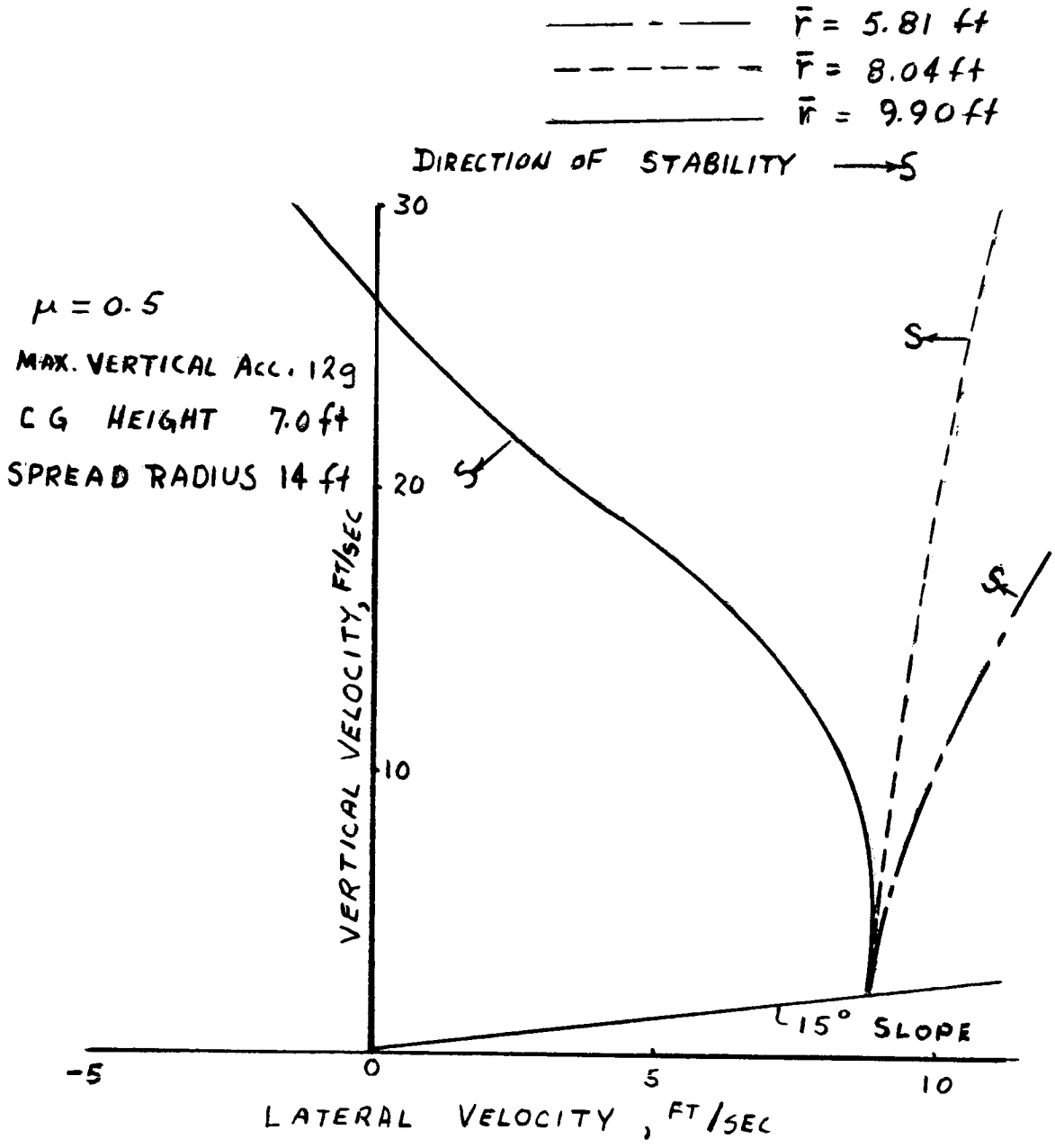

ENERGY REDISTRIBUTION ON INITIAL IMPACT
(BASED ON ASSUMPTION FIRST CONTACT POINT LIFTS OFF PRIOR TO SECOND IMPACT)
$F = \sqrt{\frac{I_o}{m}}$
R
F
100
80
60
40
20
0
PERCENT TOTAL ENERGY
ENERGY IN TRANSLATION
ENERGY ABSORBED
ENERGY IN ROTATION
0
.2
.4
.6
.8
1.0
1.2
1.4
1.6
F/R

FIGURE 60



EFFECT OF RADIUS OF GYRATION ON STABILITY

FIGURE 61

Where: 1. V_{v1} and V_{L1} are the vertical and lateral velocities causing overturn using a decelerator of magnitude F_1.

2. V_{v2} and V_{L2} are the predicted vertical and lateral velocities causing overturn using a decelerator magnitude of F_2.

Figures 62 and 63 show the critical velocity-stability profile for a three and four leg version of the LEM, respectively. Also shown on these figures are the corresponding stability profiles for a decelerator of one-half the original magnitude and the predicted shift of these profiles. Note that the stability profile of Figure 62 bends to the left while that of Figure 63 bends to the right. For low velocities, the direction in which the stability profile bends is related to the ratio of leg spread to radius gyration. When this ratio is large, the bend is to the right. At higher landing velocities a second phenomenon related to multiple contacts will cause the curve to bend to the right. That is to say, if during the initial contact the stabilizing forward legs make contact, then stability is enhanced and the stability profile breaks to the right.

It can be seen then that softening the leg decelerators may or may not aid the vehicle's stability depending on the nature of the family of curves involved.

In addition to translational accelerations during landing impact, large objectionable angular accelerations are encountered. These loadings may be alleviated by the gear force-stroke design. Figure 64 presents computer predicted angular accelerations for two landing force-stroke designs. Large lever arm reactions occur at initial contact and by their very nature are quickly neutralized after which the ground reaction force operates with a reduced lever arm associated with a reversal in friction force. For this reason, a two staged decelerator results in reduced angular accelerations as illustrated in Figure 64.

3. Elastic Flexibility

For the most part, past analyses of vehicle landing stability have been made assuming that no portion of the impact is stored in recoil energy. In order to determine the magnitude of this effect an existing digital computer program was revised such that a spring could be incorporated in series with the previously treated plastic absorbers. The result of this study indicated that the elastic recoil energy degrades stability and a weight tradeoff study on increasing leg-spring stiffness and/or leg spread is likely in optimum landing gear design. Figure 65 illustrates the effect of spring stiffness on landing stability noted for a LEM vehicle.

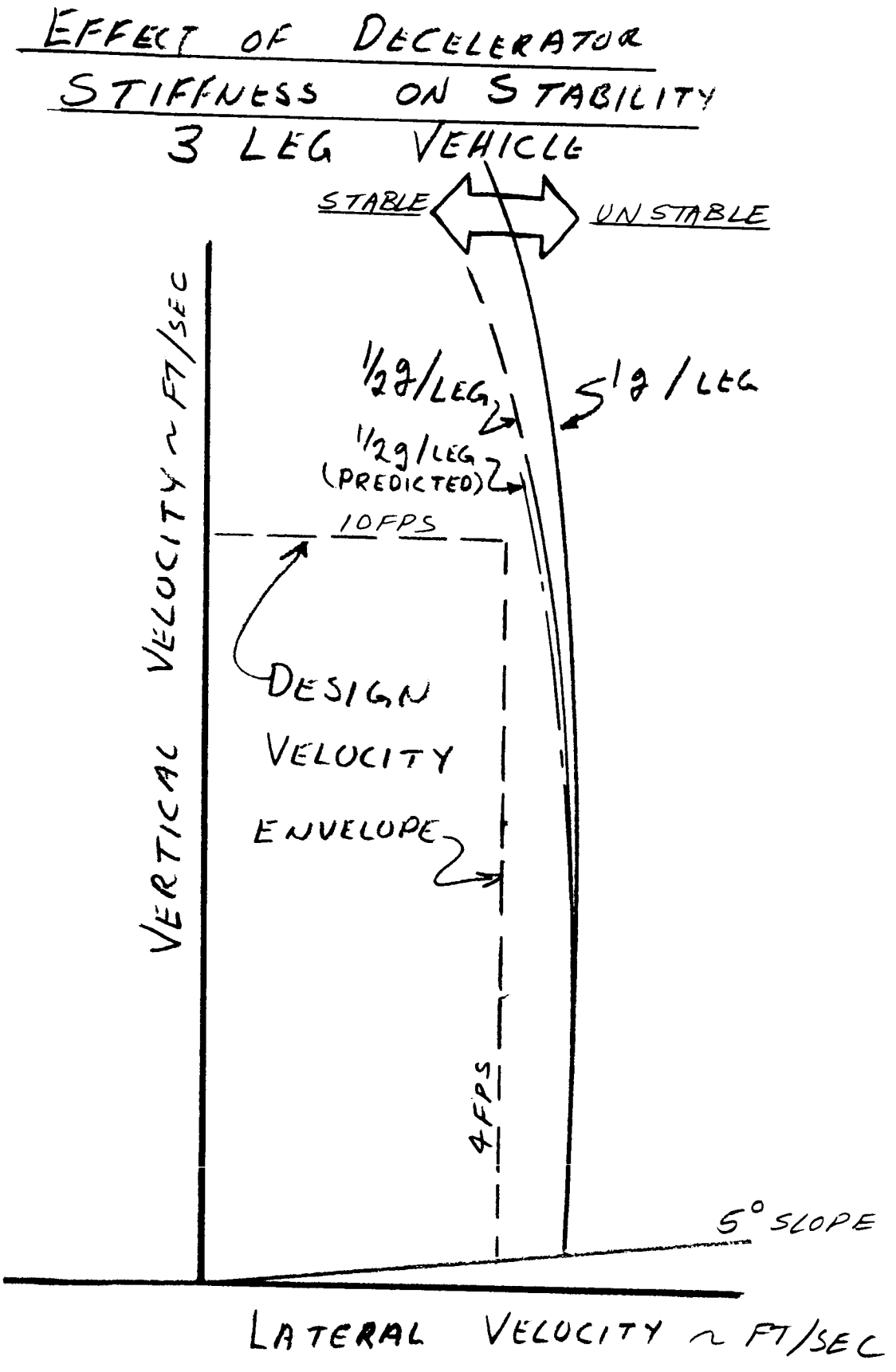
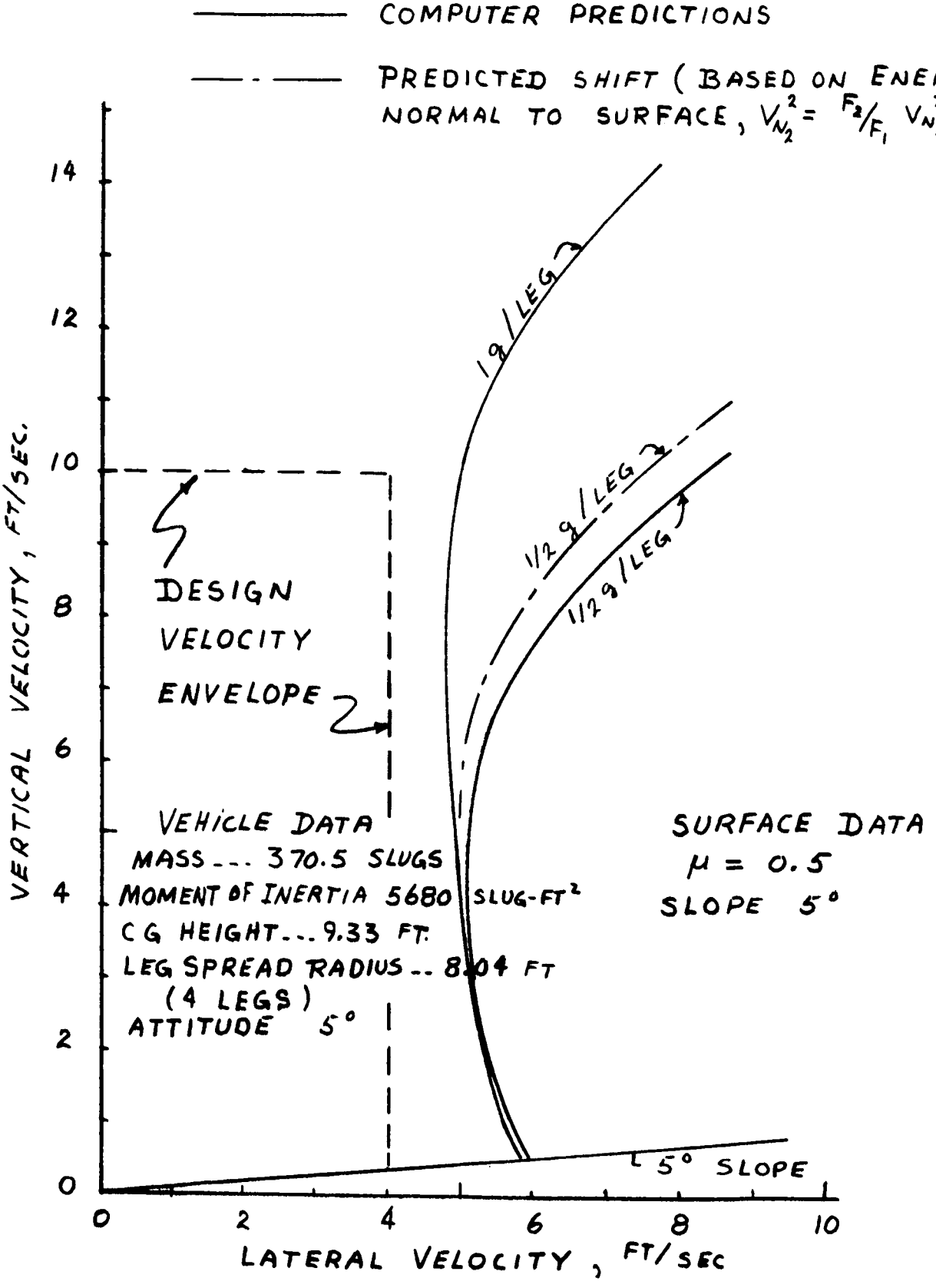

EFFECT OF DECELERATOR STIFFNESS ON STABILITY
3 LEG VEHICLE
STABLE
UNSTABLE
1/2g/LEG
5 g/LEG
1/2g/LEG (PREDICTED)
10FPS
DESIGN VELOCITY ENVELOPE
4FPS
5° SLOPE
VERTICAL VELOCITY ~ FT/SEC
LATERAL VELOCITY ~ FT/SEC

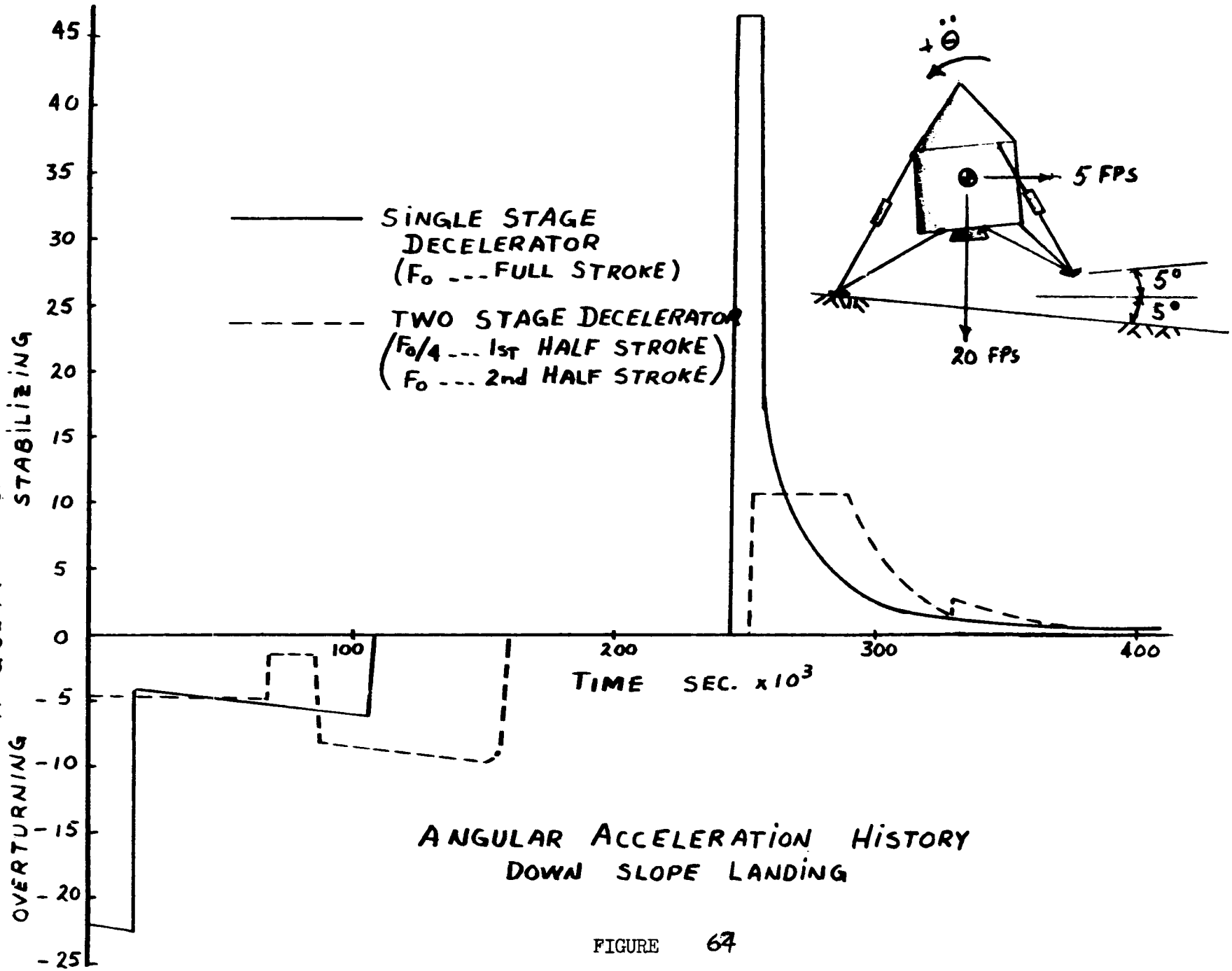
FIGURE 67



EFFECT OF DECELERATOR MAGNITUDE

FIGURE

SID 63-1251



ANGULAR ACCELERATION HISTORY
DOWN SLOPE LANDING

FIGURE 64

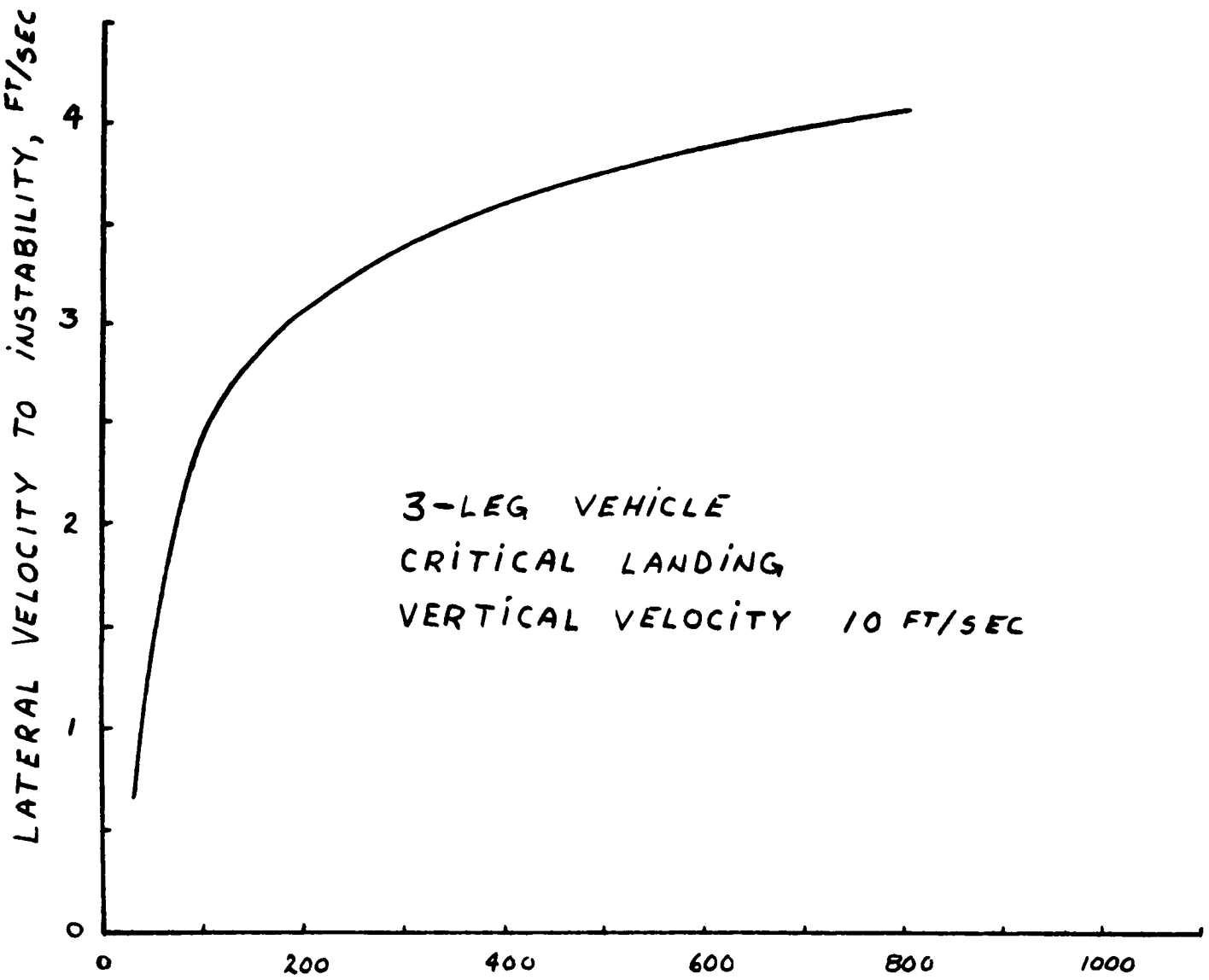


FIGURE 65 EFFECT OF LEG SPRING ON STABILITY

Although the spring stiffness study was limited both in time and scope, it may be concluded that elastic flexibility (1) generally results in degraded stability and (2) is an important parameter in arriving at an optimum gear design. Inasmuch as the required gear spread radius decreases with the number of landing legs as does the maximum loading per gear, it would be logical to expect that the spring stiffness effect will be less critical for the multi-leg configurations.

4. Size Effect

A comparison of the stability characteristics of two or more vehicles of various sizes is possible utilizing dynamic similarity laws. Assume one of the vehicles is selected as a "base" vehicle and that the others are sealed up or down to this size. The basic quantities used in making this comparison are:

(1) mass scale - ratio of vehicle masses

$$\beta = M/M_{Base}$$

(2) length scale - ratio of C.G. height of vehicle to C.G. height of base vehicle

$$\alpha = h/h_{Base}$$

(3) Acceleration scale -

$$\gamma = 1.0$$

From these basic scaling factors, scaling laws may be determined for mass moment of inertia, velocity, spring stiffness, etc.

In this manner such parameters as $\bar{r}/R$ and spring stiffness may be utilized in determining which vehicle offers the most stability at this common "base" size. The next step is to scale the velocity back to the original true scale and relate this effect to the comparison made at the "base" vehicle size.

It will generally be found that, employing dynamic similarity laws, the smaller vehicles are very nearly scale models of large versions. The smaller vehicle will usually have a slightly larger moment of inertia than a scaled down vehicle. This in terms of comparative stability means, that if pertinent vehicle dimensions, g loading, and surface characteristics are scaled uniformly, the larger vehicle will be more stable. The reasons being: (1) The velocities causing instability are $\sqrt{\alpha}$ times greater.

(2) The moment of inertia of the scaled down vehicle being smaller than the smaller vehicle represents a decrease in $\bar{r}/R$ ratio and, therefore, an increase in stability.

5. Surface Slope and Vehicle Attitude

The leg spread requirements become more demanding as the initial (vehicle to surface) contact angle is increased. The misalignment angle is composed of two components: vehicle initial attitude and surface slope.

The vehicle attitude at touchdown will vary depending on the required maneuver, vehicle's control system, pilot's ability and visability. In past studies vehicle alignment has commonly been accounted for by allowing the vehicle to be rotated $\pm 5°$ from the intended orientation about any axis.

Crater walls and the more severe slopes in the lunar mountain ranges have been estimated not to exceed 5°. The possibility of local mounds and depressions exceeding the 5° prediction does exist, however, and any chosen design slope is naturally subject to criticism. In past studies, S&ID has used design surface slopes of 5, 10 and 15 degrees. Figures 66 and 67 present analog computer stability predictions for varying surface slope. In Figures 66 and 67 lateral velocity causing instability is plotted versus the surface slope for three and four legged vehicles.

6. Number of Landing Legs

A digital computer study has recently been conducted by S&ID to assess the comparative landing merits of a three, four and five leg version of an Apollo II lunar vehicle. The maximum vertical deceleration for this study was held constant at 3 earth "g" and the leg spread radius was altered until the computer analysis predicted stability throughout a given landing velocity envelope. Figure 68 and 69 illustrate the results obtained for each the three, four, and five legged vehicles using a friction coefficient of 0.5 and 1.0, respectively. Figure 70 presents a plot of required leg spread versus coefficient of friction for three, four, and five legged vehicles having similar stability. The leg spread radius required for the four leg vehicle, varying friction from 0.5 to 1.0, was 80.5% to 82.5% that required of the equivalent three leg vehicle. The five leg vehicle required from 70 - 72.5% the leg spread required for the equivalent three leg vehicle. The relationship observed was, however, for a given vehicle and will vary with: design velocity envelopes, vehicle mass and mass moment of inertia, deceleration magnitude, and gear spring stiffness.

The weight advantage associated with the number of landing legs will depend primarily on:

(1) The base diameter of the landing vehicle and the required leg spread radius.
(2) The structural arrangement of the landing vehicle's primary structure.

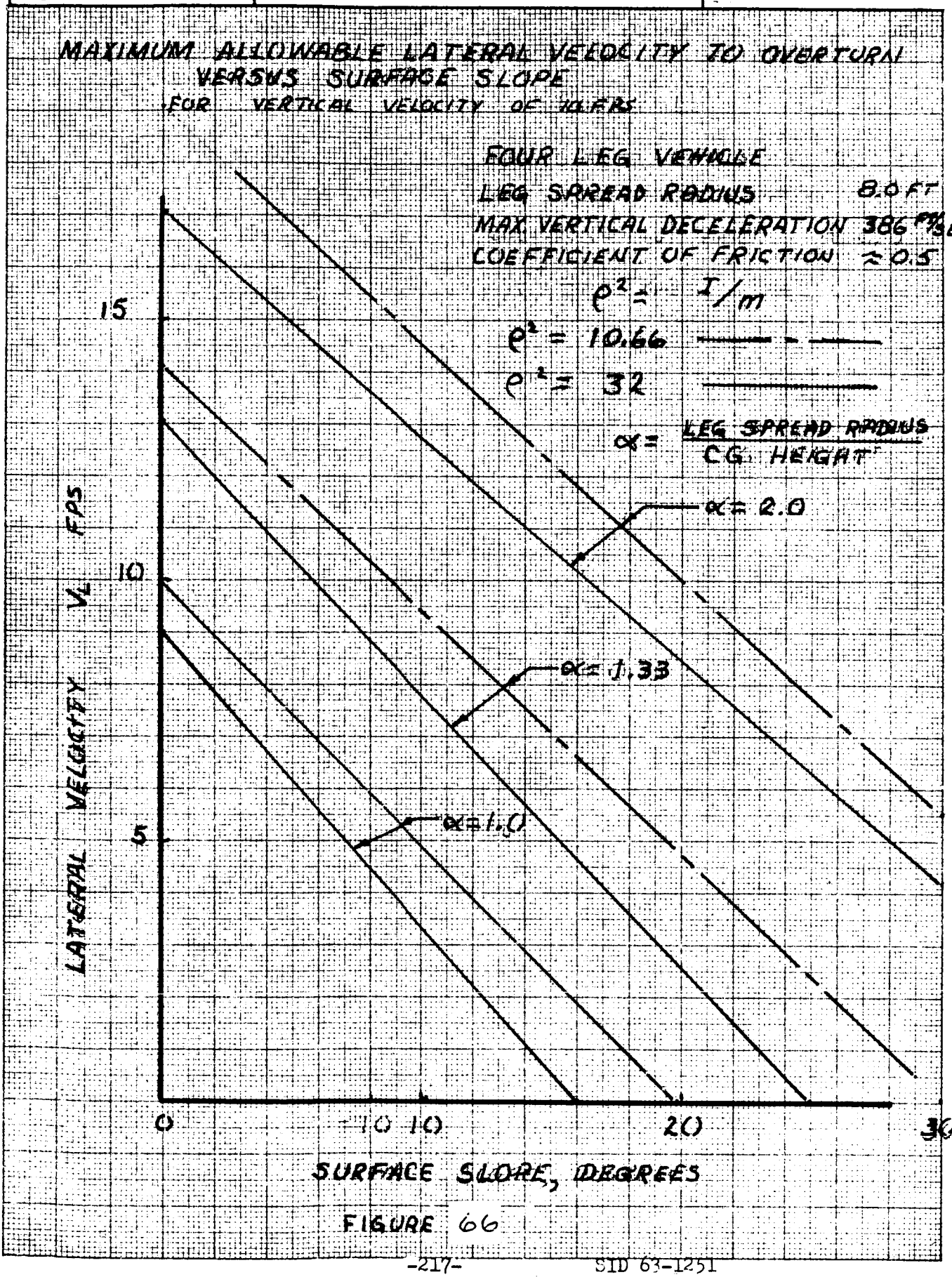

MAXIMUM ALLOWABLE LATERAL VELOCITY TO OVERTURN
VERSUS SURFACE SLOPE
FOR VERTICAL VELOCITY OF 10 FPS
FOUR LEG VEHICLE
LEG SPREAD RADIUS 8.0 FT
MAX. VERTICAL DECELERATION 386 FT/SEC²
COEFFICIENT OF FRICTION ≈ 0.5
ρ² = I/m
ρ² = 10.66
ρ² = 32
α = LEG SPREAD RADIUS / CG HEIGHT
α = 2.0
α = 1.33
α = 1.0
LATERAL VELOCITY VL FPS
15
10
5
0
-10
10
20
30
SURFACE SLOPE, DEGREES

FIGURE 66

MAXIMUM ALLOWABLE LATERAL VELOCITY TO OVERTURN VERSUS SURFACE SLOPE

(FOR VERTICAL VELOCITY OF 10 FPS.)

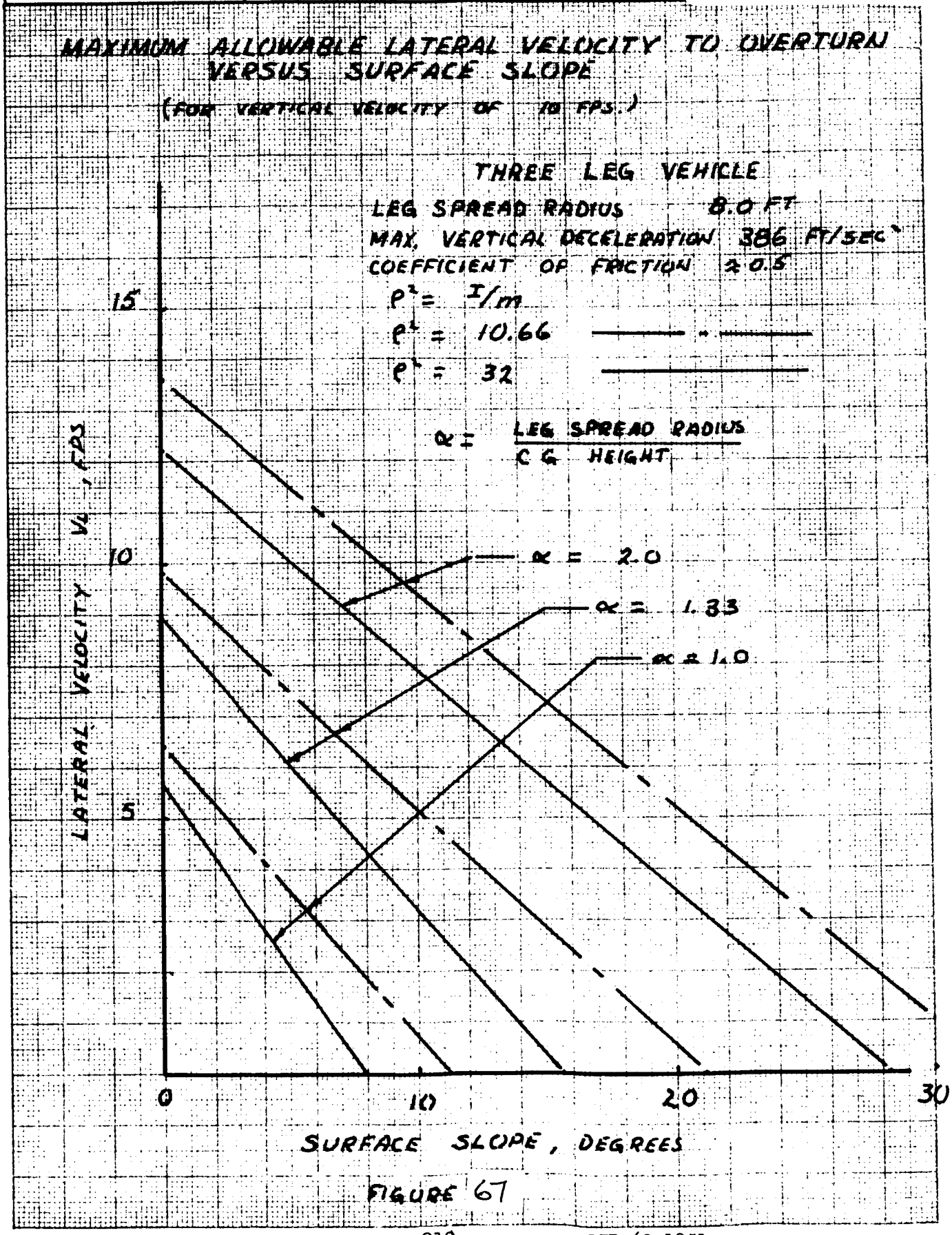


FIGURE 67

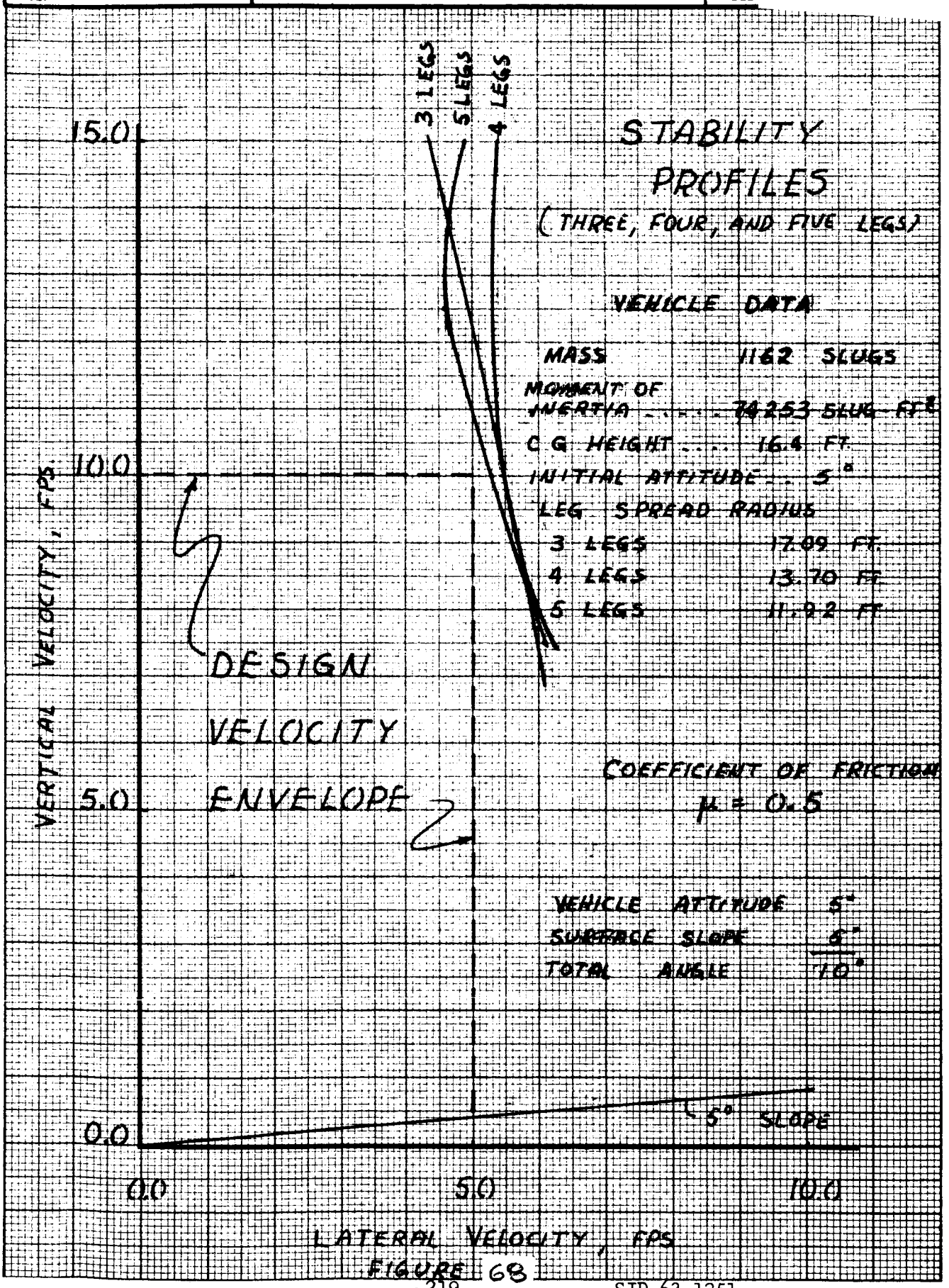

3 LEGS
5 LEGS
4 LEGS
STABILITY PROFILES
(THREE, FOUR, AND FIVE LEGS)
VEHICLE DATA
MASS 1162 SLUGS
MOMENT OF INERTIA ... 74253 SLUG-FT²
C G HEIGHT ... 16.4 FT.
INITIAL ATTITUDE ... 5°
LEG SPREAD RADIUS
3 LEGS 17.09 FT.
4 LEGS 13.70 FT.
5 LEGS 11.92 FT.
DESIGN VELOCITY ENVELOPE
COEFFICIENT OF FRICTION
μ = 0.5
VEHICLE ATTITUDE 5°
SURFACE SLOPE 5°
TOTAL ANGLE 10°
5° SLOPE
15.0
10.0
5.0
0.0
0.0
5.0
10.0
VERTICAL VELOCITY, FPS
LATERAL VELOCITY, FPS

FIGURE 68

PREPARED BY:

CHECKED BY:

DATE:

NORTH AMERICAN AVIATION, INC.
SPACE and INFORMATION SYSTEMS DIVISION

STABILITY PROFILES

(THREE, FOUR, AND FIVE LANDING LEGS)

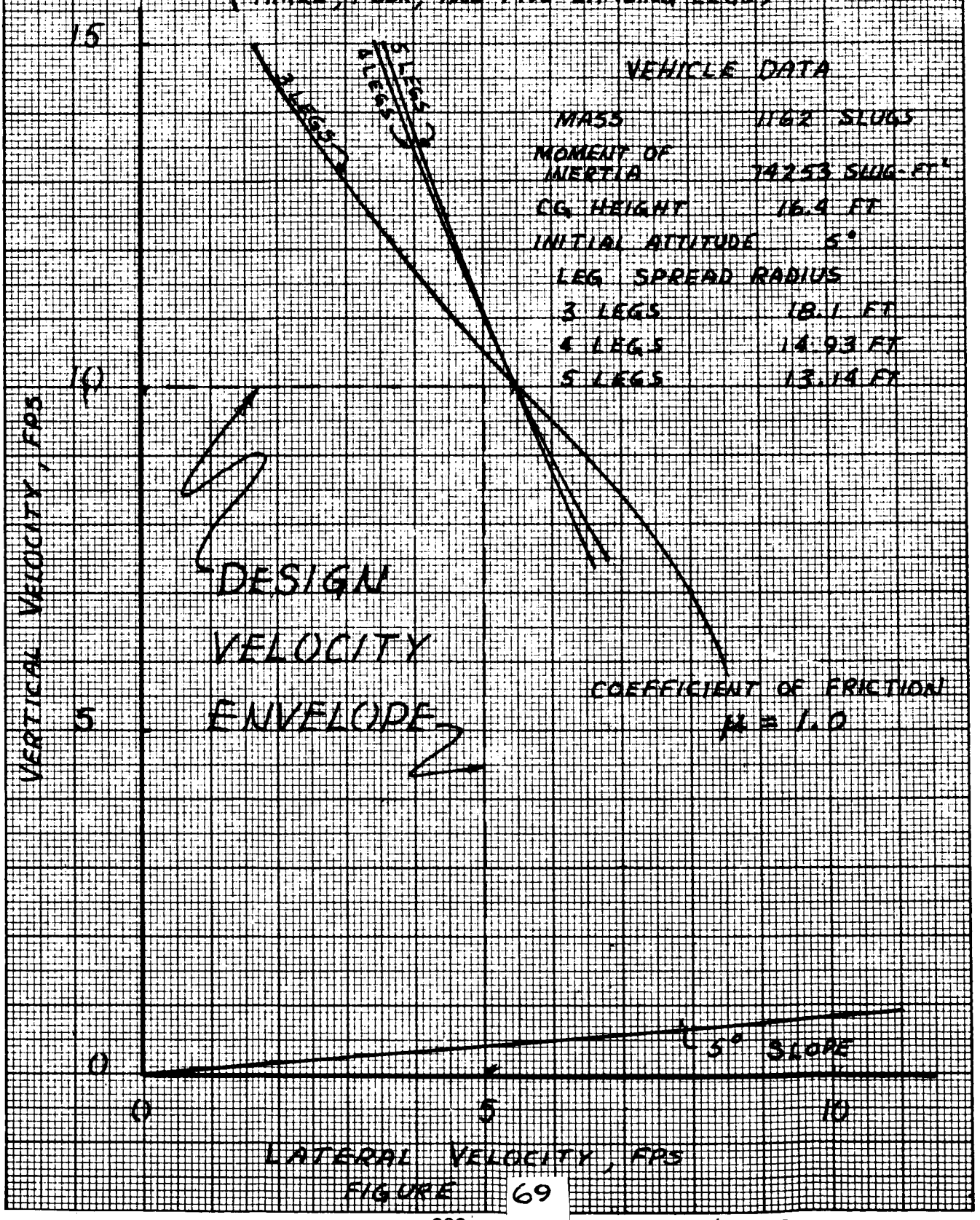


FIGURE 69

PREPARED BY:	NORTH AMERICAN AVIATION, INC. SPACE and INFORMATION SYSTEMS DIVISION
CHECKED BY:	
DATE:	

7. Effect of Friction Coefficient

Figure 70 presents analytical predictions of the leg spread radius required versus friction coefficient for three, four and five legged Apollo II vehicles. In this study the leg spread radius was increased until the entire design velocity envelope was within the predicted stable landing capabilities of the vehicle.

The study revealed the following:

(1) For friction coefficients above 0.75 the required leg spread radius essentially remains unchanged.

(2) As the friction coefficient is decreased below 0.75 a significant but still small reduction in the required leg spread was noted. The reduction in required leg spread for friction coefficient of 1.0 to 0.5 was 5%, 7%, and 9% for three, four, and five legged vehicles, respectively.

