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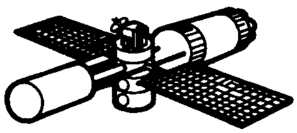
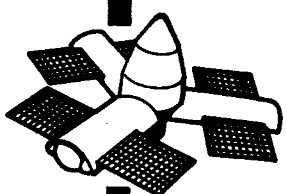
Vol. III

PRELIMINARY TECHNICAL DATA FOR EARTH ORBITING SPACE STATION



SYSTEMS

VOLUME III



NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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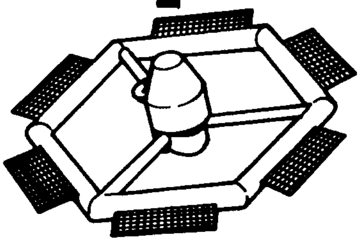

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PREFACE

This document, Volume III of IV, contains the Manned Spacecraft Center's technical data on Systems for the Earth Orbital Manned Space Station Study. The data is concerned with electrical power, environmental control/life support, instrumentation, communications, and cryogenic storage systems. A discussion of the system requirements for a Mars Mission is also included in some of the sections. This data is submitted in response to a NASA Headquarters' initiated study which includes requirements data from Langley Research Center, and experiment integration data from Marshall Space Flight Center. The complete integrated study will include the data from all three Centers.

The contributions of the various organizations within the Manned Spacecraft Center are acknowledged at the beginning of each section. Some of the data within these sections may differ slightly from the summary document since the summary presents the technical data in an integrated form. Any design philosophy presented in this volume represents the judgement of the contributing organization and has not necessarily been approved for the final study.

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PRELIMINARY TECHNICAL DATA

FOR EARTH ORBITING SPACE STATION

VOLUME III

SYSTEMS

SECTION 1.0

ELECTRICAL POWER

PROPULSION AND POWER DIVISION

ENGINEERING AND DEVELOPMENT DIRECTORATE

MANNED SPACECRAFT CENTER

1.0 ELECTRICAL POWER SYSTEM

1.1 GENERAL BACKGROUND

A critical constraint on manned spaceflight duration is adequate electrical power. The Apollo power generation subsystem, designed for a 14-day duration, can be extended to about 1 month when operated in a powered-down condition. Beyond a month duration, major modification and system redesign would be required. As manned spaceflight mission planning moves from the 2- to 4-week duration regime into the 2- to 5-year regime, energy requirements increase from the order of 10^2 kW-hr to the order of 10^5 kW-hr. Even with resupply of expendables and conversion units, chemical systems are prohibitively heavy. Solar and nuclear energy sources with their associated conversion devices remain to be traded-off and analyzed for optimum system selection.

An extensive effort has been initiated by NASA Headquarters to assist in further definition of this country's role in space exploration. As part of the Manned Spacecraft Center's participation in this definition effort, an electrical power system study has been requested. This study, parametric in nature, is in response to and in support of the above request. Since no firm spacecraft configurations have been established at the time of this study, the power systems selected for consideration are integrated in reference vehicle configurations to provide a basis for comparison.

In addition to a treatment of physical characteristics such as life, performance, weight, and volume, the study hopefully increases its usefulness by including consideration of areas such as cost, availability, and maximum utilization. Moreover, further objectives of the study are to point out specific areas that should be treated in a more detailed study and to define critical areas requiring early initiation of technology development programs.

1.2 GENERAL OBJECTIVES AND GUIDELINES

The general study objectives, in order of priority, were as follows:

a. Evolve and select power systems for the MSC-selected space station mission combinations.

b. Assess these systems for the MSC-selected Mars Flyby Interplanetary Mission; if necessary, evolve different systems and then evaluate their applicability to the the same space stations.

c. Ascertain use of precursor missions for flight tests.

The study guidelines for space station power were as follows:

a. Provide system capabilities to afford accomplishment of multiple mission objectives.

b. Maximum independence of earth support.

c. Simplicity of design.

d. No orbital assembly.

Within a, it was required that the mission objectives include: (1) conducting fundamental zero and partial gravity research, (2) conducting Earth-oriented remote sensing, (3) conducting astronomical scientific observations, and (4) supporting interplanetary research and development. Within b, the objectives were: (1) minimum logistics support; (2) accommodation of onboard emergencies, including functional facility failures; (3) subsystems redundancy and maintainability -- within reasonable limits; (4) onboard control of routine operations; and (5) logistics support consistent with crew duty cycles. Within c, the requirements were: (1) trade lower cost for higher weight and less sophistication, (2) use large design margins to maximize lifetime and reduce ground testing and documentation, and (3) minimize subsystem integration/interdependency. Guideline d contained no detailed requirements, but was considered sufficient within itself and with reason applied to any given situation.

No guidelines were given for the Mars Flyby Mission, but the guidelines given for the space station were considered to be sufficiently broad to include the necessary power system requirements.

Specific groundrules for the study are given in table 1-1. Figures 1-1 through 1-6 are the associated spacecraft configurations. As will be noted, mission modules -- called "cans" -- and basic 5 kWe power levels or combinations thereof are used wherever possible. This is, of course, to provide a basic independent building block for mission/configuration flexibility. Using various mission combinations, the four primary configurations cover the spectrum of candidate methods of system utilization.

TABLE 1-1.- MISSION/CONFIGURATION GROUNDRULES

	Configuration no. 1	Configuration no. 2	Configuration no. 3a	Configuration no. 3b	Configuration no. 4
Orbit	260 n. mi.	260 n. mi.	Sync	Sync	Mars flyby
Inclination	60°	60°	28.5°	0°	-----
Power	Hybrid or dual system; 5, 10, and 15 kWe per location. Locations at Hub and Bell.[b]	5, 10, 15, 20, 25, and 30 kWe at 5 and 10 kWe/can[b]	5 and 10 kWe at 5 and 10 kWe/can	Same	5, 10, and 15 kWe at 5 and 10 kWe/can[a]
Crew size	9 to 24 men	3 to 9 men/can	3 to 6 men		4 to 6 men
Resupply	4 to 6 months	4 to 6 months	4 to 6 months		-----
Gravity	Art-G, 0-G Hub	0-G	0-G		0-G, art-G
Orientation	Spin axis perpendicular to ecliptic plane, 60° inclination. Spin axis parallel to ecliptic plane, 60° inclination.	Gravity gradient. Sun oriented.[c]	Gravity gradient. Sun oriented.[c]		As needed
Life	2 years[d]	2 years[d]	2 years[d]		700 Days[d]
Launch	First quarter, 1973	First quarter, 1973	First quarter, 1973	First quarter, 1973	First quarter, 1975
Subsystem need date	First quarter, 1972	First quarter, 1972	First quarter, 1972	First quarter, 1972	First quarter, 1974

[a]Reactors may use conversion modules of an optimum size. Power level is 15 kWe.

[b]Reactors may use conversion modules of an optimum size. Power levels are 15, 20, 25, and 30 kWe.

[c]Control moment gyro.

[d]Discuss problems, costs, et cetera associated with extending system to 5 years.

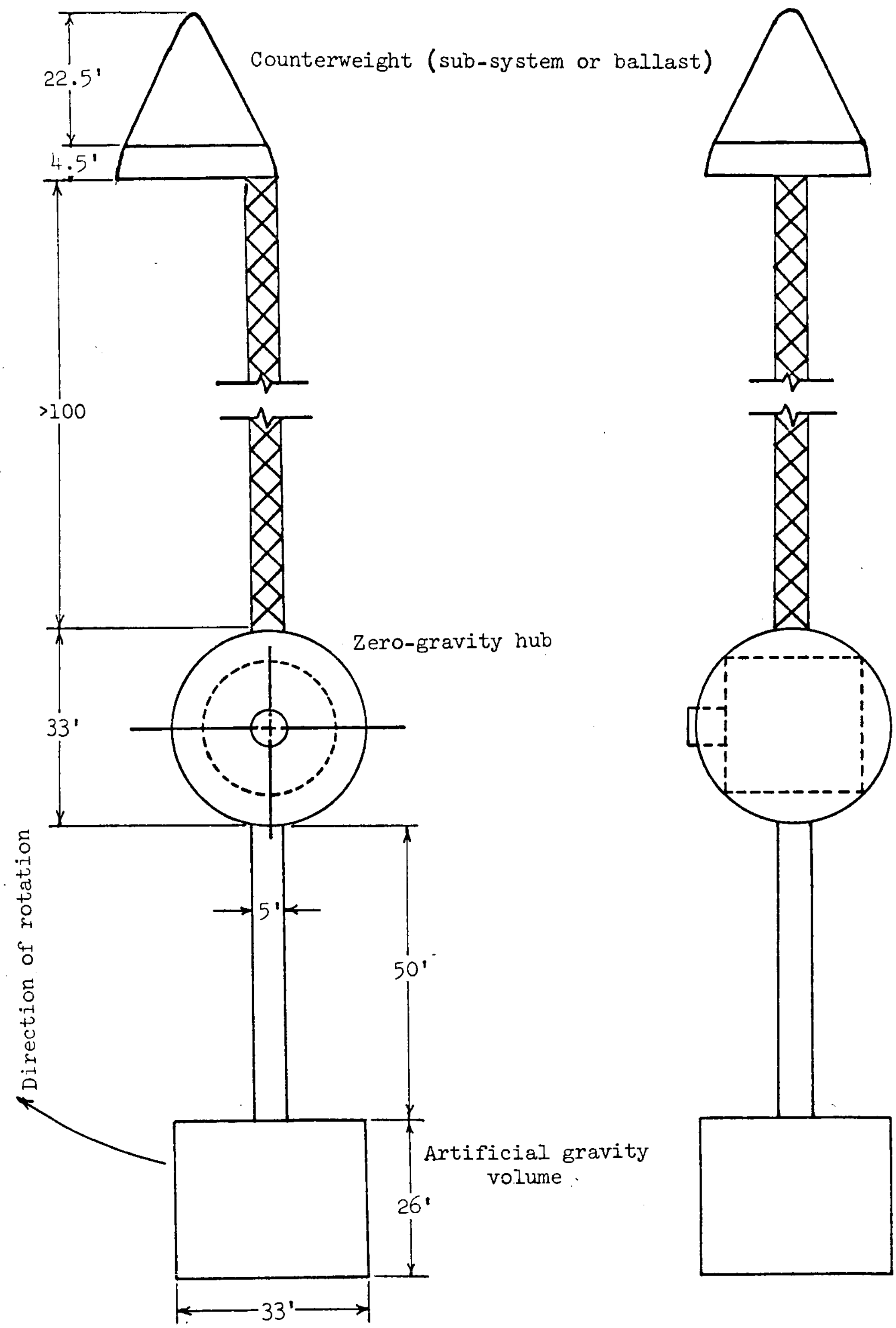

Figure 1-1 - Configuration no. 1, dumbbell space station, deployed.

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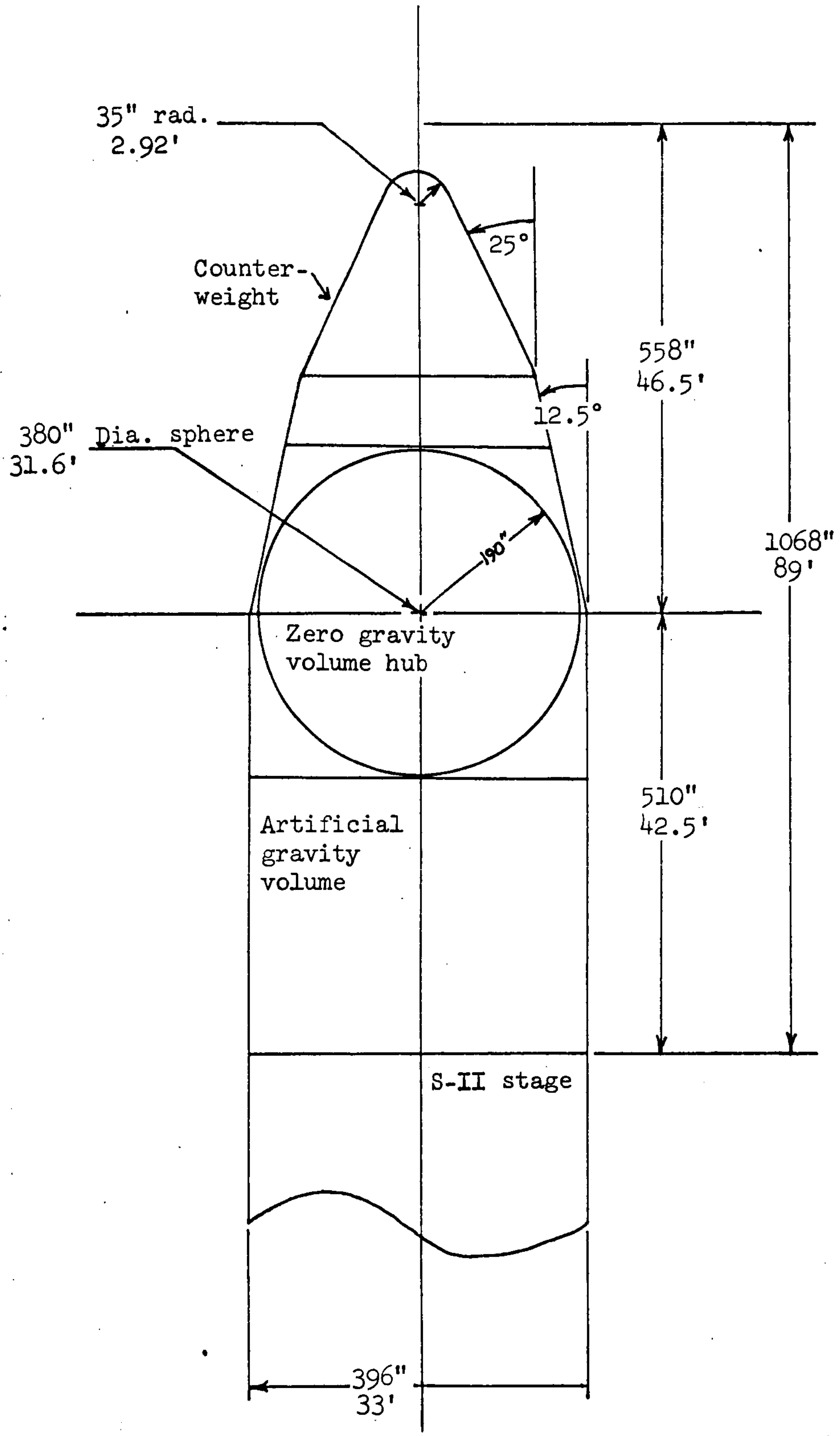

Figure 1-2 - Configuration no. 1, dumbbell space station, launch.

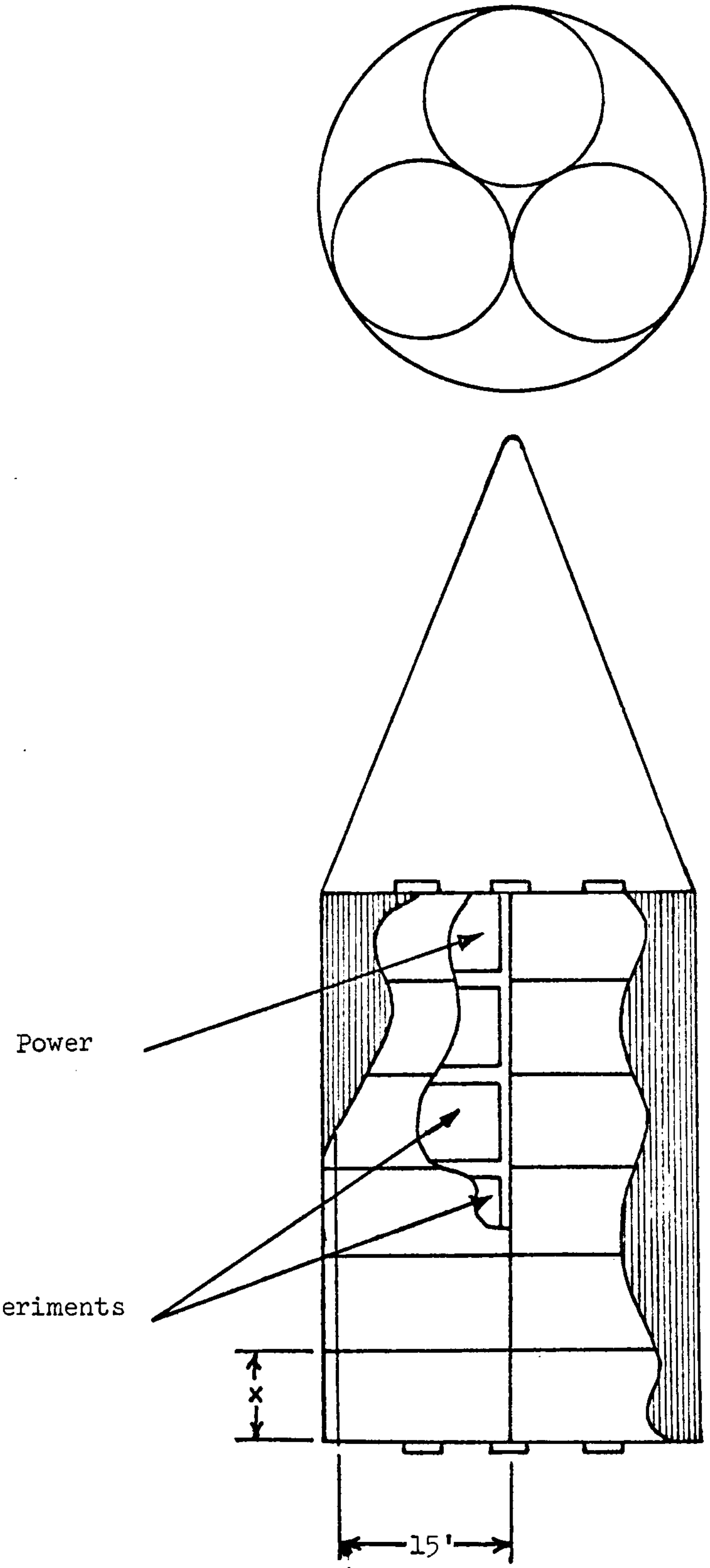

Figure 1-3 - Configuration no. 2, zero-g space station.

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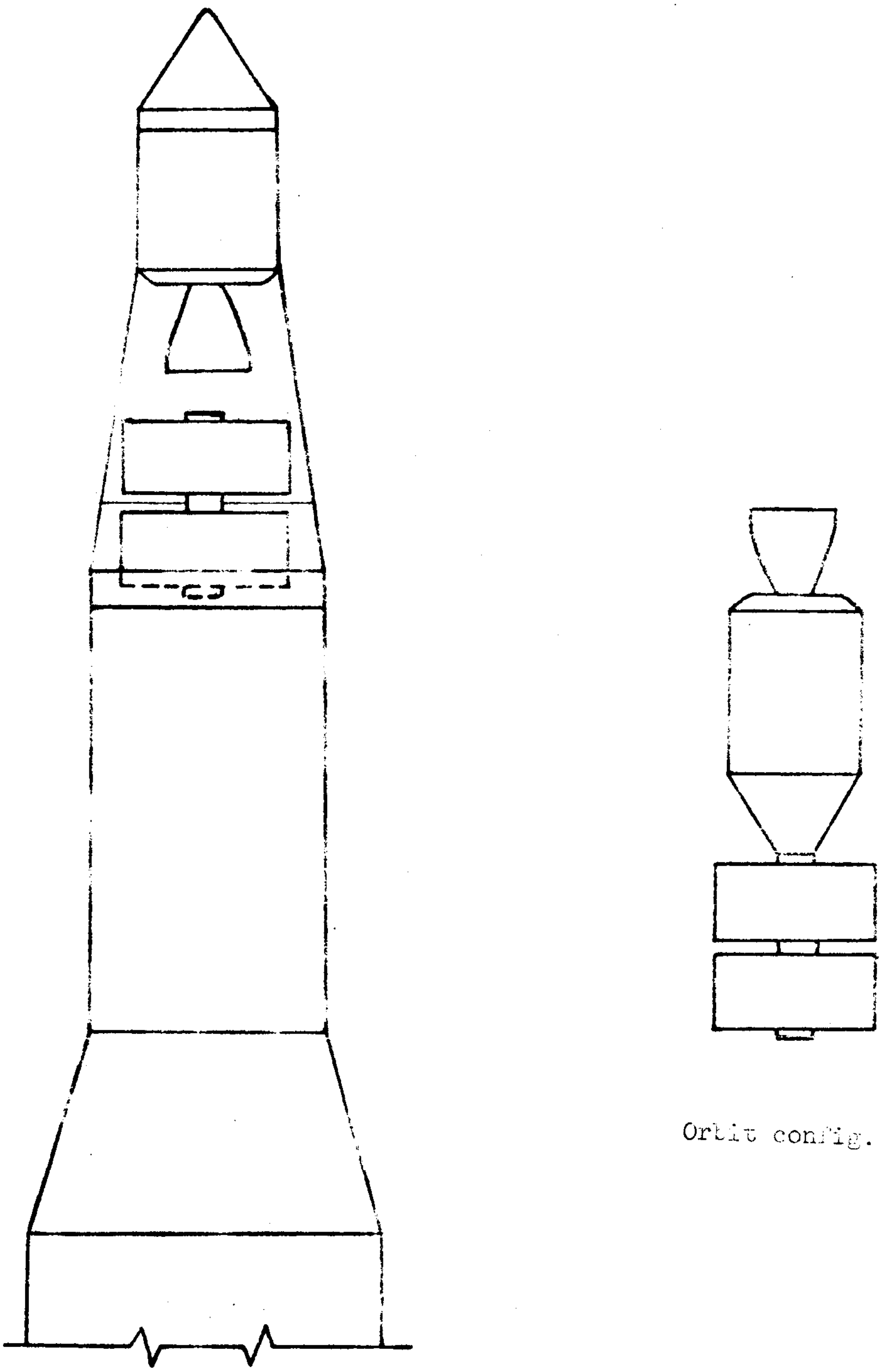

Figure 1.4. - Configuration no. 3, sync orbit space station.

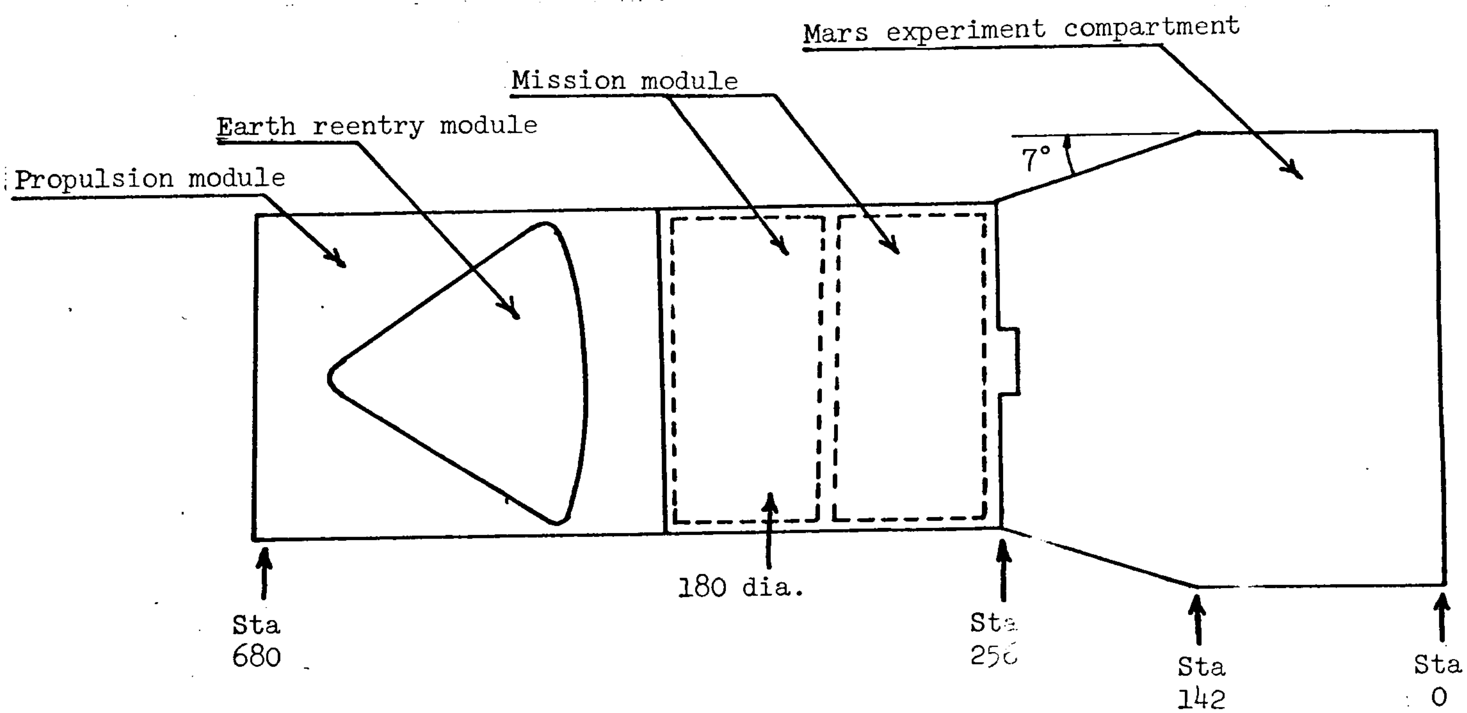

Figure 1-5. - Mars flyby spacecraft.

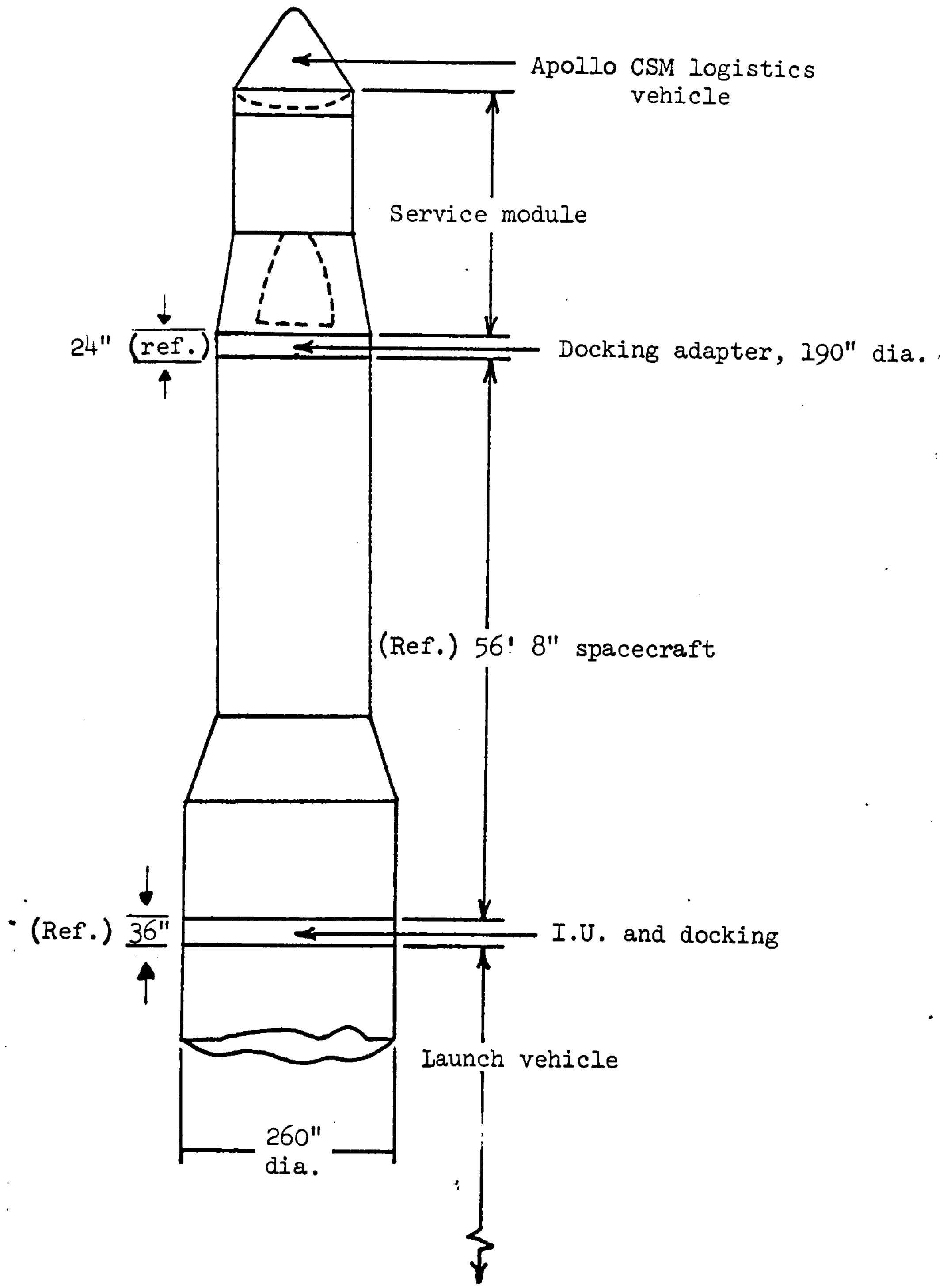

Figure 1-6 - Mars flyby spacecraft, launch.

1.3 BACKGROUND FOR DETAIL DATA

Due to the number of spacecraft configurations and the range of power requirements it was necessary to investigate candidate power systems on a parametric basis. The guidelines and groundrules dictated modularization and standardization wherever possible -- utilizing mission modules (cans), as given by the study configurations, for added mission flexibility. Because of the system need date constraints and to minimize technical risk and development cost, existing power system concepts were used.

1.4 SPECIFIC OBJECTIVES AND APPROACH

The specific objectives and approach were as follows:

a. Develop energy source and conversion device technology charts that show the attainable state-of-the art based on reasonable programs.

b. Assimilate these data and select the candidate system concepts for each mission/configuration.

c. Formulate the system concepts in detail.

d. Formulate schedules and costs.

e. Present development problems and advantages/disadvantages.

f. Assimilate all data and evaluate.

g. Select prime and alternate systems.

h. General:

(1) Technology used shall be based on the system launch-need date.

(2) Design philosophy: modularize where possible.

(3) Shield reasonably, but basically to man, not experiments.

The power requirements for this study were based on continuous basic power levels as determined by crew size, plus peaking power profiles representative of Earth orbital and Mars flyby mission duty cycles.

Figure 1-7 shows the reference continuous power levels as a function of crew size.

The logic for combining these continuous base levels with appropriate peaking profiles for the nominal 5, 10, and 15 kWe design points desired for modular system design is shown in figure 1.8. A mission average power level of 5 kWe typically consists of a 4.3 kWe minimum continuous load with sustained peaks up to 6 kWe and spikes up to 9 kWe. The continuous base and peak levels are also shown in figure 1.8 for the 10- and 15-kWe design points. Referring to figure 1.7 it may be seen that crew sizes of 3, 12, and 24 correspond to the 5, 10, and 15 kWe average power levels, respectively.

Figures 1.9 through 1.17 depict the various composite profiles used for the nominal 5 kWe design point.

The profile in figure 1.9 used for low altitude (260 n. mi.) Earth orbital system design, represents a duty cycle typical of Earth-sensing operations. The broad peaks above the 4.3 kWe continuous base level from hour zero through hour 23 represent housekeeping-type peaks, while the shorter pulses represent Earth sensor operation in duration and repeat interval.

Figure 1.10 shows the reference power profile for synchronous Earth orbit system design at the 5 kWe design point.

The next seven figures indicate the Mars flyby power profiles as the mission progresses through pre-encounter, encounter, and post-encounter phases. Encounter was assumed to occur 150 days into the flight. The short pulses in figures 1.12 and 1.16 represent high-power transmitter requirements at large Earth-vehicle distances. Not shown on these figures is the "storm cellar" power requirement, estimated to be 2.5 kWe for a total of 6 days. This requirement was considered in secondary systems tradeoffs, using independence from the primary EPS as a major groundrule for the storm cellar EPS.

1.5 STUDY IMPLEMENTATION

1.5.1 State-of-the-Art Assessment

1.5.1.1 Chemical energy storage nomenclature.- The terms primary and secondary have been used for years for both energy conversion and chemical energy storage systems. The terms generally are accepted as follows:

a. Primary energy conversion system -- supplies all basic power needs, including recharging energy for chemical energy storage if so required.

b. Secondary energy conversion system -- supplies all additional power needs such as when the primary system is not producing energy, for low power/long term peaks, and for high power/short term peaks. System may or may not require recharging.

c. Primary chemical energy storage -- normally used for one-time applications, or at most 5 to 6 charge/discharge cycles.

d. Secondary chemical energy storage -- has capability of being charged and discharged hundreds, even thousands, of times.

Because of the overlap of energy conversion system applications and nomenclature with those of chemical energy storage, the latter are hereinafter described as complementary and supplementary. The former is when chemical energy storage is used in lieu of the primary energy conversion system -- as Earth dark-side operation with solar cells or when systems are shut down for maintenance. The latter is when chemical energy storage is used for peaking power requirements. Complementary and supplementary systems for this study may be either fuel cells or batteries and be either rechargeable or unrechargeable.

1.5.1.2 BATTERIES AND FUEL CELLS

1.5.1.2.1 Batteries: The manned spacecraft applications under consideration herein require the use of supplementary and complementary rechargeable batteries only. These applications include two requirements in particular: (1) a supplementary battery subsystem to meet repeating and irregular peak loads and (2) a complementary battery subsystem to meet the dark-side loads with an electrical power system utilizing solar energy.

The specific types of batteries under consideration for the above applications are nickel-cadmium (Ni-Cd), silver oxide-cadmium (Ag-Cd), and silver oxide-zinc (Ag-Zn).

Ni-Cd batteries have been flown in the majority of the satellite applications to date. These batteries are especially adaptable to the long duration, high rate applications presented by the low earth orbit missions. Ni-Cd batteries have an energy density of nominally 10 watt-hours per pound and a nominal charge discharge watt-hour efficiency, if properly charged, of 75 percent. Battery temperature plays an important part in the amount of overcharge required, because efficiency is a function of temperature. At 120° F it may be difficult to restore full charge even with several hundred percent overcharge. At 60° F as little as 10 percent overcharge may be all that is necessary.

Figure 1.18 presents data taken on Ni-Cd batteries of various sizes and depths of discharge (ref. 1). The test conditions range from 55-min charge/35-min discharge (90-min orbit) at 75° F battery or cell temperature to 65-min charge/35-min discharge (100-min orbit) at 70° F battery or cell temperature for the illustrated depths of discharge. An additional example of Ni-Cd battery performance is evident in the TIROS 7 satellite launched June 19, 1963, and still transmitting. TIROS 7 has a period of 97.4 minutes and orbit of approximately 340 nautical miles (n. mi.). The depth of discharge is only about 3 percent, however, TIROS 8, launched December 21, 1963, in an approximately 390 n. mi. orbit, also is powered by solar cells and Ni-Cd batteries at 3 percent depth of discharge, and is still transmitting. Other satellites have operated for similar period durations utilizing Ni-Cd batteries at widely varying depths of discharge due to the particular orbits and load requirements of these satellites.

Ag-Cd batteries have an energy density of approximately 23 watt-hours per pound for vented cells, decreasing to about 10 to 15 watt-hours per pound for sealed cells depending on the rate of charge. These batteries also have a nominal watt-hour charge/discharge efficiency of 75 percent. The optimum temperature for Ag-Cd batteries has been shown to be around room temperature as far as performance is concerned. Lower temperature improves life, but degrades performance. The Ag-Cd battery will not accept charge as rapidly as will the Ni-Cd, which apparently makes the Ag-Cd better suited for the longer orbit applications. For faster charge rates, the plate area of this type of battery must be increased. Ag-Cd batteries have been flown in a number of satellites, primarily because of their nonmagnetic properties. These satellites include Explorers XII, XIV, XV, XVI, XXVI, IMP I, II, III, FR-1, and OGO. Most successful of these flights has been Explorer XXVI launched December 21, 1964 (perigee 135 n. mi.; apogee 13 980 n. mi.; period 449.7 min) and still transmitting. The depth of discharge varies widely with load requirements from 10 to 100 percent at less than 86° F. The battery has sustained at least one hundred 100 percent depth of discharge cycles. It is a 5 ampere-hour, 13-cell battery weighing 6.3 pounds. A ground test at the Naval Ammunition Depot at Crane, Indiana, has a similar battery on a 5-hour charge/1-hour discharge cycle at 40 percent depth of discharge. This test has lasted 2 years. The FR-1, a French satellite utilizing Ag-Cd batteries, may belie the long duration orbit stipulation mentioned previously, as may the data in figure 1.19 (ref. 1). The FR-1 was launched December 6, 1965, in a 405 n. mi. perigee, 410 n. mi. apogee, 99.9-min orbit. Ag-Cd batteries (at approximately 9 watt-hours per pound, sealed cells) were utilized for their nonmagnetic properties. The satellite is currently still transmitting. The depth of discharge is not known, but probably varies widely with load requirements. Figure 1-19 represents a compilation of data over the short-duration orbit regime. The data were taken on 65-min charge/35-min discharge cycles at 70° to 75° F battery temperature.

Ag-Zn batteries, as far as is known, have never been flown in space in a true rechargeable-battery application. Data from this type of battery are limited to ground testing, of which figure 1-20 (ref.1) is an example. The Boeing data were all taken over a 65-min charge/35-min discharge cycle at 70° F battery/cell temperature. The Inland-Delco data were taken on a 55-min charge/35-min discharge cycle at 75° F battery temperature. The energy density for a rechargeable Ag-Zn battery depends on the cycle life required, and varies from 15 to 50 watt-hours per pound. Watt-hour charge/discharge efficiency is approximately 75 percent. They may be discharged rapidly, but also have limited capability to accept high rates of charge and cannot tolerate high overcharge rates. As seen in figure 1-20, the reported cycle life of the Ag-Zn battery varies over a wide range with very similar test conditions, and so is somewhat inconclusive. Until further information is obtained, the Ag-Zn battery will not be considered further for this study. Work should be continued in this area, however, especially for the less-than-1000-cycle type applications, because this battery can offer a significant weight savings over the Ni-Cd and Ag-Cd battery subsystems.

The Quality Evaluation Laboratory of the U.S. Naval Ammunition Depot at Crane, Indiana, is performing a battery evaluation program for NASA under Contract W11,252B. A document entitled "Evaluation Program for Secondary Spacecraft Cells, Second Annual Report of Cycle Life Test" was published May 13, 1966, in which results of the test data were summarized in a Report Brief (ref. 2). This summary along with table B-1 and figures B-1(a) through B-1(g) are included in this study out of the body of the report, as appendix B, to provide supporting test data. The program covers a total of approximately 1525 cells and is still in progress. Results are in general agreement with the data discussed in this section.

1.5.1.2.2 Fuel cells: Fuel cell development has progressed to the point where a 400-hour life system is now qualified for the Apollo missions. The fuel cell system which will be used in the Apollo spacecraft consists of three fuel cell modules, or assemblies, each capable of supplying 563 to 1420 watts of direct-current electrical power at 29 ± 2 volts under normal operation. Each fuel cell assembly (FCA) weighs approximately 248 pounds with its mount assembly and stands 44 inches high by 22.5 inches in diameter. The normal operating temperature range of the system is 385° to 440° F. The system is designed to run continuously throughout the mission; that is, no provision is made for inflight start. Energy density based on the qualification test total

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energy is 1450 watt-hours per pound.[1] Efforts are now under way to extend the design life of the Apollo Bacon-type configuration by replacing the molten KOH electrolyte with a KOH/ceria matrix for the purpose of preventing dendrite growth within the cell which ultimately results in shorting. The presence of the ceria matrix, however, results in a performance decrease due to partial obstruction of ionic conduction paths within the electrolyte. To offset this performance decrease, an improved activation procedure is used on the hydrogen and oxygen electrodes. Potential energy density for this fuel cell is 10 000 kilowatt-hours per pound.[2]

Since fuel cell life is inversely proportional to temperature, an increase in life can be obtained by lowering the operating temperature of the cells, but this can only be done at the expense of a performance decrease. For some applications, this method may be desirable.

Other types of fuel cell systems are also being developed, one of which is the capillary asbestos matrix system. Since this fuel cell operates at a relatively lower temperature (200° F) than some of the others, its potential life capabilities are enhanced over those of other systems. Operating pressure and electrolyte concentration are also lower for this system than for other systems. Major problems associated with development of this system are controls problems such as those with the electrical monitoring and control subsystem and the water removal system. Potential energy density for this system is 12 500 kilowatt-hours per pound of inert fuel cell weight.

1.5.1.3 Solar cells.- Table 1-2 gives general solar cell technology as a function of calendar year. The efficiency extrapolations are based on normal production improvements and execution of present applied research plans of and by other Government agencies. Improvement of cell temperature capability is not planned and is not considered necessary.

Of the cell sizes considered, a cell size of 2 by 2 centimeters is preferred for several reasons, as follows:

a. A slightly higher packaging density can be achieved because the gap that is between two 1 by 2 centimeter cells is eliminated.

b. A savings in the purchase and installation cost can be realized. The 2 by 2 centimeter cells cost only 50 percent more than 1 by 2 centimeter cells, yet produce 100 percent more power. Installation costs will be lower since only half as many cells must be handled.

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[1] Based on fuel cell inert weight only.

[2] Based on fuel cell inert weight only.

c. The 3 by 3 centimeter cells are not yet available in large quantities and will probably not be available in the quantities required for at least 2 years.

Depending on availability trends, the wrap-around (also called back-connected) cell is preferred for the systems under consideration. Not only is assembly greatly simplified, but the wrap-around cells have 5 percent more active cell area than the present standard front-connected cells. The conventional cell of today is typically 0.013-inch thick. Figure 1.21 illustrates the relative performance of cells of various thicknesses. Cell efficiency decreases rapidly with cell thicknesses less than 0.012 inch. If weight is of primary importance, the power-to-weight ratio is a maximum at 0.008 inch. However, the 0.008-inch thick cells are not available in the quantities required and may not be for a number of years. Hence, the conventional 0.013-inch cell was selected for the solar cell system described herein.

The depth of this study does not permit final selection of a solar cell panel substrate material to be made. Most paddle-type solar cell arrays flown to date have used aluminum honeycomb of various thicknesses. However, the fiberglass honeycomb structures have demonstrated an improvement in strength-to-weight ratios. The thermal conductance properties are not as favorable as the aluminum honeycomb, although this is not considered to be a large problem. Therefore, the weight calculations performed in this study were all based on fiberglass honeycomb substrates.

The Boeing fiberglass-tape array substrate concept, although lightweight, is relatively fragile. The ability of this substrate to withstand the dynamic stresses of the launch and docking environment is questionable. Further investigation of this area is necessary before a final decision concerning the substrate material can be made.

The flexible, Teflon-impregnated substrate requires external supporting members, possesses good thermal characteristics, and requires no coatings for thermal control. However, the use of this substrate is limited to concepts that are unfurlable, due to its high flexibility and need for external support.

The thickness and, therefore, weight of cover glass used on solar cells depends primarily on the radiation environment to be encountered. The value given in the technology chart serves as an example of typical specific weight for a conventional cover glass thickness.

The rigid-panel deployment system technology is generally available. Flights such as the Pegasus satellite and certain classified Air Force flights have successfully deployed areas up to 1500 ft^2. With appropriate funding, the 1966 level of technology can be utilized as a basis, along with experimental ground and flight data, to provide a good engineering design of the rigid-panel deployment system along with substantiating hardware.

The Hughes unfurlable concept is presently being evaluated under Air Force contract. To date, the concept has not been flight-tested or qualified. However, present results indicate that the concept is feasible for certain missions where retractable arrays are necessary. Based on the present level of effort, sufficient engineering data for this system should be available to proceed with development of this concept on a large scale (greater than 1000 ft^2) by late 1967 or early 1968.

Orientation systems for large solar cell arrays have been designed, but have not, to date, been flown due to a lack of requirements. However, the orientation system design does not appear to present any major problems.

1.5.1.4 THERMOELECTRICS

1.5.1.4.1 General: Thermoelectric power generators are attractive devices due to their inherent reliability and simplicity. The attractiveness is impaired by the basic inefficiency of actual flight configured devices when compared to more conventional heat engines. All radioisotope fueled thermoelectric devices flown to date have utilized lead telluride, a moderate-temperature material. Since the first thermoelectric generator was flown in 1960, improvements have been sought in existing lead telluride couples, as well as advanced concepts.

Improvements in the lead telluride couples have been shown in systems where higher hot junction temperatures on the order of 1125° F have been used for long duration missions. The increase in operating temperature, while minimizing couple degradation, has allowed systems to be designed which have reasonable efficiency, about 5 percent, but still can reject heat at 400° F or higher. The net effect has been to reduce system radiator area with the same efficiency.

Advanced concepts emphasize efficiency gains with higher temperature materials. As shown in figure 1.22 silicon germanium is an attractive material with a capability of being used at temperatures as high as 1800° F. Additional gains are expected when segmented or cascaded modules of SiGe and PbTe are perfected. These advanced thermoelectric concepts were not given extensive consideration in this study. The utilization of these concepts is predicated on obtaining a successful development of a high temperature fuel capsule and other high temperature auxiliaries. The potential benefit of using these advanced concepts is higher efficiency; however, little information is available on usable components or couples which give assurance that these efficiency benefits are worth the system development risk.

In figure 1.22 is shown the obtainable efficiency at common average cold junction temperatures for candidate thermoelectric concepts. This figure shows that no efficiency gain is evident in going to elevated fuel clad temperatures (as high as 1800° F). When going to these elevated temperatures, the technological limit is not the thermoelectric materials. Considerable data exist showing thermoelectric couple and converter performance over long periods of time in various environments. The support technologies of fuel capsules, insulations, and liquid metals become the present limits. For the purpose of this study, two reference designs were considered: one lead telluride system utilizing only state-of-the-art technologies and the other based on the SiGe compact converter which requires extensive developments in the areas of (1) fuel capsule, (2) insulations, and (3) liquid metal loops and pumps.

1.5.1.4.2 Concepts: The SiGe compact converter is presently being developed by RCA for the AEC under Contract AT(30-1)-3582. The RCA compact converter submodule consists of a closely packed array of rectangular thermoelectric elements metallurgically bonded between hot and cold NaK containment channels. SiGe pellets are arranged in three rows of 12 on each side of the hot NaK channel for a total of 72 elements. The addition of a coolant channel on the cold side of each row of pellets results in a total of five layers for each converter submodule. The individual pellets in the three rows on each side of the hot channel are connected in parallel to minimize the effects of open circuit failure. Submodules so arranged can then be connected in series-parallel combinations to meet load requirements.

The tubular PbTe compact converter design is under development by Westinghouse for the AEC under Contract AT(30-1)-3584. Each thermoelectric module consists of a stack of n- and p-type PbTe washers designed to give a designated voltage, for example, 14 V at matched load, at specified operating conditions. Therefore, this converter differs from other designs both in geometry and in the high voltage obtained from the basic module. The initial system design uses converter modules that consist of four tubular modules hydraulically connected in parallel and electrically connected in a series and/or parallel arrangement to provide the required system voltage. The basic converter module is approximately 10 inches wide by 23 inches long by 2.5 inches high, and the converter module weight is about 210 pounds.

Although the pellet-type PbTe converter is not under active development in the mechanical arrangement required for this application, it does utilize existing thermoelectric technology. The preliminary analysis of such a converter was performed by the Westinghouse Aerospace Electrical Division (WAED) by using technology that has been applied to other systems. The configuration adopted for this study consists of flat rectangular arrays of couples, hermetically sealed, and sandwiched between NaK coolant channels.

1.5.1.4.3 Thermoelectric converter degradation: The principal failure mode of thermoelectric converters is not an abrupt failure, but degradation of material properties and/or interfacial resistances.

Although the PbTe tubular modules consist of many series elements, these elements have never been observed to suffer open circuit failure among the more than 3000 modules built to date. This performance is attributed to positive pressures that are maintained over relatively large contact areas. In other types of design, the paralleling of individual elements is also possible to minimize the effects of open circuit. The short circuit failure mode analysis of compact generators has not been completed, but the use of design features to prevent shorting appears feasible.

All the thermoelectric systems suffer some degree of degradation in output. The data for the predicted degradation of the compact and direct-radiating systems are presented in figure 1.23.

The data for the PbTe pellet compact converter are representative of tests that have been conducted with small systems for periods as long as 23 000 hours. Similarly, some of the SiGe direct-radiating material data are based on tests lasting about 20 000 hours.

Long-term operating data appropriate to the current module designs are not available for either the PbTe or the SiGe compact converters. The RCA prediction is based on data similar to that developed for the direct-radiating converter. The PbTe compact prediction is based primarily on existing low-temperature (1000° F) performance data with allowance for increased degradation at higher operating temperatures.

1.5.1.5 Closed cycle dynamic power conversion.- Closed cycle dynamic conversion space power systems of several different types and capabilities are currently receiving interest and support by NASA and other government agencies. This interest stems primarily from the vast amount of experience with rotating machinery for terrestrial application and from the potentially high specific power of these systems. Energy sources for these systems, primarily solar concentrators and nuclear sources, have also received considerable development support.

Table 1.3 lists the promising dynamic conversion system concepts investigated to date and shows an assessment of their applicability to the planned space station and Mars flyby missions. A brief discussion of each of the systems listed in table 1.3 follows.

The SNAP-2 system is a 3 kWe nuclear reactor-powered Rankine-cycle system employing mercury as the working fluid. Development of the system was initiated by the AEC in 1958. Atomics International is contractor for the reactor (SNAP-2 reactor) and overall system with Thompson-Ramo-Wooldridge as the subcontractor for the power conversion system. The objectives of the SNAP-2 program were to develop, qualify, and flight test a 3 kWe nuclear auxiliary power unit for space application.

The power conversion system development program has undergone several redirections since its beginning. In 1964 the program was reoriented from system development to component development. Since initiation of the program, more than 30 000 test hours have been accumulated on essentially six different turboalternator designs. Individual units have been operated for 2500 to over 4000 hours. The design power level of the latest turboalternator model (CRU V) is 4.1 kWe at about 6 percent efficiency.

The basic power conversion system is capable of operating in conjunction with a radioisotope heat source. The only major item which would require new design and development would be the mercury boiler. For an isotope system, the boiler would be constructed integral with the heat source, thereby eliminating a liquid metal loop and separate boiler as used with the SNAP-2 reactor system.

To achieve the high power levels of the missions under study, multiple SNAP-2 conversion units would have to be coupled to a reactor of high power output such as the SNAP-8 reactor.

The SNAP-8 system is a 35 kWe nuclear-reactor powered Rankine cycle system employing mercury as the working fluid. The SNAP-8 dynamic power generating system is similar to the SNAP-2 system, but employs an indirect heat rejection system (condenser heat exchanger) and an organic fluid bearing lubricate. The SNAP-2 system utilizes the system working fluid (mercury) as the bearing lubricate.

Aerojet General was awarded a NASA Lewis Research Center contract for development of the SNAP-8 power conversion system in 1959. Atomics International is the prime AEC contractor for the SNAP-8 reactor. The present design goal for the SNAP-8 system is to operate continuously for 10 000 hours at 35 kWe output. At present, the conversion system program at Aerojet-General is in the component development stage.

The SNAP-8 conversion system coupled with the SNAP-8 reactor would be suitable only for the highest power requirements -- 30 kWe -- specified for the missions under study. This would negate the possibility of modular approach to achieving the design power level, however. Also, because a rather conservative design approach was taken initially in the SNAP-8 program, the system is heavy -- about 7000 pounds per power conversion system. To achieve 2-year life reliability, redundant standby components and possibly loops would be required, resulting in a large, comparatively heavy and complex system.

Based on the above considerations, the SNAP-8 conversion system is ruled out for further investigation in this study.

In recent years, increasing interest and recognition have been directed to the closed Brayton cycle as an attractive power-conversion cycle for space electrical power systems. Considerable progress has been made in the experimental demonstration of the early theoretical component performance predictions and the design approaches required to adapt the Brayton cycle for use in space.

NASA Lewis Research Center (LeRC) initiated a program in 1963 to investigate the closed Brayton cycle for onboard space vehicle electrical power generation from solar, isotope, or nuclear reactor energy sources. The program evolved into several parts for component investigation and evaluation. The reference cycle arrangement employs a two-shaft system for adaptation to a solar energy source.

This year, LeRC initiated a new Brayton-cycle research and development program. This system is aimed specifically toward utilization with a radioisotope heat source. The conversion system utilizes a single-shaft combined rotating unit (CRU) — radial flow turbine and compressor and high speed alternator all on a common shaft. Based on the current LeRC schedule, the single-shaft Brayton rotating unit (BRU) should be available for system level evaluation testing in 1969. The BRU is being designed to produce power levels ranging from 2.25 to 10 kWe with the same CRU. The desired power level in this range is determined by loop pressure level.

To achieve long life capability, it is desirable that the Brayton-cycle rotating machinery utilize gas lubricated bearings. There may be problems in the design and manufacture of gas bearings because the technology has not yet reached a high level of engineering maturity. Considerable progress has been made in recent years, however, in the adaptation and utilization of gas bearings.

The closed Brayton-cycle system has potentially high system efficiency and long life capability. Research and development work conducted to date has been fairly successful. Turbine and compressor efficiencies of about 88 and 80 percent, respectively, have been demonstrated. The absence of corrosive and toxic fluids is also desirable. Material compatibility problems are almost nonexistent in the inert-gas Brayton-cycle system.

Although a considerable amount of development work remains to be done, the closed Brayton cycle is considered attractive for mission application study.

A notable amount of effort has been expended in the development of Rankine-cycle systems utilizing an organic working fluid. The chief merits of this concept are good performance at relatively low system temperatures and pressures and noncorrosive working fluids. The

Sundstrand Corporation under contract to the Navy and Air Force designed, fabricated, and tested a 1.5 kWe system utilizing Dowtherm A as the working fluid. Approximately 2000 hours of operating time was accumulated on a single turboalternator with total funding of about $500 000.

The major problem associated with the organic-Rankine system concept is the degrading characteristic of organic fluids when exposed for long duration to moderately high temperature and/or nuclear radiation. This degradation results in a loss of system performance and fouling of heat transfer surfaces.

Since the missions under study have durations of up to 2 years, it does not appear reasonable, based on the present knowledge of organic-Rankine systems, to further consider their application.

The dynamic power conversion system concepts considered applicable to the space station are the closed Brayton-cycle currently being developed by NASA-Lewis Research Center and the SNAP-2 mercury-Rankine cycle system currently being developed by the AEC. These systems are capable of operating in conjunction with either nuclear reactor or radioisotope heat sources. The closed Brayton-cycle system is capable of achieving overall system efficiencies[1] of 18 to 20 percent when coupled with a high temperature radioisotope heat source (1600° to 1800° F maximum fuel temperature). Overall system efficiencies of 13 to 15 percent would be realized with the SNAP-8 reactor heat source because of its lower temperature capability. The mercury-Rankine system is temperature limited by mercury corrosion of system components; therefore, the high temperature capability of radioisotope heat sources does not greatly benefit this system. Maximum overall system efficiencies are 5 to 7 percent when coupled with either a reactor or radioisotope heat source.

Based on the availability figures for the preferred radioisotope (Pu-238) (see section 1.5.1.6) and the efficiencies quoted above, the maximum attainable power level for the Brayton-cycle system would be 9 to 10 kWe, and for the mercury-Rankine system 2.5 to 3.5 kWe.

The radioisotope-mercury-Rankine system is therefore ruled out for space station application because of its inability to meet the minimum 5 kWe power module requirement. The radioisotope-Brayton system could be considered for use at the 5 kWe power level per location or up to 10 kWe in one location in a hybrid system.

The SNAP-2 mercury-Rankine system can be considered for use with the SNAP-8 reactor for power levels up to 20 kWe total (10 kWe per location).

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[1] Overall system efficiency = $\frac{\text{net conditioned power}}{\text{heat input}}$

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However, because of the relatively low power output capability and reliability of the present SNAP-2 turbomachinery, an excessive number of active and redundant standby power conversion systems (PCS) would be required to achieve high reliability and long operating life at the higher power levels. The reactor Brayton-cycle system can be considered for all power levels.

The list below summarizes the preceding information and delineates the dynamic system concepts for continued system study.

	Net Power Per Location, kWe			
	5	10	15	Remarks
1. Isotope-Brayton	X	X	-	Only one location can be served at 10 kWe
2. Reactor-Brayton	X	X	X	
3. Isotope-Hg Rankine	-	-	-	Low system efficiency, isotope availability
4. Reactor-Hg Rankine	X	X	-	System complexity rules out 15 kWe per location

Mars flyby application concepts - The Pu-238 isotope availability will have improved somewhat by 1974. With the available quantity of isotope, however, the mercury-Rankine system could provide only 5 to 7 kWe. The isotope Brayton-cycle system could be capable of producing the required 5, 10, and 15 kWe power levels.

Reactor systems utilizing either the Brayton- or mercury-Rankine conversion systems can be considered for the Mars Flyby Mission.

1.5.1.6 Radioisotope heat sources.- This section discusses various isotope systems under various stages of development and selects the isotope for further study. Details of the state-of-the-art of fuel capsules are deferred until section 1.5.3.4 for convenience.

1.5.1.6.1 Radioisotope power systems: Table 1.4 outlines various radioisotope SNAP power systems that have been developed or are at various stages of development or study. The largest such device flown thus far is the 25-watt(e), plutonium-fueled SNAP-9A. The plutonium-fueled SNAP-27 being developed for an Apollo auxiliary power requirement is to produce 50 watts(e) at the end of 1 year's operation.

NASA and the AEC have funded numerous design studies for multikilowatt isotope power systems using mercury-Rankine, organic-Rankine, Brayton cycle, and thermoelectric conversion devices. Hardware development of such isotope systems has not yet evolved. These studies did, however, conclude that such systems are feasible -- particularly at power levels of 10 kWe or less.

1.5.1.6.2 Radioisotope selection: Application of isotopes as heat sources for space power systems is affected by isotope half-life, required weight of heat source and shield, isotope availability and cost, and nuclear safety considerations. The characteristics of isotopes generally considered for space power applications are given in table 1.5. The following conclusions are made based on reference mission requirements:

a. For space power applications in the early to mid-1970's, promethium cannot be considered because of non-availability.

b. Curium-242 is eliminated because it has no apparent advantages over the more available and less expensive polonium-210.

c. Cobalt, strontium, cesium, cerium, uranium, and curium-244 are eliminated because of heavy shielding requirements (relative to other isotopes).

d. Thulium-170 offers no advantages over polonium-210. It has a shorter half-life and a lower power density and is therefore not considered further. Another thulium isotope, thulium-171, has many advantages but cannot be readily produced. A very high reactor flux, 10^{16} neutrons/cm^2/sec, is required for reasonable irradiation times; such "reactor space" is so severly limited that this isotope also cannot be considered further.

e. Polonium-210 is an α-emitting isotope with a half-life of 138.4 days; consequently, it has a high theoretical power density. This isotope can be considered only for short missions (~100 days) or for missions with comparably short resupply periods. Even so, when 30-day hold plus 10-day miscellaneous contingency is accounted for, this requires that the initially-encapsulated source be twice the activity or thermal power as required at end-of-life. Hence, the following establishes the pre-launch requirements of a 5 kWe output for different conversion efficiency systems:

Net conversion efficiency, percent	Effective thermal power requirements, kWt	Cost, \$, 20/thermal watt
5	200	4×10^6
10	100	2×10^6
15	67	1.3×10^6

The potential availability figures could support up to two, 200 kWt sources per year. However, for the purposes of this study, the logistics difficulties associated with a short-lived isotope make its use impractical.

Based on the above discussion, plutonium-238 (Pu-238) is the only isotope selected for further study. Discussion of Pu-238 follows.

Pu-238 is produced by reactor irradiation of uranium-235 via neutron capture and beta decay of neptunium-237. Neptunium-237 is separated from the uranium and further irradiated via neutron capture to neptunium-238. The beta decay of neptunium-238 produces Pu-238.

Pu-238 is a difficult isotope to produce and is therefore limited in availability. In the irradiation process, uranium-235 (U-235) undergoes neutron capture to uranium-236 in competition with fission -- the fission process being approximately five times more probable. Because U-235 is irradiated primarily for the fission process production of energy, the production of Pu-238 depends solely on the amount of U-235 which is irradiated for commercial and military uses.

Plutonium dioxide ($Pu^{238}O_2$) is presently the recommended fuel form for mission application. Plutonium dioxide emits alpha particles, beta particles, gamma rays, and neutrons; however, only the gamma rays and neutrons require shielding.

The only appreciable source of neutrons results from the alpha-neutron reaction with oxygen in $Pu^{238}O_2$. These (α,n) neutrons constitute about 91 percent of the total neutron activity, the remainder being produced primarily by spontaneous fission of Pu-238. Measurement of the as-produced, total neutron activity of $Pu^{238}O_2$ gives approximately 2.1×10^4 neutrons/sec-g Pu-238. This source strength, however, does not include induced fissioning.

Under the condition of subcriticality, the subcritical multiplication factor, or the ratio of neutrons emitted in a multiplying medium to the number of neutrons originally present in the medium, is given by

$$\frac{n}{n_o} = \frac{1}{1 - K} \qquad (K < 1)$$

where K is the neutron multiplication factor of the medium. Thus, for shielding calculations, the neutron source strength is n rather than n_o, where n_o is 2.1×10^4 neutrons/sec-g $Pu^{238}O_2$. As discussed below, K is a function of material and geometric properties; therefore, the resultant neutron activity for a Pu-238 source is a function of heat source design. Critical masses of Pu-238 under various configurations have been studied extensively; however, there remains disagreement because of the limited cross-section data.

Pu-238 has been encapsulated and tested by Mound Laboratory. However, before Pu-238 is used routinely in manned space applications, additional data are needed on fuel forms, capsule and heat exchanger materials, shielding, and nuclear safety requirements. For example, complete physical engineering data, together with data on chemical kinetics, sea water and soil reactions, and dissociation and stoichiometric effects resulting from time and temperature are needed for the radioisotope.

The mission groundrules require mission readiness of the heat source approximately 6 months prior to launch. For the 1975 mission -- because of the assumed 6- to 9-months required for processing and encapsulation -- only isotope production through FY74 can be relied upon to produce all the isotope required. Based on current requirements for SNAP-19 and SNAP-27 plus other inevitable Pu-238 requirements prior to 1975, a limit of 100 kWt per mission heat source was established for this application. Using the same rationale, a limit of 50 kWt is applied for a 1973 application.

In figure 1.24, required isotope thermal power and isotope cost versus conversion system efficiency are plotted for various power levels. This figure establishes the maximum electrical power level attainable for the 1973 and 1975 missions for thermoelectric, mercury-Rankine, and Brayton-cycle conversion systems.

1.5.1.7 NUCLEAR REACTORS

1.5.1.7.1 Hydride reactors considered: The SNAP-8 is the largest (600 thermal kW) of a hydride fuel element class of SNAP reactors which are currently under development by the AEC. Other, lower power, reactors in this class are the SNAP-10A/2 and SNAP-10B. A reactor power system is generally considered uncompetitive for manned mission application below 10 electrical kW, due to weight and volume constraints. The SNAP-8 reactor, which can provide 10 or more electrical kilowatts, is the most promising of the hydride reactors for mission applications in the early to mid-1970's. (See table 1.6 for a summary of SNAP reactor power systems currently under development or study.)

The SNAP-8 is conservatively designed to produce 600 thermal kilowatts with a 1300° F NaK outlet temperature for 10 000 hours. Typical reactor core life, temperature, and power operational capability for the current SNAP-8 design are discussed in section 1.5.3.3.

SNAP zirconium-hydride reactor technology has already been demonstrated at the 1300° F level with the SNAP-8 Experimental Reactor (S8ER). The S8ER was a test of the reactor core and reflector, operated without a power conversion system. The core and reflector closely represented the SNAP-8 reference flight design at the time S8ER was built in 1962. Design data for S8ER are summarized in table 1.7. The S8ER deviates from the flight design in that it does not use the flight system automatic startup and control components. The feasibility of the flight-design control-drive system has been demonstrated in the SNAP-10A flight test.

The S8ER test program provided experimental verification of the reactor performance characteristics including:

a. Capability of sustained power operation. During 500 days (12 000 hours) of nuclear operation, S8ER accumulated 365 days of operation in the 400 to 600 thermal kilowatt range with a 1300° F NaK outlet temperature. The longest continuous run at power was 5000 hours, which at the time was a new record for uninterrupted operation of a power reactor in the United States. Subsequently, a SNAP-10A ground test system exceeded this record for continuous operation.

b. Static and dynamic stability. The reactor was inherently stable during both steady-state and transient operation over the entire power range of the nuclear system.

c. Capability of tolerating rapid changes in power level. During power coefficient measurements, the reactor power level was changed 100 kWt in 1 minute. This transient was repeated 115 times.

Examination of S8ER fuel elements subsequent to reactor shutdown disclosed that 167 of the 211 fuel elements had hoop stress cracks. The test, consequently, did not prove fuel element design adequacy. Design modifications to relieve stress buildup and hydrogen loss will be incorporated into fuel elements for the follow-on developmental reactors.

1.5.1.7.2 Other reactor concepts: In addition to the hydride class of thermal reactors, several fast reactor concepts are under study by the AEC. The most promising are:

a. The SNAP-50 reactor

b. The 710 reactor

These reactors are not funded for complete system development but are proof-of-principle experiments to ascertain concept technical feasibility. Briefly, the principal design characteristics of these reactors are:

SNAP-50 - The reactor has a design objective of 10 000 hours, uses liquid lithium as coolant, and has a coolant outlet temperature of 2000° F. Various reactor designs have been investigated in the 2- to 10-megawatt (thermal) range. Development work has centered on irradiation testing of fuels – principally uranium nitride and uranium carbide.

710 - The 710 reactor concept is similar to SNAP-50 in size and power level. Coolant would be an inert gas, rather than a high temperature liquid metal. Development work for this reactor has also centered on fuel irradiations – principally refractory-metal-clad, refractory-metal uranium dioxide fuel elements.

The above two reactors are in the preliminary study concept and fuels-criticality testing stage. Because of advanced materials technology required for reactor operation in the 2000° to 3500° F range, advanced manned mission programing cannot contemplate utilization -- or need -- of fast reactors (producing thousands of thermal kilowatts) for auxiliary electrical power until past the mid-1970 time period, and, more realistically, the 1980's.

The SNAP-8 class is therefore considered the only type of reactor applicable to the current study. The basic SNAP-8 would have to be modified, as discussed later, to increase reliability and core life, particularly at higher thermal power levels.

1.5.2 SELECTION OF STUDY CONCEPTS

Because of the many possible electrical power system concepts and combinations, it was necessary to assess the state-of-the-art in light of mission/configuration groundrules to eliminate further study of those concepts which were obviously not appropriate. The results are given in table 1.8 as systems selected for detailed study.

For all mission/configurations, radioisotope availability severely limits all conversion methods except solar cells. This is reflected by the maximum power availabilities given in the table, as based on conversion efficiency/radioisotope availability relationships. A reactor energy source can possibly be developed for the space station concepts (1, 2, and 3) but could definitely be made available for the Mars Flyby Mission. Appropriate supplementary and complementary batteries were to be used for Earth dark-side and all peaking requirements. Fuel cells were preliminarily considered both in place of and as hybrid with batteries, but were excluded from further study with primary power systems pending separate detailed evaluation of fuel cell and battery subsystem characteristics.

The reduction in the number of study concepts did not in itself eliminate evaluation of more than one approach per conversion method.