Structural Analysis

The stability design envelope obtained from the preceding structural dynamics analysis was utilized to determine the required leg spread for the friction coefficient value of 1.0. The analysis was conducted using a vehicle with the following characteristics:

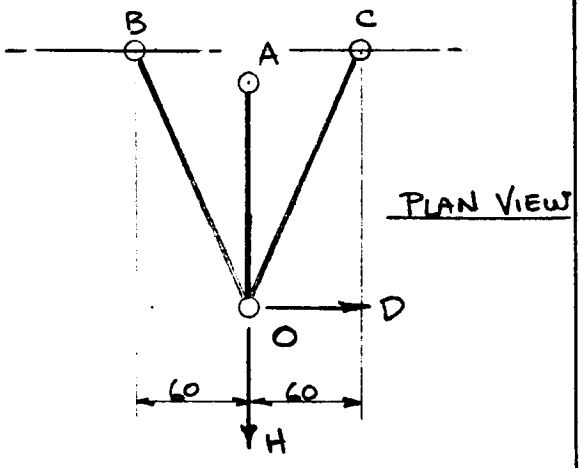
Mass	1162 slugs
Moment of Inertia	74,253 slug-feet^2
CG Height	16.4 feet
Initial Attitude	5°

The structural analysis considered a constant vehicle shall radius of 16.65 feet with varying leg extension dimensions of 5, 10 and 15 feet from the vehicle shell to the landing shoe. Gear geometry is shown in Figure 71. The tripod structural concept permits the center compression strut to compress while allowing rotation of the lower struts and joints. The deflected position of the tripod is shown in Figure 72. Honeycomb material, provided inside of the center compression strut, serves as a crushable element. A Maximum vertical load factor of 3 g's applied to the vehicle and equally reacted among the landing shoes was the ultimate design condition for the crushing element. A safety factor of 1.5 times this load was used to design the remaining gear in order to insure proper function of the crushing element. Horizontal load components were computed for various orientation angles of impact applied to the landing shoes. As shown in Figures 73, 74, and 75 for the 3, 4 and 5 leg gears the maximum horizontal load component occurred when the orientation angle, Θ, equals zero. This condition developed maximum compression load in the center compression strut. The

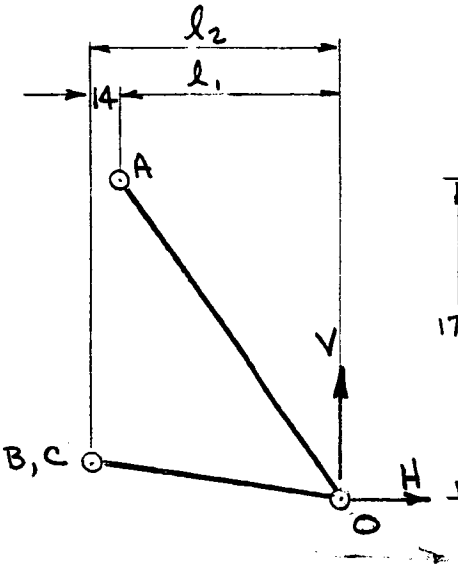
LANDING GEAR STUDIES OF 3, 4 OR 5 LEGS

GEOMETRY Figure 71

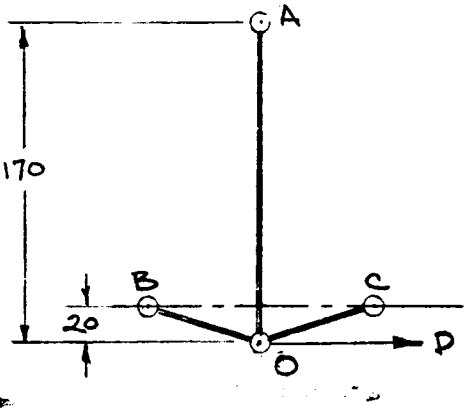
Config. No.	l_1	l_2
1	60	74
2	120	134
3	180	194



PLAN VIEW



SIDE ELEVATION



FRONT VIEW

TENSION COEFFICIENTS:

Config No.	Member	V	D	H	L
1	OA	170	0	60	180
	OB	20	-60	74	97.2
	OC	20	60	74	97.2
2	OA	170	0	120	208
	OB	20	-60	134	148
	OC	20	60	134	148
3	OA	170	0	180	247
	OB	20	-60	194	204
	OC	20	60	194	204

FIGURE 72 DEFLECTION OF TRIPOD

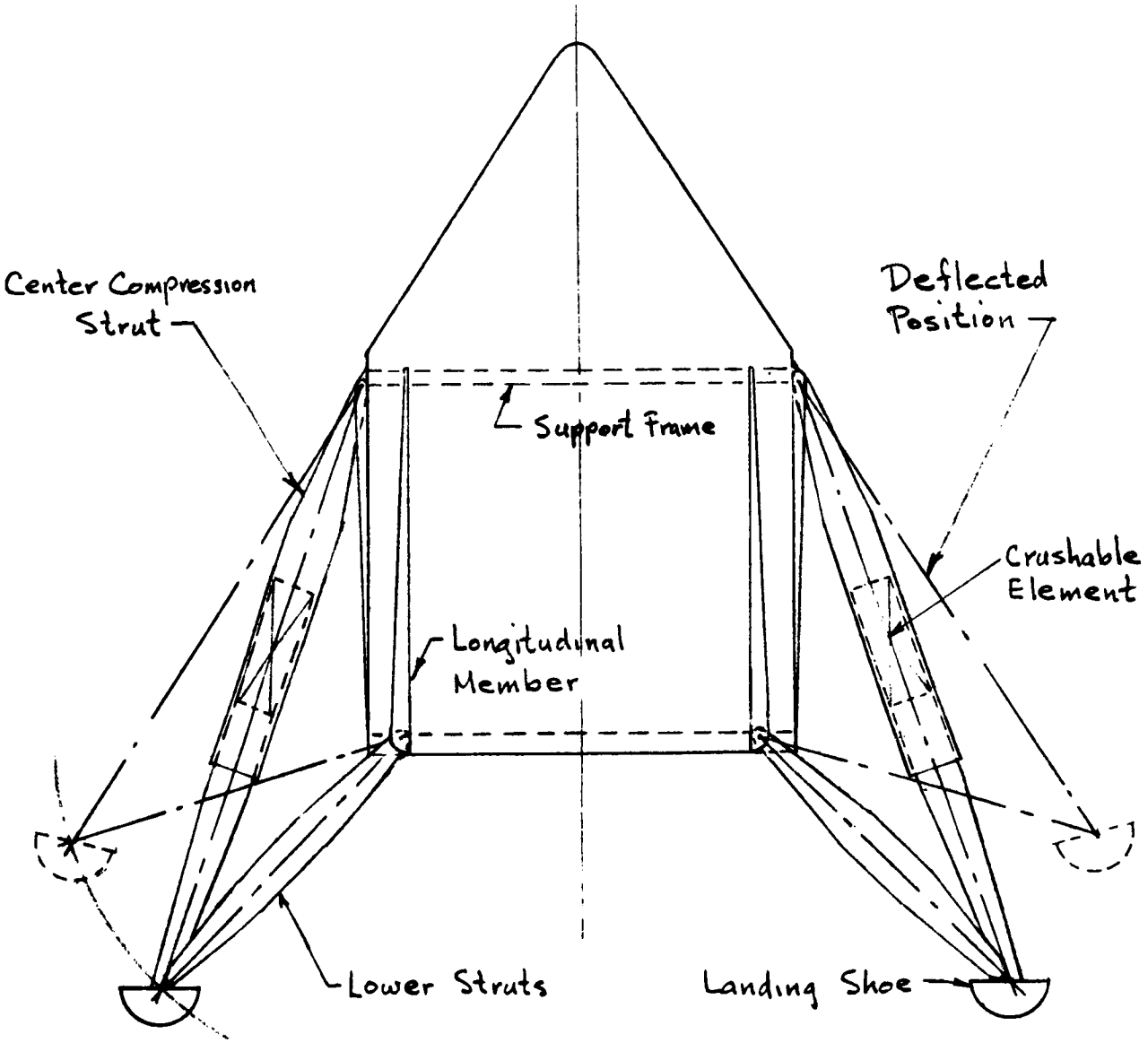
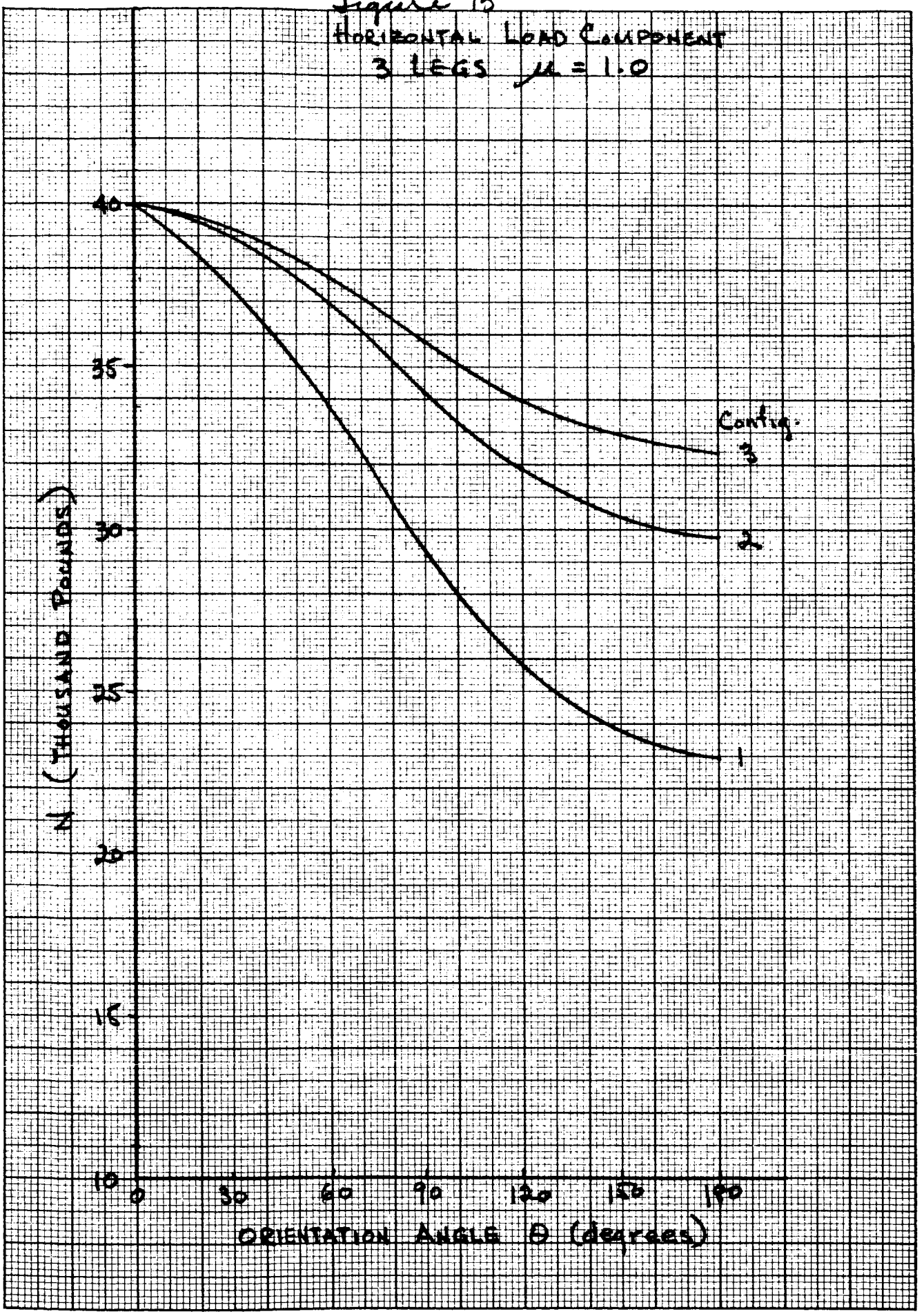
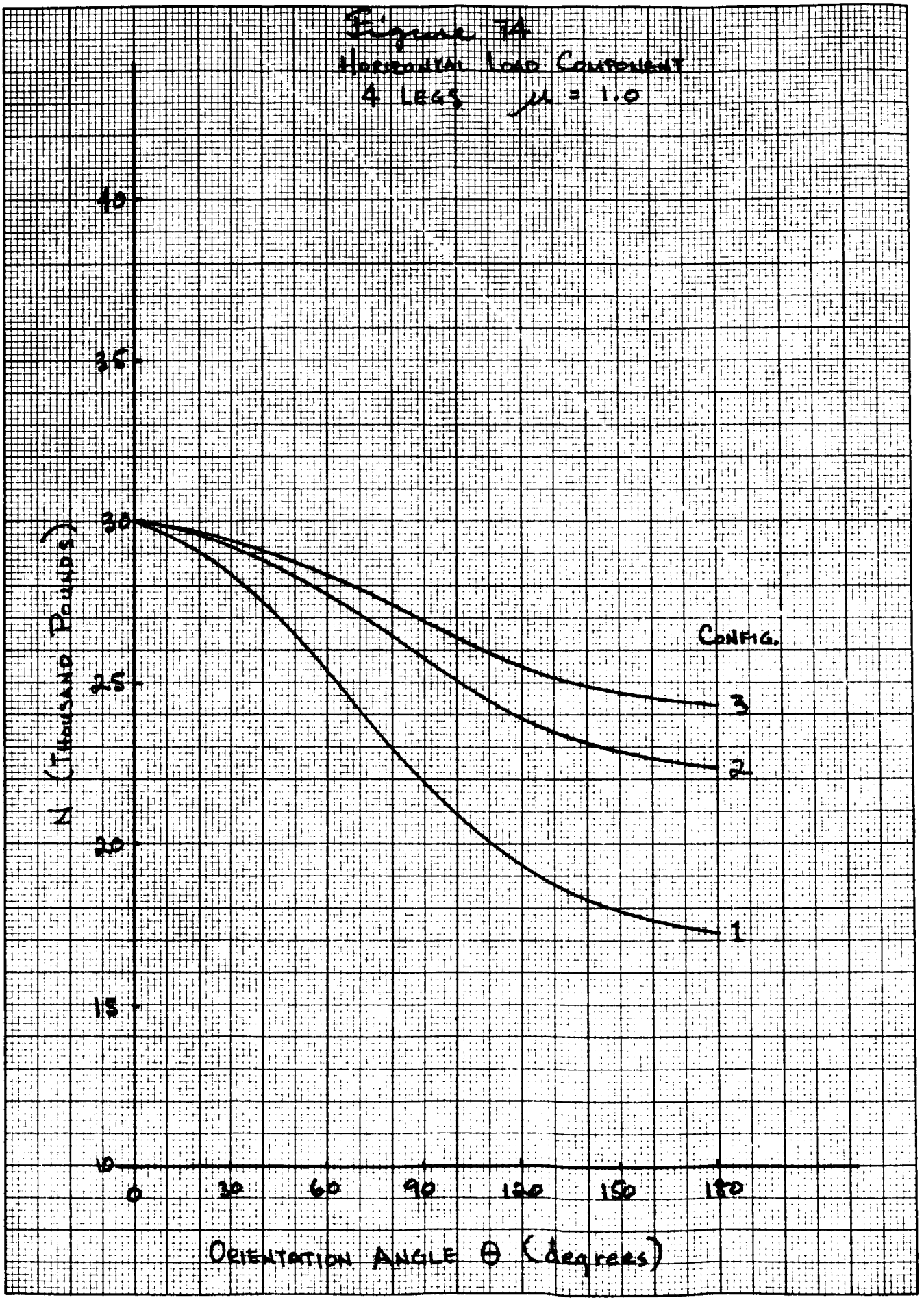
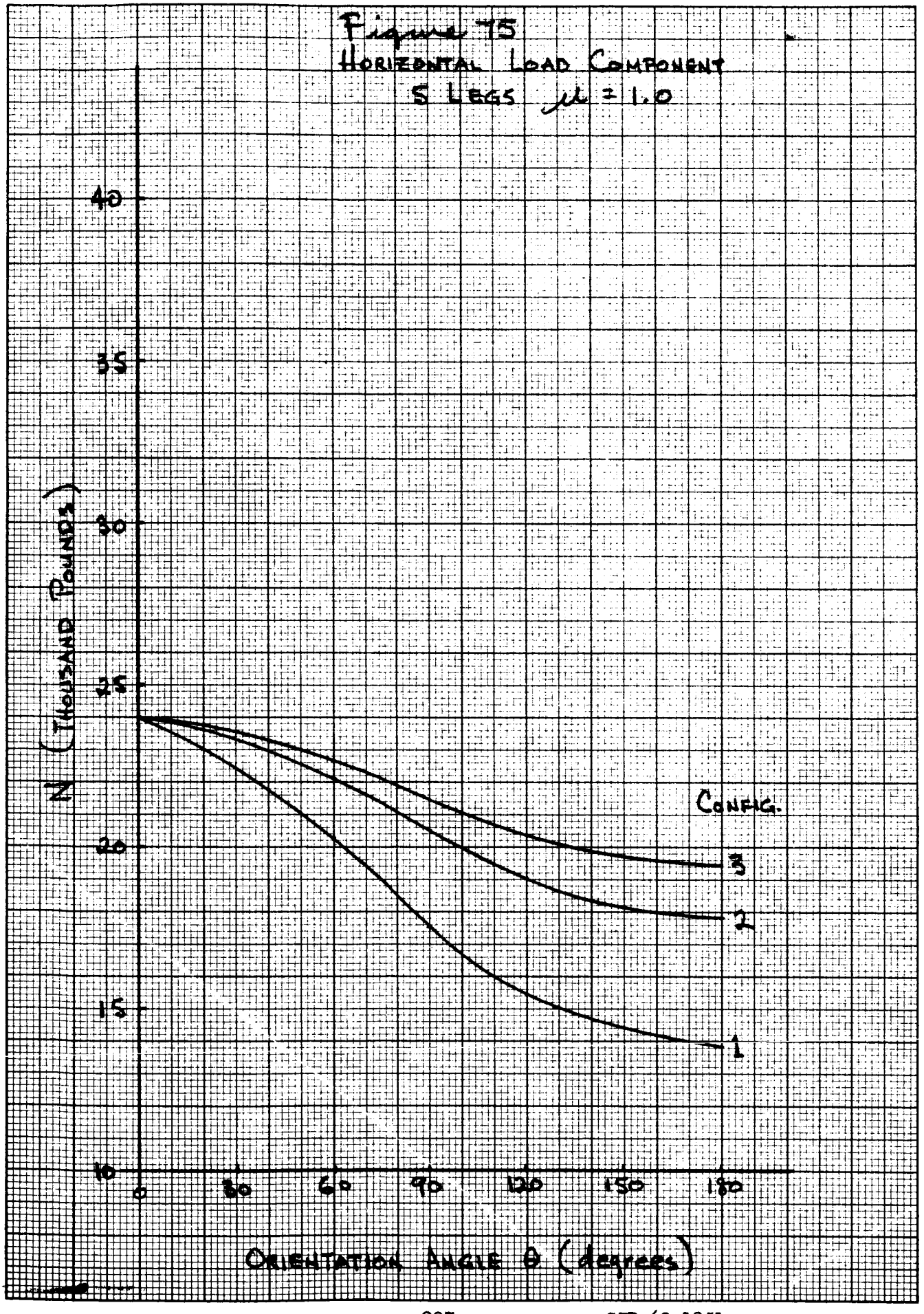


Figure 73


HORIZONTAL LOAD COMPONENT
3 LEGS μ = 1.0
40
35
30
25
20
15
10
N (THOUSAND POUNDS)
Config.
3
2
1
0
30
60
90
120
150
180
ORIENTATION ANGLE θ (degrees)


Figure 74
Horizontal Load Containment
4 Legs $\mu = 1.0$
40
35
30
25
20
15
10
H (Thousand Pounds)
Config.
3
2
1
0
30
60
90
120
150
180
Orientation Angle θ (degrees)


Figure 75
Horizontal Load Component
5 Legs $\mu = 1.0$
40
35
30
25
20
15
10
N (Thousand Pounds)
Config.
3
2
1
0
30
60
90
120
150
180
Orientation Angle θ (degrees)

maximum compression load applied to the lower struts would occur when the impact orientation angle becomes in close alignment with the strut geometry.

The method of analysis was simplified by solving 3 unit loading conditions applied along the three coordinate axes at the landing shoe. The actual member loads were determined by multiplying the actual vertical and horizontal load components by their unit solutions and combined by superposition.

Support frames and longitudinal members are provided to distribute loads to the shell. The frames providing support for the horizontal load component and the longitudinal members support load for the vertical component.

The structural weight of the entire gear including support structure for required leg spread is shown in Figure 76. Results of the study show that the 5 leg configuration is the optimum design with the 4 leg configuration very competitive. A trade-off should be made between the additional structural weight of the 4 leg configuration. At this design phase, the 4 leg configuration was recommended for the vehicle.

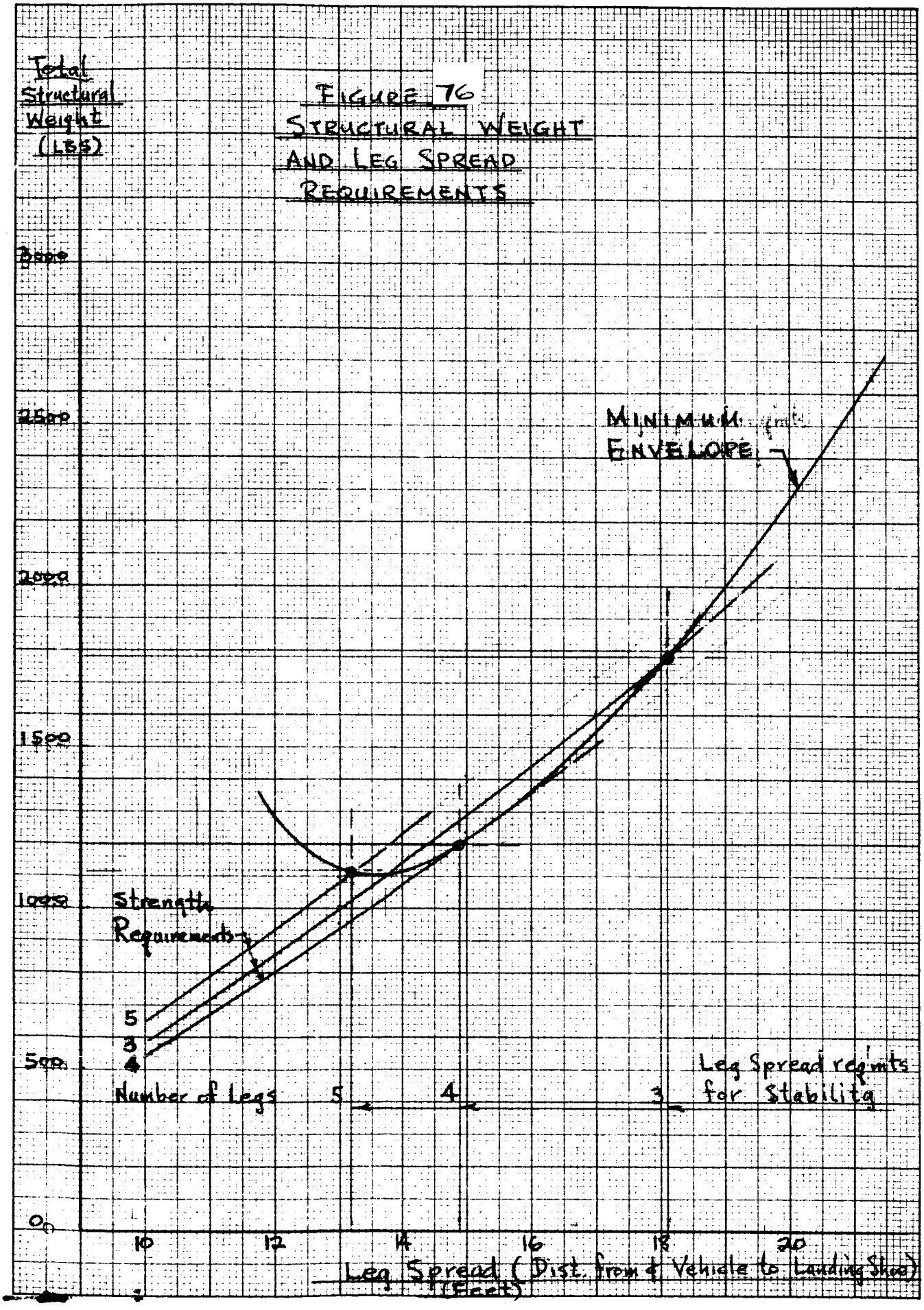
THERMAL ANALYSIS

The requirement that the various lunar shelter concepts must be capable of withstanding long standby periods on the lunar surface is the controlling factor in the design of temperature control systems for such vehicles. The extremely cold environment encountered during the lunar night, and the high temperatures encountered during the day, require highly effective insulation in order to rely upon the thermal capacitance of the internal structure to keep temperatures within prescribed limits. This, in turn, means that the heat loads generated during manned occupancy will necessitate the constant usage of a heat rejection system, even during the lunar night. It becomes apparent that passive temperature control requirements, particularly for standby during the lunar night, will greatly influence the design of both the vehicle shell and the temperature control system.

A preliminary investigation has been made to determine the thermal protection requirements necessary for long-term passive standby. This involved the three different lunar shelter concepts: (1) a CLEM landed shelter with standby up to a week, (2) an Apollo I SM-landed shelter with standby up to 90 days, and (3) an LLV-landed shelter with standby up to one year. The results of this investigation are presented below.

Structural Shell

In order to minimize heat loss or gain through the outer walls of the shelter, it will be necessary to utilize some sort of insulated structure


Total Structural Weight (LBS)
FIGURE 76
STRUCTURAL WEIGHT AND LEG SPREAD REQUIREMENTS
2500
2000
1500
1000
500
0
MINIMUM ENVELOPE
Strength Requirements
5
3
4
Number of Legs
5
4
3
Leg Spread reqmts for Stability
10
12
14
16
18
20
Leg Spread (Dist. from ℄ Vehicle to Landing Shoe) (Feet)

that minimizes heat leaks by eliminating direct metallic paths. From a thermal standpoint, it makes little difference how this is accomplished, as long as two requirements are met: (1) the required insulation must be protected from crushing loads and such things as aerodynamic loads, etc.; (2) the supporting structure must be non-metallic with minimum cross-sectional area. Fortunately, the present Apollo CM structure utilizes a double-wall type of structure that can satisfy the above requirements, with certain modifications. Therefore, all analyses performed by this investigation were based upon a double wall structure filled with "super" insulation, such as Linde SI-62 or NRC-2, with supporting structure (frames, longerons) composed of fiberglass-epoxy. All cases were analyzed assuming the double walls were 1.0 inch apart. The heat leak area through the fiberglass-epoxy was assumed to be 12.7 in^2 per 100 square feet of surface area. This is considered to be the minimum practical value at the present time from a structural point of view.

The total heat transfer through the shelter walls is composed of (1) heat transfer through the insulated double walls and the fiberglass-epoxy supporting structure, and (2) unavoidable heat leaks such as electrical conduits, hatch frames, etc. In accordance with the assumptions made in Reference 1*, it was assumed that 40 percent of the total heat leak occurred through the insulated double wall and 60 percent occurred through parallel heat leaks.

Lunar Surface Temperatures

There are several lunar temperature distribution curves which have been used at North American and elsewhere. Unfortunately, there is considerable difference between these curves; and, as a result, a certain amount of confusion exists relative to which is the most realistic. Recently, however, there has been considerable study in this area, and several conclusions have been reached which are in general supported by the majority of investigators. These are:

(1) Subsolar point temperature is between 230 and 240°F. The most widely quoted figure (242°F) is that of Sinton (Reference 2), who refined the earlier work done by Pettit (Reference 3).

(2) The best model for the lunar surface is a two layer configuration consisting of an outer dust layer (varying in thickness from a few mm. to a few cm.) which overlays a substratum of material such as pumice or basaltic rock. The emissivity of the lunar surface is very close to 1.0. This approach to black body characteristics is probably due to the porous nature of the dust layer.

* References listed on last page of the section.

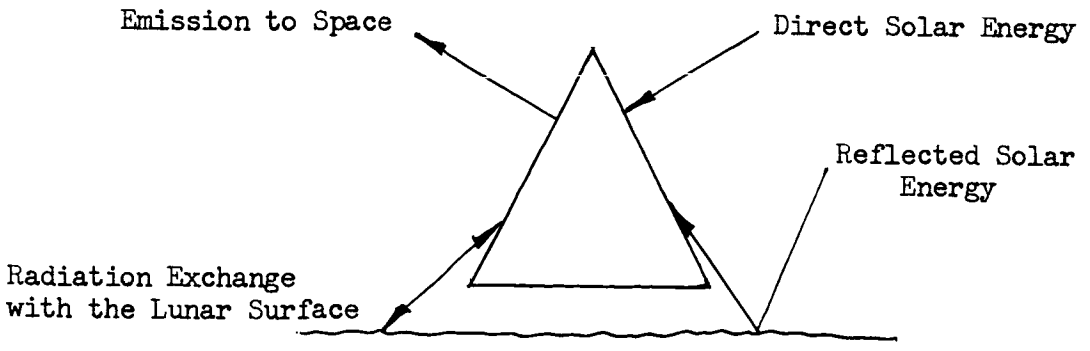
(3) The distribution of planetary radiation over the sunlit lunar surface varies approximately as the cosine of the angle, θ, from the sub-solar point. Therefore, the temperature varies as the cos $1/4\theta$. Pettit (Reference 3) reported that the apparent distribution of planetary radiation over the lunar disk at full moon varies as cos $2/3\theta$. However, it was recognized and demonstrated that the experimental data exhibited a deviation from the cosine distribution due to a directional effect caused by the rough surface. Considering total hemispherical radiation from points on the lunar surface would probably demonstrate a cosine distribution, noting that the lunar sphere must receive energy from the sun in this fashion. These conclusions were reached by Piddington and Minett (Reference 4) and several others.

(4) The ratio of the solar energy reflected from the sub-solar point to that received by it is 0.124, according to Pettit. This may be considered as the albedo of the subsolar point and is the value that is recommended for local areas of the lunar surface. Admittedly, there is considerable controversy concerning the albedo of the moon, and the most widely quoted figure is 0.073, which is the visual albedo of the moon, i.e., the ratio of light reflected from the whole hemisphere to that received by it. Fortunately, whether the true value is 0.07 or 0.124 is somewhat immaterial from a thermal analysis standpoint, since the amount of reflected energy is small relative to direct solar energy and lunar emission.

The lunar temperature curve used in this study is shown in Figure 77, and was obtained from Reference 5, which has been attached to the latest RFP's emanating from MSC. This curve appears to be quite satisfactory, inasmuch as it adheres closely to the quoted figures of Sinton, and follows a cos $1/4\theta$ temperature distribution. (See also Part I, Lunar Environment.)

Method of Analysis

The heat exchange between an object on the lunar surface and its environment is shown below:



The net heat lost or gained by the object can be expressed by:

q = (absorbed direct solar energy) + (absorbed reflected solar energy) (1)
+(net radiation exchange between object and lunar surface)-
(emission to space)

With the assumptions that (1) the lunar surface acts as a black absorber and black emitter of infrared energy, and (2) reflected solar energy is handled on a "one-bounce" basis (no interreflections), Eq. 1 can be written as

$$q = \alpha A_p S + \alpha r_M S F_{O-M} A \cos\phi + \sigma \epsilon F_{O-M} A (T_M^4 - T^4) - \sigma \epsilon F_{O-S} A T^4 \quad (2)$$

where
- α = solar absorptivity of vehicle surface
- A_p = projected area of vehicle surface relative to the sun
- S = solar constant, 443 Btu/(hr) (ft^2)
- r_M = albedo of lunar surface
- F_{O-M} = geometric configuration factor from vehicle surface to moon
- A = vehicle surface area
- ϕ = angle between vehicle center line and vector between vehicle and sun
- ϵ = emissivity of vehicle surface
- σ = Stefan-Boltzmann constant, 0.1713×10^{-8} Btu (hr) (ft^2) ($^\circ R^4$)
- T_M = absolute temperature of the lunar surface, degrees R
- T = absolute temperature of the vehicle surface, degrees R
- F_{O-S} = geometric configuration factor from vehicle surface to space

Noting that the quantities (F_{O-M}) and (F_{O-S}) must add up to unity, Eq. 2 can be simplified to

$$q = \alpha A_p S + \alpha r_M S F_{O-M} A \cos\phi + \sigma \epsilon F_{O-M} A T_M^4 - \sigma \epsilon A T^4 \quad (3)$$

Due to the requirement that the shelters have a highly effective insulated shell, it can be assumed that the heat transfer through the shell is so small that the outside surface temperature is equal to the adiabatic wall temperature. This allows evaluation of the outside surface temperature on the basis of the α/ϵ ratio and as a function of sun angle (time), i.e.,

$$\sigma A T_A^4 = \frac{\alpha}{\epsilon} \left[A_p S + r_M S F_{O-M} A \cos\phi \right] + \sigma F_{O-M} A T_M^4 \quad (4)$$

where T_A = adiabatic wall temperature, degrees R

[redacted]

Once the adiabatic wall temperatures are determined, the final step in the problem solution is to evaluate the transient (temperature-time) history of the lunar shelter for the particular case at hand. The equation describing this process is written as,

$$Wc_p \frac{dt}{d\theta} = \frac{k}{x} A(t_A - t) \qquad (5)$$

where W = mass of the vehicle inside the insulated shell, lb

c_p = specific heat of vehicle mass, Btu/(lb)(°F)

t = average temperature of the vehicle, °F

θ = time

k = overall thermal conductivity of the insulated shell, Btu/(hr)(ft^2)(°F/ft)

x = thickness of insulation, ft

t_A = adiabatic wall temperature, °F

The transient analyses performed in this study were solved by an IBM 7094 general heat transfer program, which achieves solution of Eq. 5 by converting it to finite-difference form. The following assumptions were made:

(1) The effective thermal conductivity of the insulated shell was based upon a weighted average of the following:

$k_{\text{SUPER INSULATION}} = 2.0 \times 10^{-5}$ Btu/(hr)(ft^2)(°F/ft)

$k_{\text{FIBERGLASS-EPOXY}} = 0.2$ Btu/(hr)(ft^2)(°F/ft)

(2) The specific heat of the vehicle mass inside the insulated shell was assumed to be 0.2 Btu/(lb)(°F).

(3) The thermal conductivity of the internal vehicle structure was assumed to be infinite in comparison to that of the insulated shell.

Concept 1

Figure 78 shows the temperature variation of the short term CLEM landed shelter as a function of time after landing. It was assumed that the initial temperature of the shelter was 70°F. Total weight of the shelter was assumed to be 7000 lb, of which 6400 lb was contained within the insulated shell. (The 600 lb of structure external to the thermal barrier has no effect on the variation of internal temperature.)

Referring to Eq. 5, it can be seen that the heat transfer to the interior of the vehicle is a direct function of k/x (the ratio of thermal conductivity to thickness). As pointed out previously and described in the structures section of this part of the report, the structural shell is composed of a double wall configuration, filled with super insulation, with fiberglass-epoxy supporting structure. Allowing for parallel heat leaks through electrical conduits, etc., the effective value of k/x for the shell becomes approximately 0.006. This is higher than the value of 0.0035 used in Reference 1, but is considered the minimum practical value at the present time from a structural point of view.

Concept 2

Figures 79 and 80 show various temperature-time histories of the Apollo I SM-landed shelter for three complete lunar days (approx. 84 earth days). It was assumed that the initial temperature in all cases was 70°F. In all cases except one, the total weight of the shelter was assumed to be 9000 lb, of which 8350 lb was inside the thermal barrier. For the other case, the total weight of the shelter was assumed to be 4500 lb, of which 3850 lb was inside the thermal barrier.

Note that cases were run for k/x values of 0.006 and 0.0038. The latter figure corresponds to the same double wall configuration that was used originally, except the heat leak through the fiberglass-epoxy structure was assumed to be zero. These cases were included more for purposes of comparison than as a recommendation, since to achieve such a low figure as 0.0038 would be extremely difficult, if not impossible.

Figure 81 shows the temperature time history of the shelter assuming 0-180 watts of available internal heating. This amount of heat may be supplied by the isotopic dynamic prime electrical power source used during the standby period. The temperature in this case would never be lower than -28°F nor higher than 27°F (for $\alpha/\epsilon = 1.0$). The internal heater would be required only during the lunar night, as shown on the curve.

It has been suggested that an outer shroud completely surrounding the shelter would aid in thermally isolating the shelter. However, the extremely effective insulation requirements needed for passive standby would not be accommodated by this arrangement. Note that the insulation used in the shell is of the multiple-foil radiation shield type of vacuum insulation. The number of foils per inch for this type of insulation would be on the order of 50 to 150, depending on the manufacturer. In effect, the addition of an outside shroud would only add one more radiation shield to the 50 to 150 already present. Even allowing for elimination of a certain amount of structural heat leak, it becomes apparent that the addition of a shroud would have a negligible effect on the overall effectiveness of the insulation.

[redacted]

Concept 3

Figure 82 shows various temperature-time histories of the LLV-landed shelter for one complete lunar cycle. All cases were machine solved for a period covering one year of passive standby after landing (12 lunar cycles). The curves shown represent the temperature cycling that occurred after the effects of the initial vehicle temperature had become negligible. Initial temperature effects damped out in about three or four months, after which time there was no change in maximum and minimum temperatures. Total weight of the shelter was assumed to be 28,000 lb, of which 25,400 lb was assumed to be inside the thermal barrier.

Discussion

The results of this study indicate that a major problem area exists with respect to the tendency for unmanned shelters to reach a low temperature during the lunar night standby period (in the region of -100°F to -120°F). However, this level can be raised by careful control of surface coating properties, or by internal heating, as can be seen from the curves for Concepts 2 and 3.

A second problem area concerns the establishment of a realistic allowable temperature range, both for equipment on standby or otherwise. This has tentatively been set at +160°F to -65°F, corresponding to several widely used military aircraft specifications. However, this is rather arbitrary, and further investigation should be made into this subject. If, for instance, the allowable minimum equipment temperature were to be set at -120°F or lower, the problem mentioned in the preceding paragraph would cease to exist and no internal heating would be required.