1.5.3 SYSTEM CONCEPT FORMULATION AND EVALUATION

1.5.3.1 BATTERY, FUEL CELL, AND POWER CONDITIONING SUBSYSTEMS.

1.5.3.1.1 General: A consideration of electrical power systems for Earth-orbital and interplanetary flyby missions must include an examination of secondary power requirements which cannot be reasonably met by the primary power systems. These requirements manifest themselves both as peaks too large to fall under the capabilities of the primary EPS and as dark-side power requirements in conjunction with Earth-orbital solar cell systems. The energy storage subsystems considered to meet these requirements were rechargeable batteries and 2500-hour life regenerative and non-regenerative fuel cells in 6-month resupply and 2-year operation Earth-orbital missions and a 700-day Mars flyby mission.

Two basic types of application were considered: dark-side Earth-orbital operation for complementary use with solar cells and supplementary (peaking) needs. The only complementary application is the 260 n. mi. Earth-orbital mission utilizing solar cells for power during the light period of 58.5 minutes and rechargeable batteries for the dark period of 35.8 minutes. The synchronous Earth-orbit mission with a 22 hour 50 minute light time/70 minute dark time was considered a supplementary application because of the fewer cycles and longer charge time available.

The various combinations of batteries and fuel cells evaluated are given in table 1.9. Table 1.10 shows the secondary energy requirements, available energy for charging, number of cycles, power levels, and mission duty cycles for each of the several missions under consideration.

1.5.3.1.2 Battery subsystems: Unrechargeable batteries were not extensively considered due to the power requirements outlined by the power profiles; therefore, some of the more advanced energy storage systems such as the dry tape battery concept and the Li-CuF_2 cell are not discussed. Some information pertaining to these systems may be found in the figure preprints to the Space Power Systems Advanced Technology Conference held at Lewis Research Center on August 23 and 24, 1966.

Ni-Cd batteries are tentatively chosen to perform the 2-year low Earth-orbital mission (LEO), primarily because of cycle life and cycle frequency requirements. It should be understood that the selection of Ni-Cd over Ag-Cd was based to a large extent on a lack of information on large Ag-Cd subsystem capability. Work now in progress should be continued to ascertain and develop this capability. This recommendation is based on the potential weight savings of approximately 50 percent at the same depth of discharge which could be made by a Ag-Cd system over a Ni-Cd system. However, because of the fewer number of cycles and the lower frequency at which they occur, supplemental requirements and the 6-month resupply LEO solar cell application will be considered to be met by the Ag-Cd system. For the majority of the applications, Ag-Zn subsystems were not considered because of the high cycle life/mission duration requirements. It is not felt that these subsystems could be developed to the extent necessary in the time required.

The percent depths of discharge for each application were chosen based on the nominal consensus of the life test data to date.

The volume calculations are based on 0.725 ft^3/kWh for Ni-Cd systems and 0.38 ft^3/kWh for Ag-Cd systems. The radiator requirements (maximum) are presented in table 1.11, for each mission, both with solar and nuclear primary electrical power systems. These maximum radiator characteristics are based on continuous rejection at the maximum thermal rejection levels produced by all batteries, charge/discharge controller, and all power conditioning. These maximum levels are not constant, but are transient, and so should be considered extreme case values for area and weight characteristics.

For each mission application as presented by the power profiles, the total energy requirement to the loads was calculated. The energy available for charging the batteries was then calculated based on a primary EPS design level of 5 kWe net for the basic module. From this was calculated the "usable" energy to the battery which includes the inefficiencies of the battery and charge/discharge controller. Component efficiencies are given as follows: battery charge/discharge efficiency = 0.75, charge/discharge controller efficiency = 0.9, power-conditioning efficiency = 0.75 (assumes a 50/50 ac/dc power split and approximately 95 percent distribution efficiency). An absolute scale of energy was then determined for the peaking power profile by balancing energy content; that is, -, energy out of the battery and, +, energy into the battery. The absolute position on this energy scale was calculated after each charge and discharge period. The battery was sized by the worst case; that is, the point at which the - energy content, or discharged energy, is at a maximum. This energy value in kWh was made equivalent to the lowest state of discharge allowed in the battery. It is noted that if available energy from the existing profiles was not sufficient to recharge the batteries with the various inefficiencies considered, an additional increment of power was added to the basic capability of the primary EPS. This added increment, if found necessary, was calculated such that its addition balanced out the energy output by the battery including inefficiencies. The previously-mentioned energy content calculation made due consideration of the energy contribution both to the battery charging and feeding the load during battery discharging. Considering table 1.12 which summarizes the battery subsystems for all missions considered, the battery capacities shown and the percent depths of discharge shown are based on the above procedure which inherently produces the worst case point, so far as battery output capability is concerned, and so does not give a true indication of the battery's required performance level over the entire mission duration. If a cursory examination is made of each power profile, an apparent variation in the demands made on a supplementary battery subsystem is evident. This is especially in the solar EPS Mars Mission Profile, where maximum demands are made on the battery from day 281 to day 450 – some 170 days out of 700 days total. Before and after this 170-day period, the peaking requirements barely tax the battery's capabilities. This basic approach adds to the conservatism of the power system design approach; however, it is an approach which is inherently required since a battery is an energy storage device.

In order to compare equitably a fuel cell and battery subsystem for the same application, the following procedure for a battery was adopted: (1) no redundancy was assumed; (2) a charge/discharge controller weight penalty is added for the battery system; (3) if an additional increment of power for charging is required by the batteries, a weight penalty is assessed for this increment. For solar cells, the penalty is 0.38 lb/W for Mars missions, 0.18 lb/W for low Earth orbit, and 0.124 lb/W for synchronous Earth orbit; radioisotope systems, 1 lb/W; and reactor systems, 0.5 lb/W. No penalty is assessed for utilizing energy available within the capability of the basic EPS. These weights were calculated for each mission and plotted versus net mission power in figure 1.25 through 1.28. These "system" weights with their weight penalties included may be analyzed directly by comparing similar fuel cell plots, as may the volume data in figure 1.29.

1.5.3.1.3 Fuel cell subsystems: The approach to designing and evaluating fuel cell subsystems is essentially the same as for batteries. The specific groundrules and assumptions used are as follows:

WEIGHT:

a. Since any fuel cell subsystem considered will be used in conjunction with some primary system (other than fuel cells), only the weight of the fuel cell subsystem will be presented.

b. The following reactant tankage weights are used:

H_2: 2.5 (lb tank + controls) per lb usable H_2

O_2: 0.3 (lb tank + controls) per lb usable O_2

or 0.545 (lb tank + controls) per lb usable reactant ($H_2 + O_2$) (reactants for purging are neglected)

c. Fuel cell reactant consumption:

0.8 lb ($H_2 + O_2$) per kWh

d. EPS distribution efficiency: 80 percent

e. Charge (reactant regeneration) efficiency for a regenerative fuel cell: 50 percent; that is, for every 1 kWh gross furnished by the regenerative fuel cell, it takes 2 kWh supplied to the regeneration device to regenerate the water produced to hydrogen and oxygen.

A controller efficiency (for the regeneration system) of 90 percent is used.

Hence, if the regenerative fuel cell supplies 1 kWh gross to the spacecraft load, then the energy required of a primary system for purposes of regenerating the reactants is:

$$\frac{1 \text{ kWh}}{0.5 \times 0.9} = 2.22 \text{ kWh}$$

f. In the case of a non-regenerative fuel cell subsystem, the water produced by the fuel cell is used on the spacecraft and is therefore not charged to system weight.

g. A fuel cell life (hot time) of 2500 hours is used for this study.

h. Fuel cell fixed weight:

(1) Non-regenerative FCA[1] = 200 lb

(2) For a regenerative fuel cell subsystem, a fixed-weight penalty of 100 lb was added for three small tanks (H_2, O_2, and H_2O), controls, mounts and compressor. It is assumed that one equipment package of this type can handle the requirements of all FCA's on load at any given time.

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[1]FCA = Fuel Cell Assembly

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(3) Since the weight penalties associated with thermal control of the fuel cell subsystems are only a very small part of the total system weight for most cases, they are not included in the weight calculations.

i. Fuel cell parasitic power: 100 watts dc/FCA

j. In the case of the regenerative fuel cell used in conjunction with a solar or nuclear primary system, the weight penalties assessed for oversizing the primary system to provide recharge capability (power) or reactant regeneration are calculated as follows:

Solar Cell System Weight

Mars	0.38 lb/watt
Low E/O[1]	0.18 lb/watt
Sync E/O	0.125 lb/watt

Nuclear System Weight

Isotope	1.0 lb/watt
Reactor	0.5 lb/watt

VOLUME:

a. Use 12.9 ft^3/FCA + 5.1 ft^3 for small H_2 and O_2 tanks in a regenerative subsystem, + compressor.

b. Use 4.5 lb/ft^3 for H_2

c. Use 70 lb/ft^3 for O_2

$\longrightarrow \frac{62.7\ ft^3}{lb\ RCTNT}$

d. Assume no volume penalty for water produced in a non-regenerative configuration.

e. Assume no volume penalty for primary solar or nuclear systems.

1.5.3.1.3.1 Mars flyby - Figure 1.30 compares subsystem weights for fuel cell supplementary power subsystems used in conjunction with solar and nuclear primary systems, for regenerative and non-regenerative 2500-hour fuel cell modules. The lightest of these systems (curve A) is the supplementary regenerative fuel cell subsystem used in conjunction with a solar cell primary system.

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[1]E/O = Earth Orbit

Figure 1.31 gives weights for fuel cell subsystems (all non-regenerative) for the storm cellar (A), encounter experiments systems (B), midcourse power system (C), and for a single complementary power system (D) for (A), (B), and (C). Because fuel cell reactants are not chargeable to system weight due to the production of potable water, the single EPS (D) used with a solar cell primary system is only 40 pounds heavier than that of the storm cellar. For both the solar and nuclear cases, the integrated or single complementary system (D) is lighter than the combined weights of (A) plus (B) plus (C), at the expense of requiring more complex circuitry.

1.5.3.1.3.2 Earth orbit - In figure 1.32 are shown comparisons of various fuel cell complementary subsystems used with a solar cell primary system for low Earth-orbit missions. Both regenerative and non-regenerative fuel cell weights are given, for 6-month and 2-year resupply intervals. The regenerative subsystems are consistently lighter due to the absence of the large cryogenic tankage penalty associated with the non-regenerative subsystems. No complementary subsystems as such are required with a nuclear primary system.

Figure 1.33 shows the comparisons between various fuel cell supplementary subsystems used with nuclear and solar primary systems. The non-regenerative subsystems are considerably heavier than their comparable regenerative subsystems, the difference becoming extraordinarily high as the resupply interval is extended from 6 months to 2 years. The regenerative fuel/cell/reactor systems are slightly lighter than the regenerative fuel cell solar cell systems, the difference between them increasing as net mission power is increased. The regenerative fuel cell/radioisotope systems are heavier than the regenerative fuel cell/reactor systems due to the larger primary system weight penalty associated with the radioisotope system due to primary system power requirements between peaks for regeneration of reactants.

Figure 1.34 compares various fuel cell supplementary power subsystems used for synchronous Earth-orbit missions. Weights are given for regenerative and non-regenerative fuel cells, for both 6-month and 2-year resupply intervals. Due to the very large primary system weight penalty associated with regeneration, for the 6-month resupply interval, the non-regenerative fuel cell subsystem actually is lighter than the regenerative subsystem. Appendix C shows the detailed weight tabulations for all fuel cell configurations.

Total volume as a function of net mission power is shown for all configurations in figures 1.35 through 1.39.

1.5.3.1.4 Battery/fuel cell conclusions: A comparison of fuel cell and battery subsystem weights is shown in table 1.13 for a 5 kWe primary EPS. In general, these weight relationships will become even more favorable for batteries as net mission power and resupply intervals (in the case of Earth orbit) increase. The table shows approximate fuel cell weights relative to comparable battery subsystems. It can be seen that both the regenerative and non-regenerative fuel cell subsystems are considerably heavier than the battery subsystems for all missions. The regenerative fuel cell weight is competitive with the battery subsystem only for the low Earth-orbit mission peaking power system (6-month resupply interval, 5 kWe primary system).

Although not summarized in a table, evaluation of previously-shown data indicates that batteries have an even greater advantage over fuel cells in volume considerations. Further, preliminary cost analyses show batteries again to have significant advantages over fuel cells. Costs are examined in detail in section 1.5.5.2

1.5.3.1.5 Power conditioning: Electrical power conditioning was an area in which only a minor effort could be made due to the critical time constraints. The approach taken for this study was to base all weights and volumes on 5 kWe power system modules, with component redundancy not being considered per se. The weights and volumes were based on the general component values given in table 1.14. From this table, the values shown in figures 1.40 and 1.41 were calculated for the various systems. The change in slope at 15 kWe occurs because peaking requirements are constant thereafter. These values are considered to be -0, +50 percent, although final system values might be +150 percent if the redundancy philosophy so dictates. However, the power conditioning trend is valid.

A detailed study should be conducted to evolve a system application approach, namely, (1) type of redundancy, including number of units for reliability, basic power level (module) desired, derating factors required, et cetera; (2) optimum component use in the various power systems; and (3) component technology improvements that can be made with reasonable time and funds.

1.5.3.2 SOLAR CELL BATTERY SYSTEM

1.5.3.2.1 Design: Two solar array concepts were considered for this study. Concept "A" uses rigid panels deployed from a stack configuration by a scissors-link mechanism similar to the Pegasus satellite panel deployment system as shown in figure 1.42; concept "B" uses a flexible substrate for mounting the cells and is deployed as shown in figure 1.43 Table 1.15 presents a comparison of the two concepts. Concept "B" is lightweight, easily and compactly stowed, and can be retracted after deployment; however, its development status is not as advanced as concept "A." Hardware experience with concept "B" is practically nil and it has never been flight-tested. The retractability of "B" makes it a candidate for the Mars Flyby Mission, where midcourse and other maneuvers present potential shock-loading problems for conventional arrays. However, concept "B" was excluded from further consideration for the 1973 Earth-orbital missions because of the high risk involved in attempting to develop this concept by the launch need date of 1972.

Table 1.16 shows the anticipated radiation environment for the low-Earth orbital missions for 1968. These data are for omnidirectional fluxes and were provided by the Space Science Division of the Manned Spacecraft Center. Predictions for the year 1968 are presented since this year represents an anticipated peak in the solar cycle, and thus, worst case conditions. The damage threshold integrated fluxes for bare N-on-P silicon solar cells for equivalent 1 MeV electrons (e) and protons (p) are, respectively, 10^{11} e/cm^2 and 10^{13} p/cm^2. From table 1.16 the approximate integrated 2-year equivalent fluxes are 10^{12} e/cm^2 and 2.5×10^{10} p/cm^2. Although the protons present no anticipated problems, the electron fluxes are near the threshold damage value. For this reason, 0.006-inch thick cover glasses will be used for the system and are included in all low-Earth orbit calculations. Covers of this thickness, along with the substrate on which the cells are mounted, will provide more than adequate protection from the radiation environment.

Information from the Space Science Division indicates that the integrated spectra for synchronous missions will be less severe by approximately one order of magnitude. This factor has been substantiated by presently contracted effort out of the Propulsion and Power Division of the Manned Spacecraft Center (Contract NAS9-5266 with Radio Corporation of America). Therefore, 0.006-inch thick cover glasses will suffice for the synchronous orbit missions also and are included in the corresponding calculations.

The data available for the Mars Flyby Mission radiation environment from various sources do not agree, and are subject to many assumptions. The primary anticipated radiation spectra are solar-flare generated, and hence, are difficult to predict with much accuracy. The depth of this study does not permit detailed evaluation of the radiation spectra for this mission. A general survey of the available data indicates that cover glass of approximately 0.010-inch thickness will be sufficient for protection from solar flare events, and hence this thickness was used for the Mars Flyby Mission solar-cell subsystem calculations.

Basic cell stack weights were calculated as given in table 1.7. For the low-Earth orbit (LEO) missions - configurations nos. 1 and 2 - drag penalties were next calculated, as given by the 5 kWe example in table 1.18 Drag fuel penalties were based on a specific impulse (Isp) of 250 seconds, slightly lower than that of the Apollo Reaction Control System engines when operating in a continuous pulsing mode. This lower-than-optimum value was used because the Isp is lower when the engines are operated in a transient pulse mode, as will most likely be the actual case in flight. Drag was not calculated for the synchronous mission - configuration no. 3 - because past calculations have shown it to be negligible.

The next calculation was to determine gross array power required for supplementary batteries. These data are given in table 1.19. Array areas and actual power outputs versus load power were then calculated as given in table 1.20 and shown in figure 1.44 using nickel-cadmium batteries and basic efficiency and orbit times as shown. It should be noted that the solar-cell output in LEO is 9.33 watts/ft^2 compared to 10.59 watts/ft^2 in synchronous orbit. This LEO reduction is due to higher cell operating temperatures, as calculated in table 1.21 The next calculation was solar array subsystem weights, excluding batteries and power conditioning. These are given in table 1.22 The final array weights, excluding batteries and power conditioning, are given in table 1.23. Included are weights for systems with and without orientation subsystems. Although array orientation becomes more of a problem as inclination angle increases and/or gravity-gradient vehicle orientation is used, the worst case weights are used in all calculations because the weight advantage of less-stringent orientation requirements is of minimal consequence to total system weight. A two-degree-of-freedom orientation subsystem was included for all cases where the vehicle was not sun-oriented. For simplicity, this system was assumed to consist primarily of a four-quadrant solar aspect sensor which controls, through error-sensing logic circuitry, a set of servo motors which in turn drive the array to the required orientation. It was assumed that the arrays could be manually or automatically oriented during orbital darkness in such a way as to effectively eliminate or at least minimize the aerodynamic drag and thus conserve attitude control fuel during these nonoperating periods. Figure 1.45 shows array subsystem weight for LEO mission/configurations, where array orientation is and is not required. Figure 1.46 shows array subsystem weight for the synchronous mission/configuration for the same cases.

For the Mars Flyby Mission, table 1.24 gives basic solar-cell performance characteristics. A sample calculation is also shown in this table for clarity. The actual output power for a 5-kWe subsystem is shown in figure 1.46. The array must be designed for 5-kWe minimum power at 2.2. astronautical units (A.U.). At 2.2. A.U., effective solar intensity is approximately 3.5 times less than Earth orbit, with improved cell output considered due to lower cell temperatures. Thus, the array is essentially overdesigned as operated in Earth orbit.

However, the overdesign is not completely wasted, because direct cell output can be used in lieu of or to aid supplementary batteries for nearly the entire mission. Figure 1.48 shows array subsystem area versus load power. No extra area for battery charging is required. Figure 1.49 gives array subsystem weights versus load power for both the rigid and unfurlable arrays.

Figures 1.50, 1.51 and 1.52 show the solar array subsystem weights, battery weights (where applicable), power conditioning weights, and the total system weights for all mission/configurations that were considered. Figures 1.53 and 1.54 schematically show potential electrical subsystem for LEO, Sync, and Mars flyby missions.

1.5.3.2.2 Integration and operation: Figures 1.55 and 1.56 show the general array stowage configuration and deployment sequence. For those configurations where little or no space is available between the power module or "can" and the vehicle shroud, it was assumed that the array could be stowed in the outer volume occupied by the "can" itself, since reasonable volume is allotted for EPS utilization. Thus, the power "can" may be nothing more than a structure designed for mounting and stowage of the array and the battery/electronics package.

Array deployment is automatic in all cases. For Earth-orbital missions, deployment of the rigid array is accomplished by flat springs located at the panel hinges. Deployment rate is controlled by harmonically-driven servo motors which serve not only to control deployment rate but also provide a back-up deployment system. This type of deployment system has been proven on the Pegasus satellite.

Orientation is accomplished by servo drive motors which are controlled by inputs from a four-quadrant solar aspect sensor. It is anticipated that this system will be designed to maintain orientation within a predetermined range of accuracy, such as $\pm 15^\circ$. As was previously discussed, the arrays are oriented (manually or automatically) on the dark side in an attitude which generates the smallest aerodynamic drag forces on the array.

1.5.3.2.3 Advantages and disadvantages: Probably the greatest advantage of a solar cell/battery EPS is the wealth of flight data and associated experience available from the many unmanned satellites that have been placed in orbit. No other types of EPS have logged as much cumulative operating time in space. Solar cells are repeatedly used for electrical power on unmanned scientific satellites. For this reason, supporting applied research programs continue to be funded at high levels, thus bringing about significant system advances in the technology within relatively short periods of time. For the same reasons, the development time for such a system would be relatively short.

The solar cell is basically a simple device with no moving parts. As is the case with most static devices, it is intrinsically reliable. Since solar cells on large arrays are interconnected in matrix fashion (series-parallel) and thousands of cells are required per kWe, failure of an interconnect means failure of only a small part of the array. This method of interconnection yields a high partial-power reliability over long periods of operation.

The solar cell/battery EPS components contain, for all practical purposes, no hazardous (nuclear or toxic) materials, and hence are easily handled during assembly, transportation, checkout, and launch. This system also requires no fuel other than solar energy and therefore has low development and recurring costs. For the same reason, the system is safe in space operation -- no radiation -- and is relatively easily maintained, if required, in flight.

Solar arrays, by virtue of their large exposed areas, present vehicle "drag" problems in low-Earth orbits. A weight penalty must be assessed to account for the additional Reaction Control System (RCS) propellant required to maintain vehicle altitude. Also, since the arrays must be continually pointed toward the sun, the vehicle may be severely attitude-constrained if the arrays are not independently oriented. Another possible problem is the effect of moving the large panels with respect to the spacecraft. This motion may impart undesirable forces to the vehicle, thus requiring even more RCS propellant for altitude maintenance.

Another disadvantage of the solar cell system is that it must rely on the sun for energy and hence cannot provide power during orbital darkness. Therefore, some other energy source, normally batteries, must be available to provide dark-side power.

For interplanetary missions, another problem arises because of the variation of solar intensity over the mission. For a Mars flyby, the solar cell EPS would have to be designed to provide minimum mission power at 2.2 astronomical units; thus the array would produce approximately 3.5 times the required power in near-Earth space.

The advantages and disadvantages are summarized in table 1.25.

1.5.3.2.4 Photovoltaic/battery subsystem - areas requiring further investigation:

1. Radiation profiles and asteroid fluxes for all missions to determine true cover glass thicknesses.

2. Ability of unfurlable concept to withstand inflight vibrations and thermal transients of consequence.

3. Actual status and ability to project development of unfurlable concept.

4. Real hardware experience with unfurlable concept is severely limited. Concept has never flown. Development costs difficult to ascertain.

5. Tradeoff between solar-oriented vehicle/2-axis oriented array(s) (no spacecraft roll) /1-axis oriented array (spacecraft roll).

6. Spacecraft attitude definition to determine requirements of solar cell orientation system.

7. Midcourse and other acceleration loads while in flight (Mars flyby).

8. Power transmission sliprings may require significant development effort (for oriented arrays); thus, investigate alternate transmission approaches.

9. Detailed study of effects of asteroid environment for Mars Flyby Mission.

10. Tradeoff/optimization of photovoltaic array and battery weights for various power levels and missions.

1.5.3.3 Reactor design.- In addition to the basic design/operating information given in and required for the state-of-the-art assessment, uprated SNAP-8 reactor safety considerations and shield design and integration data are required for system design. These items are discussed herein.

As stated previously, the SNAP-8 is the only basic reactor concept to be considered for the 1970's. The final reference design development reactor, the S8DS, will be assembled in FY1967 and initial criticality is scheduled for the early FY1968 time period.

Although the S8DS is in hardware development, it is presently envisioned for unmanned mission use in the early 1970's. Man-rating this basic reactor concept would delay flight availability until 1972. Man-rating such a reactor would involve:

a. Increasing the number of fuel elements to extend core life at temperature and power level reliability. if it occurred before orbit,

b. Increasing control drum reliability.

c. Incorporate system operation monitoring provisions.

d. Incorporate provisions for automatic or manned shutdown and restart or control.

e. Demonstrate the required life (20 000 hours).

Reactor temperature versus power and power level versus core life are presented in figures 1.57 and 1.58 for the modified SNAP-8. From these figures, it is reasonable to expect 20 000 hours of continuous operation at the thermal power and temperature requirements of at least 30 kWe thermoelectric or Brayton-cycle systems.

Reactor safety considerations: Launch of a nuclear reactor involves a finite risk of inadvertent criticality which, if it occurred before orbit, could subject launch operations personnel or the general populace to a definite radiation hazard. Such a possibility is made quite low (10^{-6} or lower probability) by designing control safeguards which prevent control drum movement until the reactor is safely inserted into the proper orbit.

Once the reactor has operated for a few weeks and has built up a fission product inventory, a hazard exists for the case of inadvertent atmospheric reentry burnup. Such an accident would result in the dispersal of the radioactive fission products into the atmosphere. A normal mode of reactor "disposal" is to place the reactor — either during the operational phase or subsequent to final shutdown - into a sufficiently long-lived orbit so that in the event of atmospheric reentry the fission product will have decayed to a sufficiently low level.

The crew radiation hazard for in-space operations is controlled to prescribed levels through the use of radiation shielding. The shielding requirements for the thermoelectric and dynamic cycle systems are given parametrically in figures 1.59 and 1.60.

Shield design and integration: The following work on reactor shielding was performed by Atomics International for Douglas Aircraft under Contract NAS1-5547, and is reported in Douglas Report no. DAC 59213, May 1966. (See ref. 3.)

To meet the exposure limitations for men in a space vehicle, shielding must be provided to attenuate the radiation dosage from the reactor by about six orders of magnitude. The shielding design and integration analyses performed to define the most appropriate shields for this application have been concerned with (1) the shielding material selection problem, (2) generalized parametric analyses for shadow shields, (3) analysis of the scatter dose from a 154-inch-diameter cylindrical radiator projecting up to the plane of the bottom of the second shadow shield, and (4) a study to determine the dose rates normal to the reactor (outside the shadow-shielded zone), as well as directly below the second shadow shield after reactor shutdown. The results obtained are summarized in the following sections.

Material selection: Extensive experience in both the SNAP and ANP programs has shown that lithium hydride is an efficient fast neutron shield material with excellent thermal and radiation stability. Its melting point is sufficiently high to permit application in regions close to the reactor, where temperatures as high as 1000° F may occur. Additional properties that have led to the selection of natural lithium hydride are its low density (48.3 lb/ft^3) and its low decomposition pressure (25 mm Hg at its melting point of 1267° F).

However, when lithium hydride is used for neutron shields, it must be suitably contained to assure containment of the hydrogen. Experiments have shown that the hydrogen loss from uncontained lithium hydride at 800° to 950° F in a vacuum is sufficiently high to cause the loss of nearly all of the hydrogen during a 1000-hour period. Fortunately, several solutions to this problem are immediately apparent. One possibility would be to enclose this material in a casing thick enough to give a high meteoroid nonpuncture probability. A zoned casing to localize the shield degradation is another possibility. Tests have shown that type 316, 321, and 347 stainless steels are suitable canning materials for these neutron shields.

In the shields under consideration for this study, the gamma shielding is a very high percentage of the total shield weight. As a result, a depleted uranium alloy (U-8 Mo) is the recommended gamma shield material for both shadow and 4-π shields because of its superior gamma attenuation and low secondary-gamma production characteristics.

Shielding parametric analyses: In manned systems, the degree of attenuation required in the radiation exclusion zone is sufficiently high that radiation from reactor coolant in the primary coolant system heat exchanger(s) as well as that generated in the reactor must be considered. Such situations are best resolved by using a dual shadow shield that accomodates the primary coolant heat exchanger(s) in a gallery between the two shield assemblies. With this approach, the more intense reactor radiations are attenuated by two gamma and neutron shield assemblies, and the less intense primary coolant emissions by a single gamma and neutron shield assembly.

An analysis of this split-shield concept has shown that shield weight minimization is possible by proper apportionment of shielding materials between these two assemblies. Such optimum apportionments have been made in the shadow analyses that follows.

Shadow-shield analysis: The shadow shield configuration consists of an assembly of two depleted uranium alloy (U-8 Mo) gamma shields and two natural lithium hydride neutron shields enclosed in a stainless-steel casing. The shield thicknesses used in the generalized shield model derived from these configurations have been established by calculations based upon experimentally determined radiation attenuation characteristics for these two materials, a relative biological equivalent (RBE) for neutrons of five, and the minimum weight constraint discussed previously. Neutron shield thicknesses for an RBE of six would be about 1/2-inch thicker than those calculated in this analysis. This generalized shield model also assumes that there are no projections beyond the basic reactor-spacecraft envelope other than the neutron shield casing.

Analysis of this generalized shadow shield model considering all shielding aspects associated with the MORL has shown that shadow shield geometry and, in turn, shield weight are affected by the following set of primary variables:

a. Envelope diameter at the reactor core midplane.

b. Reactor fuel-element length.

c. Reactor core diameter.

d. Primary coolant system heat exchanger gallery height.

e. Reactor thermal output.

f. Reactor/dose-plane separation distance.

g. Dose-plane diameter.

h. Dose rate at the dose plane.

Moreover, these investigations have shown each of these design variables or parameters, with the exception of parameters f and g, to be largely independent and separable variables. When parameters f and g are considered jointly, however, sufficient linearity exists with the remaining parametric set to permit these to be considered independently, within reasonable limits of shield weight accuracy considered satisfactory for this study.

The reference shield design criteria utilized in this study are listed in figures 1.59 and 1.60. Parametric curves were developed by varying each primary design variable or parameter separately, or jointly, as the case may be, about this reference point. Shield weights were then determined as a function of each parameter. These variations were then normalized to the reference value to produce the set of curves as shown in figures 1.59 and 1.60 from which relative shield weight factors may be found when more than one primary design parameter is varied at one time.

1.5.3.4 Radioisotope design.- In addition to the basic Pu-238 characteristics given in the state-of-the-art assessment, Pu-238 heat source block and heat exchanger design data and nuclear safety criteria are required for system design. These items are discussed in the following sections.

Heat source block and heat exchanger design: The two isotope capsule clad design temperatures considered are 1400° F and 1800° F. The requirements and technology status of these two classes of heat sources will be discussed separately.

a. 1400° F - A 1400° F capsule clad temperature is considered the state-of-the-art of isotope capsules; this technology status is represented by the SNAP-27 radioisotope thermoelectric generator capsule. For these temperatures, the superalloys - for example, Haynes Alloy 25 - can be used as the capsule structural member to withstand helium pressure buildup and impact forces as well as for the corrosion/oxidation resistant coating. The structural and coating component thicknesses are designed, per nuclear safety requirements, to completely contain the radioisotope for about 10 half-lives (900 years for Pu-238). These basic requirements result in 10 to 12 pounds per thermal kilowatt for isotope capsules alone. Reentry protection has been shown to give a minimum weight if applied on a per capsule basis, particularly if a reentry protective coating can be designed to yield small ΔT's while in the operating configuration and still provide acceptable heat rejection. An example is using sublimation upon reentry to prevent overheating at the fuel centerline. Should the temperature drop through the reentry protective coating be excessive for operating conditions, the basic capsule clad temperature would have to be raised. This higher basic capsule temperature requirement (1500° to 1600° F) would require a refractory metal for the structural member with a superalloy oxidation-resistant clading superimposed.

The reentry-protected capsules are placed in the heat source block which is designed with temperature-activated louvers to provide inflight emergency cooling. The source block is coupled through a NAK flow loop to a counterflow heat exchanger which serves as the effective heat source for a thermoelectric energy conversion device. Other conversion devices may not require a NaK loop, but will have equivalent heat exahanger loops. All these components combined result in an effective weight for the heat source-heat exchanger-emergency heat rejection component combination with a specific weight of about 45 pounds per thermal kilowatt, which includes a configuration dependent factor of 10 to 12 pounds per kilowatt for radiological shielding to 3 to 5 mrem per hour at the dose plane with a 10 to 15 foot separation.

b. 1800° F - The design requirements of an 1800° F clad requires a significant advancement to the present state-of-the-art because of the need for refractory metal structural member and a high-temperature, oxidation-resistant outer clad. In order to meet the overall requirements of the heat source, the high temperature capsule becomes complicated as evidenced by the following discussion.

1. Primary container - For high temperatures, refractory metals possess the strength properties required to meet the capsule structural requirements in accommodating helium pressure buildup and impact resistance. The tantalum base alloy, T-222, can be used because it is commercially available and appears to have the required creep rupture and impact strength-to-weight ratio for an optimum capsule. Associated time-temperature-mechanical properties data to support the design requirements of the reference missions are virtually non-existent. Obtaining such data would constitute a major portion of the development effort for an 1800° F capsule.

2. Oxidation barrier - refractory materials are subject to extensive oxidation so that outer, oxidation-resistant barriers are required. Superalloys, such as Haynes 25 or Hastalloy X, are acceptable for low-temperature applications; however, a ductile-to-brittle transition

occurs after long-term, high-temperature exposure. This brittleness of the outer clad or barrier leaves the refractory material subject to exposure following Earth impact. Platinum or platinum rhodium appear to be the best choice for the outer capsule because of superior corrosion and oxidation resistance at high temperature.

3. Diffusion barriers - at the high temperatures considered, platinum would probably diffuse into the tantalum alloy primary capsule over a long period of time so that thoria or a high-temperature glass diffusion barrier is required between the inner and outer capsules. Also, T-222 contains hafnium which may be incompatible with the Pu-238 source. A tungsten liner would alleviate this problem.

4. High-emissivity coating - if the capsule is to be used in a radiant heat transfer heat exchanger as in the case of the Brayton-cycle, the platinum outer clad must be coated with a high-temperature, high-emissivity coating, such as iron titanate.

As in the 1400° F case, reentry heat protection would have to be provided for each capsule or for the entire heat source block. For a system requiring an intermediate heat exchanger to transfer heat to the power conversion loop, the design approach would be the same as for the 1400° F system. However, the 1800° F requirements again impose additional design constraints probably requiring a liquid metal other than NaK coupled with either stainless steel tubing lined for corrosion-erosion resistance or a more advanced tubing material.

To meet the design requirements, the heat source-heat exchanger combination exhibits a specific weight of about 60 pounds per kilowatt including about 10 to 12 pounds for radiological shielding per the aforementioned shielding criteria.

Nuclear safety (general): The Pu-238 O_2 microsphere fuel form has been selected as the most attractive fuel form for this isotope. The low volatility and solubility, as well as nonrespirable features of the microsphere fuel form, result in significant relaxation of containment requirements. Absolute containment is highly desirable but not mandatory for most cases. The most hazardous potential Pu-238 dispersal mode has been evaluated as high-temperature volatilization to the atmosphere. Two principle sources of such high temperature exist: internal heat generated by radioactive decay when the capsule is thermally isolated, as with soil burial or with loss of coolant, and atmospheric reentry heating without adequate protection.

Recovery of the fuel material at mission termination is obviously the most desirable plan from a safety viewpoint. If such recovery is not feasible, impact and permanent disposal in deep ocean areas is acceptable. With such disposal, long-term containment is not absolutely necessary. Reentry burnup is not desirable for large Pu-238 sources.

Fabrication, handling, transportation, and prelaunch integration of the heat source/fuel capsule will require considerable design and procedural safeguards, but the potential accident environment during these activities is much less severe than with mission activity phases.

The development of a heat source/fuel capsule design and overall operational plans which assure an adequate safety level have been studied extensively under several Government-sponsored efforts. Detailed study of safety factors associated with the use of large quantities of radioisotopes is beyond the scope of this study. It should be noted, however, that the safety program associated with development of a large isotope power system would constitute a major portion of the overall development effort. Investigation of many of the physical, biological, and chemical properties of the fuel form is still incomplete so that the proposed applications of these isotopes may suffer from imposing unduly restrictive safety requirements.

Items requiring further study: Before Pu-238 can be used routinely on manned missions, considerable development effort is required. Some areas requiring further investigation are discussed below.

a. Further investigation of many of the physical, biological, and chemical properties of the Pu-238 O_2 fuel form. Because of insufficient information, proposed applications of these sources may suffer unduly restrictive safety requirements.

b. Further investigation of subliming materials such as AlF_3 for reentry protection of individual fuel capsules; further study of the tradeoffs of protecting individual capsules versus putting entire heat block in reentry body.

c. Considerable testing to determine mechanical properties of superalloys and refractory alloys after long-term (>10 000 hour) exposures at high temperatures. Such data are needed to minimize capsule thickness overdesign and to determine the temperature at which it is necessary to change over from superalloys to refractory-based materials.

d. Further investigation of the feasibility of controlled intact reentry and subsequent recovery of multikilowatt isotope heat sources. Such an approach is desirable for Pu-238 sources from both safety and economic viewpoints.

e. Considerable effort must be spent in refractory metal design requirements for isotope capsules. Examples are machining, compatibility testing, developing of high temperature oxidation-resistant coatings, bonding techniques, et cetera.

f. Development of acceptable fuel form simulants should be pursued. If adequate fuel simulants are obtainable, costs and availability limitations for a Pu-238 program could be relaxed.

1.5.3.5 Thermoelectric systems.-

1.5.3.5.1 Pu-238 radioisotope thermoelectric module:

1.5.3.5.1.1 Design - Early work in the study was directed to configuring a radioisotope fueled thermoelectric power generating device capable of producing 5.0 kWe net power for 2 years. However, when isotope availability was considered, it became apparent that such a thermoelectric system would require more Pu-238 than would be available even as late as 1975. From the availability of isotope as discussed in the isotope section of this report, a limit per system in 1972-1973 of 50 thermal kilowatts was established. In 1974-1975, a 100 thermal kilowatt system can be considered. Using the above heat source constraints, a basic thermoelectric module was configured utilizing 50 thermal kilowatts.

The two systems considered in detail were the two compact convertor concepts under study by the Atomic Energy Commission. The systems configured herein are based on performance data obtained during early development work done in those programs. Use of either of these concepts will require extensive life qualification to verify performance after 2 or more years operation.

The lead telluride compact convertor was designed with a primary constraint of a NaK inlet temperature to the convertor of 1100° F. The parameters of the basic 2 kWe module as configured in this study are shown in figure 1.61 It must be noted that this basic module configuration can be optimized to better meet mission objectives, vehicles, or other constraints once they are formulated; however, this system is representative of a typical state-of-the-art concept. All support equipment required to meet the requirements of this system or any other lead telluride system designed for 2 year life can be designed, built, and qualified with minimum development.

The silicon germanium compact convertor module was designed to accept a NaK inlet temperature of 1500° F. The system parameters are summarized in figure 1.62. As in the discussion of the PbTe concept, this system has not been fully optimized. It should be noted that there is no efficiency gain experienced in a system using the SiGe over one using a PbTe module. The weights for the SiGe system will be significantly higher than for lower temperature systems and the development program would have higher risk. Chief among the questionable technological areas is that of the high-temperature fuel capsule and the high-temperature liquid-metal developments required to support such systems.

The primary heat rejection systems are configured for easy comparison. Both systems utilize a NaK loop which required the use of proven technology. The thermoelectric electromagnetic pump required in the radiator loop as in the PbTe conversion system, are based on well established design methods developed in the SNAP reactor programs. The loop configurations and list of system component comparisons are shown in figure 1.63.

1.5.3.5.1.2 Operation and integration - The radioisotope power conversion systems considered for these advanced missions have been shown incapable of meeting total mission prime power. They were not deleted from further study due to the potential usefulness of a basic 2.0 kWe module in a hybrid system, or as a source of emergency electrical power. A cursory examination of the various mission vehicles involved in the Earth orbit or interplanetary missions shows that radioisotope thermoelectric systems present no constraints on mission inflight operations. The only area of potential concern involves the integration of these devices with the spacecraft. The isotope availability limit imposed on Pu-238 systems considered in the study limits the thermoelectric systems to 4.0 kWe (net). At the 4.0 kWe level, this system will require approximately 600 ft^2 of radiator area. This radiator would be integrated with vehicle structure, either in a subsystem vehicle or with a living or experiment module. The heat source and conversion systems have not been considered as an internal part of living or laboratory module. The shield weights are based on a nominal 16-foot separation from the crew.

1.5.3.5.2 Thermoelectric conversion with nuclear reactor heat source:

1.5.3.5.2.1 Design - Based on the long lead times for developments in the field of nuclear reactors, it is evident that the only reactor which can be considered for use with a thermoelectric convertor by the mid-1970's is the SNAP-8 reactor. Meeting the requirements of up to a 30 Kwe thermoelectric system does not tax the projected capabilities (power, temperature, or life) of the planned manrated SNAP-8 reactor. The temperature limit assumed in this study is 1300° F (NaK out of reactor). This temperature limit on reactor outlet limits the thermoelectric modules to be considered to only the lead telluride family of convertors.

The temperature limitation of 1300° F excludes the silicon-germanium compact convertor from consideration. To develop a reactor capable of operating with a NaK outlet of 1500° F would require a long development and verification which would preclude its use for flight until the late 1970's or early 1980's. The lead telluride system modules which would be coupled with the reactor for use on a mission in the mid-1970's would be similar to those described in the radioisotope EPS discussion. The basic approach change from the radioisotope system module is to use a 2.5 kWe net basic module or some other convenient size. The maximum power capability of a system utilizing PbTe thermoelectrics and the SNAP-8 would be approximately 30 kWe. Characteristics of the reactor thermoelectric systems considered are shown in figure 1.64 and table 1.26.

1.5.3.5.2.2 Operation and integration - The reactor considered in these analyses is an improved SNAP-8 reactor. The normal design of the reactor includes automatic start-up, self-control, and fail-safe provisions. The improved version envisioned for use in these missions has added manual overrides to some of the automatic shut-downs.

The shield weights for the reactor thermoelectric systems were calculated based on a 22.5-foot dose plane diameter for interplanetary missions and a 50-foot separation distance. For the Earth orbital (Dumbbell Configuration), shields were based on a 60-foot dose plane diameter and a separation distance of 100 feet. The 60-foot dose plane diameter was used to give allowance for a shielded area for rendezvous and resupply operations.

The radiators used in these analyses are conical units which are integrated with the reactor, shield, and conversion package. The radiator areas were based on the use of coatings with an emissivity of 0.8 and an α/ϵ of 0.66. The radiator weights were based on the use of stainless steel tubes, with meteoroid protection, and aluminum fins. The specific weight of such radiators was assumed to be 2 pounds per square foot. Figures 1.65 and 1.66 show the two conceptual configurations of reactor thermoelectric systems.

1.5.3.5.3 General considerations: The radioisotope and reactor thermoelectric systems configured for this study reflect no attempt to optimize with respect to any parameter. Wide variations in the systems could be realized if this were to be done. If radiator area were to be a limiting factor, an optimum cold junction condition could be established which would minimize the radiator area. Should weight be the overriding consideration, the cold junction could be varied to optimize weight of radiator and heat source components to yield a minimum-weight system for a particular mission and vehicle. These types of optimizations require either a precise definition of the mission and vehicle involved or extensive parametric analyses which were beyond the scope of this study.

Figure 1.67 shows schematically the basic electrical configuration upon which electrical-to-electrical efficiencies for all thermoelectric systems were based.

1.5.3.6 NUCLEAR DYNAMIC SYSTEMS

1.5.3.6.1 Design

Radioisotope Brayton-cycle systems - A typical schematic flow diagram of a radioisotope Brayton-cycle system is shown in figure 1.68. The major system components are the shielded isotope heat source, recuperator, heat sink heat exchanger, space radiator, and combined rotating unit (CRU). The CRU consists of a high speed alternator mounted on a common shaft with a radial flow turbine and radial flow compressor. This assembly is supported by gas lubricated bearings and contained within a hermetically-sealed housing. The alternator stator is liquid cooled by the same coolant used in the heat rejection system. The working fluid for the system is pure argon gas. Gas mixtures (helium and xenon) could be considered to increase performance and reduce heat exchanger size; however, selective leakage of helium may prove unacceptable for long term application.

At the 5 kWe net power level, the major loop components (excluding the heat source) may be packaged within a volume approximately 4 ft by 3 ft by 2.5 ft. At higher power levels (up to 15 kWe) per power conversion system (PCS), the heat exchanger and gas duct sizes would increase somewhat, whereas the CRU would remain about the same size.

Table 1.27 shows a representative list of system operating parameters for radioisotope Brayton-cycle systems. The values shown are for systems producing 5 kWe and 10 kWe net power. The radiator area values shown are based on a cylindrical design.

Reactor Brayton-cycle systems - The schematic flow diagram for a reactor-powered Brayton-cycle system would be similar to that for the radioisotope system with the exception that the heat source would be a liquid metal-to-gas heat exchanger instead of the isotope fuel elements. Table 1.28 shows the reference design point data for the reactor Brayton-cycle system. The valves shown are for 10 kWe modules.

Redundancy requirements - To achieve the required system life, a number of redundant standby PCS modules will be required, depending on power level per module. Based on a PCS unit life of 1 year and a total system life of 2 years, the approximate number of redundant and operating 5 and 10 kWe units required to achieve the various power levels are as follows:

5 kWe modules

5 kWe	-	three units installed, one operating
10 kWe	-	five units installed, two operating

10 kWe modules

10 kWe	-	three units installed, one operating
20 kWe	-	five units installed, two operating
30 kWe	-	eight units installed, three operating

The redundancy values given are very preliminary and subject to further review.

Radioisotope mercury-Rankine systems - As previously stated, radioisotope mercury-Rankine systems cannot be considered for the space station applications because of isotope availability limitations.

Reactor mercury-Rankine systems - The SNAP-2 mercury-Rankine system can be considered for use with the SNAP-8 reactor for power levels up to 20 kWe total. Figure 1.69 shows a schematic flow diagram for the reactor system.

The major system components are the reactor heat source with associated liquid metal (NaK) coolant loop and pump, mercury boiler, radiator-condenser, and the combined rotating unit (CRU). The CRU consists of a turboalternator and mercury pump mounted on a common shaft which is supported by mercury-lubricated bearings. This assembly is contained within a hermetically-sealed housing. The design speed of the turboalternator is 36 000 rpm. The alternator stator is cooled by the turbine exhaust vapor.

The SNAP-2 power conversion system design is based upon application of the present CRU-V turbomachinery, which has a design power level of 4.1 kWe at the alternator terminals. CRU's have been operated at 5.6 kWe for short periods of time. Most of the test hours have been accumulated, however, at the 3 to 4 kWe power level. A 65

A brief assessment of the problems associated with extending the gross alternator output power to approximately 6.67 kWe (to achieve 5 kWe net conditioned) indicated that significant design modifications would be required. An inherently greater degree of confidence would be achieved by further development of the existing turbomachinery design on which extensive test and operating experience has been accumulated. Therefore, for application to the space station, it is proposed that two independent active power conversion systems operating in parallel be utilized to produce the net 5 kWe conditioned power level, each unit supplying 3.33 kWe at the alternator terminals.

The higher power levels would be achieved by grouping pairs of the 3.33 kWe units. Another approach to achieving higher power levels with less weight penalty and complexity is to uprate the individual units to slightly higher power output and operate fewer systems in parallel. For example, 10 kWe net could be achieved with three systems of 4.44 kWe rating each. This power rating could be achieved with minor changes to the current SNAP-2 turbine (CRU-V).

Redundancy requirements - In addition to the active systems, installed redundant standby systems would be required to satisfy the long mission life requirements. The number of standby systems required is a function of many variables including total system life reliability required, life reliability of individual modules, and reliabilities associated with system startups.

There are some inherent uncertainties associated with projecting lifetime capabilities of individual CRU's from the available analytical and test data, even with the large number of test hours thus far accumulated. Corrosion by mercury and CRU bearing cavitation erosion are basically the only fundamental life-limiting mechanisms thus far identified. These phenomena allow a potential lifetime capability of 2-1/2 years for the CRU and the remainder of the power conversion system (PCS) components. However, this study is based on the more conservative component lifetime objective of 10 000 hours (about 1 year). To maintain PCS module redundancy within reasonable limits, a corresponding overall system lifetime of 2 years is assumed.

Table 1.29 shows typical numbers of initial active PCS modules required and the corresponding number of redundant standby modules to achieve 2-year system life at the 5, 10, 15, and 20 kWe net power levels.

It is fairly obvious from examination of table 1-29 that at the 15 to 20 kWe power levels, the power subsystem becomes very complex with the large numbers of standby PCS loops.

Consideration has been given to scaling-up the existing SNAP-2 turbo-machinery design to produce a net conditioned output power of 10 kWe. This would greatly reduce the redundancy requirements imposed by multiple units to achieve the desired power level, system lifetime, and reliability. Also, the turbomachinery performance would be enhanced by larger size. Although these potential advantages apparently favor eventual application of the scaled-up design, a large and expensive program would be required to achieve the same level of confidence as is now realized with the existing turbomachinery. For this reason, a 10 kWe net mercury-Rankine system will not be further considered in this study.

Because of the complexity associated with a large number of redundant standby PCS modules, a maximum power level of 15 kWe is placed on the reactor SNAP-2 mercury-Rankine system. At this power level, the reactor SNAP-2 mercury-Rankine system can be considered further in a hybrid system for space station configuration number 1 (Dumbbell) or as primary power for configurations 2, 3a, 3b, and the Mars flyby.

1.5.3.6.2 Operation

Radioisotope Brayton cycle: The power conversion system could be started either on the ground prior to launch or in orbit. If started on the ground, a supplementary gas bearing pressurization system would be required to hydrostatically support the rotating assembly during boost acceleration. Also, whether started on the ground or in orbit, a water boil-off cooling system or heat absorbing medium would be required for system heat rejection.

In operation, the working fluid is heated at nearly constant pressure in the isotope heat source to the desired turbine inlet temperature. It then expands through the turbine producing mechanical power to drive the alternator and compressor. The gas then flows through the recuperator where a portion of its energy is transferred to the cooler gas from the compressor. The minimum cycle temperature is reached in the heat sink-heat exchanger where the waste heat is transferred to the radiator coolant. The radiator coolant is circulated through the radiator and heat exchanger by small electric motor driven pumps.

The electrical power is generated at high frequency by the compact alternator. The frequency of the system is maintained by a parasitic-load-type speed control which provides a constant load operational mode for the CRU(s) at all times. Also, a secondary battery system which handles peak and spike loads is connected in the alternator(s) output circuit and are charged as required when not in use. The use of the parasitic speed control permits essentially constant conditions of gas loop parameters to be maintained independent of load demands.

Reactor Brayton cycle: Operation of the reactor Brayton system is essentially the same as the isotope system with the exception that the heat source output is controllable. The reactor would not be started until the spacecraft is in Earth orbit. This eliminates many of the ground operational requirements associated with the radioisotope system; that is, shielding and water boil-off heat removal.

It may be necessary to include a secondary liquid metal loop between the reactor coolant loop and the gas heat exchanger. The estimated weight of this loop will be included in the total system weight.

Reactor mercury-Rankine cycle: As with the reactor Brayton system, the reactor would not be started until the spacecraft has been placed in orbit.

In operation, the reactor coolant(NaK) would transfer the reactor heat output to a secondary liquid metal coolant loop through a compact heat exchanger located in the gallery. This loop would transfer the heat to the mercury boiler. The liquid mercury is converted into superheated vapor in the boiler and flows to the turbine. The vapor expands in the turbine, producing mechanical power to drive the alternator. Mercury from the turbine exhaust flows to the radiator-condensor where it is condensed and subcooled – the latent and sensible heat being rejected to space by direct radiation from a tube-fin radiator. The mercury condensate is then returned to the boiler by a pump to complete the cycle.

The electrical power is generated at high frequency (1800 cps) by the compact alternator. The frequency of the system is maintained by a parasitic-load-type speed control which provides a constant load operational mode for the CRU at all times. The use of the parasitic speed control permits essentially constant conditions of temperatures, pressures, and flows to be maintained independent of load demands.

1.5.3.6.3 Integration

Brayton-cycle systems - configuration no. 1: It is assumed that the counterweight portion of the space station can be utilized to house a portion of the power generation system. Approximately 800 to 1000 ft^2 of surface should be available for radiator installation. This area is sufficient to accommodate up to a 10 kWe system (five 5 kWe modules or three 10 kWe modules). Because of the limitation on power imposed by insufficient radioisotope, this is the maximum power level attainable with the radioisotope Brayton-cycle system.

System weights: Figure 1.70 shows total system weights as a function of power level. The system weights shown include the heat source -- reactor or isotope heat block -- and the required shielding and peaking batteries.

Radiator area requirements: Figure 1.71 shows the radiator area requirements as a function of power level. Area requirements for both radioisotope and reactor systems are shown.

Configuration nos. 2, 3a, 3b, and 4: A detailed integration study for all of these configurations is beyond the scope of this study. Figure 1.72 shows the volume requirements as a function of power level. It is assumed that a cylindrical section of the vehicle surface will be available to accommodate the radiators. Figure 1.73 shows system weights for the Mars Flyby Mission (configuration 4).

Mercury-Rankine systems - configuration no. 1: System weights - Figure 1.74 shows total system weights as a function of power level. The system weights shown include the heat source (reactor) and the required shielding and peaking batteries. A maximum of 15 kWe is considered an upper limit for the multiple SNAP-2 system.

Radiator area required: Figure 1.75 shows the cylindrical radiator requirements for the reactor mercury-Rankine system for power levels up to 15 kWe.

Configuration nos. 2, 3a, 3b, and 4: Shown in figure 1.76 is total system weight as a function of power level for the reactor mercury-Rankine systems applicable to the zero-G space station. The radiator area required is the same as shown in figure 1.75 Figure 1.77 shows the radioisotope and reactor systems weights as functions of power level for the Mars Flyby Mission. Again, the radiator area required will not be significantly different from the values shown in figure 1.75 for this case.

1.5.3.6.4 Advantages and disadvantages.- As described in the previous sections, the nuclear Brayton-cycle and mercury-Rankine cycle (SNAP-2) systems both appear applicable in selected power ranges. Some of the advantages and disadvantages of these systems are listed below.

Brayton Cycle		SNAP-2 Mercury-Rankine	
Advantages	Disadvantages	Advantages	Disadvantages
1. High system efficiency	1. Relatively low development status	1. Advanced development status	1. Corrosion buildup
2. Single phase working fluid	2. Gas bearing load sensitivity	2. Relatively small high temperature radiator	2. Adverse "G" effects; two-phase working fluid
3. No materials compatibility problems	3. Large radiator required	3. Moderate temperatures	3. Low efficiency

1.5.3.6.5 Major system development areas.- The closed Brayton-cycle system will require development in the following major areas:

a. Gas lubricated bearings; manufacturing techniques, reproducibility, predictability, type (tilting pad, journal, or foil).

b. Compact system packaging.

c. System startup and re-start procedures; gas impingement or motor start.

d. Turbine and compressor manufacturing techniques to insure quality and long life capability.

e. High speed alternators, type, construction.

All of the major development problems associated with the mercury-Rankine system (SNAP-2) appear to be identified with the exception of problems which may arise during zero-G operation. The current AEC program is aimed toward solution of the known problems and demonstration of endurance capability.

1.5.3.7 Radiators.- Radiator areas presented for all systems are based on steady-state analyses for the desired system conditions. Areas are presented for the maximum environmental sink temperature (T_s)

for each mission considered. Radiator configurations are assumed cylindrical unless otherwise stated. A radiator effectiveness (η) of 0.85 was assumed.

Since some of the systems can accommodate or be designed over a fairly large radiator temperature range which approaches the upper temperature limits of low α/ϵ coatings, two values of α/ϵ were chosen. These values are 0.23 and 0.66. The attainment of value of 0.66 for those temperatures where the low α/ϵ coatings begin to degrade seems reasonable. It might also be pointed out that degradation of a selected low α/ϵ coating might be such that it would still be more attractive than a stable high α/ϵ coating.

It was assumed that the radiator would be an integral part of the spacecraft structure and weight penalties would only be incurred for that material over and beyond structural requirements.

1.5.4 DEVELOPMENT PROGRAM

1.5.4.1 Schedules.- The development schedules for the primary power generation system were derived by a four-step process as follows:

a. A schedule requirements sequence was developed to show the typical phasing which a hypothetical subsystem should follow in building up to delivery of flight units.

b. A ground test logic schedule was formulated which meets the major milestones of the above schedule requirements while accomplishing flight certification with a minimal ground test program.

c. Development requirements were investigated from the technology viewpoint, that is, for optimum development phasing, rather than phasing to meet a particular milestone sequence.

d. Development schedules were generated which incorporate likely technology leadtimes while preserving the essential elements of the space station EPS test logic and spacecraft development milestones.

Figure 1.78 shows the schedule requirements chart. This milestone sequence was developed from prime contractor data from MSC-sponsored AAP, Multipurpose Space Station and Mars flyby study programs. It will be noted that the milestone timing is a function of launch date only (no dependence on go-ahead date); the reference launch date for the operational, or permanent, space station is 1st quarter 1973.

The ground test logic is based on the following major guidelines:

a. Completion of qualification/certification testing must occur no later than delivery of the first operational flight article.

b. Flight certification will be based in part on a long endurance test (2-year goal) at high levels of assembly for life-critical assemblies, in part on integrated performance testing of a system rig in a configuration simulating the spacecraft installation, and in part on design limit dynamic and performance testing of a flight bill-of-materials (full-up) system.

c. Two manufacturing cycles are required during the ground test program, that is, there should not be a single production run which yields all test articles from design feasibility class through qualification units. This is discussed below.

The second guideline implies that the endurance basis for flight certification is not generated in tests of classical "qualification units." A realistic endurance test is still required, however, and must be conducted in a somewhat formal manner even if the test article design is of a developmental class.

The third guideline is intended to avoid a major pitfall of success-oriented planning: failures in the field of hardware of sound design, due to manufacturing defects. Applying this guideline to the ground test schedule, a manufacturing "breathing spell" is provided following completion of the last design verification test (DVT) article, so that early DVT results can be used to update tooling, process procedures, assembly flow, and inspection operations prior to <u>initiating</u> the production run which will include the flight unit(s).

The overall ground test logic is summarized in figure 1.79. The first manufacturing run produces design feasibility and design verification hardware to accomplish the following primary test objectives:

A. DFT - demonstrate specification compliance;
- generate design data for thermal balance and plumbing, electrical, and mechanical integration;
- provide endurance feasibility basis for design freeze;
- determine failure modes, failure propagation, and ultimate system effects.

b. DVT - provide performance and environmental basis for design freeze;
- provide endurance basis for flight certification;

- provide overstress tolerance data;
- provide test basis for debugging manufacturing operations.

The second manufacturing run produces a system prototype to verify integrated performance in a simulated installed configuration, qualification hardware for final performance and environmental certification, and flight systems.

Figure 1.80 shows a hardware support schedule for the ground test program. This schedule was generated from the test logic using more or less leisurely lead times, without any particular go-ahead or technology constraints. The manufacturing breathing spell discussed earlier is clearly evident. It is significant to note that design activities for the early test hardware appear as early as CY 1967.

Development schedules for isotope Brayton-cycle and solar cell primary power systems were selected for detailed analysis because they were believed to be the most likely candidates for the 1973 operational space station. These two systems suffer no severe constraints <u>external</u> to their own development problems, such as isotope availability.

The most comprehensive development schedule analyses available in these two areas were based on programs that were either non-manrated or success-oriented in the extreme. These reference schedules showed a total time-from-go-ahead through delivery of the first flight article of 28 months for solar cells and 44 months for the Brayton-cycle.

Figure 1-81 shows the estimated development schedule for the power conversion portion of the Brayton-cycle system. It appears that flight system delivery within some 56 months after program go-ahead might be feasible. If go-ahead occurs on July 1, 1967, this would support the reference 1st quarter 1973 launch.

Figure 1.82 shows the program phasing for isotope heat block development.

The development schedule for a solar cell system is shown in figure 1.83. The estimated lead time from go-ahead through flight hardware is 41 months.

Figure 1.84 was prepared to show the relative capabilities of these two systems to support possible developmental space station flights prior to the 1973 operational launch.

The estimated development schedules for batteries and fuel cells are shown in figures 1.85 and 1.86, respectively.

Figure 1-87 indicates the capabilities of these two items to support early space station flights.

As discussed elsewhere in this report, isotope thermoelectric systems will not be available in 1973 for primary power in the 5 to 10 kWe range due to limitations in isotope availability. The development schedule for this system is shown in figure 1.88 reference, however, to show when the conversion portion of an isotope system could be flight ready for lower power (1 to 3 kWe) applications, if no external constraints existed. It should be noted, however, that the isotope heat block development is the pacing item for the thermoelectric system (see figure 1.82), rather than the power conversion section.

The estimated schedule for the development and man-rating of a reactor system (based on the SNAP-8 reactor concept and either thermoelectric or Brayton-cycle conversion) is indicated in figure 1.89.

1.5.4.2 Costs.- Program costs for the systems under study were estimated for the following major cost categories:

a. Non-recurring:

1. Development - All expenses associated with analysis, technology and component development, and the design feasibility and design verification test programs.

2. Qualification - All expenses associated with materials, manufacturing, and operations for the qualification test equipment and test articles.

3. Prequalification deliveries - All materials, manufacturing, checkout, and shipping expenses directly associated with deliverable simulators, mock-ups, training equipment, and GSE.

b. Recurring:

1. Production - All materials, manufacturing, check-out, and shipping expenses directly associated with the delivered flight article(s).

2. Field support - All engineering and associated off-site expenses for spacecraft ground test support, training operations, flight spacecraft installation and checkout, and ground support services during flight.

Costs were not estimated for the following categories:

a. Spacecraft prime contractor costs: This is expected to be in the range of $20-30 million for all of the systems considered.

b. NASA costs for management, test, travel, et cetera.

c. Spares.

Cost estimates for the isotope Brayton-cycle, solar cell, battery, and fuel cell systems are broken out by fiscal year in tables 1.30 through 1.32. The phasing for these estimates follows the development schedules shown in section 1.5.4.1.

Estimated costs for isotope thermoelectric, reactor thermoelectric, and reactor Brayton-cycle systems are shown in tables 1.33 through 1.35, as total cost without a fiscal year breakout. All costs are shown in millions of dollars. It should be noted that the reactor system estimates correspond to the 20 kWe level, rather than the 5 kWe module design point.

Figure 1.90 illustrates the cost growth pattern for the above power systems as a function of the number of flight systems delivered. Since the design philosophy for these systems is based on modularity, variation in non-recurring costs with system size is considered insignificant relative to the accuracy of the estimates given.

The recurring costs will vary with module and system size as shown in figure 1.90 because of production economies realized with larger modules and because field support is more dependent on system complexity than on system power level. For any given system, for example, field support is higher for two 5-kWe modules than for one 10-kWe module.

The cost comparison for battery and fuel cell secondary systems is given in figure 1.91.

1.5.5 RESULTS AND DISCUSSION

In all Earth-orbit cases where nuclear power systems are considered, the radioisotope systems are availability-limited because of the 1972 need-date constraint. The thermoelectric system is limited to 4-kWe net and the Brayton-cycle system is limited to 10-kWe net. Radioisotope mercury-Rankine systems were eliminated from detailed study because of their inability to meet the 5-kWe minimum power level. Radioisotope thermoelectrics was studied due to its potential use as a highly reliable, long-life emergency or auxiliary power system. For all cases where a mercury-Rankine cycle was shown with a reactor heat source, the power level was limited to 15-kWe net because of an excessive number of conversion units (up to 30) to meet reasonable life/reliability goals at higher power levels.

Shown in figure 1.92 is the total weight charged to the selected systems versus net electrical power for the dumbbell configuration. At all power levels, the Brayton-cycle system is lightest. From 5 to 10 kWe the radioisotope/Brayton-cycle/battery system is lightest, 10 kWe being the limit of isotope availability. From 10 to 30 kWe the reactor/Brayton-cycle/battery system is lightest, with its advantage increasing rapidly with increasing power level. Up to 10 kWe, the solar cell/battery system is competitive with the Brayton-cycle. At the 10-kWe level, a choice based on weight of radioisotope/Brayton-cycle/battery, solar cell/battery, and reactor/Brayton-cycle is available.

Figure 1.93 shows the total system weight versus net power for the zero-gravity configuration. The decrease in assumed separation distance for the reactor systems causes significant increases in shield weights, in turn being reflected in the total system weight increase above the dumbbell configuration.

Figure 1.94 shows system weight versus net electrical power for the synchronous orbit mission. It was assumed that a maximum of 10 kWe would be required for this mission, therefore no reactor systems were considered. For primary power, the solar cell/battery system is lightest, although the radioisotope/Brayton-cycle system is competitive.

Figure 1.95 depicts system weight versus net electrical power for the Mars flyby. The solar cell/battery system is lightest for all power levels. Isotope availability limits allow radioisotope/Brayton-cycle to be used for the assumed upper power requirement of 15 kWe. However, it is assumed weight competitive with solar cells only to 10 kWe. From a system weight standpoint, radioisotope/thermoelectrics may be attractive as an auxiliary power system in the 2 to 4 kWe range.

Fuel cell or battery secondary power systems were required for power peaks with all primary power systems and for dark-side operation with solar cells. Figure 1.96 shows system weight versus net electrical power for both low Earth orbit missions. As secondary systems for use with solar cells, fuel cells are definitely not competitive with secondary batteries. A 6-month resupply case is shown to illustrate the trend typical of all batteries as being competitive from a weight standpoint: this is true only for the low Earth orbit configurations. The synchronous orbit configuration is shown in figure 1.97. The decrease in number of cycles required in a synchronous orbit allows the batteries to be discharged to a greater depth, thereby more fully utilizing their energy density capability. This is reflected in both the 6-month and 14-month resupply cases. However, for the 24-month case, the short fuel cell life relative to that of batteries causes the most significant difference in system weights.

Shown in figure 1.98 are the volume requirements for the various systems as they are configured for orbit for the dumbbell and synchronous missions. The solar cell system is the lowest because only batteries, power conditioning, and orientation mechanical controls require internal spacecraft volume. In figure 1.99, the reactor system volumes for the zero-gravity configuration increase due to the increase in shield size (less separation distance). The solar cell/battery and radioisotope system volumes are approximately the same for the zero-gravity and synchronous configurations.

Mars flyby power system volumes are shown in figure 1.100. The decrease in battery requirements for all systems is reflected most significantly in the solar cell system.

Shown in figure 1.101 is a representative secondary system volume comparison. The comparison reflects the significant difference in packaging and energy density characteristics between batteries and fuel cells.

Shown in figure 1.102 are the deployed area requirements for the four reference configurations. The dotted curve depicts area in excess of that available on the zero-gravity and dumbbell configuration nose cones for the reactor/Brayton-cycle radiator requirement. This area may have to be deployed if nose extension is not possible.

Figure 1.103 depicts radiator area requirements for the various power systems for all configurations. The solar cell/battery area requirements shown are only for thermal control of the batteries. For Mars flyby, the radioisotope/thermoelectric and Brayton-cycle curves can be extended to account for additional radioisotope availability. For the synchronous configuration, the radiator requirements are valid for 10 kWe and below.

Figure 1.104 summarizes the development schedule estimate for solar cell, radioisotope/Brayton, and reactor system concepts. As discussed above, these are the only systems of interest for further consideration in the 10-kWe class. A fully operational solar cell system could be delivered by 1971 (to support a 1972 launch), with potential capability to support precursor flights as early 1969 with design verification class hardware.

All of the nuclear systems could support a launch in the first half of 1973 provided a go-ahead is given in mid-1967 and no schedule slippage occurs. A full-scale reactor flight test system could be available for a 1971 precursor mission launch, while early flights using an isotope/ Brayton-cycle system would have to be limited in power level (due to fuel limitations in the precursor flight time period) or duration (if a short half-life isotope has to be used for early developmental flights).

Total program costs are shown in figure 1.105 for solar cells, isotope/Brayton, and reactor systems as a function of power level. The solar cell system can be acquired for roughly half the total program cost of any of the nuclear systems. Recurring costs are higher for solar cells, but the total costs do not approach those of the nuclear systems until a nominal 80-kWe power level (eight 10-kWe flights, four 20-kWe flights, etc.) is reached.