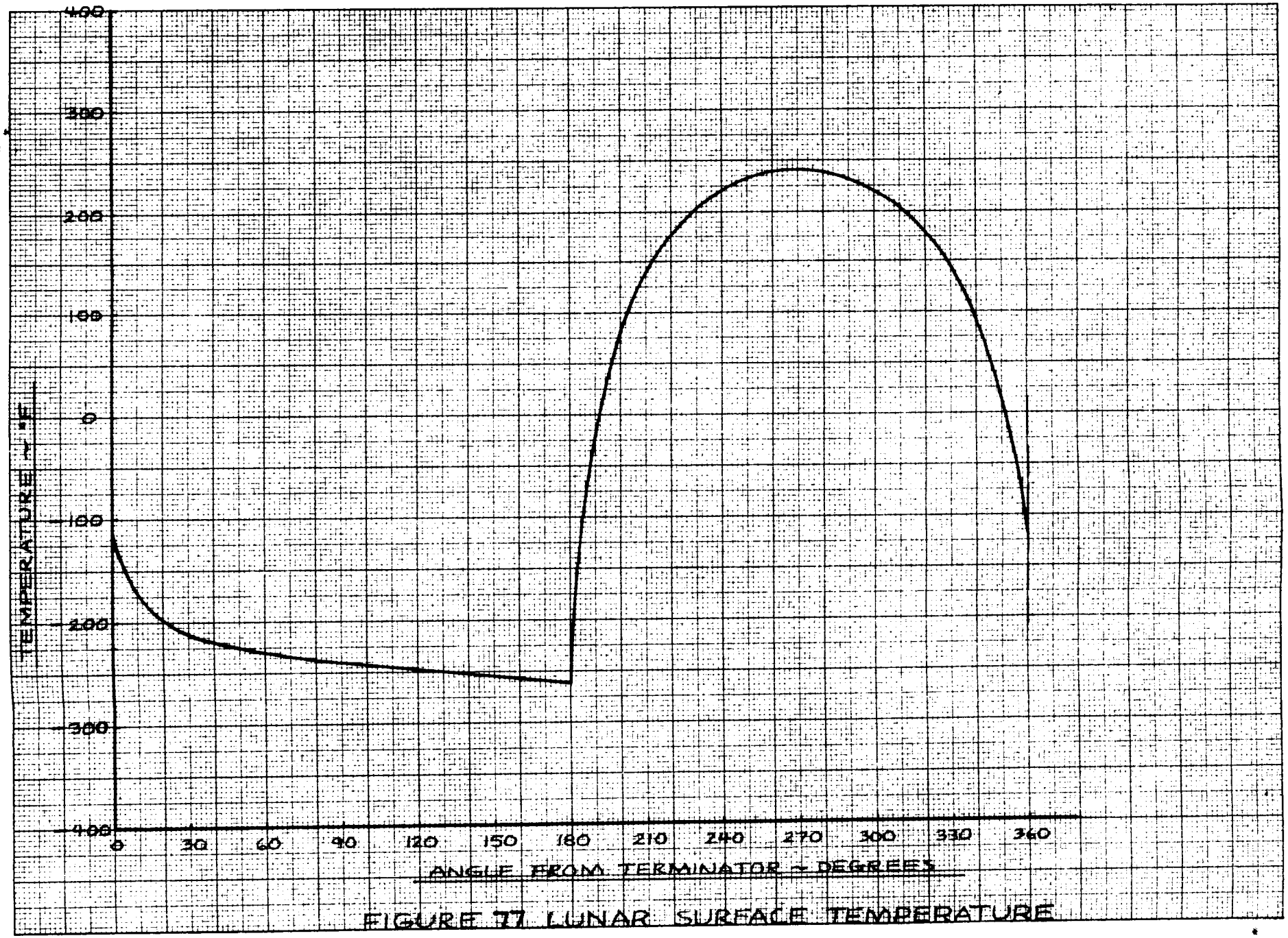


FIGURE 77 LUNAR SURFACE TEMPERATURE

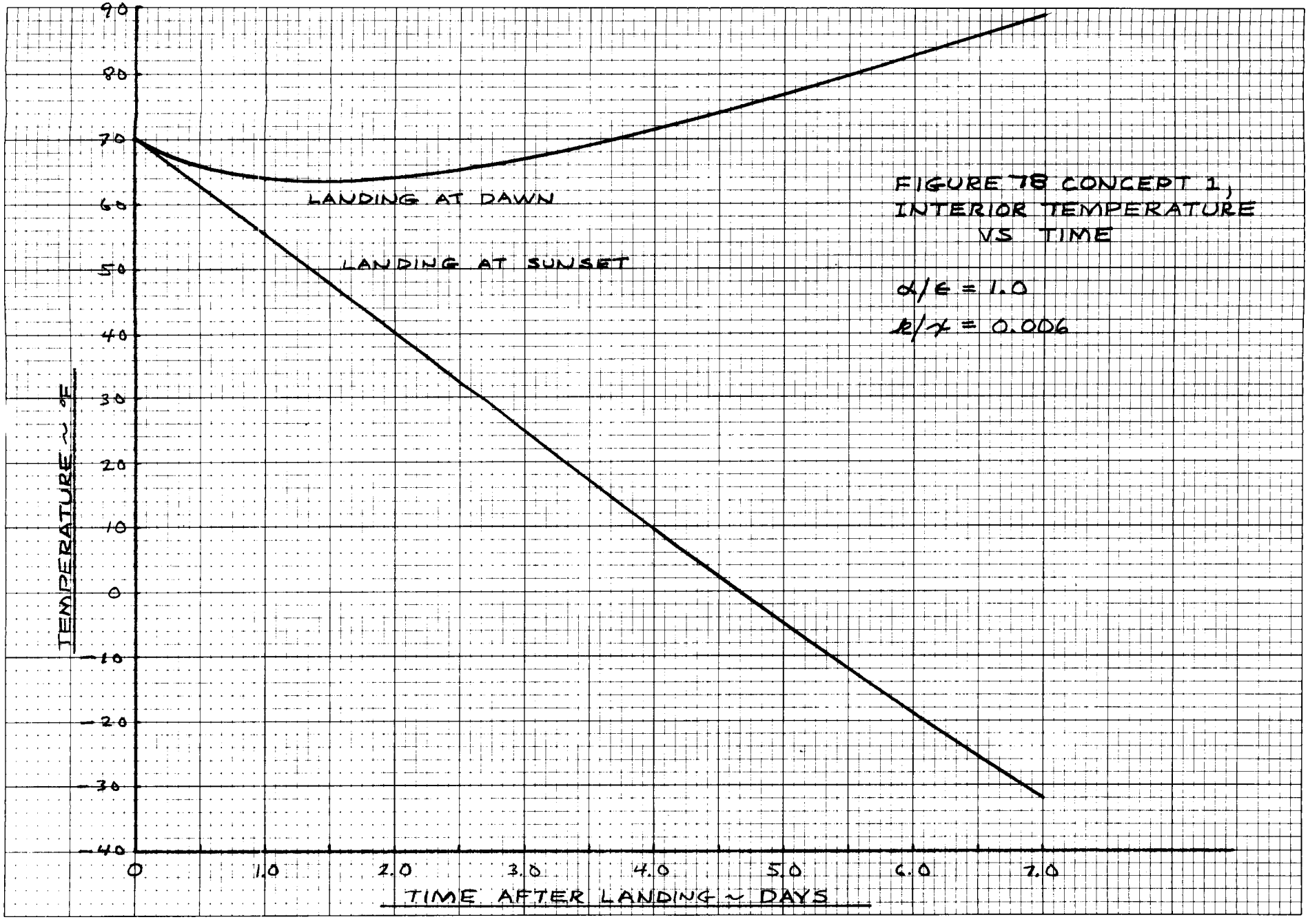


FIGURE 78 CONCEPT 1, INTERIOR TEMPERATURE VS TIME

$\alpha/\epsilon = 1.0$

$k/\chi = 0.006$

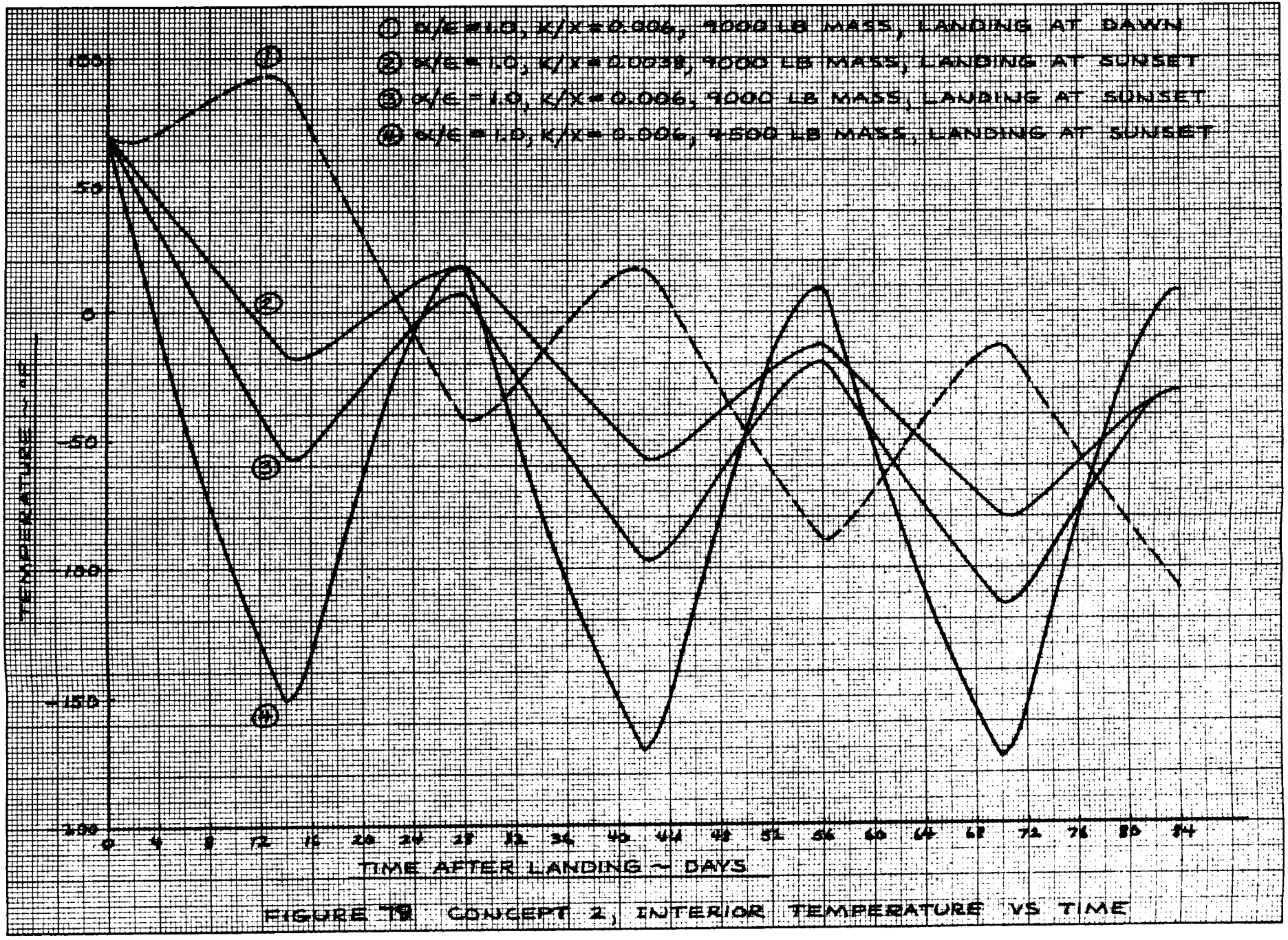


FIGURE 78 CONCEPT 2, INTERIOR TEMPERATURE VS TIME

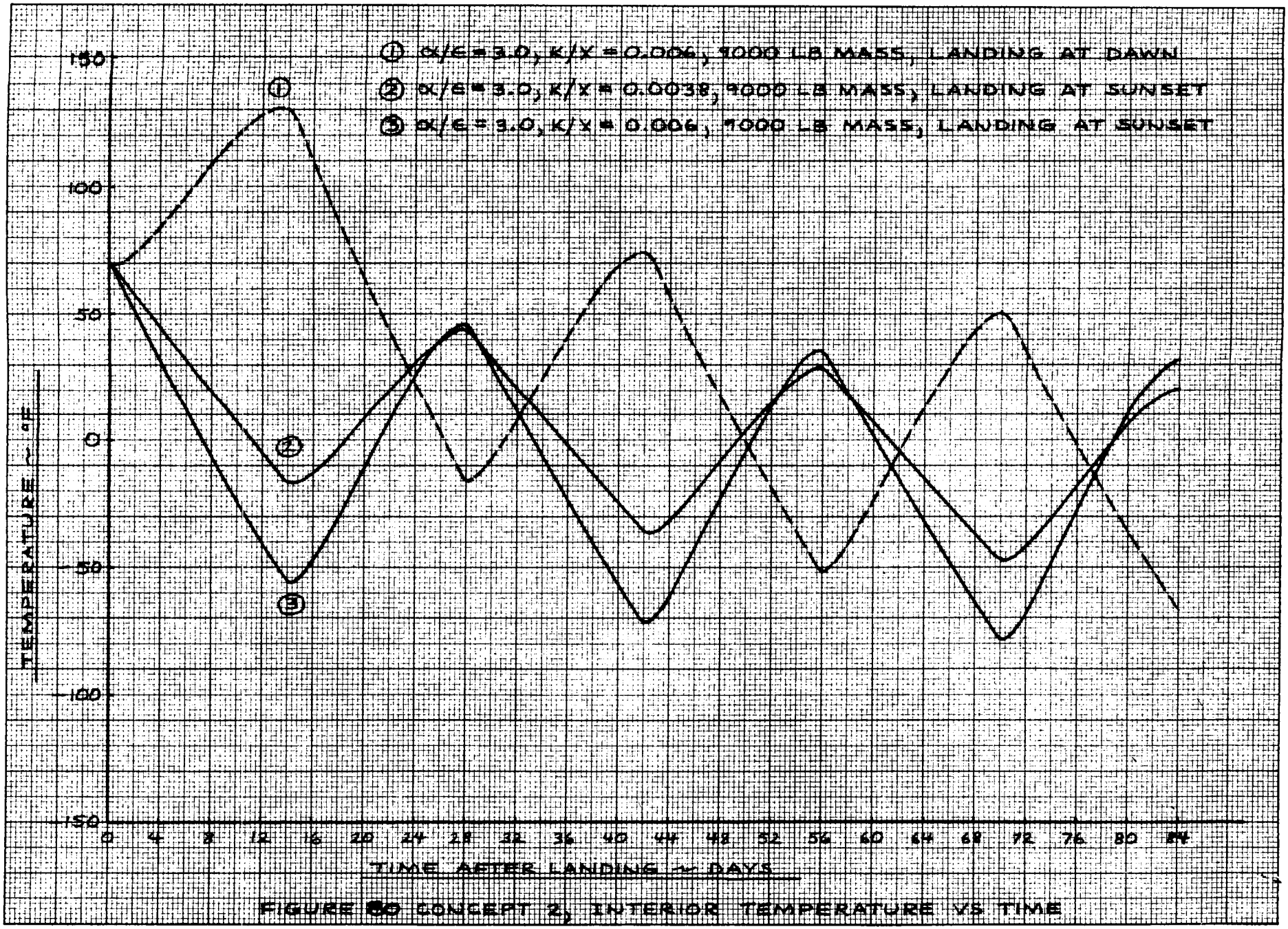


FIGURE 60 CONCEPT 2, INTERIOR TEMPERATURE VS TIME

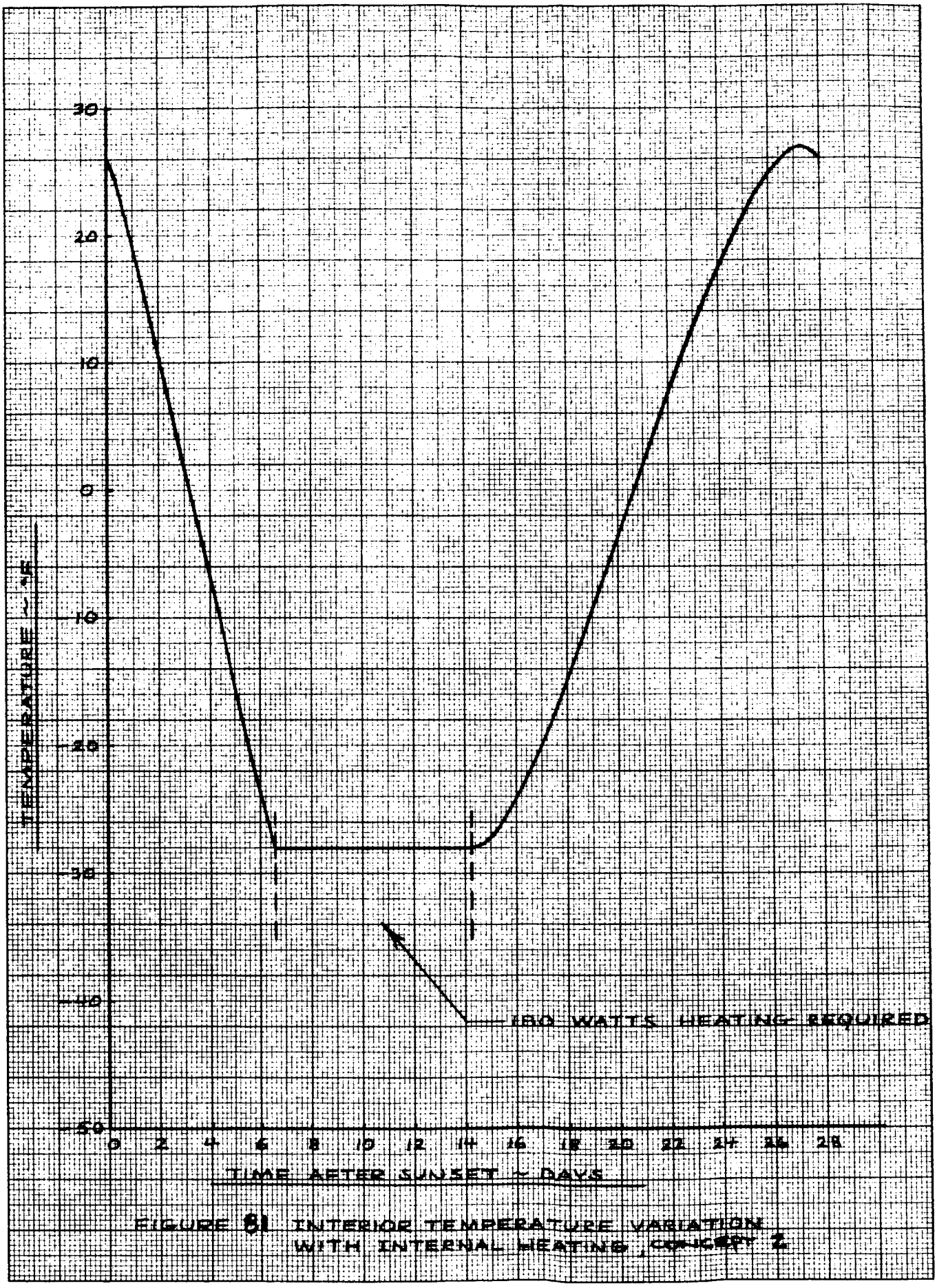


FIGURE 81 INTERIOR TEMPERATURE VARIATION WITH INTERNAL HEATING, CONCEPT 2.

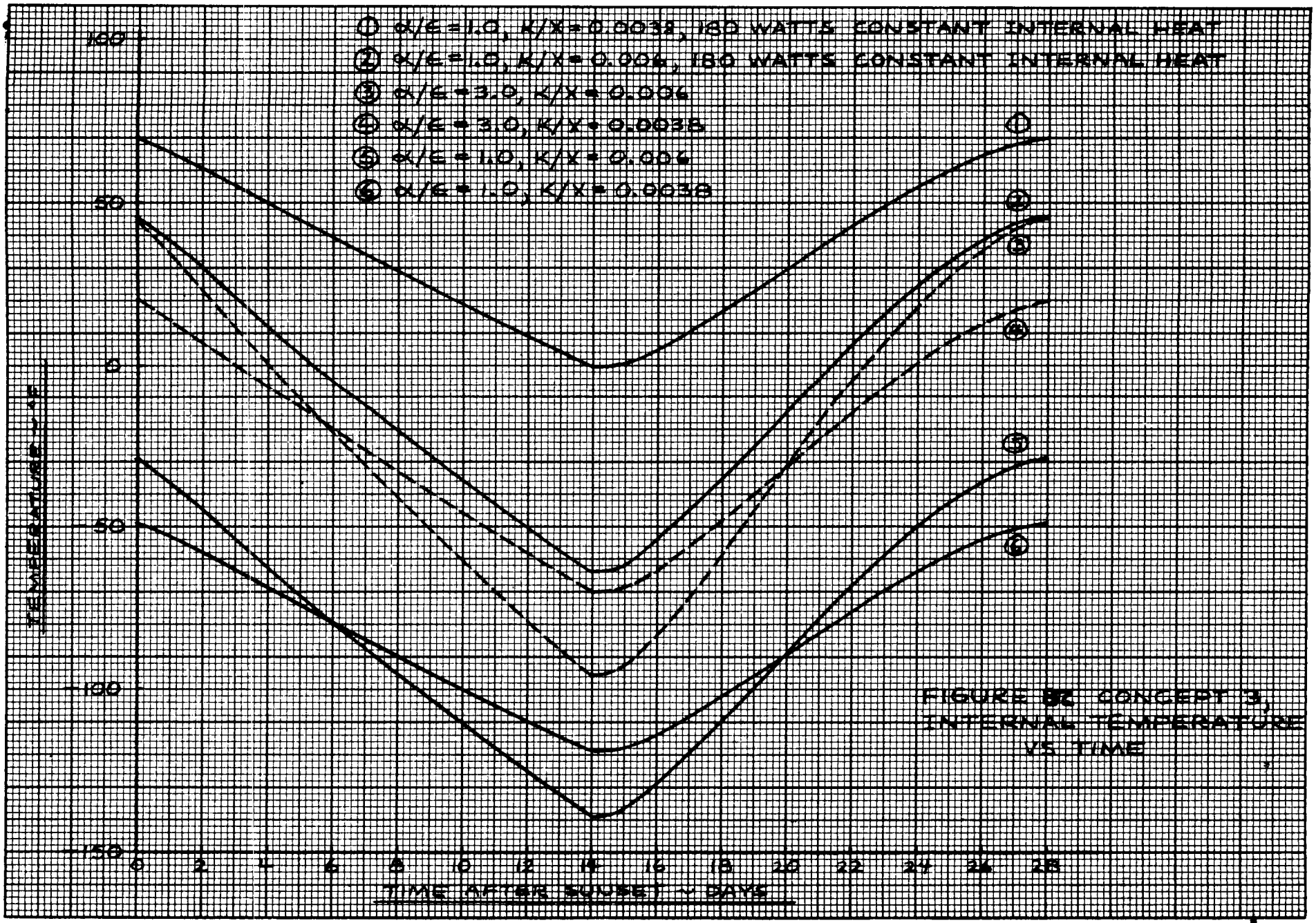


FIGURE 82 CONCEPT 3, INTERNAL TEMPERATURE VS TIME

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UNMANNED TERMINAL GUIDANCE

The analysis described herein is concerned with that period of lunar base development immediately following the initial LEM landings and the selection of one or more lunar base sites. During this period, the unmanned landing of shelters and logistic supplies will be performed. To alleviate the problems associated with moving landed modules over the lunar terrain to assemble a base complex, spot-landing accuracy is required.

A previous study (SID 62-1466, "Lunar Logistic Vehicle Preliminary Study") indicated that a single lunar beacon or radar transponder, emplaced earlier by the LEM crew, in combination with a radar-transponder detector, a three-beam Doppler velocity sensor, a radar altimeter, and a variable thrust propulsion system aboard the vehicle provided the simplest possible terminal guidance system for the LLV.

It was established from the above mentioned earlier study that a lunar landing with first establishing a lunar orbit was preferrable over a straight-in approach (no lunar orbit) for various reasons:

a. lower propellant requirements & lower T/W

b. increased probability of survival for a manned payload

c. possibility of unlimited landing sites on the lunar surface

The vehicle will initiate its descent from a nominal 100 NM orbit with a Hohman transfer down to 50,000 ft perilune, guided by a stellar-inertial guidance system. At perilune, the main retro descent is initiated based on the knowledge of the vehicle trajectory, range, range rate, and direction to a known radar transducer on the surface. At a range of about 10 NM from the landing site, the Doppler Velocity Sensor, consisting of a three-beam CW system with fixed antenna, is turned on upon command from the computer. The altimeter, with a gimbaled antenna, may already be turned on at a range of 100 NM. From the landing initiation point (altitude 1000 feet, vertical velocity 10 feet/sec., horizontal velocity = 0) on, the guidance and steering of the vehicle is based on the information from the three radars only. Upon having oriented the vehicle properly as a result of the radar transducer detector information, the Doppler frequencies of the three-beam velocity sensor monitor the LLV velocity and orientation. The objective of the guidance system at this stage is to guide the vehicle such that, just prior to touch-down, the Doppler frequencies along each beam are equal. Touch-down shall occur when all these frequencies are equal to zero at an altitude of a few feet.

The following analysis, resulting from a survey of the state-of-the-art of both present-day and future systems, discusses other possible terminal guidance methods that might be employed, compares them to each other, and attempts to describe qualitatively an optimum choice.

Terminal Guidance Methods

A survey of the literature (see bibliography) indicates that the

terminal guidance problem for Lunar-directed vehicles has undergone considerable analysis, and that several methods have been proposed, or will be utilized. At least seven basic methods, some with several variations, have been discussed, not all of which are applicable to an unmanned, precision-directed multiple-vehicle situation. To this array is added one more, suggested by S&ID, and all will be described in a qualitative sense. The techniques are listed, in broad categories, below:

A. Video, Earth-control

B. Vehicle Radar

1. With remote display, Earth-control
2. With Lunar surface beacon
3. With on-board control

C. Pre-programmed Control, Earth command

1. With altimeter

D. Celestial-Inertial

1. With altimeter
2. With horizon-scanner

E. Vehicle Interferometer

1. Lunar-surface beacon
2. Earth-radar illumination

F. Moon-site GCA

1. Manual
2. Automatic

G. Vehicle radar, two-beacon

H. Vehicle radar, three-beacon

The following paragraphs will briefly describe the method, and qualitatively discuss the applicability to the specific problem.

A. Video Guidance consists of a real-time video sensor-display, the camera in the vehicle or on the Moon's surface, and the display on the Earth. Observation of the display, perhaps aided by local computation and other displays, permits an operator to initiate commands to the vehicle to direct and control the vehicle's attitude and thrust. The method, when used alone or in conjunction with other sensor-control devices, has three characteristics that must be considered.

1. Response time

2. Resolution
3. Power/Bandwidth/weight

Response time is constrained by the propagation delay from Moon to Earth and return, and the operator's reaction delay. The two-way propagation time lag is 2.5 seconds; an operator's minimum reaction time is about 0.5 seconds. Resolution for any practical TV system is relatively poor for image quality; resolution in detecting distances, angular displacements, and their rate of change is very poor. Power and bandwidth are of lesser importance on large vehicles, but a real-time, non-intermittent TV image would require a bandwidth of 6.0 megacycles and a transmitter power of about 35 watts.

B. Vehicle Radar, with Earth-display and Earth-control, suffers all the disadvantage of response time delay that the video method has, plus a less easily interpreted display. When used with a surface beacon, however, the distance, angle, and rate measurement resolution becomes very good. Adding on-board computation reduces the response delay to nearly insignificant values. However, the power/weight factor is considerably higher for a 200-mile radar, fully-gimbaled, with a course computer, and would probably account for 50 pounds of the overall weight. It would also be necessary to either arrange several modes and power levels, or use additional sensors to cover the full range of magnitude and precisions required: a 300-mile radar is different from a 500-foot altimeter.

C. Pre-programmed Control would require an accurate ephemeris measurement, probably from Earth, considerable computations, and precise time control. Present Earth station equipments cannot provide the necessary precision of trajectory measurement; the computation includes several factors whose uncertainty is greater than the accuracy needed. While the time control possible may be adequate, the force to be controlled (thrust, angle of descent, and Moon's gravity) is uncertain to such a degree that pin-point landing accuracy is not possible.

D. Celestial-Inertial navigation will probably be included in the vehicle for other purposes. This method is best applied to the long-distance situation where a very long smoothing time (with drift correction) contributes to the accuracy. Its response time is, therefore, somewhat long, although response-time and accuracy are reciprocal functions.

E. Vehicle Interferometers require two, three, or more vehicle antenna-receivers, widely spaced (on extension arms or a gimbaled antenna.) When combined with a lunar-surface beacon or transponder and the capability of measuring rates (Doppler), as well as unambiguous range and direction, it is very precise and has negligible response delay. Course corrections would be computed on-board. Its only disadvantage lies in the need for multiple precision-spaced or gimbaled antennas.

An interesting variation of this technique has been proposed wherein the Moon's surface is illuminated by an Earth-based transmitter. Re-

flected R.F. energy would induce concentric "shells" due to interference, and it might be possible to navigate by sensing the surface of these shells. It is felt that this proposal is unsuitable for the immediate problem.

F. Moon-site GCA Sensing and controlling the vehicle's course from the Lunar-base, utilizing tracking radars (beacon on vehicle) is very attractive and feasible after the base is inhabited, and probably would become part of the base operations. However, prior to base occupancy, this method has too many disadvantages to be considered. As an Earth-based controller, it has all the propagation delays as indicated before; as an automatic system, it would probably be too complex and costly to implement.

G. Vehicle Radar, Two-Beacon This method, examined breifly by Autonetics, uses two beacons to perform the altitude and descent rate calculation, primarily during the positioning and flare maneuvers, and to provide a local vertical reference. It requires on-board computation and a gimbaled radar.

H. Vehicle Radar, Three-Beacon This method, examined by S&ID, uses three beacons to form a surface-based interferometer. Therefore, it can measure two direction cosines, or two range sums, and thus define the landing point more explicitly. Furthermore, by minor computation, the slant range, range-rate, and angular rates may be determined. The vehicle radar is not gimbaled, all antennas are omni-directional, and all data measurement and computation is electronic. A further advantage is that the automatic system can also be converted to a GCA-type system when the Moon-base is activated. The disadvantages, as with all beacon schemes, are that the beacons must be installed on the Moon and survive and operate in the exposed Moon environment.

Conclusion

1. The interferometer methods appear to have a great probability of of success. In particular, the three-beacon method is most efficient in terms of equipment weight and power requirement and has been used in determining guidance system weights for the remote-landed vehicles discussed in this report (Table 37). However, there are two problem areas related to any multiple-beacon scheme, as follows:

   a) Installation and calibration - the three beacons must be arranged on the Moon's surface as an oriented spaced array. Logically, the LEM crewmen would perform this task; one beacon could be installed on the abandoned LEM descent stage.

   b) Lunar environment - the beacons must survive the extremes of the lunar surface environment for very long periods.

   The first problem does not appear to be a serious deterrent. The second, however, points out that extreme reliability is needed

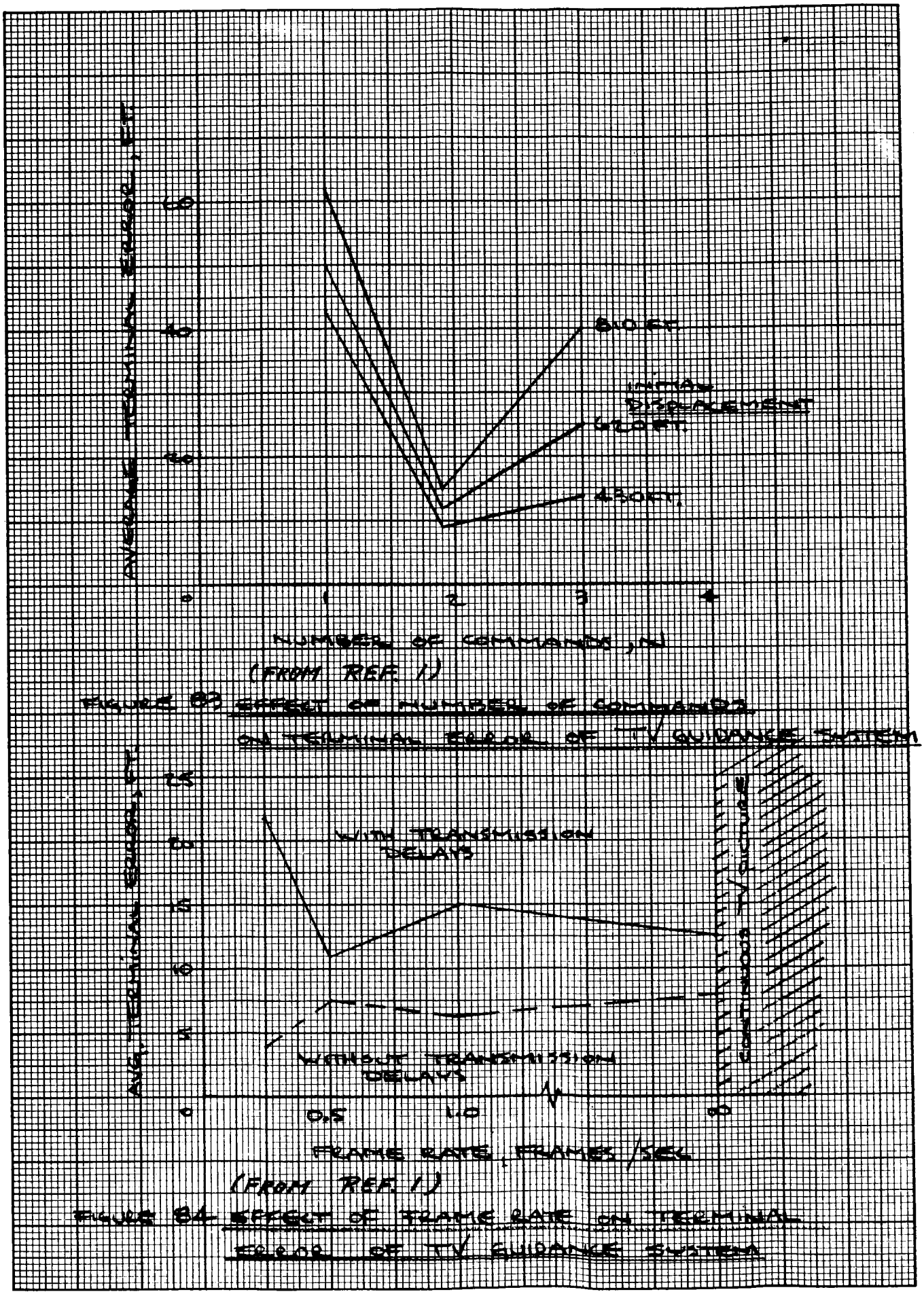
and that some form of back-up mode is highly desirable. Referring to the listing of the various techniques, the choice of back-up appears to be limited to either visual guidance or the celestial-inertial-altimeter combination. It is quite probable that the vehicle will have inertial sensor elements installed for attitude control and cis-lunar guidance and that one of the three beacons could be used with an altimeter sensor. Therefore, equipment that is already installed could probably function as an inertial-altimeter combination, back-up terminal guidance. However, it would not be completely independent of the beacons. On the other hand, a video system, consisting of TV sensors, will perform, as an independent system, a) performance monitoring, b) back-up control, and c) correction of "last minute" position errors.

2. The video terminal guidance method employing real-time video sensor display, in combination with an inertial measurement unit and Earth control, also appears to have a great probability of success. Examination of a terminal guidance scheme of this type (Reference 1 and 2), involving a hovering phase, has shown that television command could give accuracies in the order of five feet from a visual reference. Since that time, a number of studies have been made of guidance systems that do not require a hovering phase, with its attendant velocity penalty, and that allow continuous corrections of up to several miles during the landing phase. Lunar landing television-link guidance studies utilizing a simulator system (References 3 and 4) have been performed recently. Figures 83 and 84 indicate the landing accuracies obtainable using a television command link from Earth with extremely simple control laws and hardware requirements. It is clear, from these results, that landing accuracy should not be a problem for lunar logistic-support vehicles.

Recommendations

Since the preceding analysis has pointed out that a video terminal guidance system holds great promise as either the back-up mode for a beacon system or the prime terminal guidance method, further studies should be conducted to investigate the capabilities of near-future video sensors in the areas of image quality and resolution in detecting distances, angular displacements, and their rate of change. In addition, power and bandwidth requirements should be determined.

The effect of two-way propagation time lag and the operator's reaction time on terminal guidance accuracy should be investigated realistically and is best determined through the utilization of a simulator system. During the next phase of the Apollo II research and development studies, it is intended that a simulation system, including lunar terrain maps, video sensors, and guidance control panels will be fabricated and utilized in a test program to determine, finally, the feasibility of visual terminal guidance.



(FROM REF. 1)

FIGURE 63 EFFECT OF NUMBER OF COMMANDS ON TERMINAL ERROR OF TV GUIDANCE SYSTEM

(FROM REF. 1)

FIGURE 64 EFFECT OF FRAME RATE ON TERMINAL ERROR OF TV GUIDANCE SYSTEM

TABLE 37

COMPONENT WEIGHTS

3-BEACON TERMINAL GUIDANCE SYSTEM

EQUIPMENT	PRIME PURPOSE	SECOND PURPOSE	WEIGHT
3 Transponders (on Moon)			
1 Trans/Receiver	Homing	Altimeter	25 lb
Inertial	Attitude	Back-up Approach	10 lb
Doppler Receiver	Tracking	Altimeter	13 lb
Television	Monitor	Back-up Landing	5 lb
DSIF Transponder	Tracking	Data Link (telemetry, television)	5 lb

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APOLLO SM ENGINE THROTTLING

Concept 2, the SM-shelter, employs the current Apollo SM fitted with landing gear, unmanned guidance system, and a throttleable engine to accomplish the lunar landing. In addition to the guidance and landing gear analyses which were described earlier, a feasibility analysis for throttling the Apollo SM engine was performed and is described herein.

The feasibility of throttling pressure-fed, storable liquid propellant rocket engines has been demonstrated. Many tests of various throttling injector designs and techniques have shown that thrust control for ratios as high as 15:1 are within the current state-of-the-art for liquid propellant engine thrust variation. Several current engine development programs (e.g., Apollo LEM, Advanced RL-10-A3) are working to specifications calling for thrust throttling to ratios of 10:1. For the Apollo II, thrust variation in the throttling range of 5:1 to 8:1 will be necessary for the maneuvers anticipated during the hover, descent, and landing operations of the vehicle in the lunar gravitational field. The throttling requirement is similar, but less severe than that required for the LEM vehicle. The duration of this required engine throttling capability for the Apollo II mission is approximately ten percent of the total engine operating time.

Throttling of the present Apollo SM engine is an entirely feasible concept. The life of an ablatively cooled thrust chamber is primarily a function of the combustion temperature of the propellant combination and the operating chamber pressure. Any circumstance that lowers the value of either of these factors will be materially advantageous for a given engine. Since, in effect, the throttling of a rocket engine is accomplished by a lowering of the chamber pressure, the incorporation of throttling capability for the Apollo SM engine should pose not technical problems nor require any physical changes to the current chamber design. In fact, operation in a deep throttled mode should enhance the engine's service life.

The throttling of liquid propellant rocket engines has been examined by the industry through the application of several techniques. Included in these are throttling by upstream propellant valves, mechanically variable area injector orifices, variable area concentric tube injectors which are mechanically coupled with a variable area cavitating venturi throttle valve, and upstream propellant valves in combination with an inert gas dilution-aeration throttling technique. The two latter approaches are currently receiving the greatest attention and development of these techniques holds the most promise for the operational flexibility and reliability desired. Both of these methods are currently being evaluated and are under development for application to the Apollo LEM descent engine program.

On the basis of simplicity, ease of incorporation, demonstrated functionality, and the lowest potential weight and cost penalties, the inert gas dilution-aeration method is recommended for the Apollo II throttleable SM engine application. Its incorporation would involve only minor modifications to the current Apollo SM propulsion system. A brief description of this throttling technique, a comparison to other methods, and preliminary

estimates of the physical characteristics entailed for the "worse case" operating condition follow. Figure 89 presents a schematic sketch of the design.

Throttling with Helium Injection

The concept of throttling a rocket engine by means of a combination upstream valving and the injection of helium has been conceived, demonstrated, and is being developed for the LEM propulsion requirements wherein a throttling ratio of 10:1 will be achieved. It is particularly well suited to this application because it insures stable operation of the engine in the deep throttled condition and, at the same time, requires only the minimum possible propellant inlet pressure to the engine.

As shown schematically in Figure PS-1, the concept is merely that of providing a conventional upstream valve throttle system with additional helium lines running from the pressurant system to the downstream side of the throttling valves. The helium lines are provided with shut-off valves to permit introduction of helium at a predetermined thrust level. In practice, an extremely small amount of helium may be employed because the purpose of the helium is merely to maintain the injector pressure drop at an acceptable level while in a throttled condition. It is important to note that propellant flow control is still accomplished by the throttle valves alone. The introduction of the helium merely decreases the valve pressure drop by an almost insignificant amount. The injector pressure drop, however, is significantly increased, thus providing the desired margin of stability. A simple numerical example will illustrate this point:

> Assume that, at full thrust, a chamber pressure of 100 psi is used with an injector ΔP of 30 psi. Further, assuming a 50 psi drop for the feed system, a tank pressure of 180 psi will be required. Now suppose that this chamber is throttled to 1/10 rated thrust by means of the upstream valves only. It is seen that the chamber pressure will be approximately 10 psi, the injector ΔP will be about 0.3 psi, and the pressure drop across the throttle valves will be near 169.7 psi. This analysis assumes that the flow rate is in direct proportion to the thrust and a constant specific impulse is maintained:

$$\dot{w} = \frac{F}{Isp} \quad \text{and} \quad (\dot{w})^2 \sim \Delta P,$$

$$\text{so that:} \left(\frac{\dot{w}_2}{\dot{w}_1}\right)^2 = \frac{\Delta P_2}{\Delta P_1}$$

where: $\dot{w}$ = flow rate

P = injector pressure drop

F = thrust

Isp = specific impulse

If helium is introduced in the manner described, the injector ΔP may be raised to 10 psi with the result that the throttle valve pressure drop will stand at 160 psi. Thus, it is seen that sufficient injector pressure drop may be readily obtained with the introduction of helium while still vesting control of the engine thrust entirely with the throttle valves.

Based on alaysis and the preliminary results of the experimental programs currently being conducted, it is anticipated that total helium flow rates of approximately 0.04 lb/sec will be employed to achieve throttling of the Apollo SM engine at the highest ratio desired. Further, it is assumed for this analysis that this helium flow rate will be maintained at a constant value while in the deep throttling mode of operation. This would be accomplished by opening the helium valves at slightly less than 50 percent of rated thrust to insure sonic flow through the helium control orifices. Thus, if the engine operates in a throttle mode below this point for 100 sec., it is evident that the total helium requirement will be only four pounds, an increase of a modest 4% in total pressurant requirements assuming maximum throttling for the 100 second period under consideration. It is readily apparent that the final pressurant pressure should be insignificantly changed by the use of this small amount of helium to augment the throttling process.

Comparison with Other Throttling Schemes

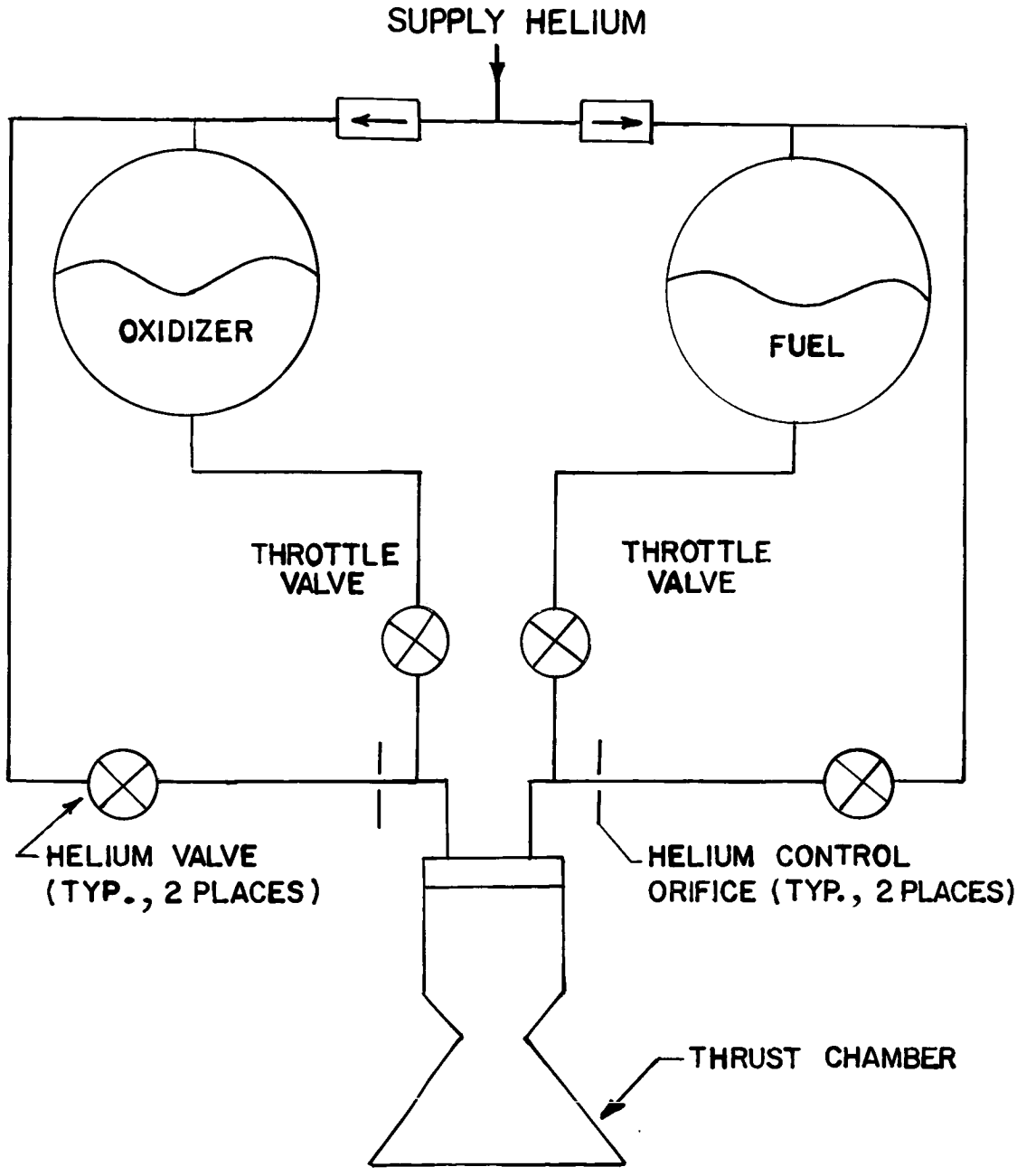
An inspection of the helium injection method as compared to other possible techniques indicates that it is superior for the Apollo II mission. It appears that there are primarily two other possibilities; namely, variable-area injectors and stepped injectors, wherein a large portion of the flow through the injector is either shut off entirely or drastically reduced while the remaining flow is throttled by means of upstream valving. In comparing the helium injection with the variable-area injector, it is seen that the helium injection method has the following advantages:

1) The attainment of high performance and good thrust chamber life can be more readily achieved because the design is completely unrestricted in the selection of injector patterns. The problem may be attacked as if the engine were a simple fixed thrust unit. The use of a variable-area injector, on the other hand, necessitates the application of injector patterns that are restricted by the basic design of the throttling mechanism.

2) There are no moving parts of the throttle exposed to combustion in the chamber or the hard vacuum of outer space during shut-down of the engine.

3) The helium injection system is readily amenable to the provision of redundant throttling valves. Redundancy of the variable area throttling elements can only be obtained at the expense of further

FIGURE 89



SIMPLIFIED HELIUM INJECTION THROTTLING

complication of the injector.

Moreover, it is seen by the example above that the inlet pressure requirements of the helium injector system are no greater than those for a variable area system. In fact, conceivably, it can be even smaller because of the less difficult problem of obtaining full-thrust high performance. In comparison with the stepped injector, the following advantages of the helium injection system are apparent:

1) Since there is always flow through all elements of the injector, the helium aeration system will not suffer the difficulties of injector-face cooling.

2) The same continuous flow characteristics of the helium injection method give much greater assurance that combustion flow characteristics will not change during throttling, with the attendant difficulty of localized combustion chamber hot spots, because of the uniform dispersion of propellants which is realized.

3) Fewer main propellant valves are required with the helium injector system.

Physical Characteristics and Development

Based on preliminary design estimates, it is anticipated that the additional hardware required to effect application of the helium injection (aeration) throttling method to the Apollo SM engine will increase its weight by a maximum of three percent. The additional hardware involved includes only the necessary lines, valves, and controls for helium injection.

Since this method is currently in development for other programs (e.g., LEM), it should incur the smallest development time and costs for application of throttling capability to the Apollo SM engine. A conservative estimate of the development time required would be approximately twenty-four months from inauguration through qualification for an Apollo SM engine program incorporating these modifications.

As stated previously, it is anticipated that the propulsion system weight increase when incorporating aeration throttling would be relatively minor. However, in order to completely evaluate this method compared to a mechanical system, a substantiating detailed weight analysis will be required.

AIRLOCK OPERATIONS

One of the operational features which determines the size of the environmental control system is the method and frequency of airlock operations. Before the ECS can be designed, the number of airlock operations must be determined from an evaluation of astronaut activities and an airlock design concept postulated. Then the total daily atmospheric loss from the shelter may be calculated and the ECS sized accordingly.

In order to realistically evaluate airlock operations to permit airlock design, it is necessary to first envision what the astronaut activities were likely to be and how much time each basic activity could be expected to consume. Based on this information, a basic work-rest cycle is developed which, in turn, provides a basis for determining how often an astronaut will leave and re-enter the shelter. The number of airlock operations thus obtained is used to determine the atmosphere loss from airlock operations. The data presented in this analysis is shown in Figure 90.

Astronaut Activities

There appeared to be three basic time-consuming functions that would be performed by a lunar-landed crew. These functions are those of the capsule monitor, the outside explorer, and rest (sleep).