1.5.6 Power Systems Viewpoints on Station Configurations

Configuration 1 - solar cell systems

a. Vehicle attitude constrained with solar cells on nose cone.

b. Undesirable to have one central power system (slip rings)

Configuration 1 - nuclear systems

a. Undesirable to have one central power system.

b. Contributes to ballast requirements.

c. Advantageous from separation distance standpoint.

d. Radiator-area limited above 15 kWe unless nose extended.

e. Potential for reactor detachment and subsequent use on Mars flyby.

Configuration 2 - solar cell system

a. Apparent stowage advantage for launch.

b. No significant impact due to solar cell integration.

Configuration 2 - nuclear system

a. Constrained because of separation distance limits.

b. Radiator-area limited above 15 kWe unless nose extended.

c. Minimum rendezvous constraints, for nuclear systems.

Configuration 3 - solar cell systems

a. Apparent stowage advantage for launch.

b. No significant impact due to solar cell system integration.

Configuration 3 - solar cell systems

a. Apparent stowage advantage for launch.

b. No significant impact due to solar cell system integration.

Configuration 3 - nuclear systems

No significant impact due to radioisotope systems integration.

Configuration 4 - solar cell systems

No significant impact due to solar cell system integration.

Configuration 4 - nuclear systems

Possibly constrained because of separation distance limits, but contributes to ballast requirements if artifical gravity is used.

TABLE 1.2 - SOLAR CELL TECHNOLOGY CHART

	1966	1967	1968	1969	1970	1971	1972	1973	1974
N/P Silicon solar cells, solderless:									
Efficiency (AMO) per cell (ref. 28° C)									
Conventional contact, 10 to 14 mils	10.7	10.85	11.0	11.1	11.1	11.2	11.2	11.3	11.3
Wrap-around contact, 10 to 14 mils	[a]11.0	11.2	11.5	11.5	11.6	11.6	11.7	11.8	11.8
Temperature capability, maximum range									
Operating	-196° + 150° C								
Storage	-196° + 175° C								
Weight/2 by 2 cm cell	0.34g	0.34g	0.34g	0.33g	0.33g	0.33g	0.32g	0.32g	0.32g
Weight/3 by 3 cm cell	0.79g	0.79g	0.79g	0.78g	0.78g	0.77g	0.76g	0.76g	0.76g
Availability in quantity required:									
2 by 2 cm cell	Yes								
3 by 3 cm cell	No	No	Yes						
Cost/cell, dollars (quantity purchase):									
2 by 2 cm	4.5 to 5	4.5 to 5	4.5	4.5	4.25	4	4	4	4
3 by 3 cm	[a]9 to 10	[a]9 to 10	[a]9	9	8.75	8.75	8.5	8.5	8
Substrate weight, lb/ft^2:									
Aluminum honeycomb, conventional 1 in.	0.46								
Fiberglass honeycomb, 1 in.	0.35								
Fiberglass tape (Boeing concept) including frame	0.2								
Flexible, teflon-impregnated (Hughes rollup concept)	0.012								
Coverglasses (6 mils), lb/ft^2	[a]0.069								
Availability:									
Rigid panel deploy-system, if funded	No	Yes							
Rollup array system, if funded	No	Maybe	Yes						
Orientation system, if funded	No	Yes							

[a]Not available in large quantities

TABLE 1.3 - DYNAMIC POWER CONVERSION SYSTEM CONCEPTS AND APPLICABILITY ASSESSMENT

Conversion system	Sponsoring agency	Energy source	Mission applicability
SNAP-2 mercury Rankine-cycle	AEC	Designed for SNAP-2 reactor, can be used with radioisotope heat source.	Considered applicable for use with nuclear reactor and/or radioisotope heat source. Power level may be limited because of complexity and/or isotope availability.
SNAP-8 mercury Rankine-cycle	NASA (conversion system only)	SNAP-8 reactor	Considered unsatisfactory for further mission application study because of its high power rating and weight.
Closed Brayton-cycle	NASA	Solar, radio-isotope	Considered applicable for use with reactor and/or isotope heat sources. Power level may be limited by isotope availability.
Organic Rankine cycle	Navy, AF, AEC	Solar, nuclear	Some success has been realized with low investments. However, potential problems with working fluid decomposition make these systems unattractive for long term use. This system concept will not be investigated.

TABLE 1.4 - RADIOISOTOPE SNAP SPACE POWER SYSTEMS

Designation	Application	Power, watts, electrical	Weight, lb	Isotope	Design life	Status
SNAP-3	Demonstration device	2.5	4	Polonium-210	90 days	Program completed.
SNAP-3 (modified)	Navy navigational satellites	2.7	4.6	Plutonium-238	5 years	Generators launched June and Nov. 1961. First still operable; second failed after 190 days.
SNAP-9A	Navy navigational satellites	25	27	Plutonium-238	5 years	Generators launched Sept. and Dec. 1963. Both still operable.
SNAP-11	Moon probe	25	30	Curium-242	90-day mission	Development complete.
SNAP-13	Thermionic demonstration device	12	4	Curium-242	90-day mission	Generator fueled Dec. 1965.
SNAP-17	-	25	28	Strontium-90	5 years	Design studies complete.
SNAP-19	Nimbus B	30	30	Plutonium-238	5 years	To become operational in 1966.
SNAP-27	Lunar surface	50	40	Plutonium-238	1 year	Early development.
SNAP-29	Earth orbit	400	>500	Polonium-210	90 days	Design studies complete.

TABLE 1.5 - CHARACTERISTICS OF RADIOISOTOPE HEAT SOURCES

		Principal mode of decay, MeV	Half-life	Typical fuel composition	Fuel form, specific gravity	Typical fuel thermal power			Fuel melting point, °C	Isotope	Permissable concen-	Shielding	Fuel costs
						watts/cc	watts/gm	curies/watt		percent	air, μc/cc		\$/watt thermal
Beta emitters	Cobalt-60	β(0.31) γ(1.17)	5.24 yr	Co	8.7	46.2	5.31	65.1	1480	34	1×10^{-10}	Heavy	33
	Strontium-90	β(2.24) γ(1.734)	28 yr	$SrTiO_3$	3.7	[a].825	[a].223	148	1900	25	3×10^{-12}	Heavy	5
	Cesium-137	β(0.529) γ(0.66)	30 yr	Cs Glass	3.1	[a].24	[a].077	207	1275	14	1.5×10^{-10}	Heavy	5
	Cerium-144	β(1.321) γ(2.18)	285 days	CeO_2	6.3	[a]21.9	3.47	126	2680	11	7×10^{-11}	Heavy	1
	Promethium-147	β(0.223) γ(0.121)	2.62 yr	Pm_2O_3	7.3	[a]2.09	[a].286	2788	2270	82	7×10^{-10}	Light	10
	Thulium-170	β(0.96) γ(0.084)	129 days	Tm_2O_3	7.7	7.9	1.03	385	2300	6.6	3×10^{-10}	Light	10
Alpha emitters	Polonium-210	α(5.3) γ(0.8)	138 days	(b)	10.0	824	82.4	31.2	1675	54	7×10^{-12}	Light	10
	Thorium-228	α(5.42) γ(0.084)	1.9 yr	ThO_2	9.0	1270	141	24	3220	83	7×10^{-14}	Heavy	40
	Uranium-232	α(5.32) γ(0.058)	74 yr	UO_2	10.0	33	3.3	26	2880	75	3×10^{-13}	Heavy	350
	Plutonium-238	α(5.49) γ(0.044)	86.4 yr	PuO_2	10.0	[a]<3.0	[a].39	29.1	2280	70	2×10^{-14}	Light	[c]1000
	Curium-242	α(6.11) γ(0.04)	162 days	Cm_2O_3	9.0	397[a]	44.1	27.7	1950	34	1×10^{-12}	Light	150
	Curium-244	α(5.80) γ(0.04)	18.1 yr	Cm_2O_3	9.0	21.6	2.40	33.8	1950	86	9×10^{-14}	Heavy	357

[a]Measured values of AEC-supplied isotopes; other values are calculated.

[b]Nonmetallic fuel forms classified.

[c]C[illegible] Pu-238 depends on production method and time period required and will vary from approximately [illegible] and dollars per thermal watt.

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TABLE 1.6.- REACTOR SNAP SPACE POWER SYSTEMS

Designation	Application	Power, kWe		Weight, lb (a)	Design life, hr	Status
		Design	Potential			
SNAP-10A	Thermoelectric demonstration system	0.5	2	1000	10 000	Development completed. Flight tested April 1965.
SNAP-2	Rankine cycle demonstration system	3	10	1500	10 000	Research and development on non-nuclear components.
SNAP-8	Unspecified	35	50	6000 (est.)	10 000	Power test of experimental reactor completed.
SNAP-50[b]	Electric propulsion and auxiliary power	300	1200	6000 (est.)	10 000	Research and technology development.

[a]Includes weight of a radiation shield to reduce cumulative dose over a year's operation to levels that will not damage sensitive instrumentation - weight also includes, except where noted, single energy conversion package.

[b]SNAP-50 was terminated as a system (reactor and Rankine cycle energy converter) development program in 1965; however, technology research and development is continuing.

TABLE 1.7 - SNAP-8 EXPERIMENTAL REACTOR DESIGN DATA

Rated Operating Conditions	
Thermal power	600 kW
Coolant	Eutectic sodium-potassium alloy (NaK-78)
NaK outlet temperature	1300° F
NaK inlet temperature	1100° F
NaK flow rate	13.6 lb/sec
NaK inlet pressure	39 psia
NaK outlet pressure	37 psia
Fuel Elements	
Fuel material	Hydride uranium-zirconium alloy
Uranium content in fuel alloy	10 wt percent (~6.56 kg U^{235} in core)
Hydrogen content in fuel alloy	6×10^{22} hydrogen atoms per cm^3
Fuel element cladding	0.010-inch-thick Hastelloy-N
Number of fuel elements	211
Fuel element OD	0.56 in.
Fuel element length	14.5 in.
Core vessel	
Material	316 stainless steel
Inside diameter	9.214 in.
Length	~ 21 in.
Thickness	0.070 in.
Reflector	
Material	Anodized beryllium
Thickness	5 in.
Length	14.5 in.
Number of reflector control drums	6
Radius of rotation of control drums	4.69 in.

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TABLE 1.8 - SYSTEMS FOR DETAILED MISSION/CONFIGURATION STUDY

Systems (a)	Configurations (b)			
	No. 1	No. 2	No. 3	No. 4
Radioisotope thermoelectric	5-kWe level	5-kWe level	5-kWe level	5-kWe level
Radioisotope mercury-Rankine	"	"	"	5 and 10 kWe levels
Radioisotope Brayton	5, 10, and 15 kWe	5,10, and 15 kWe	X	X
Reactor thermoelectric	X	X	X	10 and 15 kWe levels
Reactor mercury-Rankine	X	X	X	"
Reactor Brayton	X	X	X	"
Solar cell	X	X	X	X

[a]Batteries to be used as necessary. Hybrid systems to be used where advantageous.
[b]X means all power levels considered.

TABLE 1.9 - BATTERY/FUEL CELL CONFIGURATIONS

(a) Mars

A. Supplementary Power Systems

   a. Solar

      1. F/C Regen.

      2. F/C Non-Regen.

      3. Battery (Ag-Cd)

   b. Nuclear (R/I and R_x)

      1. F/C Regen.

      2. F/C Non-Regen.

      3. Battery (Ag-Cd)

5, 10, and 15 kWe primary EPS

B. Complementary Power Systems

   a. Storm Cellar EPS

      1. F/C Non-Regen.

      2. Battery

   b. Encounter expts. EPS

      1. F/C Non-Regen.

      2. Battery

   c. Midcourse EPS

      1. F/C Non-Regen. w/Solar Cell Primary System.

      2. F/C Non-Regen. w/Nuclear Primary System.

      3. Battery

   d. Single complementary EPS (for a and b and c)

      1. F/C Non-Regen. w/Solar Cell Primary System.

      2. F/C Non-Regen. w/Nuclear Primary System.

TABLE 1.9 - BATTERY/FUEL CELL CONFIGURATIONS - Concluded

(b) Earth orbit

(Low Earth Orbit and Synchronous Earth Orbit)

A. Complementary Power Systems (Solar Systems Only)		
a. F/C Regen. b. F/C Non-Regen. c. Battery (Ni-Cd and Ag-Cd)	5 kWe 10 kWe Prim. EPS	6-mo Re-Supp. 2-yr Re-Supp.
B. Supplementary Power Systems		
a. Solar 1. F/C Regen. 2. F/C Non-Regen. 3. Battery (Ag-Cd) b. Nuclear (R/I and R_x) 1. F/C Regen. 2. F/C Non-Regen. 3. Battery (Ag-Cd)	5 kWe 10 kWe Prim. EPS	6-mo Re-supp. 2-yr Re-supp.

TABLE 1.10- SECONDARY POWER SUBSYSTEMS DESIGN REQUIREMENTS

(a) Earth orbital missions

Subsystem Use	Energy to loads		Available recharge energy, kWh/72 hr	No. cycles per mission	Peak power level, kWe	Mission duty cycle		
	kWh/72 hr	kWh mission				Days before use	Hours on load	Hours on standby
Low Earth orbit supplemental								
5 kWe - 6 mo	37	2 200	33	720	4.0	0	1050	3 270
2 yr	37	8 650	33	2800	4.0	0	4100	12 700
10 kWe - 6 mo	56	3 300	50	720	6.0	0	1050	3 270
2 yr	56	12 980	50	2800	6.0	0	4100	12 700
15 kWe - 6 mo	74	4 400	66	720	8.0	0	1050	3 270
2 yr	74	17 300	66	2800	8.0	0	4100	12 700
Synchronous Earth orbit, supplemental (Nuclear EPS)								
5 kWe - 6 mo	16	950	33	240	1.0	0	960	3 360
2 yr	16	3 700	33	940	1.0	0	3730	13 070
10 kWe - 6 mo	24	1 430	50	240	1.5	0	960	3 360
2 yr	24	5 600	50	940	1.5	0	3730	13 070
(Solar EPS)								
5 kWe - 6 mo	34	2 020	33	360	6.0	0	1090	3 230
2 yr	34	7 840	33	1400	6.0	0	4240	12 560
10 kWe - 6 mo	51	3 030	50	360	11.5	0	1090	3 230
2 yr	51	11 800	50	1400	11.5	0	4240	12 560

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TABLE 1.10- SECONDARY POWER SUBSYSTEMS DESIGN REQUIREMENTS - Concluded

(b) Mars flyby

Subsystem use	Energy to loads		Available recharge energy, kWh/72 hr	No. cycles per mission	Peak power level, kWe	Mission duty cycle		
	kWh/72 hr	kWh/mission				Days before use	Hours on load	Hours on standby
Mars flyby, supplemental (Nuclear EPS)								
5 kWe	37	3030	0	1050	4.3	0	2630	14 170
10 kWe	56	4550	0	1050	6.6	0	2630	14 170
(Solar EPS)								
5 kWe	17	900	0	230	3.0	160	780	8 580
10 kWe	24	1350	0	230	4.5	160	780	8 580
Mars flyby storm cellar	N/A	360	N/A	3	3.1	Unk	144	Negl.
Mars flyby experiments power supply (Nuclear EPS only)	N/A	350	N/A	1	2.0	145	240	Negl.
Mars flyby midcourse power supply (Nuclear EPS)	N/A	180	N/A	6	0.8	80	220	Negl.
(Solar EPS)	N/A	60	N/A	2	0.8	400	70	Negl.
All 3 previous power supplies (Nuclear EPS)	N/A	890	N/A	10	3.1	Use 145	600	Negl.
(Solar EPS)	N/A	420	N/A	5	3.1	Use 145	220	Negl.

TABLE 1.11 - BATTERY RADIATOR CONSIDERATIONS

Mission	Max Q	Max A/Q	Area, ft^2	Weights[a]	Max[b] flow rate, lb/hr
Mars					
Solar	12 500	0.015	188.2	141	676
Nuclear	12 500	.015	188.2	141	676
Synchronous Earth Orbit					
Solar	18 800	.011	207	155	1015
Nuclear	8 200	.011	90	67.5	445
Low Earth Orbit					
Solar	25 000	.011	275	206	1350
Nuclear	10 900	.015	164	123	590

[a]Not including fluid weight, pump, or heat exchangers.

[b]For 62.5 percent ethylene glycol-water.

NOTE: Further system design analysis required to prevent low temperature flow problems. This problem can be alleviated by thermal integration of the battery system with the primary system (except solar cells).

TABLE 1.12 - BATTERY SUMMARY TABLE

5 kWe mission	Battery	Battery capacity, kWh	Maximum depth of discharge, percent	Battery weight, lb	Battery volume, ft^3	Charge control weight, lb	Primary EPS additional power required for charging kWe gross	10 kWe weight, volume, capacity	15 kWe weight, volume, capacity
LEO - supplementary 6 mo	Ag-Cd	28.5	65	1240	10.8	15 - nuclear	0.420	1.5 times	2 times
2 yr	Ag-Cd	40.0	50	1735	15.2	25 - solar			
LEO - complementary Dark side 6 mo	Ag-Cd Ag-Cd	16 8	25 50	700 350	6.12 3.06	75 75	N.A. N.A.	2 times	3 times
2 yr	Ni-Cd Ag-Cd	16 40	25 10	1600 1750	11.6 15.30	75 75	N.A. N.A.		
SYNC E-O supplementary 6 mo	Ag-Cd	17.5	75	761	6.65	10	.343	1.5 times	2 times
Solar 2 yr	Ag-Cd	26.3	50	1140	10.0	10	.343		
supplementary 6 mo	Ag-Cd	18.9	85	821	7.18	10	0		
Nuclear 2 yr	Ag-Cd	22.8	70	955	8.66	10	0		
MARS Solar 2 yr	Ag-Cd	20.8	80	905	7.9	10	0		
Nuclear 2 yr	Ag-Cd	12.7	60	552	4.76	10	1.018		

TABLE 1.13 - 5-kWe BATTERY/FUEL CELL WEIGHT COMPARISONS

Mission	Battery[a] weight, lb	Fuel cell[a] regen. wt., lb	Fuel cell[b] non-regen., wt.
Mars flyby			
Solar supplementary	920	1500	2500
Reactor supplementary	1100	3500	4900
Radioisotope supplementary	1600	4700	4900
Low Earth orbit[c]			
Solar supplementary	1400	1650	2200
Solar complementary		3500	5500
a. Ag-Cd 25 percent DOD	1850		
b. Ag-Cd 50 percent DOD	1500		
Reactor supplementary	1475	1650	2200
Radioisotope supplementary	1675	2050	2200
Synchronous Earth orbit[c]			
Solar supplementary	800	1900	2600
Reactor supplementary	875	1450	1000
Radioisotope supplementary	875	2600	1000

[a]Weights include primary EPS weight penalties for battery recharging and fuel cell reactant regeneration.

[b]Weights for nonregenerative fuel cells include a water weight credit.

[c]Only 6-month resupply intervals are shown in these missions.

TABLE 1.14.- POWER CONDITIONING EQUIPMENT

Component	Efficiency, percent	Weight, lb/kWe	Volume, in.^{3}/kWe
dc regulator	90 to 95	7.5	200
Inverter	70 to 80	28	1000
Battery charge/ discharge controller	90 to 95	10	300
Transformer/rectifier	80 to 90	4	200
Frequency changer	80 to 90	20	1000

TABLE 3-14.- CONCEPTS "A" AND "B"

	Concept "A", rigid panels	Concept "B", unfurlable
Substrate	Aluminum or fiberglass honeycomb	Teflon-impregnated fiberglass
Deployment	Motor drive	Motor drive
Status	State-of-the-art	Requires development
Retractable	No	Yes
Deployed stability	Good	Unknown
Thermal distortion	Fair	Unknown
Relative weight	1	0.5 to 0.75
Flight experience	Good	None
Ease of stowage	Fair	Good

TABLE 1.16 - RADIATION ENVIRONMENT FOR 300 NAUTICAL MILE CIRCULAR ORBIT AND STATIONARY ORBIT

[60° Inclination, 300 nautical miles circular orbit, projected 1968]

Electron integral spectrum		Proton integral spectrum	
E energy, MeV	0.5 MeV Electrons/ cm^2/day flux, E	E energy, MeV	30 MeV protons/ cm^2/day flux, E
0.0	7.92×10^{10}	1	3.1×10^{7}
.50	4.71×10^{9}	3	2.25×10^{7}
1.0	1.09×10^{9}	5	1.65×10^{7}
1.5	3.54×10^{8}	7	1.20×10^{7}
2.0	1.25×10^{8}	9	8.88×10^{6}
2.5	4.81×10^{7}	11	6.55×10^{6}
3.0	2.09×10^{7}	13	4.84×10^{6}
3.5	1.08×10^{7}	15	3.60×10^{6}
4.0	6.85×10^{6}	20	1.75×10^{6}
4.5	5.10×10^{6}	25	8.93×10^{5}
5.0	4.20×10^{6}	30	4.79×10^{5}
5.5	3.65×10^{6}	35	2.75×10^{5}
6.0	3.25×10^{6}	40	1.71×10^{5}
6.5	2.93×10^{6}	45	1.16×10^{5}
7.0	2.66×10^{6}	50	8.49×10^{4}

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TABLE 1.17 - BASIC CELL STACK WEIGHTS

	Rigid, lb/ft^2	Unfurlable, lb/ft^2
Covers (0.006 inch)	0.0690	0.0690
Cells (0.013 inch, 2 by 2 cm)	.1608	.1608
Adhesives	.0165	.0165
Interconnects	.0168	.0168
Thermal coating	.0276	
Substrate (1 inch)	.3500	.0120(TFE/fiberglass)
Totals	.6407	.2751

TABLE 1.18 - DRAG PENALTY CALCULATIONS - LEO, 5 kWe EXAMPLE

<u>260 n.mi.</u>

$F_D = 7.67 \times 10^{-6} A_n \left(ft^2\right)$ = Drag force

for orbit maintenance, $F_D = F^{th}$ (thrust)

$$\dot{W} = \frac{F_{th}}{I_{sp}}$$

where:

F_D = drag force, lb

A_n = array area normal to the orbital velocity vector, ft^2

F_{th} = RCS engine thrust, lb

$\dot{W}$ = propellant mass flow rate, lb/sec

I_{sp} = RCS engine specific impulse

$$F_C = \left(7.67 \times 10^{-6}\right)\left(1450\ ft^2\right) = 0.011\ lb = F^{th}$$

$$\dot{W} = \frac{F_{th}}{I_{sp}} = \frac{0.0111}{250} = 0.445 \times 10^{-4}\ lb/sec$$

$$\left(3.15 \times 10^7\ sec/yr\right)(2\ yr) = 6.30 \times 10^7\ sec/2\ yr$$

$$\left(0.445 \times 10^{-4}\ lb/sec\right)\left(6.30 \times 10^7\ sec\right) = 2.8 \times 10^3\ lb/2\ yr$$

This amount would be required if the array were always normal to the orbital velocity vector; at the worst, this will be true for one-third of the time.

$$\left(2.8 \times 10^3\ lb\right)(1/3) = 0.93 \times 10^3\ lb = 930\ lb/1450\ ft^2 \text{ for 2 years}$$

TABLE 1.19 - GROSS ARRAY POWER REQUIRED FOR PEAKING BATTERIES

Low Earth Orbit	
Continuous charging requirements	Array power requirements
5 kWe - 420 W	$(420\ W)\left(\frac{94.3\ min/orbit}{58.5\ min\ light/orbit}\right) = 676\ W$
10 kWe - 635 W	$(635\ W)\left(\frac{94.3}{58.5}\right) = 1014\ W$
15 kWe - 840 W	$(840\ W)\left(\frac{94.3}{48.5}\right) = 1352\ W$
20 kWe - 840 W	1352 W
25 kWe - 840 W	1352 W
30 kWe - 840 W	1352 W
Synchronous Earth Orbit	
Continuous charging requirements	Array power requirements
5 kWe - 343 W	$(343\ W)\left(\frac{24\ hr/orbit}{22.8\ hr\ light/orbit}\right) = 360\ W$
10 kWe - 515 W	$(515\ W)\left(\frac{24}{22.8}\right) = 540\ W$
15 kWe - 686 W	$(686\ W)\left(\frac{24}{22.8}\right) = 720\ W$
20 kWe - 686 W	720 W
25 kWe - 686 W	720 W
30 kWe - 686 W	720 W

TABLE 1.20.- SOLAR ARRAY AREAS AND ACTUAL POWER OUTPUTS

LEO

10-kWe battery: Ni-Cd, 11 000 cycles, 25 percent DOD

= 75 percent

P_{sc}	total gross array power
P_n	power to the loads = 10 kWe
W-H	battery watt-hour efficiencty = 75 percent
cc	charge controller efficiency = 90 percent
T_d	discharge (dark) time = 35.8 minutes
T_c	charge (light) time = 58.5 minutes
pc	power-conditioning efficiency = 76 percent

$$P_{sc} \quad \frac{P_n}{pc}\left[1 + \left(\frac{1}{W\text{-}H \quad cc}\right) \left(\frac{T_d}{T_c}\right)\right]$$

$$P_{sc} \quad \frac{10\ 000}{0.75}\left[1 + \left(\frac{1}{0.75(0.90)}\right)\left(\frac{35.8}{58.5}\right)\right]$$

P_{sc} 25 680 W or 25.7 kWe

Peaking batteries require 1014 W gross power from the array.

Solar array must be sized to 25 680 + 1014 = 26 694 watts gross array power for a 10-kWe continuous system; peaks are covered.

$\dfrac{26\ 694\ \text{W}}{9.33\ \text{W/ft}^2}$ 2861 ft^2 of array for 10 kWe, including peaks.

5 kW

For 5 kWe, Array power is: $\dfrac{25\ 680\ \text{W}}{2} = 12\ 840\ \text{W}$

Peaking batteries require = 676 W

Total array gross power is 12 840 + 676 = 13 516 watts

$\dfrac{13\ 516\ \text{W}}{9.33\ \text{W/ft}^2}$ = 1449 ft^2 of array for 5 kWe, including peaks.

15 kWe

Array power = 12 840(3) = 38 520 W gross

Peaking battery requirements = 1352 W gross

Array power = 38 520 + 1352 = 39 872 W

$$\frac{39\ 872\ \text{W}}{9.33\ \text{W/ft}^2} = 4274\ \text{ft}^2$$

TABLE 1.20 - SOLAR ARRAY AREAS AND ACTUAL POWER OUTPUTS - Concluded

SYNC

10.59 W/ft^2

5 kWe net $= \frac{5}{0.75} = 6.666$ kWe gross + 360 W batt = 7026 watts array

10 kWe net $= \frac{10}{0.75} = 13.333$ kWe gross + 540 W batt = 13 875 watts array

15 kWe net $= \frac{15}{0.75} = 20$ kWe gross + 720 W batt = 20 720 watts array

Areas

5 kWe $\quad \frac{7026 \text{ W}}{10.59 \text{ W/ft}^2} = 663 \text{ ft}^2$ of array

10 kWe $\quad \frac{13\,875 \text{ W}}{10.59 \text{ W/ft}^2} = 1310 \text{ ft}^2$ of array

15 kWe $\quad \frac{20\,720 \text{ W}}{10.59 \text{ W/ft}^2} = 1957 \text{ ft}^2$ of array

TABLE 1.21.- SOLAR ARRAY CALCULATIONS

C	Correction for cover glass losses (0.98)
D_e	Correction for environmental degradation (0.97)
D_t	Cell temperature degradation coefficient (0.45 percent/°C)
K_1	Correction for mismatch losses (0.98)
K_2	Correction for process degradation (0.985)
K_3	Correction for calibration and test errors (0.96)
K_4	Correction for solar constant uncertainty deviation (0.945)
M	Correction for misorientation (±15°, 0.966)
P_s	$SRCK_1K_2K_3K_4D_eM \left[1-(T-28)D_t\right]$
P_s	Specific power (watts/ft^2)
R	Ratio of cell area to overall array area (0.94)
$RMS(K_1K_2K_3K_4)$	0.928
S	Solar intensity (127.0 watts/ft^2)
T_c	Cell operating temperature: 65° C for 260 n. mi.; 40° C for Sync orbit
η	Air mass zero (AMO) cell efficiency (11.0 percent)

260 n. mi.

P_s	(0.11)(127)(0.94)(0.98)(0.928)(0.97)(0.966) 1 - (37) (0.0045)
P_s	9.33 W/ft^2

Sync orbit

P_s	(0.11)(127)(0.94)(0.98)(0.928)(0.97)(0.966) 1 - (12)(0.0045)
P_s	10.59 W/ft^2

TABLE 1.22.- SOLAR ARRAY SUBSYSTEM WEIGHTS, BATTERIES AND PCS EXCLUDED

LEO

5 kWe

$(1449 \text{ ft}^2)(0.6407 \text{ lb/ft}^2) = 928.4$ lb array only

a.	Orientation servos, sensors, etc.	30 lb
b.	Deployment motors, gears, etc.	45 lb
c.	Electrical transmission system	50 lb
d.	Pyrotechnics	3 lb
e.	Hinges, locks, etc.	40 lb
f.	Structure, spars, etc. (0.4 lb/ft^2)	580 lb
		748 lb

928.4 lb array
748.0 lb mech
1676.4

Total 1676 lb

10 kWe

$(2861 \text{ ft}^2)(0.6407 \text{ lb/ft}^2) = 1833$ lb array only

a.	Orientation servos, sensors, etc.	50 lb
b.	Deployment motors, gears, etc.	60 lb
c.	Electrical transmission system	90 lb
d.	Pyrotechnics	5 lb
e.	Hinges, locks, etc.	60 lb
f.	Structure, spars, etc. (0.4 lb/ft^2)	1140 lb
		1405 lb

1883 lb array
1405 lb mech
3288

Total 3288 lb

15 kWe (Configuration No. 1 Only)

$(4274 \text{ ft}^2)(.6407 \text{ lb/ft}^2) = 2738$ lb array only

a.	Orientation servos, sensors, etc.	70 lb
b.	Deployment motors, gears, etc.	75 lb
c.	Electrical transmission system	110 lb
d.	Pyrotechnics	7 lb
e.	Hinges, locks, etc.	80 lb
f.	Structure, spars, etc. (0.4 lb/ft^2)	1709 lb
		2051 lb

2738 lb array
2051 lb mech
4789 lb

Total 4789 lb

TABLE 1.22.- SOLAR ARRAY SUBSYSTEM WEIGHTS, BATTERIES AND PCS EXCLUDED - CONCLUDED

SYNC

5 kWe

$(663\ ft^2)(0.6407\ lb/ft^2) = 425$ lb array only

a.	Deployment motors, gears, etc.	25 lb
b.	Pyrotechnics	3 lb
c.	Hinges, locks, etc.	25 lb
d.	Structure, spars, etc. $(0.4\ lb/ft^2)$	265 lb
		318 lb

425 lb array
318 lb mech
743 lb

Total 743 lb

10 kWe

$(1310\ ft^2)(0.6407\ lb/ft^2) = 839$ lb array only

a.	Deployment motors, gears, etc.	40 lb
b.	Pyrotechnics	5 lb
c.	Hinges, locks, etc.	45 lb
d.	Structure, spars, etc. $(0.4\ lb/ft^2)$	525 lb
		615 lb

835 lb array
615 lb mech
1454 lb

Total 1454 lb

TABLE 1.23.- ARRAY WEIGHTS - FINAL

Leo	Area-includes area for battery charging	Array system weight, does not include battery and PCS weight	
		With orientation sys.	
5 kWe	1449 ft^2	1676 lb + 930 drag penalty Total 2606 lb	
10 kWe	2861 ft^2	3288 lb +1832 drag Total 5120 lb	
15 kWe	4274 ft^2	4789 lb +2743 drag Total 7532 lb	
Sync	**Area-includes area for battery charging**	**Array system weight, does not include battery and PCS weight**	
		With orientation system.	No orientation system.
5 kWe	663 ft^2	823	743
10 kWe	1310 ft^2	1594	1454
15 kWe	1957 ft^2	2417	2197

TABLE 1.24.- SOLAR CELL PERFORMANCE FOR INTERPLANETARY FLYBY

Mission day	Intensity W/ft^2	Cell temperatures, °C	Power, W/ft^2 (based on 127 W/ft^2)	Actual power, W/ft^2
1	127	40	10.6	10.6
40 and 690	115	15	11.22	10.15
84 and 648	88	11	11.41	7.91
112 and 626	70	1	11.89	6.55
148 and 588	55	-15	12.65	5.48
152 and 584	52	-17	12.75	5.22
174 and 560	45	-20	12.89	4.57
192 and 540	40	-21	12.94	4.08
234 and 500	33	-23	13.03	3.39
266 and 460	30	-24	13.08	3.09
360	27 min	-25	13.13	2.79

Sample calculation

For mission day 1, area-specific power is calculated as in table 3-20. An adjustment is then made for temperature using the equation

$$P = \left[P_0 \quad 1 - (T - 28)D_t\right]$$

where P = area-specific power, W/ft^2 at operating temperature

P_o = area-specific power at 28° C

T = solar cell operating temperature, °C (3rd column)

D_t = solar cell temperature degradation coefficient = 0.45 percent/°C

The results of this calculation are shown in the 4th column of the above table for the various temperatures given in the 3rd column. The intensity adjustment is performed by multiplying the data of column 4 by the ratio of the intensity for a particular day to day 1 (near-Earth) intensity:

day 84: $\left(11.41\ W/ft^2\right)\left(\frac{88}{127}\right) = 7.91\ W/ft^2$ actual power (5th column)

TABLF 1-25.- SOLAR CELL/BATTERY SYSTEMS ADVANTAGES/DISADVANTAGES

Mission	Advantages	Disavantages
Earth-orbital	Experience Highest total and partial-power reliability Safety: no hazardous matl's. Development and recurring costs Shortest development time Inherently simple: few moving parts Easily maintained, if desired No nuclear radiation effects No thermal launch constraints Ease of ground checkout No radiation handling problems	Drag and mass-movement (orientation effects) Vehicle attitude constraints Secondary system required for dark-side power
Interplanetary	Experience Total and partial-power reliability Safety: no hazardous materials Development and recurring costs Shortest development time Inherently simple: few moving parts Easily maintained, if desired No nuclear radiation effects No thermal launch constraints Ease of gound checkout No radiation handling problems	Varying power output due to solar flux variation-system must be designed to provide full mission power at 2.2 A. U. (overdesign factor of ~3.5) Possible vehicle attitude constraints Subject to damage from severe radiation environments

TABLE 1.26 - REACTOR SYSTEM WEIGHTS

	Earth orbit configuration no.1		Interplanetary	
	10 kWe, net	15 kWe, net	10 kWe, net	15 kWe, net
Reactor	760	760	760	760
Conversion	2 800	4 150	2 680	4 050
Radiator	3 300	4 900	2 600	3 900
Shield				
50-ft sep			4 800	5 400
100-ft sep	6 300	6 700		
Support boom	2 000	2 000	500	500
Batteries	2 600	3 500	750	1 100
	17 760	22 010	12 090	15 710

TABLE 1-27 - RADIOSOTOPE BRAYTON-CYCLE SYSTEM PARAMETERS

	5 kWe, net	10 kWe, net
Working fluid	Argon	Argon
Turbine inlet temperature (°F)	1 600	1 600
Compressor inlet temperature (°F)	100	100
Shaft speed (rpm)	48 000	48 000
Compressor specific speed	.1	.1
Recuperator effectiveness	.92	.92
Pressure-loss factor (B)	.903	.903
Compressor inlet pressure (psia)	17.4	28.5
Compressor efficiency	.807	.807
Turbine efficiency	.87	.87
Power conditioning efficiency	.74	.75
Generator efficiency	.90	.90
Gas flow rate (lb/sec)	.455	.910
Cycle efficiency (includes 3 percent heat loss)	19.6	19.6
Radiator area[a], ft^2	380	800

[a]Based on $\alpha = 0.23$, $\epsilon = .85$, Sink Temperature = 452° R

TABLE 1-28 - REACTOR BRAYTON-CYCLE SYSTEM PARAMETERS - 10 kWe

Working fluid	Argon
Turbine inlet temperature (°F)	1 200
Compressor inlet temperature (°F)	140
Shaft speed (rpm)	48 000
Compressor specific speed	0.1
Recuperator effectiveness	0.92
Pressure-loss factor (B)	0.903
Compressor inlet pressure (psia)	45.0
Compressor efficiency	.845
Turbine efficiency	.893
Power conditioning efficiency	.75
Generator efficiency	.90
Gas flow rate (lb/sec)	.99
Cycle efficiency (includes 3 percent heat loss)	13.4
Radiator area [a], ft^2	1 000

[a]Based on $\alpha = 0.23$, $\epsilon = .85$, Sink Temperature = 452° R

TABLE 1.29 · PRELIMINARY ESTIMATE OF INSTALLED PCS MODULE REQUIREMENTS

[SNAP-2 mercury Rankine systems]

System power, kWe net	Module power, kWe gross	System life, yr	PCS life, yr	Initial number active modules	Number of standby modules
5	3.33	2	1	2	6
10	4.44[a]	2	1	3	9
15	5.00[a]	2	1	4	9 - 12
20	5.00[a]	2	1	6	18 - 24

[a]Requires minor CRU modifications.

TABLE 1-30 - ISOTOPE BRAYTON-CYCLE SYSTEM COSTS

[Fiscal]

	FY68	FY69	FY70	FY71	FY72	FY73	FY74
Development	11.4	14.8	11.1	5.6	3.2		
Qualification		.3	4.1	9.2	.4		
Prequal deliv.		.5	2.1	2.9			
Production (1 flt)[a]				1.2	1.8		
Field support				.5	.7	1.2	0.4
Total	11.4	15.6	17.3	19.4	6.1	1.2	.4

Total Program 71.4

(Power Conversion Section 47.1,
Fuel Block 24.3)

[a]Does not include isotope costs.

TABLE 1.31 .- SOLAR CELL/BATTERY SYSTEM COSTS

[Fiscal]

	FY68	FY69	FY70	FY71	FY72	FY73	FY74
Development	7.2	5.9	0.4				
Qualification		2.2	4.6	0.4			
Prequal delivery		4.0	3.0				
Production (1 flt)			2.0	3.2			
Field support			0.3	0.5	0.2	0.3	0.5
Battery (Sub or GFE)	0.3	0.6	0.3	0.1			
Total SC	7.2	12.1	10.3	4.1	0.2	0.3	0.5
Total SC/B	7.5	12.7	10.6	4.2	0.2	0.3	0.5

Total Program SC 34.7
SC/B 36.0

TABLE 1.32 - NONREGENERATIVE FUEL CELL COSTS

[Fiscal]

	FY68	FY69	FY70	FY71	FY72	FY73	FY74
Development	4.2	6.2	3.0				
Qualification		2.1	3.5	0.8			
Prequal deliv.		1.0	1.0				
Production (1 flt)[a]			.6	1.4	1.3	1.2	0.7
Field support			.3	.5	.3	.6	.4
Total	4.2	9.3	8.4	2.7	1.6	1.8	1.1
Total program $29.1 million							

[a]One 50-cu ft hydrogen tank charged to fuel cell subsystem cost per 6 months for 2-yr flight.

TABLE 1.33 ISOTOPE THERMOELECTRIC SYSTEM COSTS

[4 kW]

	Power section	Fuel block	Total
Development	22.0	7.5	29.5
Qualification	5.0	6.5	11.5
Prequal deliv.	3.0	2.0	5.0
Production[a]	1.5	.9	2.4
Field support	1.0	1.5	2.5
Total	32.5	18.4	49.9

[a]Does not include isotope cost.

TABLE 1.34 - REACTOR THERMOLETRIC SYSTEM COSTS

[20 kW]

	Power section	Reactor	Total
Development	24.0	38.7	62.7
Qualification	5.6	18.0	23.6
Prequal deliv.	4.5	2.2	6.7
Production[a]	6.0	1.4	7.4
Field support	1.5	2.0	3.5
Total	41.6	62.3	103.9

[a]All nuclear costs included.

TABLE 1.35 - REACTOR BRAYTON-CYCLE SYSTEM COSTS

[20 kW]

	Power section	Reactor	Total
Development	38.0	38.7	76.7
Qualification	5.6	18.0	23.6
Prequal Deliv.	3.4	2.2	5.6
Production[a]	2.0	1.4	3.4
Field Support	1.5	2.0	3.5
Total	50.5	62.3	112.8

[a]All nuclear costs included.

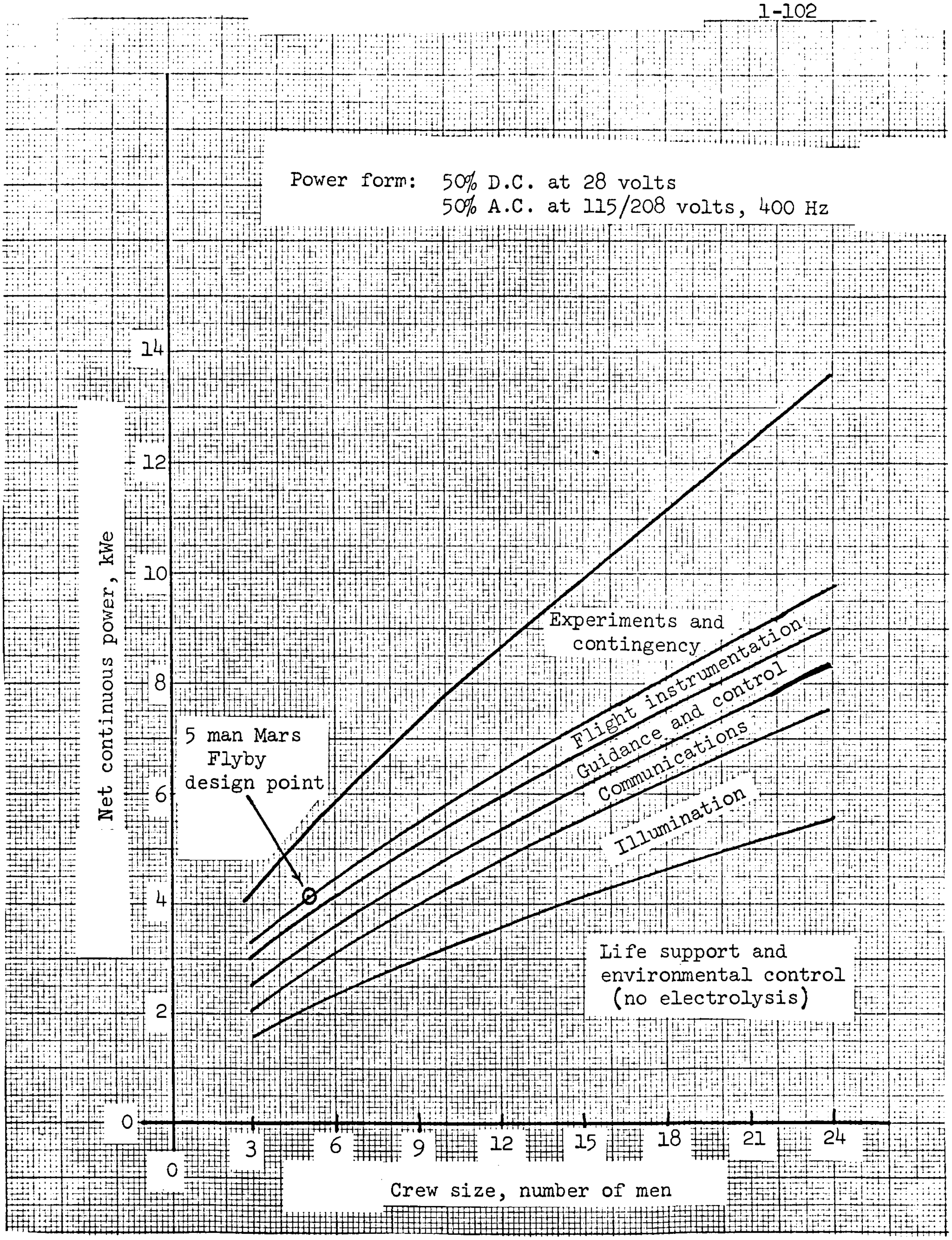


Figure 1.7 - Space station continuous power requirements versus crew size.

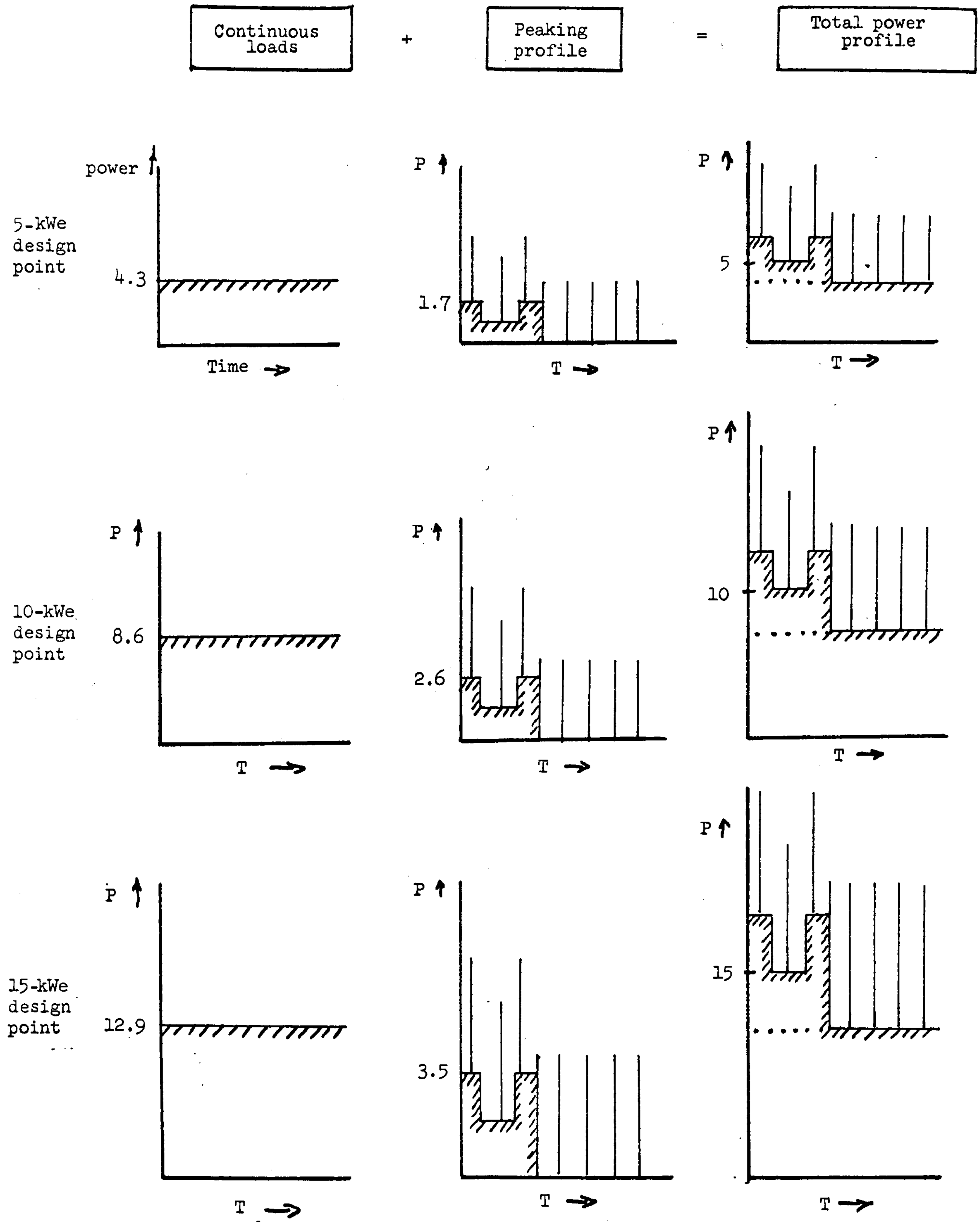


Figure 1.8 - Logic for determination of power profiles.

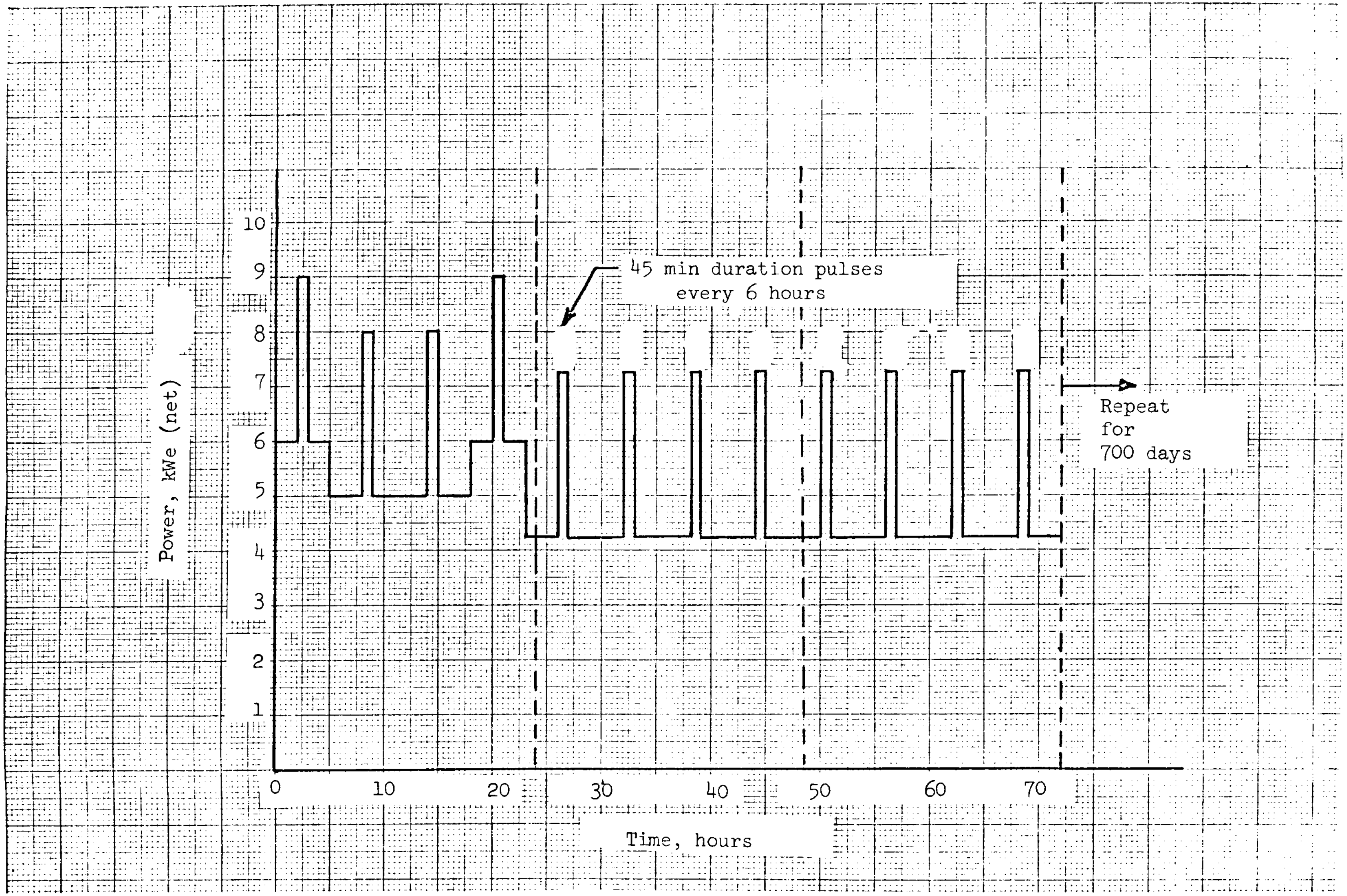


Figure 1.9 - Power profile of 260 nautical mile Earth orbit.

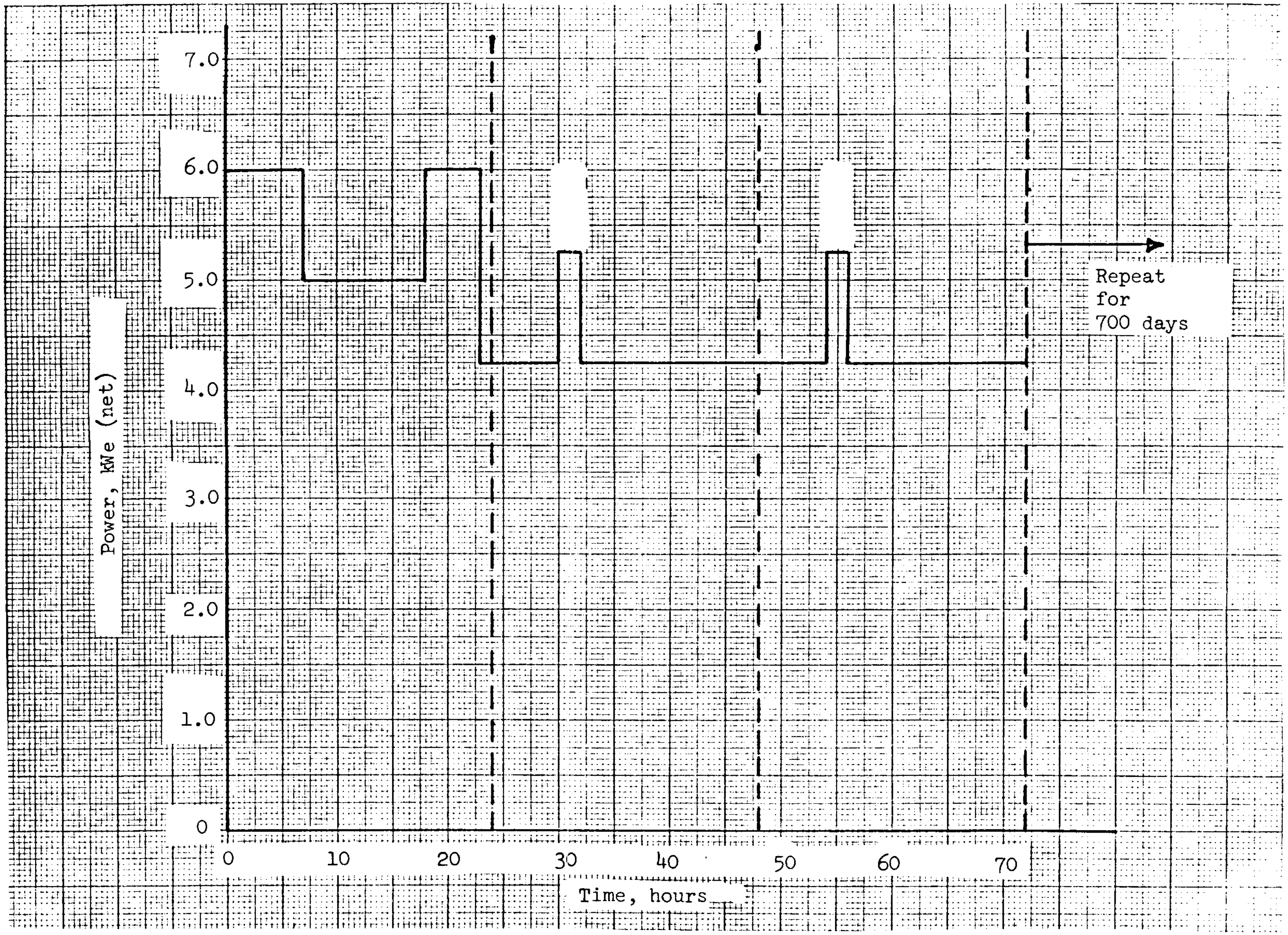


Figure 1.10 - Synchronous Earth orbit power profile.

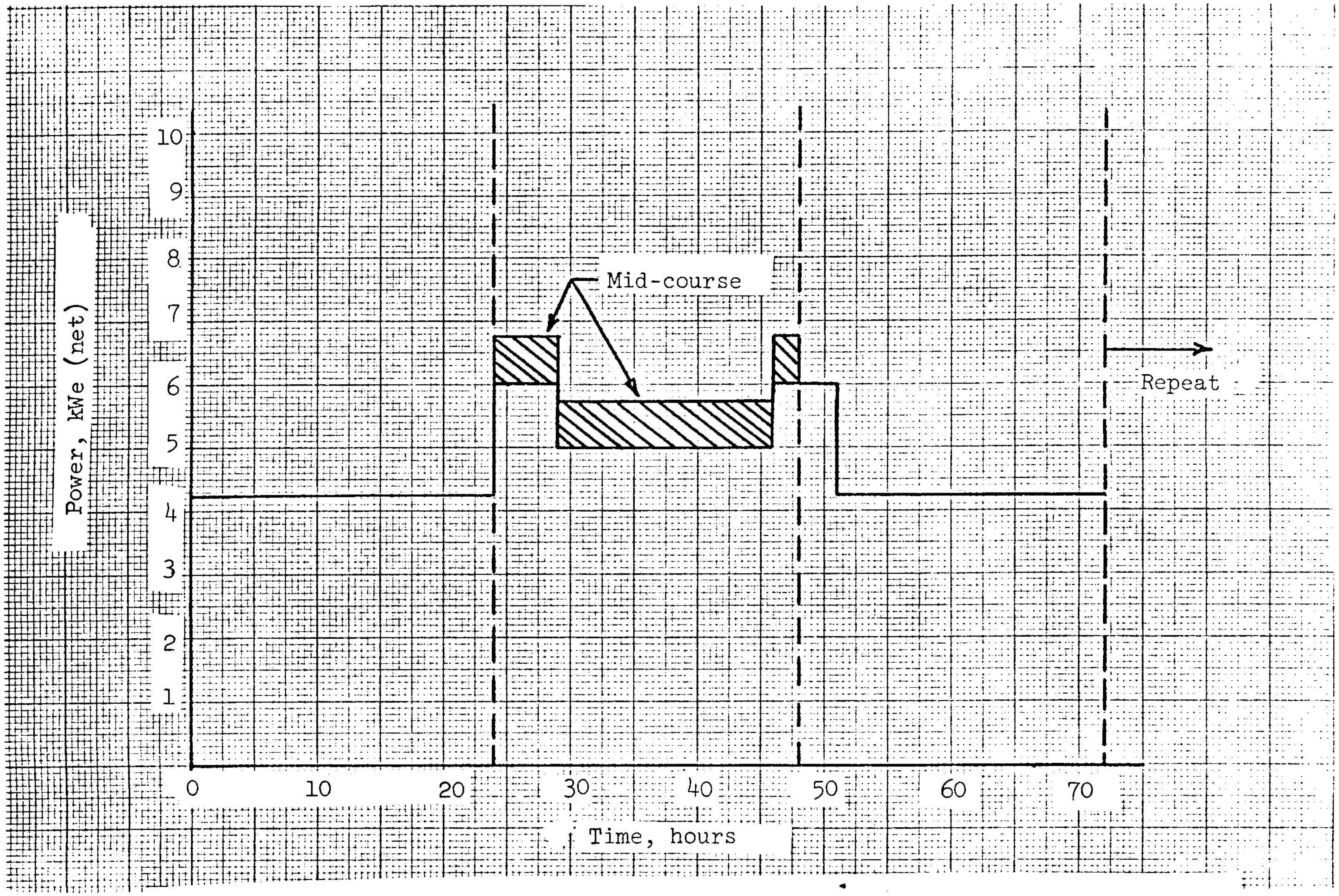


Figure 1.11 - Power profile, Mars flyby day No. 001 through day No. 119, midcourse day No. 8 and day No. 80.

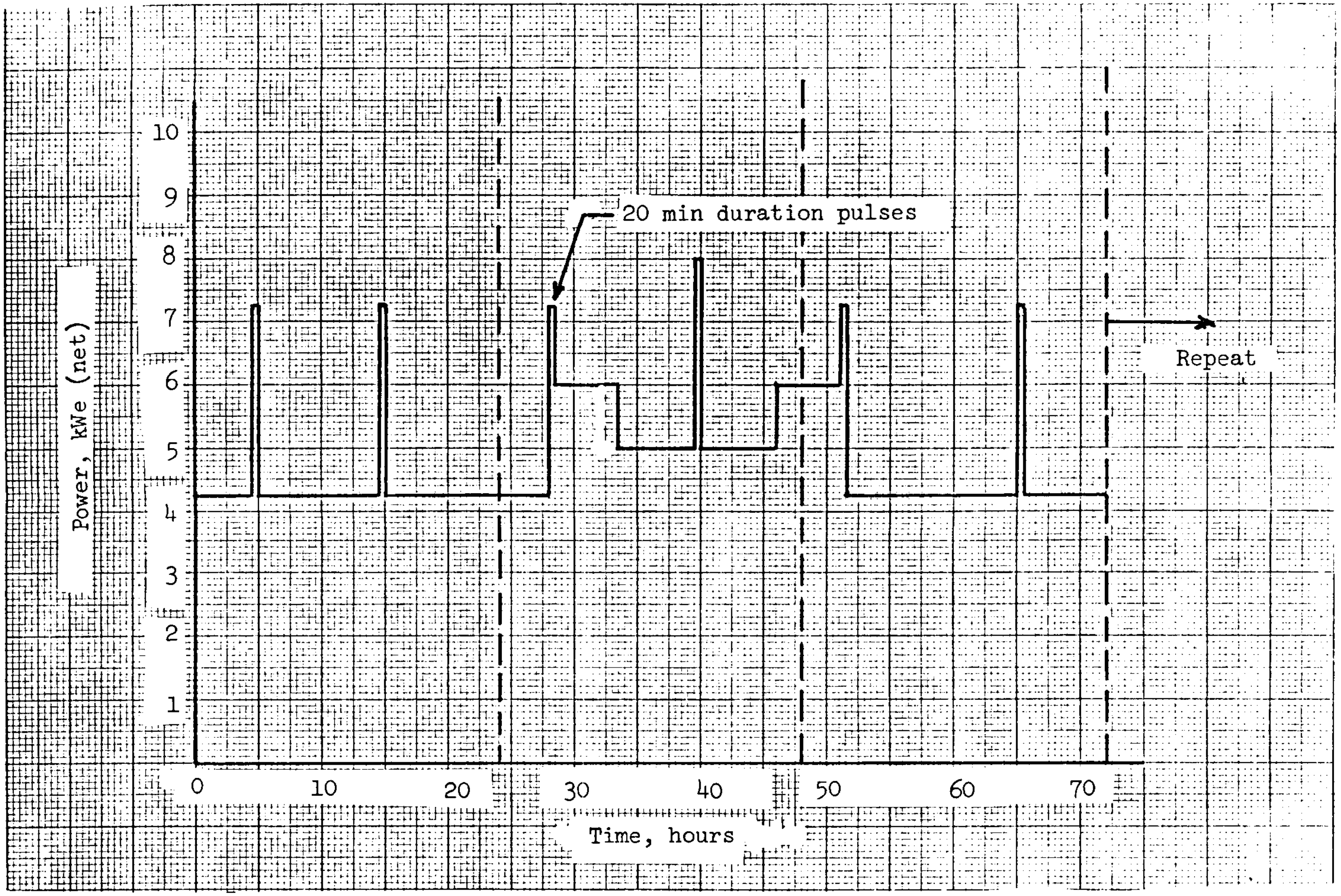


Figure 1.12 - Mars flyby power profile day No. 120 through day No. 139

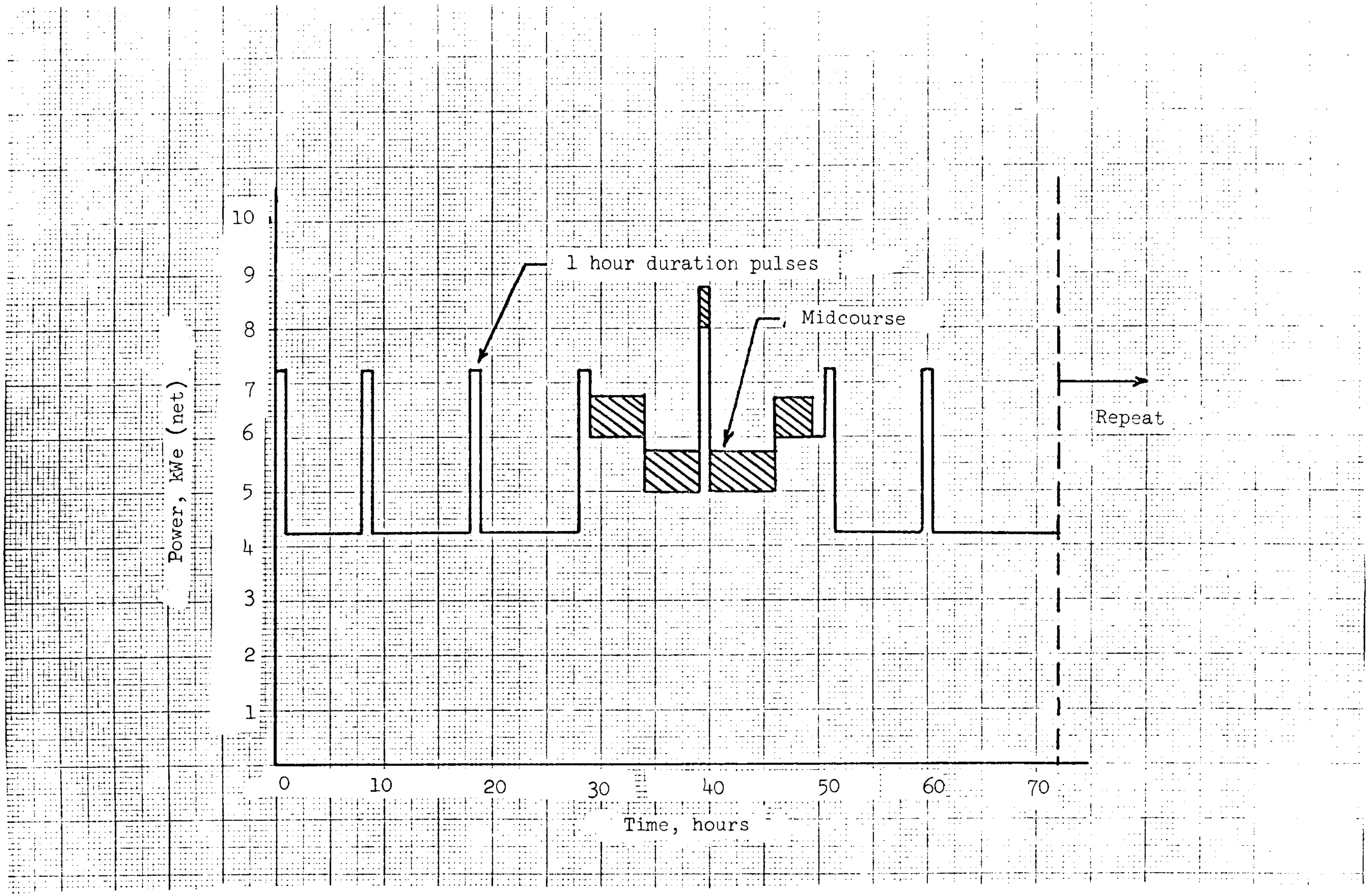


Figure 1.13- Mars flyby power profile day No. 140 through day No. 144, midcourse day No. 140.

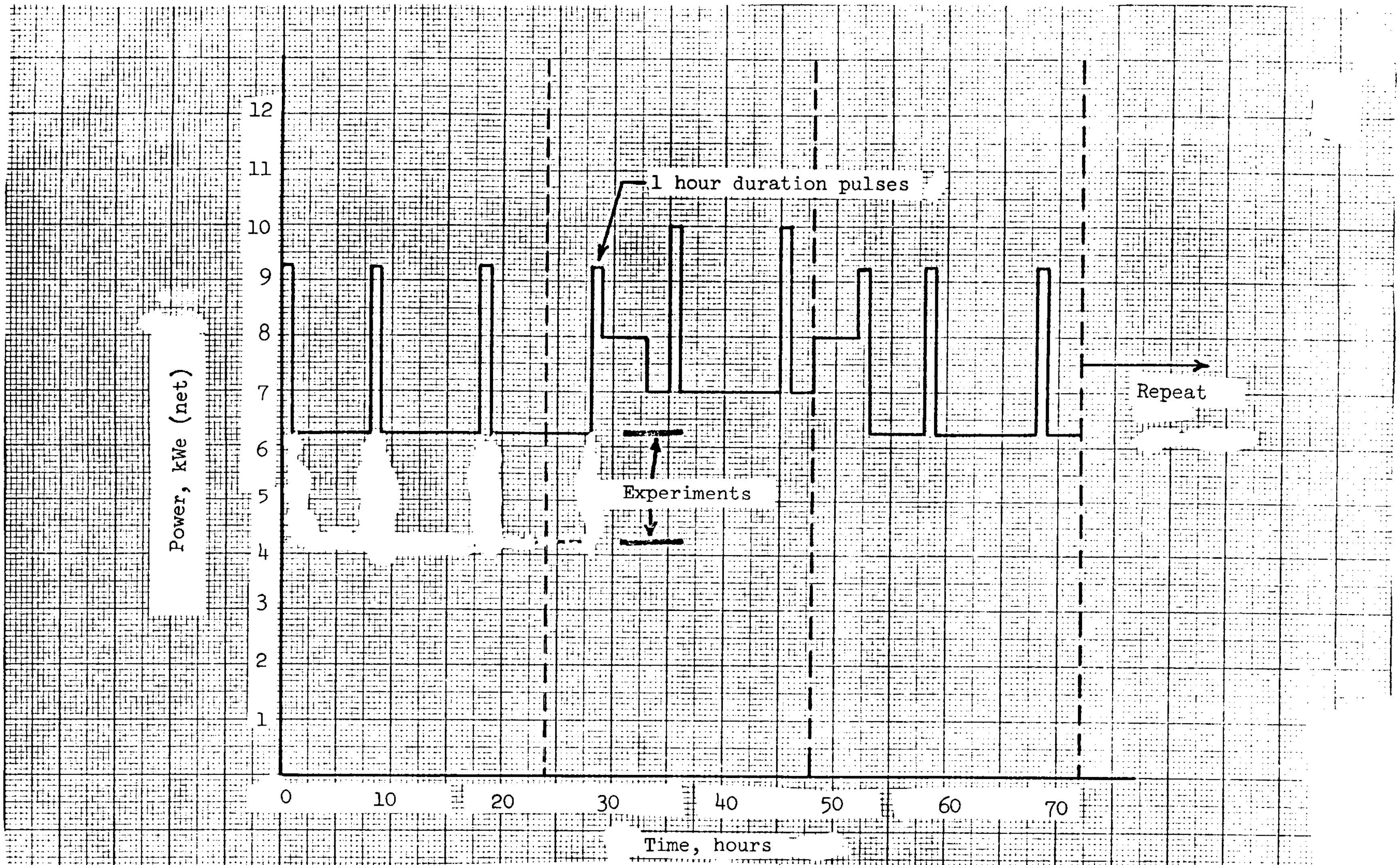


Figure 1.14 - Mars flyby power profile day No. 145 through day No. 154.

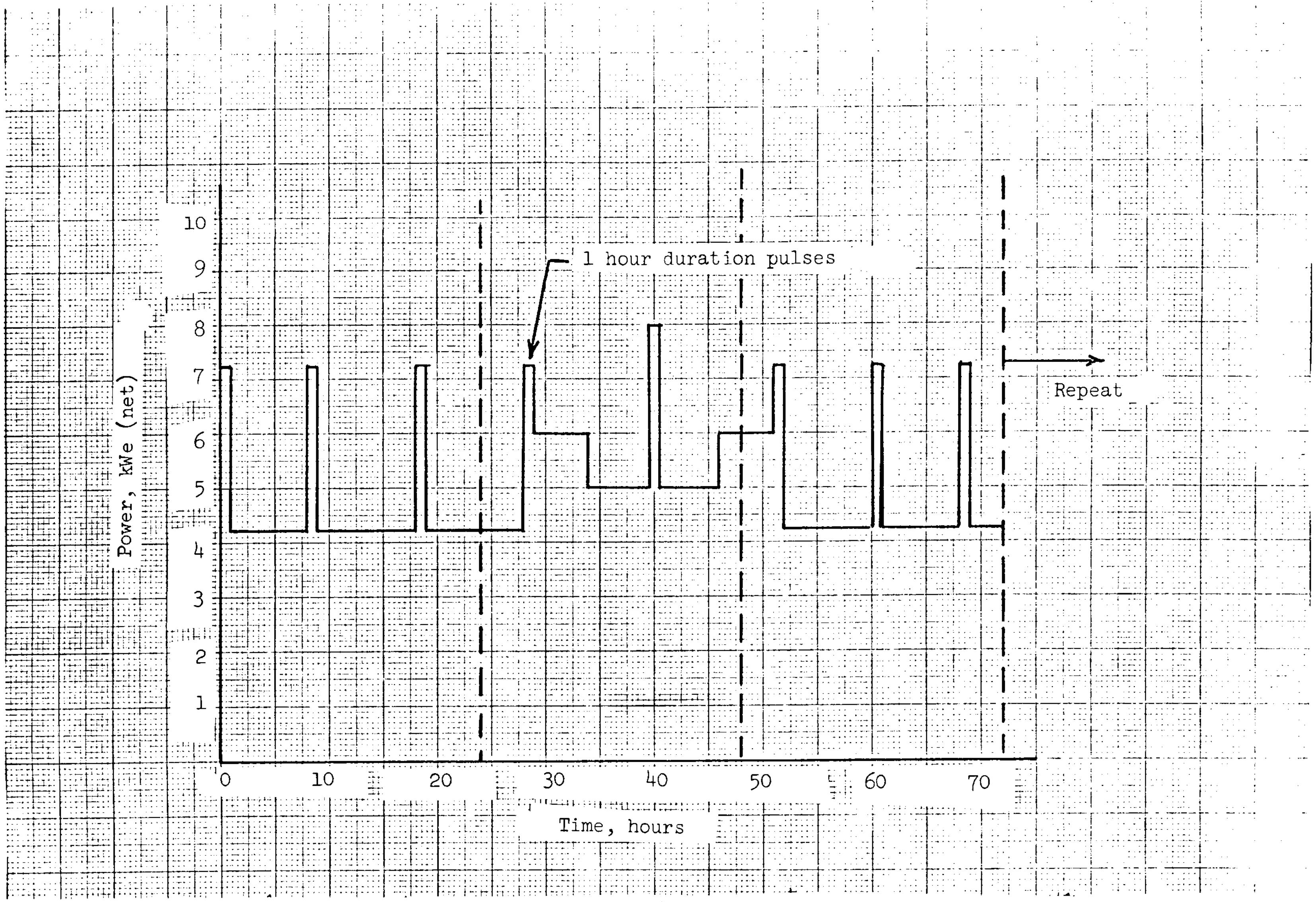


Figure 1.15 - Mars flyby power profile day No. 155 through day No. 159.

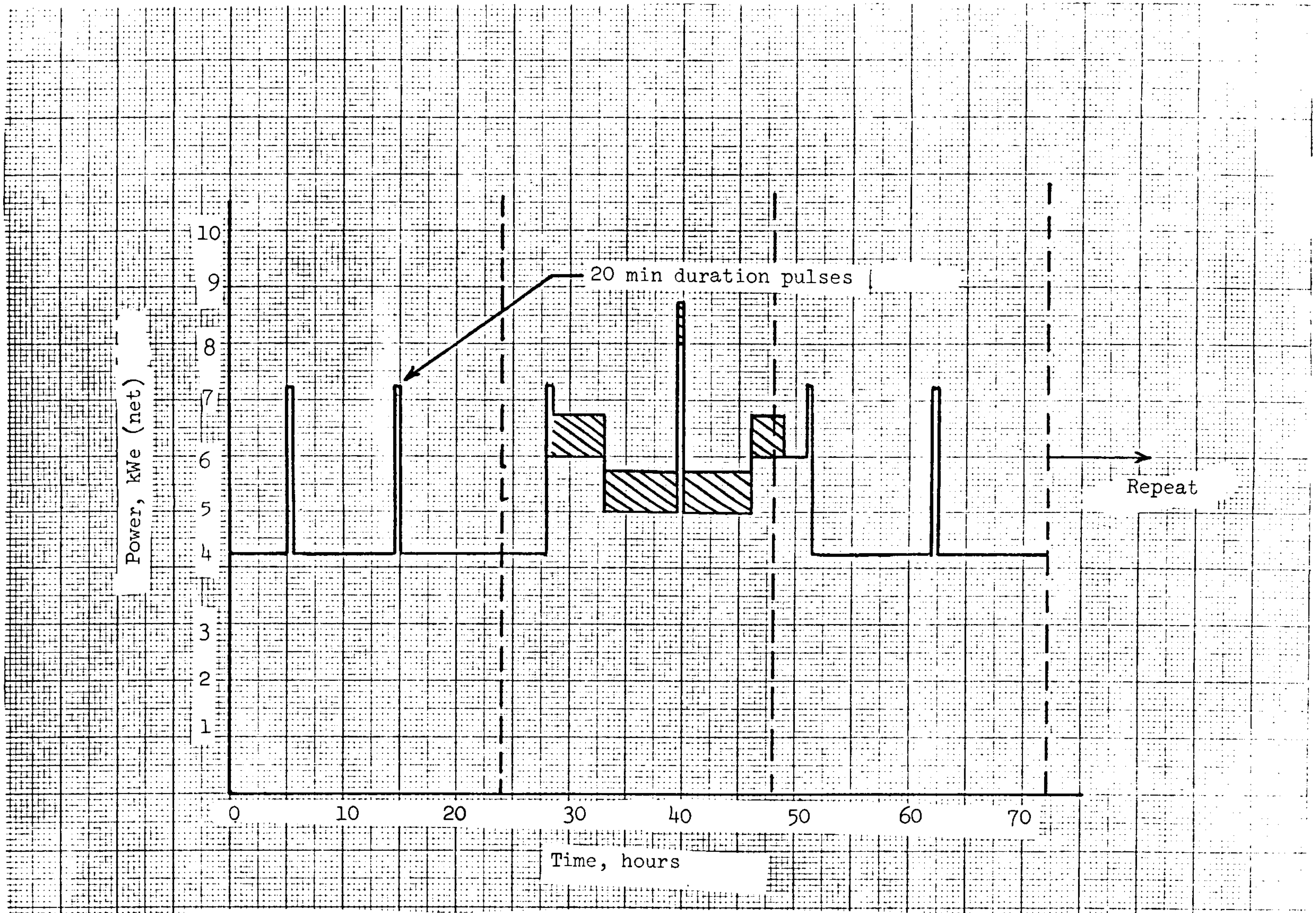


Figure 1.16 . - Mars flyby power profile day No. 160 through day No. 550, midcourse day No. 160 and day No. 400

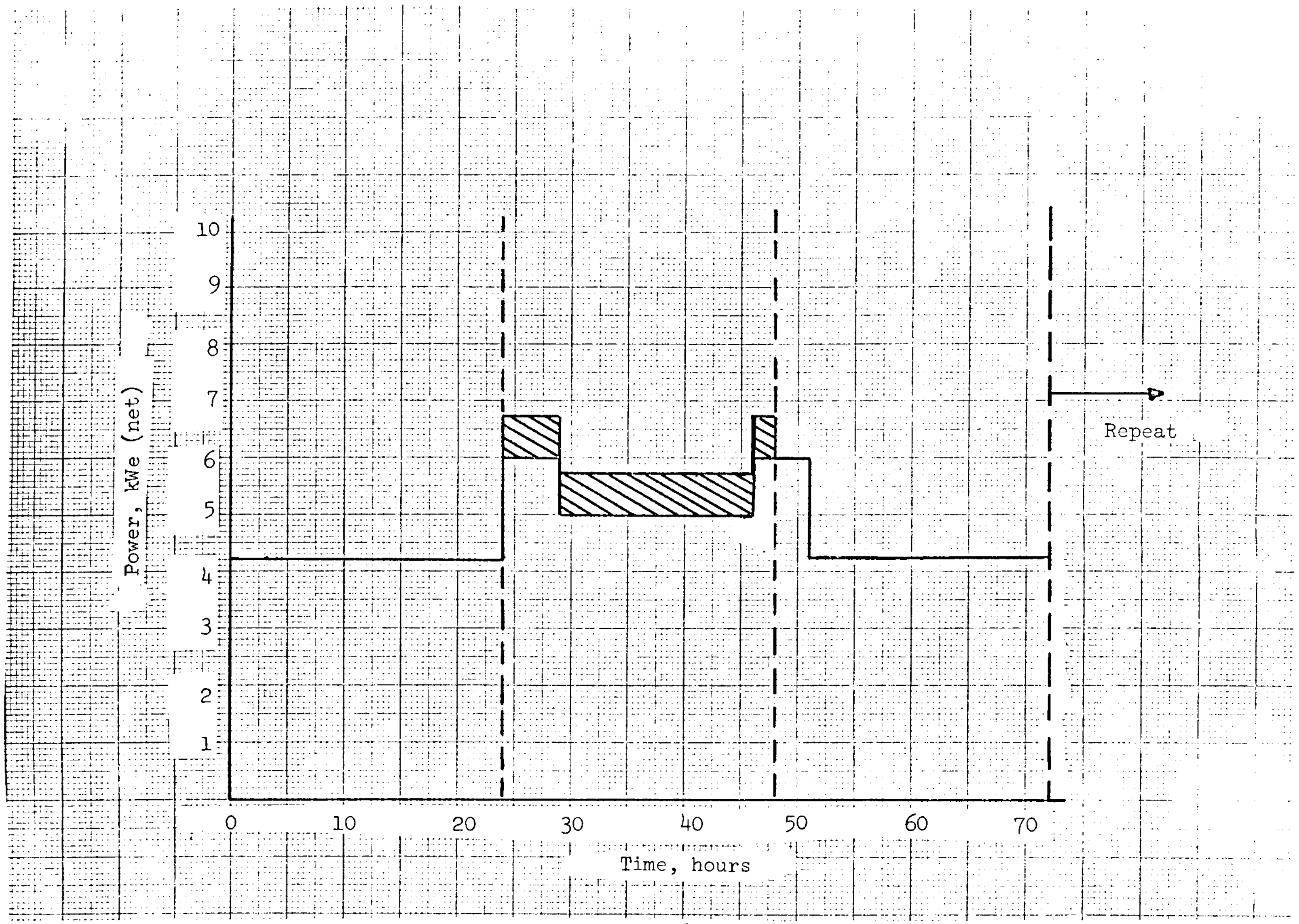


Figure 1.17 - Mars flyby power profile day No. 551 through day No. 700, midcourse day No. 680

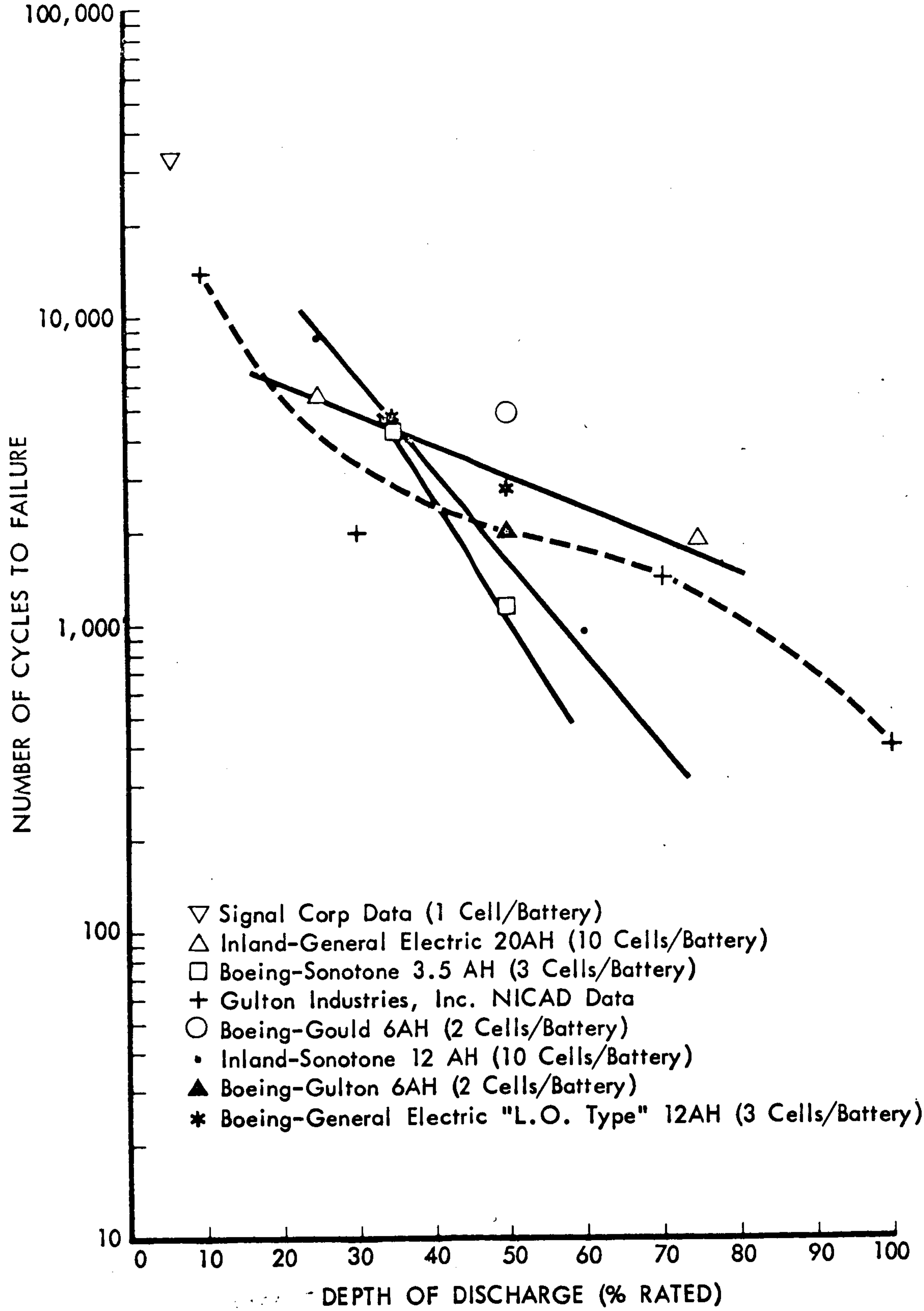


Figure 1.18 - Nickel cadmium cell cycle life data.

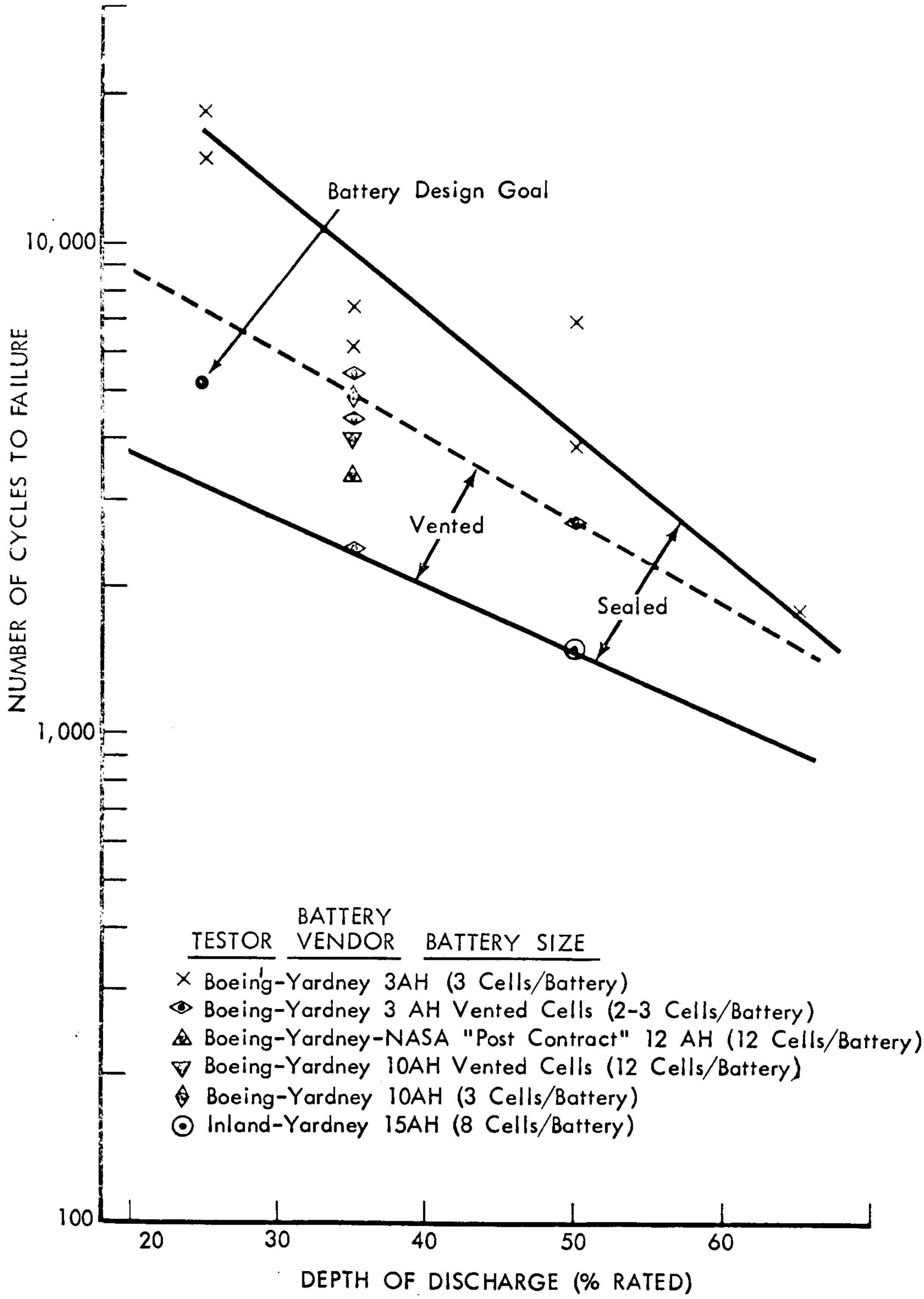


Figure 1.19 - Silver cadmium cell cycle life data.

A 127

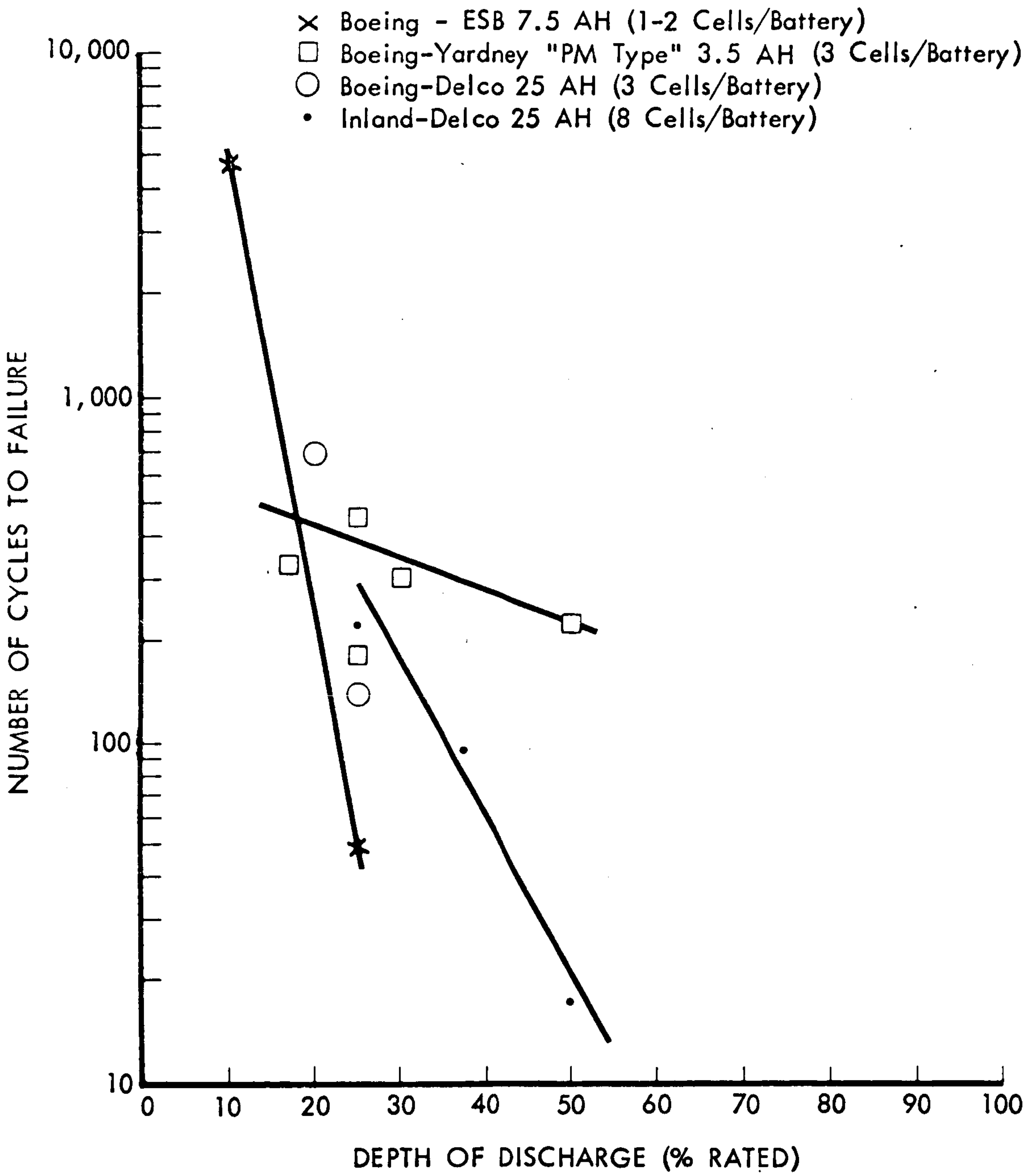


Figure 1.20 **- Silver zinc cell cycle life data.**

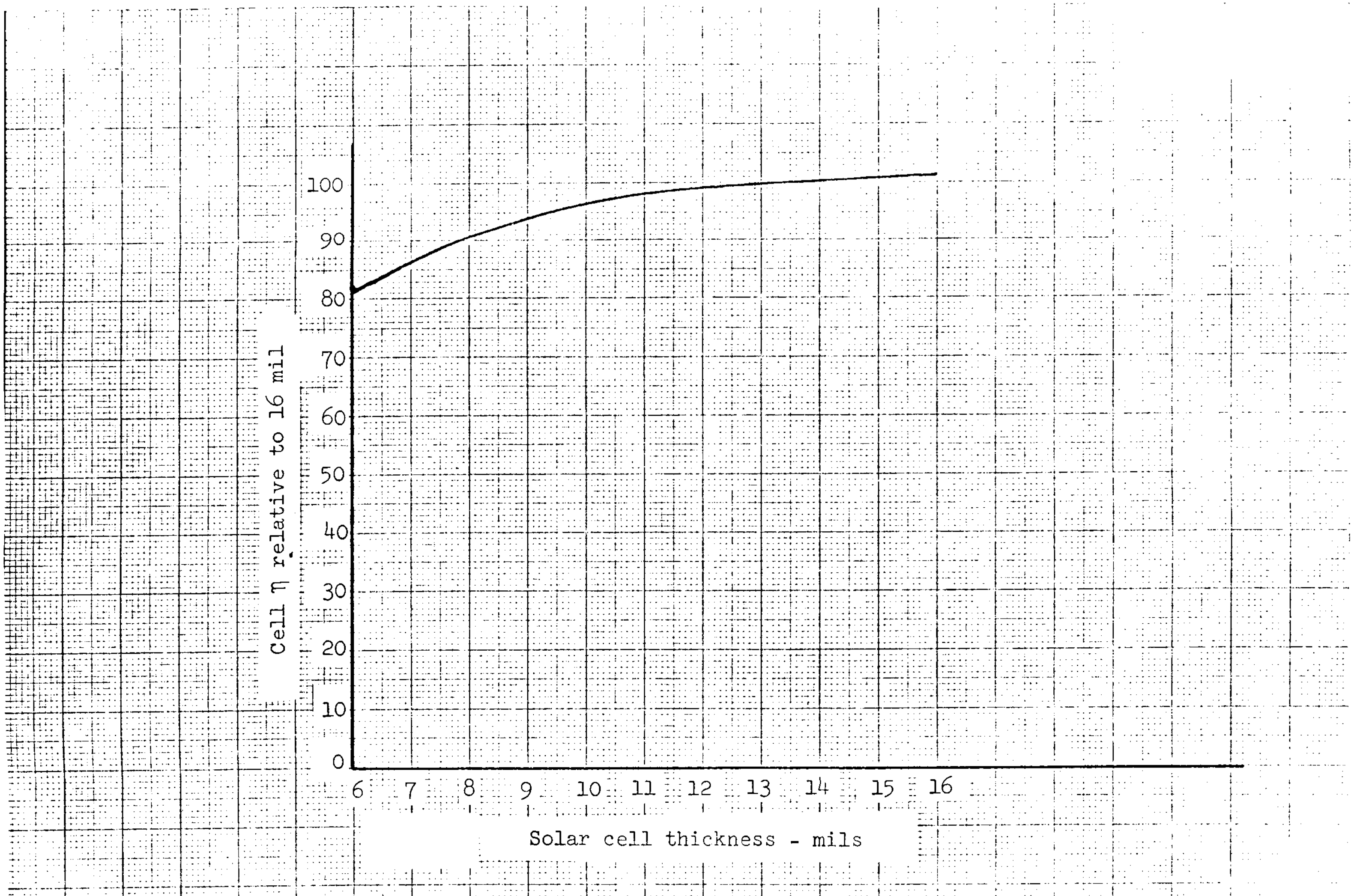


Figure 1.21 - Solar cell efficiency versus thickness.

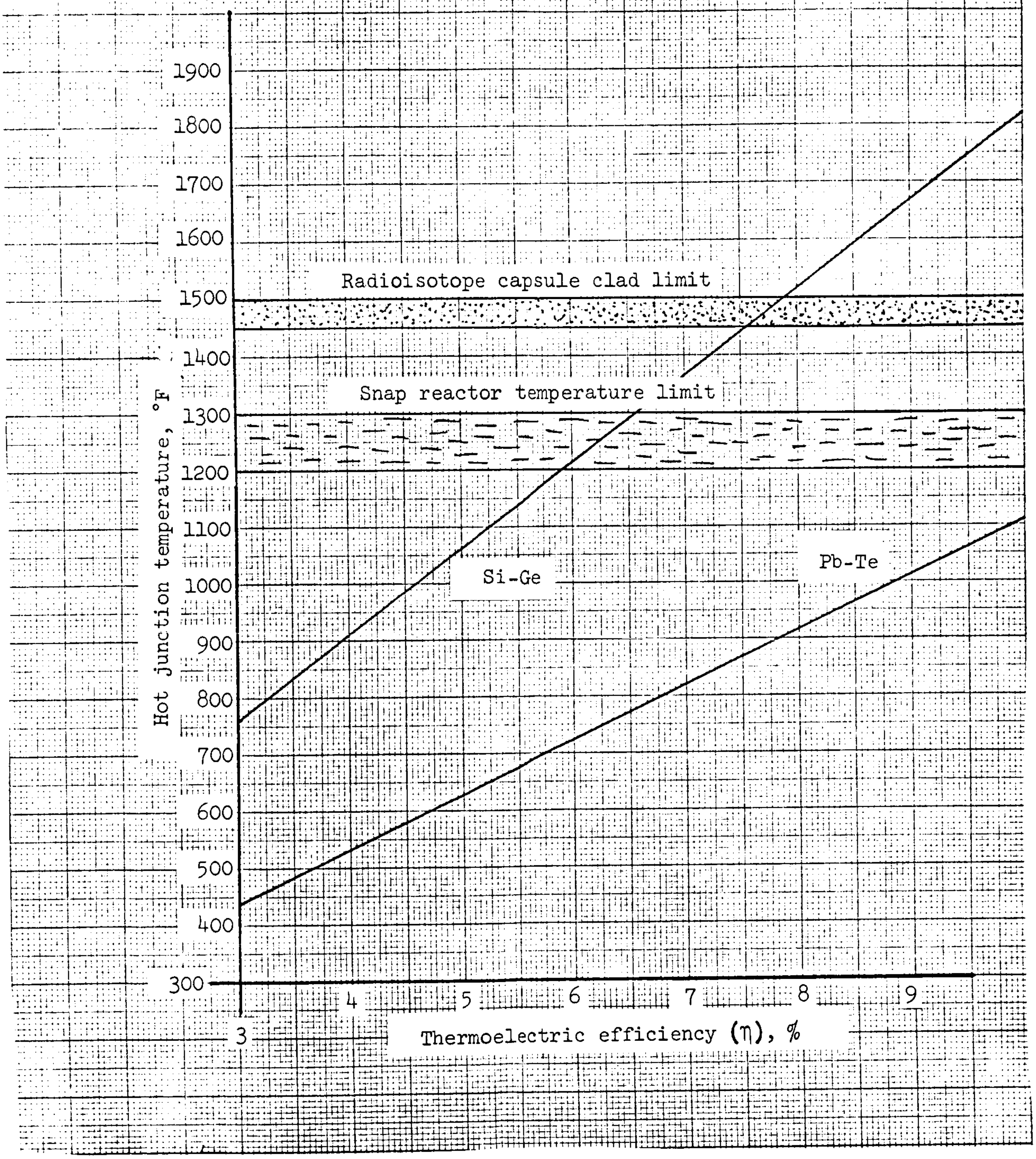


Figure 1.22 - Hot junction temperature versus thermoelectric efficiency.

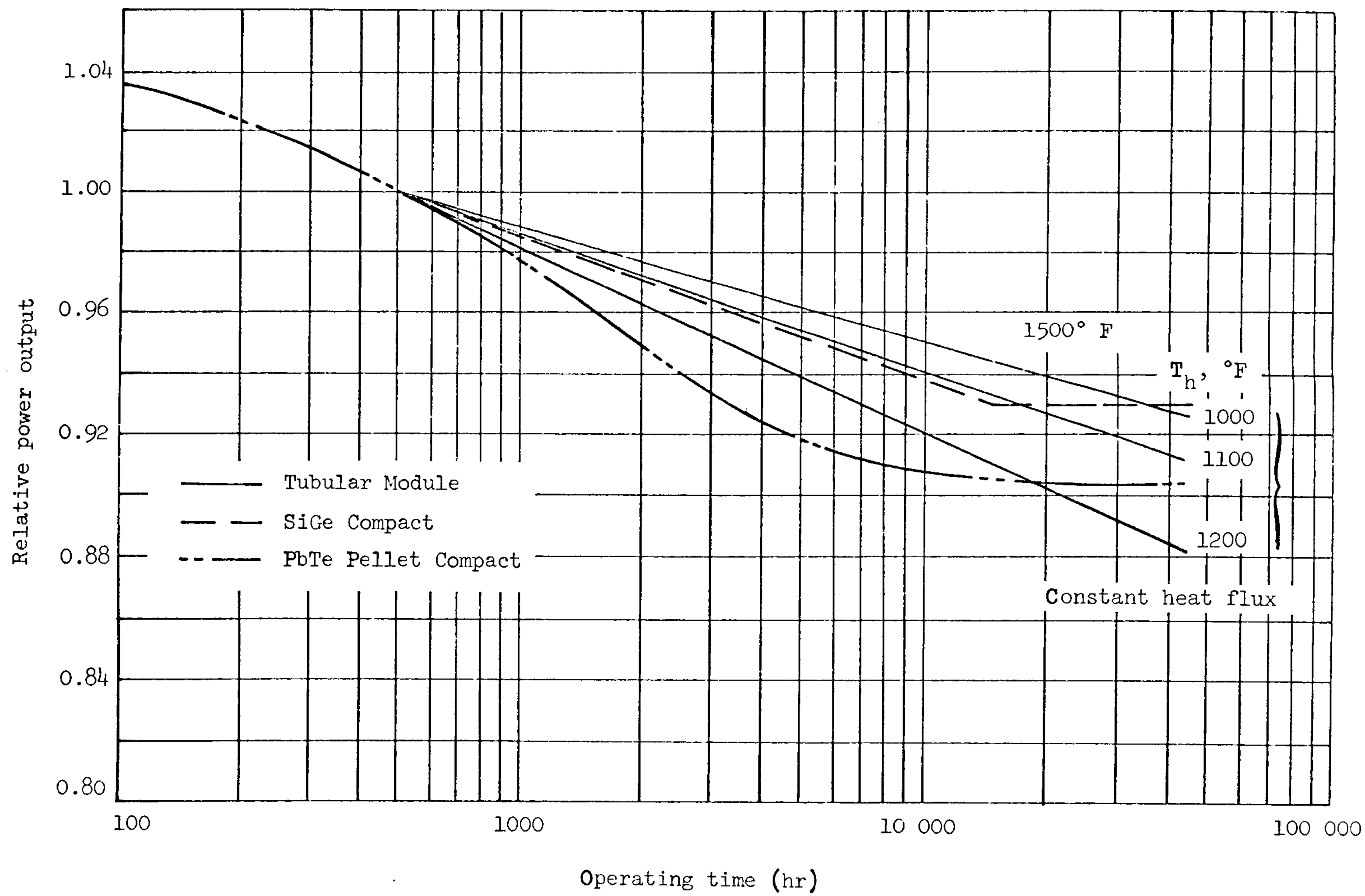


Figure 1.23 - Degradation of thermoelectric convertors.

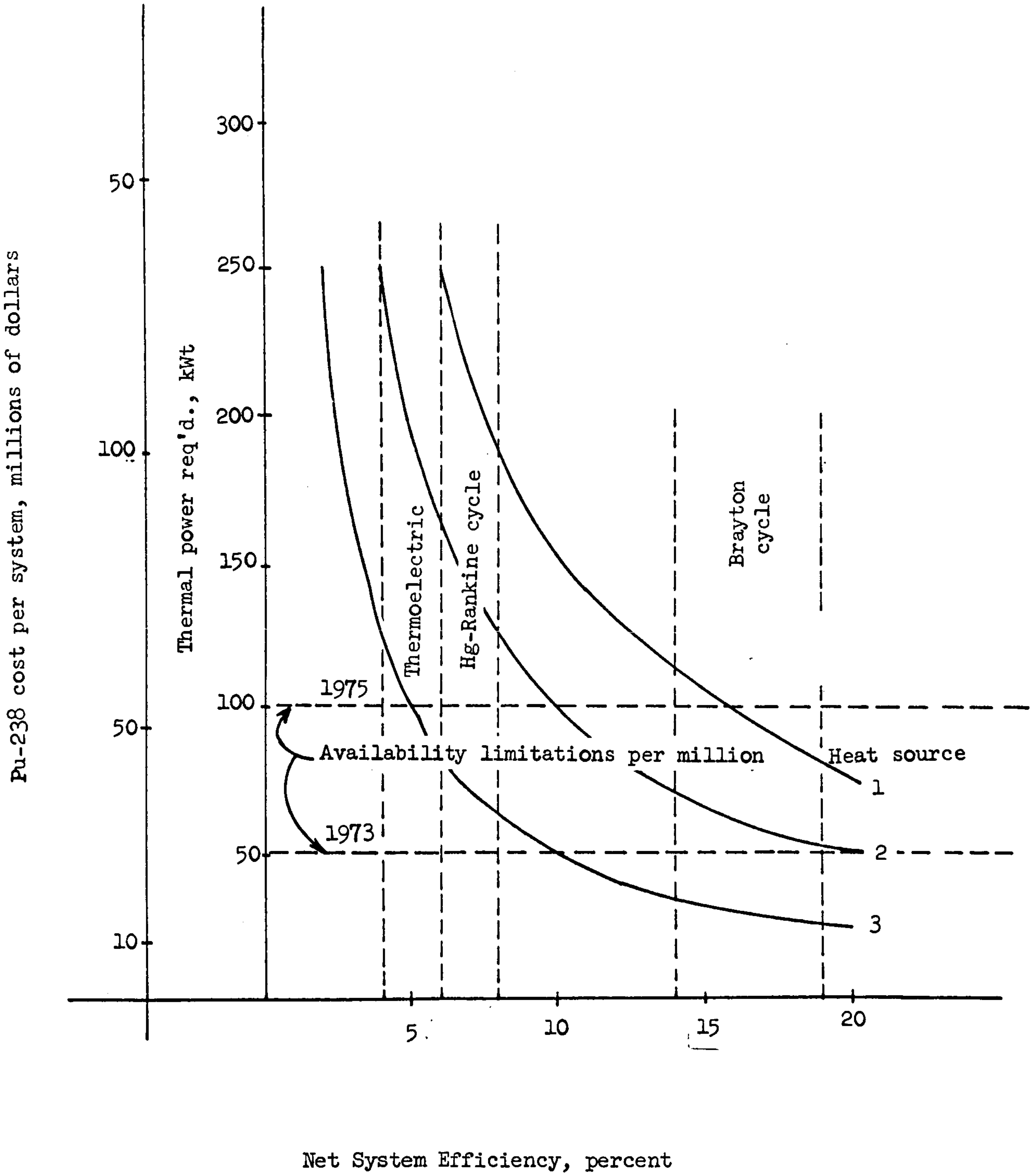


Figure 1.24 - Required thermal power and cost versus net system efficiency.

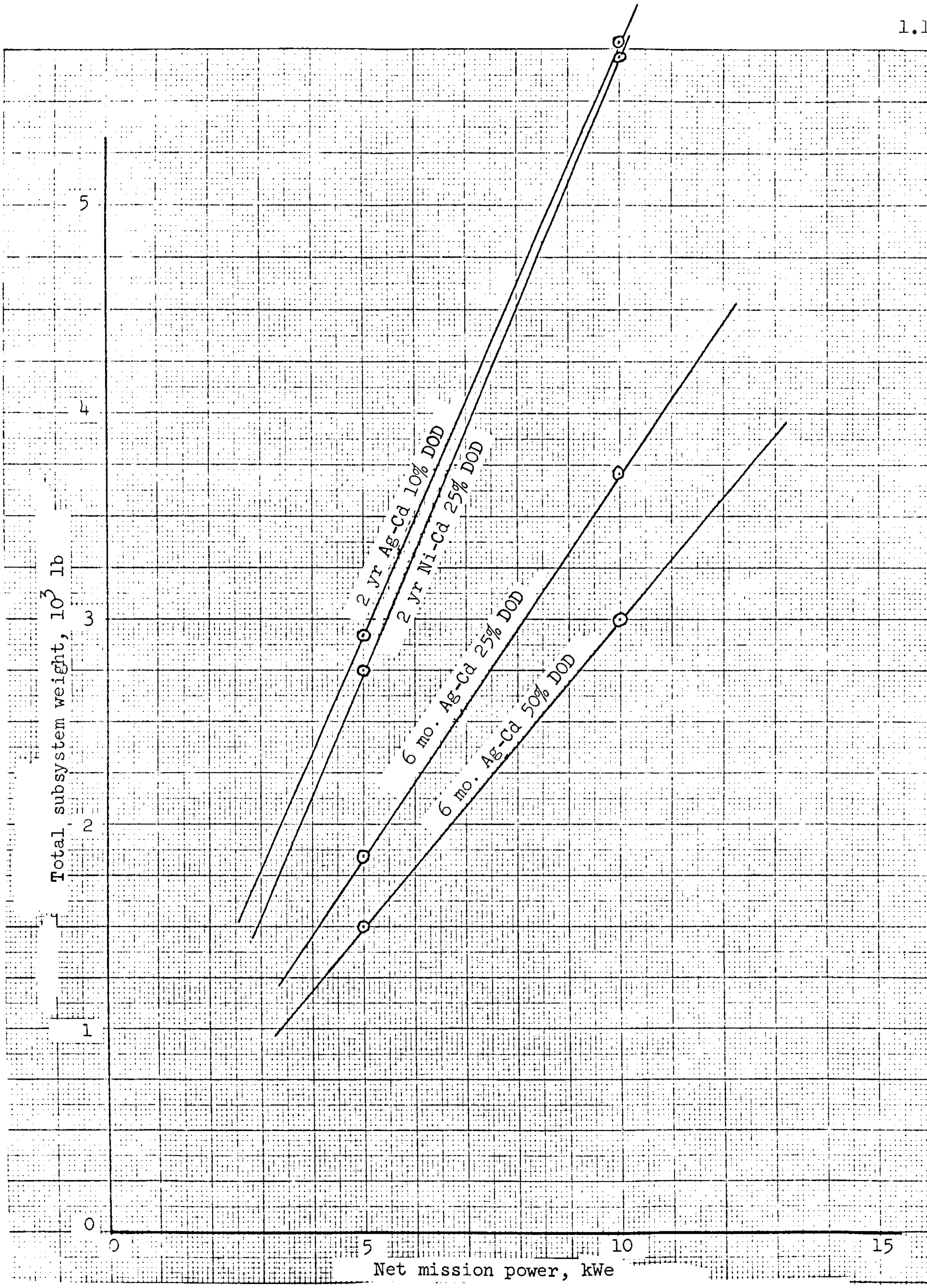


Figure 1.25 - Low Earth orbit battery complementarty power subsystems.

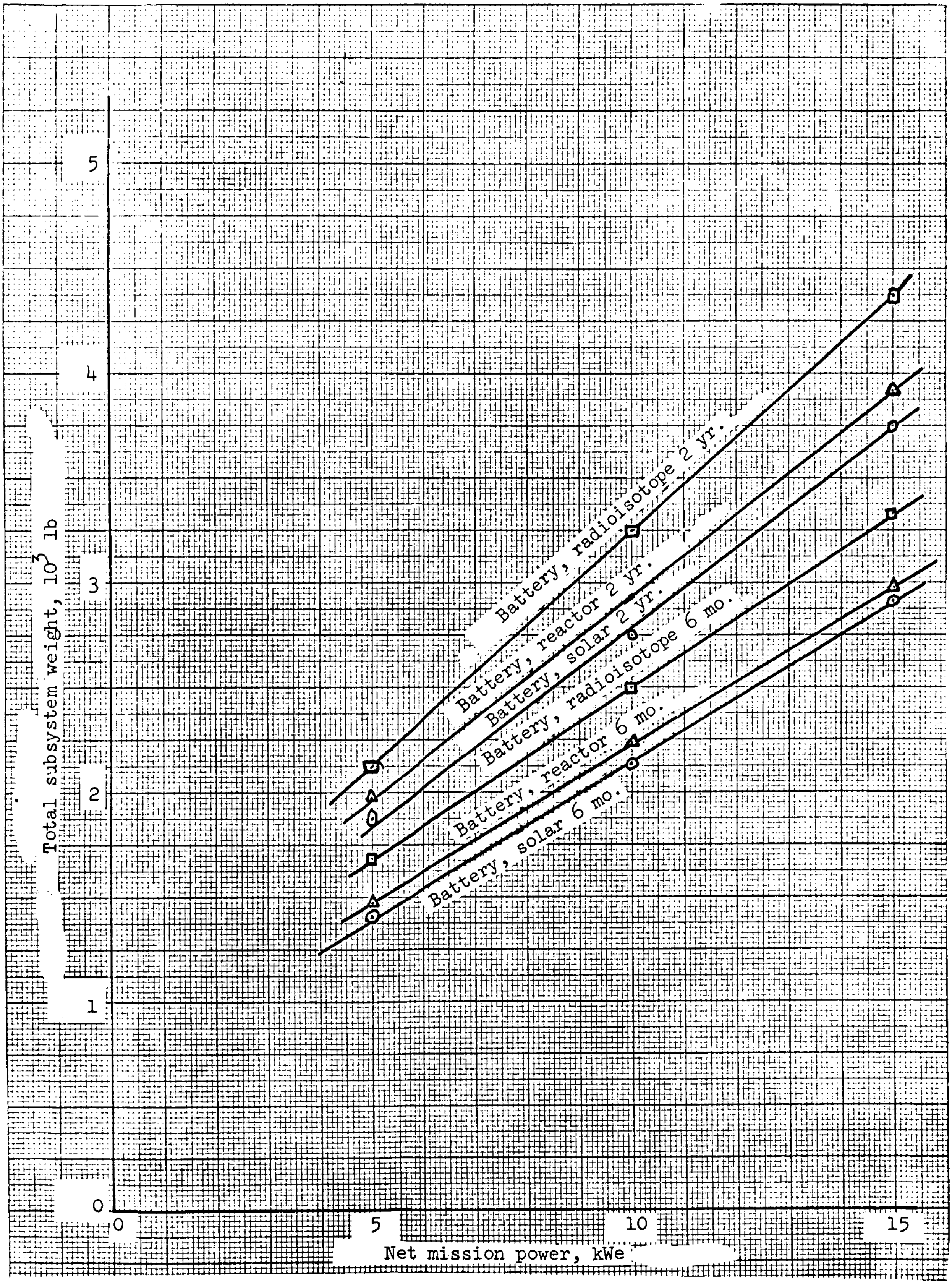


Figure 1.26 - Low Earth orbit battery supplementary power subsystems.

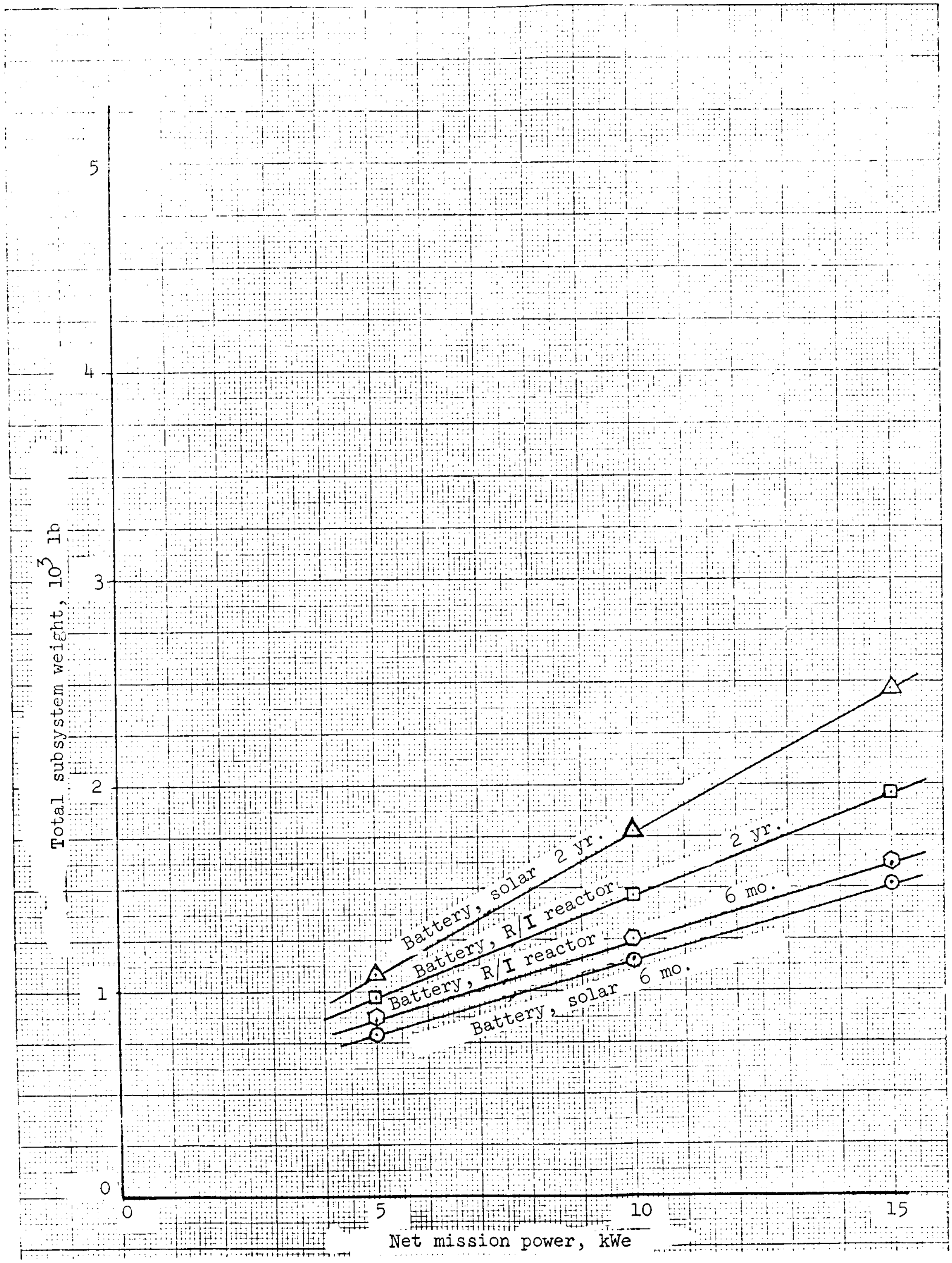


Figure 1.27 . - Synchronous Earth orbit battery supplementary power subsystems.

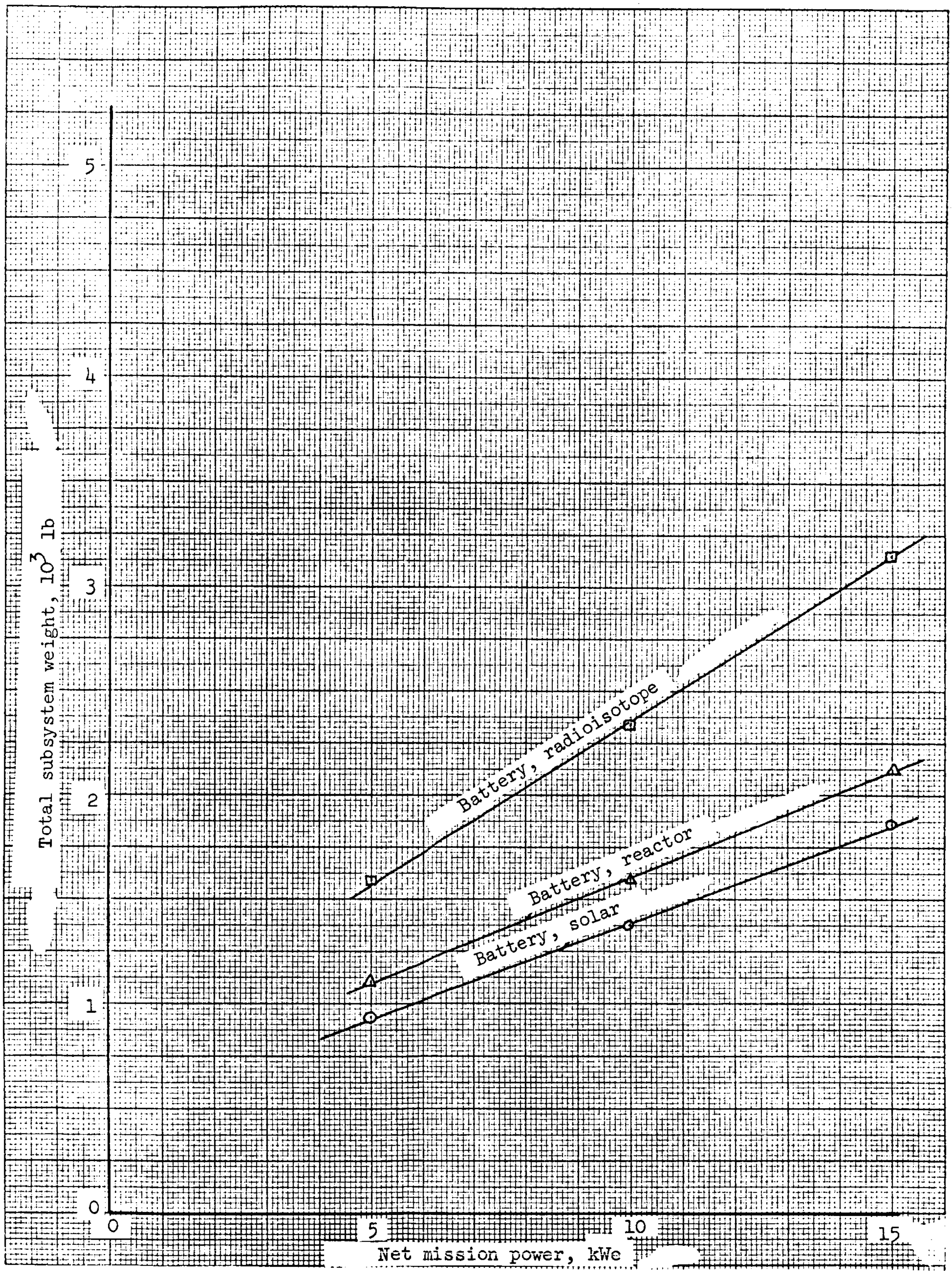


Figure 1.28 - Mars flyby battery supplementary power subsystems.

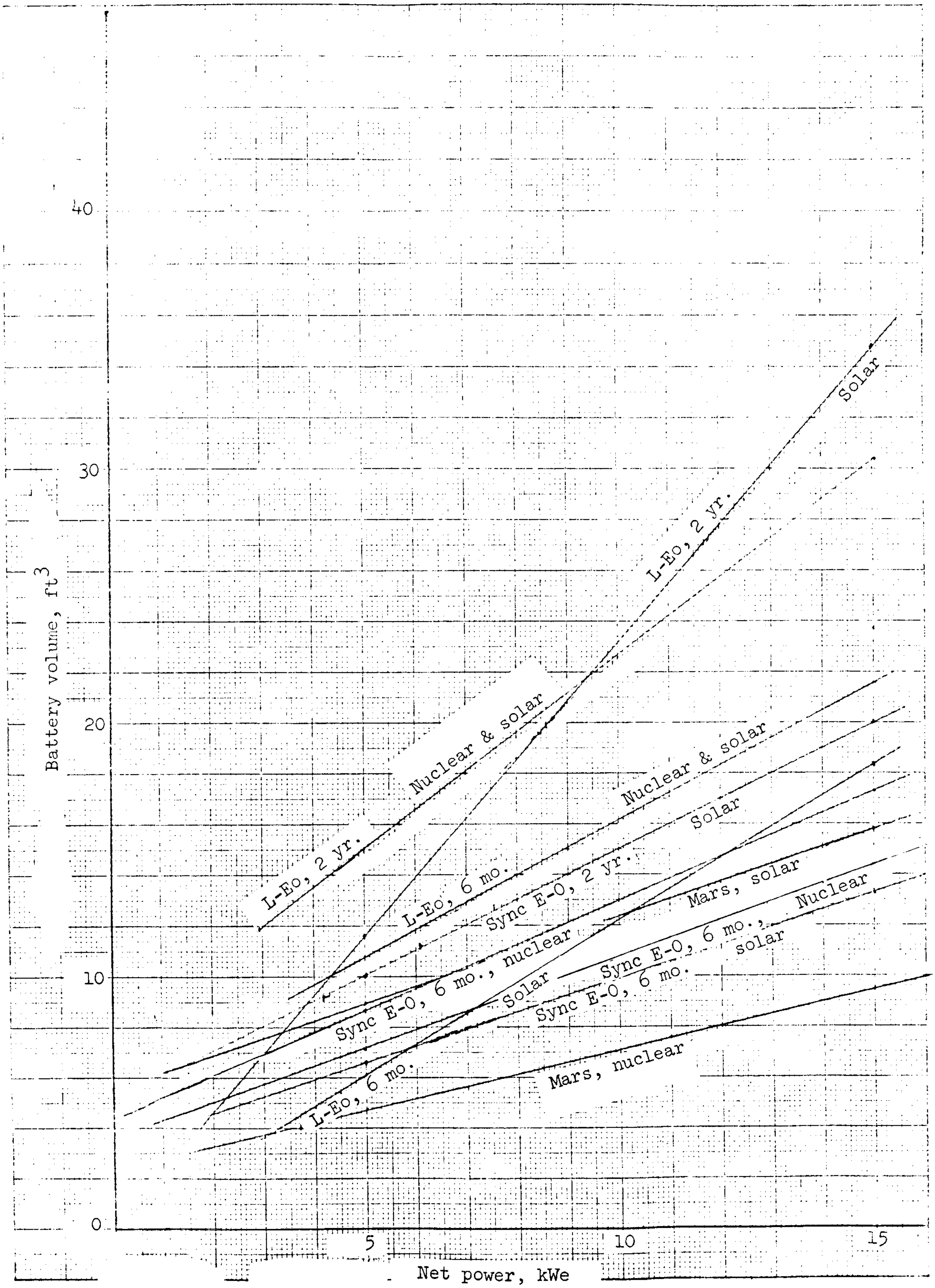


Figure 1.29 - Battery volumes versus power level for all missions.

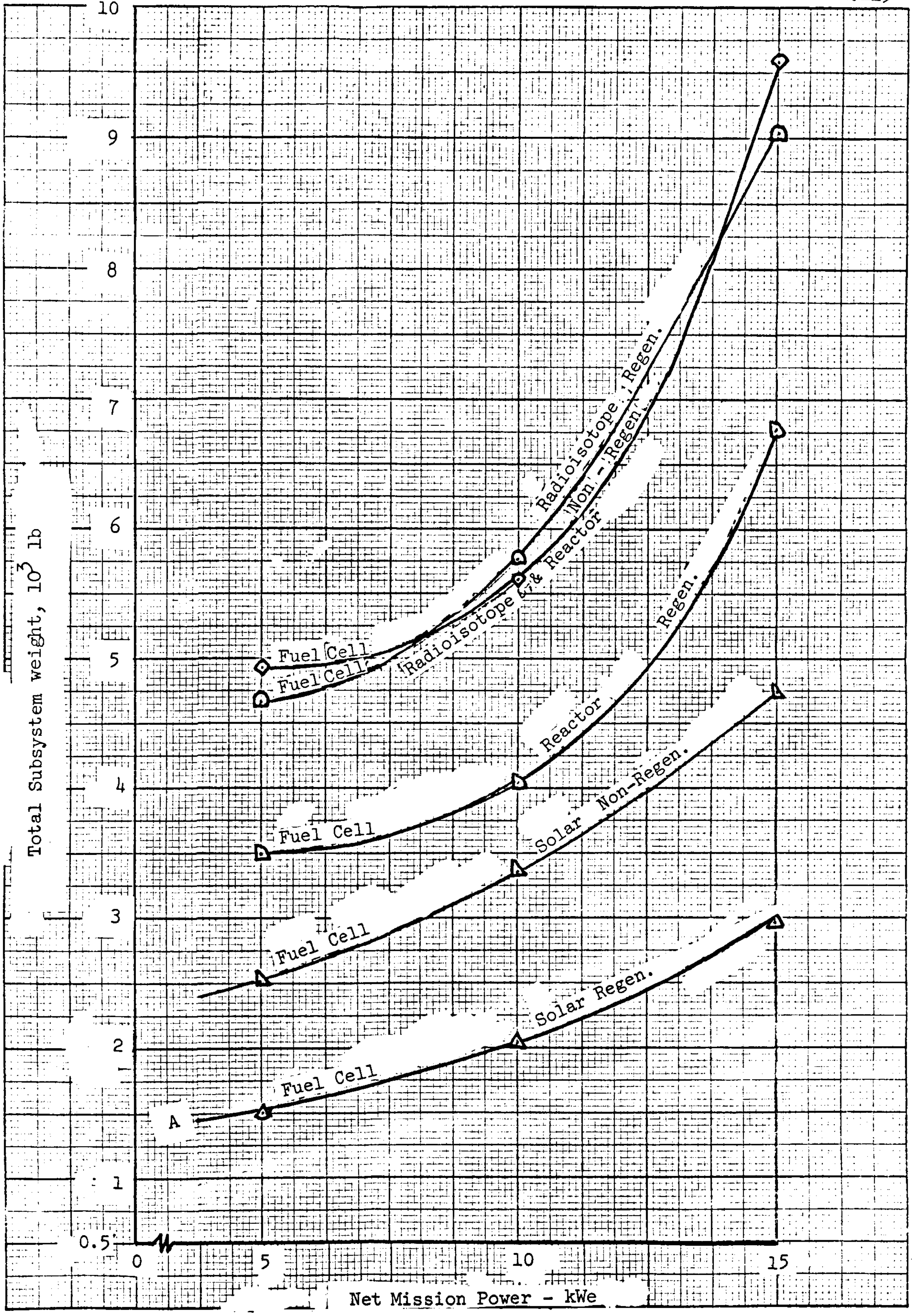


Figure 1.30 - Mars flyby fuel cell supplementary power subsystems.

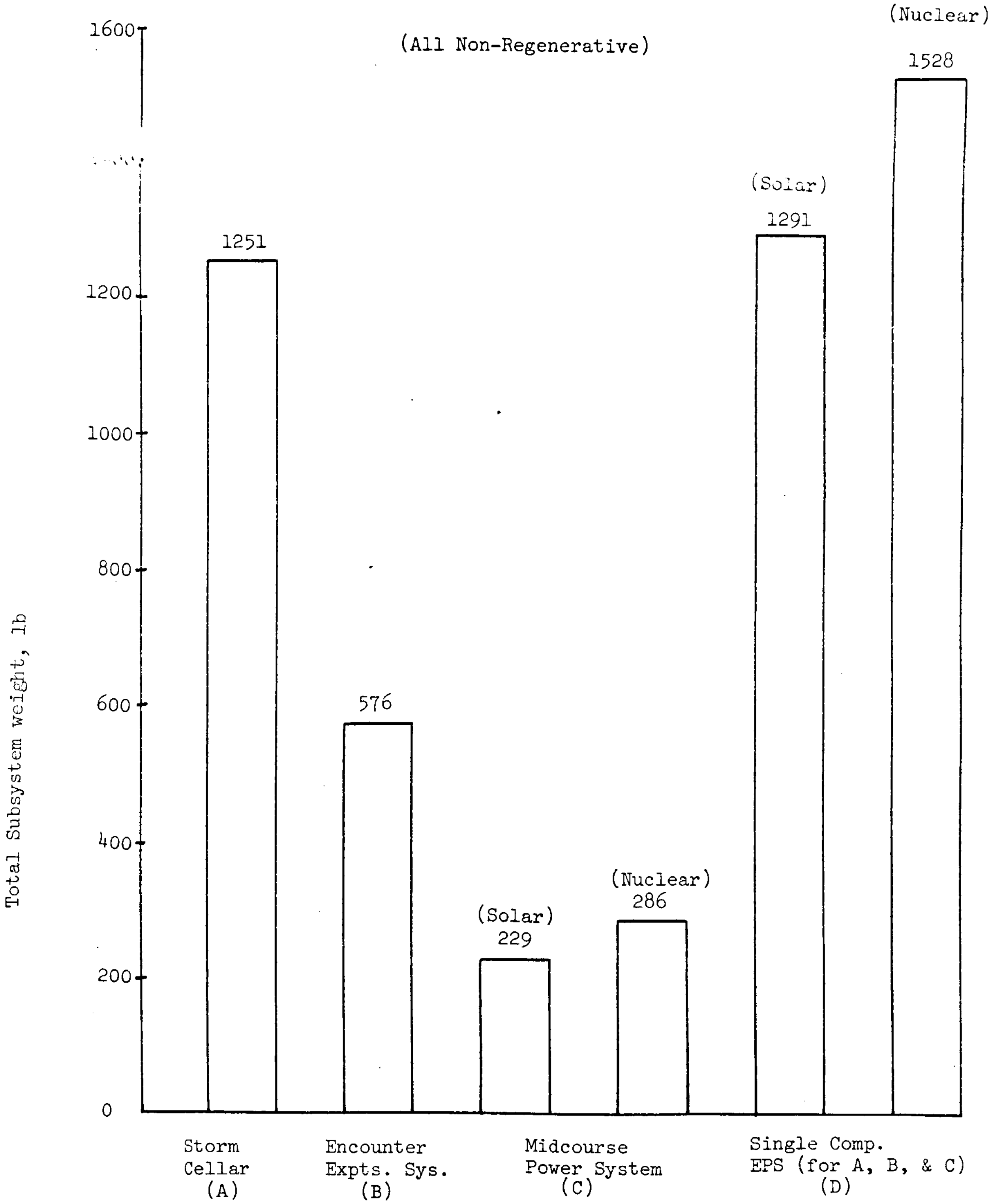


Figure 1.31 - Mars flyby fuel cell power subsystems.