1) Capsule Monitor

The specific tasks of the capsule monitor would undoubtedly vary depending upon the equipment he is using, as well as the overall mission of the lunar crew. In general, however, he would be responsible for maintaining voice and/or video communications with Earth and with the outside explorers. He would also be engaged in such tasks as visually verifying the accuracy of the telemetry signals sent to Earth, monitoring the capsule environmental system, etc. Because of the uncertaintly associated with the effect of the lunar environment on shelter systems, it was assumed that the capsule systems would be monitored at all times. During those periods when an astronaut is outside the shelter, constant communication would need to be maintained with him. It was assumed that the capsule monitor and outside explorer would alternate functions. In a given twenty-four hour period, it is conceivable that there will be certain slack hours of relative inactivity. The capsule monitor would use this time to bring the crew log up to date, eat, exercise, for personal hygiene, or just free time. In cases where a total crew is composed of two men, and continuous capsule monitoring must be provided, the proposed schedule indicates that each man will be on duty as capsule monitor for twelve hours of each twenty-four. Three-man crews would require each man to serve for eight hours as capsule monitor.

2) Outside Explorer

The outside explorer represents another function for the crew. It can be assumed that this function will be one of general exploration and observation. The outside explorer would obtain surface rock samples, take pictures, test gravitational pull, and, in general, conduct a program of scientific investigation. If radiation and other environmental conditions permit, each astronaut would spend approximately six to eight hours outside the shelter. Because of requirements for rest and spacesuit limitations, this would be divided into two shifts of three or

four hours each. In two-man crews, six hours per man was assumed to be the maximum time available for outside exploration so that other activities could be performed.

3) Sleep

Another one of the functions that will be required is sleep. Though sleep requirements vary among individuals, it will probably be not less than six nor more than seven hours. To maximize the benefits of sleep, it was assumed that these periods would be continuous and regular.

Work-Rest Cycle

The number of personnel comprising the crew has a direct influence on determining the activities that can be performed and the tasks that can be accomplished. Work-rest cycles will undoubtedly be tailored to fit both the objectives of the particular lunar mission undertaken and the size of the crew involved. The personal preferences of the crew will also strongly influence any work-rest cycle that is ultimately accepted.

1) Two-Man Crews

If the crews are comprised of two men, the work-rest cycle should consist of three-hour units. This statement is based on two assumptions:

- The capsule controls must be monitored at all times.
- A minimum of six hours of continuous and regular sleep should be provided.

If the three-hour work cycle is accepted for two-man crews, then each man will spend approximately six hours in sleep, six hours (divided into two shifts of three hours each) outside the shelter, and twelve hours as capsule monitor (some of this latter time can be free time).

a) Advantages

1. Only one bed will be required if "hot-bunk" practice is followed.
2. Capsule controls will be manned and monitored at all times.
3. Sleep will be continuous and regular.

b) Disadvantages

1. Only six hours of sleep are provided.

[redacted]

2. Limited utilization is made of lunar daylight, (i.e., the astronauts are outside the shelter only one-half the time during the lunar day).

3. Rescue of outside explorer is severely limited if not actually impossible, unless the airlock is sized and designed for such an emergency.

2) Three-Man Crews

Three-man crews allow greater flexibility in the work-rest cycle. If four-hour shifts are used, then each man can spend eight hours as capsule monitor, eight hours (divided into two shifts of four hours each) as outside explorer, and eight hours in continuous and regular sleep.

a) Advantages

1. Only one bed is required if "hot-bunk practice is followed; two beds, however, should be furnished.

2. Capsule controls will be manned and monitored at all times.

3. Maximum utilization of lunar daylight is provided by continuous operations of outside explorer.

4. Eight hours of sleep will be continuous and regular.

b) Disadvantages

1. The proposed cycle is applicable only to the lunar day.

2. Since there is not likely to be an outside explorer during lunar night, a lunar night work-rest cycle would need to be drafted.

Airlock Operations and Pressure Losses

Airlock atmosphere losses are directly related to the number of airlock operations required. If each crewman leaves and re-enters the shelter twice in every twenty-four hour period, then the specific number of airlock operations is a function of crew size.

1) Two-Man Crews

If the above work-rest cycle is observed, then there will be eight airlock operations in each twenty-four hour period, four of which would be depressurizations. Assuming that the airlock chamber will have a total volume of 34.5 cubic feet (i.e., 7 feet high

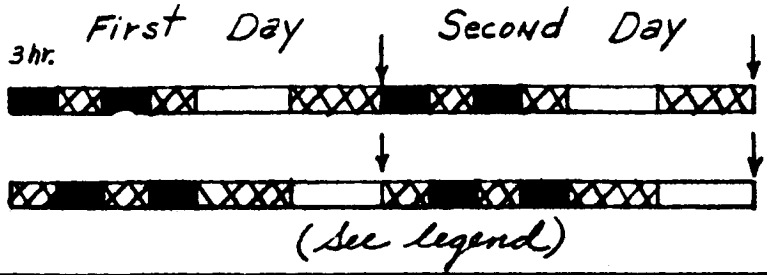
Manpower Provisions:	C
Anticipated Occupancy:	
Work - Rest Cycle:	1 2
Tasks to be Performed:	O Ca
Clock Operations:	7 u
Pressure Loss	5 0
Advantages:	1. 2. 3.
Disadvantages:	1. 2. 3

Legend:

- ■ Outside Monitor
- ⊠ Capsule Monitor
- ▭ Sleep

FIGURE 90

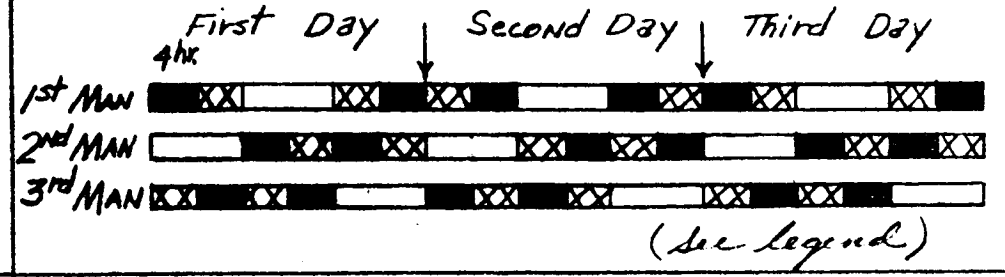
Rationale For Early Shelter Conc

Concept One LEM - LANDED SHELTER	Concept Tw APOLLO I SI
Two Men	Two Men
14 Days	30 - 60 D
 (See legend)	Same as Conc
tside Monitor: General exploration & observation; test space suit performance operation, and maneuverability; take samples; etc. sule Monitor: Communications with earth, outside monitor; take environmental readings, make maps; test samples; etc.	Similar to those Concept One. Th however, will c comprehensive scope.
ur depressurizations will be required each 24 hour period.	Same as Con
4.50 X 0.05 X 0.10 X 4.00 = 0.69 pounds/day r 9.66 pounds/14 days (See assumptions 2-5)	0.69 pounds/da 0.69 X 30 = 20.70 p (See assumption
Only one bed required if "hot bunk" practice is followed. Capsule controls are manned at all times. Sleep cycle is continuous and regular.	Same as Concept
Only six hours of sleep is provided. Limited utilization is made of lunar daylight (i.e. outside monitor in operation only one-half the time). Rescue of outside monitor is severly limited, if not actually impossible (i.e. if outside astronaut falls unconscious how can he be rescued).	Same as Concep

ASSUMPTIONS:

1. No outside monitor during lunar night
2. Size of circular airlock is 34.5 ft.^{3} (7' high with 2.6' diameter).
3. Airlock will be pressurized to 10 psi.
4. 90% of airlock atmosphere is salvageable during depressurization.
5. Density of airlock a is estimated at 0.

cpts.

o 1	Concept Three LLV - LANDED SHELTER
,	Two - Three Men
ays	120 Days
pt One.	 (see legend)
in e tasks, e more in	Similar to those in Concept One and Two, but greater detail and accuracy will be required.
ept One.	Six depressurizations will be required in each 24 hour period.
e or unds/60 days)	34.50 x 0.05 x 0.10 x 6.00 = 1.04 pounds/day or 1.04 x 60 = 62.40 pounds/120 days
t One.	1. Only one bed is required if "hot bunk" practice is followed. 2. Capsule controls are manned at all times 3. The eight hours of sleep is continuous and regular. 4. Maximum utilization of lunar daylight is provided by continuous operation of outside monitor.
t One.	
tmosphere 05 lbs./ft.^{3}	Purpose: These three concepts will form the basic phase of exploration and investigation into the lunar environment. As such, these modules will not be a physical part of any permanent lunar base complex.

with a 2.5 foot circular diameter) and that 90 percent of the chamber's atmosphere will be salvaged during each depressurization, then the approximate weight or atmosphere lost per day will be 0.69 pounds. In a fourteen-day period, this will amount to 9.66 pounds. Without salvaging the airlock atmosphere, ten times this amount, or 96.6 pounds, would be lost in fourteen days.

2) Three-Man Crews

If the above work-rest cycle is observed, then there would be twelve airlock operations, six of which would be depressurizations, in each twenty-four hour period. If the same airlock cahmber volume and slavage rates are used, then the approximate loss per day would be 1.04 pounds. During a 120-day occupancy, this would amount to 62.40 pounds if the assumption is made that no outside exploration would be conducted during lunar night. If the airlock atmosphere were not salvaged, then approximately 624 pounds of atmosphere would be lost.

Airlock Design Concepts

Outside the cabin, the astronaut's time is at a premium and so airlock operation time must be reduced to a minimum. Two systems are proposed herein in which the time loss for depressurization of the airlock can be reduced to anywhere from a few minutes to a fraction of a minute, and a large percentage of the gas, otherwise lost by venting the airlock to ambient, may be saved. This report presents an initial study of possibilities of the proposed methods for conserving part of the airlock atmosphere, using a minimum of additional components and requiring minimal power.

Three factors are of major importance:

a) Time lost by an astronaut in an airlock while depressurizing.

b) Weight of airlock atmosphere saved.

c) Power requirement for mechanical depressurization.

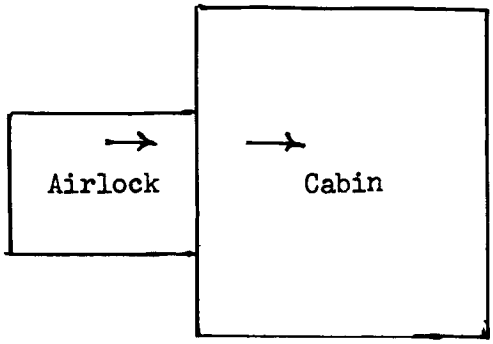
This discussion is applied to a pressurized cabin module in space or on a planetary surface having an airlock of smaller volume to act as a pressurization-depressurization chamber to enable an astronaut in a pressure suit to exit from or to re-enter the cabin module from the external space vacuum.

Cabin/Airlock Arrangements

Cabin and airlock arrangement with airlock pumped to cabin:

Case I: At $t = 0$:

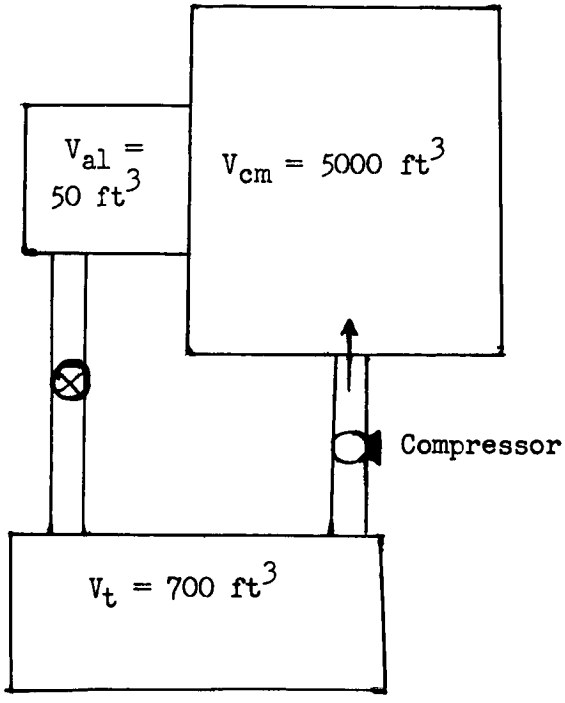
$$P_{al} = P_{cm} = 7.0 \text{ psia}$$



At time zero, the compressor starts and brings P_{al} from 7.0 psia to 0.5 psia. First, the airlock volume is held constant at 30 ft^3 and the cabin volume, V_{cm}, is varied as a parameter from 300 to 2000 ft^3. Secondly, the cabin volume is held constant at 2000 ft^3 and the airlock volume is varied from 30 to 100 ft^3. For various decompression times, from four to sixteen minutes, the average Hp requirement is computed. For calculation purposes, the pressure is lowered in 1.0 psi increments and the horsepower for each increment is found. This gives transient Hp data for any given depressurization time.

Cabin, airlock and tank - tank exhausted to cabin; airlock depressurized to tank:

Case II (a)



At t = 0:

P_{cm} = 7.0 psia

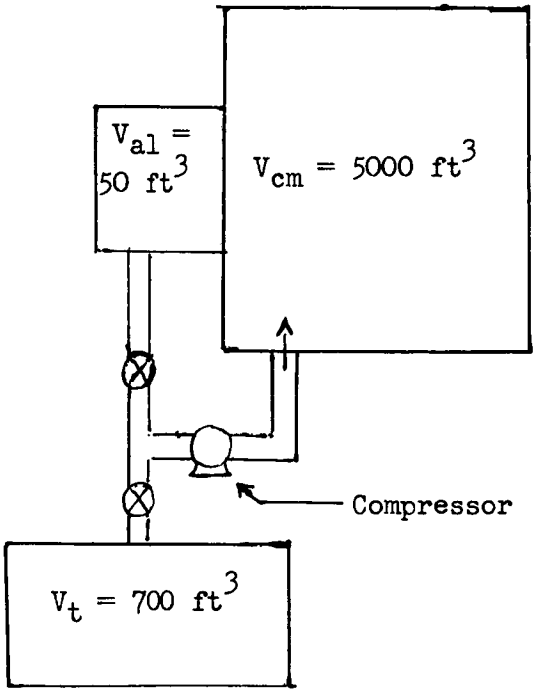
P_{al} = 0.935 psia

P_t = 0.935 psia

The compressor shown on the preceding page is designed to stop when P_t = 0.5 psia. Thus, just prior to opening the valve between airlock and tank, P_{al} = 7.0 and P_t = 0.5 psia. The volumes, V_{cm}, V_{al}, and V_t are held constant at 5000, 50, and 700 ft^3 respectively. After the valve is opened, pressure equilibrium occurs at 0.935 psia between the tank and airlock. At this time, the valve is closed and the compressor starts. Various decompression times from 15 to 180 minutes are considered and the average Hp requirements calculated. Also, for given depressurization times, P_t is decreased in 0.1 psi increments and the transient power requirements computed.

Cabin, airlock, and extra tank. Airlock depressurized to tank. Airlock and tank alternately decompressed by pump:

Case II (b)



This system is similar to Case II (a). Here, however, the compressor is capable of pumping either or both the airlock and tank atmosphere into the cabin. The airlock is vented to the tank until pressure equilibrium is reached (P_{al} = P_t = 0.935 psia). The tank then is sealed off and the airlock pumped down to 0.5 psia using a range of times from one to twelve minutes. These times are chosen so that the power requirements of the compressor will be about the same as those for tank pump-down using times from 15 to 180 minutes. After the airlock pressure, P_{al}, reaches 0.5 psia, the airlock is closed off and the tank is pumped down exactly as in Case II (a).

ENVIRONMENTAL CONTROL/LIFE SUPPORT SYSTEMS

The requirements and analysis of the environmental control and life support systems for shelter concepts 1, 2, 3, and the extended LOP as de-

scribed in this section. Several system combinations have been considered for comparison purposes and detailed weight, volume, and power requirements are presented for each.

Shelter Concepts 1 and 2

Concept No. 1 Shelter is assumed to have a totally stored system. Oxygen for a 100 percent oxygen atmosphere is stored subcritically at Earth lauch and used in the supercritical state when the shelter is occupied. Solid and liquid wastes will be collected separately, frozen, and stored in the frozen state beneath the lunar surface for future processing. Some chemicals may be required to reduce bacteriologica activity in the waste products prior to freezing. For purposes of this report, it is also assumed that all the water required in the life support subsystem will be stored. Lithium hydroxide is used, as in the Apollo vehicle, for carbon dioxide removal. The basic environmental control subsystem and the life support subsystems hardware and furnishings are assumed to be essentially the same as those presently used in the Apollo space vehicle.

Concept No. 2 Shelter is assumed to have a mixed oxygen-nitrogen atmosphere with the gases supplied from supercritical storage. For the longer standby and occupancy mission, it may be more economical from an Earth-launch weight standpoint to use regenerative systems, particularly the molecular sieve for carbon dioxide removal and water recovery systems to recover water from urine, wash water, and humidity control.

For comparison purposes, the following three system combinations are considered: System A consists of supercritical storage of atmospheric gases, lithium hydroxide for carbon dioxide removal, and stored water; System B consists of supercritical storage of atmospheric gases, molecular sieve for carbon dioxide removal, and water recovery; and System C consists of some supercritical storage of atmospheric gases, oxygen generation by electrolysis of water, methanation process of carbon dioxide reduction, a molecular sieve for carbon dioxide removal, and a water recovery system.

The basic ECLSS design criteria for shelter concepts 1 and 2 are shown below:

ECLSS Design Criteria
(Concepts 1 and 2)

	Concept 1	Concept 2
Standby (desired design point)	7 days	60-90 days
Occupancy (desired design point)	14 days	30-60 days
Crew	2 men	2 men
Atmosphere Control		
Compartment atmosphere (occupied)	O_2	$O_2 + N_2$
Total pressure	5 psia	7 psia

Oxygen partial pressure	5 psia	3.5 psia
Carbon dioxide partial pressure	0.14 psia	0.14 psia (max.)
Temperature	60-70°F	60-70°F
Relative humidity	40-70%	60-70%
Diluent	None	Nitrogen
Leakage rate	5 lb/day	5 lb/day
Compartment volume	210 ft^3	210 ft^3
Airlock volume	30 ft^3	30 ft^3
Crew Requirements - (on basis of 14,500 BTU [3650Kcal] /man-day)		
Metabolic Consumption		
Oxygen (lb/man-day)	2.26	2.26
Water (food & drink) (lb/man-day)	6.00	6.00
Food (freeze-dried) (lb/man-day)	1.66	1.66
Metabolic Output		
Carbon dioxide (lb/man-day)	2.76	2.76
Perspiration and Respiration (lb/man-day)	4.09	4.09
Urine (lb/man-day)	2.85	2.85
Fecal water (lb/man-day)	0.17	0.17
Feces (solids) (lb/man-day)	0.05	0.05

The data presented in Tables 17, 18, and 19 of the Systems Design & Analysis section of Part II of this report are applicable to concepts 1 and 2:

Table 17	Airlock Losses
Table 18	Resupply Weights for Space Suit Backpacks
Table 19	Weight of Materials Expended due to Extravehicular Excursions

The basic oxygen and nitrogen storage weights are presented in Tables 42 and 43 for concepts 1 and 2 and systems A, B, and C as a function of occupancy time. These basic weights increase with standby time as shown in Figure 91; Figure 92 presents the total storage volume as a function of storage weight.

The total ECS weight, considering supercritical storage and no standby venting allowance is shown in Table 44 for concept 1 with respect to occupancy time. Total weight of systems A, B, and C are shown in Table 45 for concept 2 with respect to occupancy time. These values are combined with those of Figure 91, which accounts for standby venting of the supercritically stored gases, and are shown in Figures 93, 94, and 95 to present total system weight as a function of standby and occupancy time.

From a viewpoint of minimum system weight, system B appears to be the

TABLE 42*

BASIC OXYGEN STORAGE REQUIREMENTS** - Pounds, for two men

FOR TWO-MAN SHELTER CONCEPTS

Item	Concept 1 14 day Occupancy	Concept 2 System A & B		Concept 2 System C	
		30 day Occupancy	60 day Occupancy	30 day Occupancy	60 day Occupancy
Metabolic	63.3	135.5	271.0		
Leakage	70.0	80.0	160.0		
Sub Total	133.3	215.5	431.0		
50% Reserve	66.7	107.7	215.5	107.7	215.5
Sub Total	200.0				
Airlock	53.1	39.8	79.9	39.8	79.9
Backpack	37.8	75.6	151.2	75.6	151.2
Repressurizations	5.9	23.4	46.8	23.4	46.8
Puncture	5.7	7.2	14.4	7.2	14.4
Total = W_t	312.4	469.2	938.8	253.7	507.8
Volume = W_t + 68.8	4.6 ft^3	6.8 ft^3	13.6 ft^3	3.7 ft^3	7.3 ft^3
Vessel Weight	160.5	232.0	465.0	129.0	269.0
Hardware and Plumbing	32.1	46.4	92.0	25.8	53.8
TOTAL BASIC STORED WT. (No Standby)	505.0	747.6	1495.8	408.5	830.6

*Tables numbered serially to other interim reports: LSS 23/63/7, LSS 23/63/8, LSS 23/63/9, and LSS 23/63/13

**No power penalties included.

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PREPARED BY: R. KRAUSE

CHECKED BY:

DATE:

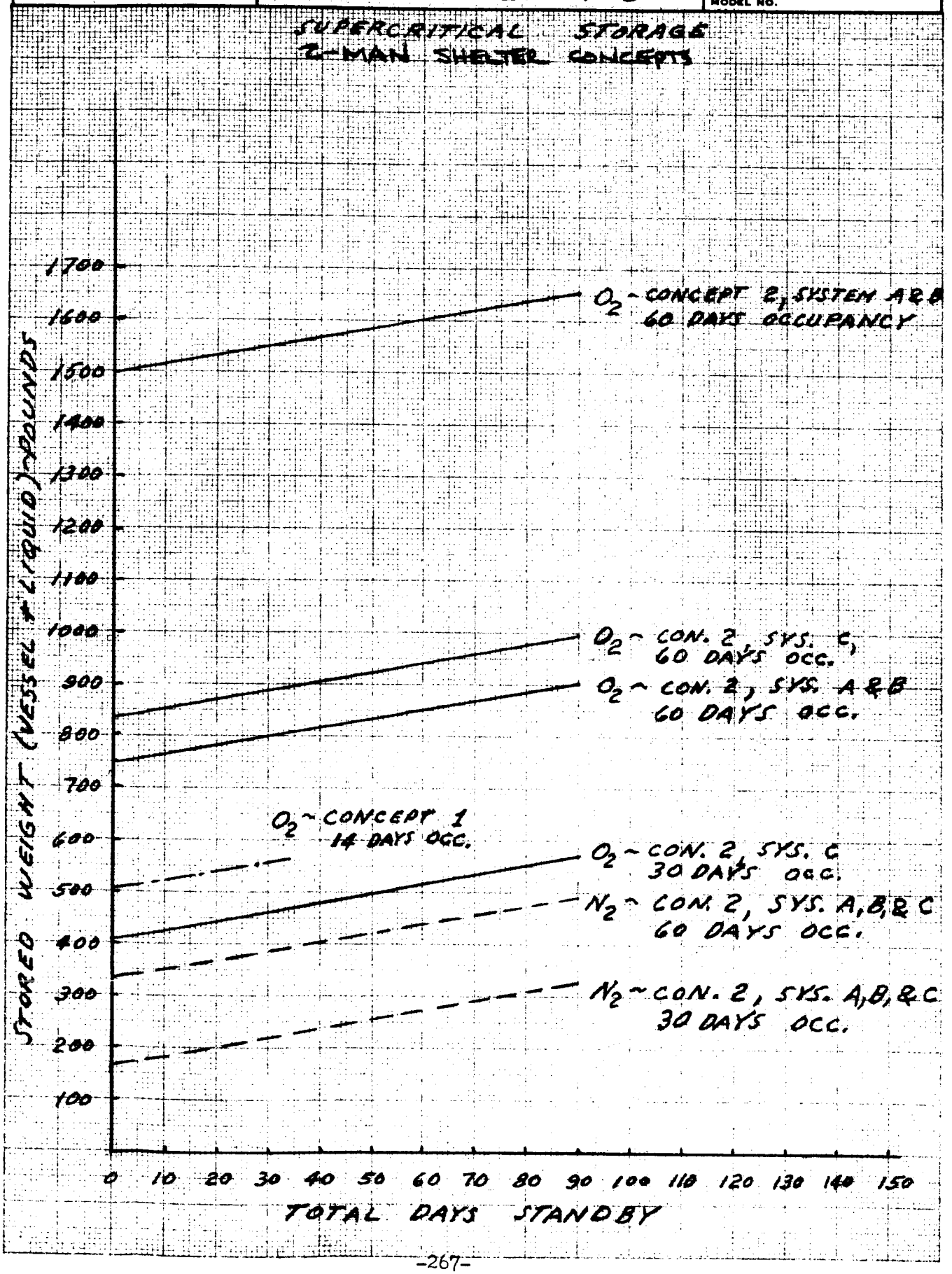
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MODEL NO.

FIGURE 91
STORED WEIGHTS



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FIGURE 92

VESSEL VOLUME

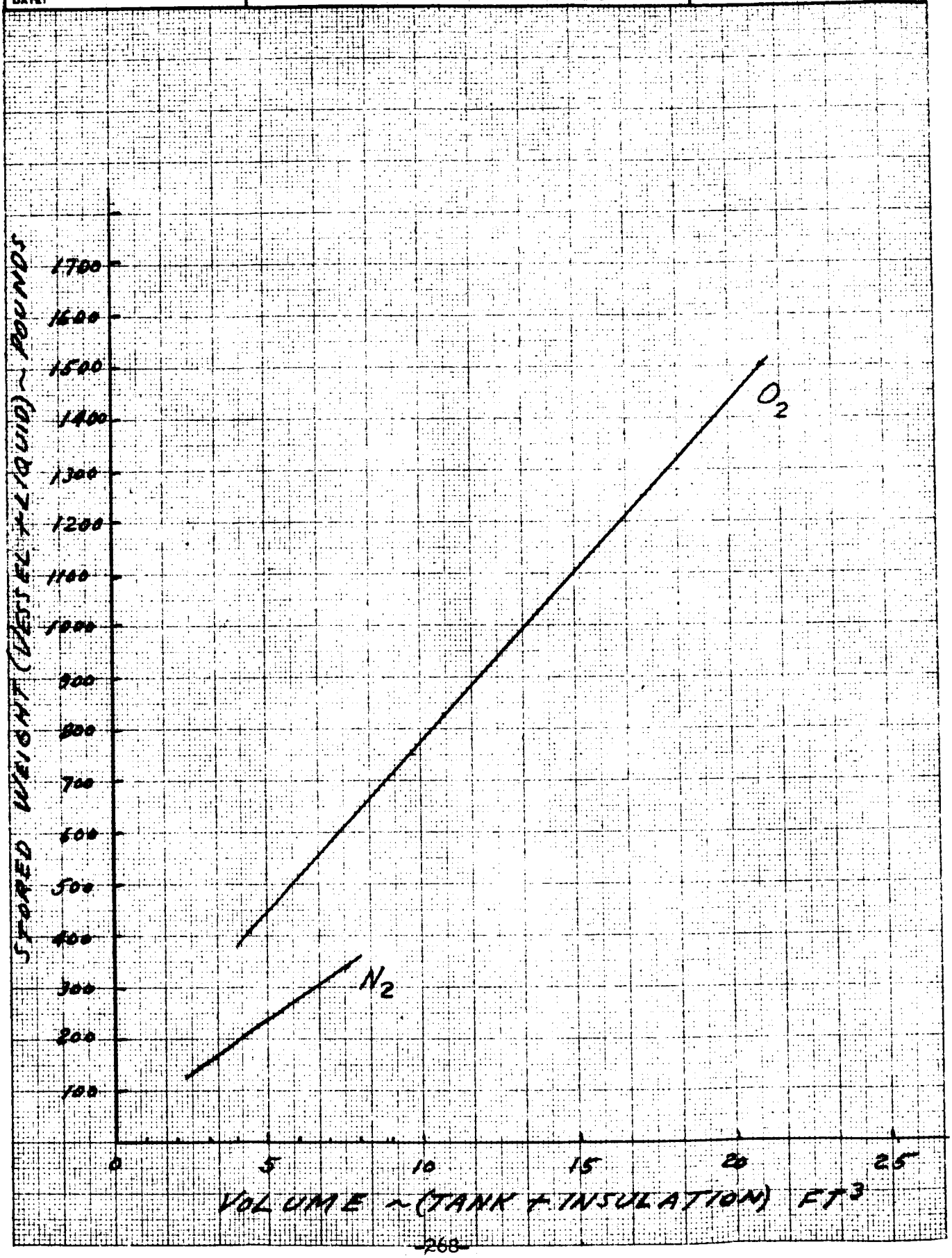


TABLE 43

BASIC NITROGEN STORAGE REQUIREMENT*

Pounds

Item - Nitrogen	Concept 2 Systems A, B, C	
	30 Day Occupancy	60 Day Occupancy
Structural Leakage	65.3	131.6
15% Margin	9.8	19.7
Sub Total	75.1	151.3
Airlock	34.1	69.6
Repressurizations	14.4	28.9
Puncture	6.3	12.6
Total W_t	130.6	262.4
Volume = W_t + 49.4	2.64 ft^3	5.30 ft^3
Vessel Weight	29.0	58.0
Hardware and Plumbing	5.8	11.6
Total Basic Stored Weight, lb (No Standby)	165.4	332.0

* No power penalties included

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TABLE 44

TOTAL SYSTEM WEIGHT* - No Standby

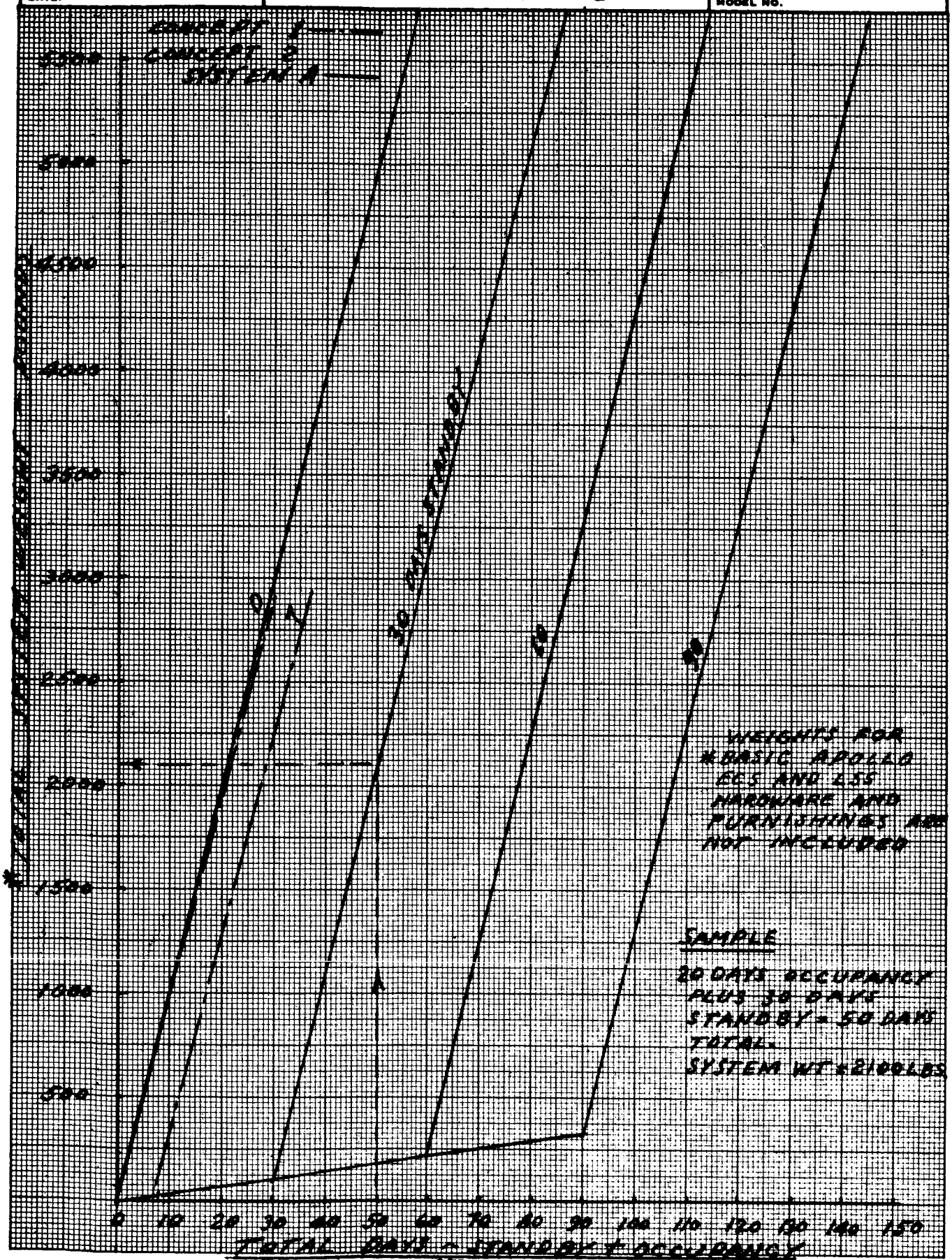
Concept 1

100% O_2 Stored System	14 Days	28 Days
CO_2 and Contaminant Removal, lbs	135.0	270.0
Supercritical O_2 (Storage Tank)	505.0	1,010.0
Water and Tank	695.5	1,391.0
Food	48.0	96.0
Total, lb	1,383.5	2,767.0

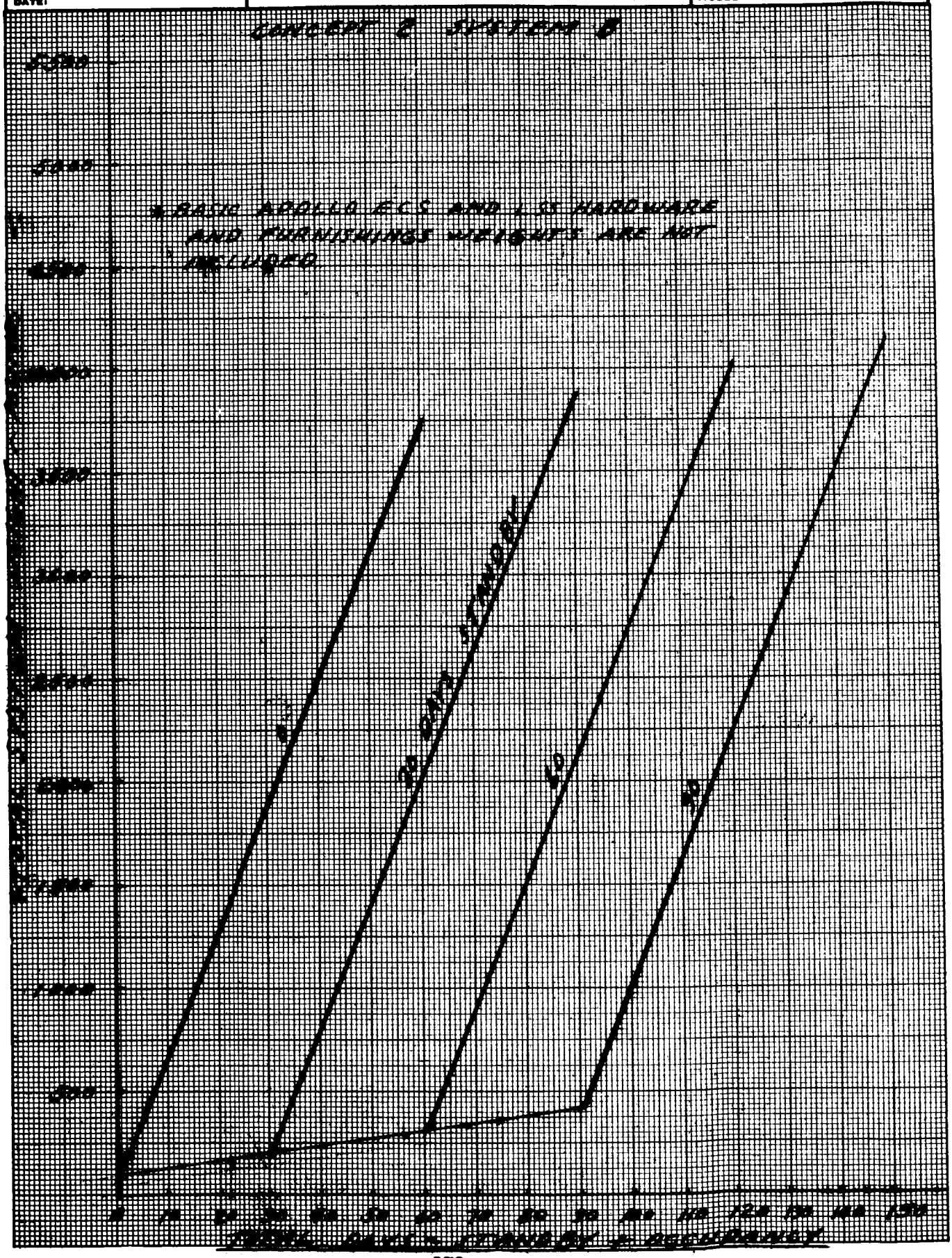
*Power penalties not included.

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CHECKED BY:	FIGURE 93	REPORT NO.
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DATE:	TOTAL SYSTEM WEIGHT	MODEL NO.



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CHECKED BY:	FIGURE 95	REPORT NO.
DATE:	TOTAL SYSTEM WEIGHT	MODEL NO.

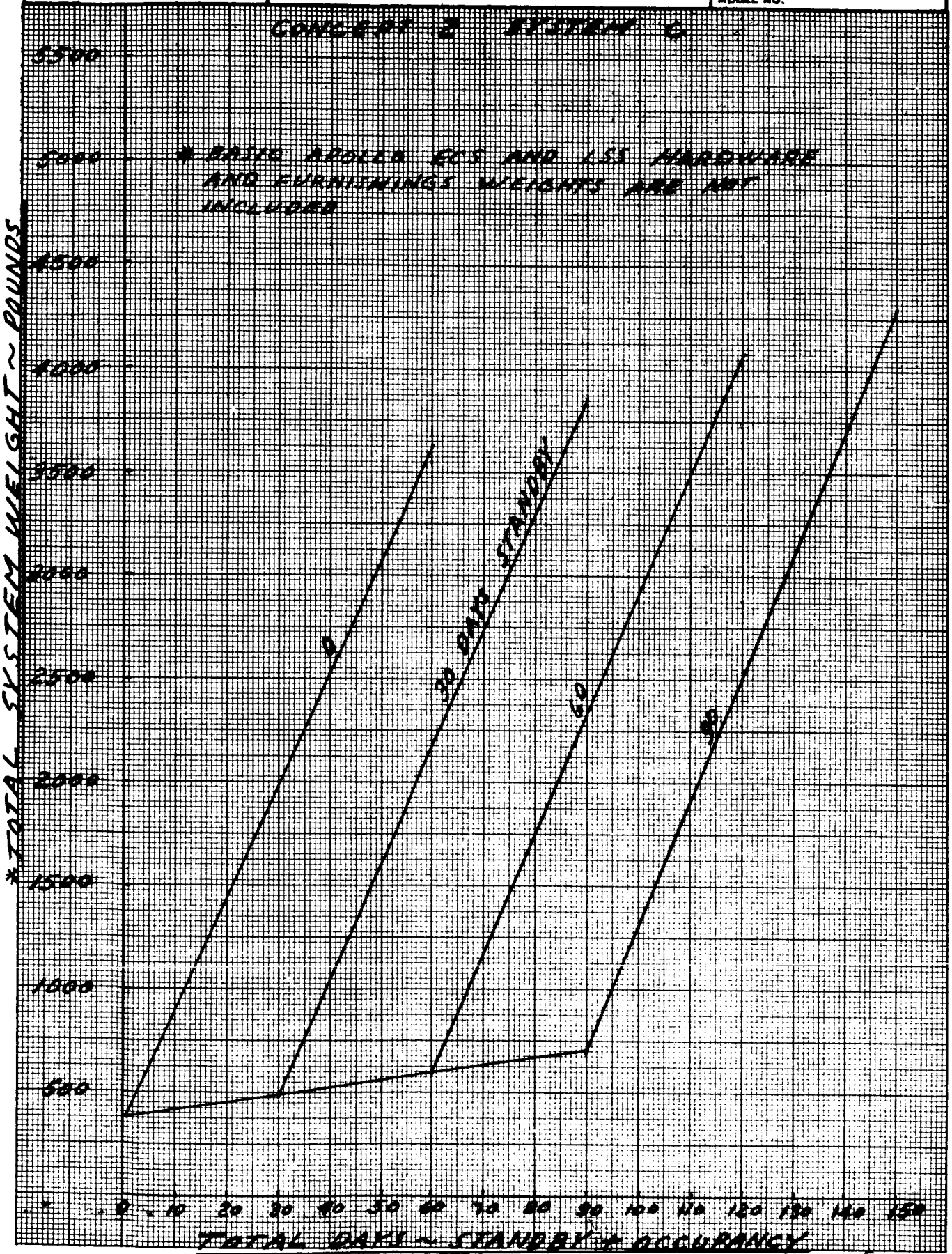


Table 45

TOTAL SYSTEM WEIGHTS* - No Standby

Concept 2

	System A		System B		System C	
	30 Days	60 Days	30 Days	60 Days	30 Days	60 Days
Fixed Weight Items, lb						
Contaminant Control Unit	----	----	----	----	2.3	2.3
O_2 Generation Unit	----	----	----	----	214.0	214.0
CO_2 Reduction Unit	----	----	----	----	66.0	66.0
Molecular Sieve	----	----	64.0	64.0	64.0	64.0
Water Recovery - Urine	----	----	25.0	25.0	25.0	25.0
- Wash Water	----	----	10.0	10.0	10.0	10.0
- Humidity	----	----	2.0	2.0	2.0	2.0
Total	----	----	101.0	101.0	383.3	383.3
Variable Weight Items, lb						
Supercritical O_2 (Gas + Tank)	747.7	1,495.8	747.6	1,495.8	408.5	830.6
Supercritical N_2 (Gas + Tank)	165.4	332.0	165.4	334.1	165.4	332.0
Water Recovery System Expendables			36.6	73.2	36.6	72.2
Make-up Water + Tank	1,470.0	2,940	774.0	1,548.0	870.0	1,740.0
Food	102.5	205.0	102.5	205.0	102.5	205.0
Lithium Hydroxide and Activated Charcoal Replacement Cartridges	568.0	1,035.0	----	----	----	----
Total	3,053.5	6,007.8	1,927.1	3,756.1	1,966.3	3,563.1

*Power penalties not included.
Basic Apollo system hardware and furnishings not included (viz. canister for the lithium hydroxide and activated charcoal).

most attractive for missions of long duration.

Electrical power requirements for the ECS of concept 1 and systems A, B, and C of concept 2 are shown in Table 46.

Shelter Concept 3 and Extended LOP

The environmental control/life support system concepts discussed herein are applicable to both shelter concept 3 and the extended mission LOP since the target times for standby and occupancy are similar (1-year standby, 120-day occupancy). Data has been generated considering crew sizes ranging from two to four men.

Requirements

The basic ECLSS design criteria for concept 3 and the extended mission LOP are presented below:

Standby	365 days
Occupancy	120 days
Crew	Two, three, and four men

ATMOSPHERE CONTROL

Compartment Atmosphere	(occupied)
Total pressure	7.00 psia
Oxygen partial pressure	3.50 psia
Carbon dioxide partial pressure	0.14 psia (max.)
Temperature	60-70°F
Relative Humidity	40-70%
Nitrogen diluent	
Leakage rate	10 lb/day
Compartment volume	3000 ft^3
Airlock volume	60 ft^3

Crew Requirements - 14,500 BTU (3650 Kcal) per man-day

Metabolic Consumption

Oxygen	2.26 lb/man-day

TABLE 46

SUMMARY OF DAILY ELECTRICAL POWER REQUIREMENTS

Item	Concept 1	Concept 2		
		System A	System B	System C
Atmosphere Circuit Compressor	150	150	150	150
Air Recirculation Blower	40	40	40	40
Temperature & Pressure Controls	7	7	7	7
Compartment Cooling EGW Pump	54	54	54	54
Electronic Cooling EGW Pump	54	54	54	54
Catalytic Filter		10	10	10
Molecular Sieve			210	210
CO_2 Reduction				276
O_2 Generation				1570
Water Recovery - Humidity	—	—	—	4
Total Watts, Continuous	305	315	525	2375
Water Recovery, Urine			123	123
Water Recovery, Wash Water			60	60
Total Watt-hours			183	183
Gas Monitoring and Analysis Unit	5	5	5	5
Water Heater (if required)	500	500	500	500
Total Watts (Intermittent)	505	505	505	505

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Water (food & drink)	6.00 lb/man-day
Food (freeze-dried)	1.66 lb/man-day
Carbon dioxide	2.76 lb/man-day
Water	
Perspiration & Respiration	4.09 lb/man-day
Urine	2.85 lb/man-day
Fecal water	0.17 lb/man-day
Feces (solids)	0.05 lb/man-day

Daily oxygen requirements are shown in Table 47. Added to the daily requirements will be quantities for initial pressurization and one emergency repressurization, as well as vent losses and residual gas. Each pressurization of the compartment will require 59.1 lb of oxygen, giving a total of 118.2 lb.