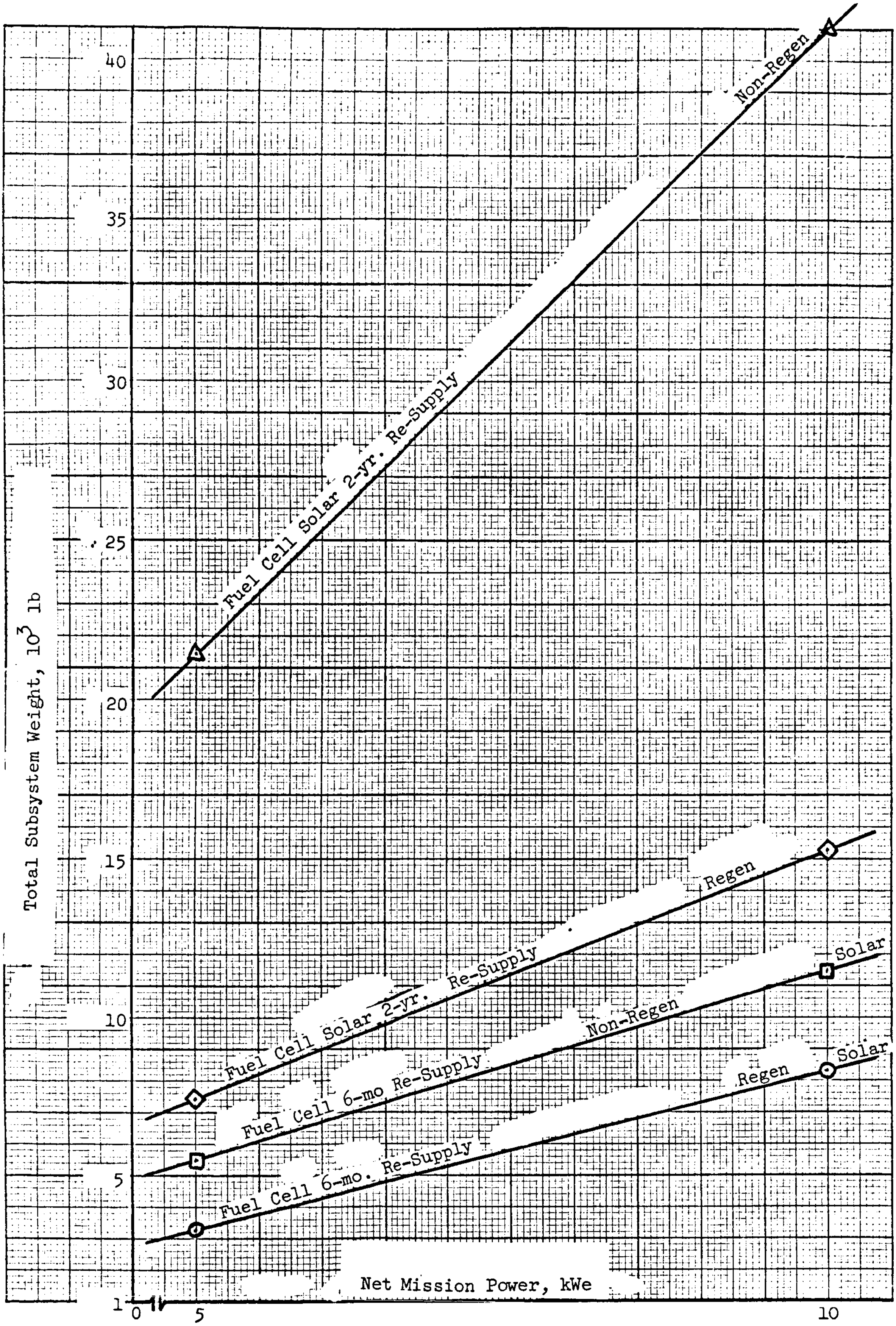


Figure 1.32 - Low Earth orbit fuel cell complementary power subsystems.

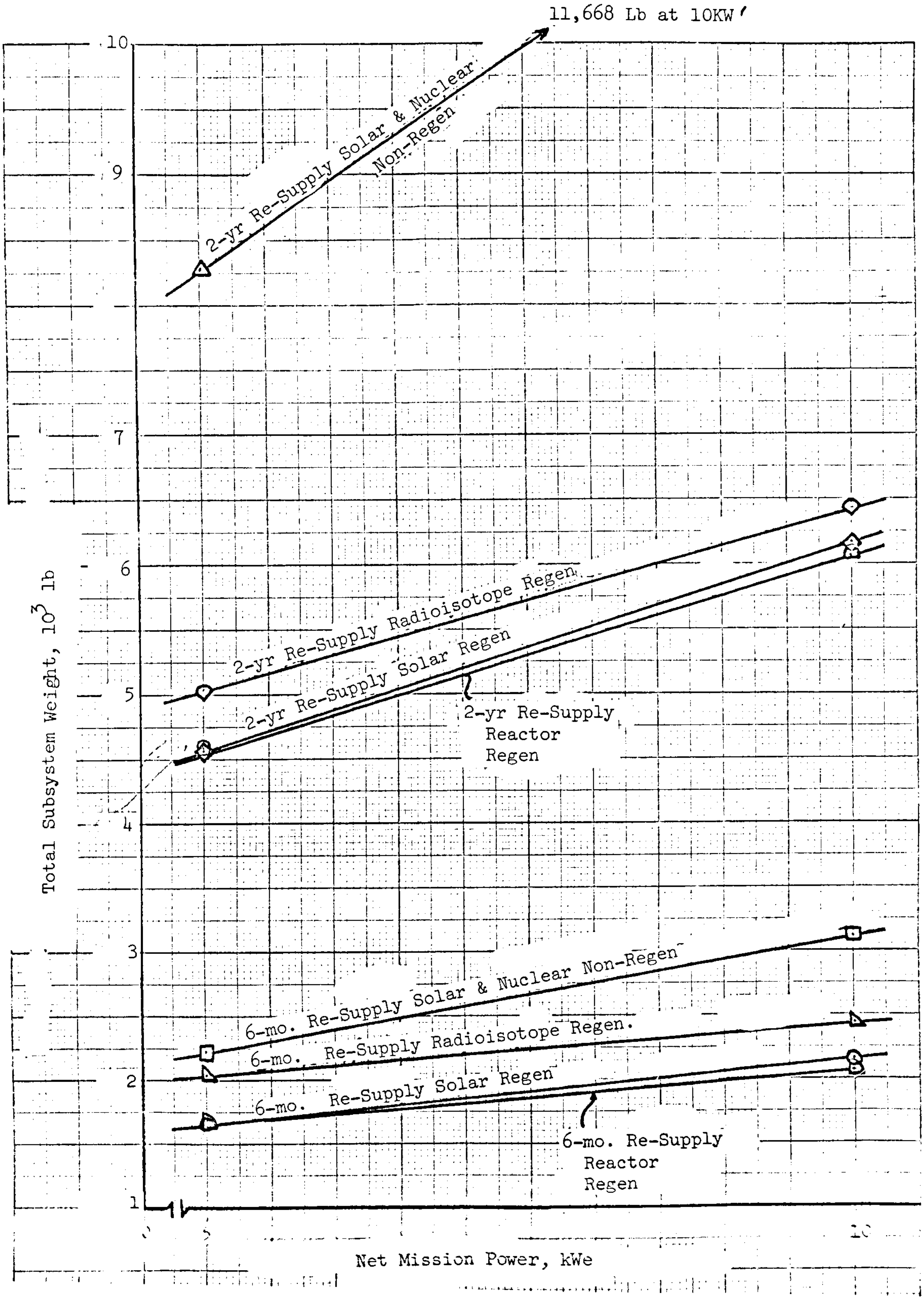


Figure 1.33 - Low Earth orbit fuel cell supplementary power subsystems.

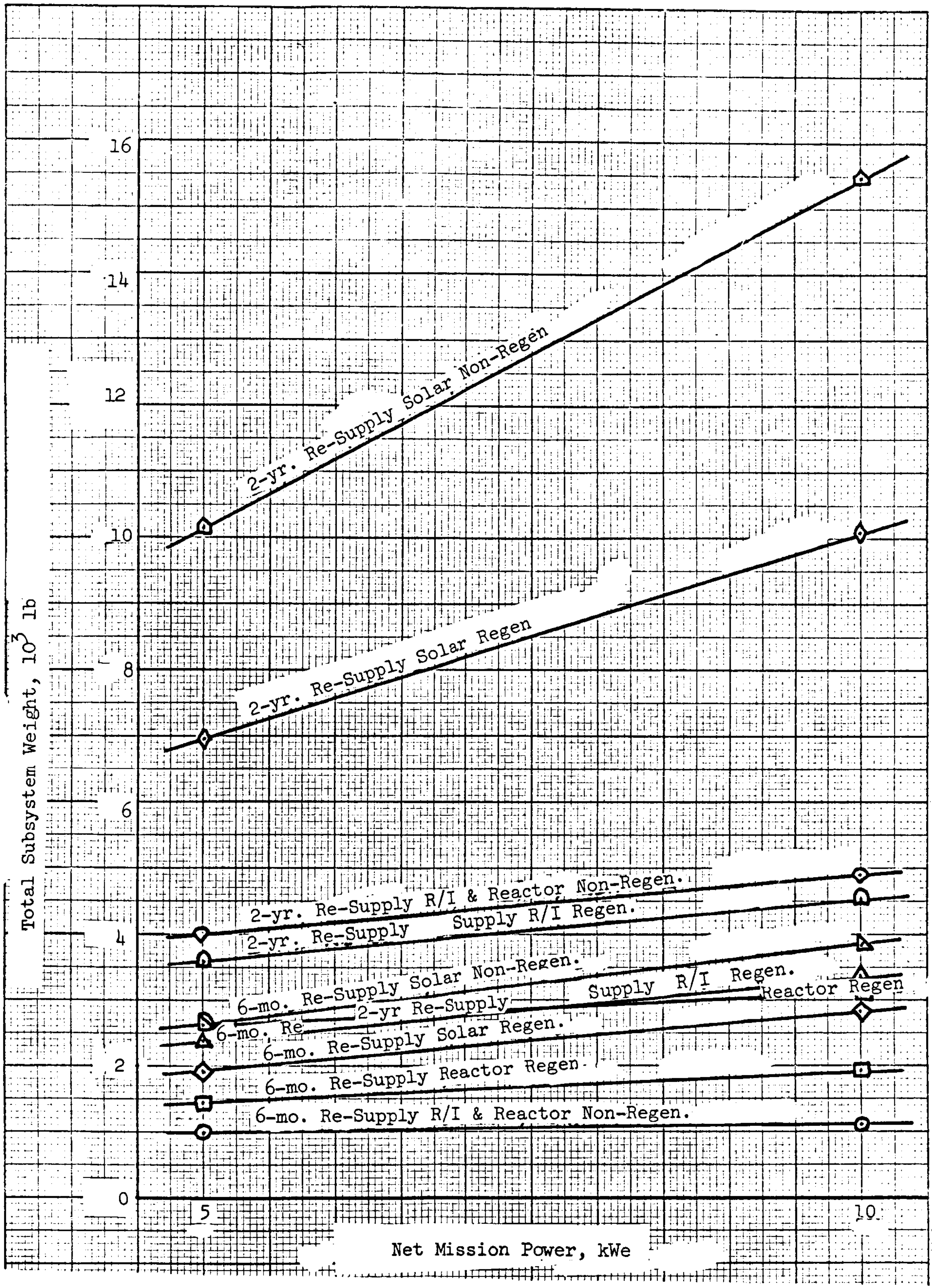


Figure 1.34 - Synchronous Earth orbit fuel cell supplementary power subsystem.

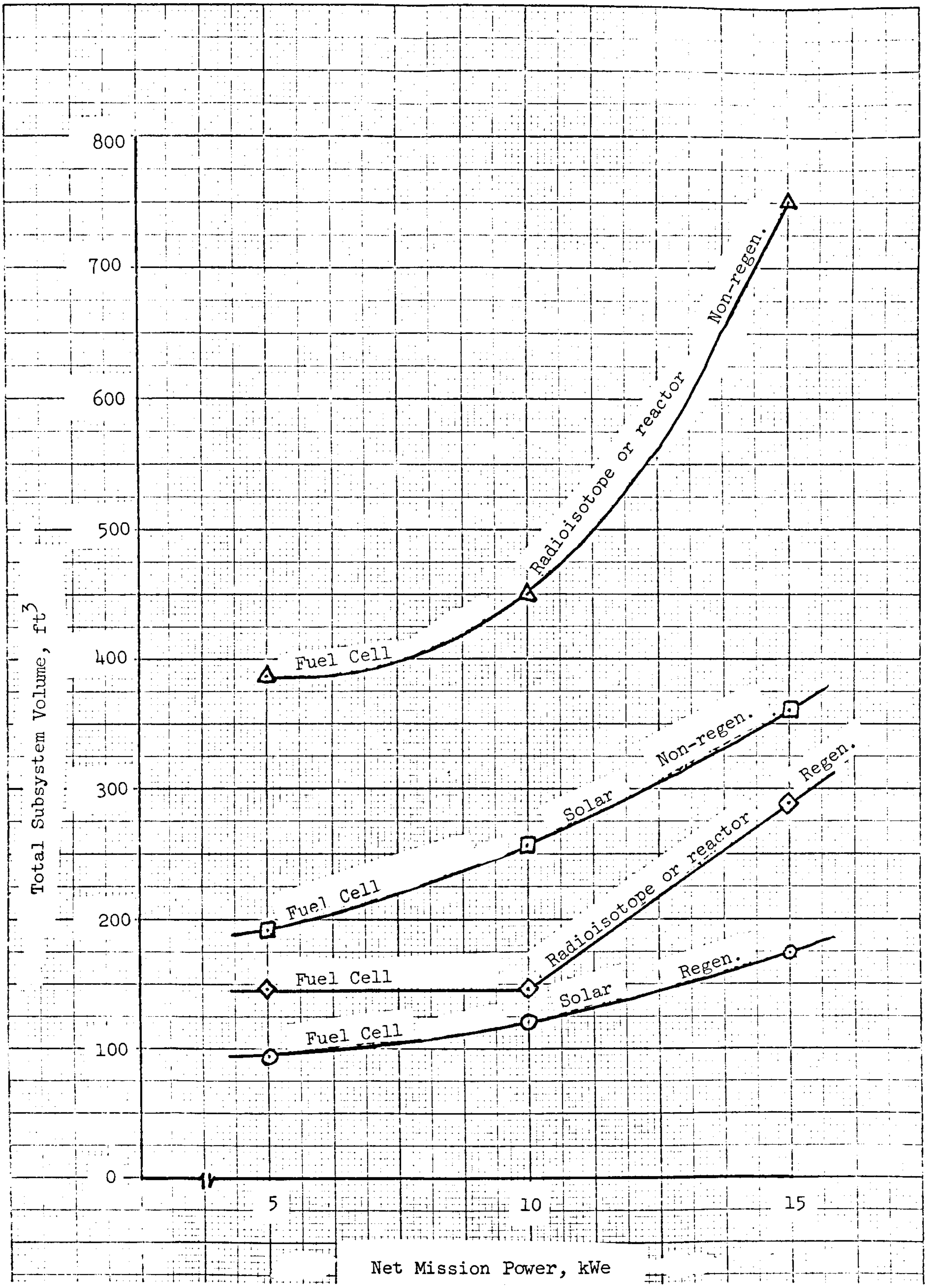


Figure 1.35 - Mars flyby fuel cell supplementary power subsystems volume.

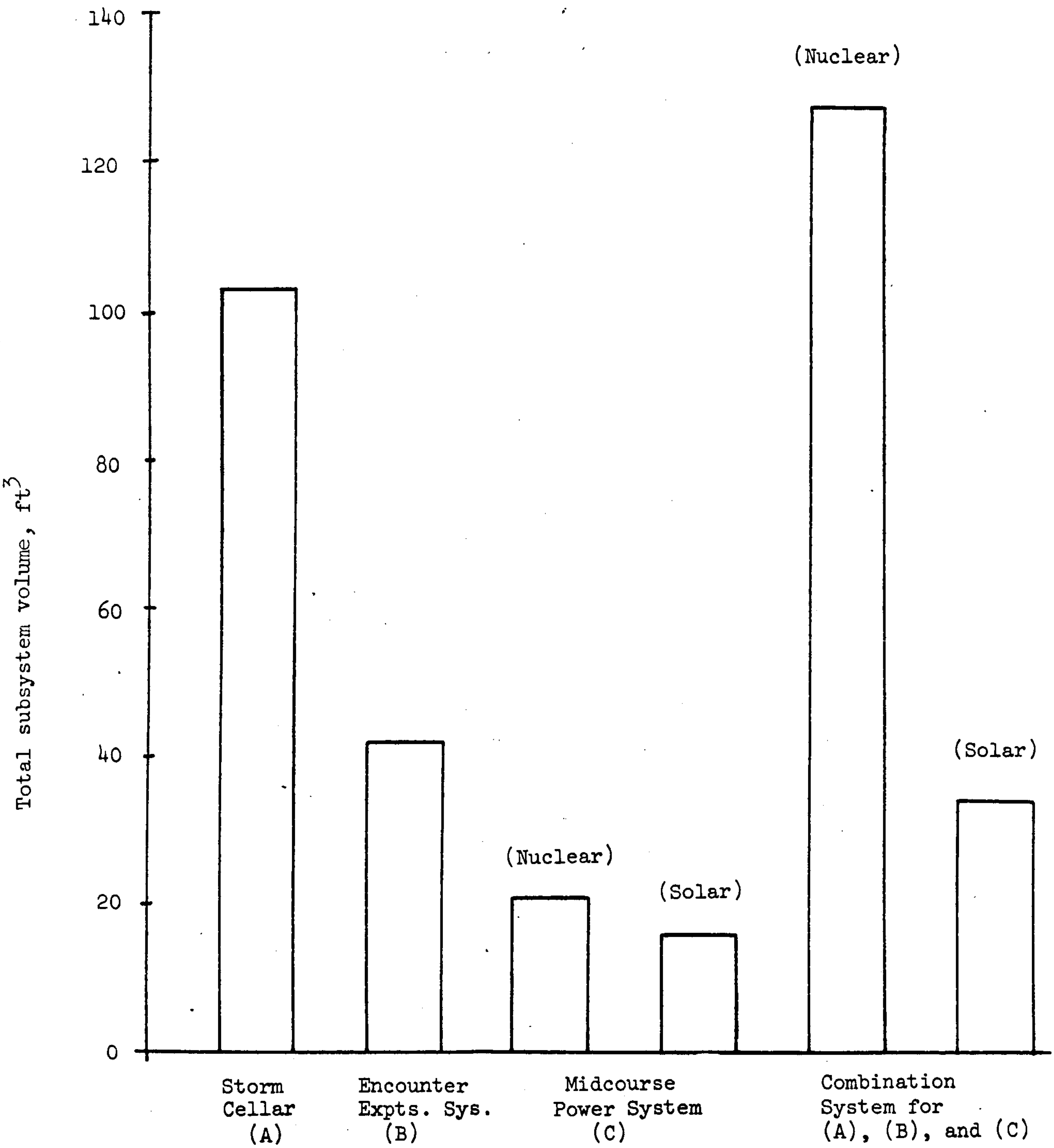


Figure 1.36 - Mars flyby fuel cell power subsystems volume.

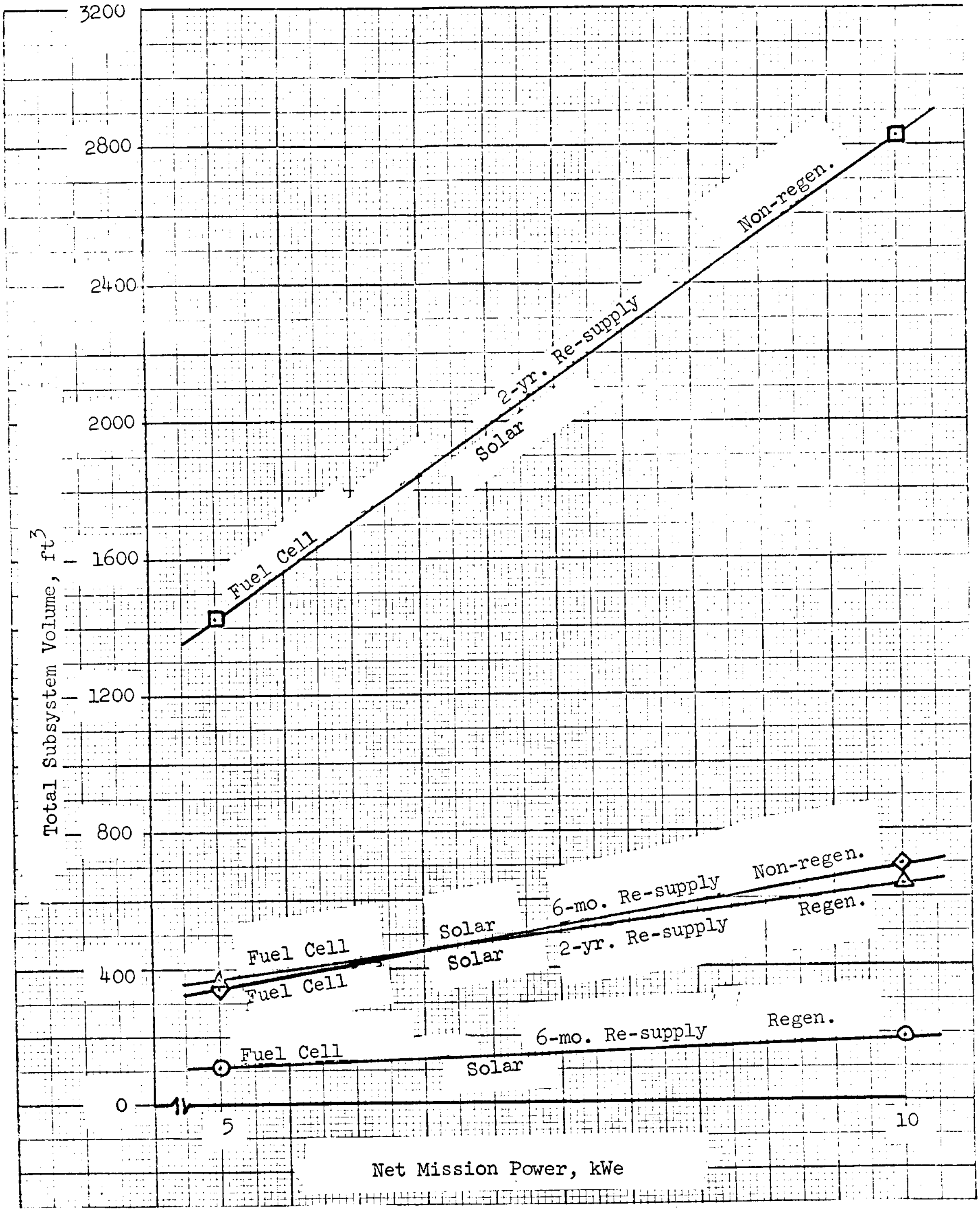


Figure 1.37 - Low Earth orbit fuel cell complementary power subsystems volume.

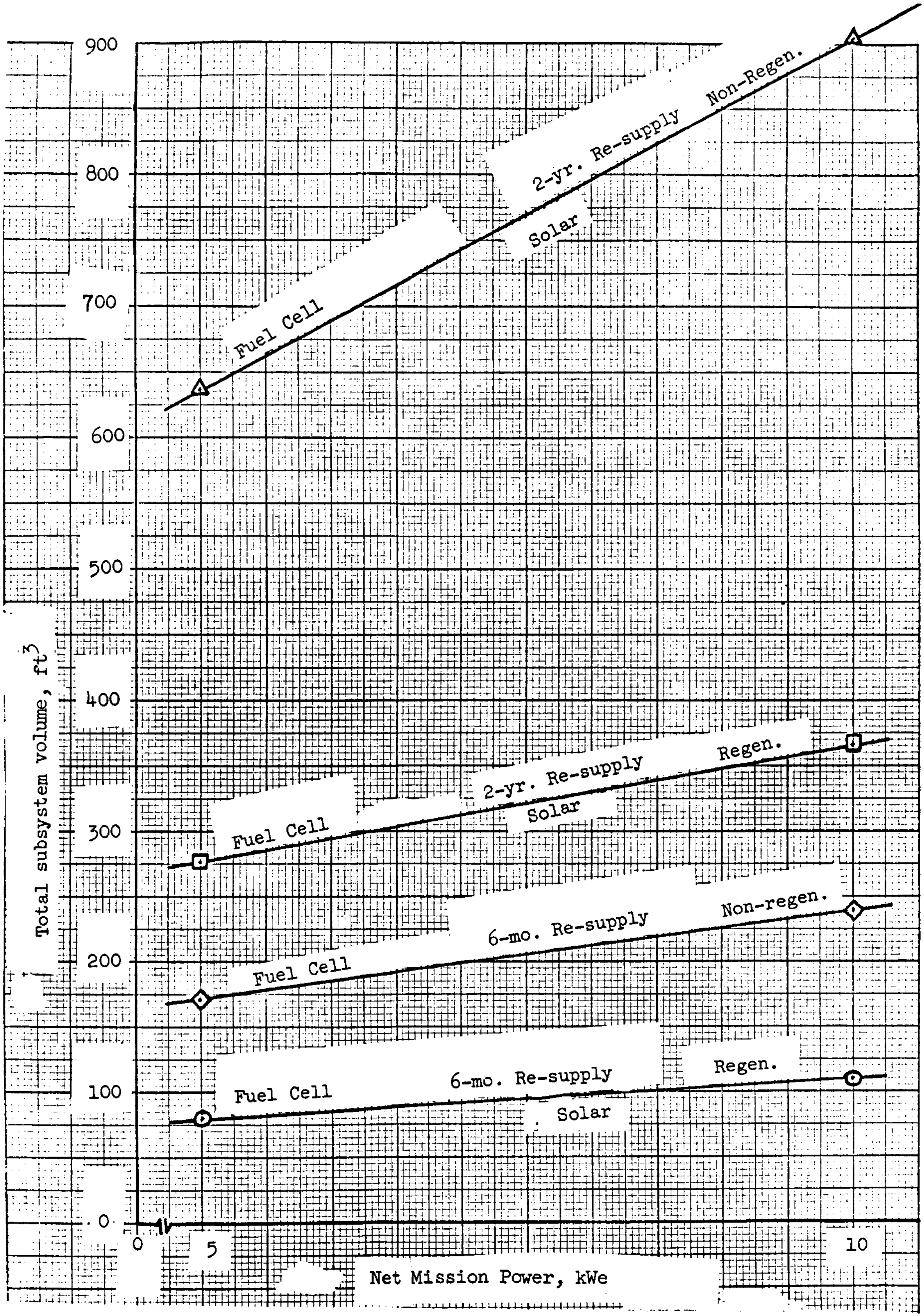


Figure 1.38 - Low Earth ordit fuel cell supplementary power subsystems volume.

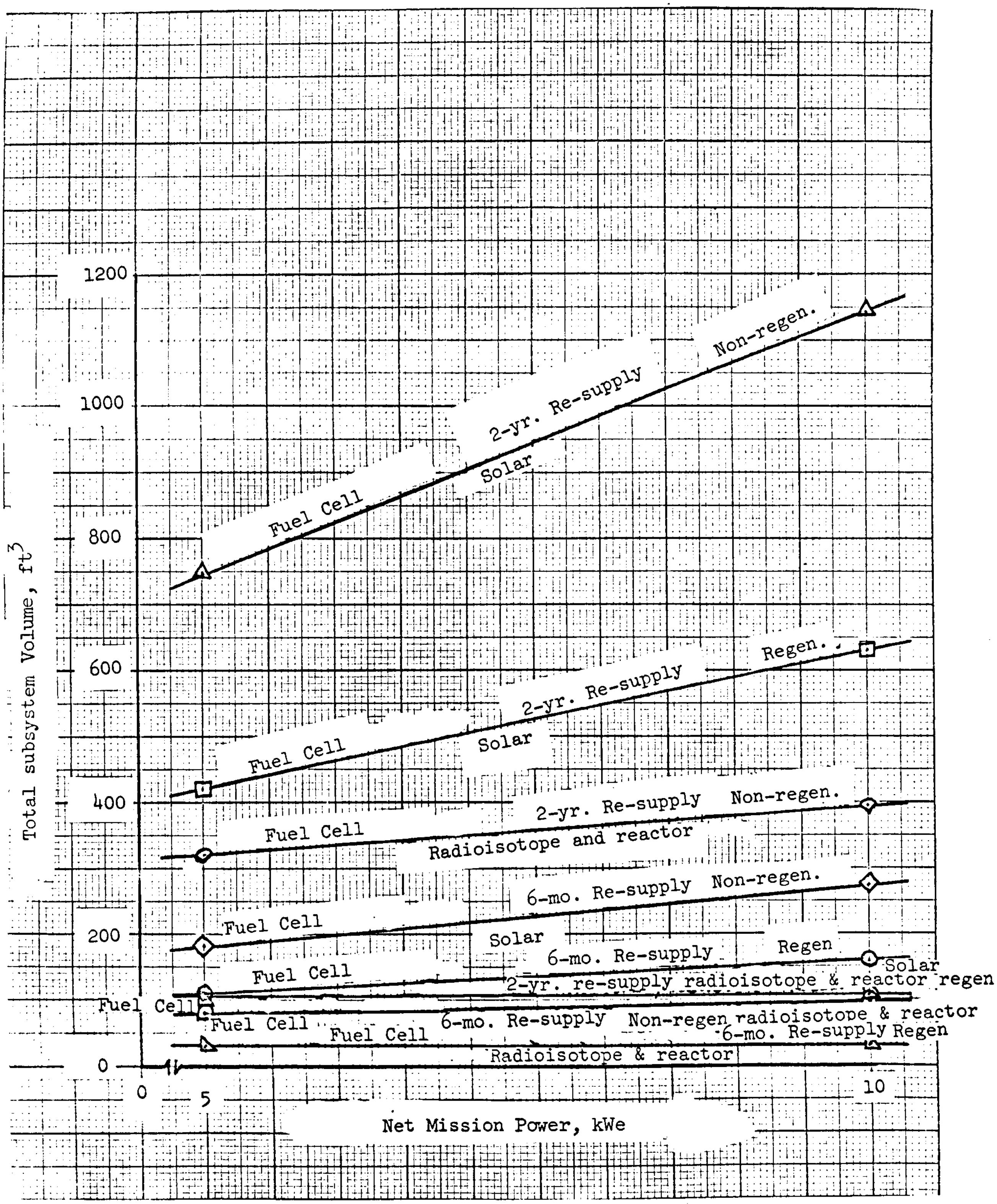


Figure 1.39 - Synchronous Earth orbit fuel cell supplementary power subsystems volume.

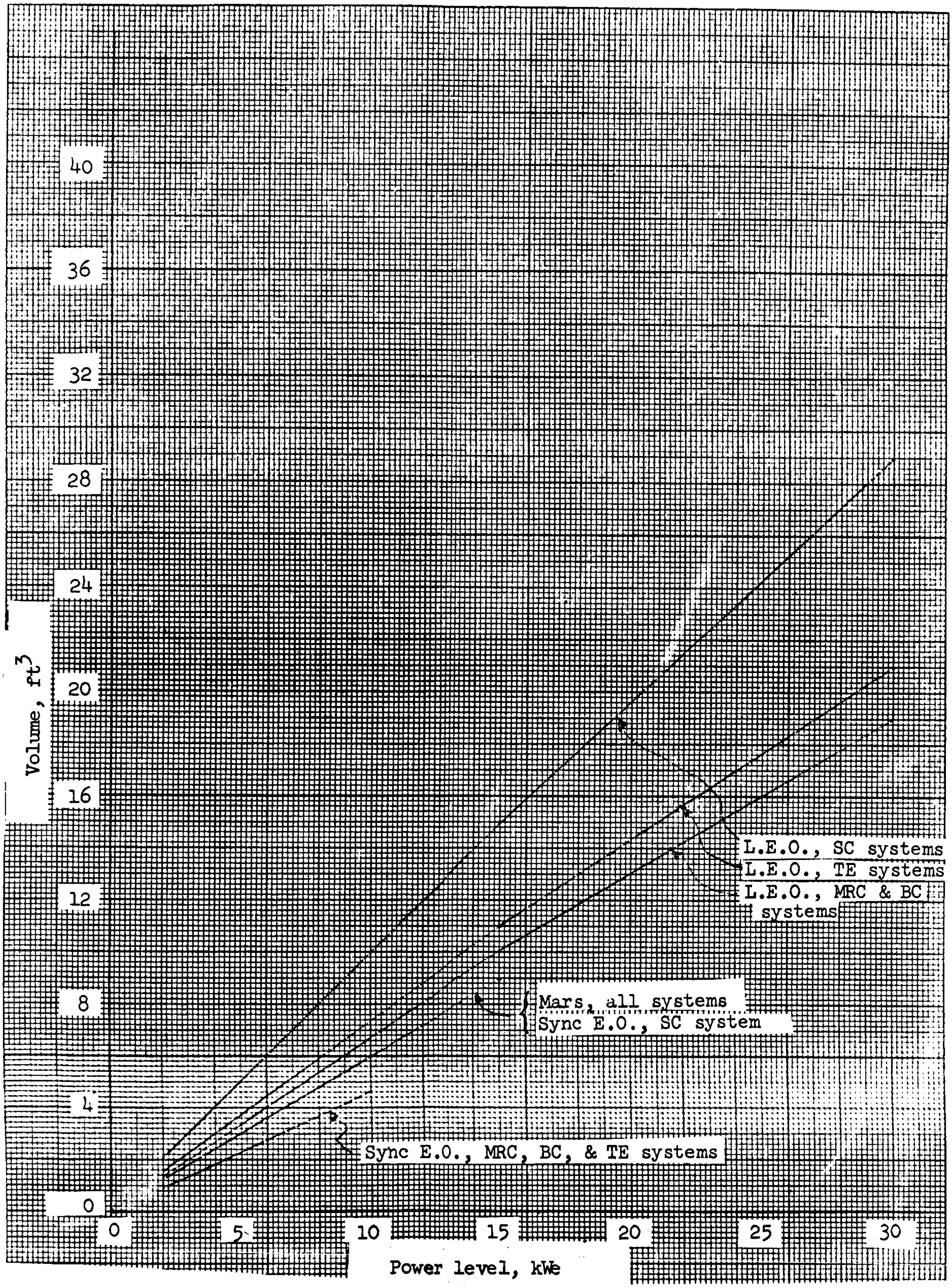


Figure 1.40 - Power conditioning volumes versus power level.

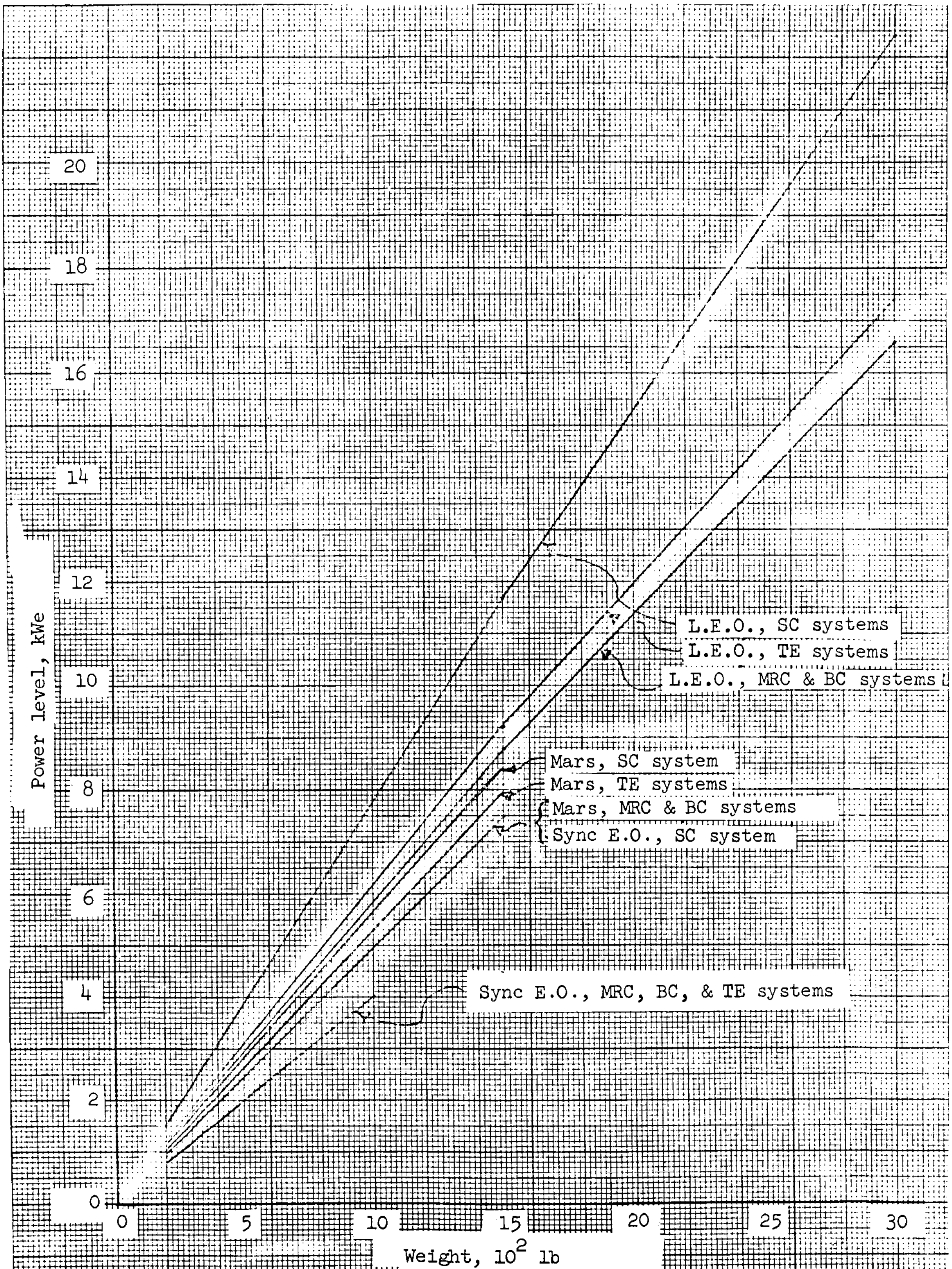


Figure 1.41 - Power conditioning weights vs power level.

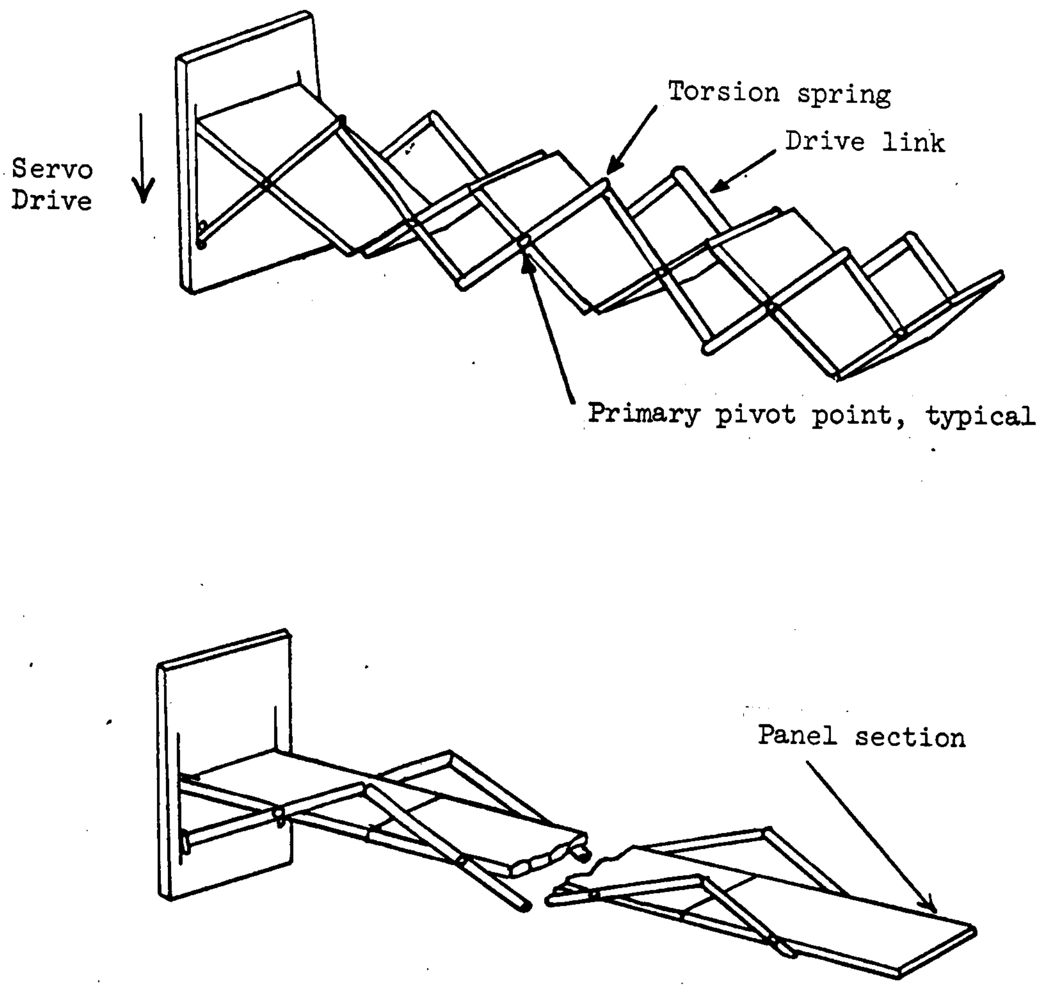


Figure 1.42 **. - Concept "A" - rigid panel deployment concept.**

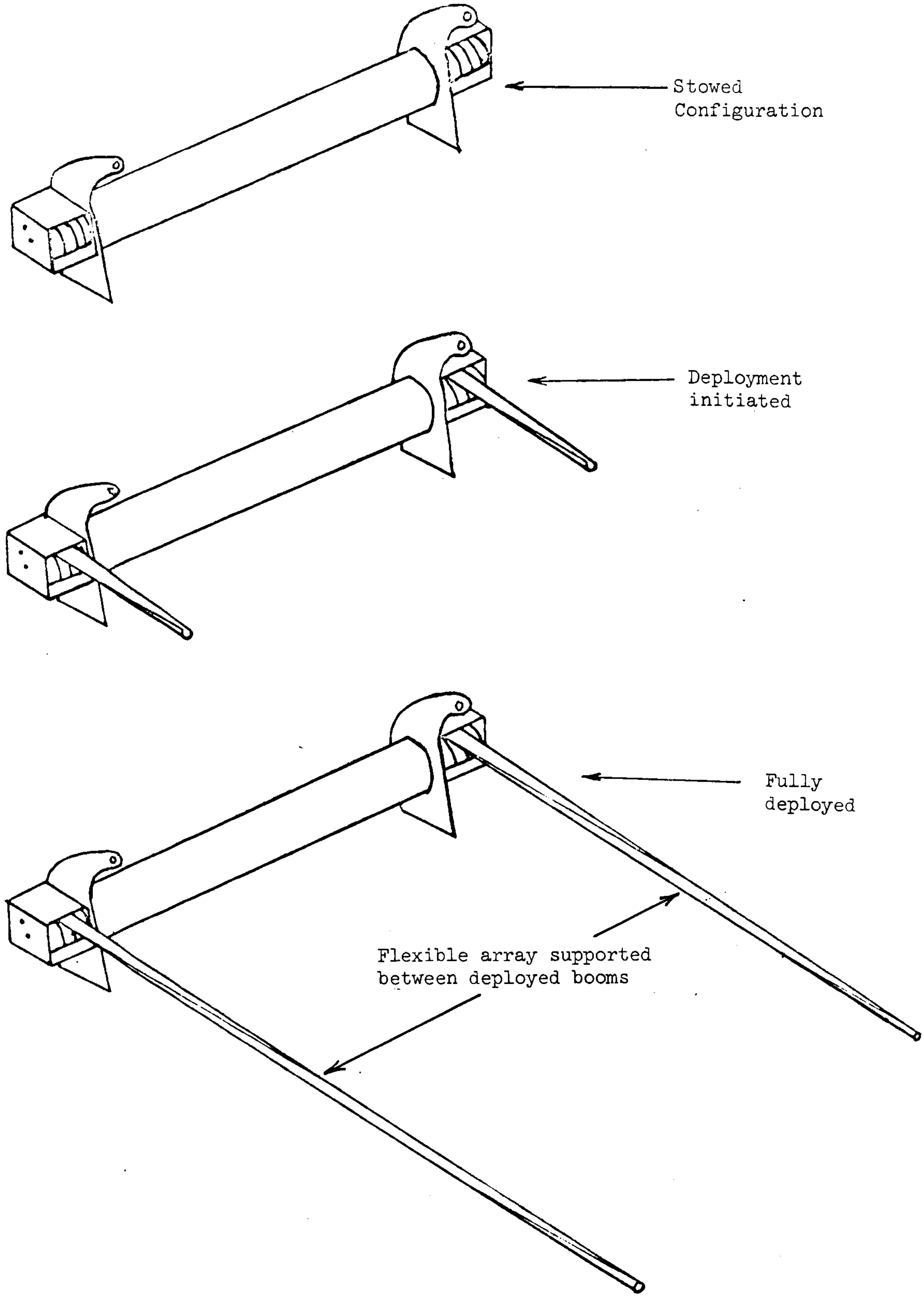


Figure 1.43 · Concept "B" - unfurlable array deployment concept.

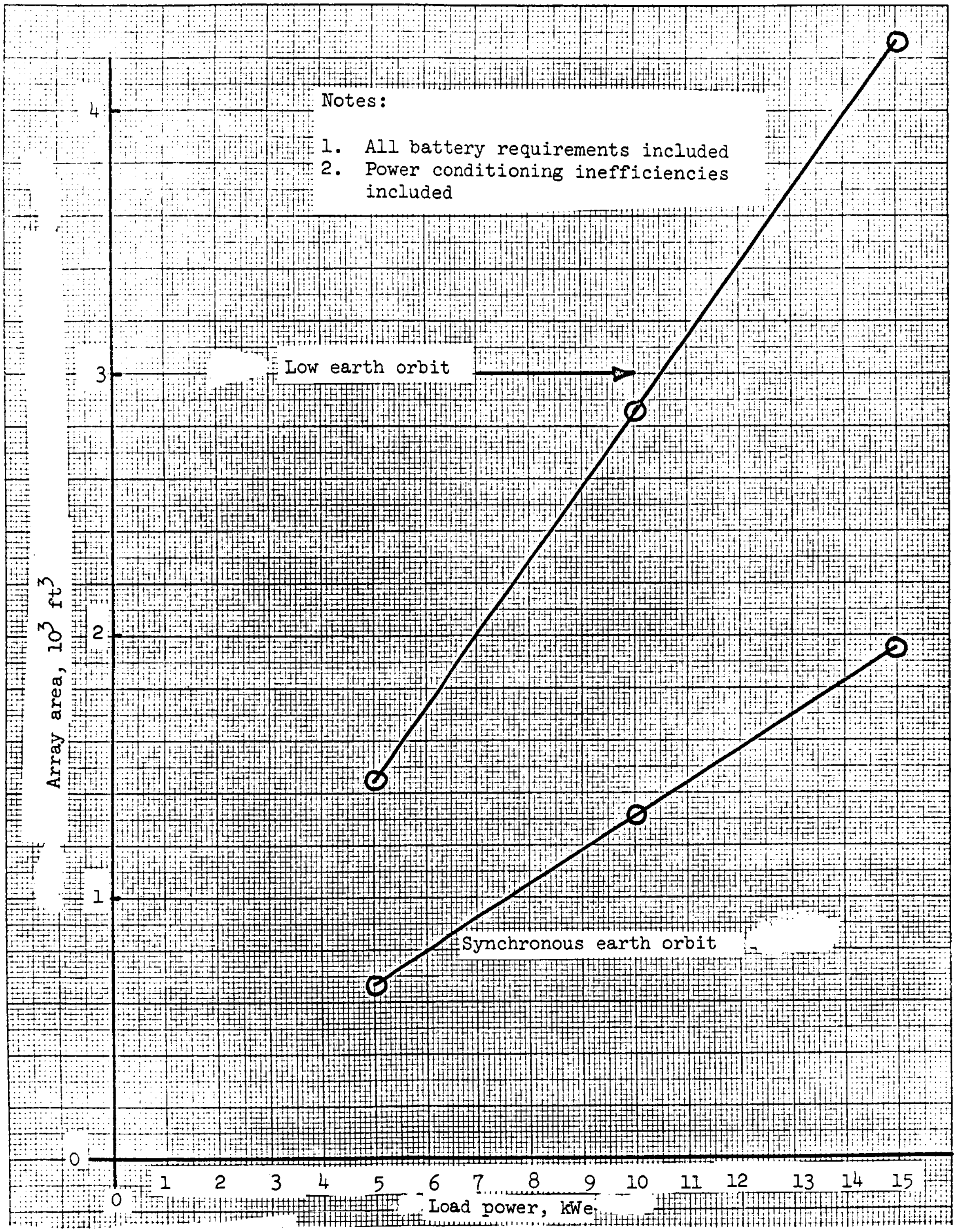


Figure 1.44 - Array area versus power for Earth orbit.

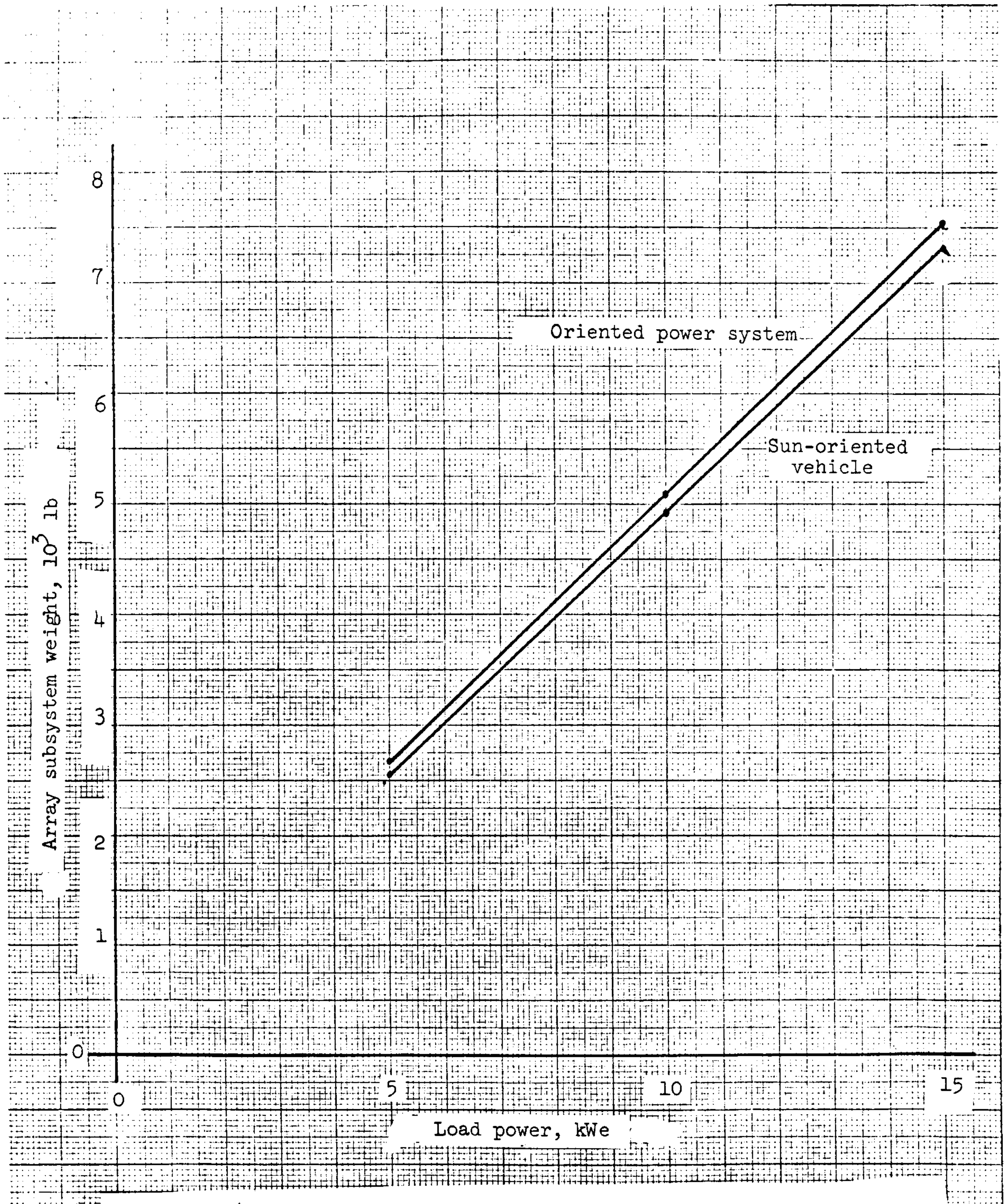


Figure 1.45 - Array subsystem weight versus load power, low Earth orbit missions.

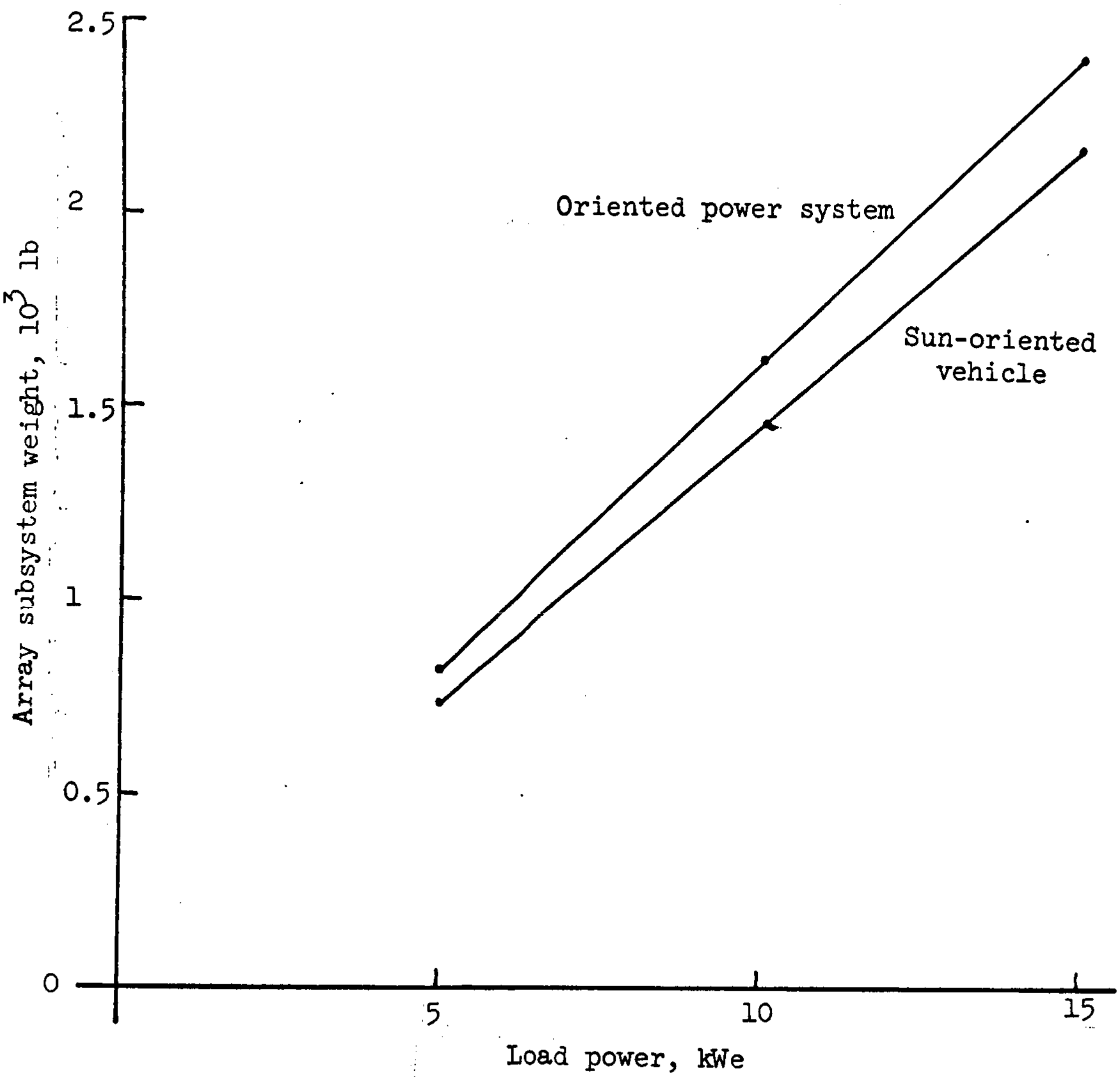


Figure 1.46 - Subsystem array weight versus load power, synchronous orbit missions.

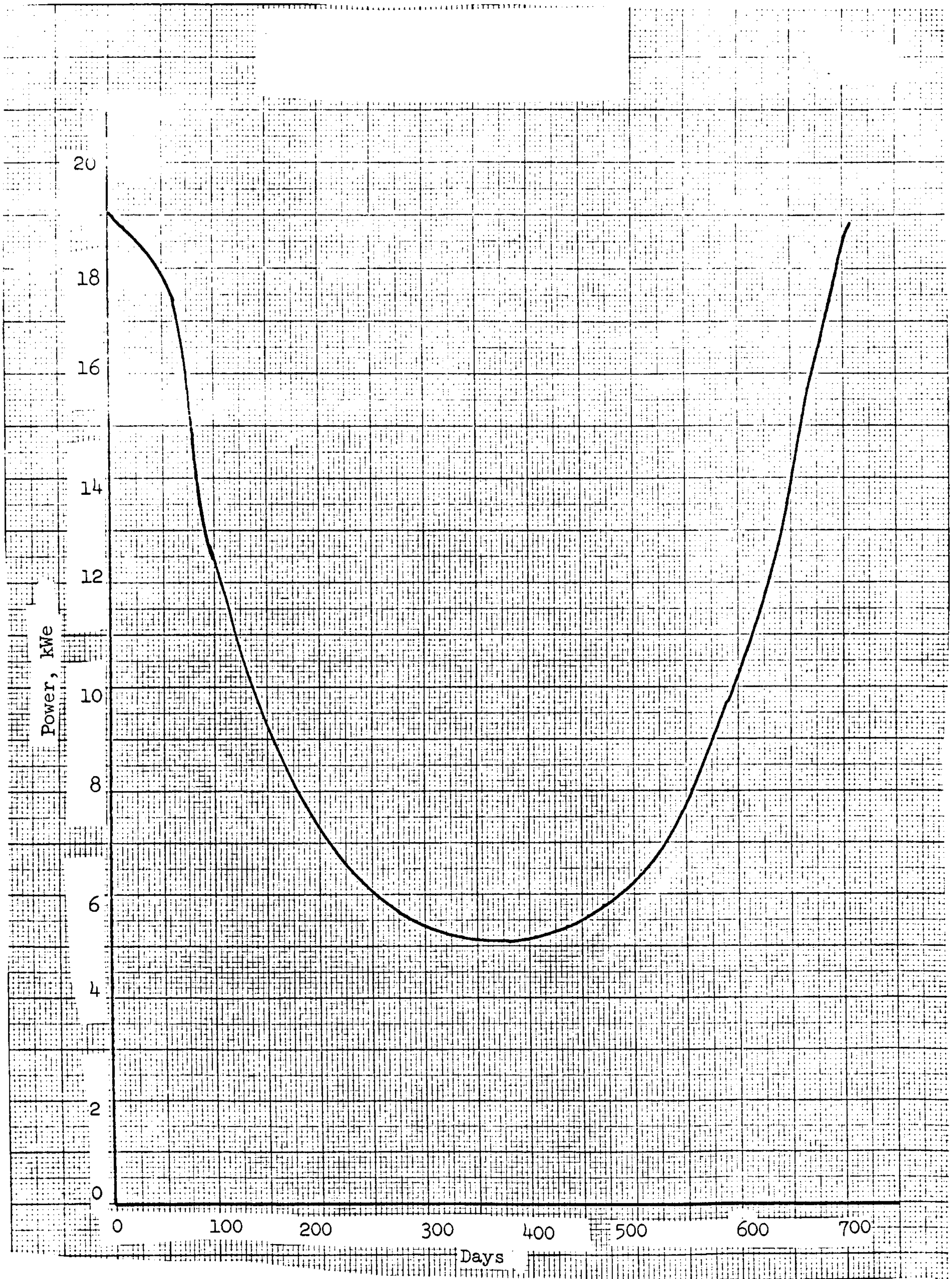


Figure 1.47 - Output of solar cell system designed for 5 kWe output at 2. 2 A. U. (Temperature effects included).

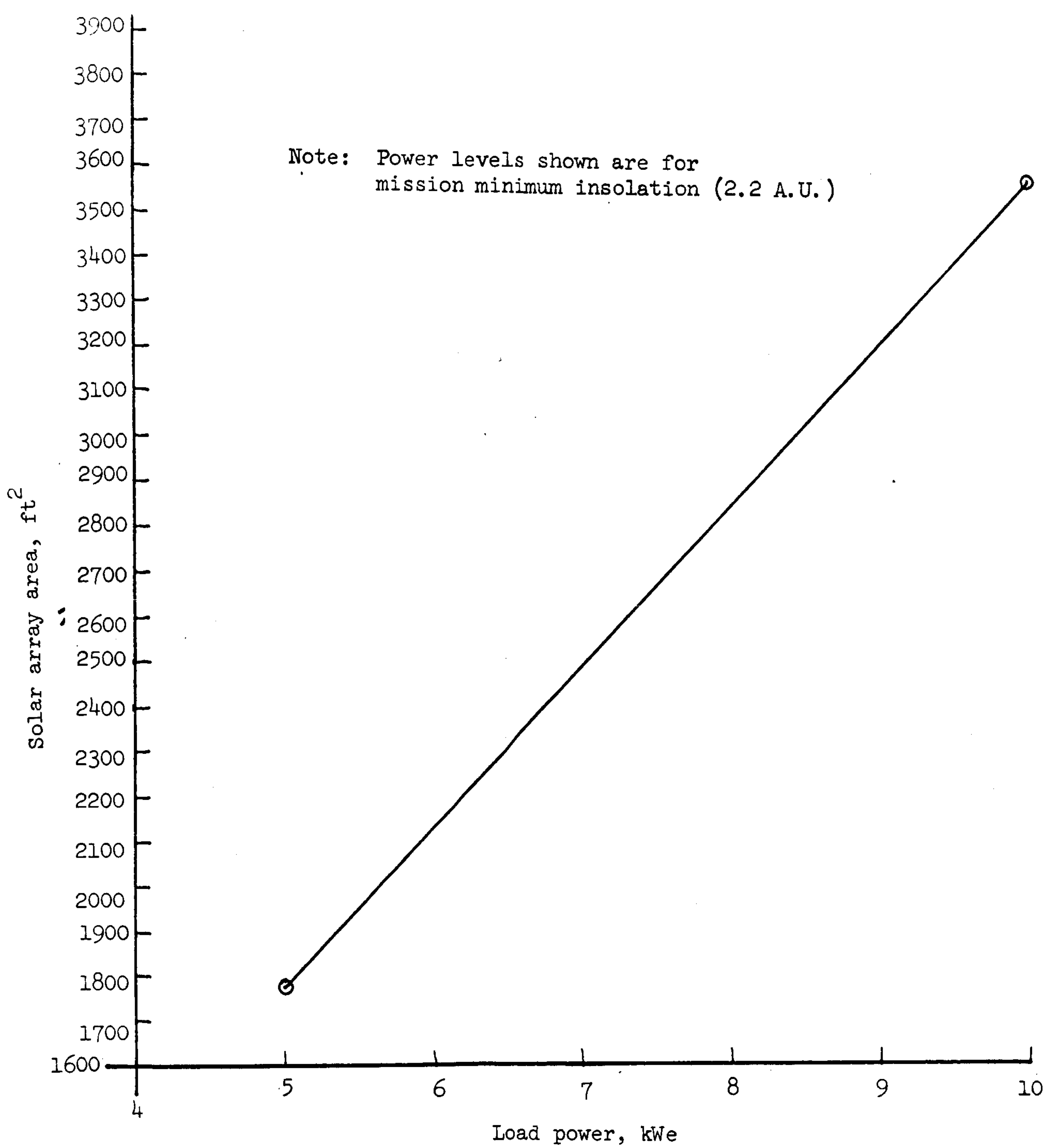


Figure 1.48 - Array area required for a Mars flyby.

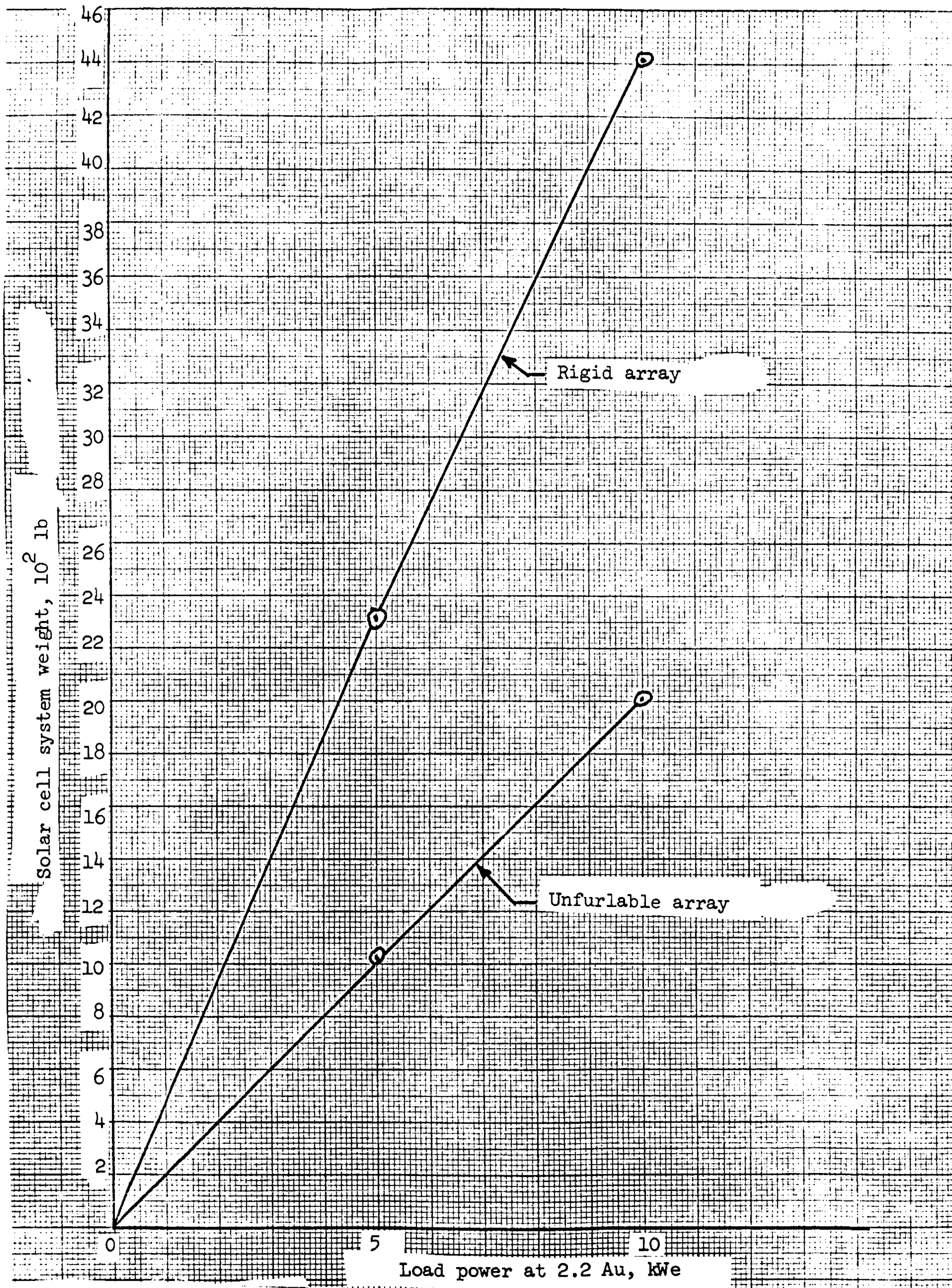


Figure 1.49. - Mars flyby solar cell system weight versus power.