TABLE 47

Daily Oxygen Requirements

Item	Number of Men		
	Concept #3 Shelter & LOP		
	2	3	4
Oxygen-Metabolic	4.52	6.78	9.04
Leakage	5.34	5.34	5.34
50% Margin	4.93	6.06	7.19
Four Airlock cycles	4.73	4.73	4.73
Back-pack 2/man-day	2.09	3.14	4.19
Total Daily Reqmt	21.61	25.75	30.49

Landing module tanks will contain, at time of lunar touch-down, approximately 1560 pounds of oxygen at 67 psia pressure. The present insulation is 0.25 inches thick. After a relatively short period of time, heat leaks will start the gas to expand and vent overboard. In order to retain some of the residual oxygen during extended standby, additional insulation will be required. For example; it appears feasible to have approximately 780 lb of oxygen for use in the manned shelter at the end of 365 days of standby on

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the lunar surface if an additional 1.00 inch of superinsulation is installed at a weight penalty of 700 lb. The 780 lb of oxygen will furnish approximately twenty-six days' supply for two men, twenty-two days for three men, and eighteen days for four men.

If the residual hydrogen is not to be used for fuel cells or other purposes, it may be used to provide a heat sink to prevent excessive venting of the stored oxygen. Further study in considerable detail is required to establish coherent trade-offs involving the effects of additional insulation on the LOX propellant tanks and the use of residual hydrogen for heat sink purposes. The daily nitrogen requirements are shown in Table 48.. Added to the daily requirements will be quantities for initial pressurization and one emergency repressurization. Each pressurization of the compartment will require 51.70 lb of nitrogen, giving a total of 103.40 lb. This represents a stored weight of 325.71 lb and a volume of 35.50 cubic feet for high pressure storage at 2500 psia.

TABLE 48

Daily Nitrogen Requirements

Item	Number of Men		
	Concept #3 Shelter & LOP		
	2	3	4
Nitrogen-	4.66	4.66	4.66
Leakage, lb	4.66	4.66	4.66
15% Margin, lb	0.70	0.70	0.70
4 Airlock Cycles, lb	4.13	4.13	4.13
Total Daily Reqmt, lb	9.49	9.49	9.49
Volume, ft^3	1.03	1.03	1.03
High Press. Vessel, lb (2500 psia)	20.41	20.41	20.41
Total Stored Weight, lb	29.90	29.90	29.90

The daily water requirements are shown in Table 49.

ECLSS Design Concepts

Based upon the design requirements presented above, four Environmental Control/Life Support System concepts were designed and compared on a weight

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TABLE 49

Daily Water Requirements

Item	Number of Men		
	Concept #3 Shelter & LOP		
	2	3	4
For Food & Drink, lb	12.00	18.00	24.00
Personal Hygiene, lb	60.00	90.00	120.00
Make-up, O_2 Generation, lb	12.50	14.85	17.60
Subtotal	84.50	122.85	161.60
Leakage Losses, lb	1.08	1.08	1.08
Airlock Losses, lb	0.20	0.20	0.20
Backpack Cooling, lb	16.00	24.00	32.00
Total Required, lb	101.78	148.13	194.88

basis. These preliminary designs assumed usage of an oxygen generation system, a water recovery system, four airlock operations per day, and two extravehicular excursions per man per day in spacesuits. Although later studies may prove beyond doubt the feasibility of cryogenic storage of oxygen, nitrogen, and hydrogen for standby periods of one year, this study assumed high pressure storage of oxygen and nitrogen.

All concepts included high pressure storage tanks for oxygen and nitrogen required for shelter pressurization. ECLSS concept A consists of a water recovery system, molecular sieve, CO_2 reduction system, and an O_2 generation system. The molecular sieve collects the CO_2 produced by the crew. The carbon dioxide is then reduced, by addition of hydrogen, to methane and water. The water is then electrolysed to form hydrogen and oxygen. The oxygen generator is sized to furnish the basic oxygen requirements. The hydrogen formed is supplied to the carbon dioxide reduction process. In order to produce the required amount of oxygen, water in addition to that formed by carbon dioxide reduction is required. Since the oxygen being generated is of relatively low pressure, a separate storage of high pressure oxygen gas is required for initial and one emergency pressurization and for recharging the backpack supply. A summary of fixed and variable system weights for two, three, and four men is presented in Table 50. Total system weight as a function of occupancy time is shown in Figure 96.

ECLSS concept B consists of a water recovery system and a molecular sieve. All atmospheric constituents are stored under high pressure. A summary of fixed and variable system weights for two, three, and four men is presented in Table 51. Total system weight as a function of occupancy time is shown in Figure 97.

ECLSS concept C consists of a water recovery system and a molecular sieve. All nitrogen diluent gas is stored but sodium superoxide is used as a basic source of oxygen; some high pressure oxygen storage is required. Water vapor and carbon dioxide generated by the crewmen combines with sodium superoxide to form sodium carbonate, sodium bicarbonate and oxygen. In this system, some water will be lost in the process. The molecular sieve is considered necessary as a back-up and to prevent build-up of traces of carbon dioxide which may not enter into this reaction. A summary of fixed and variable weights for two, three, and four men is presented in Table 52. Total system weight as a function of occupancy time is shown in Figure 98.

ECLSS concept D also consists of a water recovery system and a molecular sieve. All nitrogen diluent gas is stored but chlorate candles are used as a basic source of oxygen. The principal here is the thermal decomposition of sodium chlorate into sodium chloride and oxygen; some high pressure oxygen storage is required. A summary of fixed and variable system weights for two, three, and four men is presented in Table 53. Total system weight as a function of occupancy time is shown in Figure 99. Electrical power requirements for the various ECLSS components are tabulated in Table 54.

Comparison of the total system weights of concepts A through D as a function of mission duration as plotted in Figures 96 thru 99 indicate that, from a minimum weight viewpoint, system concept A appears to be the most attractive.

In subsequent sutides, the impact of long-term lunar surface cryogenic storage on ECLSS design and weight economics will be explored in detail. In addition, the utilization of the propulsion system residual oxygen (2-5%) will be investigated in detail.

ELECTRICAL POWER SYSTEMS

From the electric power system viewpoint, this study is an extension and intensification of those reported in References SID 62-1473, 62-1466, and 62-1467, wherein power systems best suited to lunar logistics vehicles and lunar shelters were compared. In those studies, the following systems were considered.

1) Stirling engine - alternator

2) Cryhocycle - alternator

3) Fuel cells with dual porosity electrodes (Bacon type)

TABLE 50

Item	Concept #3 Shelter & LOP Number of Men		
	2	3	4
Fixed Weight Items, lb			
O_2 Generation	240.00	286.00	338.00
CO_2 Reduction	66.00	99.00	132.00
Molecular Sieve	64.00	64.00	64.00
Water Recovery - Urine	25.00	25.00	25.00
-Wash Water	10.00	10.00	10.00
-Humidity	2.00	2.00	2.00
Hi Press. O_2 - 2 Pressurizations	120.00	120.00	120.00
Tank	300.00	300.00	300.00
Hi Press. N_2 - 2 Pressurizations	103.40	103.40	103.30
Tank	463.00	463.00	463.00
Total Fixed Weight, lb	1393.40	1472.40	1557.40
Variables, lb/day			
Water Recovery - Urine	0.43	0.64	0.85
-Wash Water	0.79	1.19	1.59
Make Up Water	28.92	38.84	49.16
Tank	2.01	2.70	3.41
Food	3.42	5.13	6.84
Hi Press. O_2 - for Backpacks	5.40	8.10	10.80
Tank	16.20	24.30	32.40
Hi Press. N_2 - Diluent Storage	29.90	29.90	29.90
Total Variable Weight, lb/day	87.07	110.80	134.95

System Concept A - Weight Summary

TABLE 51

Item	Concept #3 Shelter & LOP Number of Men		
	2	3	4
Fixed Weight Items, lb			
Molecular Sieve	64.00	64.00	64.00
Hi Press. O_2 - 2 Pressurizations	420.00	420.00	420.00
Hi Press. N_2 - 2 Pressurizations	566.40	566.40	566.40
Water Recovery	37.00	37.00	37.00
Total Fixed Weight, lb	1087.40	1087.40	1087.40
Variables, lb/day			
Hi Press. O_2 Storage	64.83	77.25	91.47
Hi Press. N_2 Diluent	29.90	29.90	29.90
Water Recovery	1.22	1.83	2.44
Make Up Water	16.42	23.99	31.56
Tank	1.15	1.68	2.21
Food	3.42	5.13	6.84
Total Variable Weight, lb/day	116.94	139.78	164.42

System Concept B - Weight Summary

TABLE 52

Item	Concept #3 Shelter & LOP Number of Men		
	2	3	4
Fixed Weight Items, lb			
Molecular Sieve	64.00	64.00	64.00
Water Recovery	37.00	37.00	37.00
Hi Press. O_2 - 2 Pressurizations	420.00	420.00	420.00
Hi Press. N_2 - 2 Pressurizations	566.40	566.40	566.40
Total Fixed Weight, lb	1087.40	1087.40	1087.40
Variables, lb/day			
Sodium Superoxide	52.48	60.95	70.80
Hi Press. O_2 for Backpacks	21.60	32.40	43.20
Hi Press. N_2 Diluent Gas	29.90	29.90	29.90
Water Recovery	1.22	1.83	2.44
Make Up Water	17.65	25.41	33.21
Tank	1.24	1.78	2.33
Food	3.42	5.13	6.84
Total Variable Weight, lb/day	127.51	157.40	188.72

System Concept C - Weight Summary

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TABLE 53

Item	Concept #3 Shelter & LOP Number of Men		
	2	3	4
Fixed Weight Items, lbs			
Molecular Sieve	64.00	64.00	64.00
Water Recovery	37.00	37.00	37.00
Hi Press. O_2 - 2 Pressurizations	420.00	420.00	420.00
Hi Press. N_2 - 2 Pressurizations	566.40	566.40	566.40
Total Fixed Weight, lb	1087.40	1087.40	1087.40
Variables, lb/day			
Chlorate Candles	50.26	58.40	67.57
Hi Press. O_2 for Backpacks	21.60	32.40	43.20
Hi Press. N_2 Diluent Gas	29.90	29.90	29.90
Water Recovery	1.22	1.83	2.44
Make Up Water	17.57	25.67	33.77
Food	3.42	5.13	6.84
Total Variable Weight, lb/day	123.97	153.33	183.72

System Concept D - Weight Summary

TABLE 54

Item	Concept #3 Shelter & LOP Number of Men		
	2	3	4
Atmosphere Circuit Compressor	150	150	150
Air Recirculation Blower	40	40	40
Temperature & Pressure Controls	7	7	7
Compartment Cooling EGW Pump	54	54	54
Electronic Cooling EGW Pump	54	54	54
Catalytic Filter	10	10	10
Molecular Sieve	210	315	420
CO_2 Reduction	276	414	552
O_2 Generation	1570	1870	2210
Water Recovery - Humidity	4	4	4
Total Watts, Continuous	2375	2915	3501
Water Recovery, Urine	122.5	183.5	245.0
Water Recovery, Wash Water	60.0	90.0	120.0
Total Watt-Hours	182.5	273.5	365.0
Gas Monitoring & Analysis Unit	5	5	5
Water Heater (If required)	500	500	500
Total Watts, Intermittent	505	505	505

Summary of Daily Electrical Power Requirements

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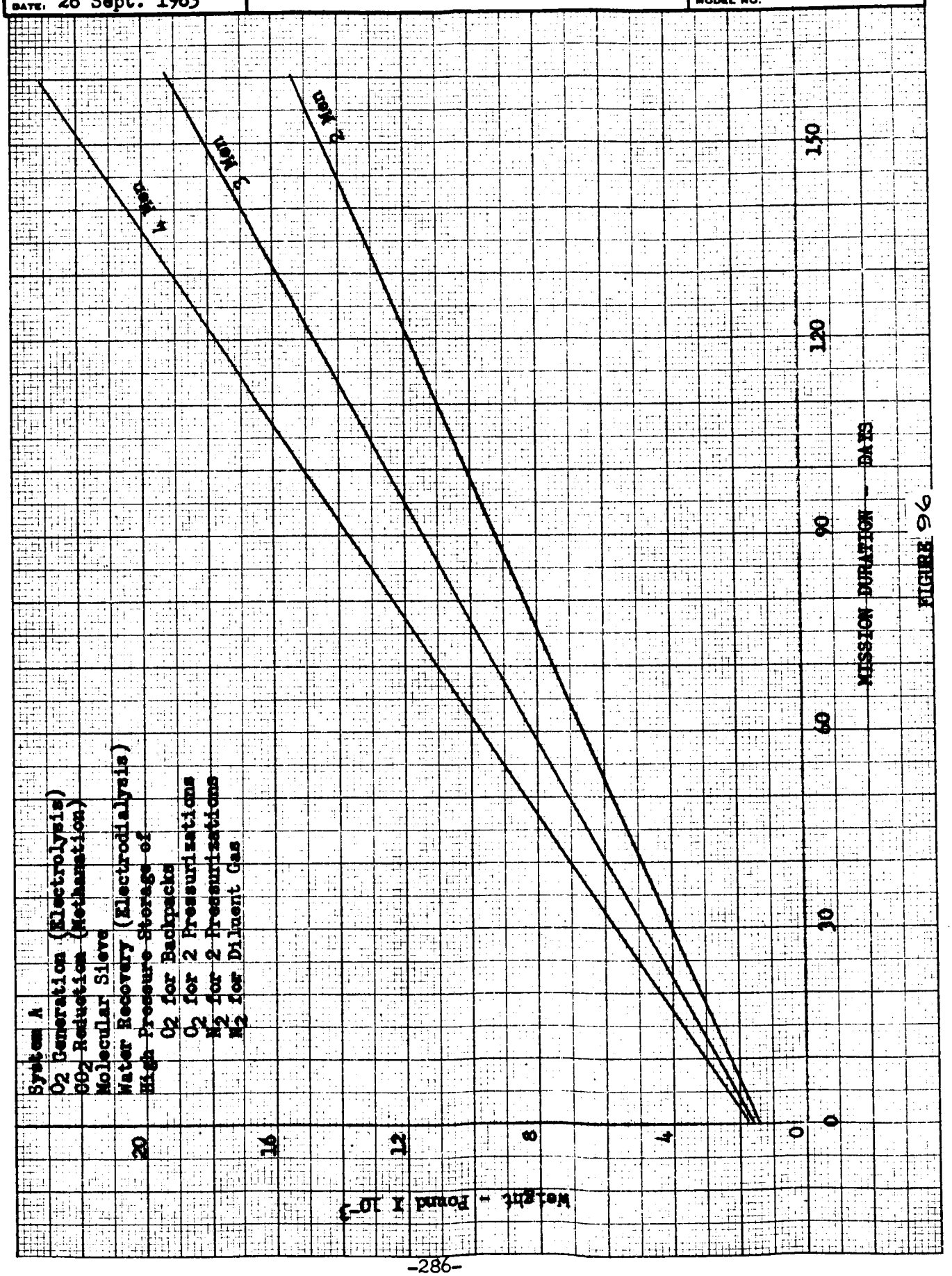


FIGURE 96

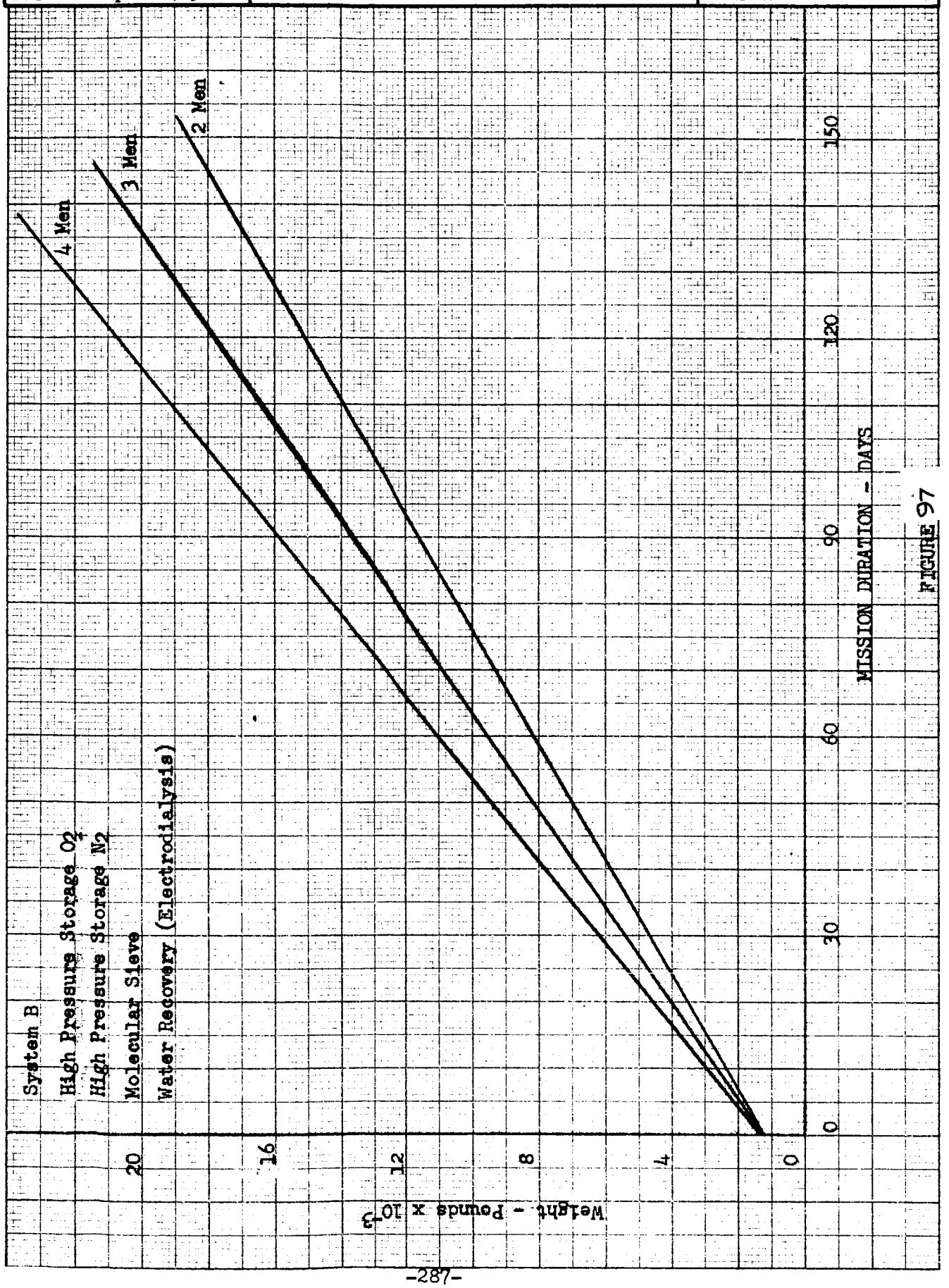

2 Men
3 Men
4 Men
System B
High Pressure Storage O2
High Pressure Storage N2
Molecular Sieve
Water Recovery (Electrodialysis)
Weight - Pounds x 10^-3
20
16
12
8
4
0
0
30
60
90
120
150
MISSION DURATION - DAYS

FIGURE 97

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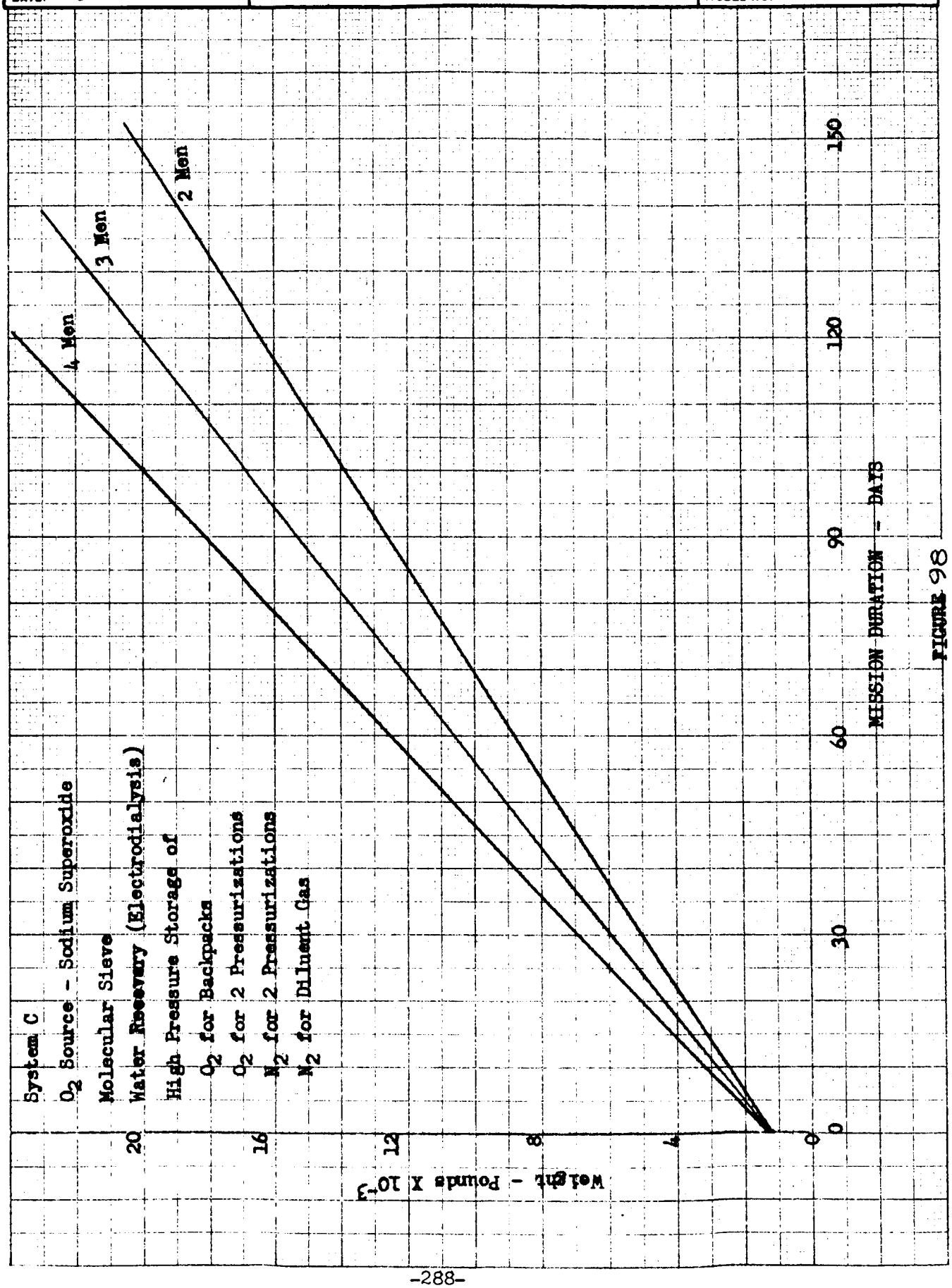


FIGURE 98

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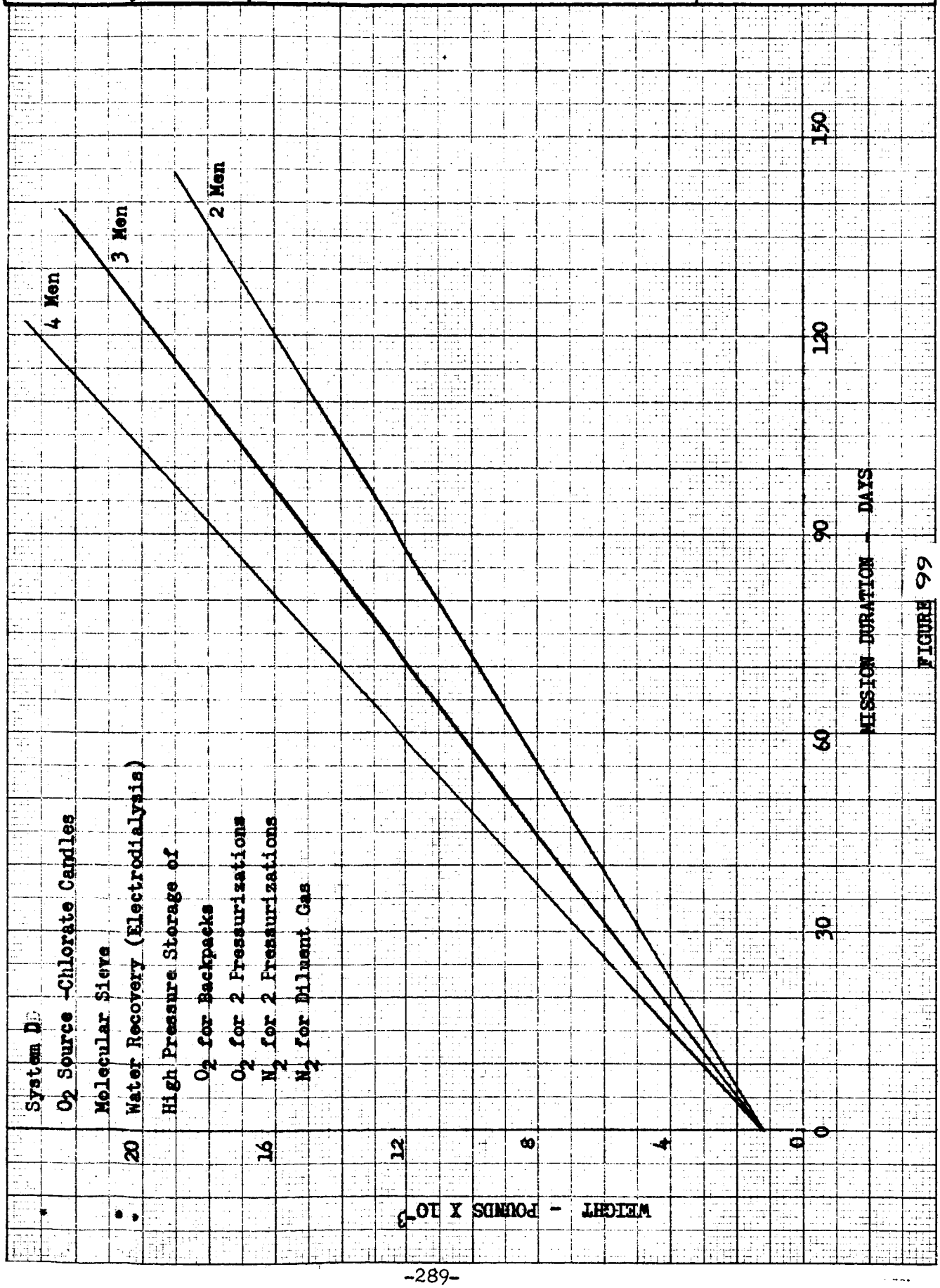


FIGURE 99

4) Fuel cells with ion exchange membrane separators

5) Brayton cycle with liquid hydrogen heat sink

6) Batteries hermetically sealed to handle peak loads and for emergency use only: rechargeable nickel-cadmium; rechargeable silver-cadmium; rechargeable silver-zinc; single discharge silver-zinc.

7) Radioisotopic dynamic - alternator

8) Radioisotopic thermoelectric

9) Radioisotopic thermionic

10) Solar heated turboelectric system (Sunflower)

11) Photovoltaic (solar cells) - rechargeable batteries

The missions have three major modes:

1) Trip to the Moon

2) Stand-by on the Moon

3) Occupancy mode on the Moon

During mode 1), most systems operate and have to be shut down after landing on the Moon. Part of this mode is the shut-down of the systems and readying for lunar storage all equipment. Some equipment has to be operated during mode 2). Storage period on the Moon such as the command receiver, telemetry, heating to maintain operation during lunar night and sufficient stand-by poer has to be available to start up the power system.

During mode 3) occupancy of the shelter, the systems are required to start up and must operate for the specified period.

Power for the storage period in most cases is very small and justifies the use of a secondary power system. For such a secondary power system, the radioisotopic thermoelectric system integrated with a battery (for peak loads) was selected as the best stand-by unit. A modified Snap 9A generator is used.

Two systems were selected for consideration in the occupancy mode. They are the Bacon type fuel cell system in the weight-saving configuration described in SID 62-1467 and the isotopic dynamic system.

In this study, the complete electric power system in both stand-by and occupancy modes was sized by electrical capacity, weight, and volume for each of the early lunar shelters. These include Concepts 1, 2, 3, and the extended LOP. All data are presented in parametric form wherever possible.

Fuel Cell Systems

Detailed estimates were prepared which gave weights and volumetric dimentsions of component parts in the fuel cell power supply system over the range 0-4 kw, for occupancy time in the shelter of 0-120 days. These data are presented in parametric form in Figures 100, 101, and 103. Fuel cell system weight in the Service Module is given in Figure 100; total fuel cell system weight is given in Figure 101, and complete power system weight, including the stand-by unit, is given in Figure 102. These curves formed the basis for the analyses reported herein. Caution must be exercised in using these figures because reactant losses during standby are ignored. The fuel system weight increases due to heat losses as a function of storage time. This is shown in Figure 109. To calculate the total weight of a fuel cell power system, compute the fuel system weight from curves 100, 101, and 102 by subtracting from the weight at actual occupancy time the weight at zero occupancy time and increase this weight according to Figure 109. Add this increased weight to the basic weight of the power system when considering stand-by time.

Example: 2kw power system operating fifteen days and having one month storage time: Use Figure 102.

> System weight for fifteen days occupancy is 1700 lb.
> System weight for zero days occupancy is 920 lb.
> Reactant system weight for fifteen days occupancy is 780 lb.
> Total energy output of the fuel cell system is 15 x 24 x 2 + 720 kw hours.

Total reactant consumption = 720 x 1 = 720 lb. This is also the weight of reactants which must be available to the fuel cells. From Figure 109 (b), the reactant system weight is 1.4 x 720 lb = 1008 lb. Add this value to the system weight for zero days occupancy. This gives 1008 + 920 = 1920 lb, which is the total electric power system weight, including standby and allowing for cryogenic feed losses.

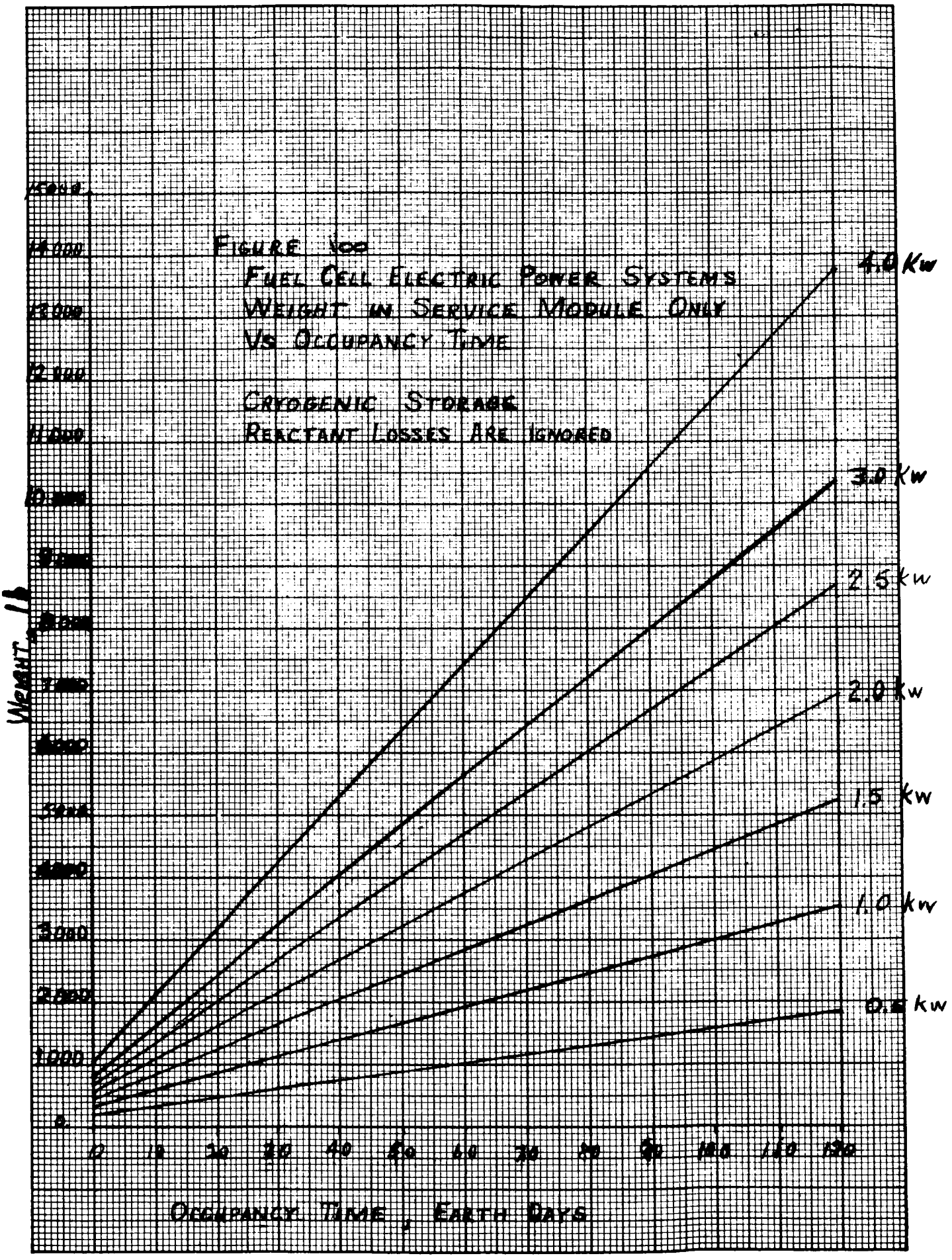
Radioisotopic-Thermoelectric Systems

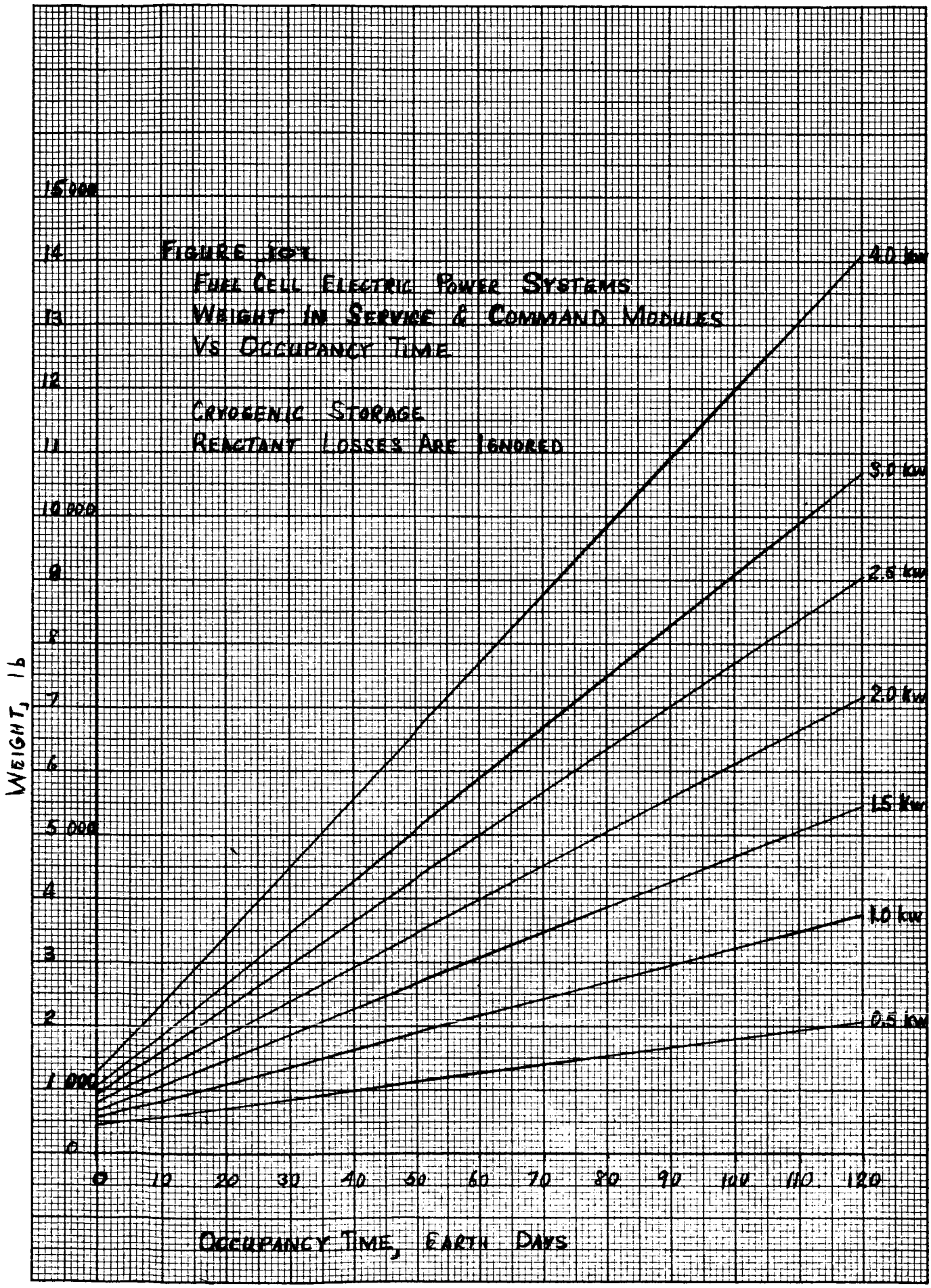
Electric power demand in the stand-by mode for all the early lunar shelters was determined by study to be 15 w, continuous average, with peaks to 40 w for a period of one week maximum. The power system as selected for this application, is a radioisotopic-thermoelectric system with a single discharge silver-zinc battery integrated by means of a heat transfer system in such a manner that the battery is maintained within required temperature limits even during the lunar night. The system is enclosed within a sealed rectangular box. One side of the box conforms to the shape of the Service Module and serves as radiator. It is assembled into the Service Module by brazing it into the aperture prepared for it. A controlled heat transfer system, in part radiative, in part natural convection of a polyphenylether dielectric heat transfer fluid, directs heat from the isotope heat generator to either (a) the battery and equipment in the Command Module, or (b) to space.

FIGURE 100

FUEL CELL ELECTRIC POWER SYSTEMS
WEIGHT IN SERVICE MODULE ONLY
VS OCCUPANCY TIME

CRYOGENIC STORAGE
REACTANT LOSSES ARE IGNORED




FIGURE 10-1
FUEL CELL ELECTRIC POWER SYSTEMS
WEIGHT IN SERVICE & COMMAND MODULES
VS OCCUPANCY TIME
CRYOGENIC STORAGE
REACTANT LOSSES ARE IGNORED
WEIGHT, lb
15 000
14
13
12
11
10 000
9
8
7
6
5 000
4
3
2
1 000
0
4.0 kw
3.0 kw
2.5 kw
2.0 kw
1.5 kw
1.0 kw
0.5 kw
0
10
20
30
40
50
60
70
80
90
100
110
120
OCCUPANCY TIME, EARTH DAYS

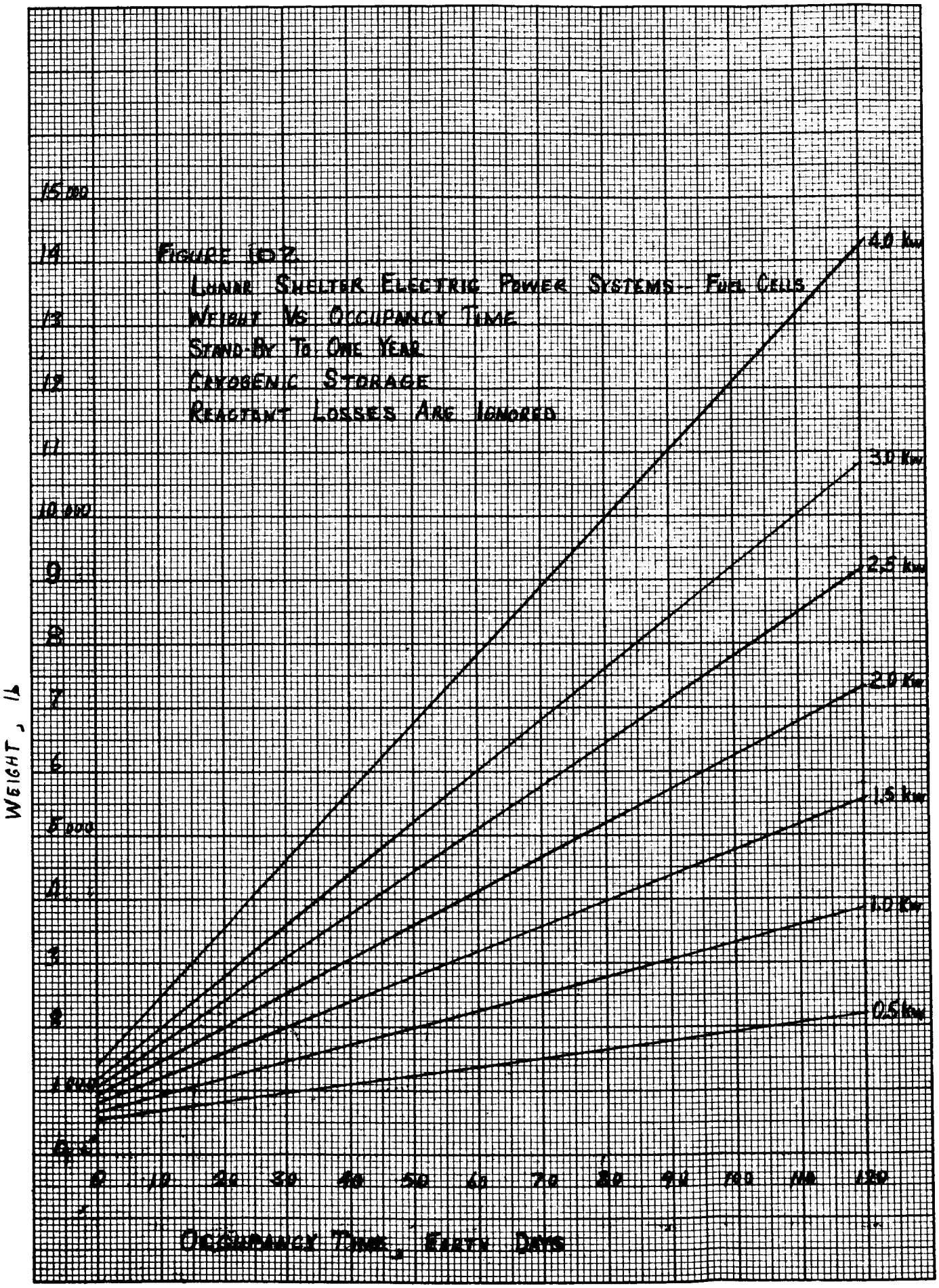


FIGURE 10.7.
LUNAR SHELTER ELECTRIC POWER SYSTEMS -- FUEL CELLS
WEIGHT VS. OCCUPANCY TIME
STAND-BY TO ONE YEAR
CRYOGENIC STORAGE
REACTANT LOSSES ARE IGNORED

Concept 1 Power Systems

The electrical power systems requirements, capacity, weight, and volume for the standby and occupancy modes of Concept 1 (CLEM-shelter) are described in the following sections.