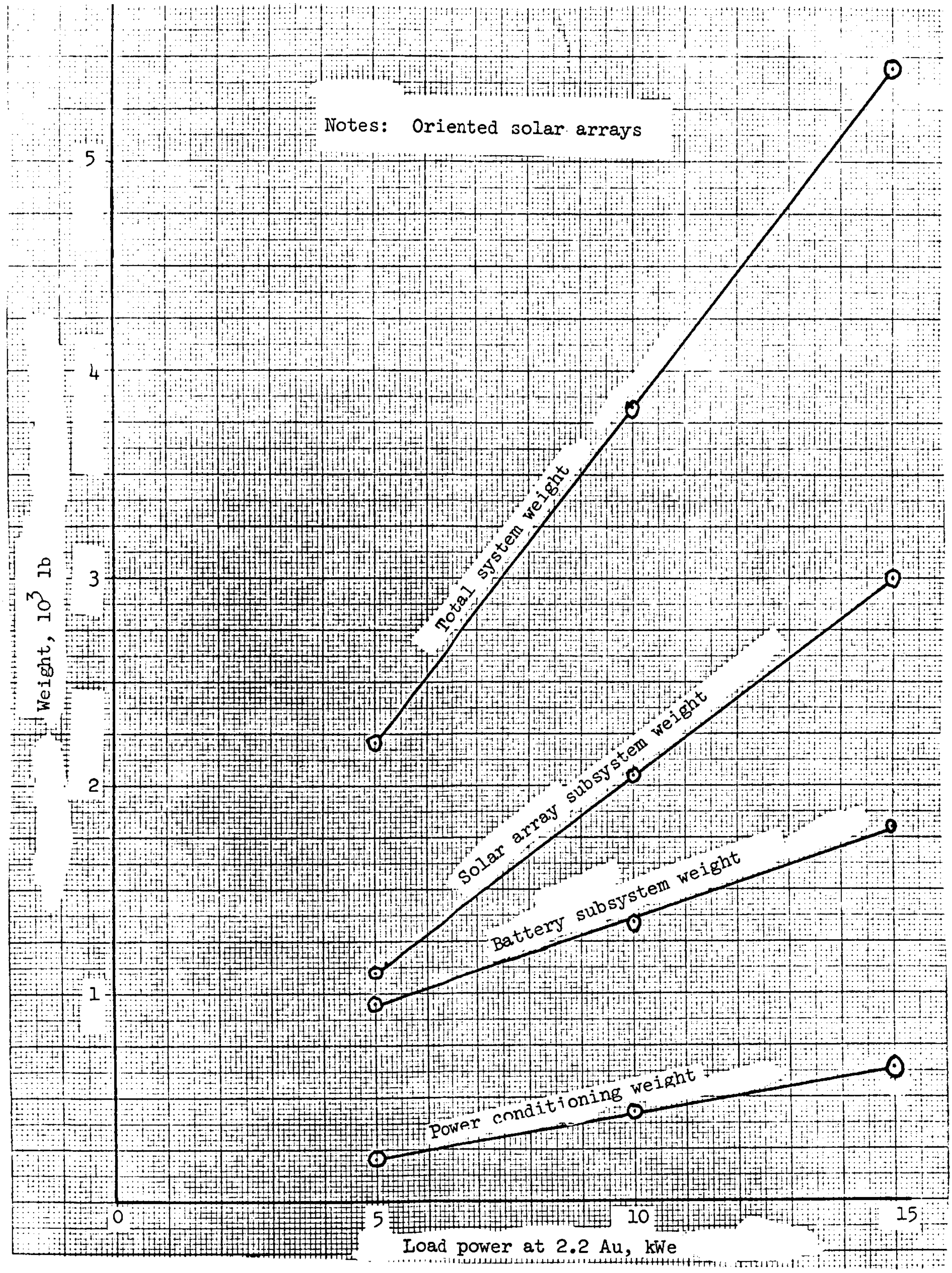


Figure 1.50 - Solar cell/battery system weight versus load power, Mars flyby.

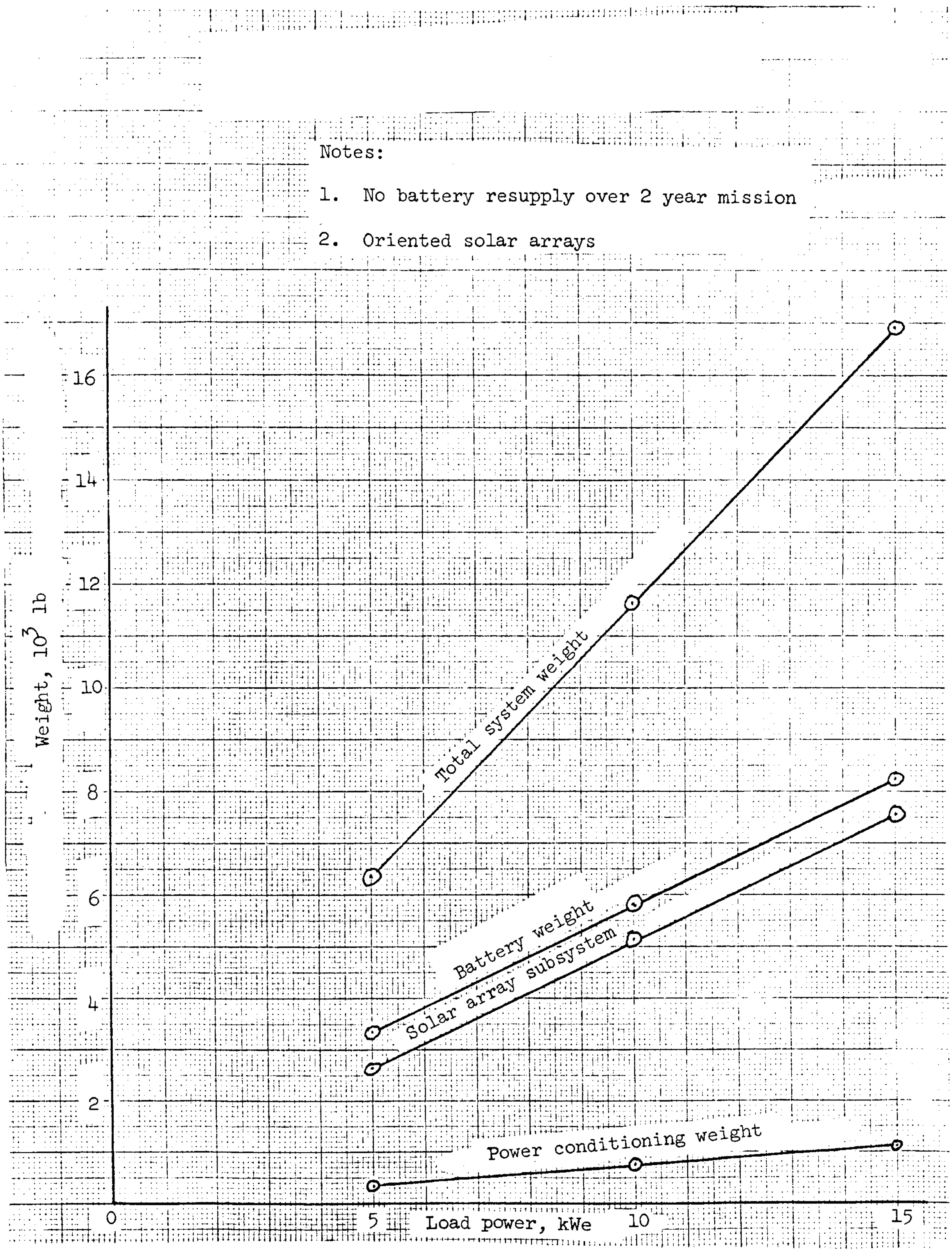


Figure 1.51 - Solar cell/battery system weight versus load power, LEO missions.

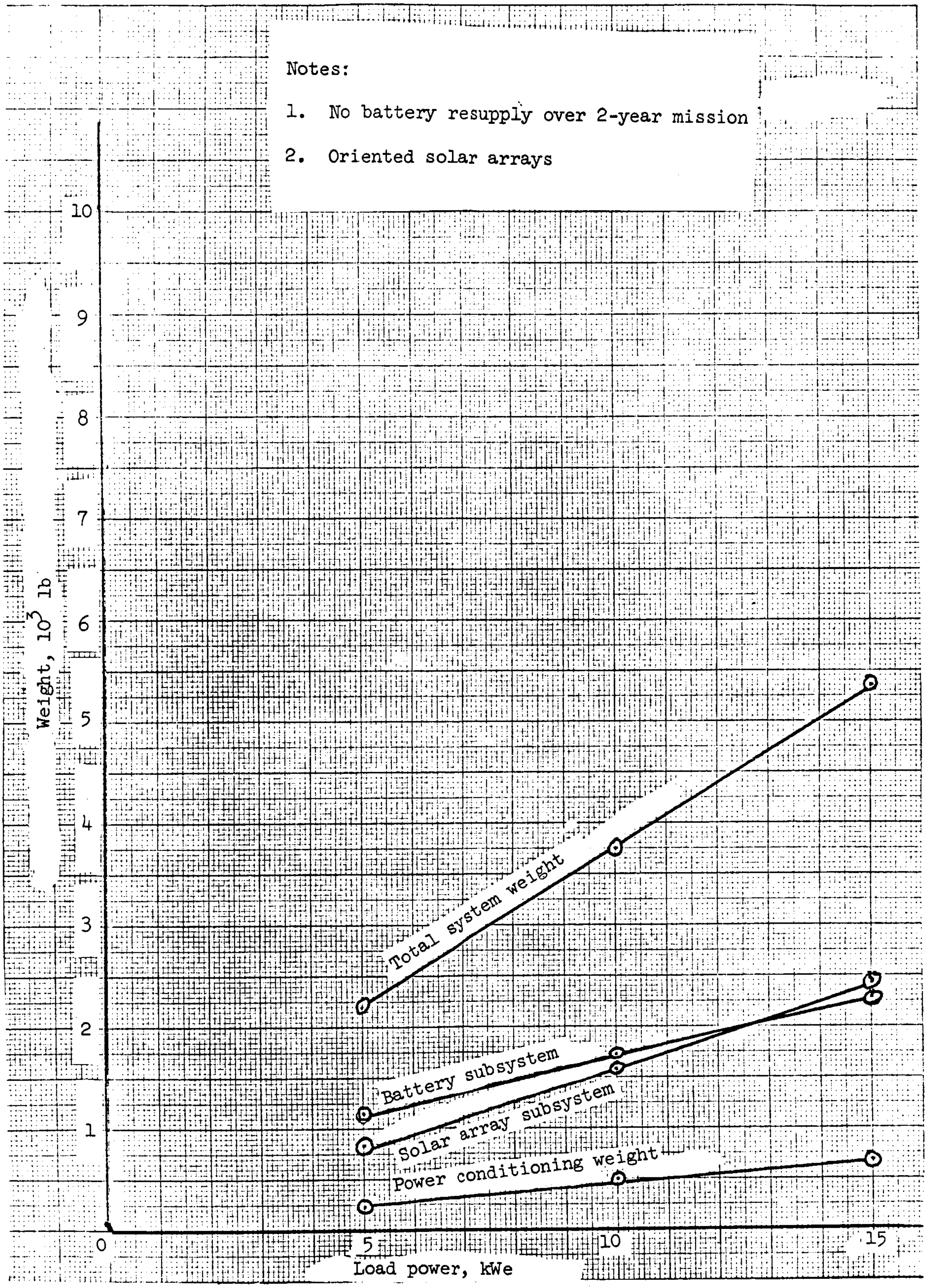


Figure 1.52 - Solar cell/battery system weight vs. load power, sync missions.

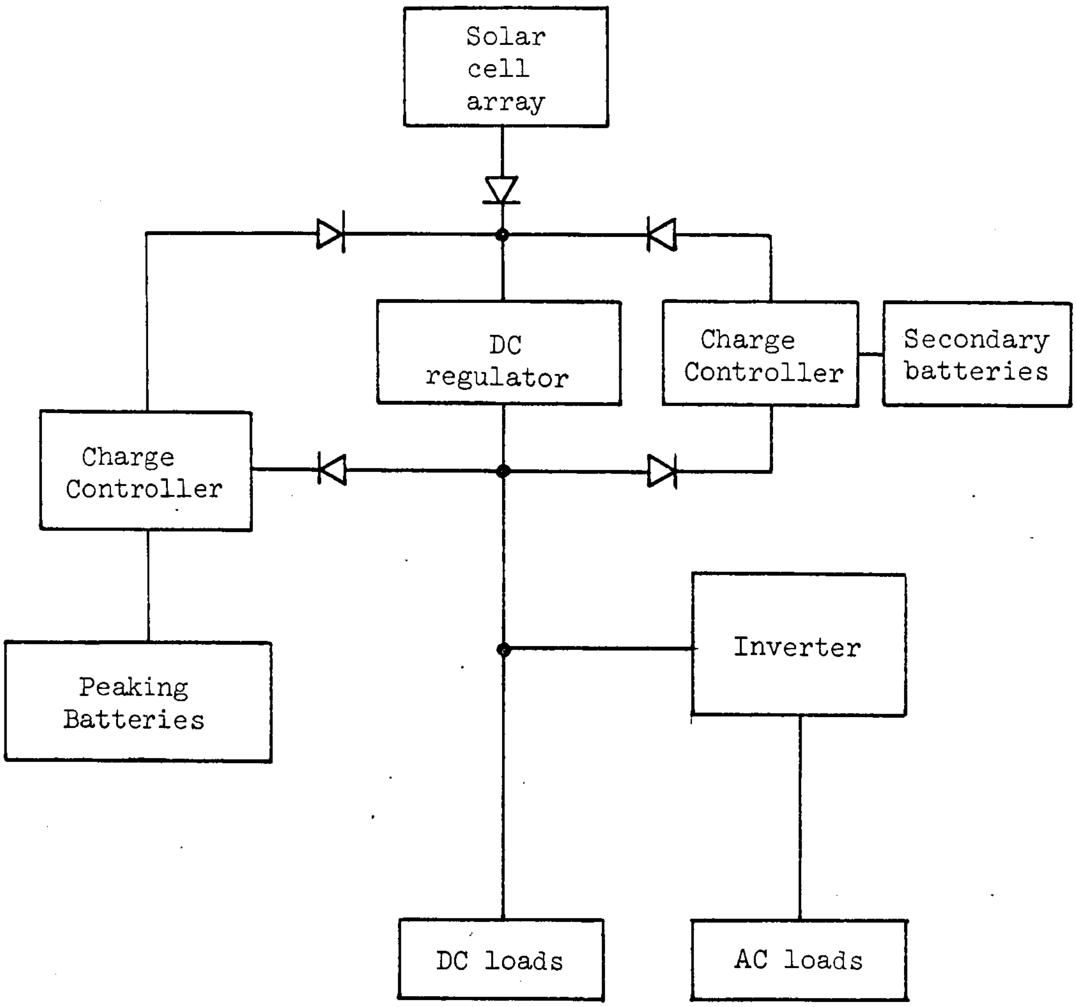


Figure 1.53 - LEO solar cell/battery schematic.

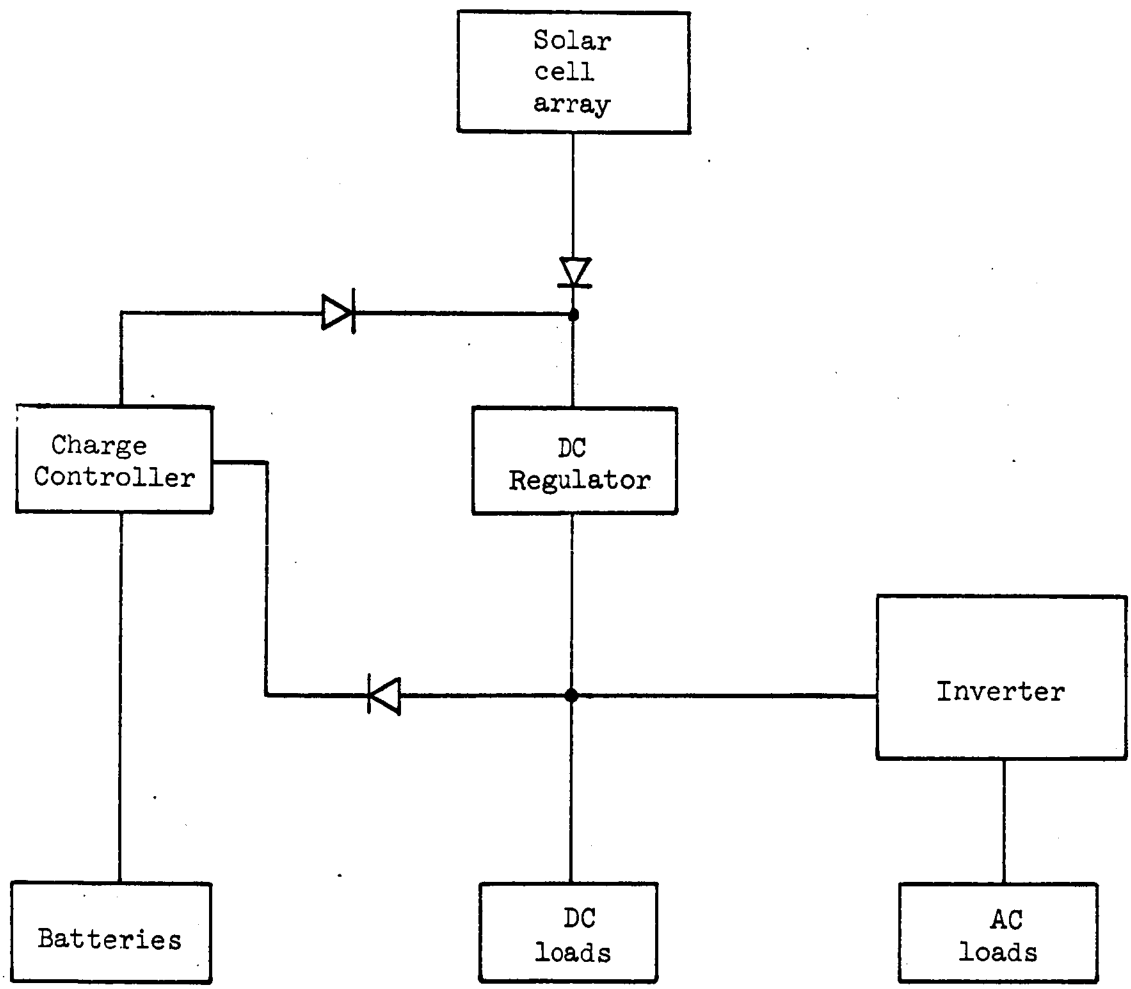
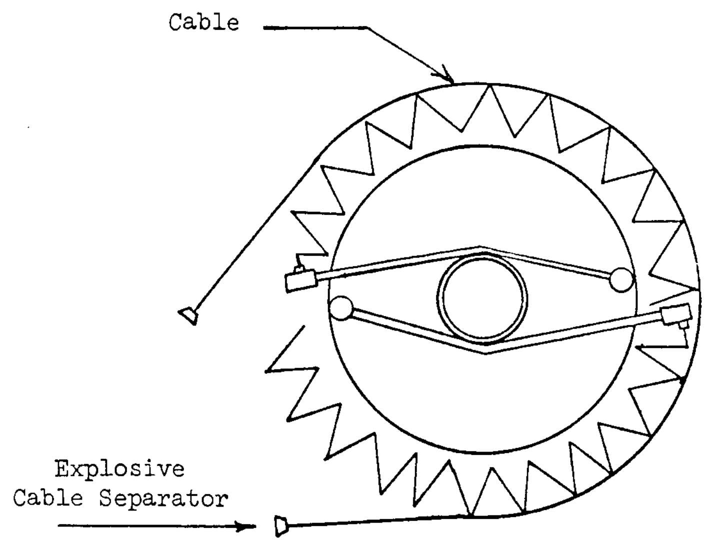
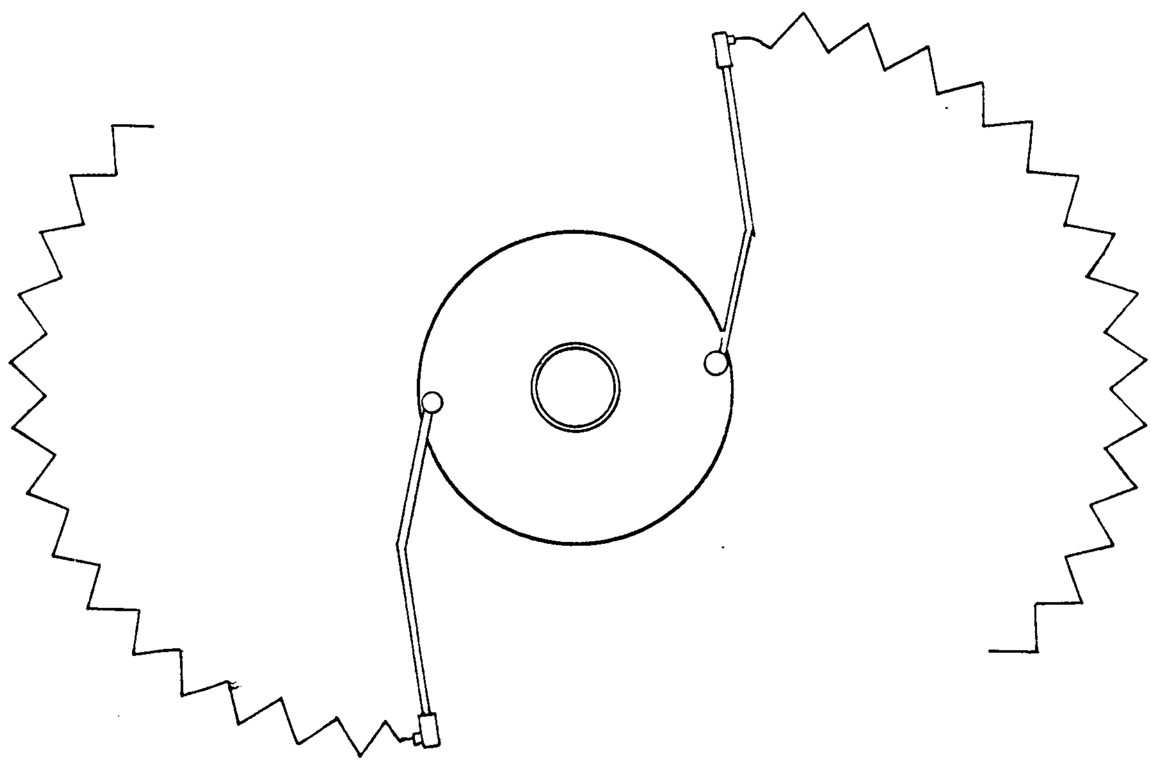


Figure 1.54 - **Mars and sync solar cell/battery schematic.**

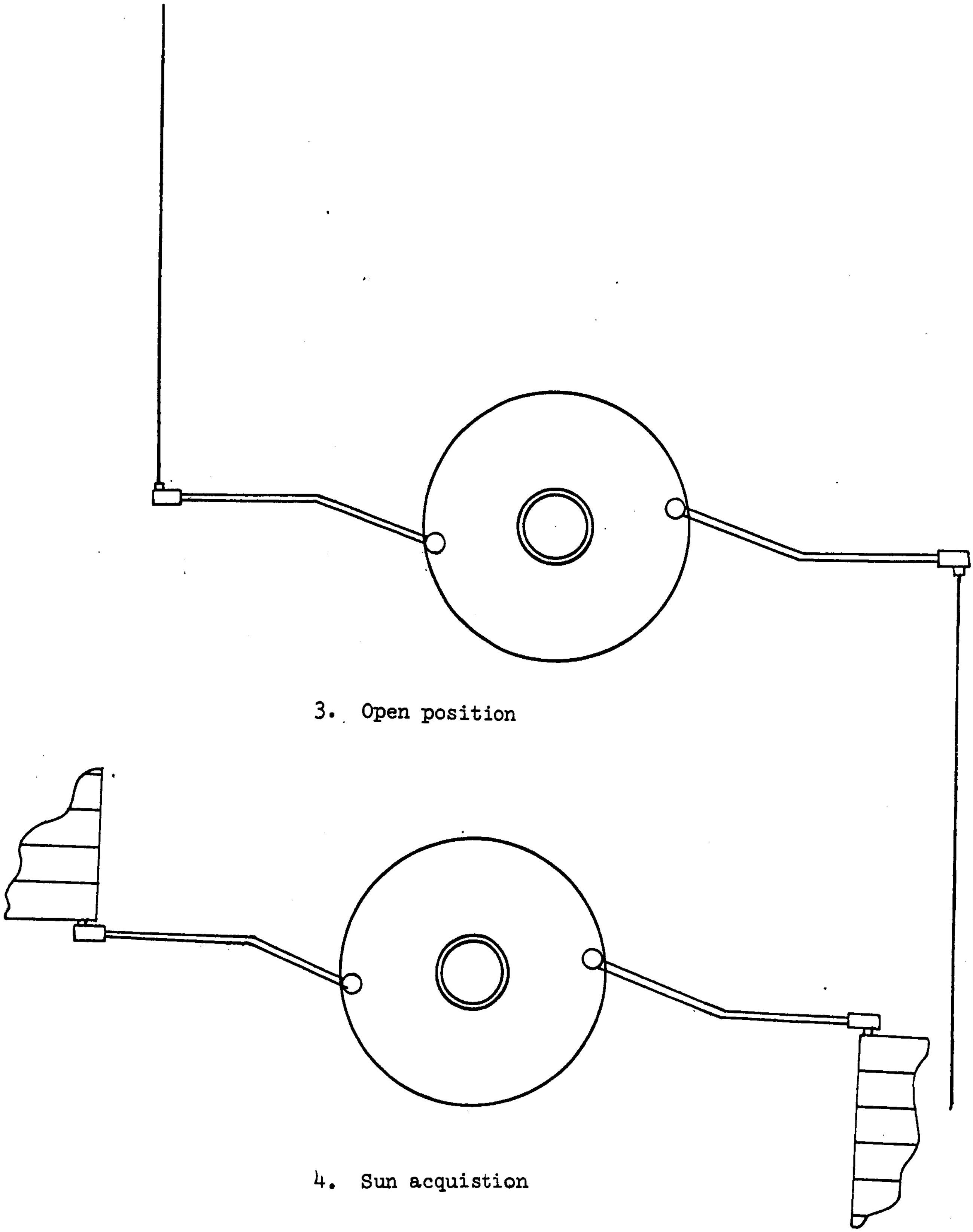


1. Closed Position
(Jettison cable)



2. Partially open position

Figure 1.55 - Solar array deployment sequences 1 and 2.



3. Open position

4. Sun acquistion

Figure 1.56 - Solar array deployment sequences 3 and 4.

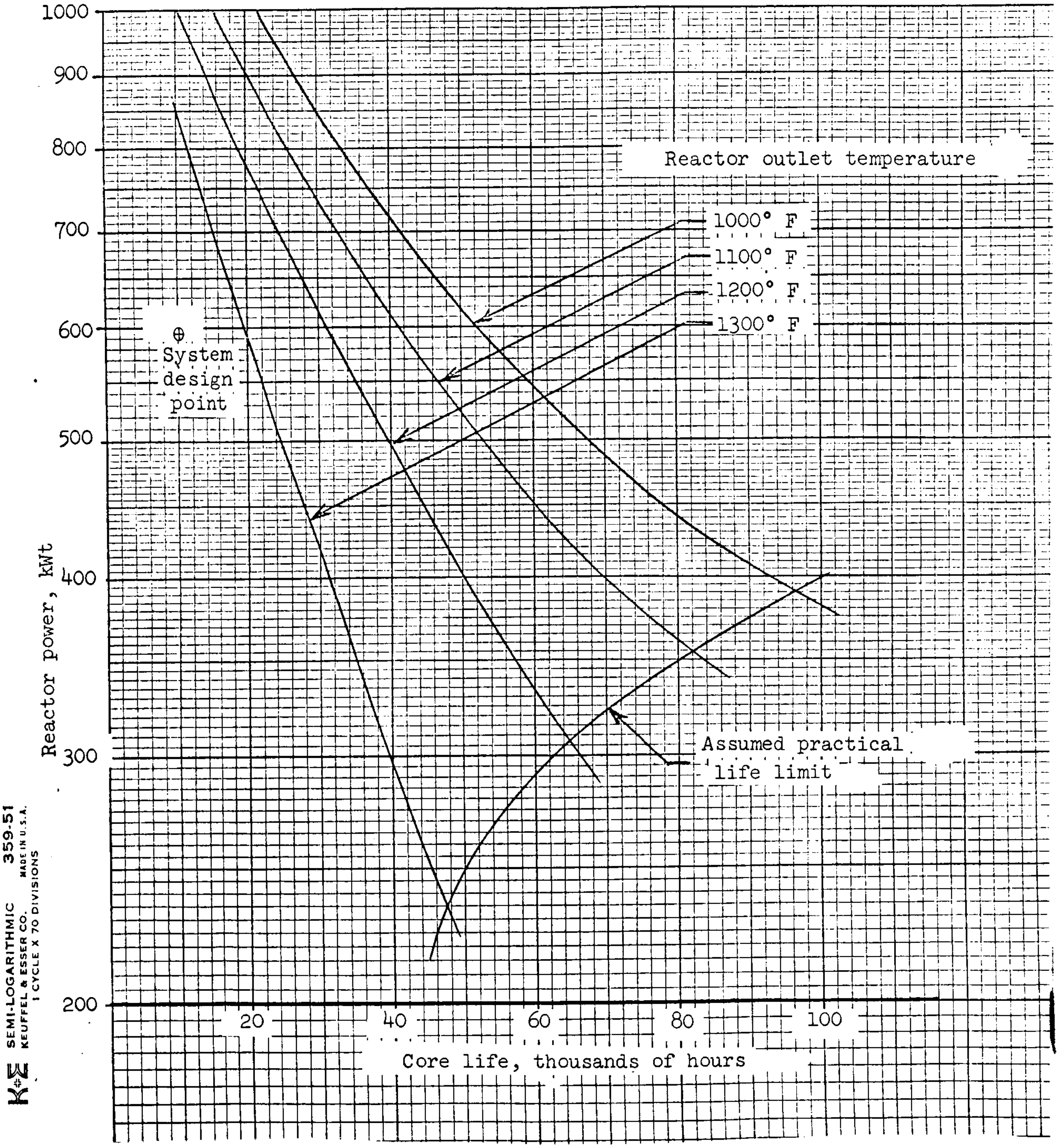


Figure 1.57 - SNAP-8 reactor operating capability.

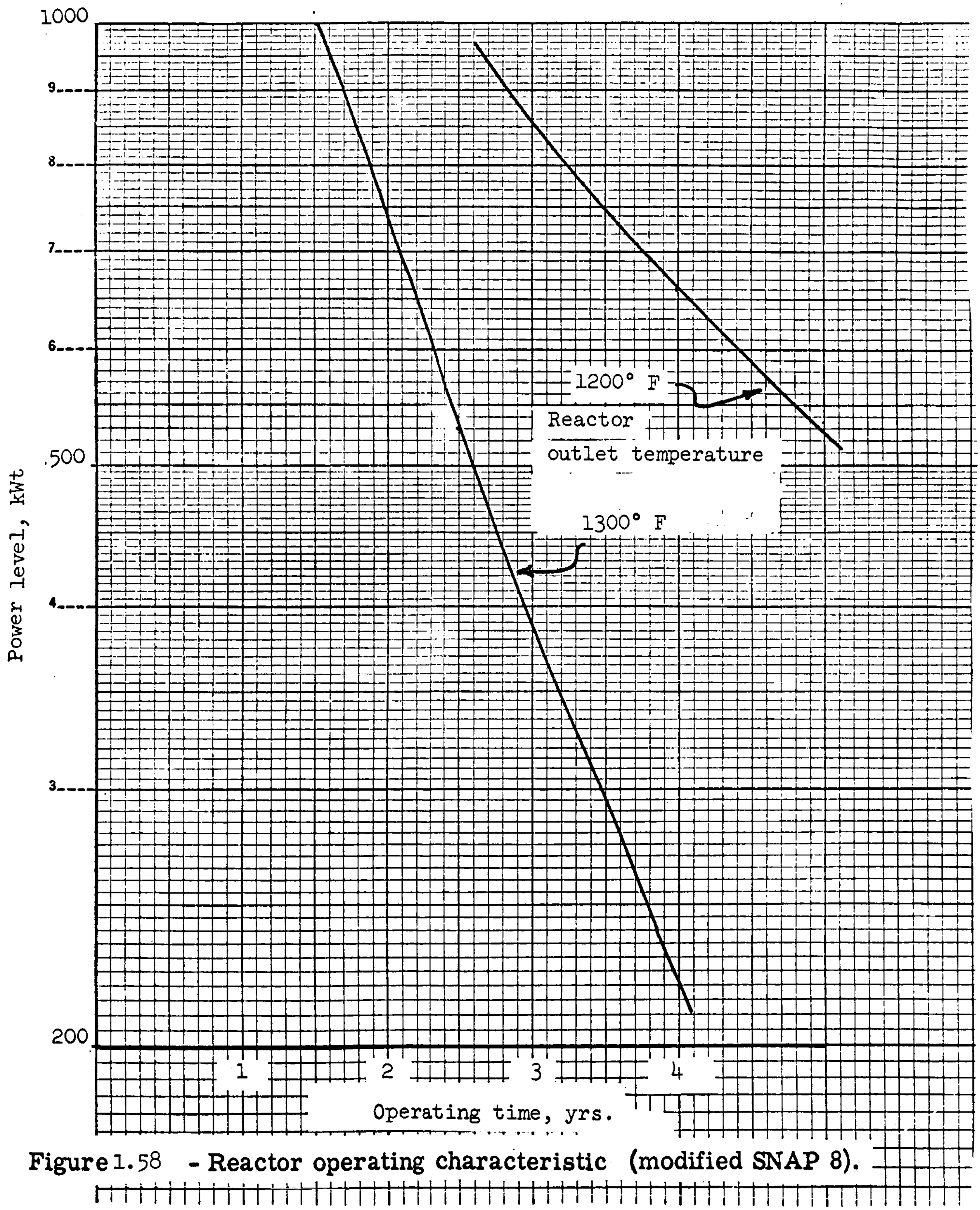
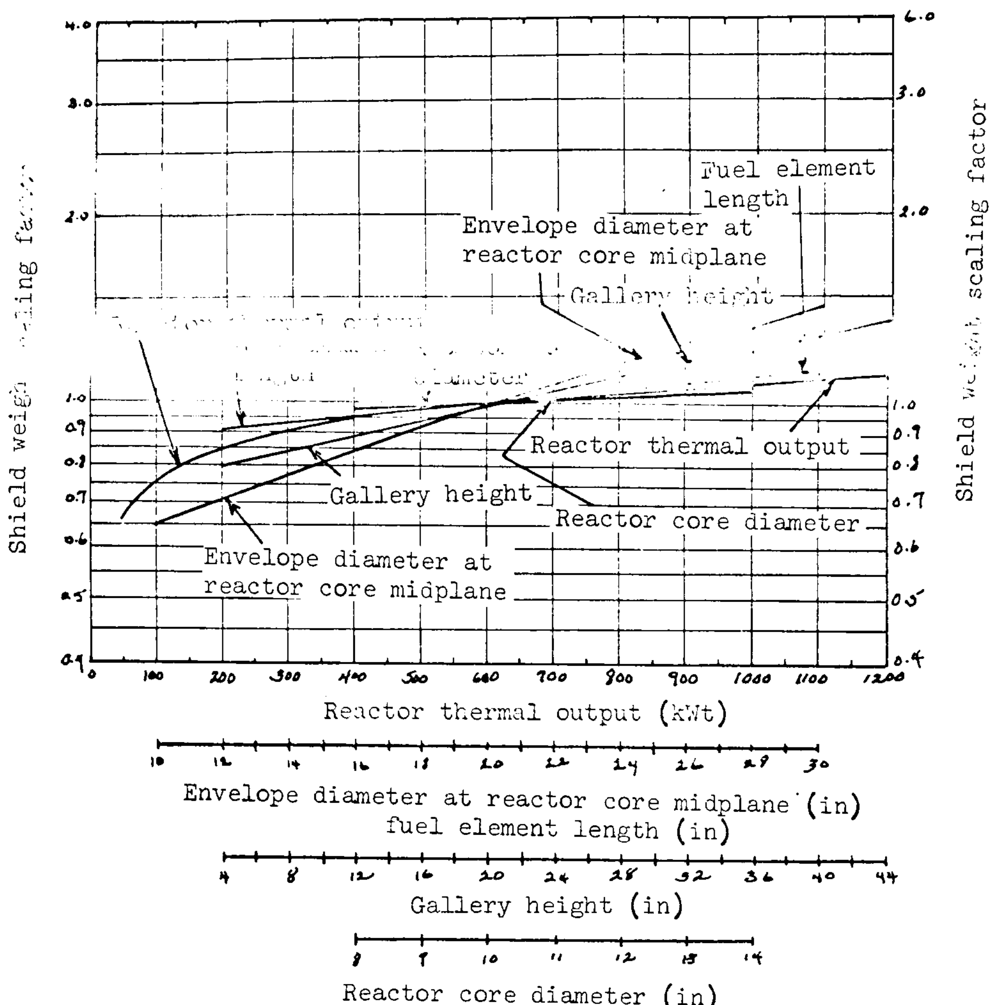
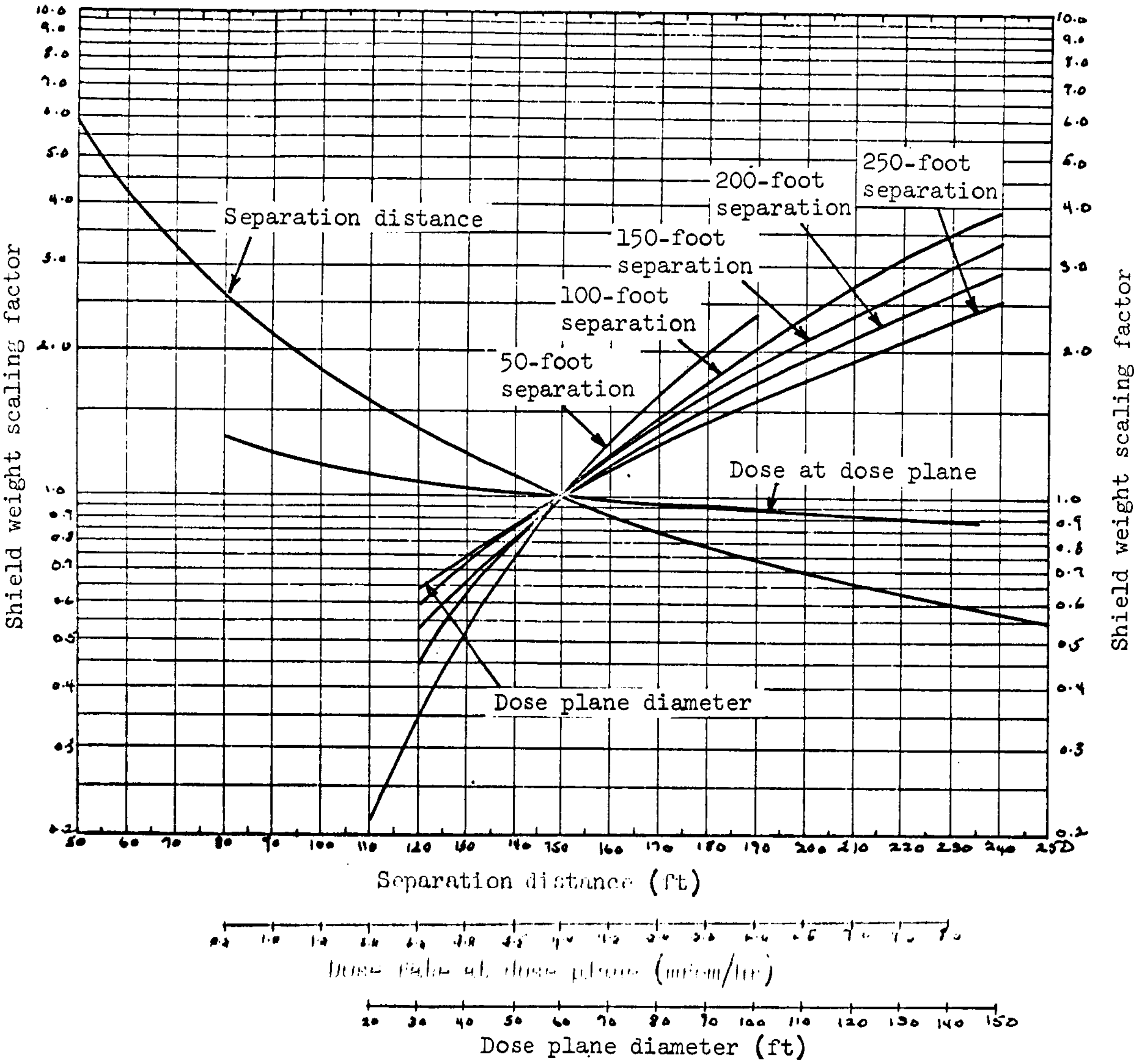


Figure 1.58 - Reactor operating characteristic (modified SNAP 8).



Reference case	
Parameter	Value
Reactor thermal output	600.0 kWt
Envelope diameter at reactor core midplane	70.0 in
Fuel element length	70.0 in
Reactor core diameter	10.0 in
Gallery height	80.0 in
Generation distance	180.0 ft
Dose plane diameter	40.0 ft
Dose rate at dose plane	4.0 mrem/hr
Shield weight	4056.0 lb

Figure 1.59 **- Shadow shield weight as a function of reactor and shield parameters.**



Reference Case	
Parameter	Value
Reactor thermal output	600.0 kWt
Envelope diameter at reactor core midplane	70.0 in
Fuel element length	70.0 in
Reactor core diameter	10.0 in
Gallery height	80.0 in
Generation distance	180.0 ft
Dose plane diameter	40.0 ft
Dose rate at dose plane	4.0 mrem/hr
Shield weight	4056.0 lb

Figure 1.60 - **Shadow shield weight as a function of envelope and dose plane parameters.**

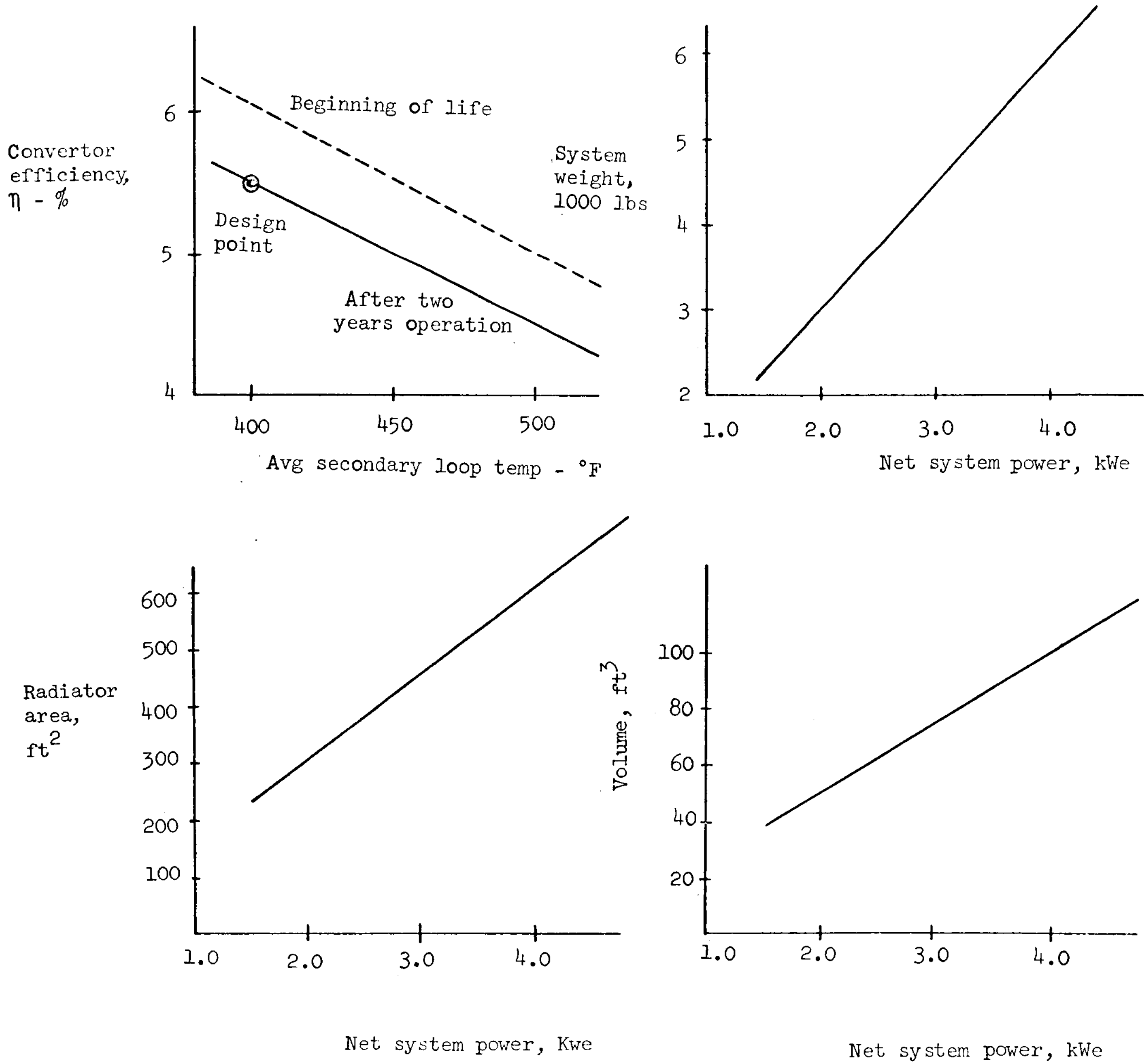


Figure 1.61 - Pb-Te convertor

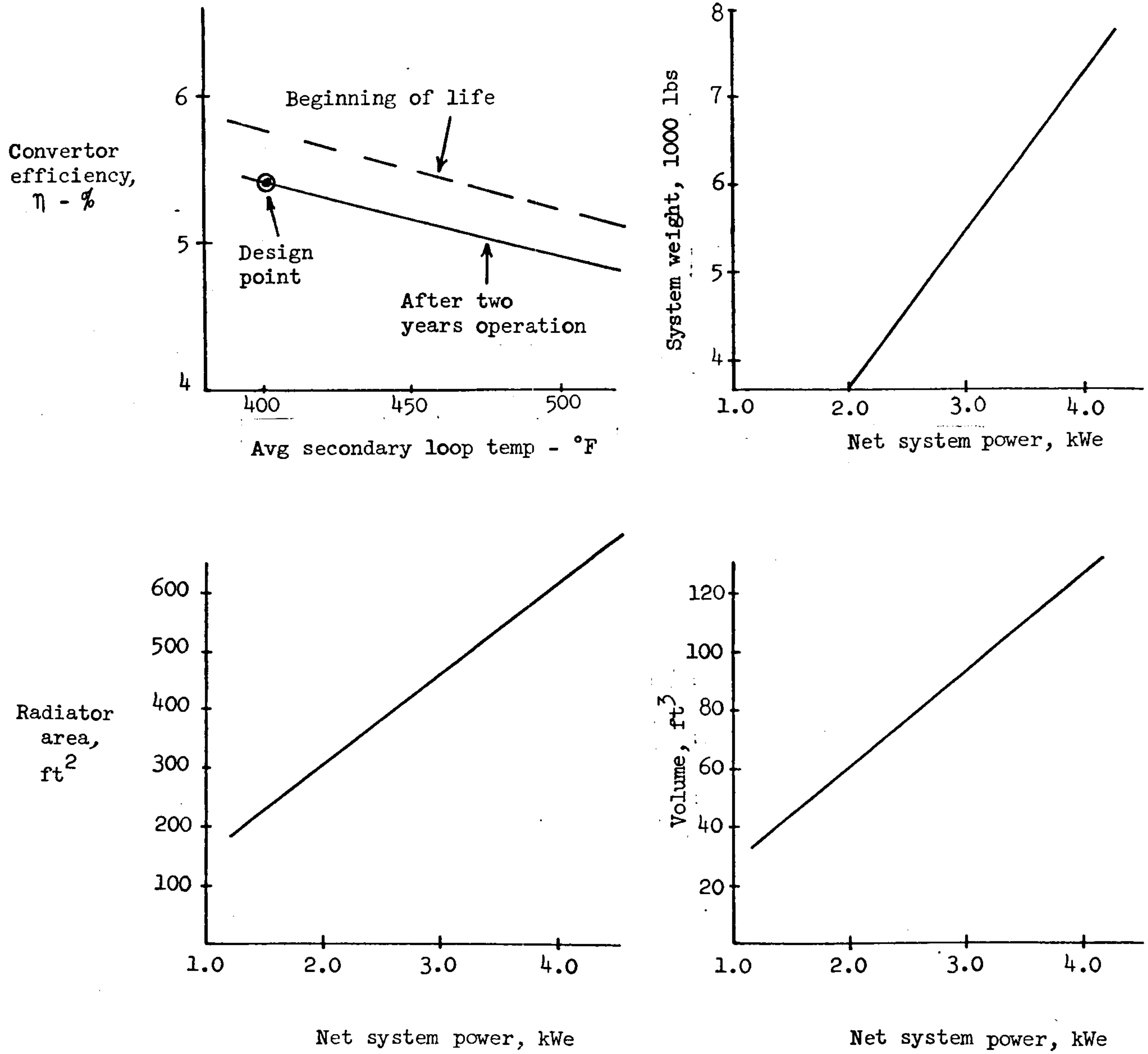
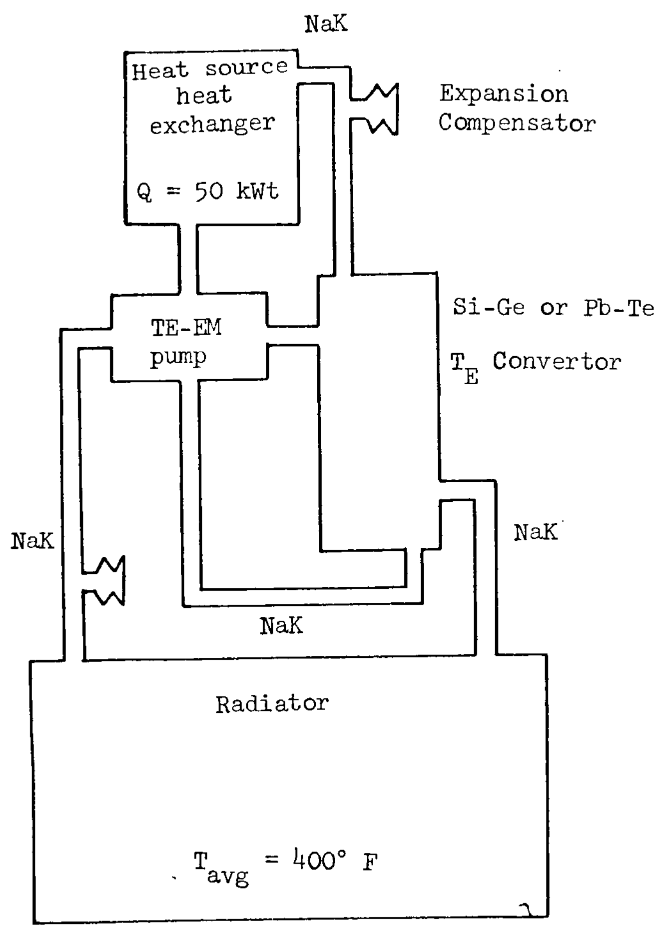


Figure 1.62 - Si-Ge convertor.



Summaries (2.0 kWe net)

	Pb-Te Compact	Si-Ge Compact
Thermal power - kWt	50	50
Primary NaK loop		
Converter inlet temp °F	1100	1500
Converter outlet temp °F	950	1350
NaK flow lb/hr	5700	
Secondary NaK loop		
Convertor outlet °F	475	457
Convertor inlet °F	325	325
NaK flow lb/hr		
Radiator area - ft^2	310	310
Weights - lbs		
Conversion	240	50
NaK pumps and plumbing	270	320
Heat source/heat exchanger	1500	2400
Radiological shield	500	500
Radiator differential weight	310	310
Total	2820	3580

Figure 1.63 . - PU 238 thermoelectric module

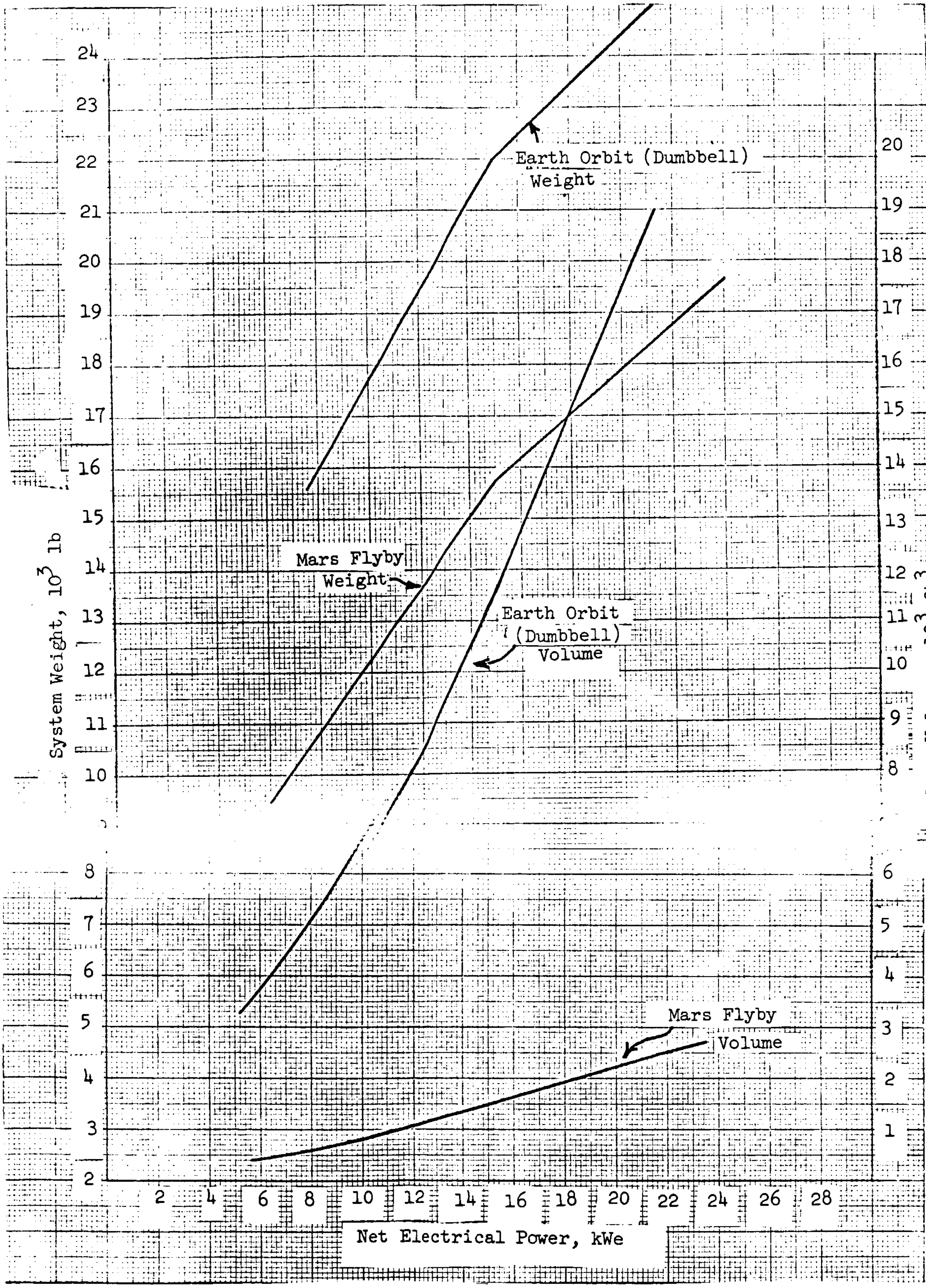


Figure 1.64 - Snap-8 reactor thermoelectric system concept.

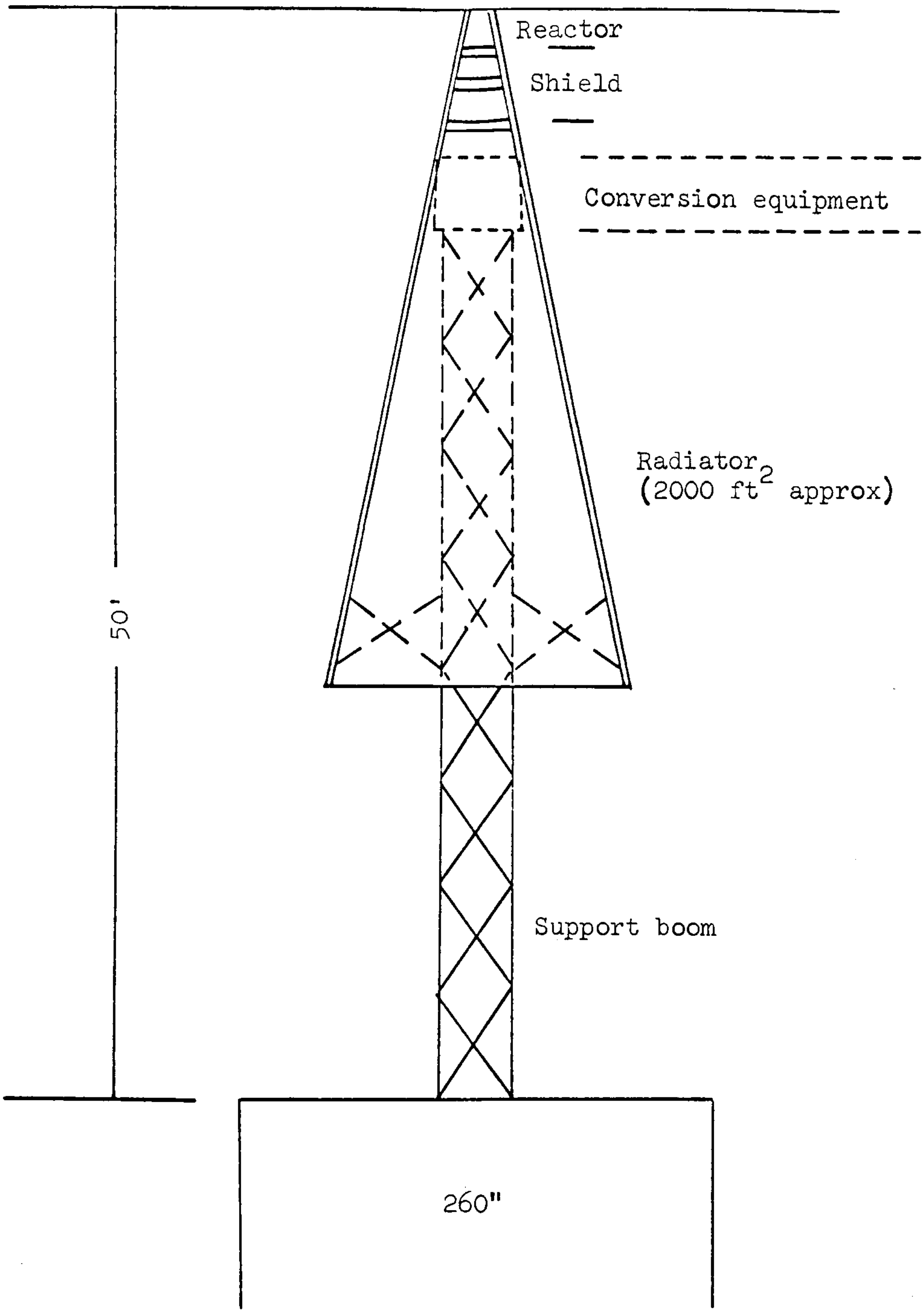


Figure 1.65 - Reactor thermoelectric system concept, Mars flyby.

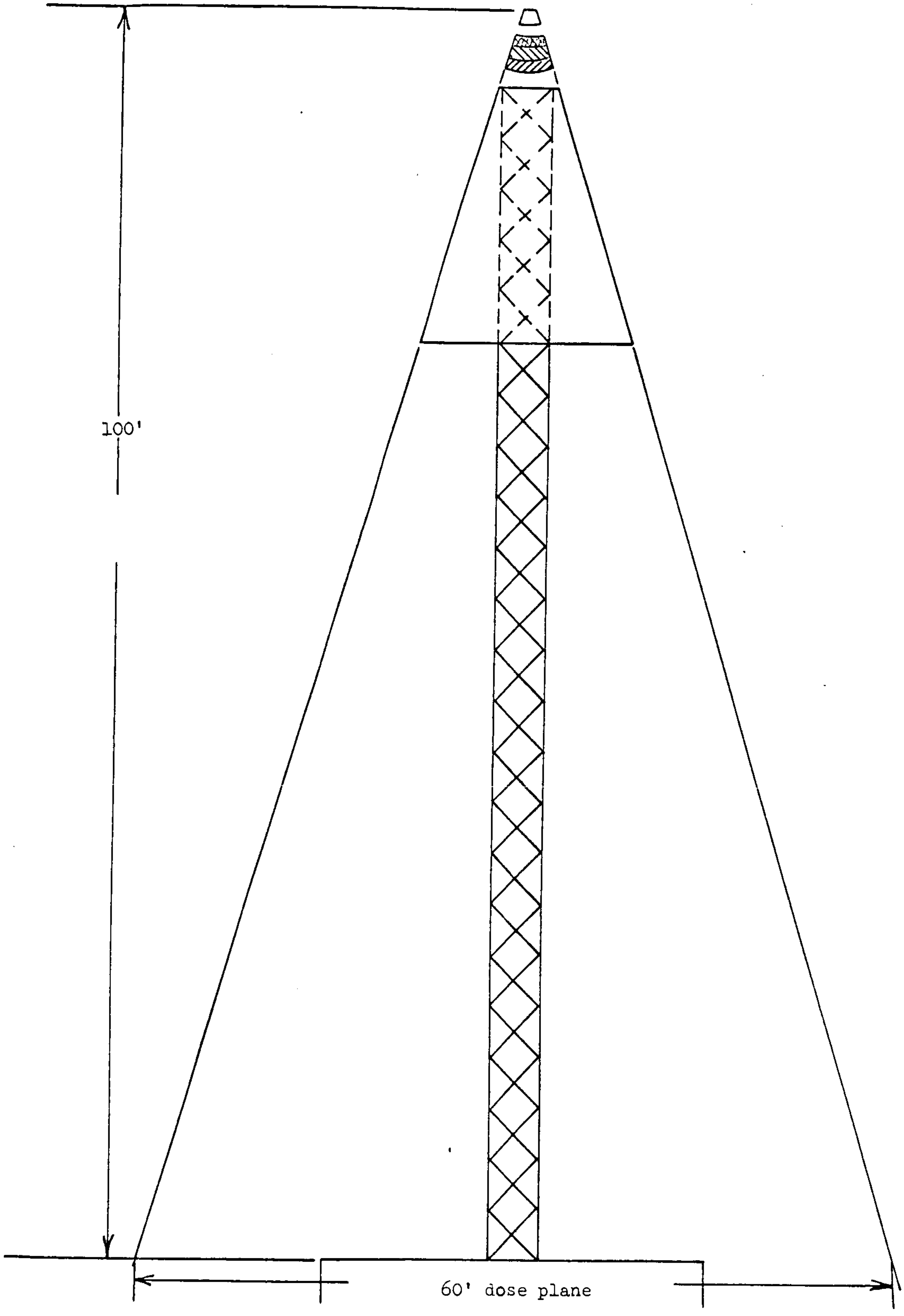


Figure 1.66 - Reactor thermoelectric system concept, LEO.

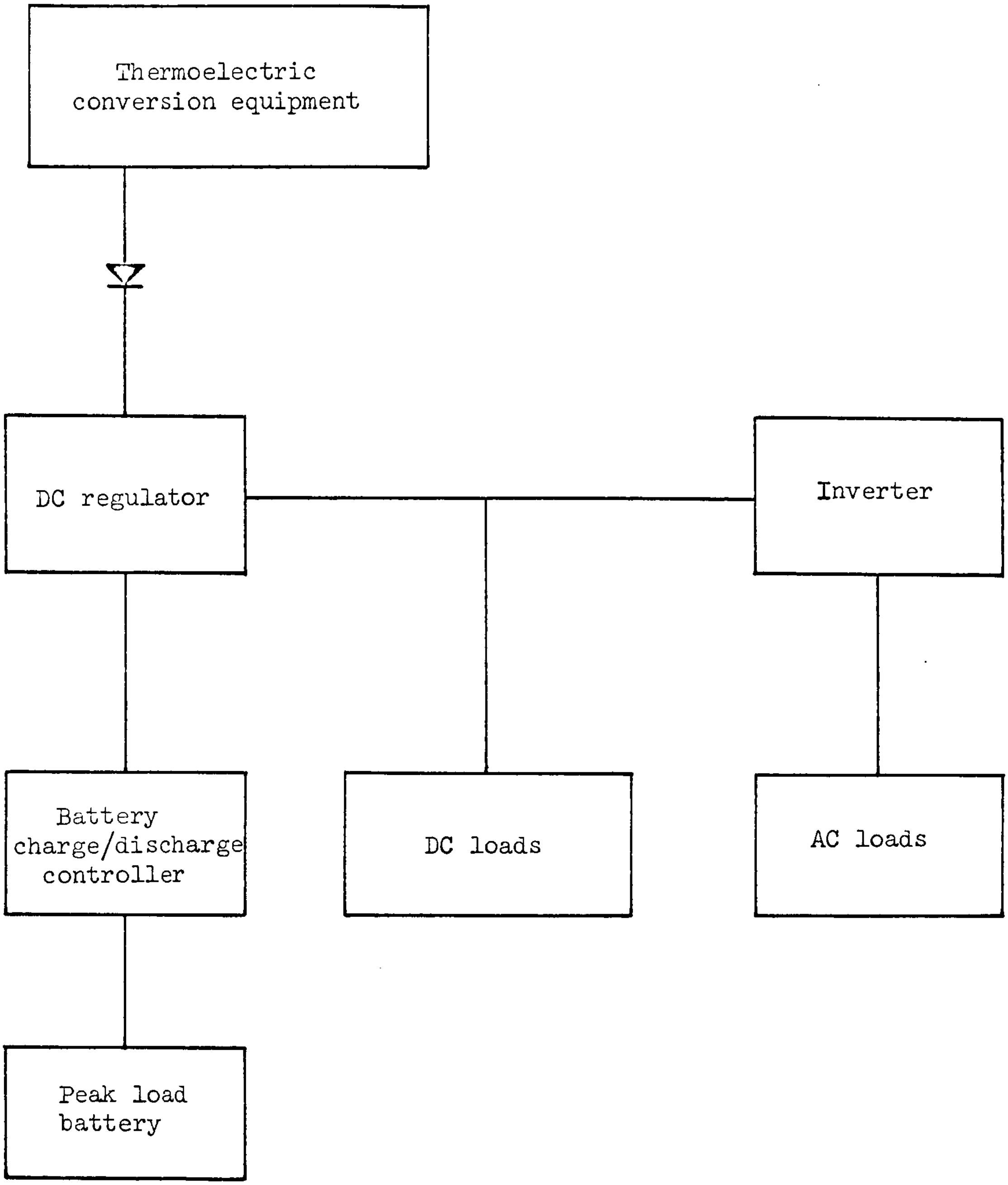


Figure 1.67.- Thermoelectric electrical schematic.