Stand-by Mode

The electric power demand of 15 w continuous average and 40 w peaks is supplied by a radioisotopic-thermoelectric system integrated with a single discharge silver-zinc battery. Weights and volumes of the system components are presented below:

TABLE 55

STAND-BY POWER UNIT FOR CONCEPT 1:

WEIGHTS AND VOLUMES OF COMPONENT PARTS

	DIMENSIONS	WT., LB
Snap 9A, modification, without fins	8½ in. dia. x 11 in	20
Battery Ag-Zn primary	8 in x 7 in x 7 in	25
Structure, walls including radiator	30 in x 12 in x 10 in	19
Insulation and reflecting panels		2
Radiation (reflector shutters) controls		3
Heat Transfer Fluid, a polyphenyl ether	4 gallons	37
TOTAL SYSTEM	30 in x 12 in x 10 in	106 lb

If the requirement for operating the stand-by unit during the lunar night is deleted, the heat transfer system would be simplifed so that the addition of the heat transfer fluid becomes unnecessary. This results in reducing the weight by 37 lb to a total of 69 lb.

Occupancy Mode

Electric power demand for the occupancy mode of Concept 1 is tabulated in Table 56.

Weight of the complete 0.9 kw fuel cell system is shown as a function of occupancy time in Figure 103. For a twenty-one day occupancy period, weights and dimensional data of the fuel cell system components are shown in Table 58.

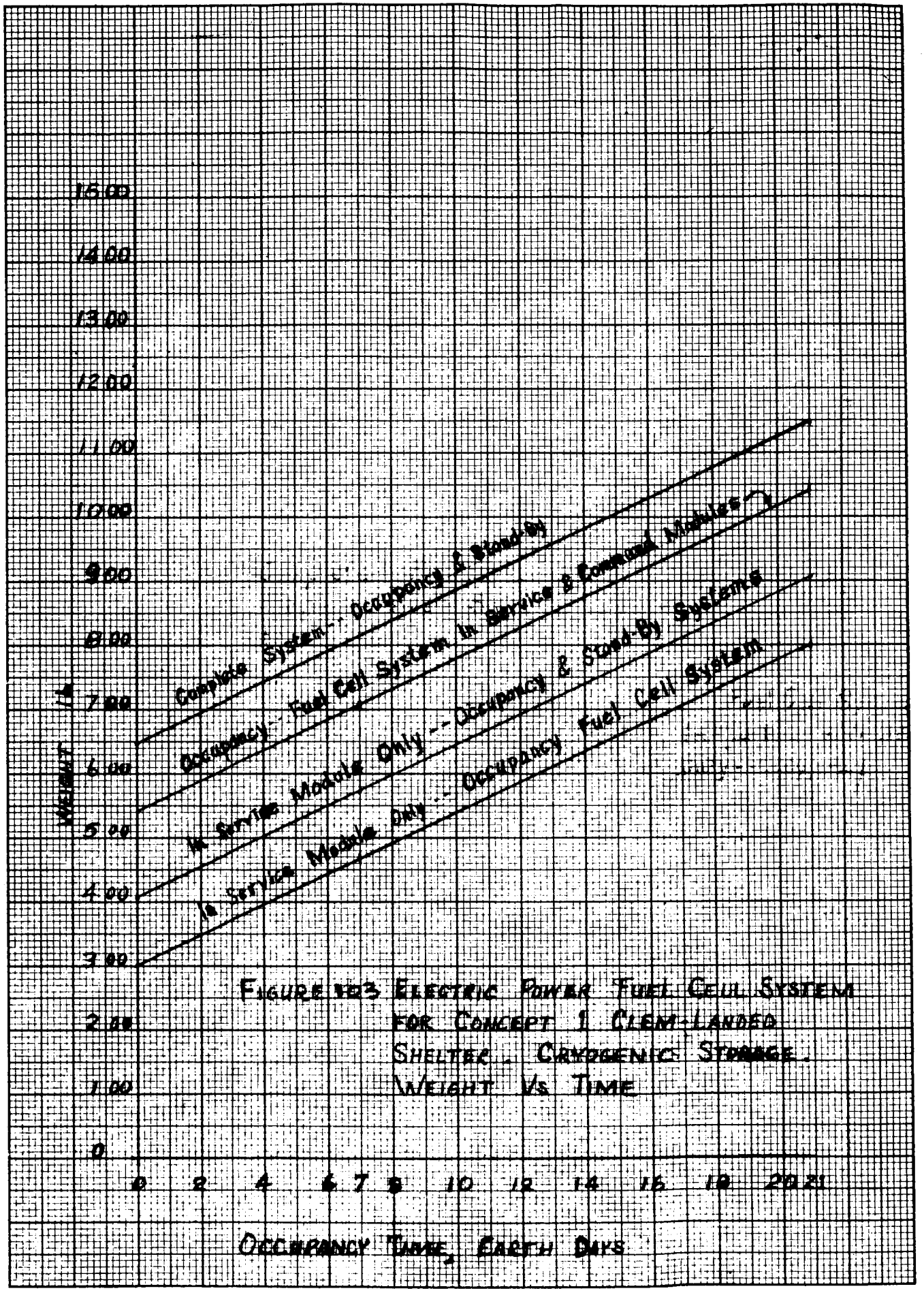


FIGURE 103 ELECTRIC POWER FUEL CELL SYSTEM FOR CONCEPT 1 CLEM-LANDED SHELTER - CRYOGENIC STORAGE - WEIGHT VS TIME

TABLE 56

ELECTRIC POWER DEMAND FOR CONCEPT 1, OCCUPANCY MODE

Reference		Continuous Average, kw
This study	ECL total demand - - - - - - - -	0.555
(1)	Lighting	0.175
(1)	Water Heating	0.100
(1)	Communications and Instrumentation	0.075
	Power Demand	0.905

Since a fuel cell power system is a water supply system as well, water demand must be considered also. It was determined to be as indicated in Table 57.

TABLE 57

WATER DEMAND FOR CONCEPT 1, OCCUPANCY MODE

	Water Demand lb/day
Food preparation	12.0
Personal hygiene	10.0
Back-packs for excursions 4 x 4	16.0
Water Demand	38.0

On the basis that 1 kw/hr of energy delivered by the fuel cell produces 1.0 lb of potable water, the system detailed in Table 58 produces 454 lb of water over a twenty-one day period. Since this amount of water production does not meet the water demand specified in Table 57, it is necessary to store water in the shelter for use during the occupancy mode. Weight of water required as a function of occupancy time is shown in Figure 104; weight of the complete fuel cell power system, including water supply, is also shown as a function of time. It is interesting to note that, if the fuel cell is sized to mmet the water demand, thus exceeding the power demand, a fuel cell system of 1.6 kw capacity is required. Weight of this system is shown in Figure 104 as a function of occupancy time and indicates that, for a twenty-one day occupancy period, the total 1.6 kw system weight would be 200 lb, or 15%, more than the 0.9 kw system plus water, but would have an increased

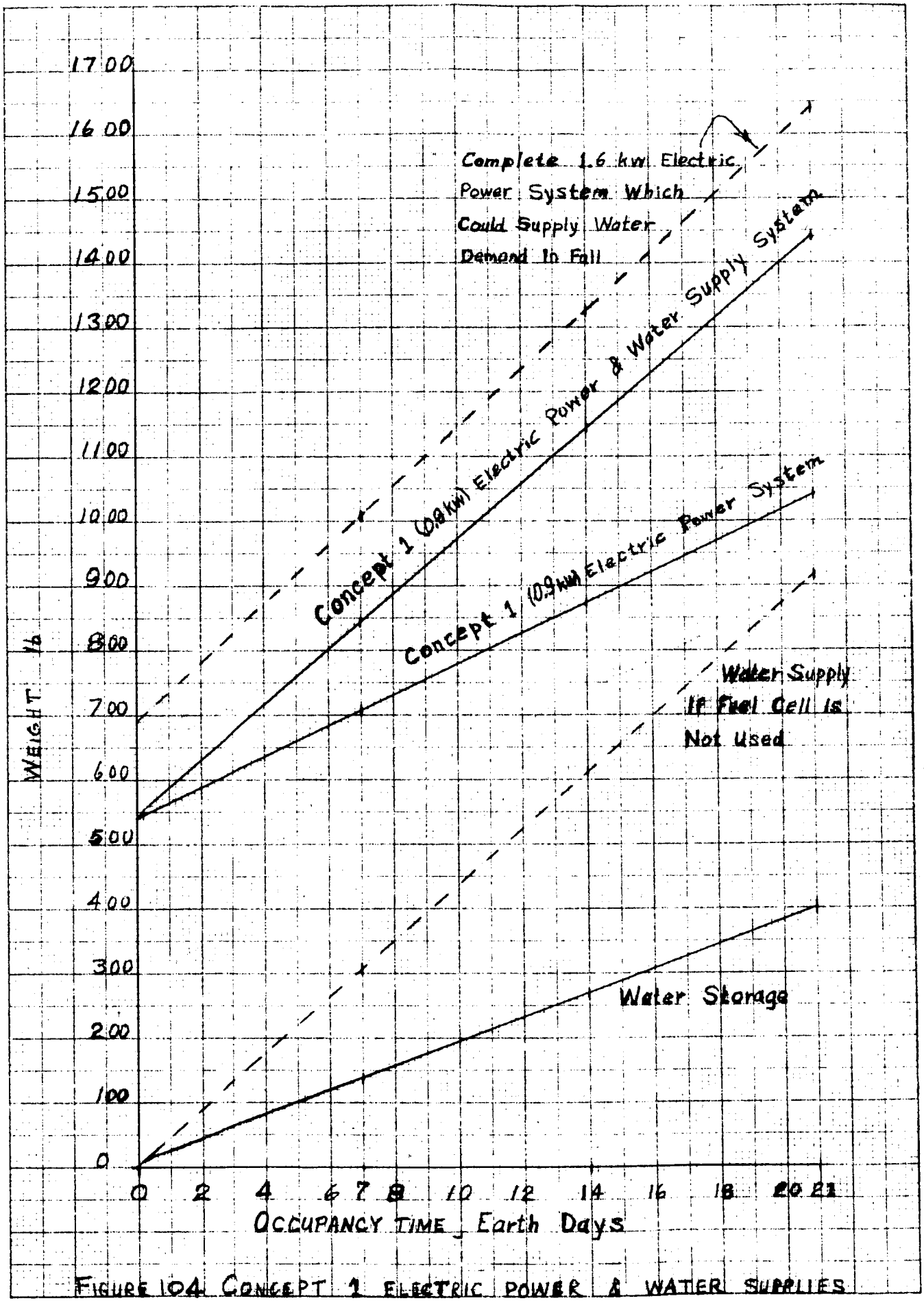


FIGURE 104 CONCEPT 1 ELECTRIC POWER & WATER SUPPLIES

electrical capacity of 80%.

TABLE 58

WEIGHT AND VOLUME OF FUEL CELL POWER SYSTEM COMPONENT PARTS

FOR

CONCEPT 1 - 21-DAY OCCUPANCY

	Dimensions	Weight, lb
In Service Module:		
Fuel cells, 5/3 redundancy per Ref (3)	20 in dia x 44 in	185
Liquid H_2		58
Liquid O_2		460
Reactant tanks Liquid H_2	14.3 ft^3	27
Liquid O_2	7.1 ft^3	13
Reactant tank supports		30
Radiator	5800 in^2	30
Hardware		15
Wiring and power distribution		70
TOTAL		888
In Command Module:		
Batteries	512 in^3	40
Chargers	150 in^3	4
Inverters	13 in x 6.5 in x 8.5 in	20
Distribution & Control	8.5 in	60
Wiring and power distribution		20
Hardware		10
Relays		65
GSE wiring		22
TOTAL		241
TOTAL WEIGHT		1129

Concept 2 Power Systems

The electrical power systems requirements, capacity, weight, and volume for the standby and occupancy modes of Concept 2 (SM-shelter/logistics modules) are described in the following sections.

Power Demand

Power demand can be treated most effectively by separating it into its primary modes; landing, stand-by, and occupancy. Demand in each of its modes was then determined from SID 62-1473 and tabulated.

TABLE 59

POWER DEMAND, 2

a) Landing Mode	Continuous Ave., kw
Propulsion AC	.420
Guidance and navigation AC	.500
Stabilization and control AC	.138
DC	.050
Telecommunication AC	.404
DC	.115
Auxiliary Power AC	.110
Total Watts AC	1.572
Including inverter AC x 1.2	1.886
Total Watts DC	.165
Total Demand	2.051

b) Stand-by Mode

Average continuous demand 15 w, with peaks to 40 w.
Time period: 60 to 90 days

c) Occupancy Mode

ECS total demand	.569
Lighting	.175
Water heating	.100
Communications and instrumentation	.075
	.919

Stand-by Mode

Since the desired stand-by time period is in the order of 60 to 90 days, a primary battery cannot be used. For example, such a battery would weigh 200 lb. In the study reported in SID 62-1473, a nickel-cadmium battery was selected and shall be used in this unit. Apart from the battery, the unit is similar to that described for use with the Concept 1 shelter. Weights and volumes of component parts of the stand-by power system are given in Table 60 and apply to Concepts 2, 3, and the extended mission LOP.

TABLE 60

STAND-BY ISOTOPIC-THERMOELECTRIC GENERATOR WITH BATTERIES

FOR

CONCEPTS 2, 3, AND LOP SHELTERS

Component Parts	Dimensions	Wt., lb
Cylindrical isotope assembly Snap 9A (mod)	8.5 in dia x 11 in	22.
Cylindrical isotope assembly supports		3.
Battery	10 in x 5 in x 7 in	30.
Battery supports		3.
Radiator controls and reflector assemblies		3.
Charger		3.
Structure interior with insulation		2.8
Structure external insulation		16.8
Dielectric heat transfer fluid	4.25 gal.	39.6
TOTAL SYSTEM	12 in x 30 in x 12 in	123.2

Occupancy Mode

Using the weight/occupancy time/fuel cell size parametric curves of Figures 100 and 101, weight of a 2.1 kw fuel cell, sized to meet the landing mode power requirement, was established and plotted as a function of occupancy time in Figure 105. The parametric data presented in Figure 105 accounts for the effect of cryogenic fuel boil-off in complete system weight for stand-by periods of 60 and 90 days. Fuel cell system component weights and dimensions are shown in Table 62.

Since the production of makeup water by the fuel cell must be considered also, the total water demand for various occupancy periods is shown in Table 61.

TABLE 61 - WATER DEMAND

Number of Days	1	30	60
Four back-packs/day @ 4 lb ea	16	480.	960.
O_2 generation; make-up water	12.5	375.	750.
Leakage of atmosphere: make-up water	0.1	3.	6.
TOTAL, LB	28.6	858.	1716.

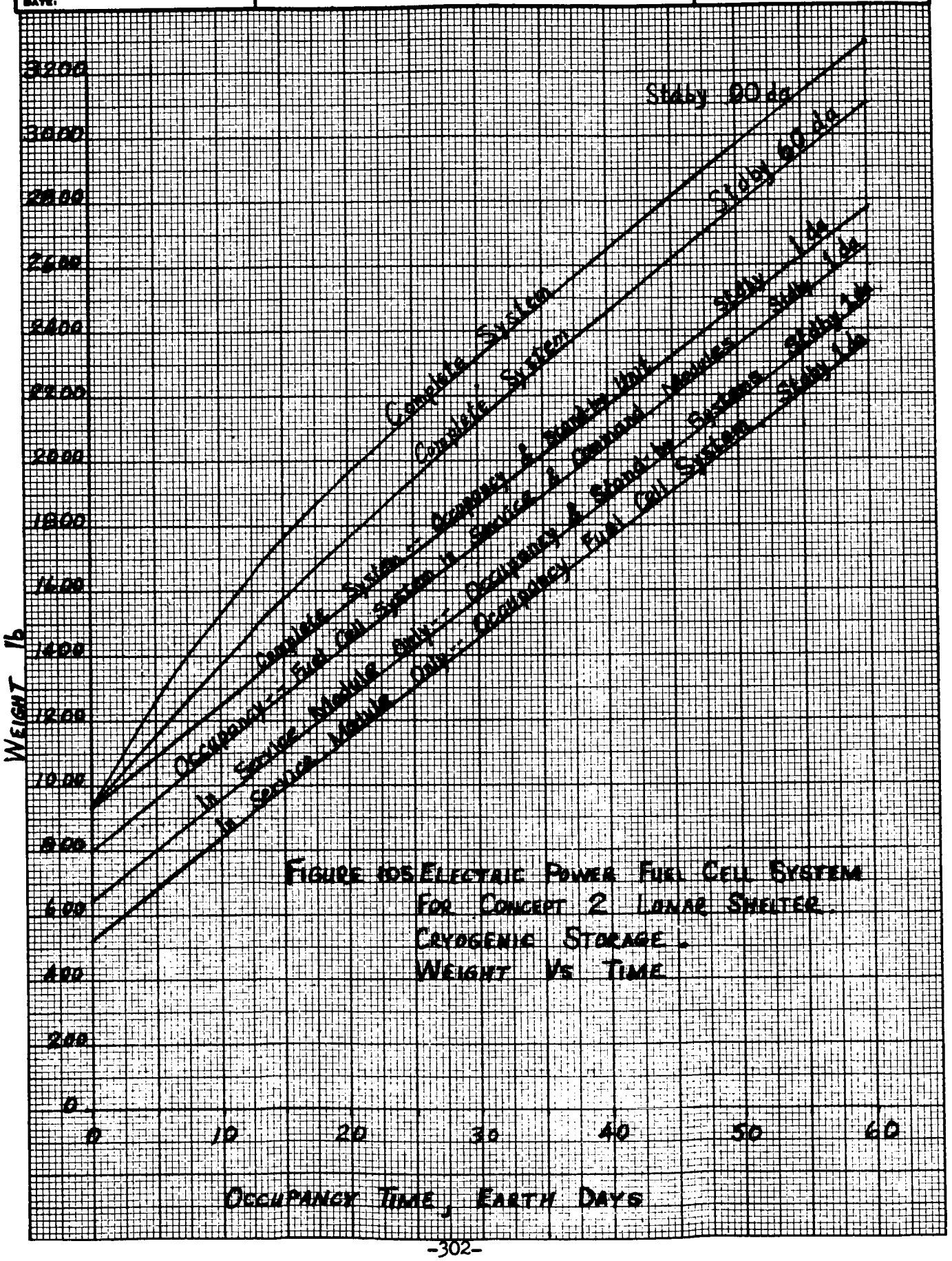


FIGURE 105 ELECTRIC POWER FUEL CELL SYSTEM FOR CONCEPT 2 LUNAR SHELTER - CRYOGENIC STORAGE - WEIGHT VS TIME

TABLE 62

2.1 KW FUEL CELL ELECTRIC POWER SYSTEM, COMPONENTS DIMENSIONS, AND WEIGHTS

	Dimensions	Wt. lb	Occupancy Time			
			30 days		60 days	
			Dim	Wt #	Dim	Wt #
In Service Module:						
2.1 kw fuel cells, 5/3 redundancy	24" dia x 55"	400.				
Reactants LH_2 avail.				90.		180.
" LO_2				720.		1440.
Reactant Tanks LH_2			28 ft^3	55.	56 ft^3	110.
" " LO_2			14 ft^3	25.	28 ft^3	50.
Reactant tank supports				40.		60.
Radiator	5800 in^2 sur.	36.				
Hardware		15.				
Wiring & power dist.		70.				
Isotopic-thermoelec.-battery stand-by unit		125.				
Subtotal wt in SM, uncorr		(646)		1576		2486
Subtotal corr for cryo loss 60 day standby				1941		2806
Subtotal corr for cryo loss 90 day standby				2026		2996
In Command Module:						
Batteries	1024 in^3	80.				
Chargers	150 in^3	6.				
Inverters	15" x 6.5" x 9"	25.				
Dist. & Control		60.				
Wiring		20.				
Hardware		10.				
Relays		65.				
GSE wiring		22.				
Subtotal Weight in CM		(288)		288		288
Total Sys. Wt uncorr for cryo loss				1864		2774
Total corr for cryo loss 60 day standby				2229		3094
Total corr for cryo loss 90 day standby				2314		3284

At the occupancy mode power demand level of 0.9 kw, the fuel cell system cannot fully supply the required water and water storage which is required as shown as a function of time in Figure 106; the weight of the complete 2.1 kw fuel cell system and the weight of the 2.1 kw fuel cell system plus water storage is also shown as a function of time in Figure 106. It is interesting to note that, if the occupancy mode power demand were increased from 0.9 to 1.2 kw, no water storage would be necessary and a weight savings of 460 lb would be realized for an occupancy period of 60 days.

A PM-147 dynamic electrical power system was also considered for the occupancy mode power source. This system is discussed in detail in Part II, Systems Design and Analysis. The weight of a 2.1 kw PM-147 dynamics system is instant as shown in Figure 106 and does not vary with occupancy time. However, the system produces no water, therefore, all required make-up water must be stored. Weight of water supply required with the PM-147 system and weight of water plus weight of PM-147 system are shown plotted as a function of occupancy time in Figure 106. The weight of this system and required water is less than the weight of the 2.1 kw fuel cell system and required water with only a 0.9 kw loss on the fuel cell; however, the fuel cell system is lighter if the demand in increased to 1.2 kw during occupancy.

Concept 3 and Extended LOP Power Systems

The electrical power systems requirements, capacity, weight, and volume for the stand-by and occupancy modes for Concept 2 (LLV shelter) and the extended-mission LOP are described in the following sections.

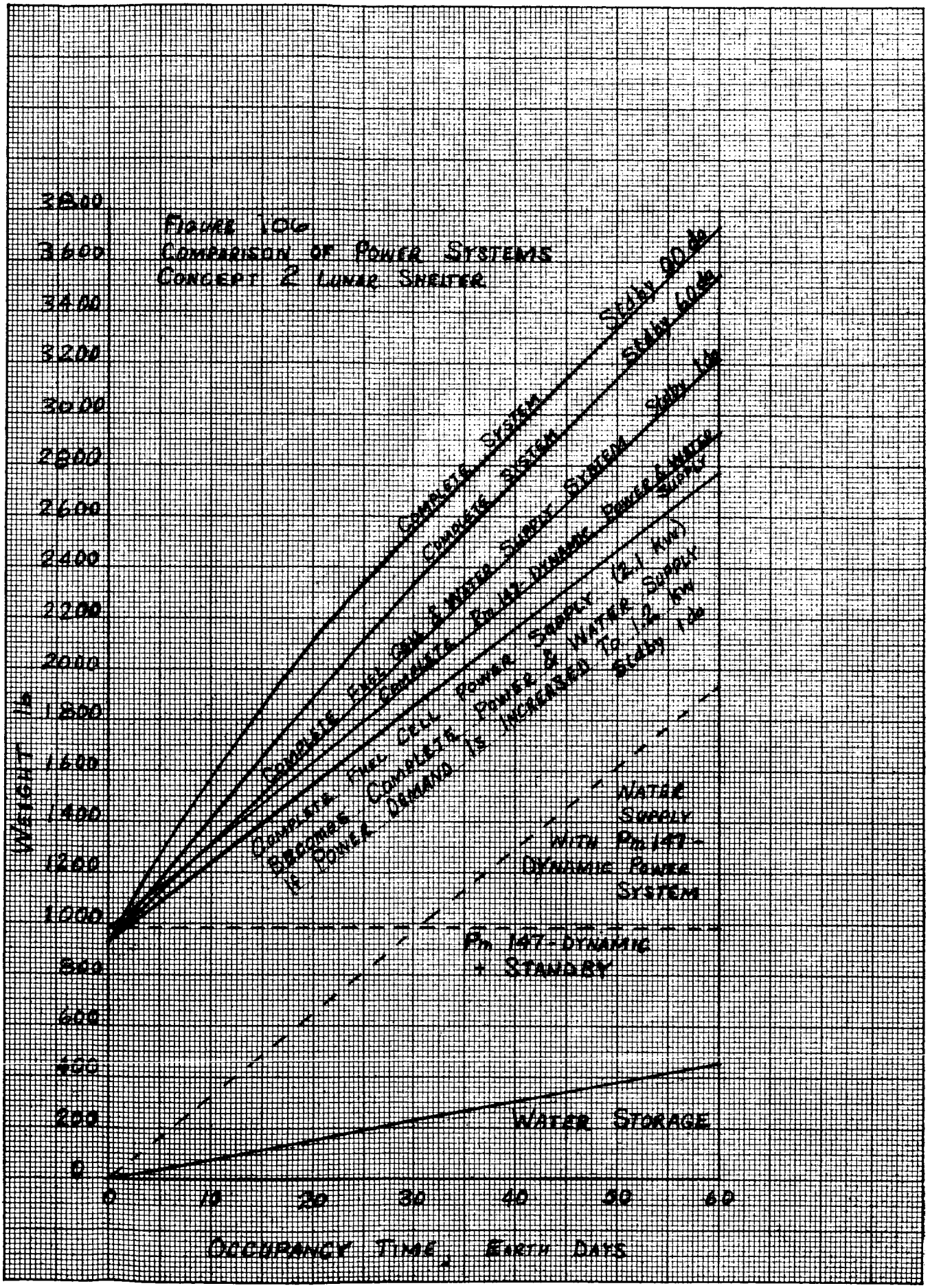
Power Demand

Power demand for the landing and stand-by modes is the same as that presented in Table 59 for Concept 2. The occupancy mode power demand is presented in Table 63 for Concept 3 and the extended mission LOP.

TABLE 63

POWER DEMAND

Number of Men	2	3	4	5
ECS	2.375	2.915	3.501	4.133
Lighting	.175	.175	.175	.175
Water Heating	.100	.100	.100	.100
Comm. & Instrumentation	.075	.100	.100	.100
TOTAL, KW	2.725	3.290	3.875	4.508


FIGURE 106
COMPARISON OF POWER SYSTEMS
CONCEPT 2. LUNAR SHELTER
3800
3600
3400
3200
3000
2800
2600
2400
2200
2000
1800
1600
1400
1200
1000
800
600
400
200
0
WEIGHT - LBS
0
10
20
30
40
50
60
OCCUPANCY TIME - EARTH DAYS
COMPLETE SYSTEM
STAND BY 90 DAYS
COMPLETE SYSTEM
STAND BY 60 DAYS
COMPLETE SYSTEM
STAND BY 30 DAYS
COMPLETE FUEL CELL & WATER SUPPLY SYSTEM
COMPLETE Pm 147 DYNAMIC POWER SUPPLY SYSTEM
POWER & WATER SUPPLY (2.1 KWH) FOR STAND BY
POWER INCREASED TO 1.2 KW
COMPLETE FUEL CELL POWER & WATER SUPPLY
COMPLETE FUEL CELL POWER DEMAND
WATER SUPPLY WITH Pm 147 - DYNAMIC POWER SYSTEM
Pm 147 - DYNAMIC + STANDBY
WATER STORAGE

[redacted]

Stand-by Mode

Since the stand-by power demand is the same as that for Concept 2, the stand-by power system for Concept 3 and the extended LOP will be the same PM-147 unit described in Table 60. Although the stand-by period is one year, the power output of the isotopic PM-147 thermoelectric system will not be affected appreciably because it has a half life of 2.6 years.

Occupancy Mode

Using the weight/occupancy time/fuel cell capacity parametric plots of Figures 100 and 101, weights of fuel cell power systems, sized to meet the power demand of a two-, three-, or four-man occupancy crew were determined as a function of occupancy time and plotted in Figure 107. System weights for one day standby and one year standby, considering cryogenic boil-off, are shown as a function of occupancy time. Storage of makeup water is not required for these shelter concepts since even the lowest power demand (for a two-man crew) requires a fuel cell which produces 64 lb of water per day. The actual water demand is shown in Table 64.

TABLE 64

WATER DEMAND FOR CONCEPT 3 AND LOP

For excursions, 6 back-packs @ 4 lb ea. -	24 lb/day
Oxygen generation: Make-up water -------	14.85
Atmosphere make-up water----------	0.15
TOTAL	39.0 lb/day

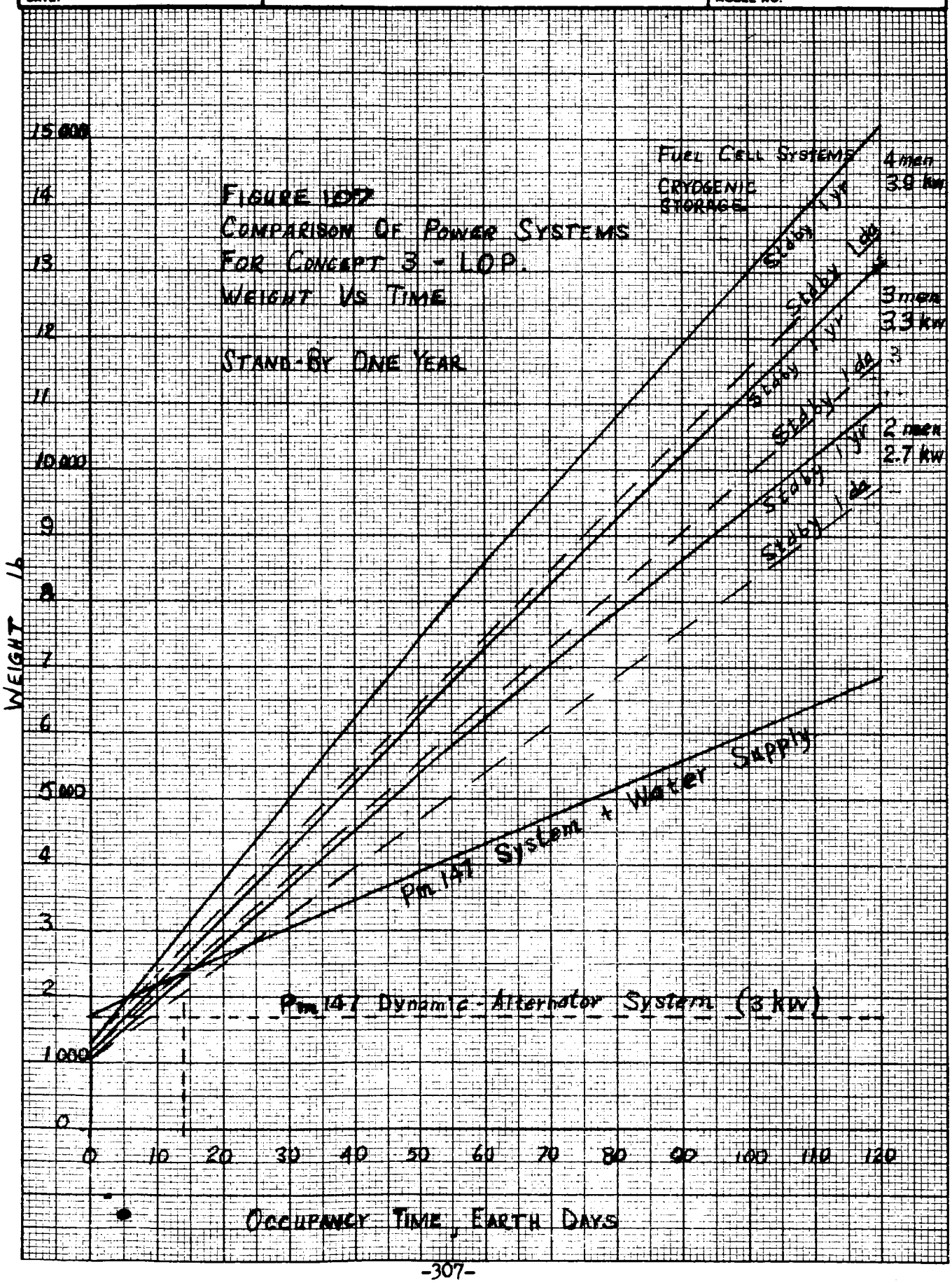
The weight, constant with occupancy time, of a 3 kw PM-147 dynamic alternator system is also shown in Figure 107. Since this system produces no water, weight of the PM-147 system and the required make-up water is shown as a function of standby time. Comparison of the curves of fuel cell and PM-147 dynamic systems weights indicates clearly that the PM-147 system is lighter than the comparable fuel cell system for periods of occupancy time greater than fourteen days. For a 120-day occupancy time, after one year standby, the PM-147 dynamic system is approximately 510 lb lighter.

The PM-147 dynamics-alternator electrical power system is recommended for use as the occupancy mode power supply for Concepts 3 and the extended LOP. However, reliability, cost, availability, and safety, besides weight and volume, are important considerations which should not be overlooked in the selection of an electrical power supply system.

Cryogenic Storage of Hydrogen and Oxygen

Throughout all considerations to this point, cryogenic storage has been assumed. Other fuel storage systems are substantially heavier and, there-

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fore, were not considered.

The fuel storage problem is limiting the use of the fuel cell power system for missions where extended lunar storage is required. The hydrogen oxygen fuel has to be loaded and stored in liquid form requiring storage tanks with superinsulation. The thickness of the insulation system is considerable to limit the boil-off to a reasonable level. The boil-off propellant losses are considerably more than the useful propellant left after a few months storage time.

Fig. 108 shows as a function time how many times more the hydrogen storage system weighs than the stored hydrogen. It was assumed that the average temperature of the external surface of the tank insulation is 80°F and the lunar days and nights are equally long. It was also figured that the insulation thickness is a function of storage time, and an optimum boil-away and insulation weight was considered for each storage time period.

Similar curves, Fig. 109, were drawn for the total fuel system weight as a function of available fuel weight and storage time. To the data of Fig. 108, which relate to hydrogen losses, were added oxygen losses, which had been estimated from consideration of oxygen losses relative to hydrogen losses. The method is described in detail below.

Assume that fuel losses are proportional to the amounts evaporated, which, in turn, are proportional to heat transferred into the tanks through their walls.

$$W_l = C \frac{q}{\Delta H_v} \qquad C, \text{ constant}$$

$$q = KS\Delta T \qquad K, \text{ constant}$$

where W_l = Losses, by weight

q = Heat transferred per unit time

ΔH_v = Heat of vaporization

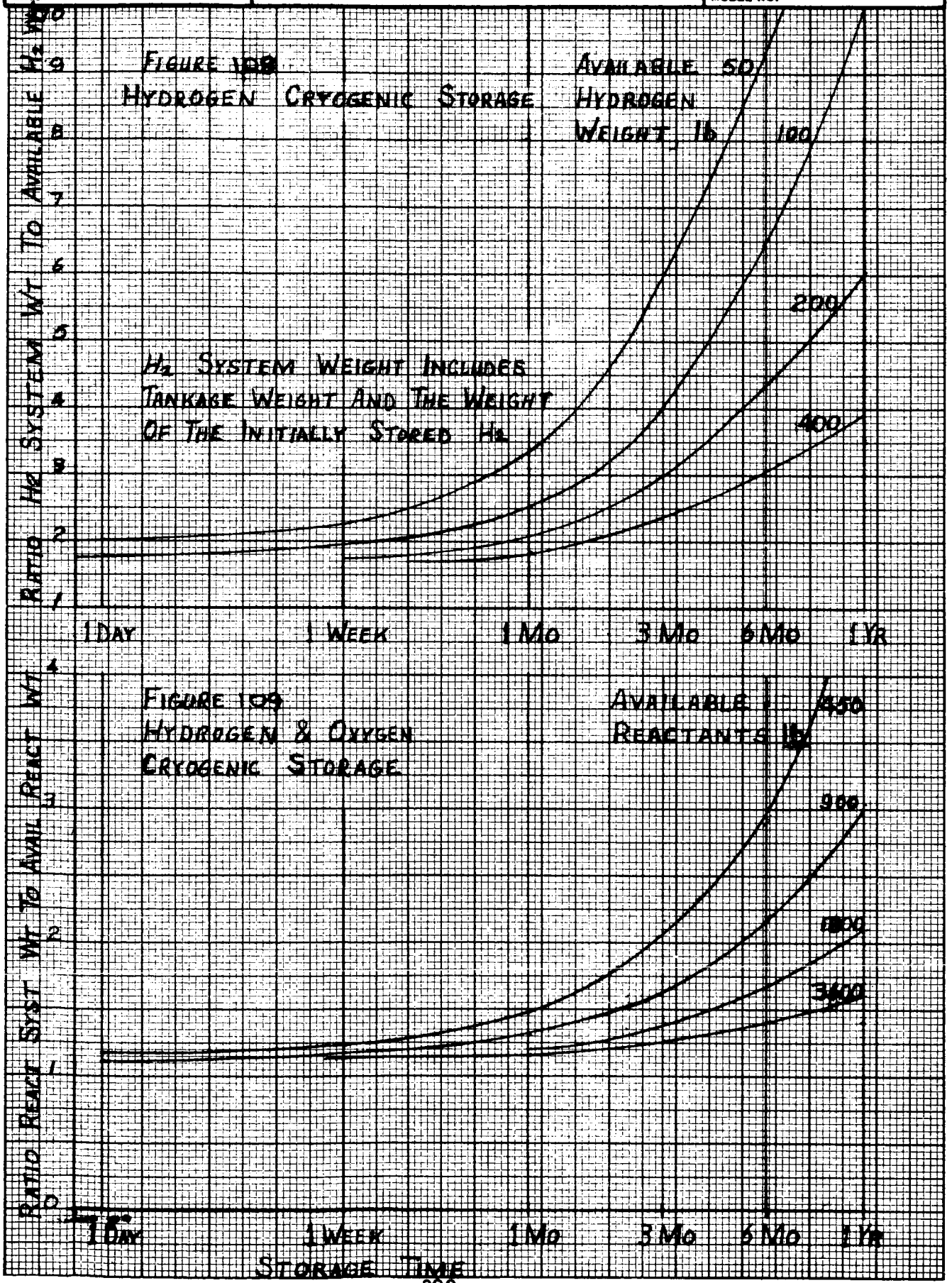
S = Surface area

ΔT = Temperature drop across tank wall

These equations may be grouped into an expression of ratios:

$$\frac{W_l\ (O_2)}{W_l\ (H_2)} = \frac{S(O_2)}{S(H_2)} \quad \frac{\Delta T(O_2)}{\Delta T(H_2)} \quad \frac{\Delta H_v(H_2)}{\Delta_{\ v}(O_2)}$$

Surface area may be minimized by choosing the tank configuration to be spherical. This minimizes losses.


Ratio H2 System Wt To Available H2 Wt
1
2
3
4
5
6
7
8
9
10
1 Day
1 Week
1 Mo
3 Mo
6 Mo
1 Yr
Storage Time
Available H2 Weight, lb
50
100
200
400
H2 System Weight Includes Tankage Weight And The Weight Of The Initially Stored H2
Ratio React Syst Wt To Avail React Wt
1
2
3
4
1 Day
1 Week
1 Mo
3 Mo
6 Mo
1 Yr
Storage Time
Available Reactants lb
450
900
1800
3600
Figure 108
Hydrogen Cryogenic Storage
Figure 109
Hydrogen & Oxygen
Cryogenic Storage

[redacted]

For spheres, $$\frac{S(O_2)}{S(H_2)} = \frac{V(O_2)}{V(H_2)}^{2/3}$$

The full system requires 8 lb O_2 per 1 lb H_2, therefore,

$$\frac{V(O_2)}{V(H_2)} = \frac{8}{71.2} \cdot \frac{4.44}{1} = 0.50$$

$$\frac{S(O_2)}{S(H_2)} = .5^{2/3} = 0.625$$

ΔT is established by assuming the ambient temperature = 300°K, and the temperature of liquid to be that of the normal boiling point.

$$\frac{\Delta T(O_2)}{\Delta T(H_2)} = \frac{300 - 20}{300 - 90} = 0.75$$

ΔH_v ratio is obtained directly.

$$\frac{\Delta H_v(H_2)}{\Delta H_v(O_2)} = \frac{192}{91.7} = 2.09$$

Substitution gives

$$\frac{W_1(O_2)}{W_1(H_2)} = 0.625 \times 0.75 - 2.09 = 1.03$$

Thus it is seen that evaporation weight losses due to heat transfer through the container wall is essentially the same for both oxygen and hydrogen.

It is interesting to note that it is more economical to store large amounts of fuel for extended periods of time because the heat leak is a function partially of the tank surface and the ratio of volume to surface decreases with increased amount of fuel.

There is also a heat leak through the tank fittings and mountings which is assumed to be 5 BTU per hour for any tank size or storage time. This heat leak affects the storage system weight as a function of available fuel weight and storage time. See Figure 109.

Where the ratio of fuel system weight to available fuel weight exceeds 2, the use of storable propellants should be considered. A water solution of hydrazene could be used in fuel cell systems under development by Allis Chalmers, Texas Instruments, and others. These fuel cell systems also have the other advantage of producing additional water compared with the pure Hydrogen-Oxygen systems. This may make them attractive for certain missions.

WEIGHTS

Major system weights and total weight estimates of the three lunar CM-shelter concepts and the extended mission LOP are presented in this section.

Concept 1, CLEM-Shelter

System weights and total lunar landed weight of the Concept 1 shelter are presented in Table 65 for various mission duration periods of one week standby in combination with occupancy periods of one, two, and three weeks for two crewmen. Since the payload capability of the Grumman proposed CLEM is only 7000 lb, it is apparent that the maximum mission duration of Concept 1 is one week standby and slightly more than three weeks occupancy.

Concept 2, SM-Shelter/Logistics Module

Systems weights and total lunar-landed weight of Concept 2 shelter module are presented in Table 66 for various mission duration periods of zero to 90 days standby in combination with 30 to 90 days occupancy time for two crewmen. The table indicates that, with an allowable lunar-landed weight of 14,854 lb, corresponding to an injection weight of 67,150 lb, the shelter standby and occupancy periods are 90 and 90 days respectively. If the maximum injection weight capability of 70,300 lb (full SM propellant tanks) were fully utilized, occupancy time would be extended about ten days.

A weight estimate for a shelter/logistics module utilizing an Apollo SM, lengthened 44-inches to increase the propulsive capability and take advatnage of the full 90,000 lb injection capabiliyt of the S-V, is down in Table 67.

As indicated by the table, the additional propulsion capability of this version of Concept 2 permits the SM to land on the lunar surface 6116 lb of cargo along with the shelter having a capability of 90 days standby and 90 days occupancy.

Concept 3, LLV-Shelter

System weights and total lunar-landed weight of the Concept 3 shelter are presented in Table 68 for a standby time of one year and an occupancy time of 120 days for various crew sizes of two, three, and five men. The present payload capability of the LLV is 27,500 based on an S-V injection capability of 90,000 lb. Therefore, it is evident that uprating of the S-V will be required to land Concept 3 shelters which will support more than three men for the above mission duration.

Extended Mission LOP

Systems weights and total lunar-landed weight of the extended mission LOP are presented in Table 69 for stand-by time of one year and an occupancy time of 120 days for crew sizes of two and three men. Since the LOP is also landed by the LLV and has a weight limitation of 27,500 lb, it is evident

that uprating of the S-V will be required for an LOP which will support more than three men for the above mission duration.