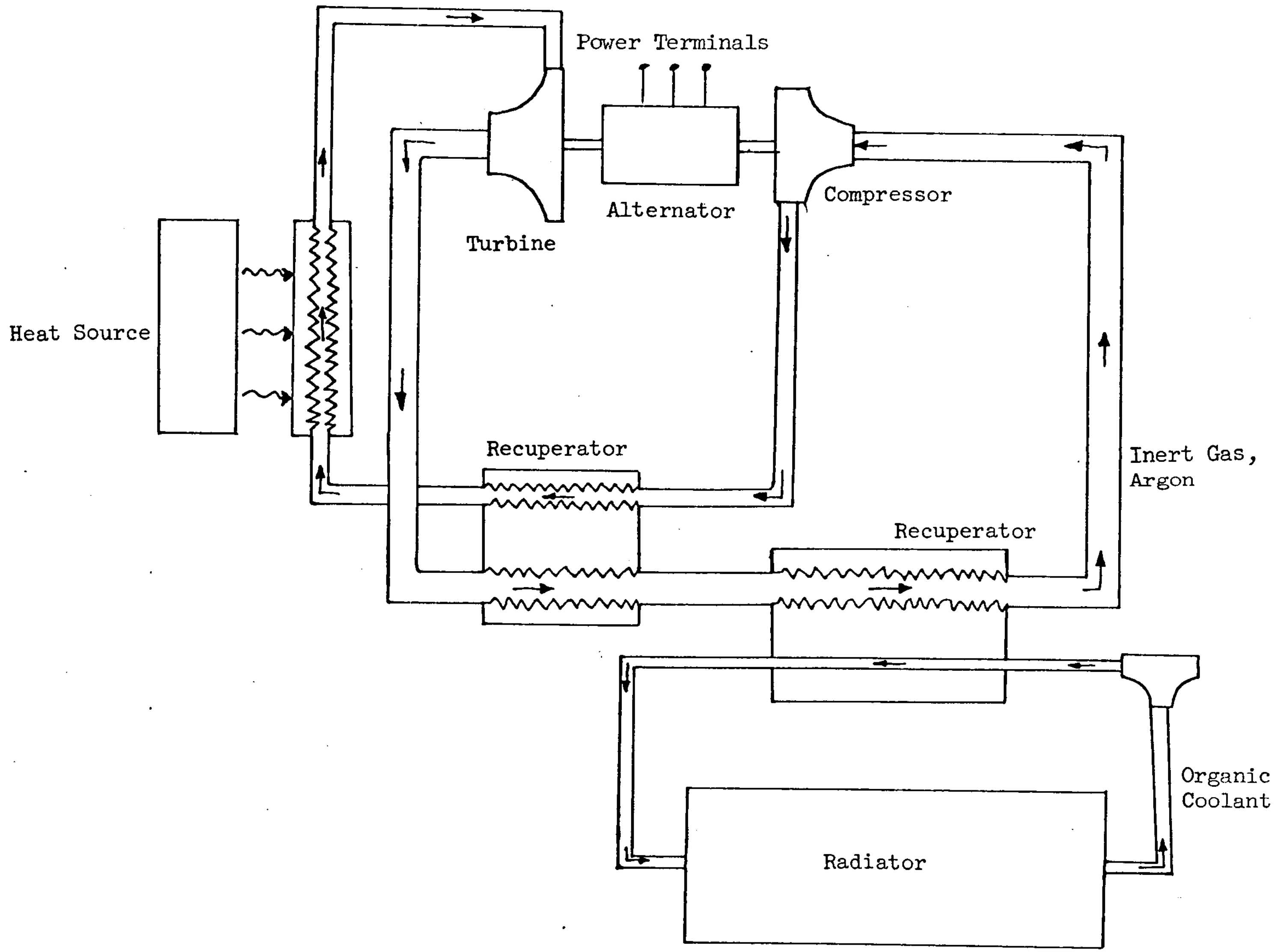


Figure 1.68 - Radioisotope Brayton-cycle system schematic.

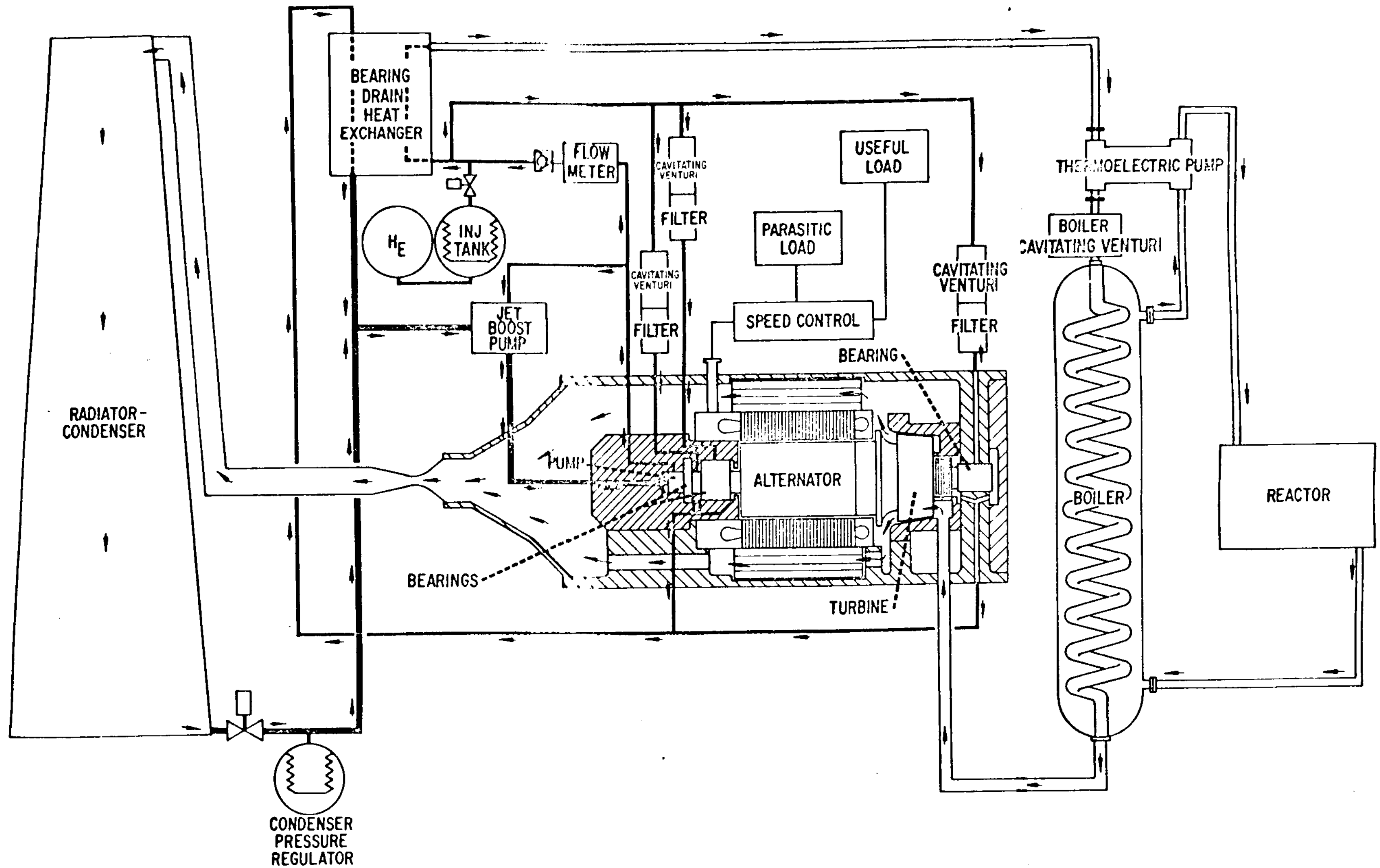


Figure 1.69 - Reactor mercury-Rånkine system.

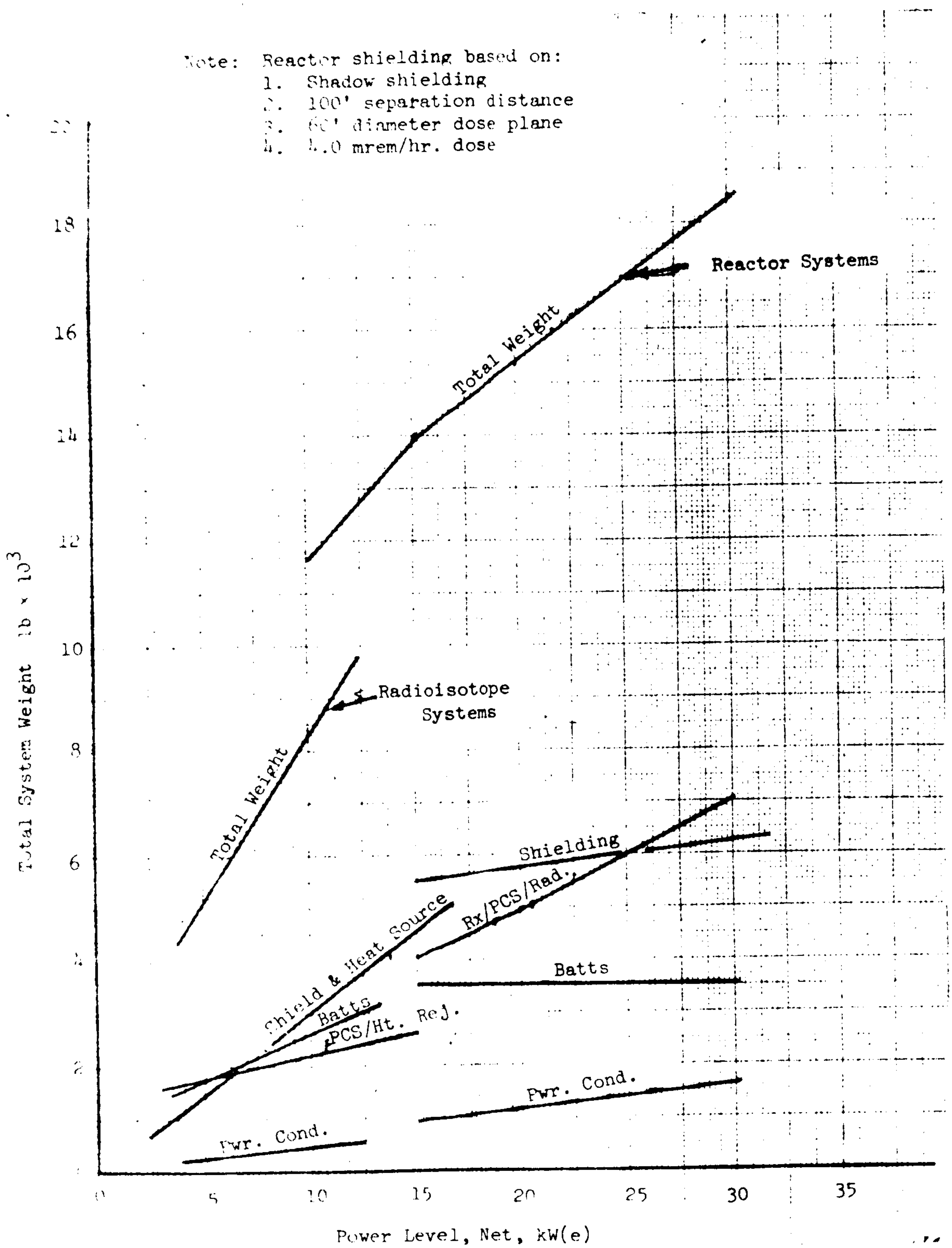


Figure 1-70 - Dumbbell space station: Total system weight versus net power level for reactor and radioisotope systems, closed Brayton-cycle.

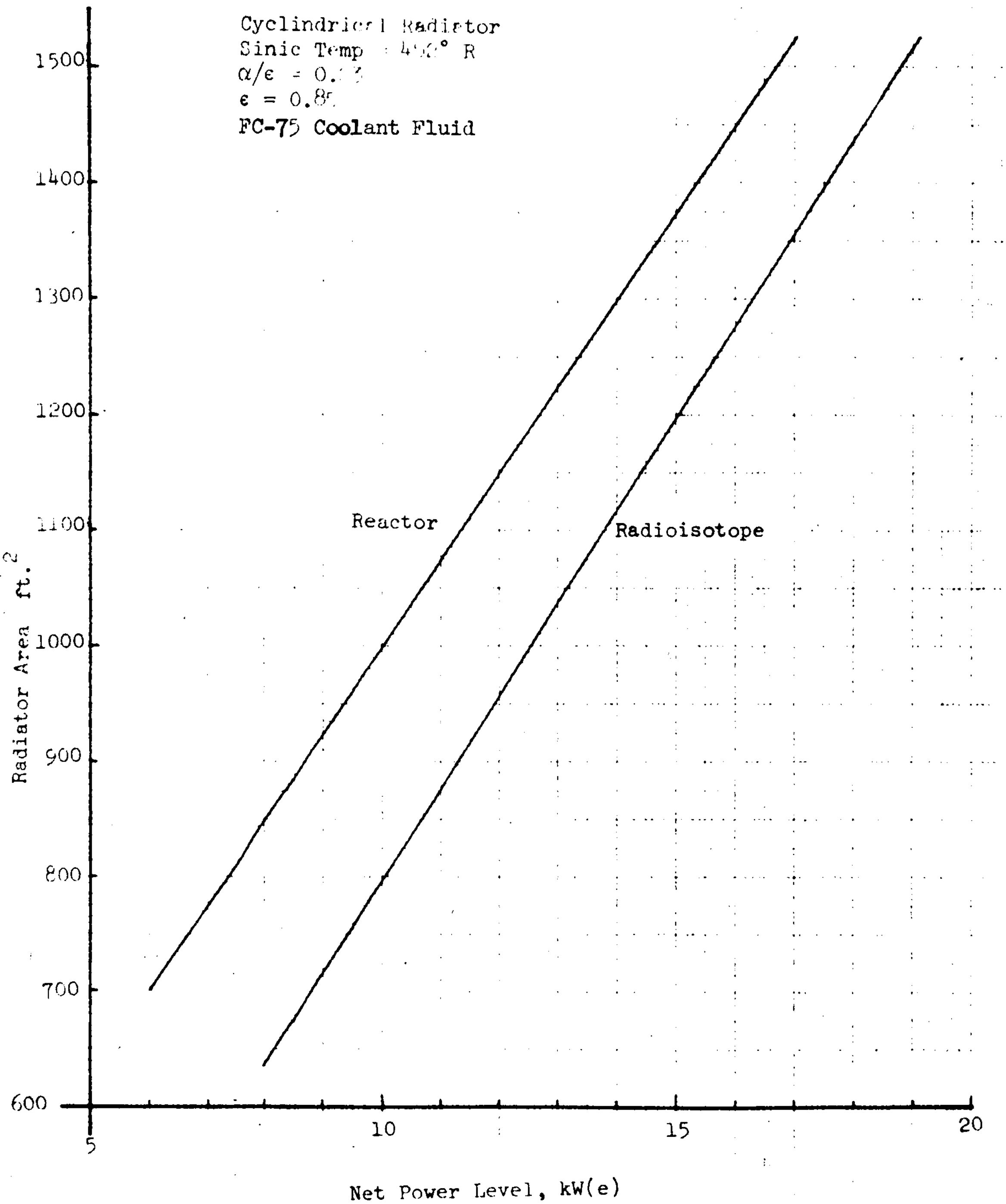


Figure 1-71 . - Radiator area required versus net power level for reactor and radioisotope Brayton-cycle systems.

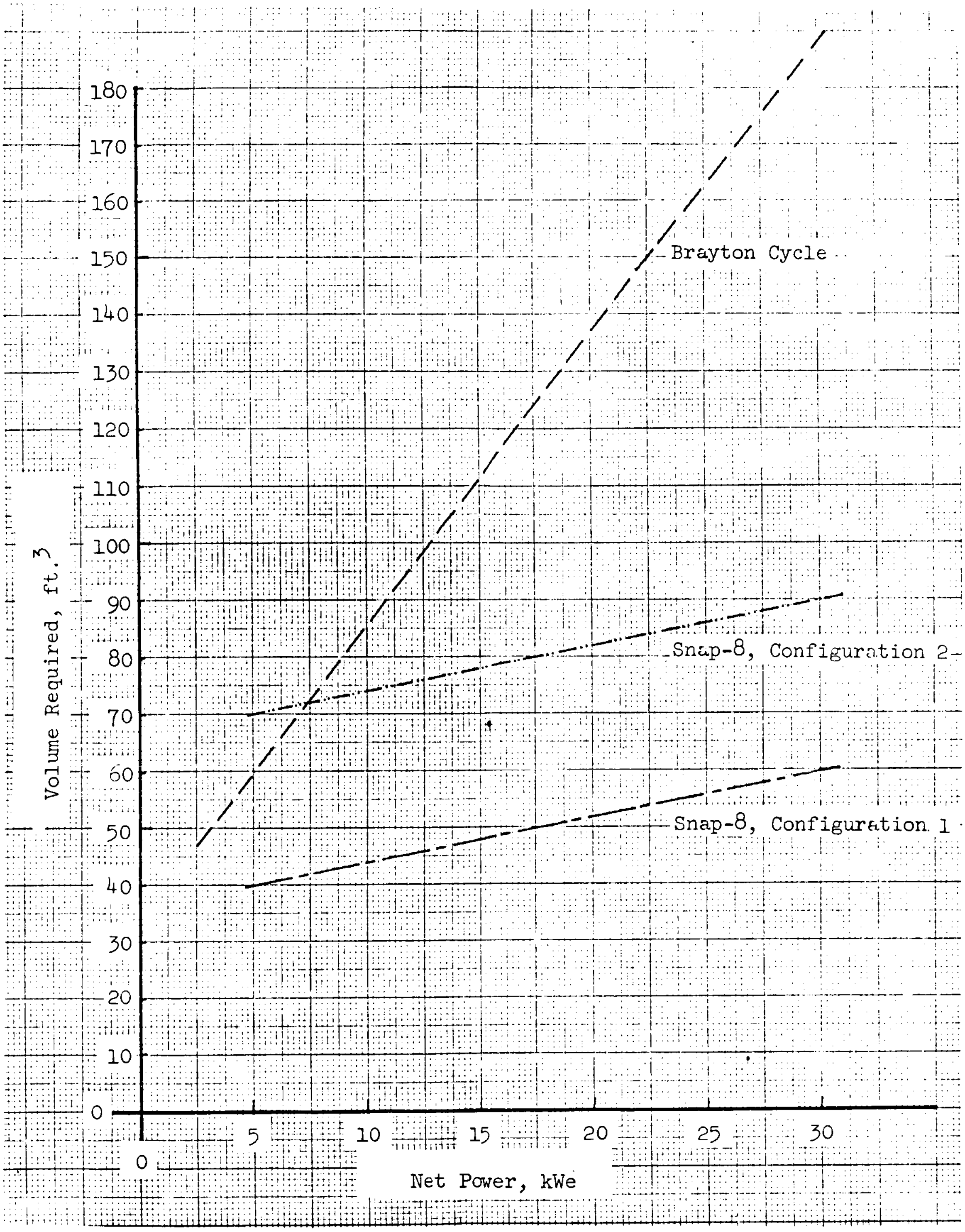


Figure 1-72 - Volume required versus net power for Brayton-cycle systems.

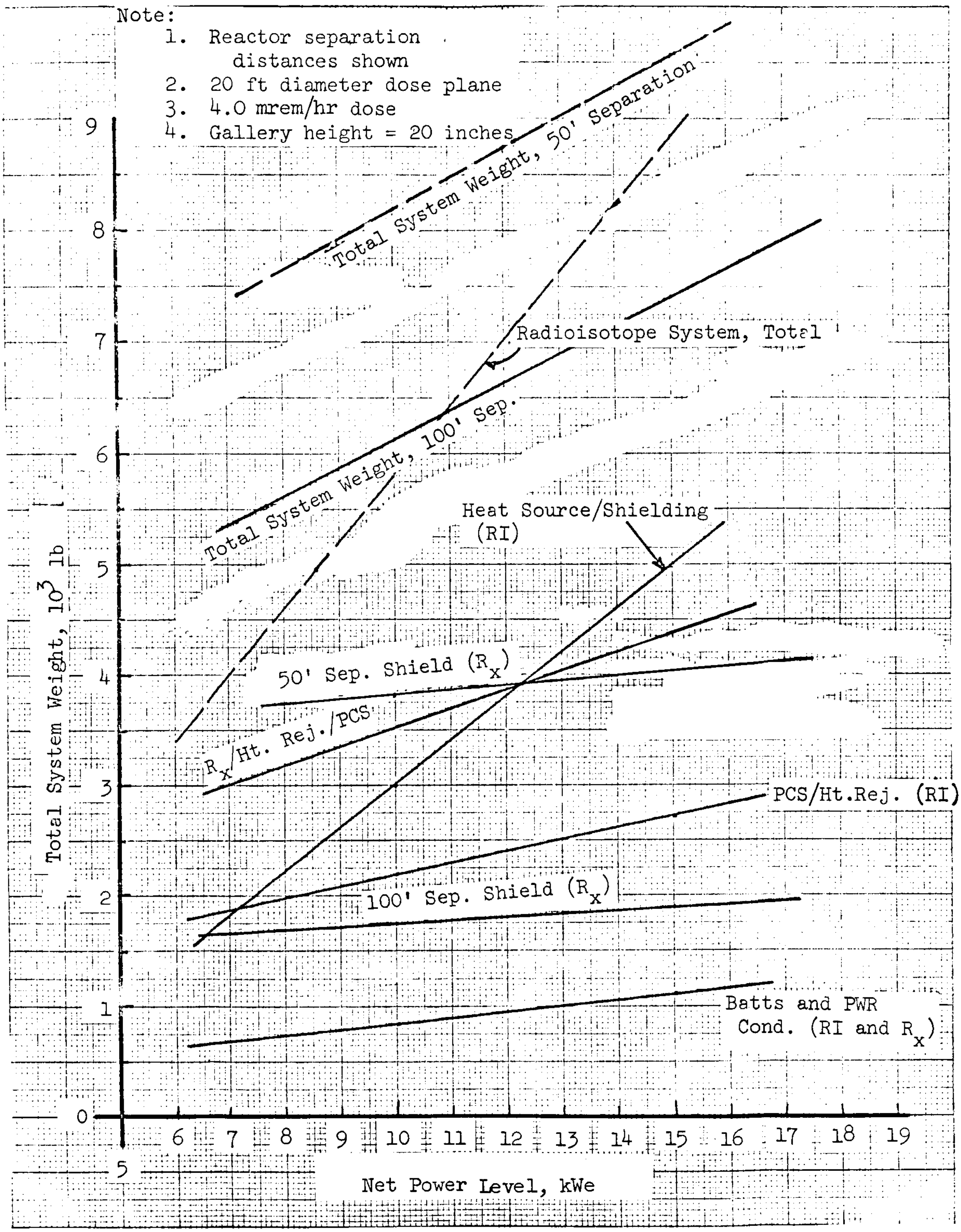


Figure 1.73 - Mars Flyby Mission: Total system weight versus net power level for reactor and radioisotope Brayton-cycle systems.

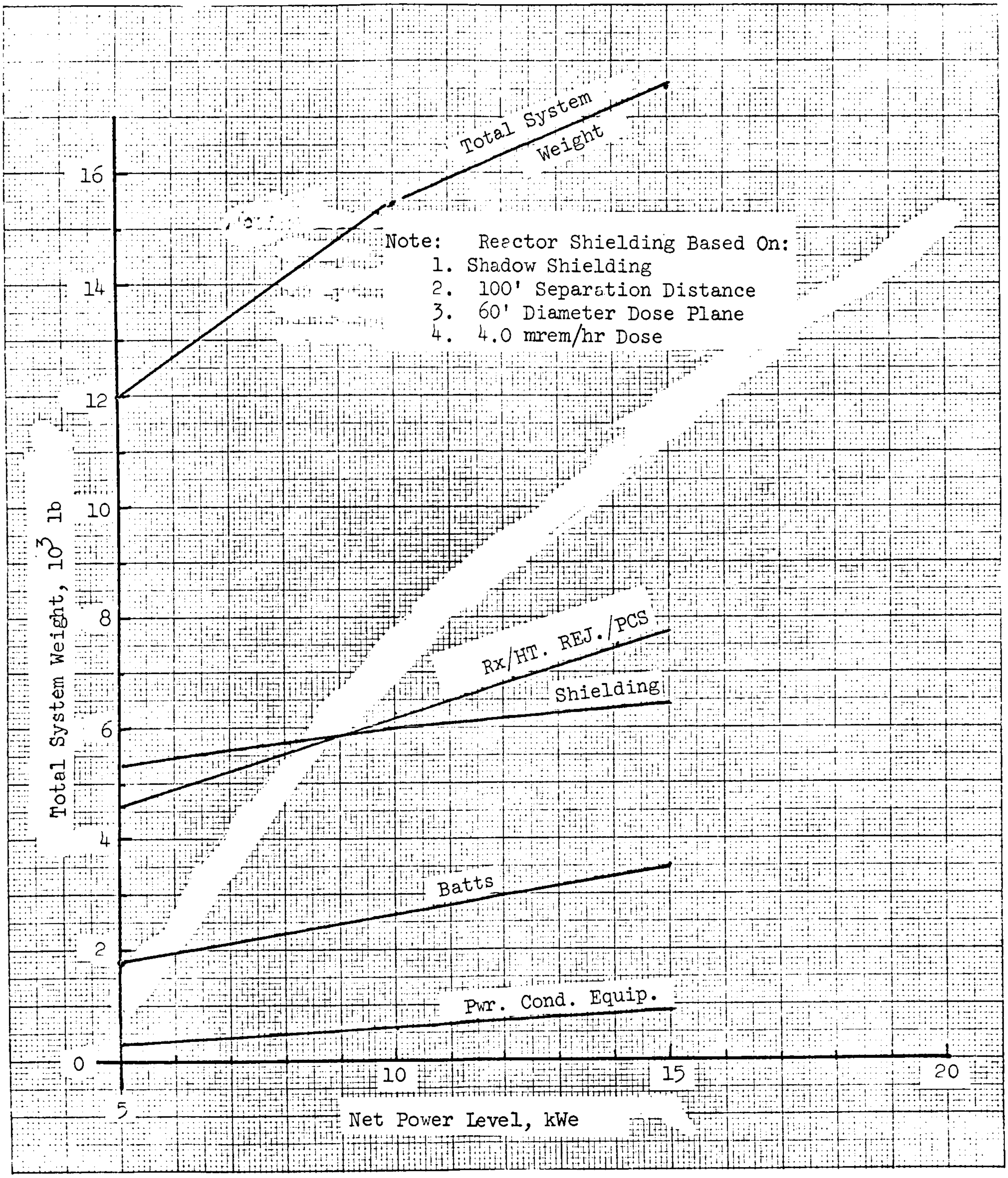


Figure 1-74 - Total system weight versus net power level for reactor Mercury-Rankine systems.

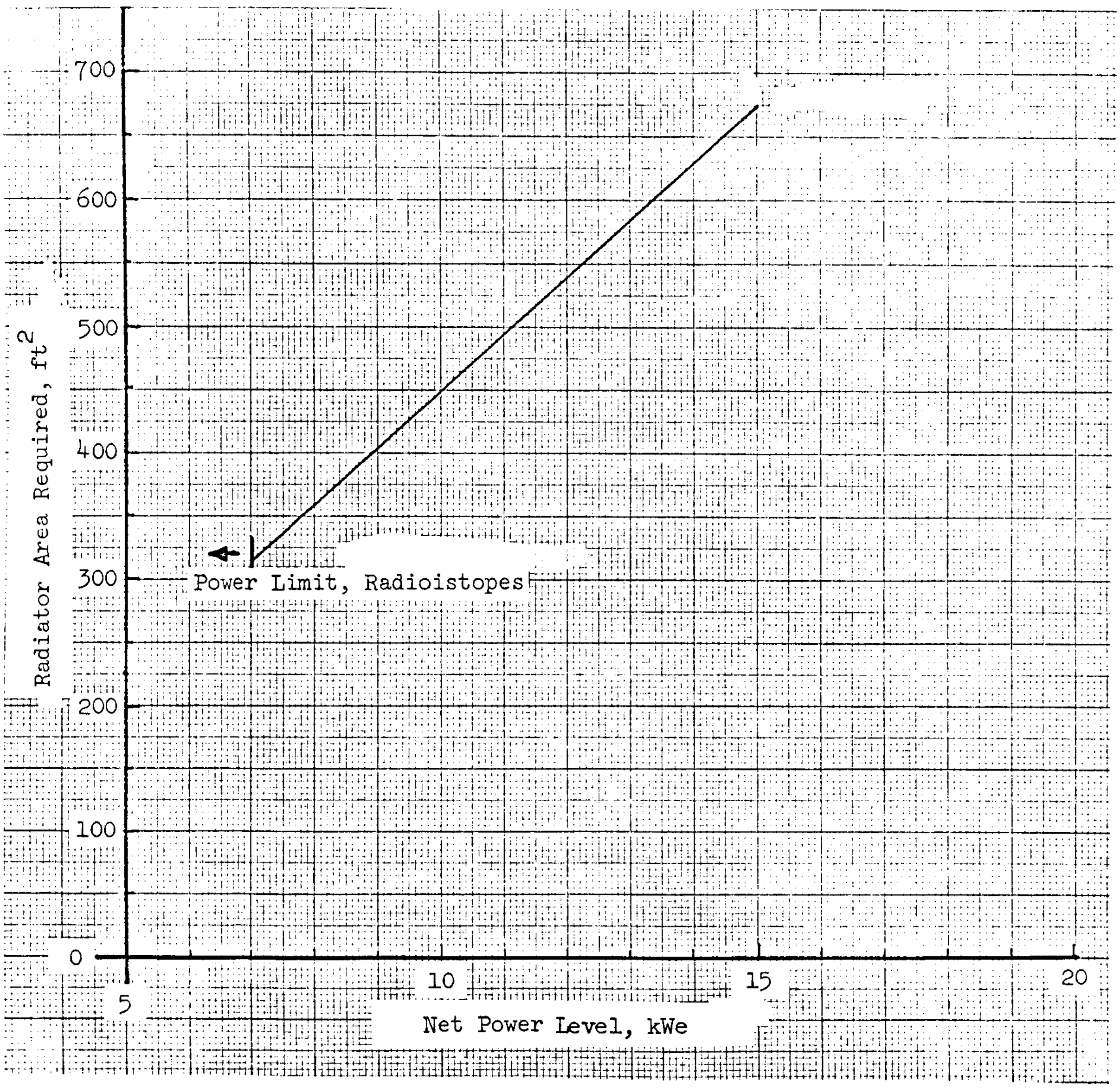


Figure 1-75 - Radiator area required versus net power level for radioisotope and reactor systems using mercury-Rankine.

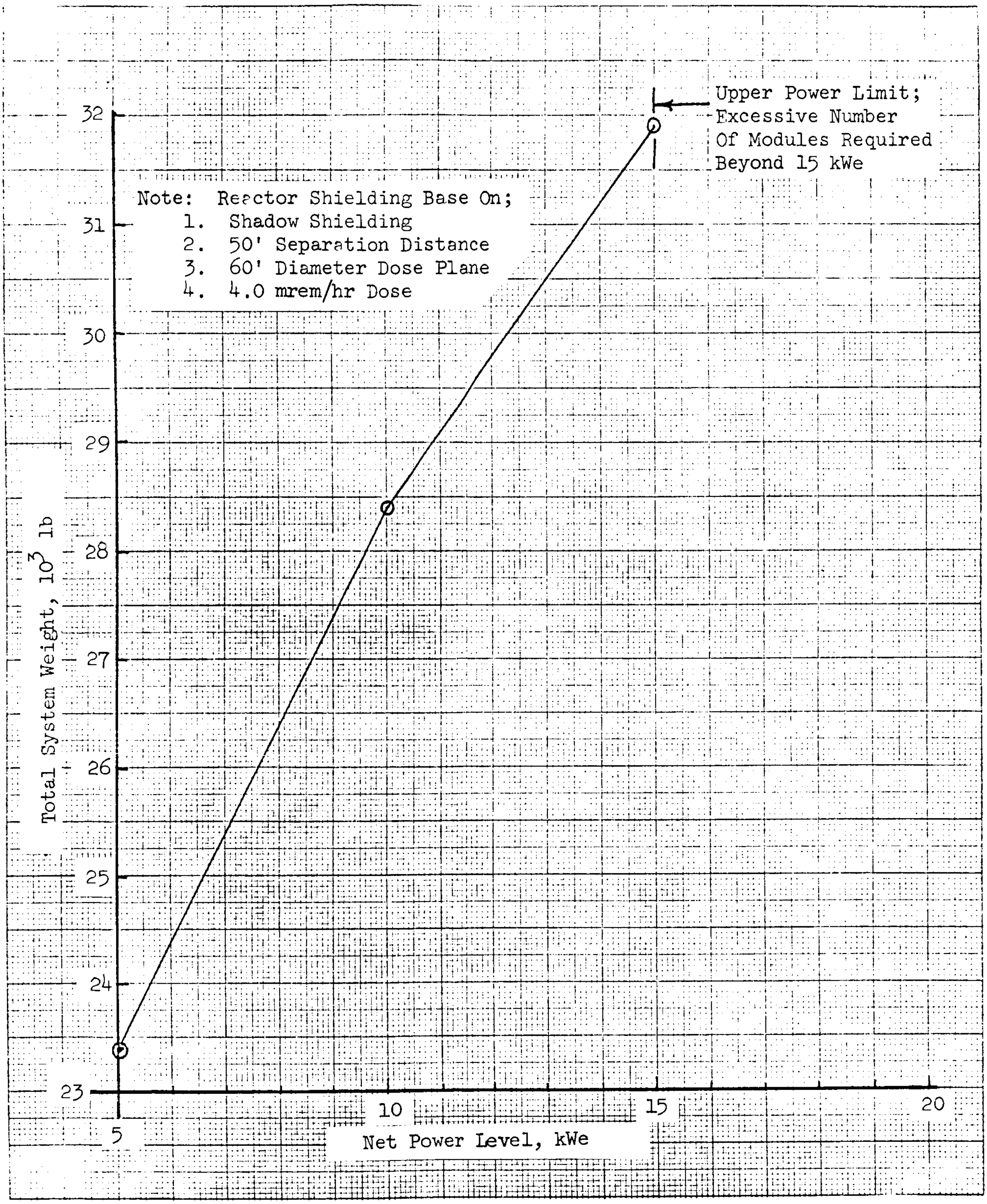


Figure 1.76 - **Zero-G Space Station: Total system weight versus net power level for reactor mercury-Rankine systems.**

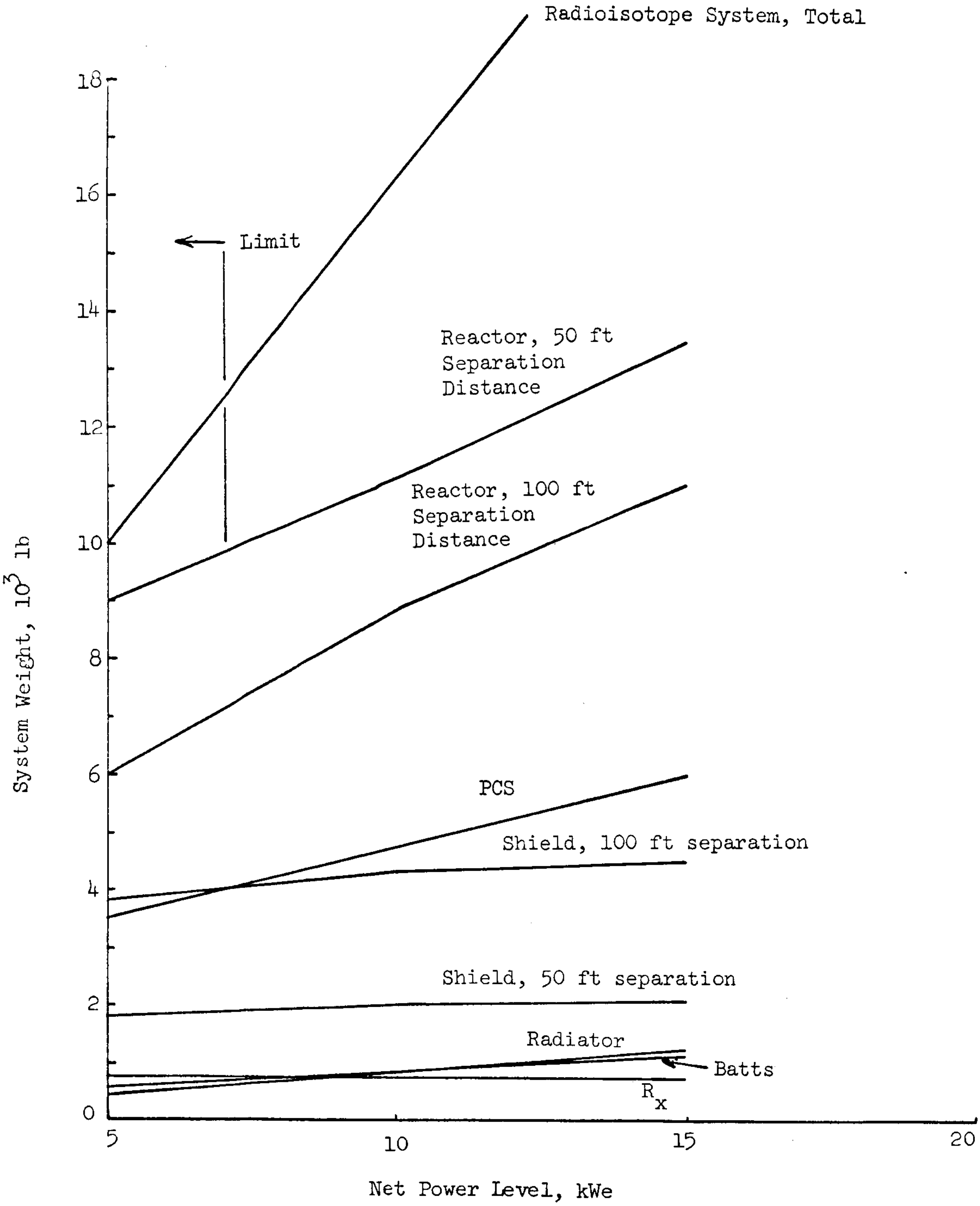


Figure 1.77 - Mars Flyby Mission: System weight vs net power level for radioisotope and reactor systems using mercury-Rankine.

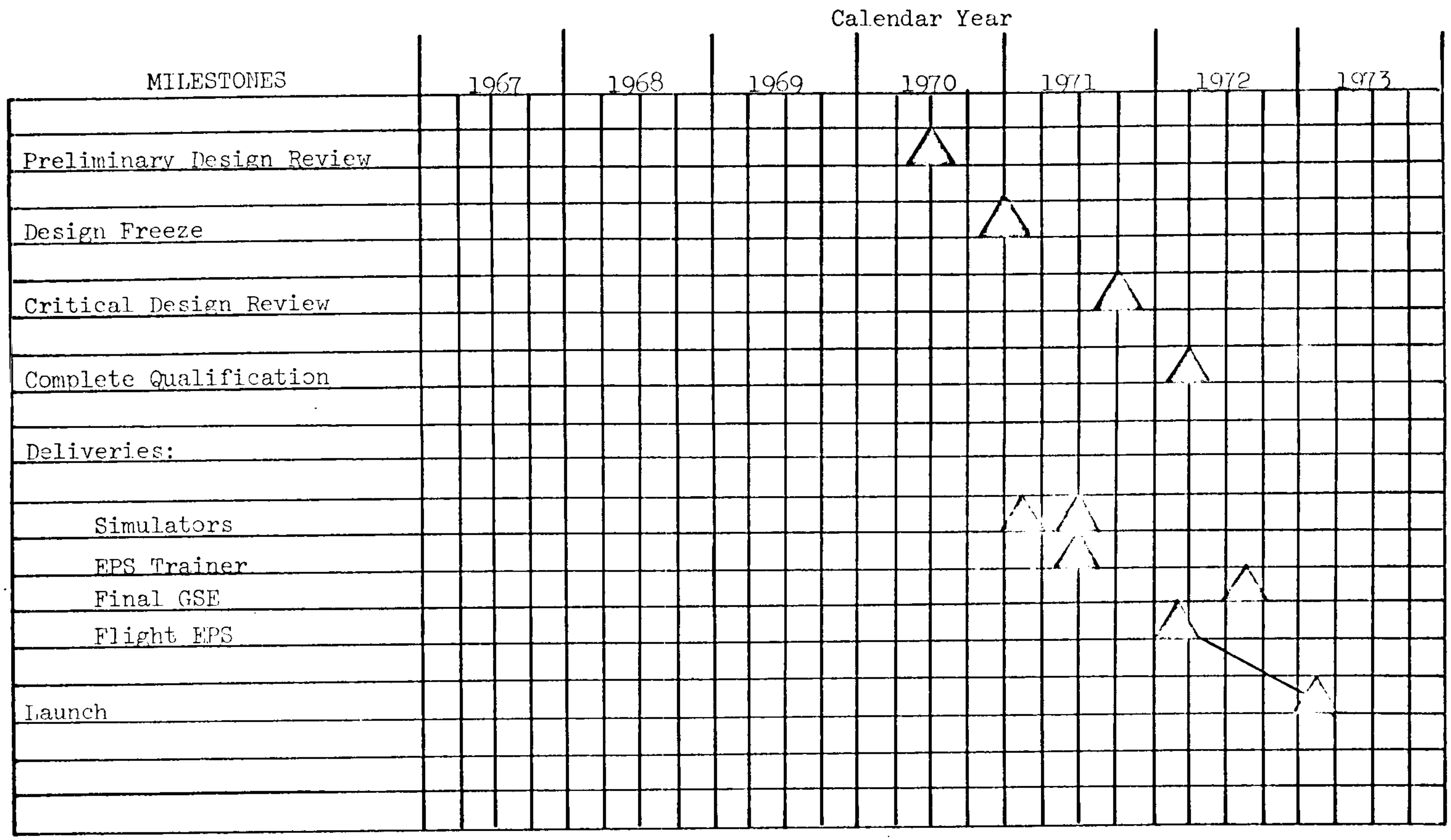


Figure 1-78 - Schedule requirements, space station EPS.

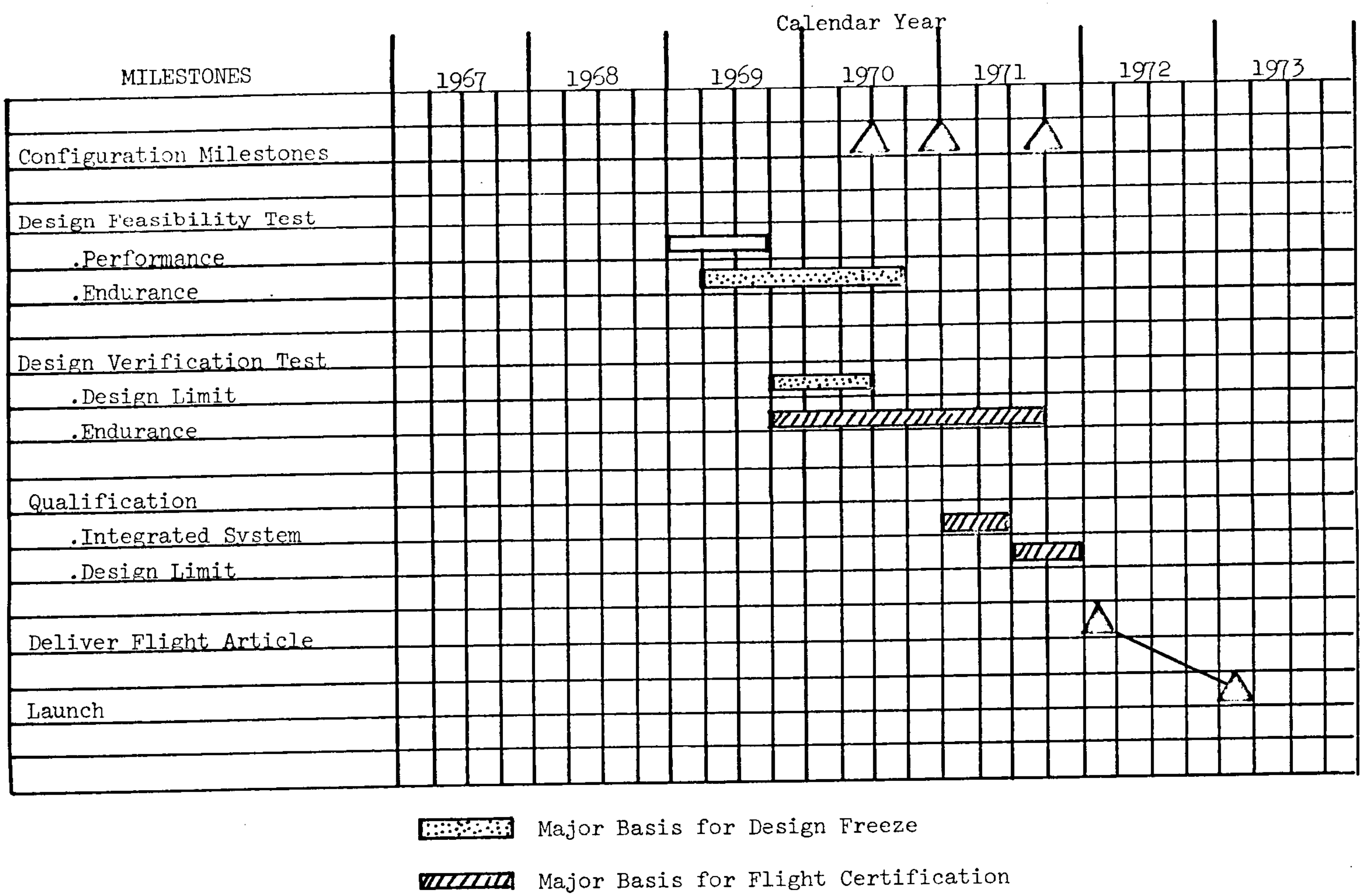


Figure 1.79 **- Ground test logic space station EPS.**

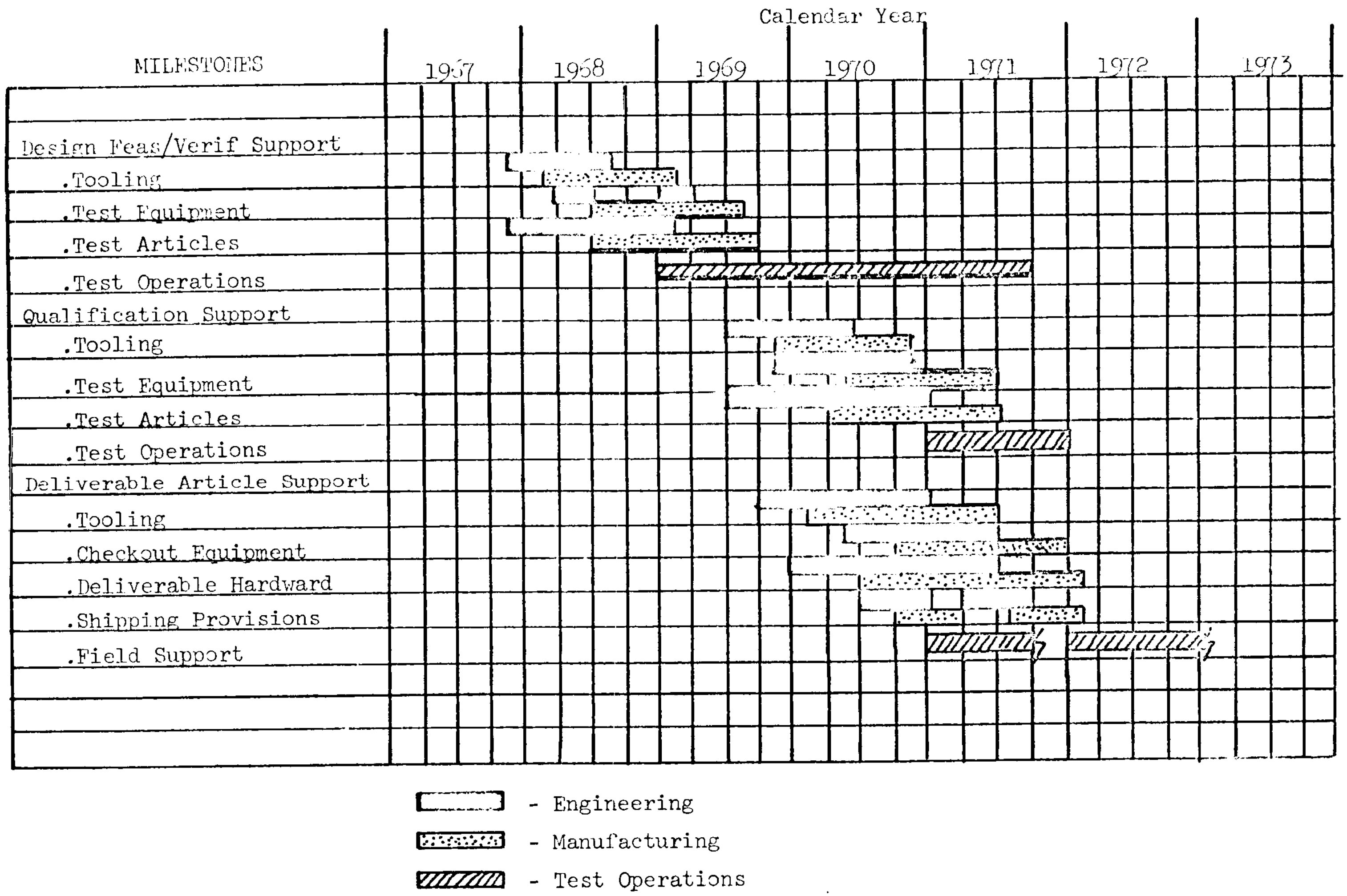


Figure 1.80 - Hardware support schedule, space station EPS.

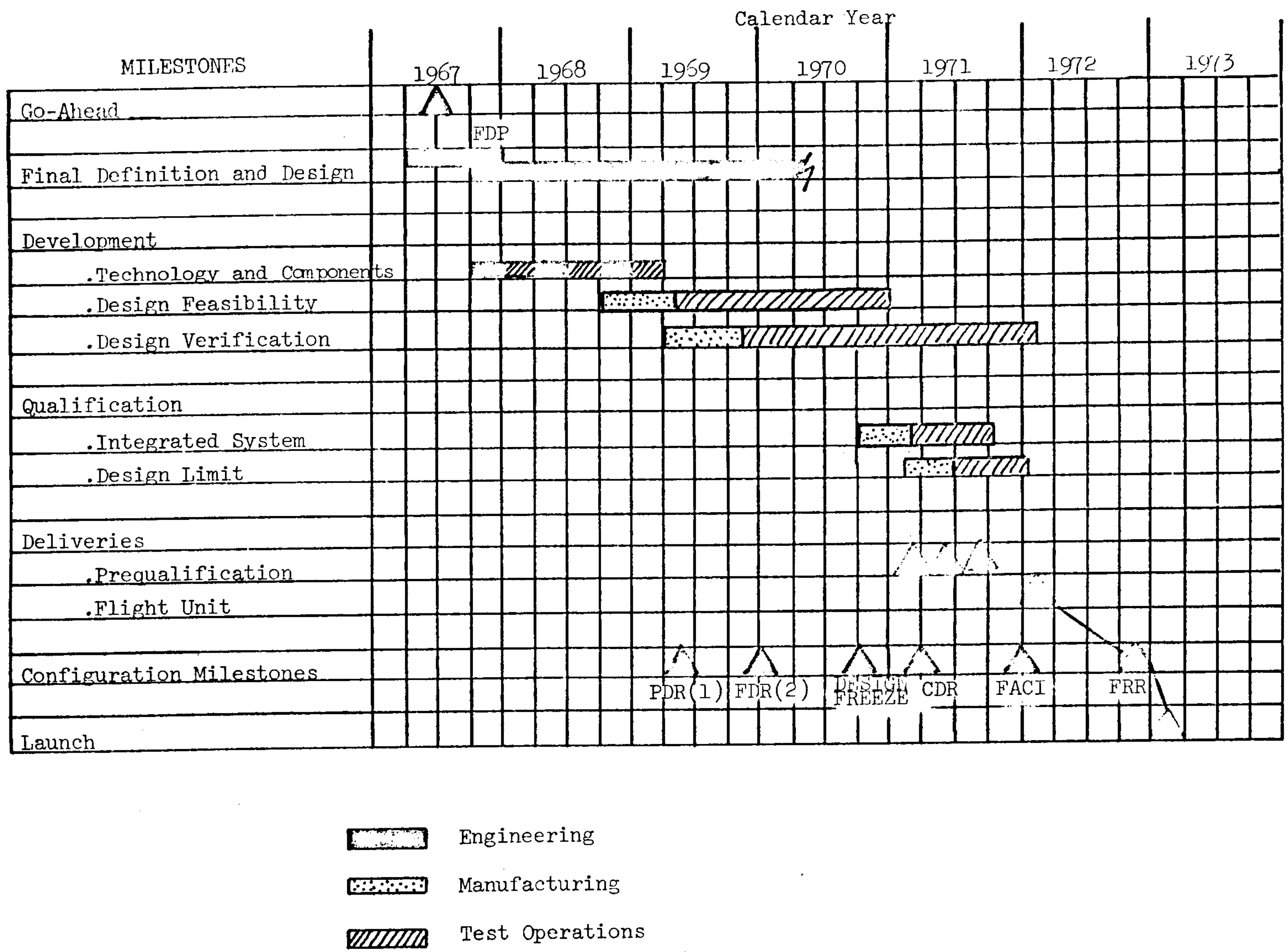


Figure 1.81 **- Isotope Brayton-cycle development schedule, power section.**

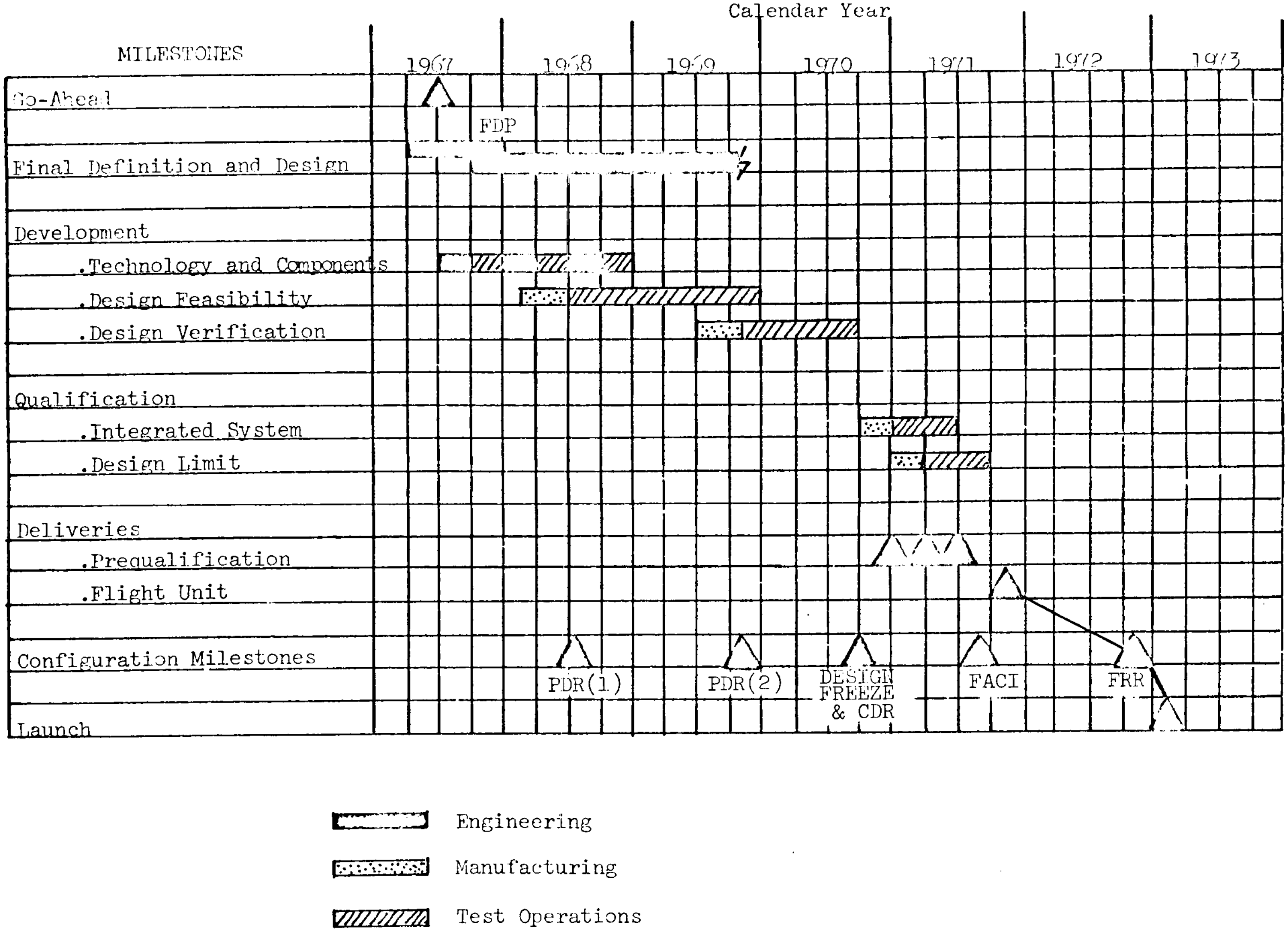


Figure 1.82 - Isotope heat block development schedule.

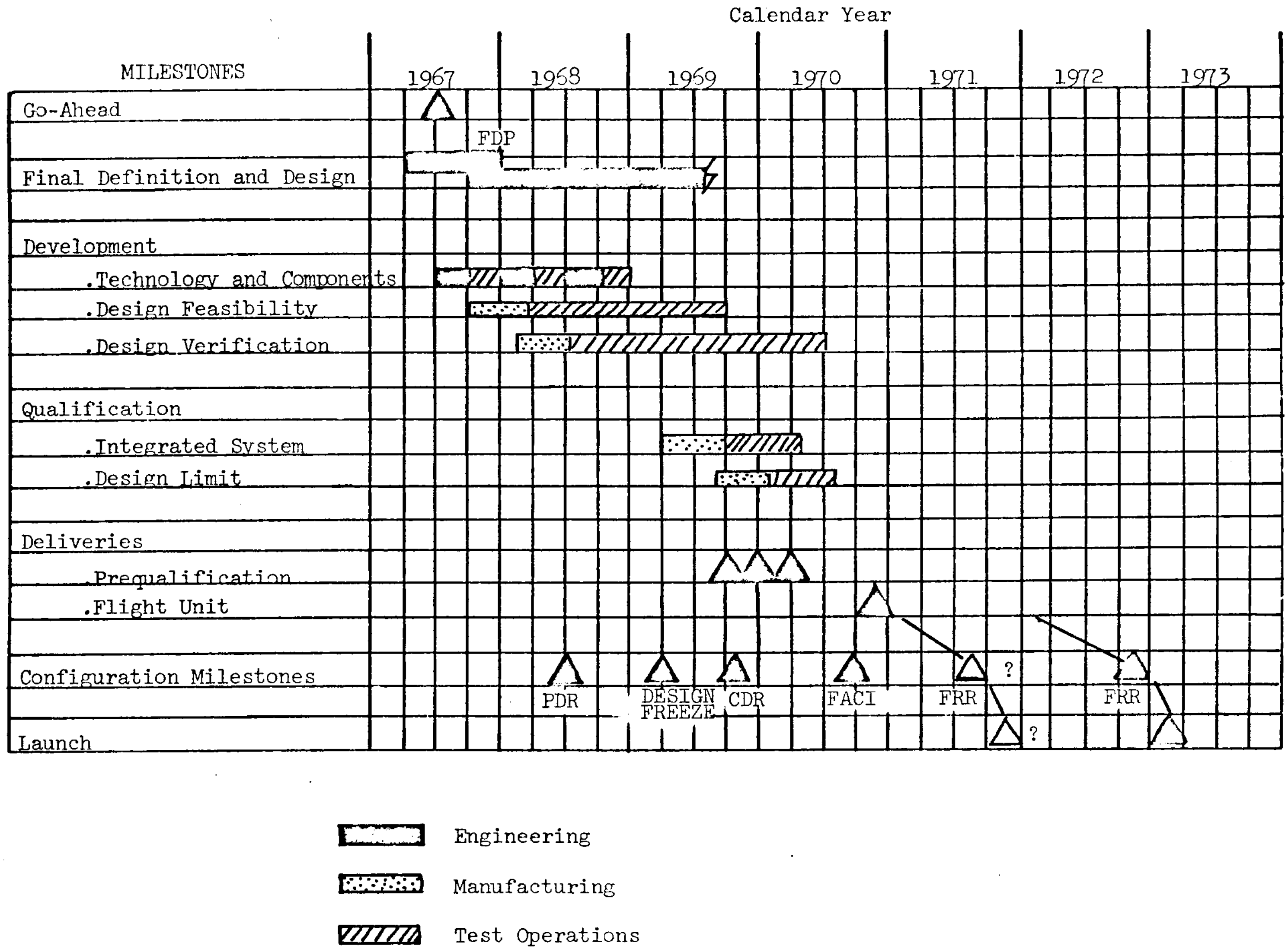


Figure 1.83 - Solar cell system development schedule.

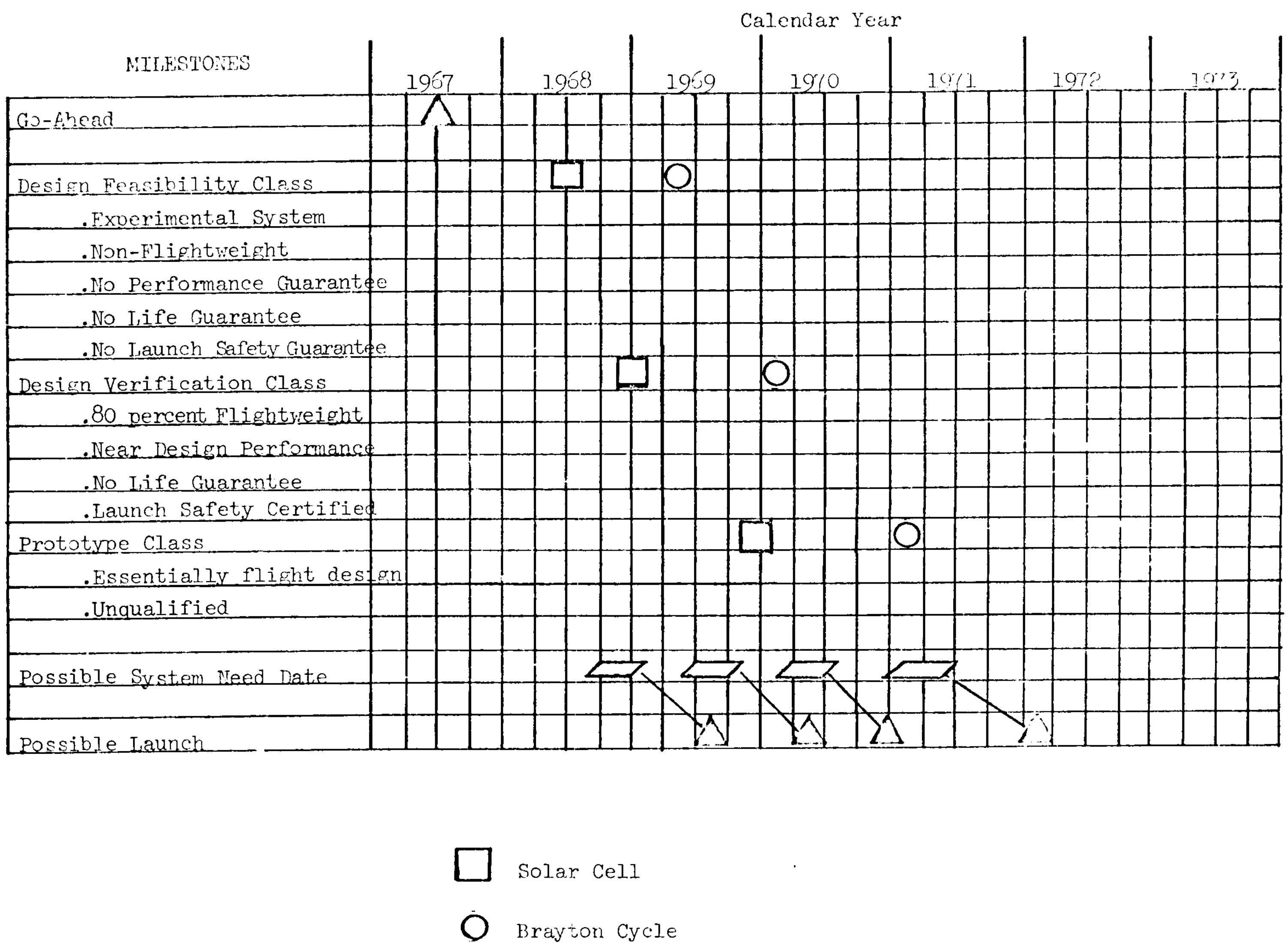


Figure 1.84 .- Primary EPS support to space station developmental flights.