[redacted]

TABLE 65

TWO-MAN SHELTER LANDED WITH LEM-TRUCK (CLEM) WEIGHT ESTIMATE

COMMAND MODULE	STANDBY OCCUPANCY 1 WEEK	1 WEEK 2 WEEKS	1 WEEK 3 WEEKS
Structure	1797	1797	1797
Crew Systems	130	130	130
Telecommunication	488	488	488
Controls & Displays	127	127	127
Electric Power	240	240	240
WEIGHT EMPTY	2782	2782	2782
EXPENDABLES			
Life Support	303	309	312
Food	24	48	96
GROSS WEIGHT	3109	3139	3190
SERVICE MODULE			
Structure	1207	1207	1207
Electronic Subsystem	150	150	150
Electric Power			
Fuel Cell	305	305	305
Standby	106	106	106
Environmental Control System	301	604	884
WEIGHT EMPTY	5178	5511	5842
EXPENDABLES			
Fuel Cell Reactants*	165	330	495
Water Storage*	140	270	400
Environmental Control System	28	28	28
GROSS WEIGHT	5511	6139	6765

*Fuel cell water production plus water storage supplies all make up water for ECLSS

[redacted]

TABLE 66

SHELTER LANDED WITH APOLLO SM (CONCEPT 2), WEIGHT ESTIMATE

DAYS STANDBY	0	60	60	90	90	90
DAYS OCCUPANCY	60	30	60	30	60	90
COMMAND MODULE						
Structure	2000	2085	2167	2252	2252	2337
Crew Systems	130	130	130	130	130	130
Telecommunications	488	488	488	488	488	488
Controls & Displays	127	127	127	127	127	127
Electrical Power	288	288	288	288	288	288
Stabilization & Control	200	200	200	200	200	200
CM EMPTY WEIGHT	3233	3318	3400	3400	3485	3570
Expendables						
Life Support	285	259	285	259	285	315
Food	205	102	205	102	205	307
CM GROSS WEIGHT	3273	3679	3890	3761	3975	4192
CM-SM						
Structure	1500	1585	1664	1664	1745	1826
Electronics	150	150	150	150	150	150
Electric Power-Fuel Cell System	743	643	743	643	743	843
-Isotope System	123	123	123	123	123	123
Environmental Control	1972	924	1897	949	1947	2946
CM + CM-SM EMPTY WEIGHT	8211	7104	8467	7290	8683	10080
Expendables						
Fuel Cell Reactants*	1620	1175	1940	1260	2130	2700
Makeup Water & Tanks*	450	230	450	230	450	649
ECS Supplies	-	325	325	425	425	425
CM, CM-SM GROSS WEIGHT	10281	8834	11182	9205	11688	14854

*Fuel cell water production plus water storage supplies all required ECLSS makeup water

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TABLE 66 (CONTINUED)

DAYS STANDBY	0	60	60	90	90	90
DAYS OCCUPANCY	60	30	60	30	60	90
APOLLO SM						
Structure	(4400)	(4400)	(4400)	(4400)	(4400)	(4400)
Apollo I	2290	2290	2290	2290	2290	2290
Landing Gear (4-legged)	2110	2110	2110	2110	2110	2110
Electronics	27	27	27	27	27	27
Reaction Control	590	590	590	590	590	590
Remote Guidance	172	172	172	172	172	172
Main Propulsion System	3007	3007	3007	3007	3007	3007
PAYLOAD + APOLLO SM WEIGHT EMPTY	18477	17030	19378	17401	19884	23050
Expendables						
Reaction Control	419	419	419	419	419	419
Propulsion	681	681	681	681	681	681
Main Propellant	35600	33000	37200	33700	38200	44000
GROSS WEIGHT	55177	51130	57678	52201	59184	67150

[redacted]

TABLE 67

SHELTER LANDED WITH EXTENDED LENGTH APOLLO SM (CONCEPT 2) WEIGHT ESTIMATE

Payload	(20,970)
Shelter - 90 day standby 90 day occupancy	14,854
Cargo	6,116
Structure	2,930
Gear	2,800
Electronics	28
Reaction Control	590
Remote Guidance	172
Main Propulsion System	3,290
END BOOST	30,780
SM Expendables	
Reaction Control	420
Propulsion	700
Main Propellant	58,100
GROSS WEIGHT	90,000

[redacted]

TABLE 68

SHELTER LANDED WITH LLV (CONCEPT 3) WEIGHT ESTIMATE

One-Year Standby - 120 Day Occupancy

	2 Men	3 Men	5 Men
Structure	9,923	9,923	9,923
Crew Systems	130	160	220
Telecommunication	488	488	488
Controls & Displays	127	127	127
Electrical Power			
Standby for one year	123	123	123
PM147 Dynamic Alternator System	1,130	1,450	2,200
Environmental Control System A	1,618	1,674	1,786
Weight Empty:	13,539	13,945	14,867
Expendables			
Makeup Water & Tanks	3,712	4,985	7,684
Environmental Control System/ Life Support	6,590	8,091	11,094
GROSS WEIGHT	23,841	27,021	33,645

[redacted]

TABLE 69

EXTENDED MISSION LOP WEIGHT ESTIMATE

One-Year Standby - 120-Day Occupancy

	2 Men	3 Men
Structure	7,810	7,810
Crew Systems	575	720
Communications	531	531
Electric Power System (PM147 Cynamic Alternator)	1,253	1,573
Environmental Control System	1,618	1,674
WEIGHT EMPTY	11,787	12,308
Expendables		
Makeup Water & Tanks	3,712	4,985
ECLSS	6,590	8,091
GROSS WEIGHT	22,089	25,384

[redacted]

PART IV - CRYOGENIC TANK INSULATION TEST

INTRODUCTION

The second generation lunar exploration system studies have indicated the need for cryogenic propulsion (LOX/LH_2). Attendant with this need is the long-term storage of cryogenic fluids in space and on the lunar surface. Propellant losses during these mission phases have to be kept to an absolute minimum. The thermal insulation and system integration of the propellant tanks must also provide thermal protection during the pre-launch and boost phases; in other words, it must afford protection in both atmospheric and vacuum environment.

In order to achieve this goal, design concepts must be developed to demonstrate the practical application and integration of structural support, propulsion system requirements, and superinsulation application techniques for atmospheric and space thermal environments.

The type of thermal insulation best suited for vacuum application is generally known as superinsulation. Superinsulations can be composite or integral materials. Two such superinsulations are manufactured under the names of "Linde" and "NRC-2." Linde superinsulation is a light-weight composite insulation material consisting of alternate layers of fiberglass cloth and aluminum foil manufactured by Linde, a Division of Union Carbide Corporation. National Research Corporation's NRC-2 is an integral insulation material consisting of multiple layers of 1/4 mil mylar, each layer coated on one side with vapor deposited aluminum. The present design concept incorporated NRC-2 superinsulation.

OBJECTIVES

The objectives of the cryogenic tank test program are to evaluate the performance of NRC-2 superinsulation applied to a practical tank configuration in both atmospheric and vacuum environment to demonstrate insulation application and structural integration techniques and develop design capability and fabrication techniuques. The complete test program consists of three phases, using liquid nitrogen and hydrogen as cryogenic test propellants.

PHASE I

The objective of Phase I is to demonstrate and obtain test data on system performance at altitude (10^{-5} Torr) using NRC-2 vacuum superinsulation. L_2 cryogenic will be used as the propellant since handling of LH_2 is more hazardous.

The liquid nitrogen test will demonstrate the over-all performance of the integrated structural support, propellant system, and application of

NRC-2 superinsulation. Boil-off of LN_2 will be used as a measure of the system performance.

PHASE II

The objective of Phase II is to design and test the application of a vacuum seal bag for ground hold (atmospheric storage) applications of LN_2.

Degradation of system performance as part of an integrated spacecraft design, due to improper thermal design for atmospheric conditions can result in excess boil-off losses for long-term storage in the vacuum of the space environment. The improvement of an integrated ground hold vacuum insulation techniques will result in reduction of propellant weight and boil-off during ground handling and flight time profiles.

PHASE III

Phase III is to demonstrate and obtain test data relating to long-term storage of liquid hydrogen.

DESIGN

The Phase I design concept is based on the integration of a half-scale propellant tank, suspended and supported in a structural sub-assembly with wires, fitted with concentric fill, feed, and vent lines, and insulated with NRC-2 superinsulation.

The Phase II design based on evaluation of the Phase I design, will incorporate a vacuum bag integrated with the superinsulation and support wires to seal off ambient atmosphere (14.7 psia) and humidity, thus minimizing LN_2 boil-off in simulated ground hold.

The Phase III design concept will be the same as the Phase II concept, incorporating any modifications necessary or desirable after evaluation of the Phase II test.

PHASE I - DESIGN - HIGH ALTITUDE STORAGE

The long-term storage of cryogens is a function of spacecraft design, vacuum performance of structural materials, and radiative and conductive heat path leaks. The integration of propellant lines, support structure, and installation of superinsulation materials must be compatible with these near-space requirements. This design is based on an integral propulsion manifold concept. The tank assembly shown in Figure 110 illustrates the system integration of the concentric propellant manifold and support wire structure; 321 stainless steel was selected for the structure because of its material properties at cryogenic temperatures. The design details of the concentric fill, drain, and vent manifold are shown in Figure 111. This particular propulsion system design has the advantage of one tank boss, which reduces the number of penetrations of the superinsulation materials.

The support wire structure tabs, Figure 112 located at eight equally

FIGURE 110

TANK ASSEMBLY AND SUPPORT STRUCTURE

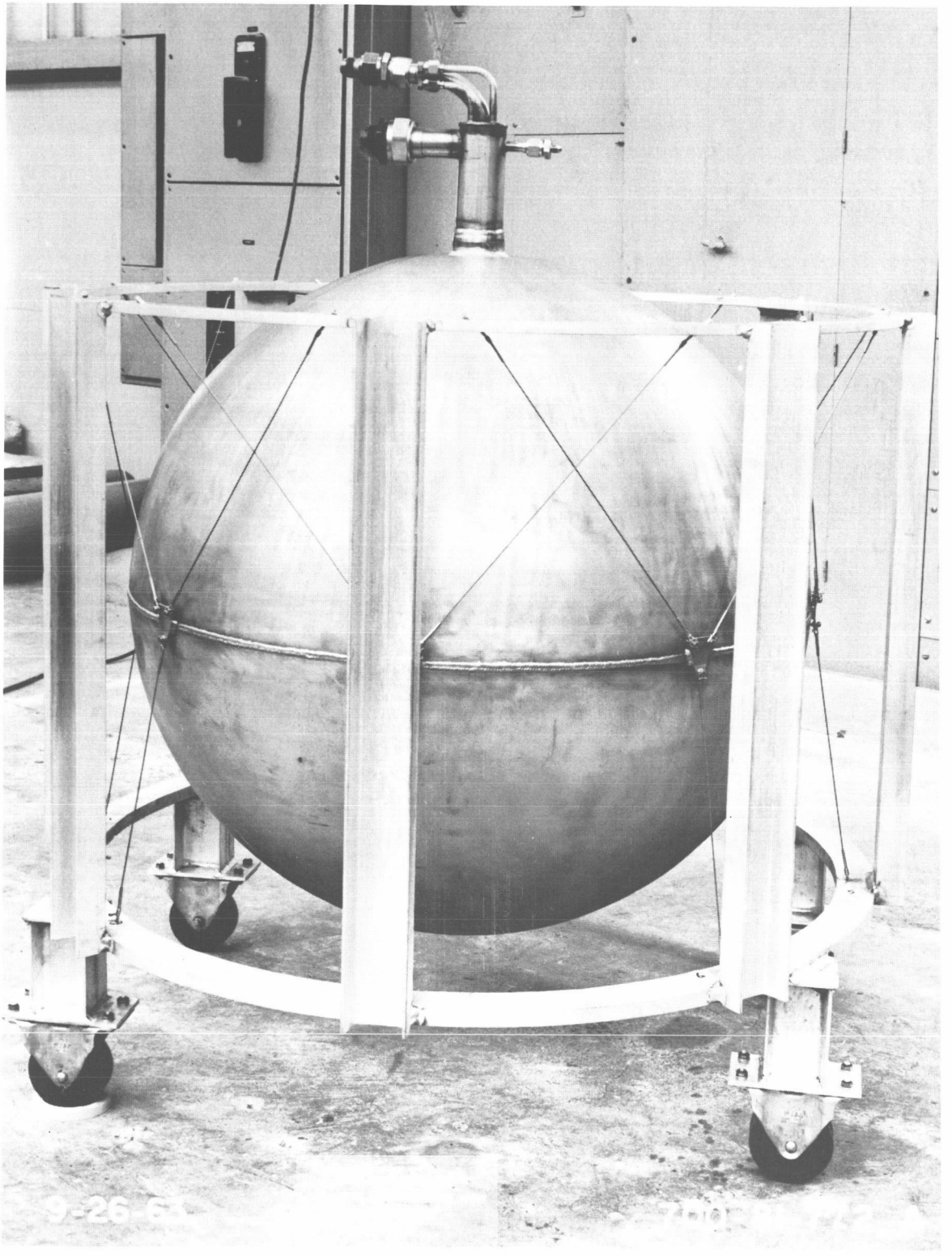
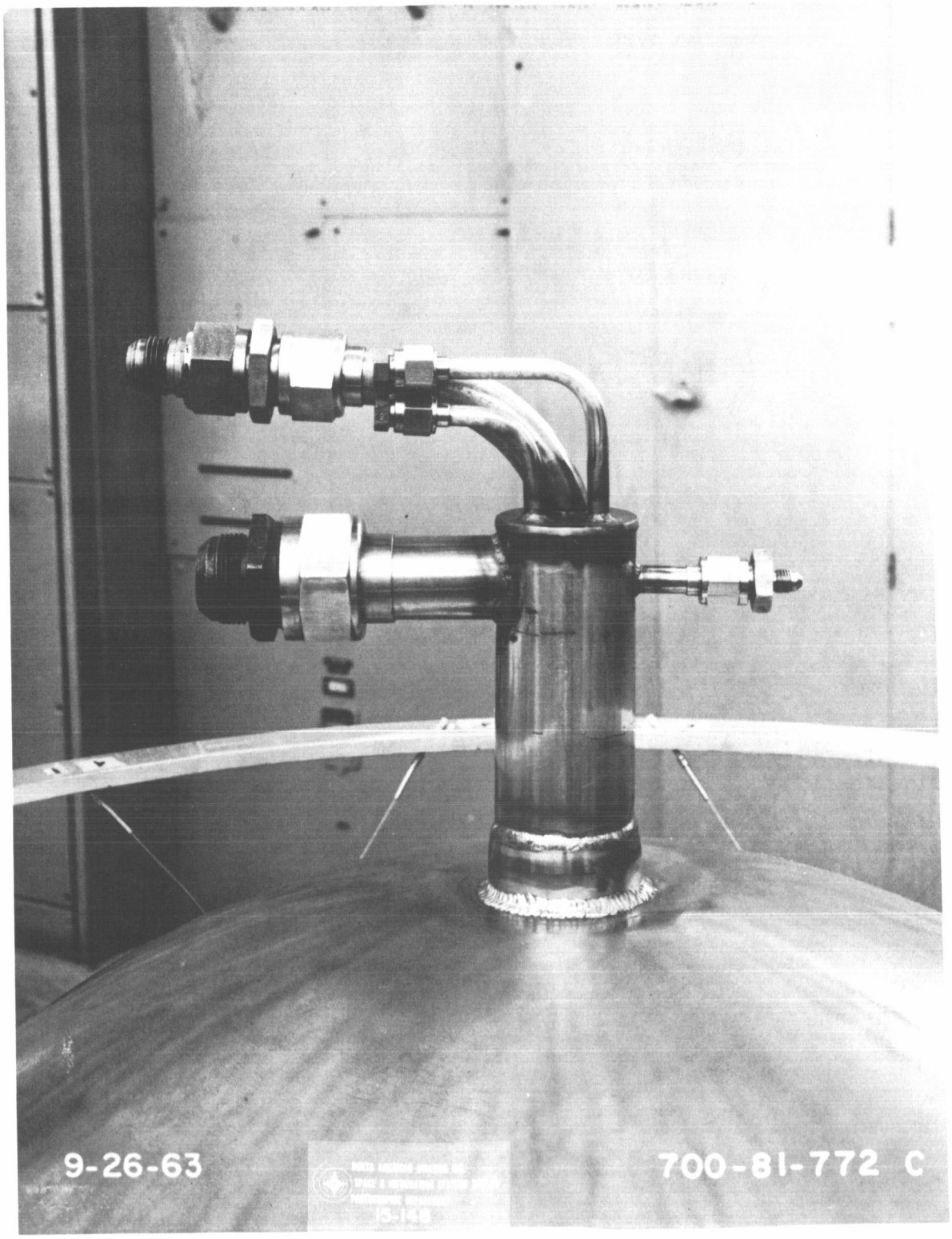


FIGURE 111

CONCENTRIC FILL, DRAIN, AND VENT MANIFOLD



spaced points around the circumference of the tank provide maximum torsional and axial rigidity, as well as providing small heat conductive paths. The use of stainless steel was selected because of the decrease in thermal conductivity and strength properties at cryogenic temperatures in preference to fiber glass or other synthetic materials. These tabs are welded flush with the tank to minimize local stress concentration at the tank wall. However, in review of the application of superinsulation materials, the tab design will have to be modified for the Phase II design. The application of superinsulation NRC-2 begins first with laying up multiple layers to form the insulation blanket, Figure 113. For this particular design, the application of the superinsulation material begins with two circumferential bands which are placed around the structure as shown in Figures 114 and 115. A portion of the upper band must overlap the meridian so that the optical paths are minimized. The positioning of the circumferential bands and polar caps is shown in Figure 116. Figure 117 shows the method of forming this cap. Figure 118 shows the fully insulated tank.

PHASE II - DESIGN - GROUND HOLD VACUUM STORAGE

The Phase II design, incorporating the Phase I design, will add a ground hold vacuum seal bag or structure. This vacuum is required so that the superinsulation material will perform at maximum effectiveness. Various concepts have been considered; these include stand-off columns with a mylar bag and a waffle grid structure molded into a mylar bag which would surround the tank. Due to the complexity of the problem, the Phase I test will be completed before starting the Phase II design. This design requires two operational phases: (1) loading the cryogenic propellant during ground operations so that boil-off and ice formation on the surface of the tank would be minimized, (2) be compatible with Phase I design performance for altitude or space flight mission performance.

THERMAL DESIGN REQUIREMENTS

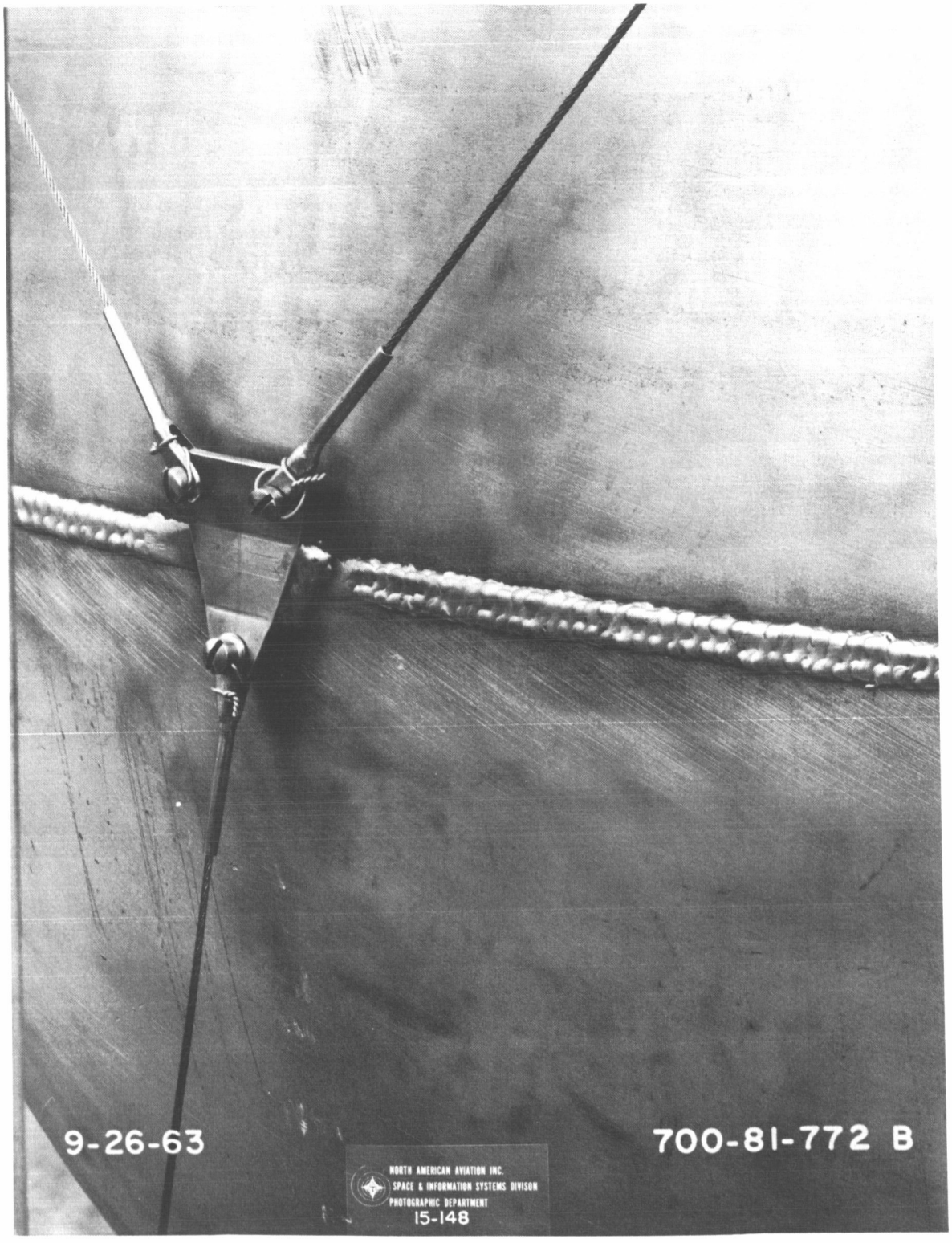
Multiple radiation shield insulation (super insulation) offers a combination of low effective thermal conductivity and material density. Theoretical performance analysis of this type of composite material for long-term storage and conservation of cryogenic propellants for space missions has been demonstrated in the past (Reference 1). However, the practical application of this insulation material has not been accomplished for the integrated design requirements of a space vehicle and its thermal environment.

CONFIGURATION

A basic thermal design criteria established for the test configuration has been that of minimizing the number of penetrations through the insulation. Careful design can minimize the heat leak through penetration (heat leak through paths other than the insulation). The presently scheduled test configuration consists of a spherical stainless steel tank, three feet in diameter. The tank is supported by wires which are attached to clips welded at eight points about the tank equator. These clips are buried under the superinsulation, which is applied after the rest of the tank has been insu-

FIGURE 112

SUPPORT WIRE STRUCTURAL ATTACHMENT TABS



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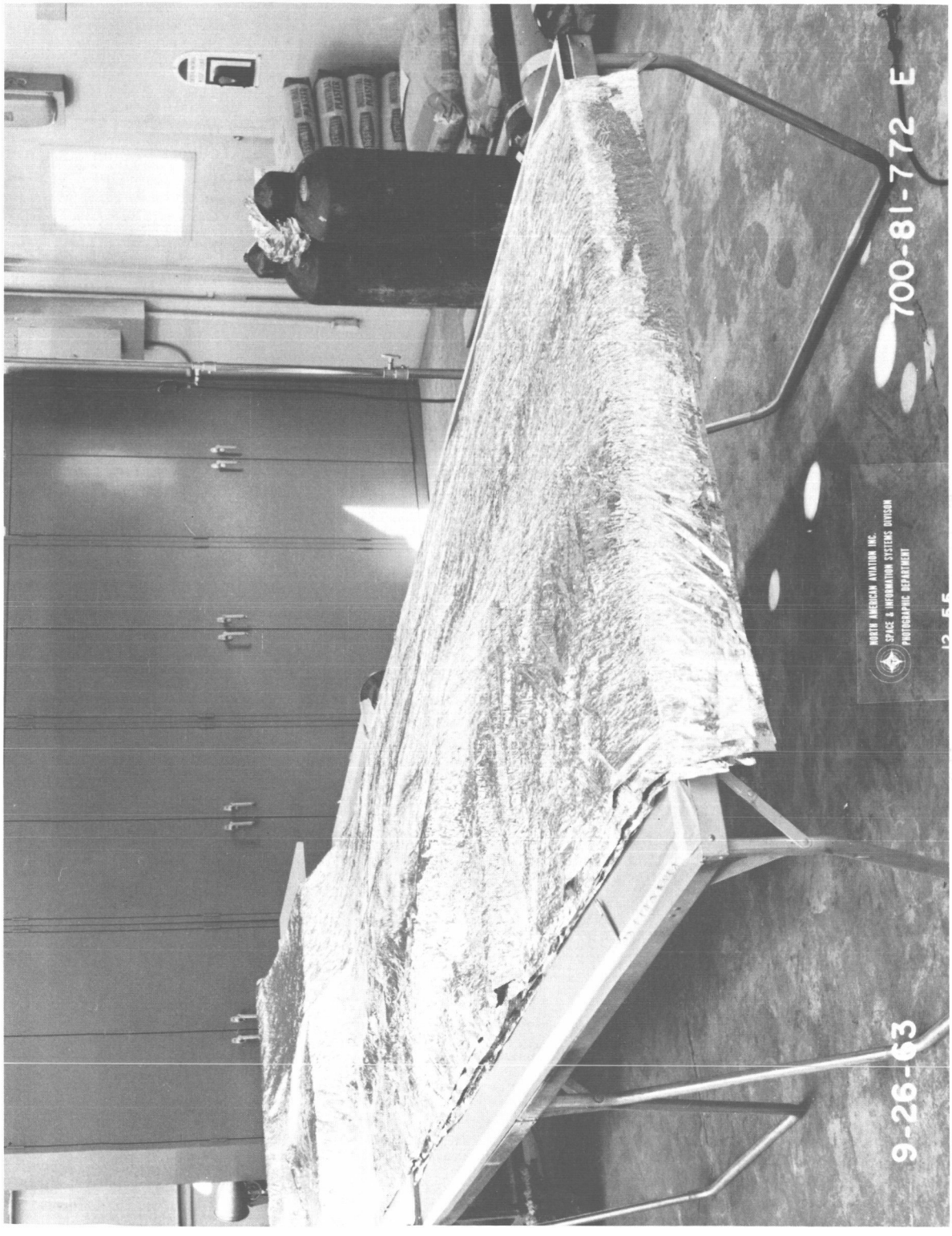


FIGURE 113
MULTIPLE LAYER INSULATION BLANKET

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FIGURE 114
INSULATION BLANKET PRIOR TO INSTALLATION

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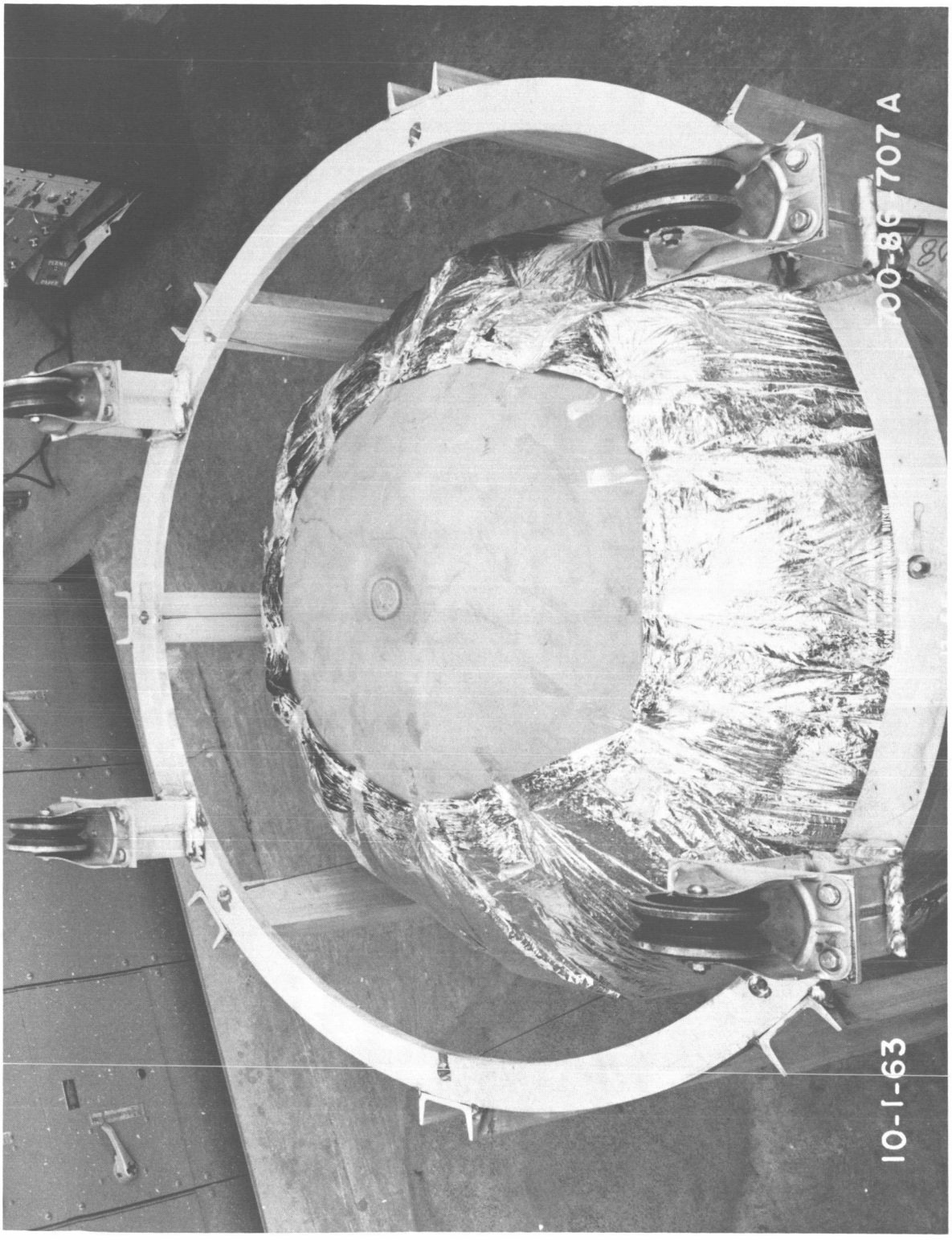


FIGURE 115
CIRCUMFERENTIAL BANDS APPLIED TO TANK

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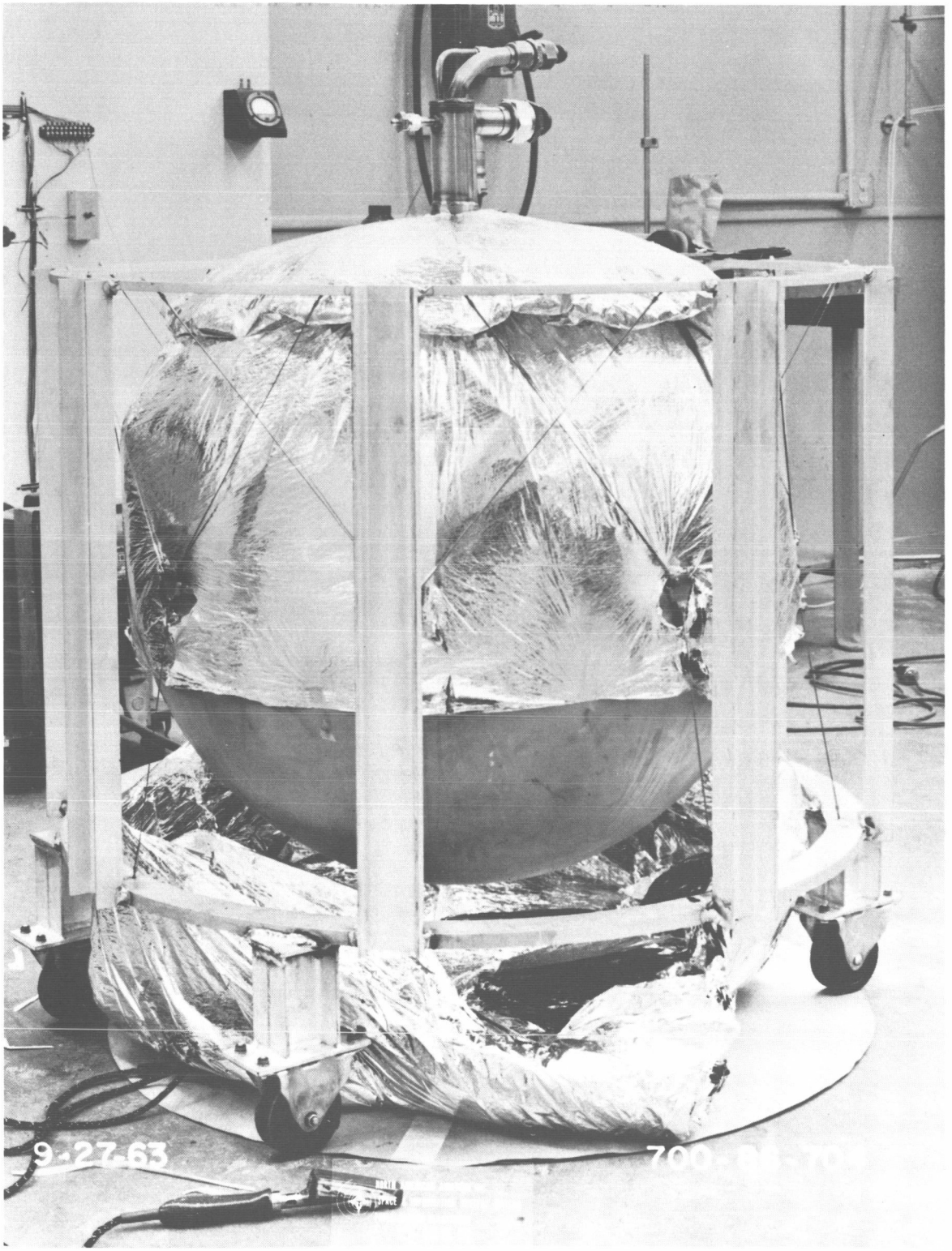
FIGURE 116

FORMING OF POLAR CAP/INSULATION BLANKET



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FIGURE 117
POLAR CAP INSULATION APPLIED TO TANK

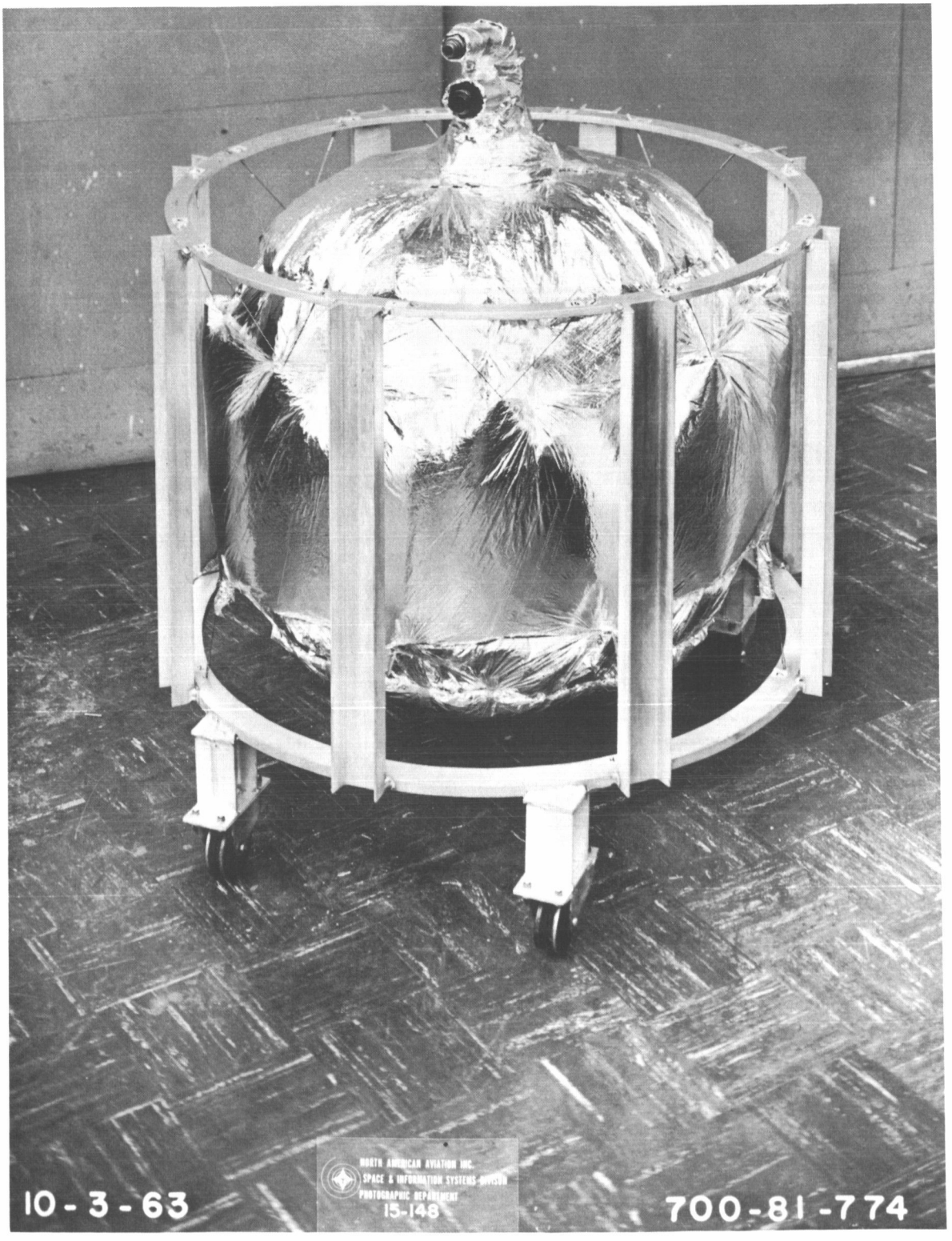


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FIGURE 118

FULLY INSULATED TANK



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lated. Thus, the only structural penetration of the insulation is made by the wire supports which provide a desirably low cross-sectional area and long conduction path for high thermal resistance.

A typical propellant tank has penetraions for a fill vent, re-circulation and pressurization lines, and electrical leads. For an efficient thermal design, these penetrations are made at one location as a concentric umbilical penetration through the wall of the test tank. This not only avoids large temperature gradients across each penetration, but simplifies the application of the superinsulation.

To further reduce heat leak through the fittings, the tank fill line has been located within the vent line. The fitting package is thus maintained with the desirable temperature gradient, with higher temperatures at the outside surface, and lower ones in the interior fill line. Superinsulation has also been wrapped around the fill and vent lines (Figure 119), essentially eliminating radial heat leak into the lines. The only heat leak to be considered through the lines would then be that conducted lengthwise.

In order to provide the necessary vacuum condition (10^{-5} Torr) for insulation performance, the first test (Phase I) will be performed in an environmental chamber located at S&ID Engineering Development Laboratories. For safety and ease of handling, liquid nitrogen will be used as the cryogen. After demonstrating the feasibility of the superinsulation and wire supports, a subsequent design and test Phase II will be developed to demonstrate a method of using the superinsulation when externally subjected to atmospheric pressure.

HEAT LEAKS

The thermal performance of the superinsulated wire-supported tank design will be demonstrated by a low leak measured for the configuration test. Paths for heat flow into the cryogen will consist of: (1) radiation propagation through the superinsulation from the chamber walls, and (2) conduction through wire tank supports, and (3) conduction along fill and vent line fittings.

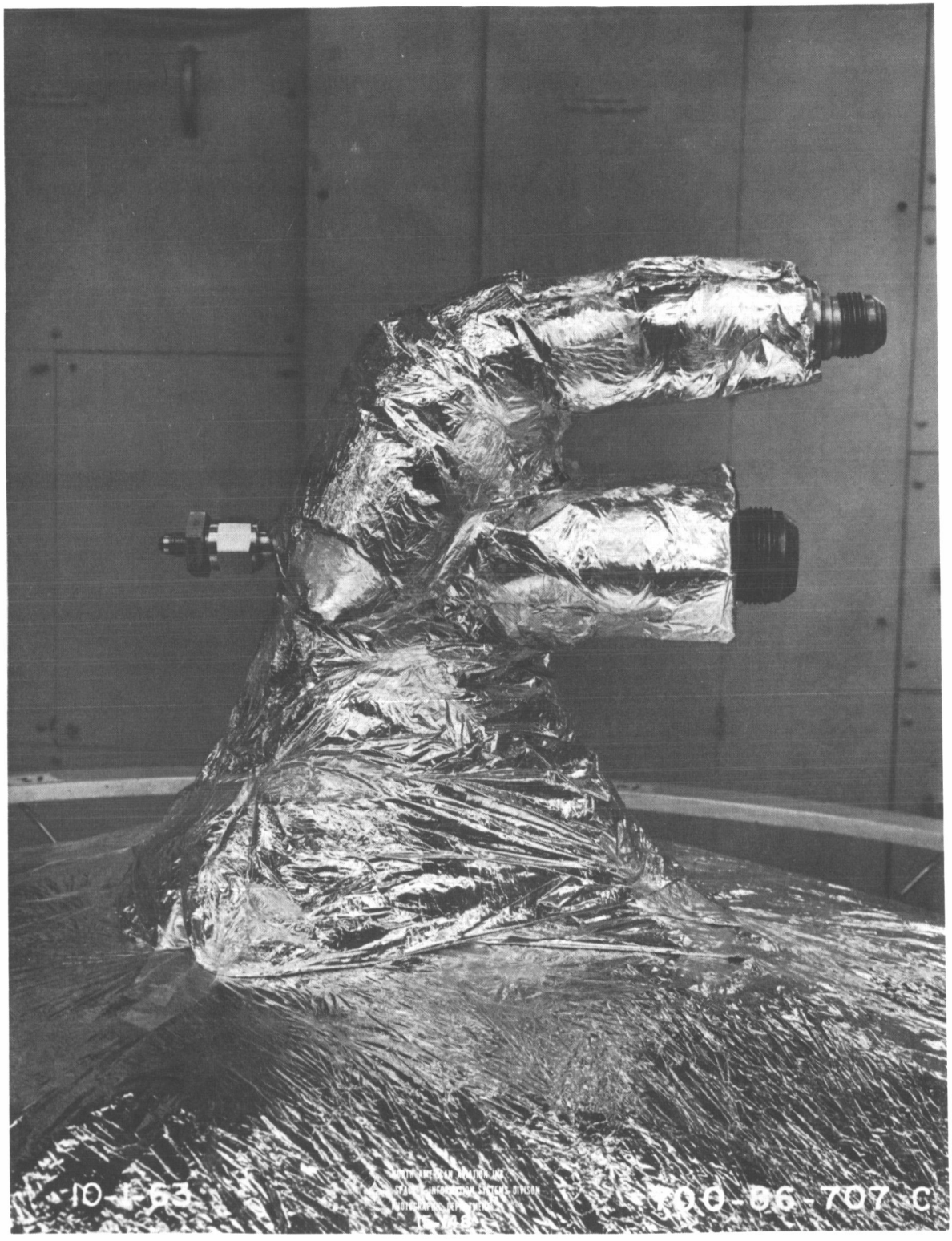
NRC-2 superinsulation was selected for use in this test because of the low material thermal conductivity and density. For some non-optimum installations, i.e., compressed, which might be used for some missions (other than long-term storage), the other leading superinsulation, made by Linde, might be more desirable. However, since the proposed application method is to demonstrate superinsulation at optimum design conditions, NRC-2 was selected.

NRC has recommended material laminar densities on the order of 80 layers per inch for best performance. In order to achieve a reasonably short test period of a few hours, rather than a few weeks, yet not use too small a thickness of insulation for performance requirements, a nominal thickness of .50 inch was suggested for the test tank. This would require 40 to 50 layers of the sheet material. The application of the superinsulation to the

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FIGURE 119

FILL AND VENT LINE INSULATION



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tank has been accomplished at S&ID under the supervision of a representative of NRC. The tank configuration has resulted in extra thicknesses of the material in some sections. As long as the insulation is not compressed below the limit, the extra thickness will add conservatism to the calculated heat leak based on one inch nominal thickness for the whole tank area. However, over-compression of the material would increase the heat leak to the cryogen.

Calculations for the heat leak into the tank has indicated about equal amounts of heat flow through the fittings and insulation and approximately one-half that amount through the tank wire supports. The heat leak values obtained were: 5.5 BTU/hr through the tank insulation, 3.3 through the wire supports, and 5.0 through the fittings for a total of 13.8 BTU/hr transferred into the liquid nitrogen during the test. A nominal one-half inch of NRC-2 superinsulation was assumed to cover the 28.3 square feet surface area of the spherical tank. Thermal conductivity for the material was taken to be 2.0×10^{-5} BTU/ft-hr-F in accordance with the performance and material specifications published by NRC. The walls of the environmental chamber will be at ambient temperature during the test and were assumed at 80°F for the calculations. Initial surface temperatures of the insulation will also be at ambient and were assumed to be at a constant 80 degrees. This simplifies calculations and makes them somewhat conservative.

Eight stainless steel clips have been welded about the equator of the tank. Three stainless steel supporting wires are attached to each clip, two to the upper ring support, and the third to the bottom one. A circumferential band of superinsulation was installed over the clips as the insulation had been cut out in these areas to facilitate application of the superinsulation around the support attachment. The wires are wrapped with superinsulation to insure heat transfer by conduction only, with no contribution by radiation from the chamber walls (Figure 120. An effective thermal conductivity of 7.0 BTU/ft-hr-F was assumed for the 1/16 inch stainless steel braided wire. An effective length of 12 inches was assumed for each of the 24 wires. The trussed ring supporting structure was assumed at a constant 80°F for the duration of the test.

A one-inch diameter stainless steel fill pipe is brought through the environmental chamber wall and will cross over into the larger vent line about eight inches above the top of the tank. Two 3/8 inch re-circulation lines also penetrate the vent line at this point, whereupon the complete umbilical package makes one penetration through the top of the tank (Figure 2). The fill and re-circulation lines extend downward almost to the bottom of the tank. The re-circulation lines are not used in the test but have been included to simulate a heat leak path which would be present in an actual propulsion system. The vent line has been sized at 1.5 inches to accommodate the large quantities of gas which will be formed during cool-down at the start of the test. The umbilical section vent area was also sized for a flow area equivalent to that of a 1.5 inch diameter line. Stainless steel tubing of the 321 series, with as thin a wall as possible, was recommended for a low heat conduction path through the lines. Pipe with a wall thickness of .035 inch was readily available and was used for the pro-

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FIGURE 120



SUPPORT WIRE ATTACHMENT AND INSULATION

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pellant lines. The fittings and lines are insulated completely to the wall of the environmental chamber.

This should eliminate radial heat transfer into the lines, allowing only heat conduction along the lines into the tank. With the small cross-sectional area and conservative effective lengths assumed for heat conduction, a heat leak of 5.5 BTU/hr was calculated for the total heat leak into the cryogen through the lines and fittings. Some approximations were necessary for these calculations because of the complexity of the umbilical configuration for heat transfer analysis.

The question of a need for optical baffling in the fill and vent lines was raised early during the configuration design. Radiation heat transfer through tubular construction is a function of geometrical configuration factors which, in turn, are very sensitive to diameter-length ratios. The configuration factors have been found to be very small for the test configuration. An analysis has shown the radiation heat transfer through the fill and vent lines to be small and is considered to be negligible for this test.

HEAT LEAK MEASUREMENT METHODS

The feasibility of a superinsulated, wire-supported cryogenic propellant tank will be demonstrated by a low-heat leak into a test configuration incorporating these concepts. The method of measuring the heat leak into the cryogen must be accurate and be capable of measuring low values within short time periods. Possible measurement methods include: (1) load cell, (2) gas flow, (3) pressure build-up, and (4) liquid displacement.

The load cell method would weigh the test configuration before and after the test period. The difference in weight is then attributed to boil-off which could be interpreted as a heat leak to the cryogen for measured temperatures and pressures. This is a simple method but could not measure accurately enough the low boil-off losses which would be expected from the tank. With the nitrogen boiling at one atmosphere and 140°R, an estimated heat leak of 20 BTU/hr (conservative interpretation of the 13.8 BTU/hr calculated) would result in a loss of 0.23 pounds of nitrogen per hour. The total system weighs approximately 2000 pounds. With a one-tech of one percent full-scale accuracy, it would take at least a forty-hour test period to measure a weight loss with 20 percent accuracy. The test set-up being contained in an environment chamber with narrow clearances would also make the installation of the load cell difficult if not impossible.

Gas flow and pressure build-up methods have been used at S&ID for previous measurement of cryogenic boil-off losses. The low flow rates which would be expected in this test could not be accurately measured continuously. The pressure build-up method would close the tank, allow the vapor pressure to increase to some value like 50 psia, then reduce the pressure by 5 or 10 psi while measuring the gas flow. This could be done as many times as desired. With careful measurement techniques, including accurate pressure and temperature histories both in measurement apparatus and within the tank, the loss in weight could be interpreted as a heat leak. Some method would also

be required to indicate either the weight or liquid level within the tank at the start of the test period. The accuracy of this method depends on a homogeneous liquid bulk temperature, which is not likely to exist. Liquid stratification has been demonstrated in many similar tests and is expected to be present in this one. Depending on the degree of stratification, the indicated heat leak could be much higher than that actually evaluated. If no stratification is assumed, starting at 10 percent ullage and pressurizing to 50 psia (T_s = 161 R), the bulk denisty will decrease with temperature from 50.4 to 46.8 pounds per cubic feet. The ullage will decrease from 10% to 3% due to the resulting rise in the liquid level. Loss of bulk liquid from vaporization would be negligible. The time required to raise the bulk temperature from 140°R at one atmosphere, to 161°R at 50 psia would be

$$t = \frac{WC_p\ (\Delta T)}{q} = \frac{641(.48)(161-140)}{20} = 323 \text{ hours}$$

This time requirement is not practical for the test. If, for example, the warmer liquid surface temperature produced by stratification caused the vapor pressure to reach 50 psia in 50 hours, this could be erroneously interpreted as a heat leak 6.5 times the actual.

In order to accomplish a heat leak measurement accurately and in a reasonable time period, a measurement of liquid displacement due to density change is considered a practical and feasible technique. This method is based on liquid density change with temperature. By measuring this density charnge and interpreting it as a change in temperature, the heat leak required to change the temperature can be calculated. Stratification would be of no consequence since this method is the only one in which a true mean bulk temperature is measurable. This is because the density curve for liquid nitrogen is approximately linear in the temperature range of the test. The only measurement required to determine heat leak would be the time required for a specified change in liquid level within the tank. The change in liquid level would have been previously calibrated for a corresponding volumetric displacement. Liquid boiling would be prevented during the test by pressurizing the tank to increase the boiling temperature well above that reached by the liquid. Point capacitance gages could be used in a calibrated arrangement to indicate the volumetric change of the liquid. Several gages of this type had been used during S-II studies.

The density reaction of the liquid nitrogen can also be utilized for measuring the heat leak by eliminating all vapor from the system and interpreting the heat leak with the adiabatic coefficient of compressibility of the liquid nitrogen. This method would measure the increase in liquid pressure due to decrease in density as the liquid accepts heat. A suitable pressure transducer is available at S&ID, and it is proposed that this method be used to measure heat leak into the tank. Accuracy with this method depends on preventing vapor from forming in the system. It is felt that this can be accomplished by pressurizing the (liquid) system to about 30 psia or more after cool-down has been reached. This point will be established by plotting flow rates with auxiliary external measurement devices. The liquid pressure would then be monitored until it reaches approximately 60 psia.

It is estimated that this pressure would be reached in 2.2 hours for a heat leak of 20 BTU/hr into the tank.

TEST DESCRIPTION

The test program is designed to demonstrate system performance and assist in evaluation of long-term cryogenic propellant storage for an integrated tank design including support structure, propulsion system plumbing, and application of NRC-2 vacuum superinsulation.

PHASE I - ALTITUDE TEST

The altitude test, representative of the thermal environment of the tank during space transit will be performed at a pressure of 10^{-5} Torr for an environment wall temperature of approximately 70°F. Basically, this test will confirm the design for long-term vacuum storage of cryogenics and indicate performance of NRC-2 superinsulation material.

The S&ID 6" diameter environmental chamber (Figure 121)will be instrumented and plumbed to obtain the heat leak to LN_2 system and ΔT temperature measurements.

Heat leak measurements shall be made at suitable intervals for determining equilibrium conditions. Because of the small heat leak expected, it is very important that the instrumentation employed obtain the necessary accuracy. Pressure measurements will be obtained through a transducer specifically designed for cryogenic liquid temperatures. Temperature gradient (thermocouple) measurements (Figure 121) will be taken during this operatinng cycle for each hour for the first thirty hours and at each two-hour interval thereafter. These temperature gradients will be used for over-all system evaluation and they are not to be construed as absolute measurements. The cryogenic storage tank/insulation/support system is shown in the altitude chamber ready for test in Figure 122.

Operating Procedure (after installation of test tank):

1. Evacuate to 10^{-5} Torr, walls at near-room temperature. Record room and wall temperature for each hour for duration of test.

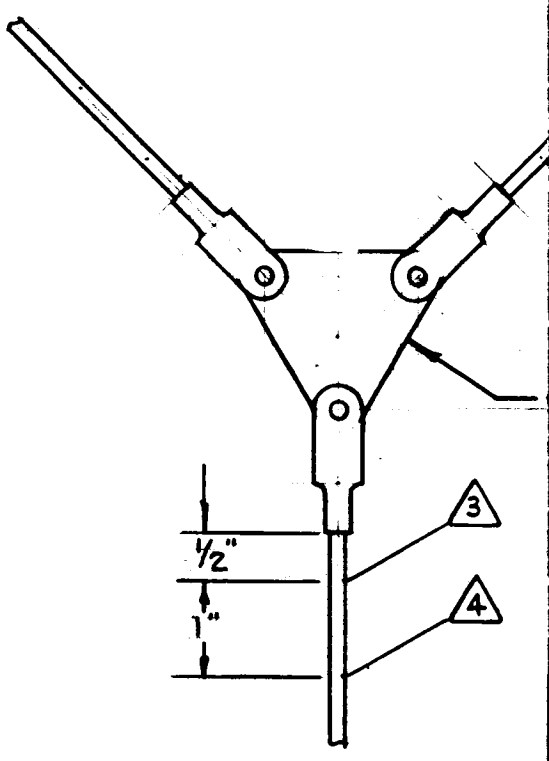
2. Cool down and fill tank with LN_2, allowing to "top-off." Establish equilibrium condition. Purge system with LN_2 after equilibrium condition is obtained.

3. Pressurize system to 40 psig.

4. Allow pressure build-up necessary for obtaining steady state heat rate. Record all data at indicated intervals or when pressure has been allowed to accumulate sufficiently for a measurement to be taken.

5. Plot heat leak (boil-off) rate vs time on log/log chart to indicate performance of test during operation.

SPACE CHAMBER
(6' DIA. X 14' LONG)



VIEW A

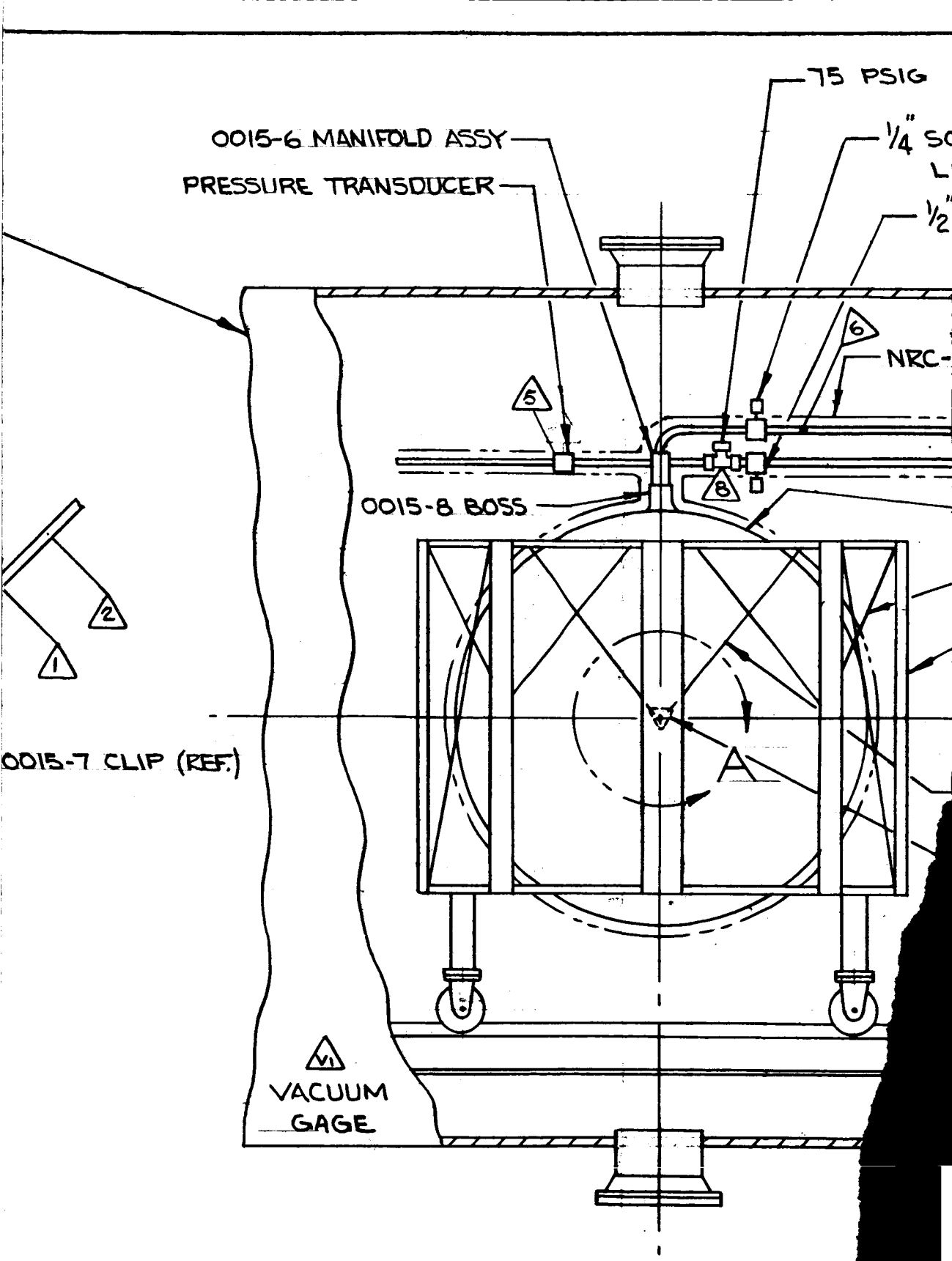
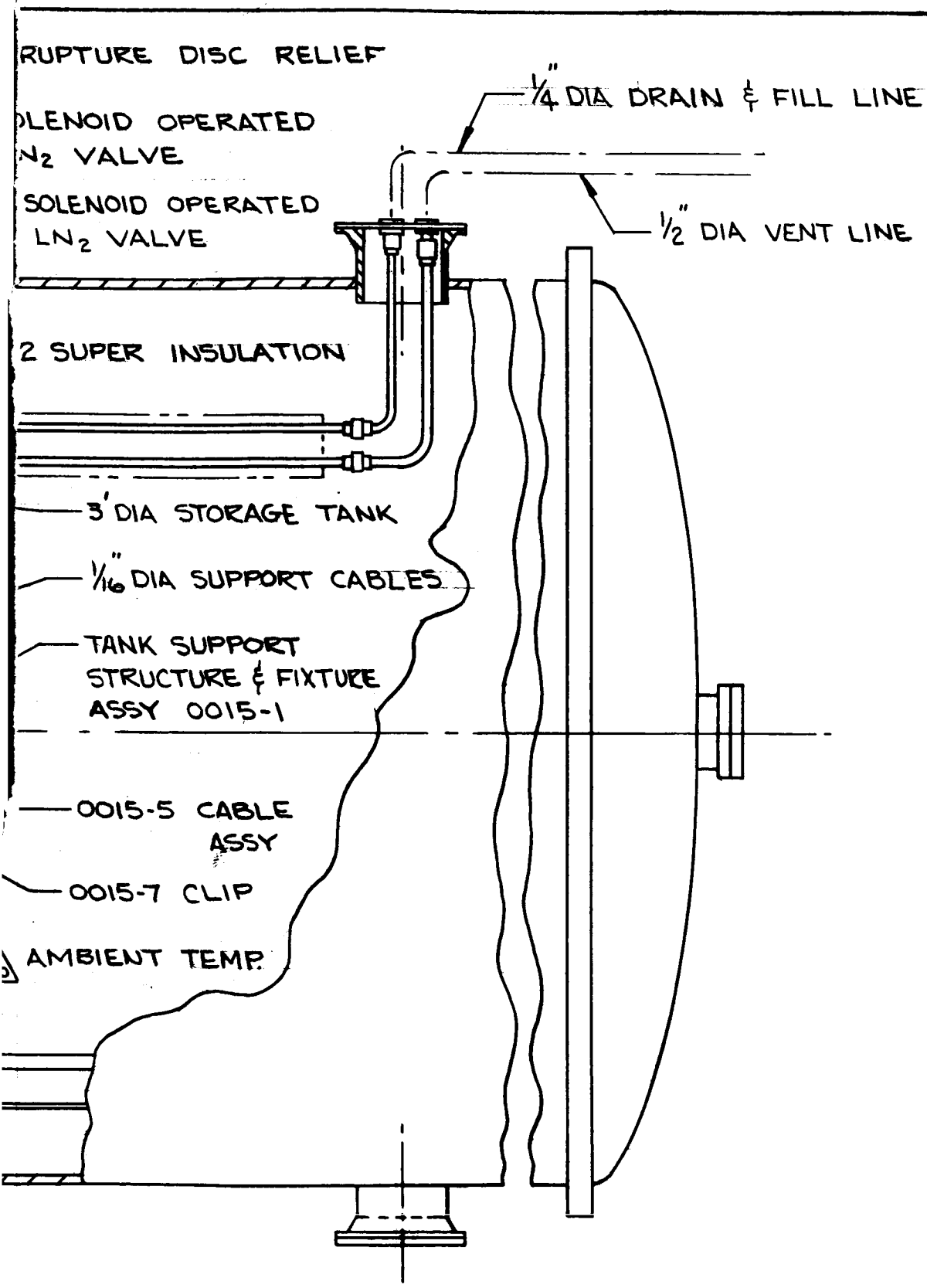


FIGURE 121

P 338, 339



TE:

△ TRIANGLES INDICATE THERMOCOUPLE LOCATIONS.

TEST SETUP FOR APOLLO II
QUID HYDROGEN STORAGE CONCEPT

6. Record thermocouple readings at intervals indicated.

7. After steady state conditions have been obtained, record heat leak data for duration of test.

8. Terminate test after three liquid pressure build-up measurements have established steady state values.

PHASE II - TEST AND EVALUATION OF NRC-2 SUPERINSULATION FOR VACUUM AND ATMOSPHERIC STORAGE USING A VACUUM SEALED BAG

The performance of the design concept for ground handling and storage will be measured by LN_2 boil-off data. A similar test procedure as used for Phase I will also be used for Phase II.

More sophisticated temperature and boil-off measurement concepts may be incorporated as a result of the Phase I test program.

PHASE III - GROUND HOLD STORAGE @ 70°F = LIQUID HYDROGEN

Atmospheric storage of LH_2 to demonstrate extreme operating conditions.

PHASE I TEST RESULTS

Summary

The cryogenic tank, insulated with NRC-2 superinsulation, was installed and tested in a vacuum chamber to determine the effectiveness of the insulation. Evaporation measurements showed a heat leak into the tank of 21.2 BTU/hr; the tank was filled with 100 to 105 gallons of liquid nitrogen. A photograph of the tank as installed in the vacuum chamber is shown in Figure 122.

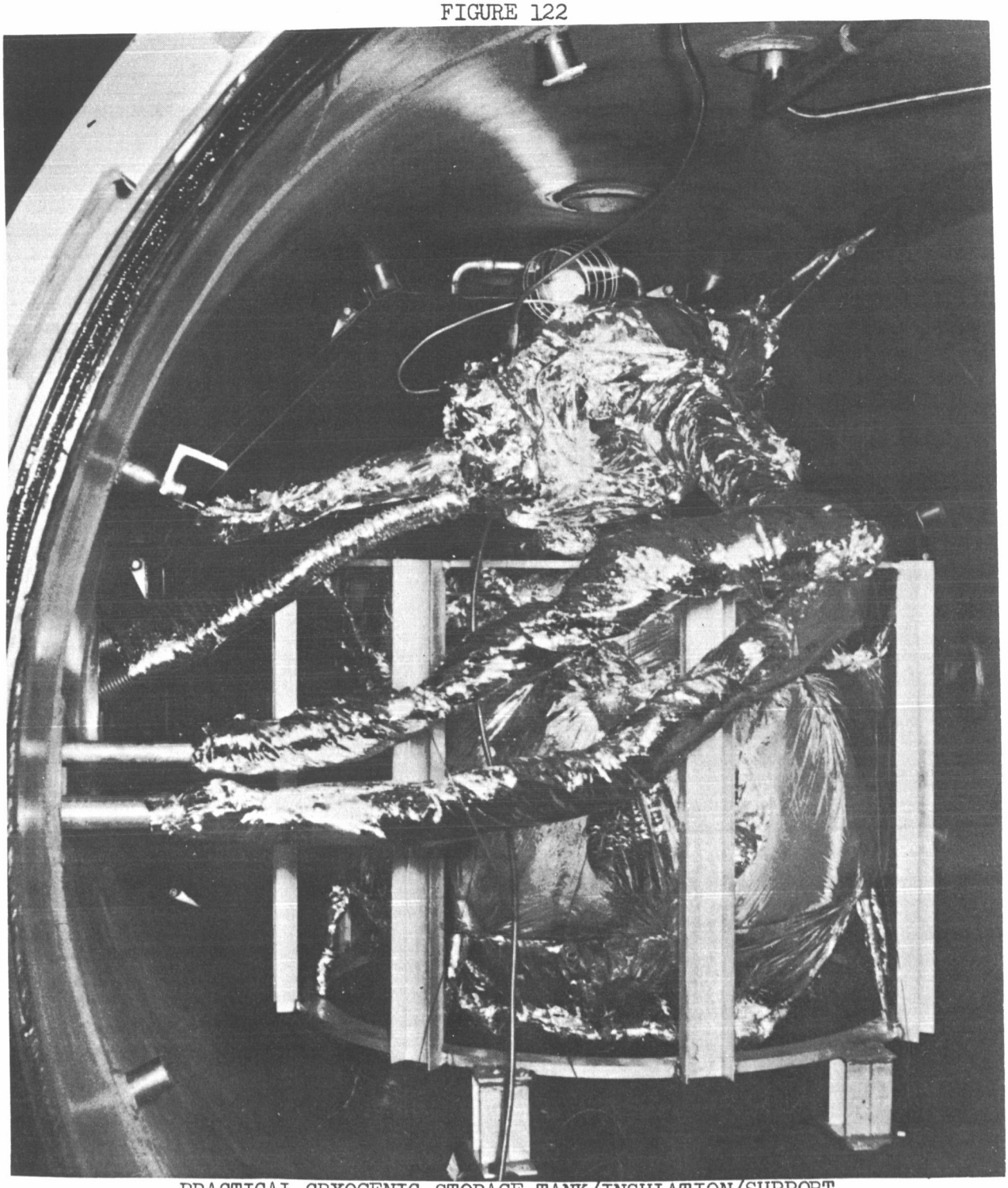
Test Set-up

The insulated sphere and holding fixture were installed in the 6' diameter by 14' long space simulation chamber. Initially, manually operated cryogenic valves, Annin Model 1510, were installed on the vent and fill lines. Liquid nitrogen barrier coils were wrapped around the valves and the piping to reduce heat paths to the tank. A cryogenic transducer, Type CPA 413 TC-200, was installed on the tank manifold for measuring pressure buildup. Additional instrumentation included a Veeco Model RG-31X Ionization and Thermocouple Cage Control for measuring chamber vacuum. Figure 121 shows location of the above instrumentation plus nine thermocouple sensors used to provide temperature data for showing the location and magnitude of heat paths. A Leeds and Northrop Potentiometer Model 8693 provided temperature readout.

The basic test set-up was later modified to eliminate all valves inside the vacuum chamber. Valves were installed external to the chamber and a Fischer and Porter Flow Meter, Model No. 05370, was installed on the vent line to measure flow of evaporated gases.

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FIGURE 122



PRACTICAL CRYOGENIC STORAGE TANK/INSULATION/SUPPORT SYSTEM IN ALTITUDE CHAMBER

Test Procedure

Prior to installing the tank in the vacuum chamber, the supporting cables were adjusted to provide even loading on all supports as indicated by a standard cable tension tester. With 300 lb dead weight on top of the tank, tension in the upper and lower cables was set at 70 $\pm$ 5 lb and 40 $\pm$ 5 lb respectively. After twelve hours, cable tension measurements were repeated to determine stability of the wire supports.

After completing installation of the speciman, the chamber was evacuated to the 10^{-4} torr range and LN_2 flow was started to the test tank. A sudden rise in chamber pressure indicating a leak in the internal piping caused the test to be aborted. A check of the valves immersed in liquid nitrogen showed the location of external leaks. The valves were subsequently modified by venting the shaft glands external to the chamber, but additional leakage developed and the valves were eliminated from the system.

The test was again started with solenoid valves installed on the vent and fill lines. The chamber was evacuated, liquid nitrogen flow was started through the barrier coils, and the test tank was filled. After a two-hour period of "topping off," the solenoid valves were closed to obtain pressure build-up data. After twelve hours, the vent solenoid valve was opened and a flow tube was installed on the vent line external to the test chamber. The test was continued for an additional seven hours until gaseous flow through the vent line stabilized. All test data was recorded every thirty minutes during the run.

A second test run was conducted after topping off the test tank to provide pressure build-up data using an external pressure gage installed on the vent line. This test was discontinued after two hours when the vent line solenoid valve failed in the closed position. The altitude chamber was brough back to sea level pressure, the vent line solenoid removed, and the test tank was drained.

After failure of the solenoid valves, a final test was conducted with internal valves removed from the vent and fill lines. The tank was filled and allowed to set over night. Seventeen hours after filling, approximately three gallons of liquid nitrogen was drawn from the test tank to provide a 3 to 5% ullage space. All test data was recorded every thirty minutes for the next seven and one-half hours until vent line flow stabilized. Maximum obtainable vacuum was maintained in the chamber during all test runs.

Test Results

Tension in tank support cables after twelve hours with a 300 lb static load on the tank is recorded in Table 70. Table 71 presents cable tension measurements after removing the load.

All tests for determining heat flux into the tank by the pressure build-up method with zero ullage space were unsuccessful because of leakage from the fill and vent valves. After several attempts, this method was abandoned

TABLE 70 - CABLE TENSION UNDER LOAD - AFTER 12 HOURS

CABLE No.	CABLE TENSION-lb UPPER	LOWER
1	62.5	55
2	67.5	
3	65	42.5
4	70	
5	65	45
6	70	
7	70	82.5
8	67.5	
9	65	55
10	70	
11	70	37.5
12	67.5	
13	67.5	42.5
14	67.5	
15	67.5	80
16	67.5	

TABLE 71 - CABLE TENSION WITH LOAD REMOVED

CABLE No.	CABLE TENSION-lb UPPER	LOWER
1	50	50
2	57.5	
3	52.5	42.5
4	57.5	
5	57.5	52.5
6	55	
7	50	62.5
8	45	
9	47.5	57.5
10	55	
11	55	40
12	57.5	
13	60	47.5
14	55	
15	47.5	55
16	50	

in favor of measuring evaporated gases with the liquid at atmospheric pressure. Figure 123 presents temperatures and gas flows in graphic form as a function of time for the test run with solenoid valves installed. Figure 124 presents the data without solenoid valves.

Discussion

The stabilized evaporation rate obtained when testing with solenoid valves was 131 scim (standard cubic inches per minute) and 101 scim without the valves. The higher rate was probably due to two factors: The poorer vacuum attained during the former test run and the heat input due to current loading of the solenoid valves.

Assuming that all heat into the tank under a stabilized condition is absorbed by the liquid nitrogen and is used to evaporate the liquid, the heat gain (Q) may be calculated as follows:

$$Q = gM$$

where:

Q = heat gain - BTU/hr
q = latent heat of vaporization (85.7 BTU/lb)
M = Mass Flow - lb/hr

By this method the heat gain for the final test run was determined to be 21.8 BTU/hr with a full tank of liquid and 21.2 BTU/hr with a 3 to 5% ullage space.

An examination of the temperature gradient across the supporting wires indicated that the heat leak rate into the test tank was approximately 2.0 BTU/hr. The heat leak rate across the insulation was calculated as follows:

$$Q_{insul} = Q_{total} - (Q_{wires} + Q_{plumbing})$$

$$+ 14.5 \text{ BTU/hr (Average value, both test runs)}$$

$$\dot{q}_{insul} = \frac{Q}{A} = 0.51 \text{ BTU/hr-ft}^2$$

Conductance across insulation ($k_{apparent}$)

$$= \frac{\Delta X}{\Delta T} = \frac{0.51 \times 1.0}{12 \times 384} = 11.0 \times 10^{-5} \frac{\text{BTU}}{\text{hr-ft}^2\text{-°F/ft}}$$

The $k_{apparent}$ calculated for this test set-up is approximately five times the value published by the manufacturer (2×10^{-5}) for a theoretical installation. It is felt that a large part of this degradation of the superinsulation properties can be attributed to local compression of the superinsulation due to the installation techniques required to insulate this configuration (i.e., compression in the area of the wire support tabs.) However, it should be pointed out that the results are, nevertheless, very encouraging. A heat flux of 0.51 BTU/hr-ft^2 is considerably lower than the values now being realized in many cryogenic designs. Furthermore, if this order of magnitude of heat leak can be obtained in an atmospheric environ-

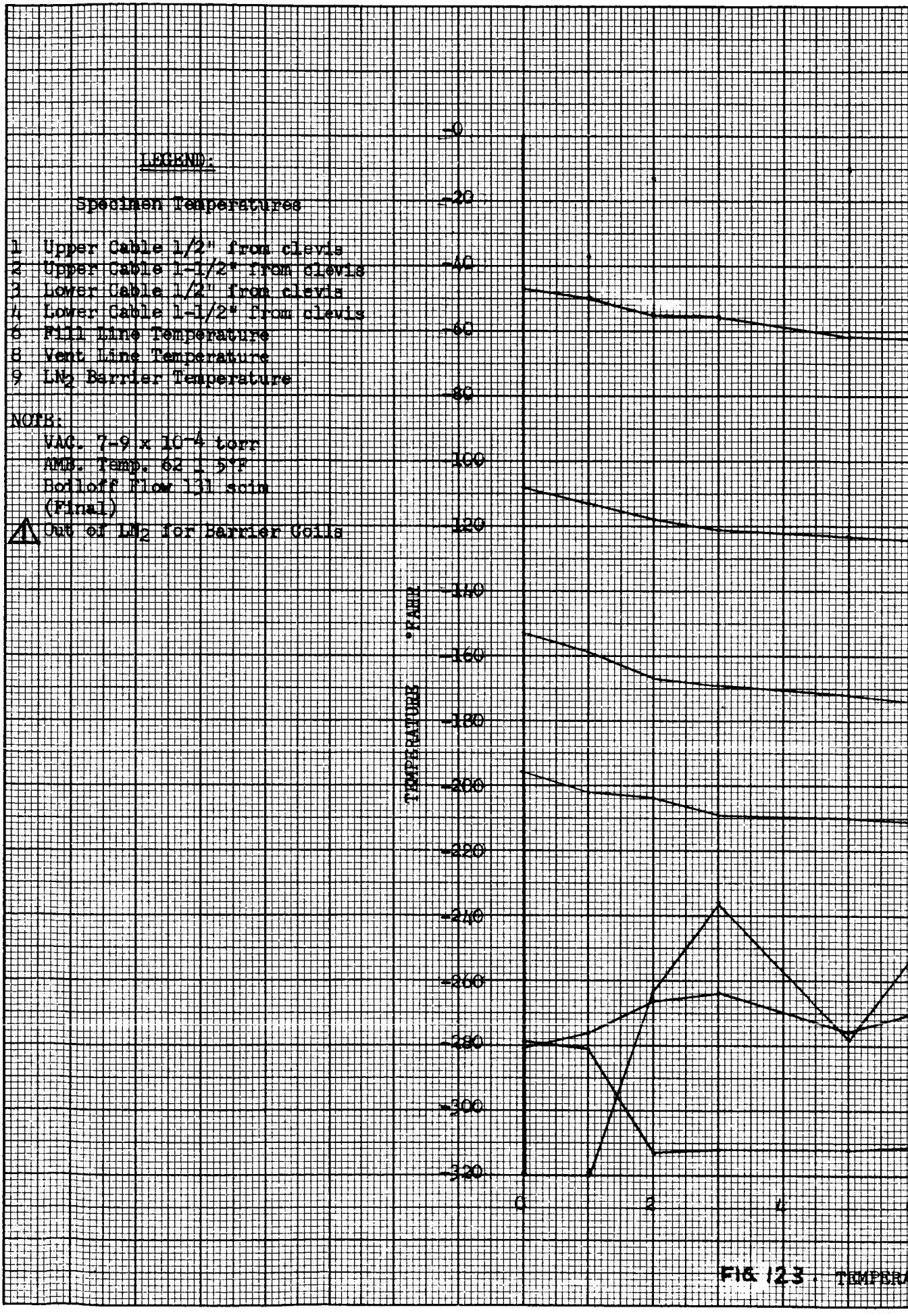
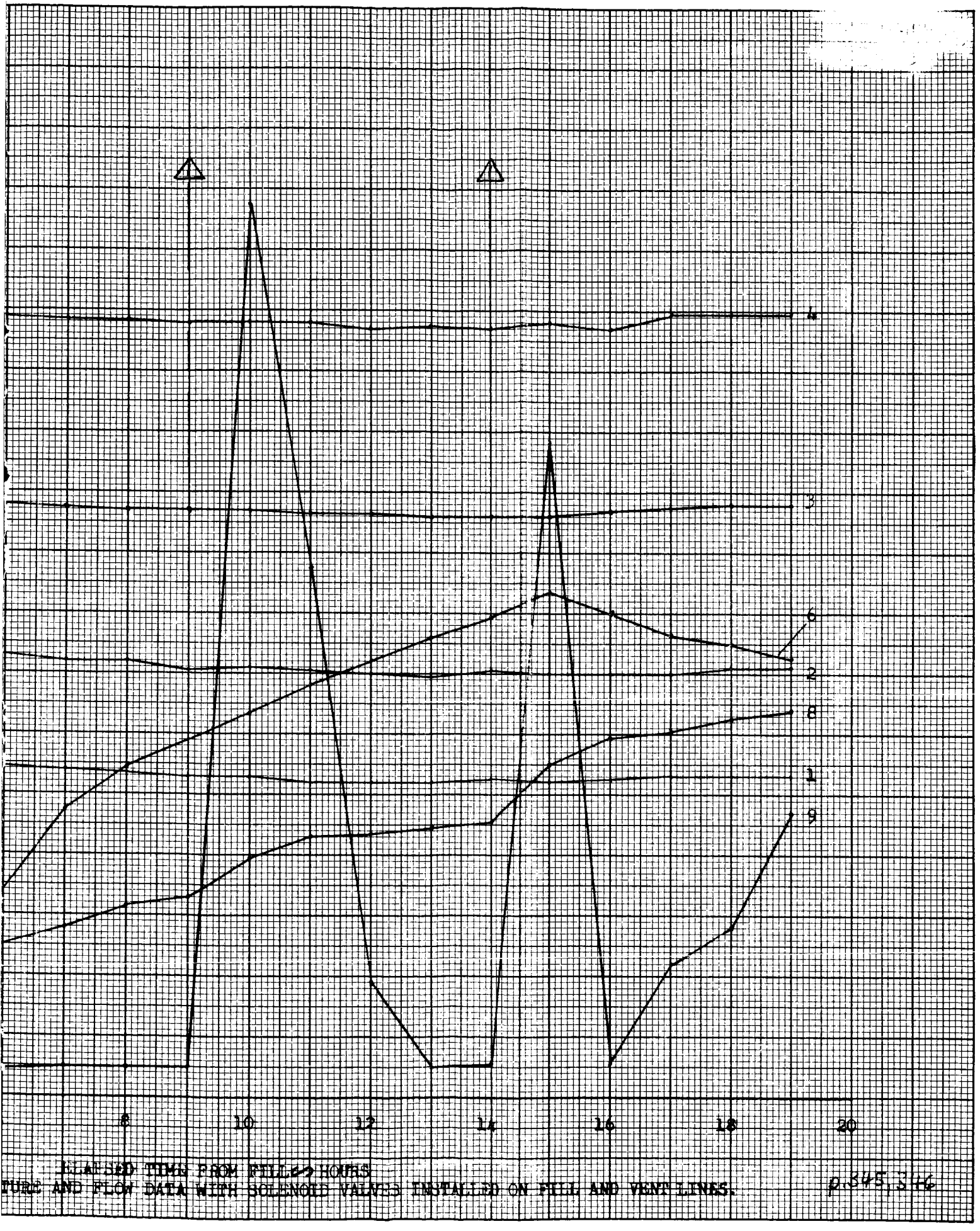


FIG. 123 - TEMPERA



ELAPSED TIME FROM FILL ⟶ HOURS

TURE AND FLOW DATA WITH SOLENOID VALVES INSTALLED ON FILL AND VENT LINES.

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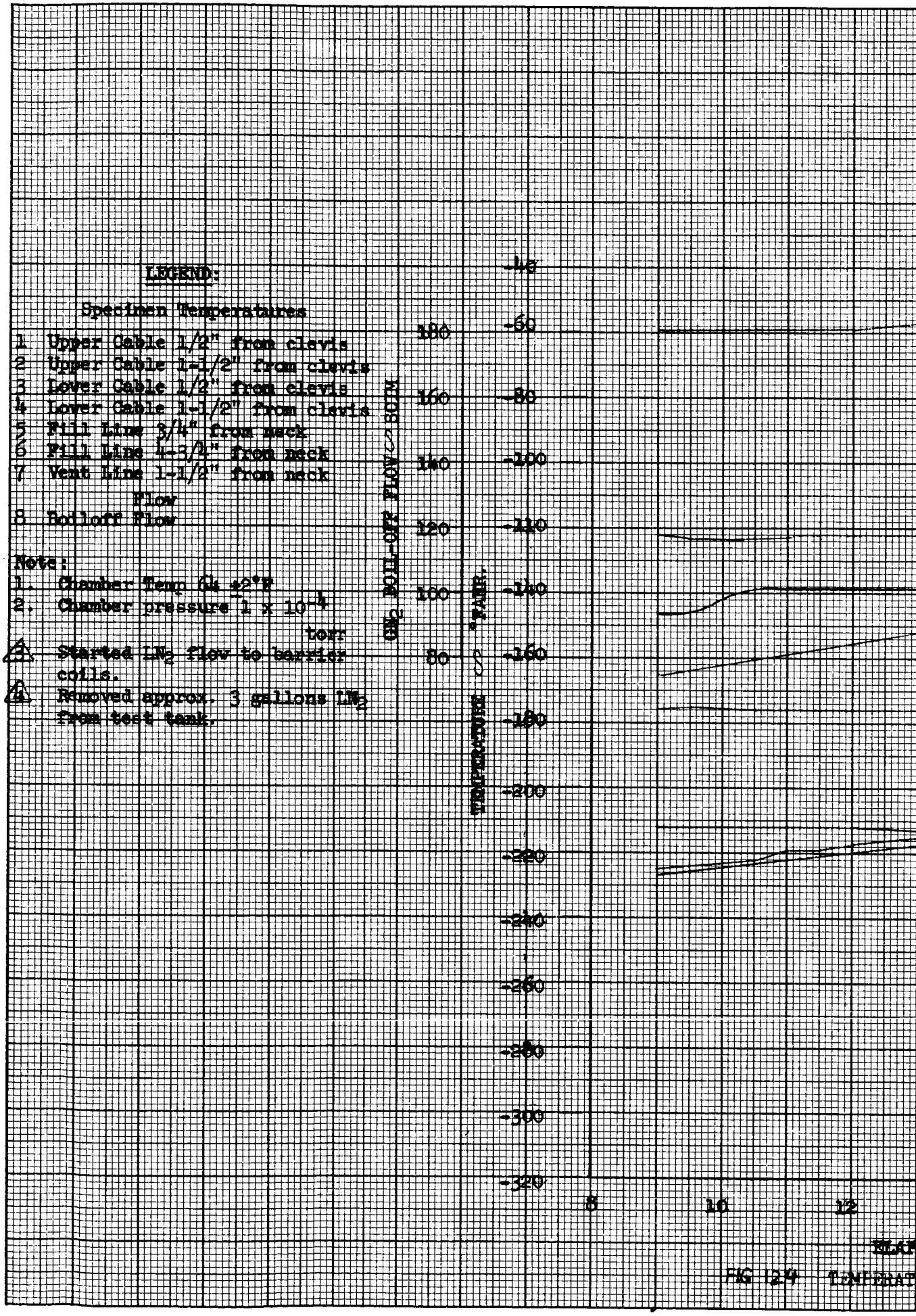
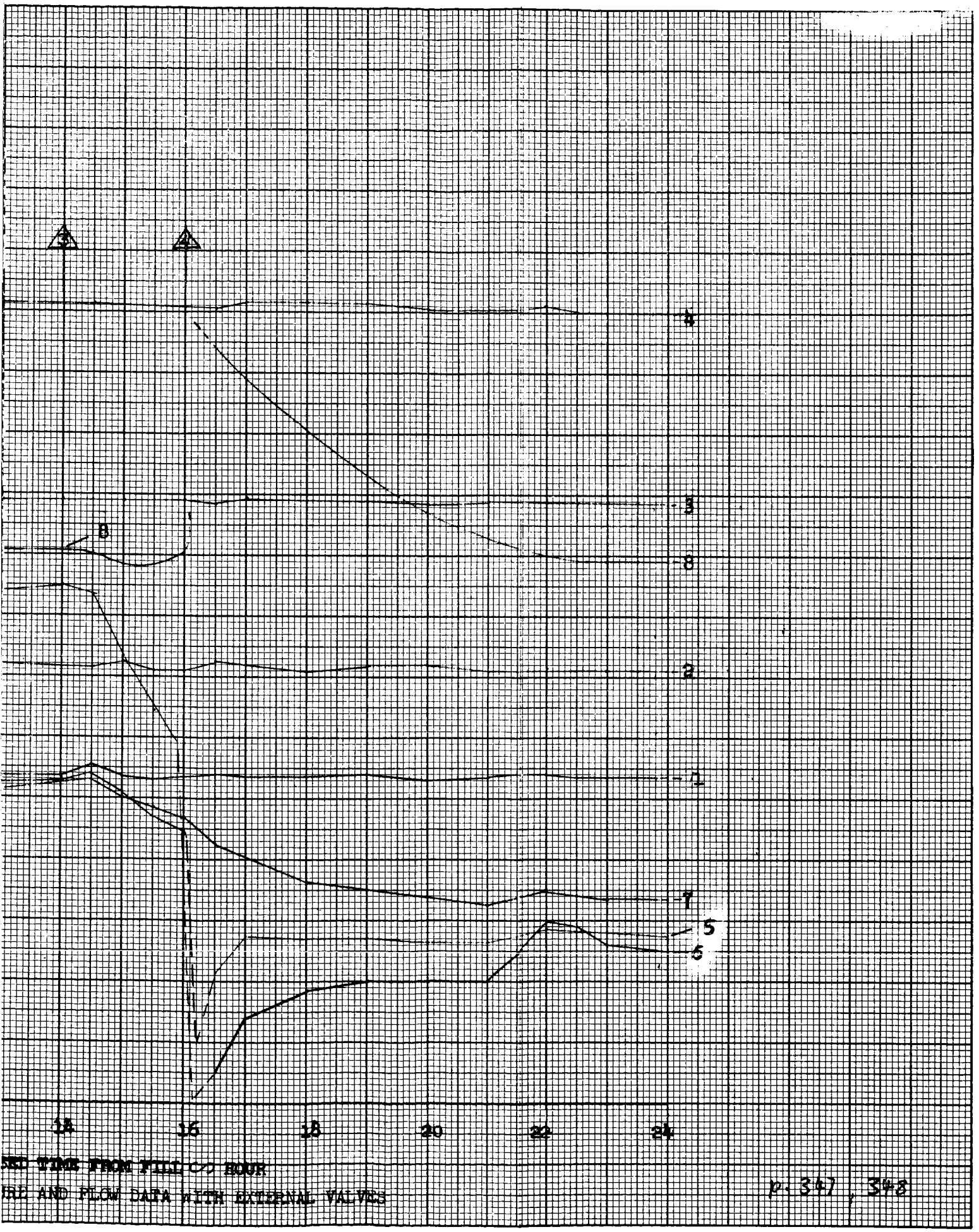


FIG 124 TEMPERAT



KED TIME FROM FILL (-) HOUR

RE AND FLOW DATA WITH EXTERNAL VALVES

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ment (by evacuation of the superinsulation), the use of sub-cooled cryogenic fluids for space vehicles becomes feasible. (Assuming the economic acceptability of sub-cooled cryogenic fluids.)

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APPENDIX

BIBLIOGRAPHY OF STORAGE AND HANDLING OF CRYOGENIC FLUIDS IN SPACE

I Space Environment

II Zero-Gravity Effect

III Thermodynamic Properties

IV Heat Transfer

V Cryogenic Materials

VI Thermal Protection Techniques

VII Refrigeration Techniques

VIII Storage Techniques

IX Cryogenic Handling Techniques

X Cryogenic Tankage

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