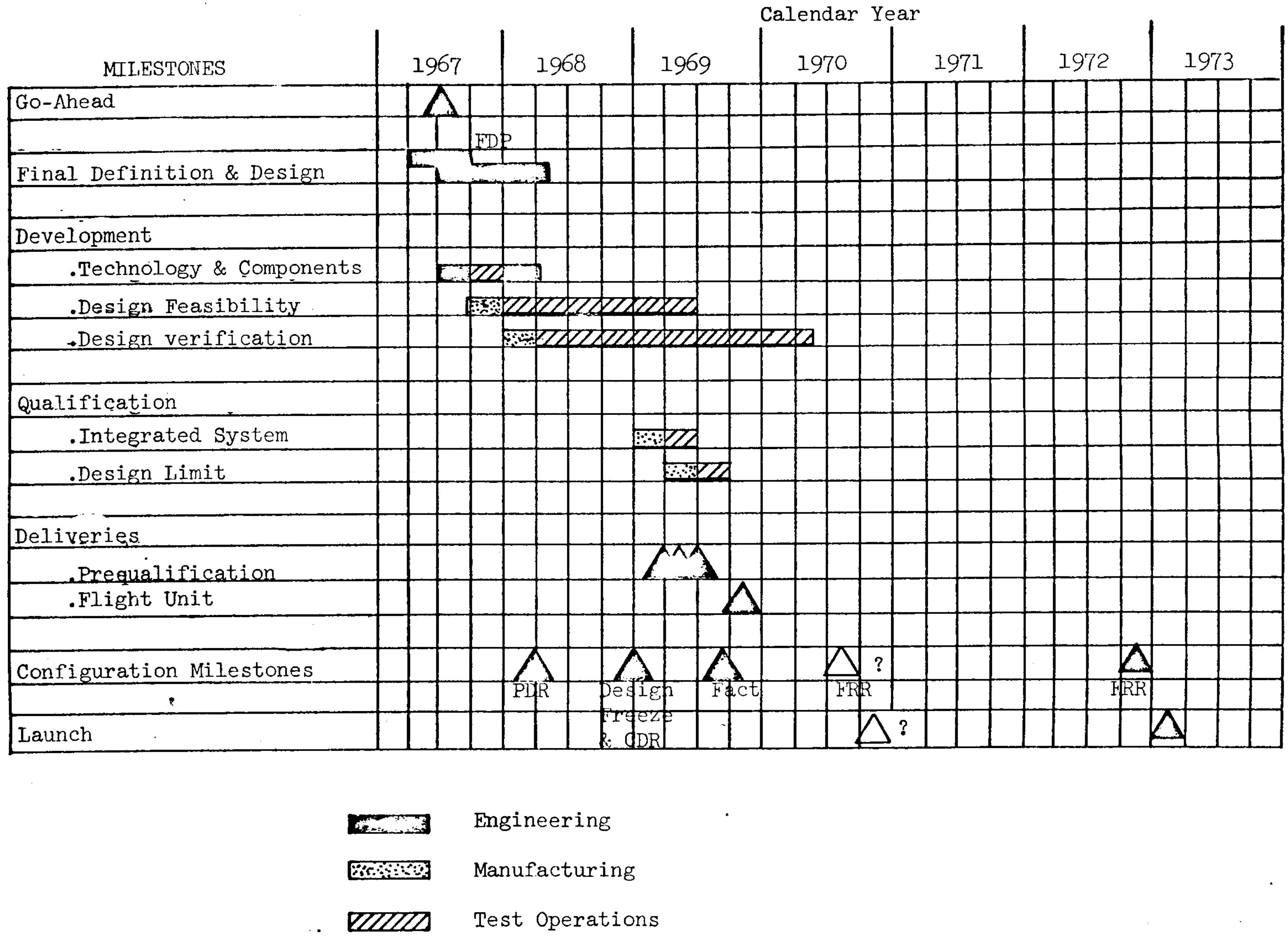


Figure 1.85 **- Battery development schedule.**

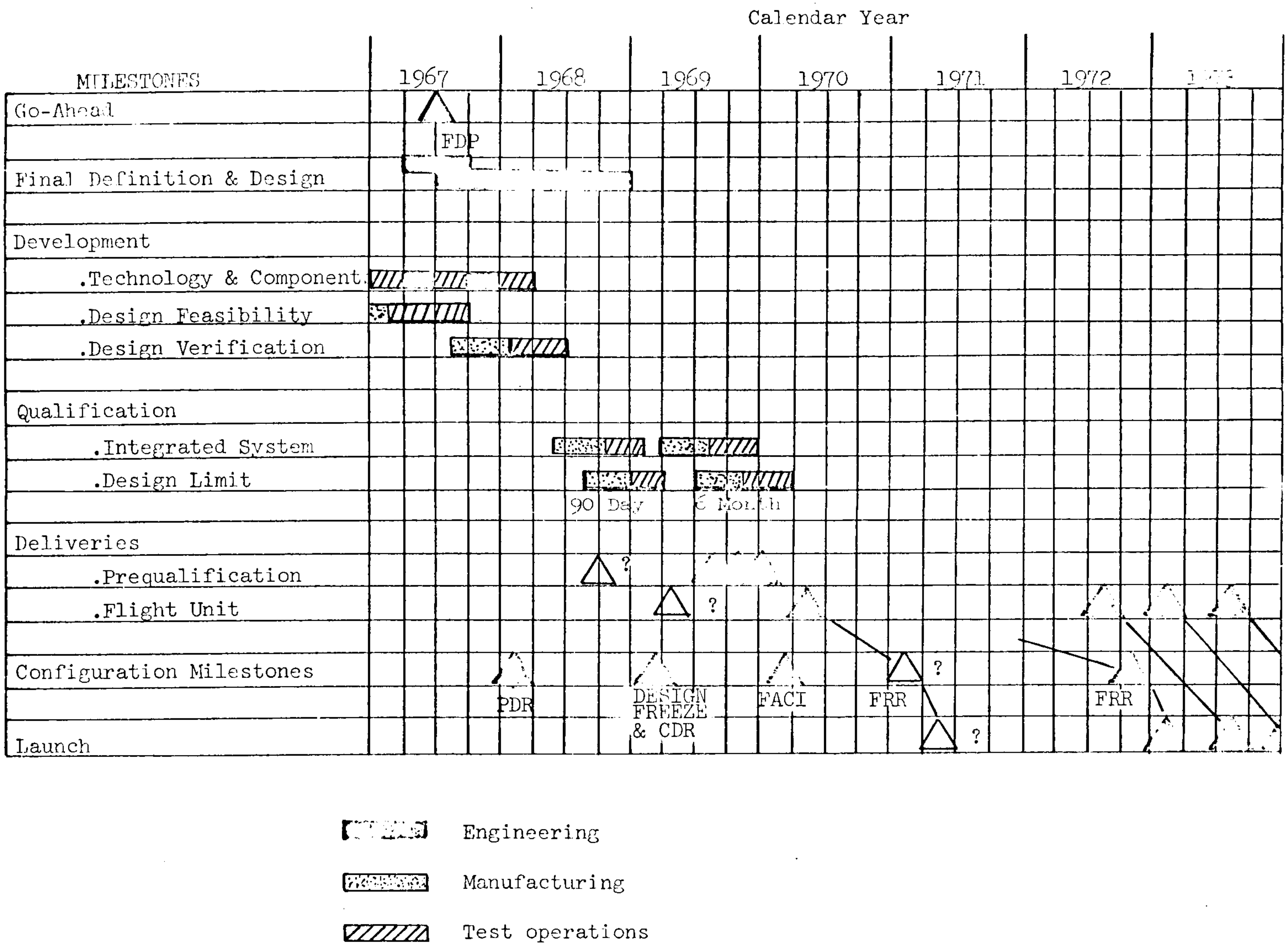


Figure 1.86 **Fuel cell development schedule (nonregenerative).**

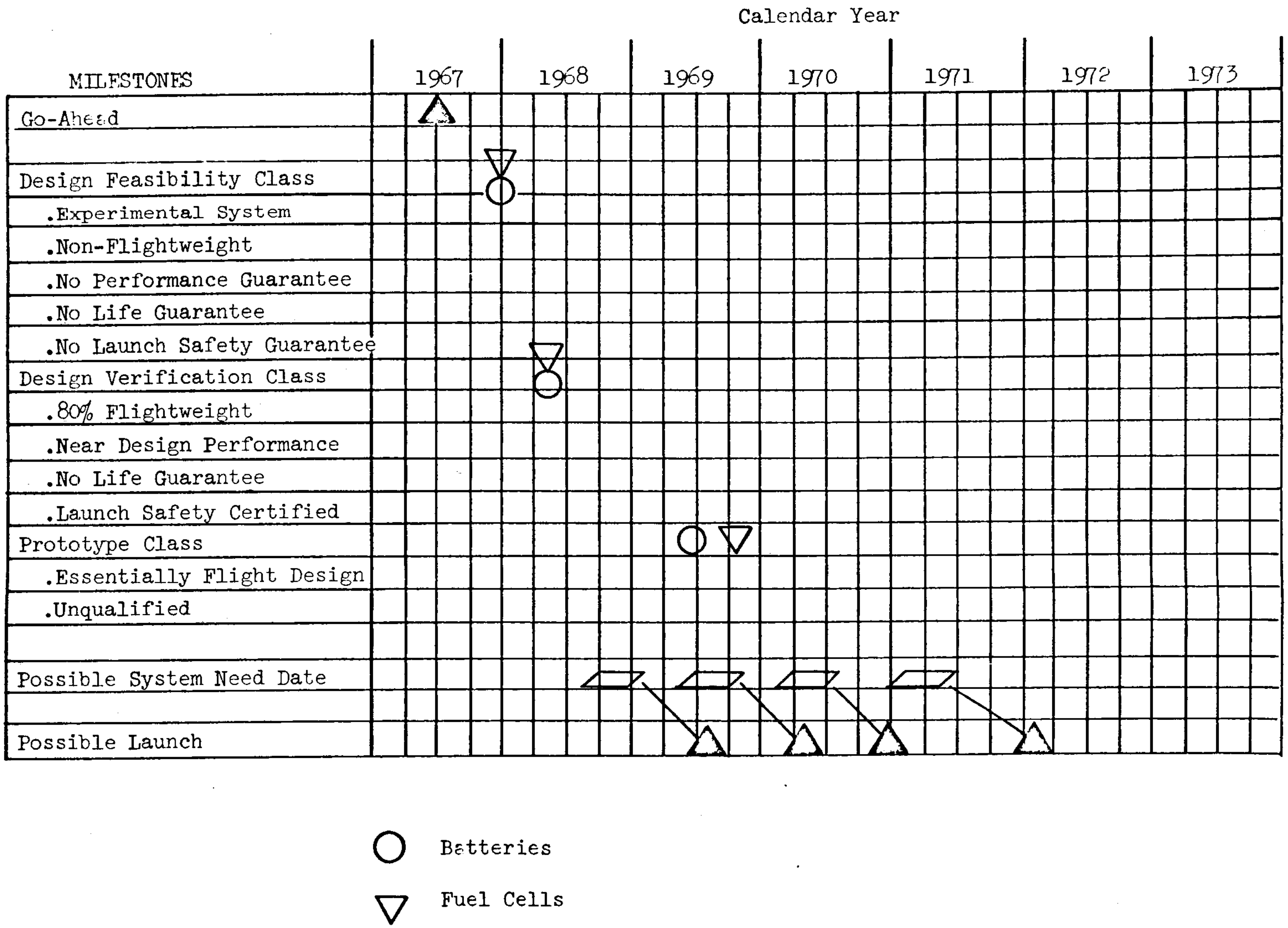


Figure 1.87 - **Secondary system support to space station developmental flights.**

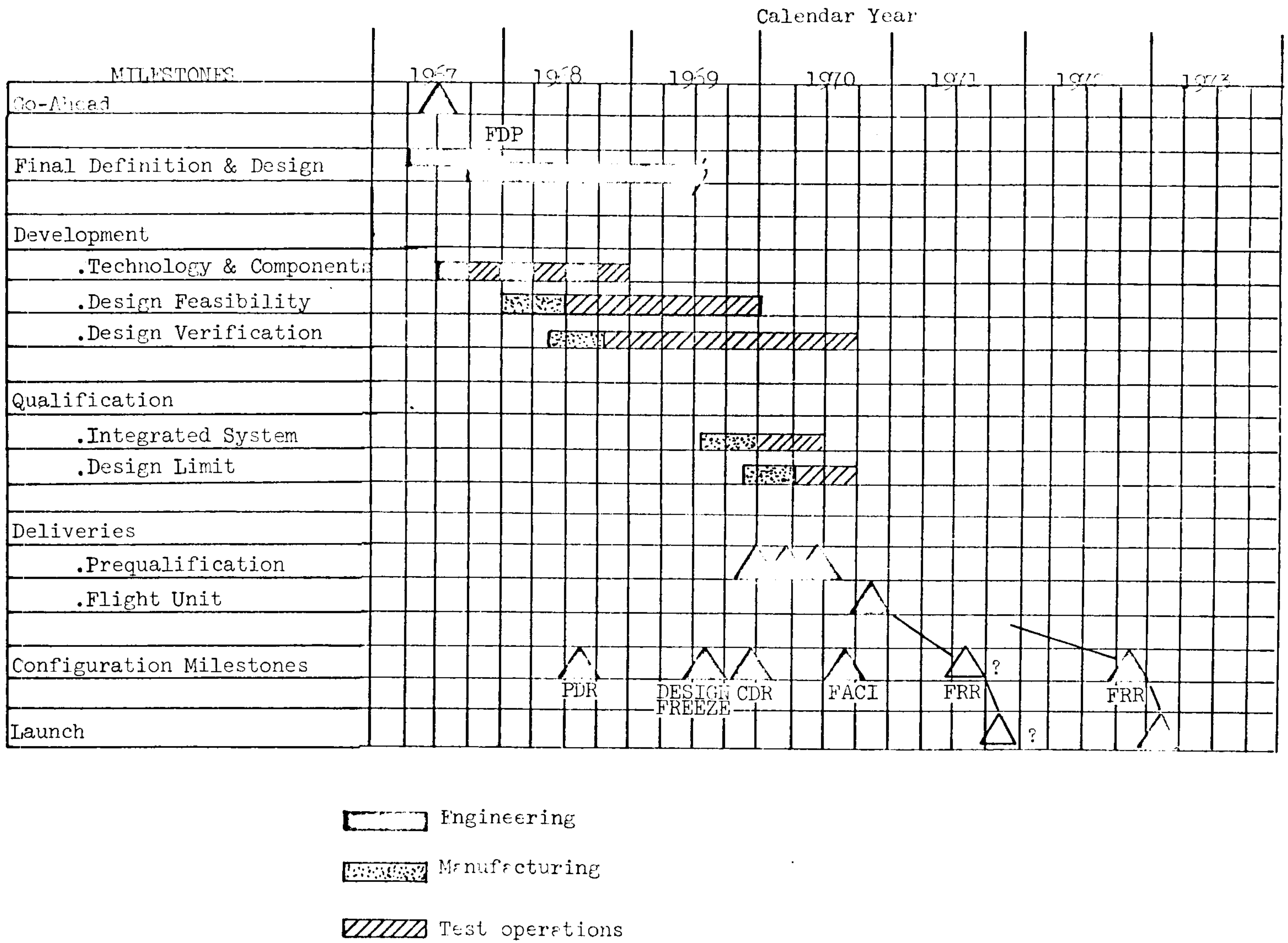


Figure 1.88 - Thermoelectric system development schedule, power sections.

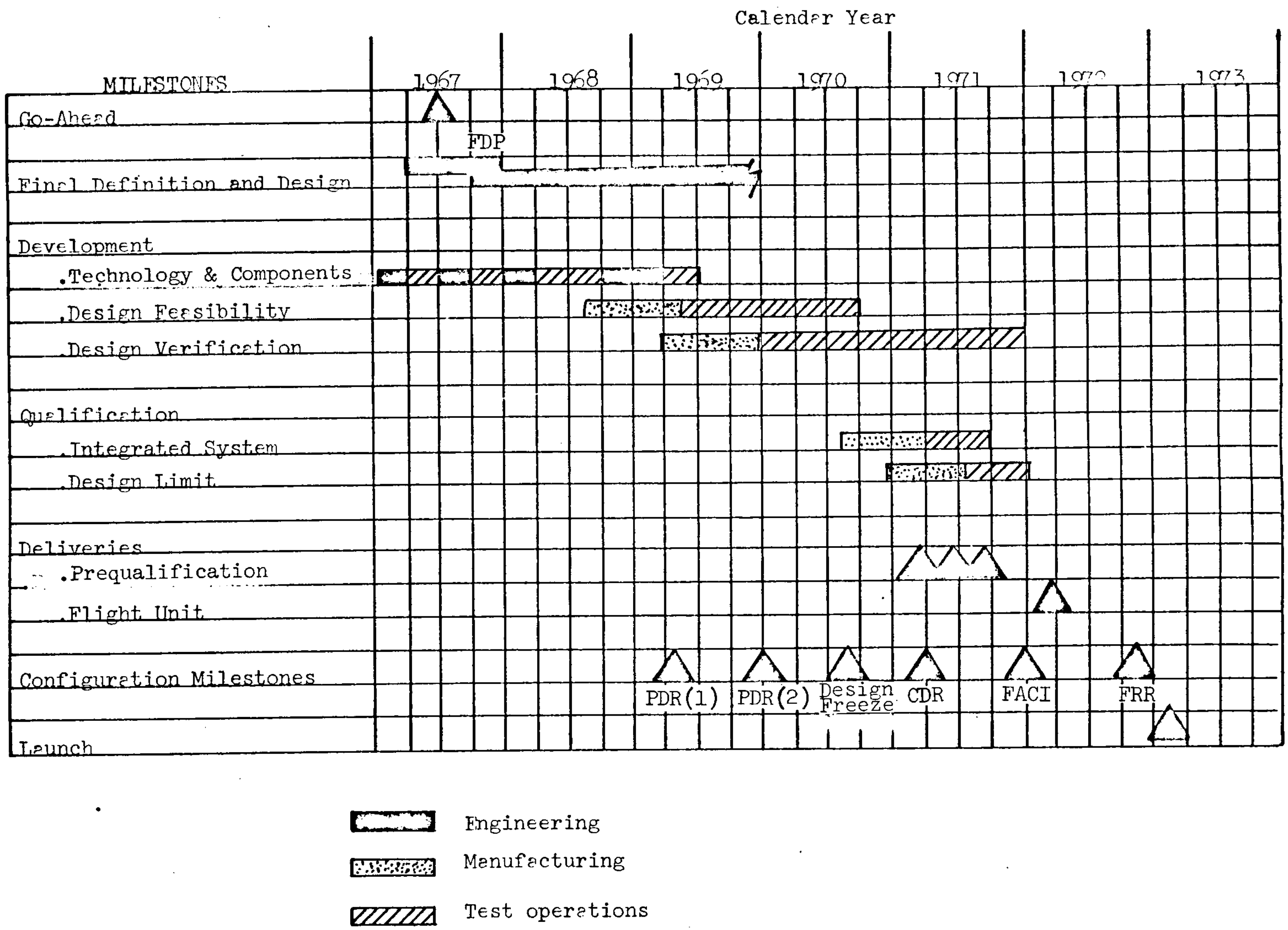


Figure 1.89 - Nuclear reactor development schedule.

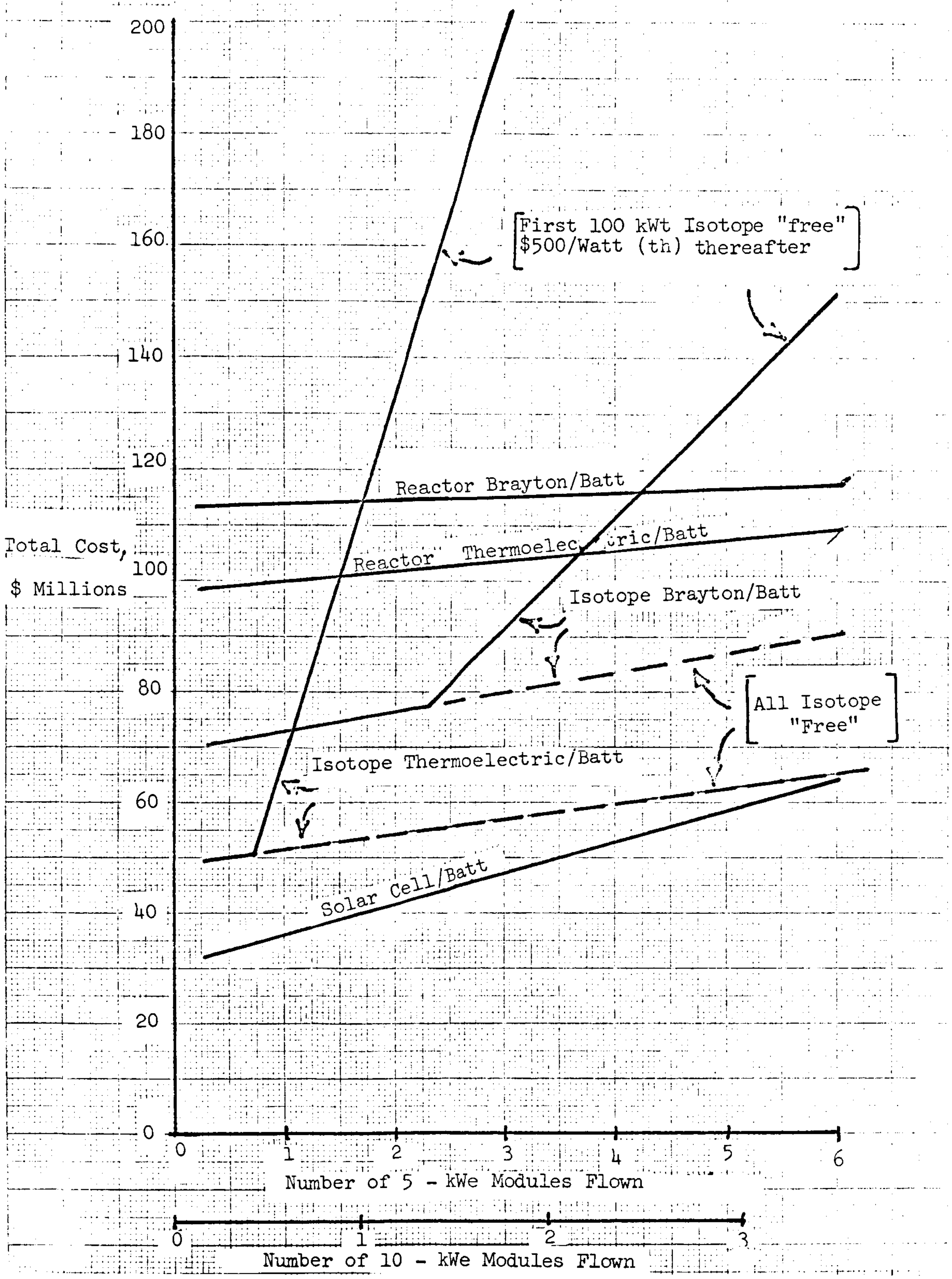


Figure 1.90. - Power system program cost versus number of systems.

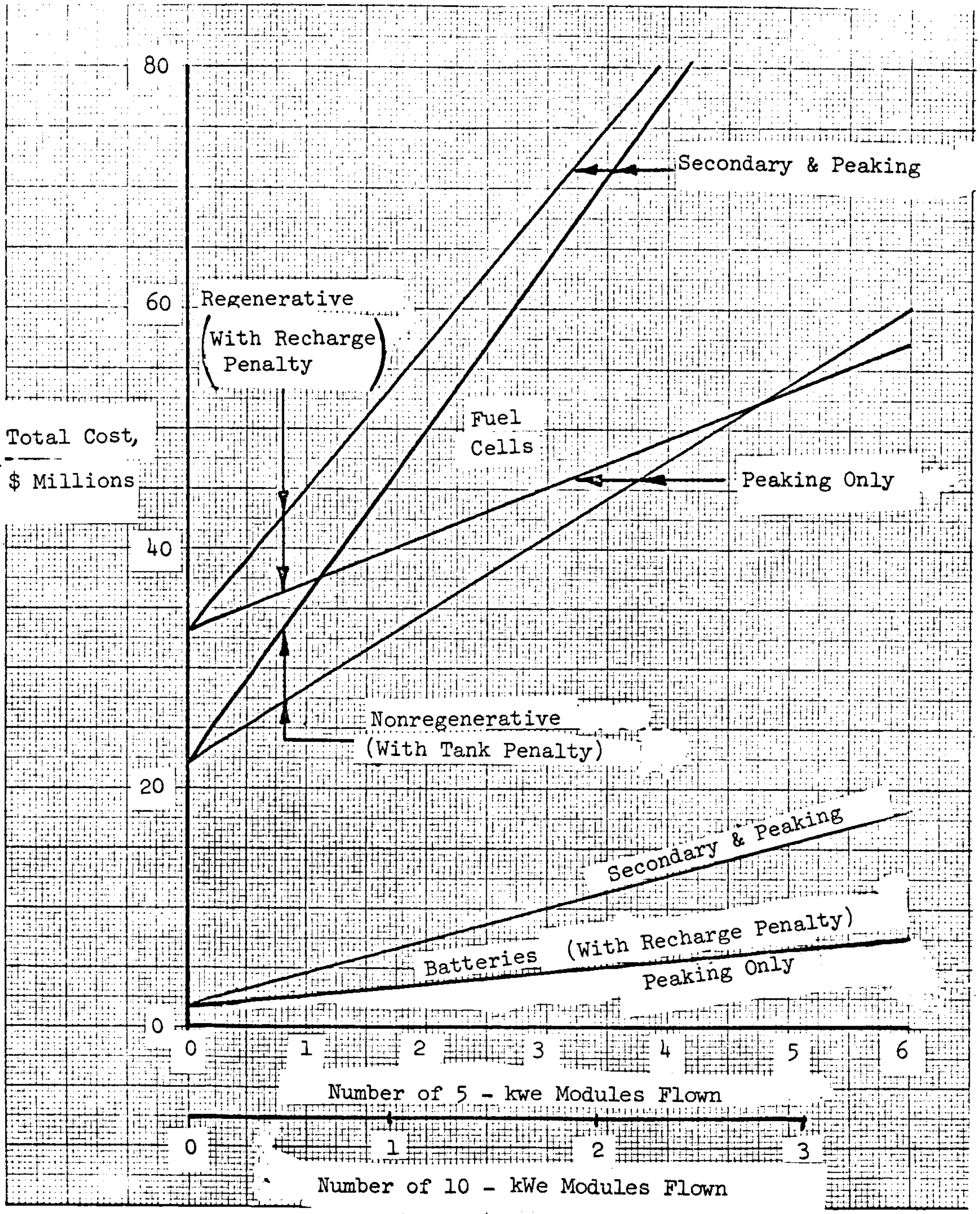


Figure 1.91 - Secondary power systems costs.

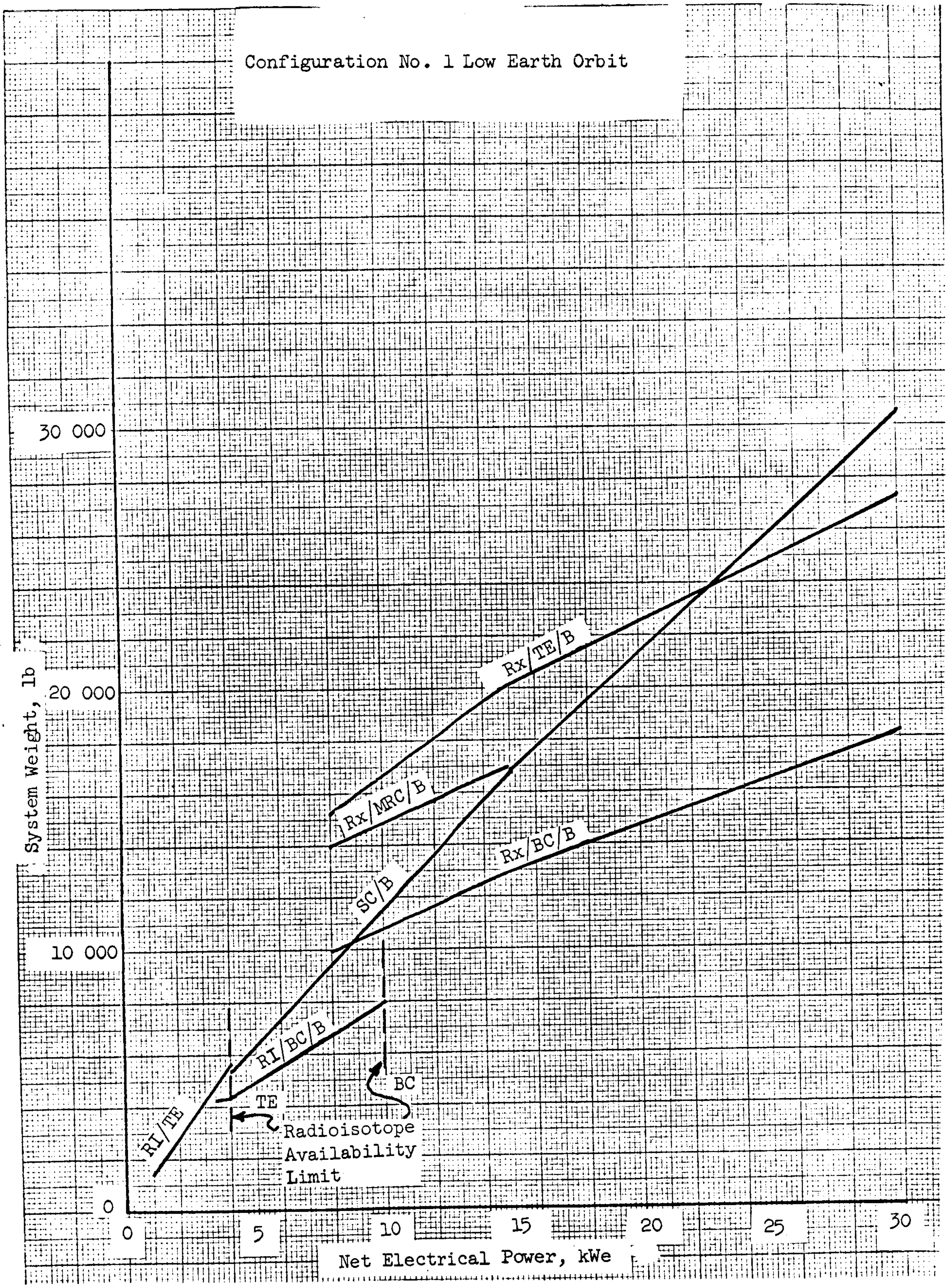


Figure 1.92 - Power system weight versus net electrical power, LEO no. 1

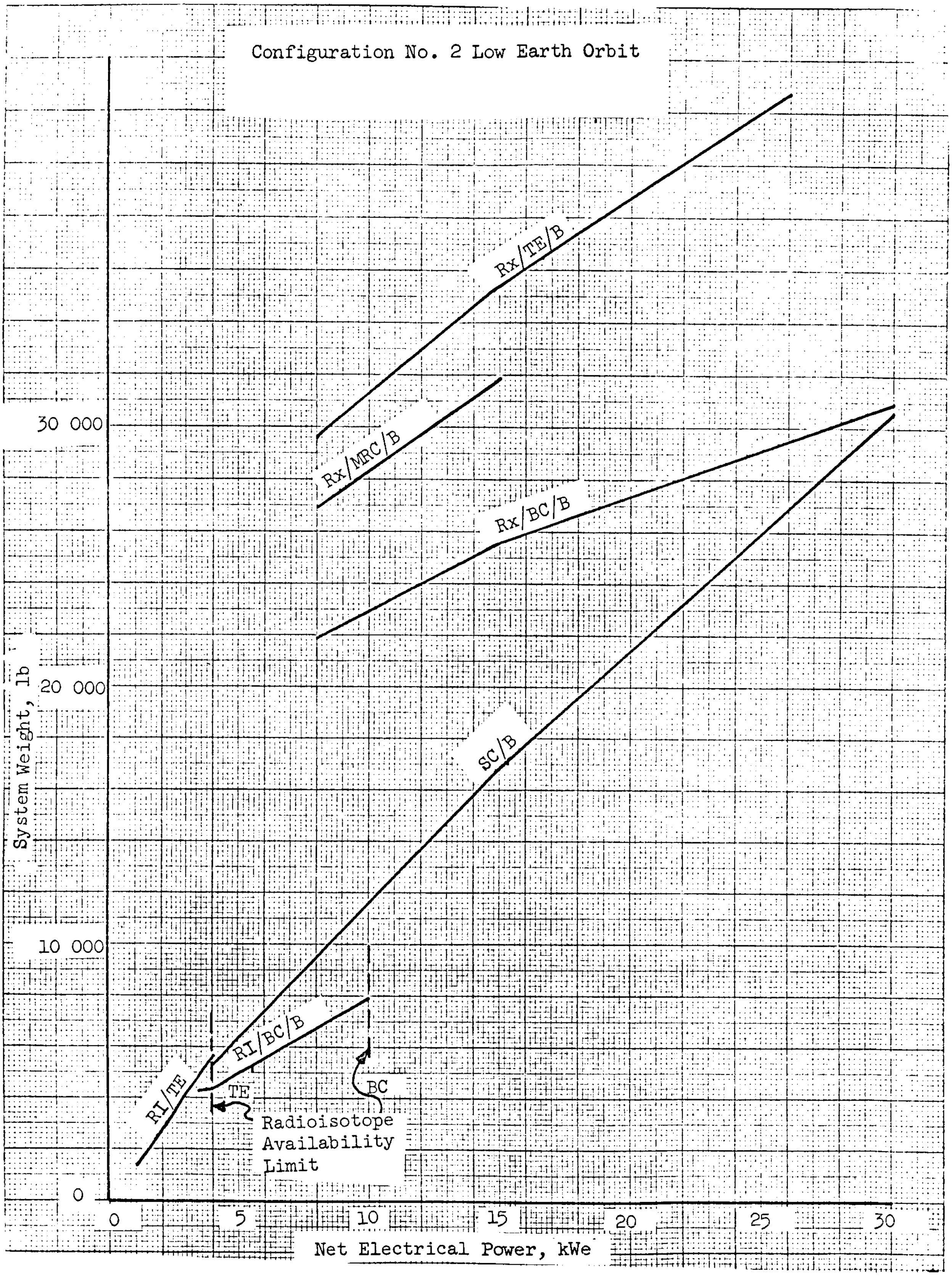


Figure 1.93 - Power system weight versus net electrical power, LEO no. 2

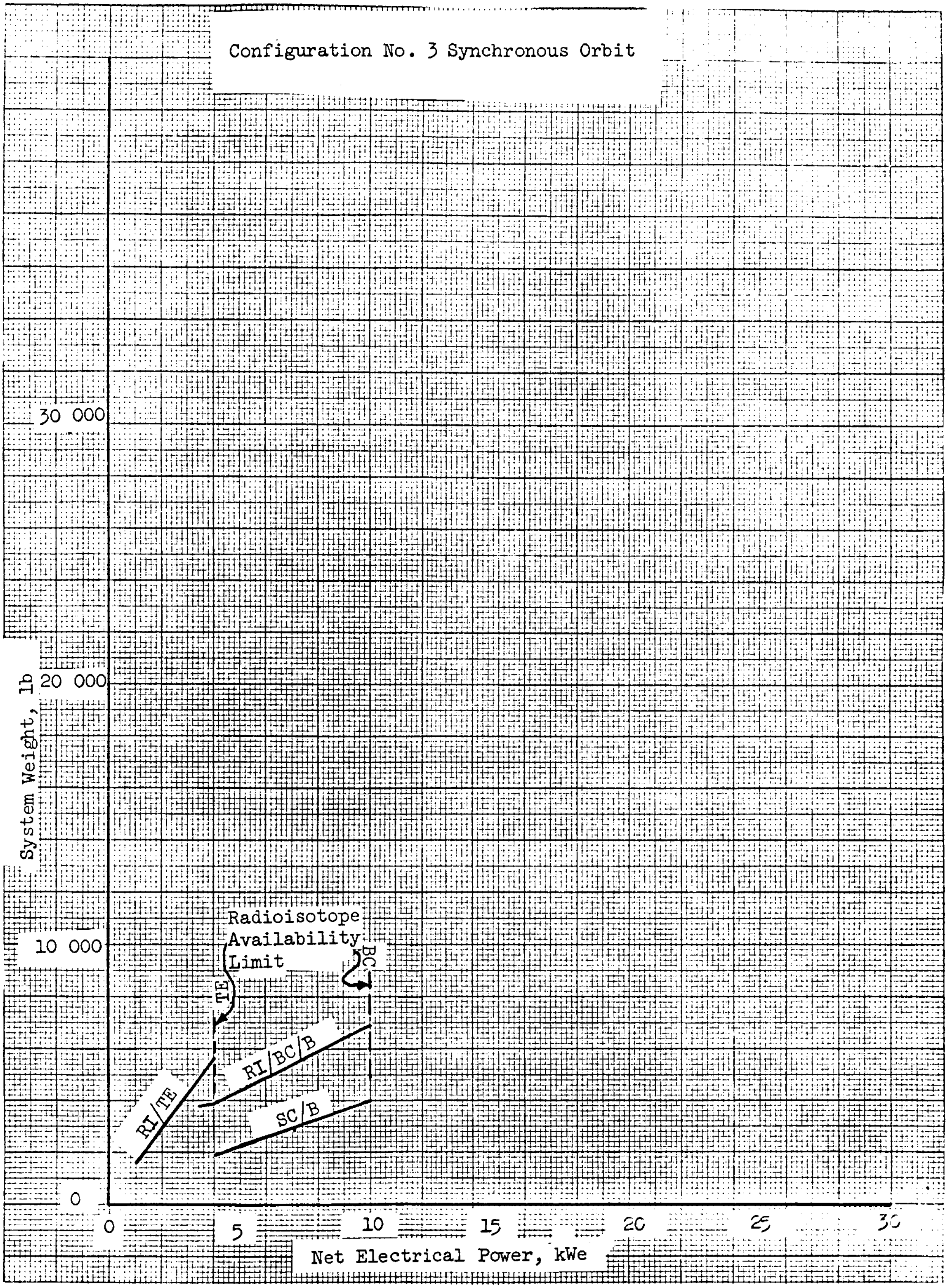


Figure 1.94 - Power system weight versus net electrical power, sync

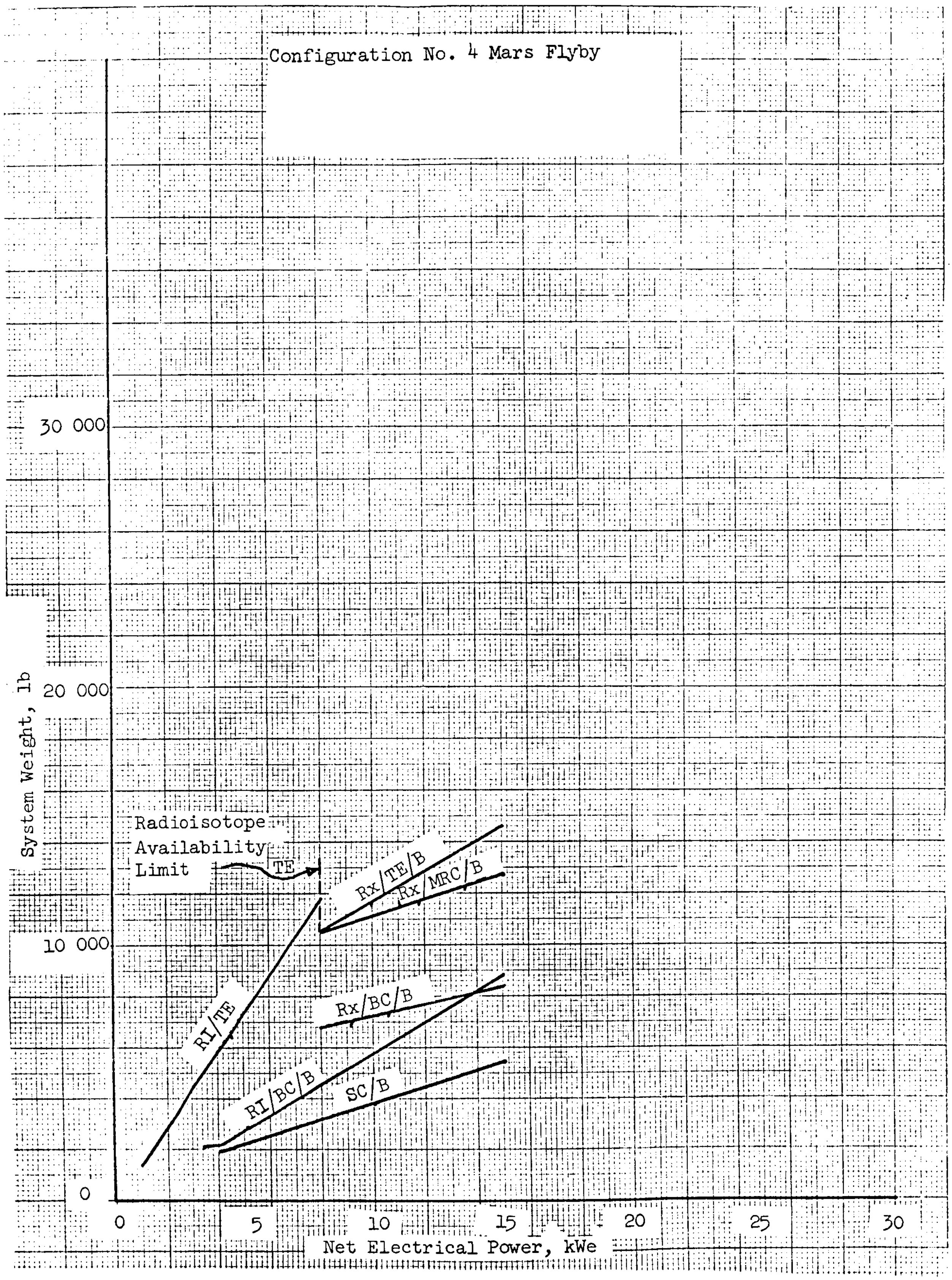


Figure 1.95 .- Power system weight versus net electrical power, Mars

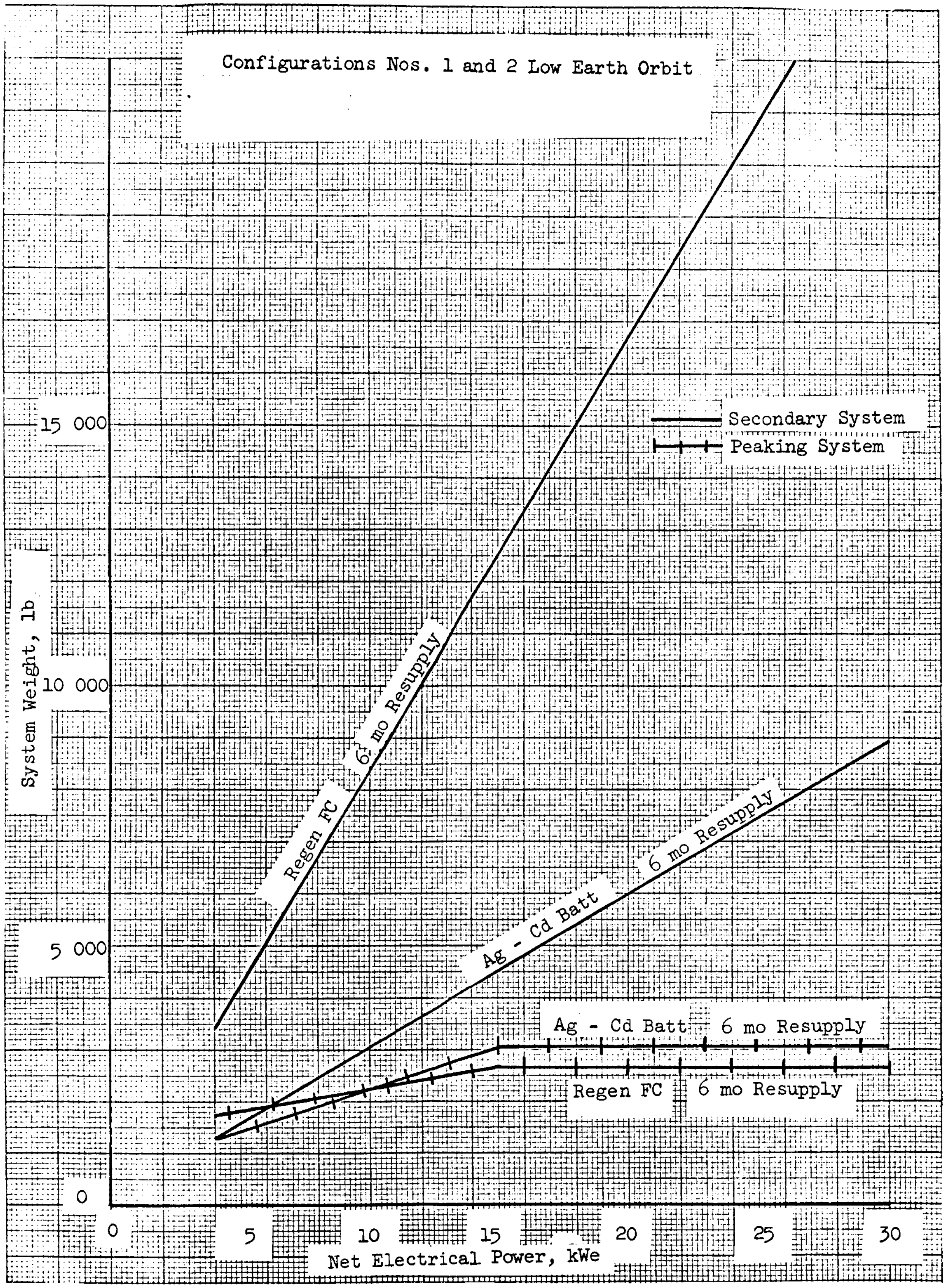


Figure 1.96 - Secondary power system weight versus net electrical power, LEO

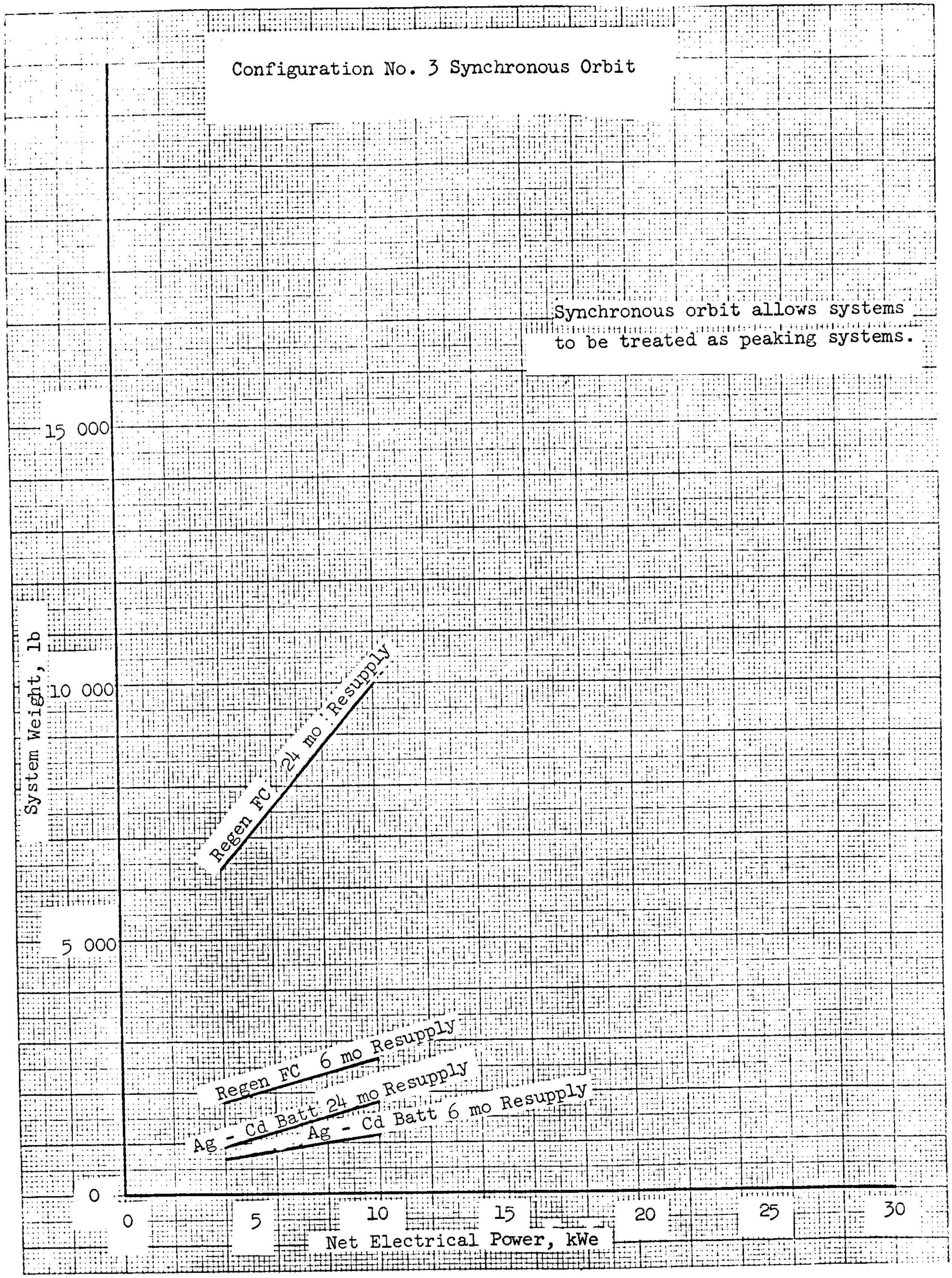


Figure 1.97 - Secondary power system weight versus net electrical power, sync

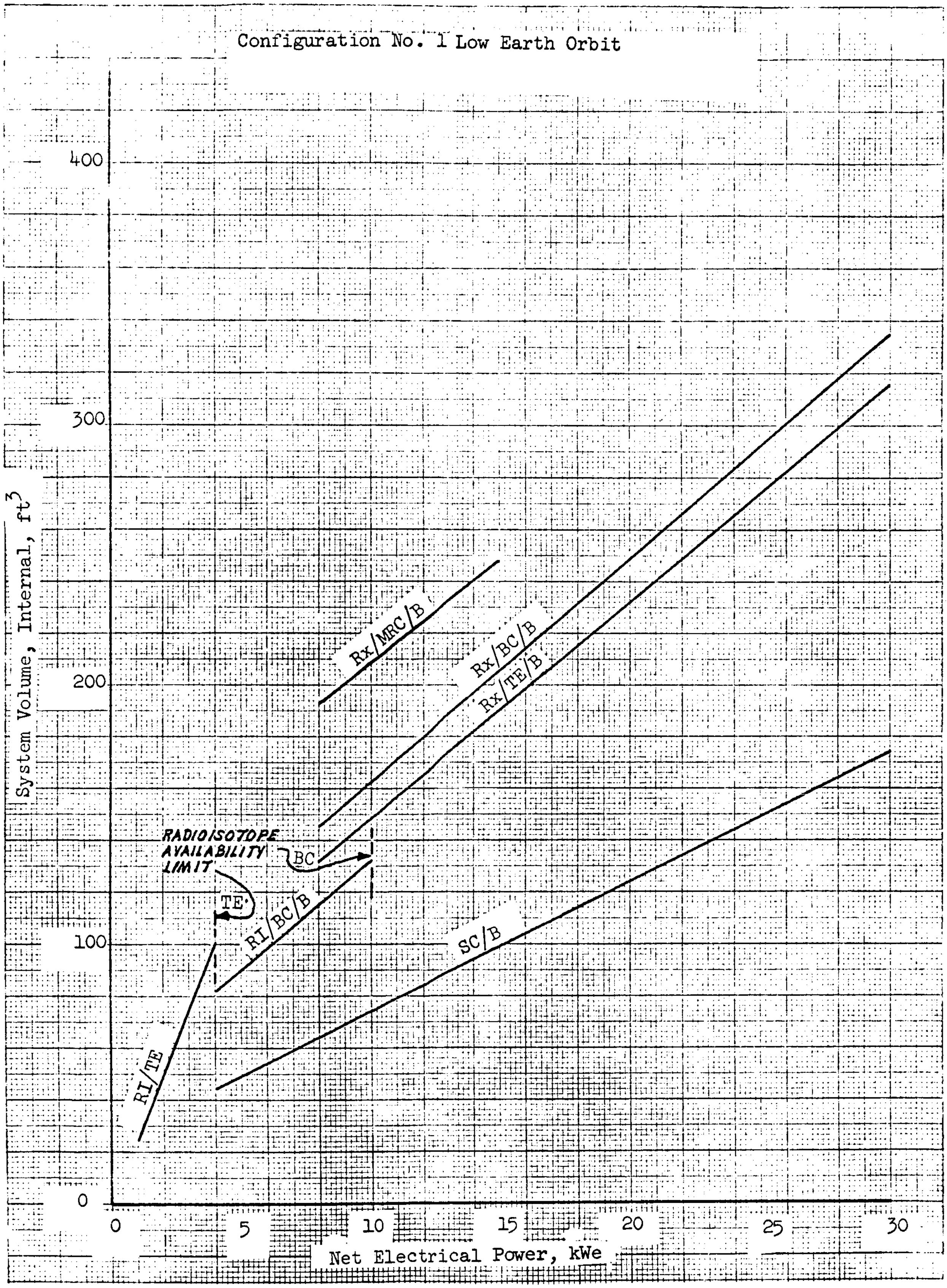


Figure 1.98 Power system volume versus net electrical power, LEO no. 1

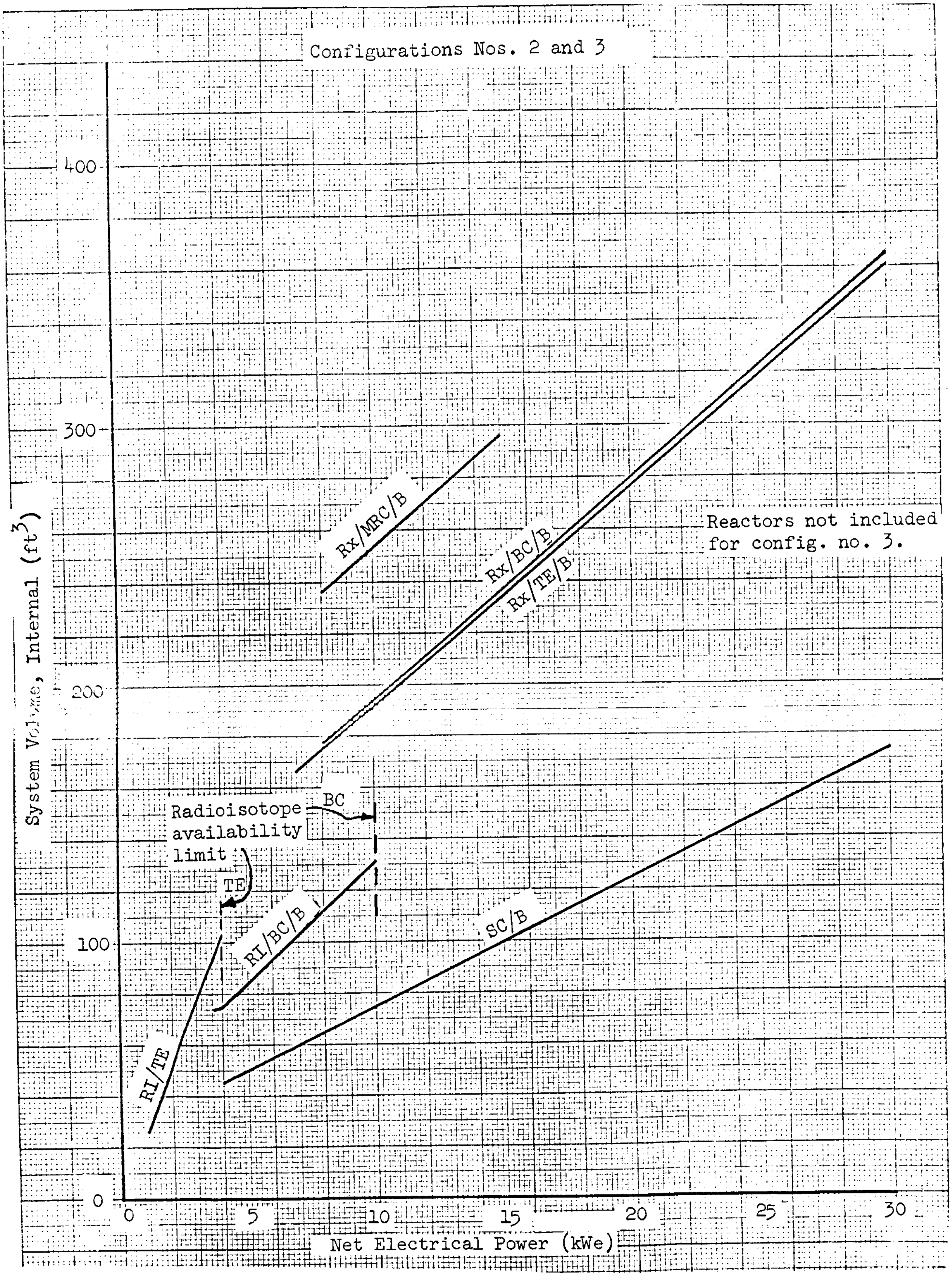


Figure 1.99 - Power system volume versus net electrical power, LEO no. 2 and sync

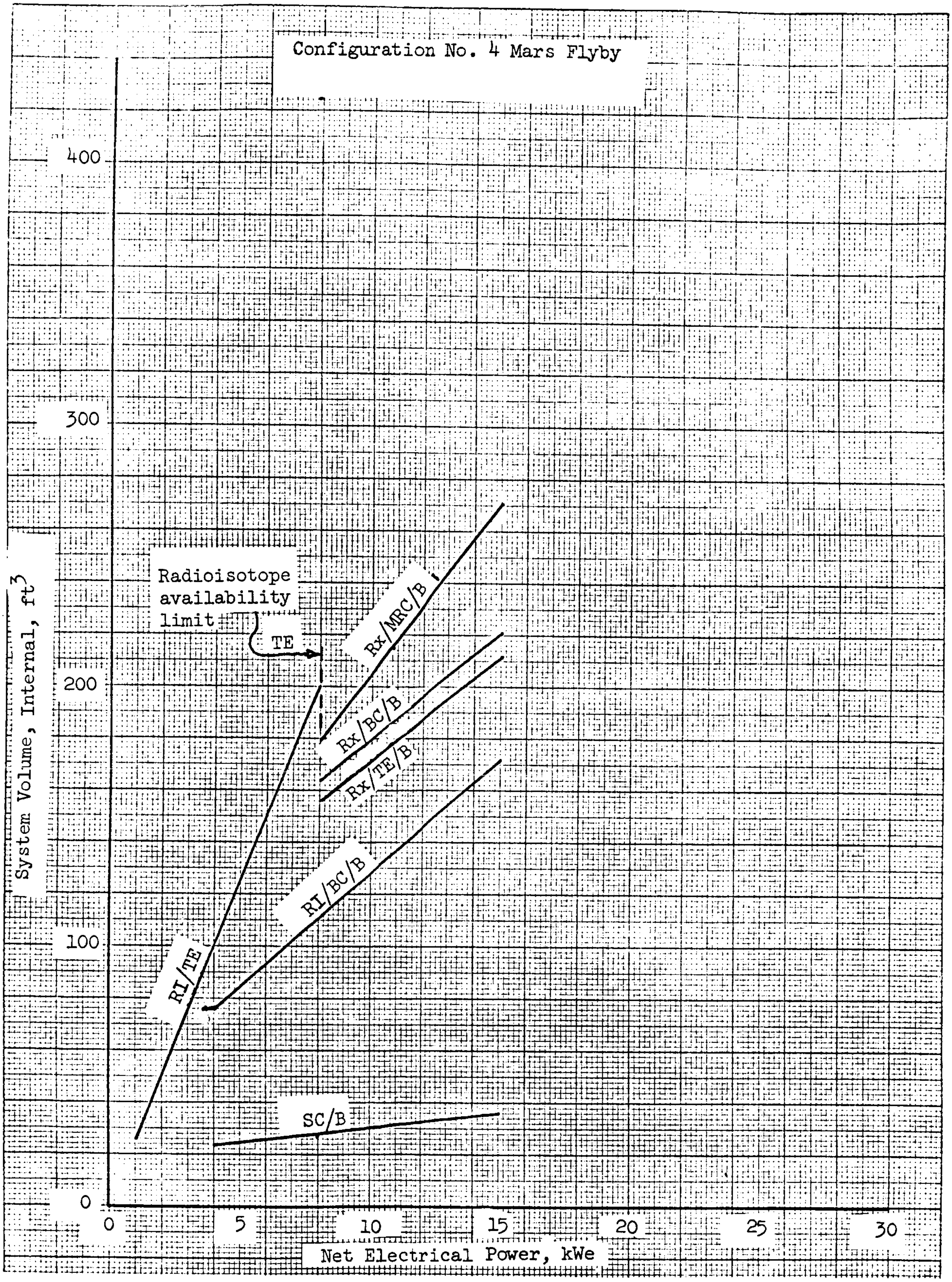


Figure 1.100- Power system volume versus net electrical power, Mars

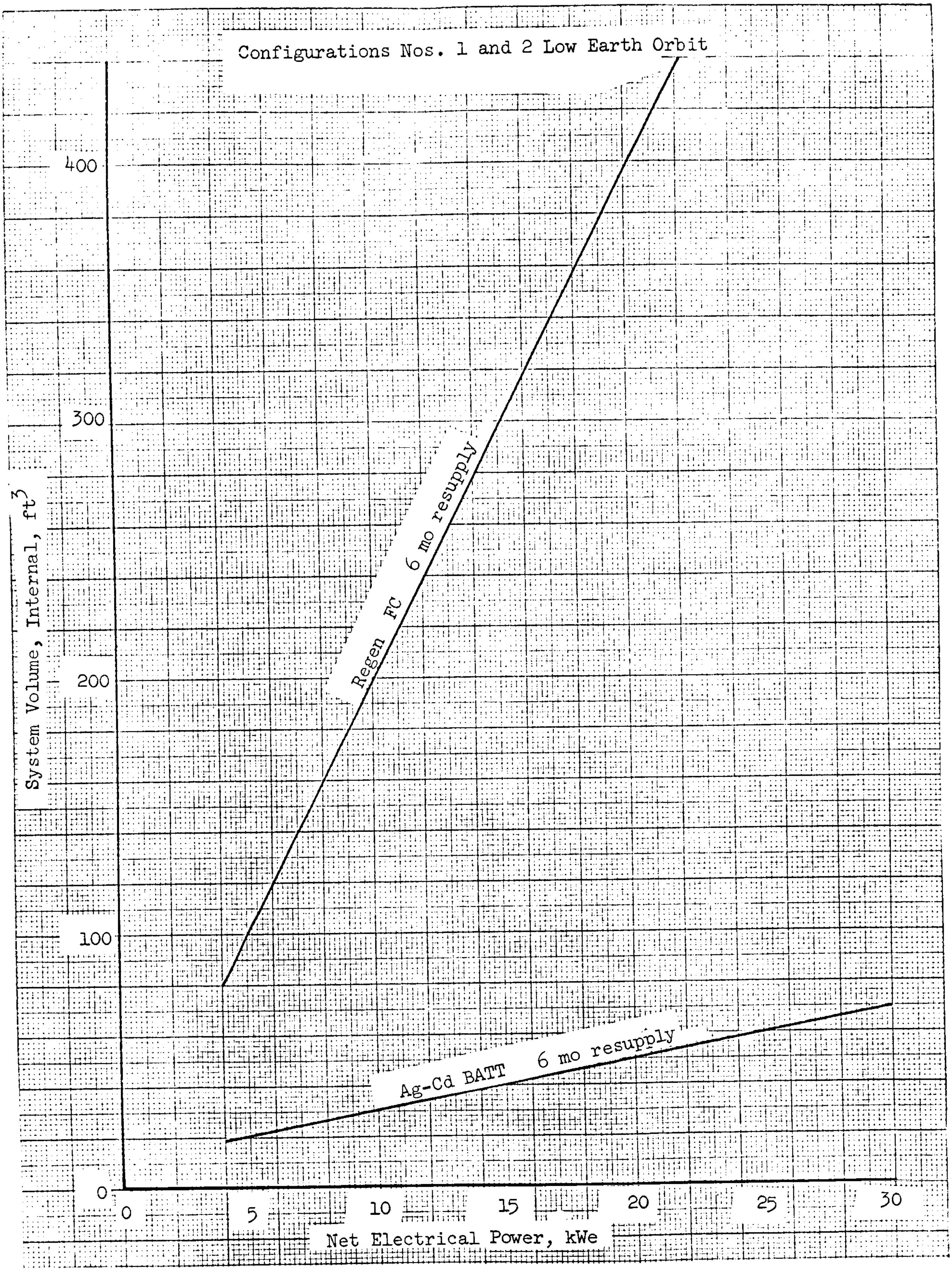


Figure 1.101 - Secondary power system volume versus net electrical power, LEO

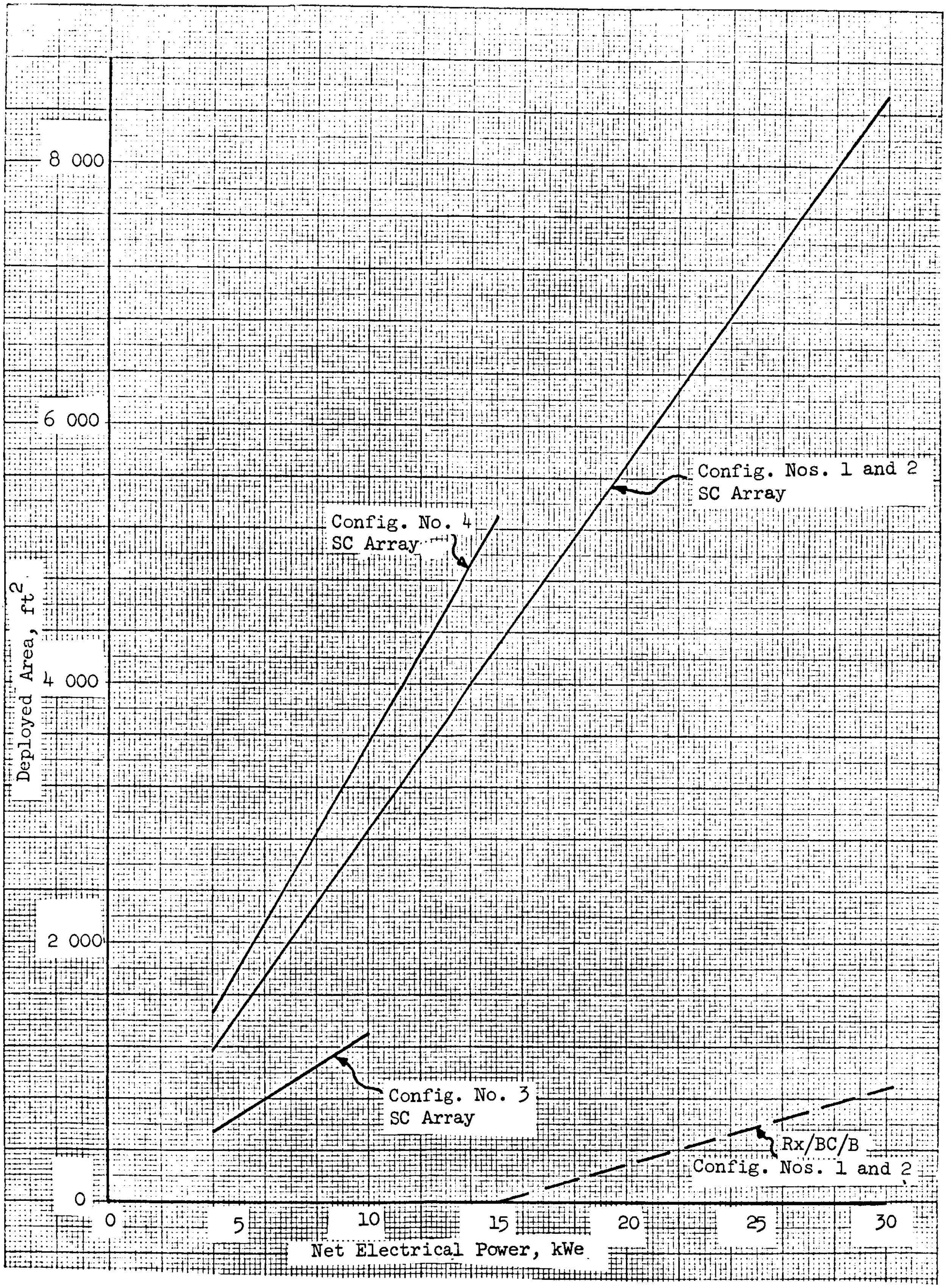


Figure 1.102- Power system deployed area versus net electrical power

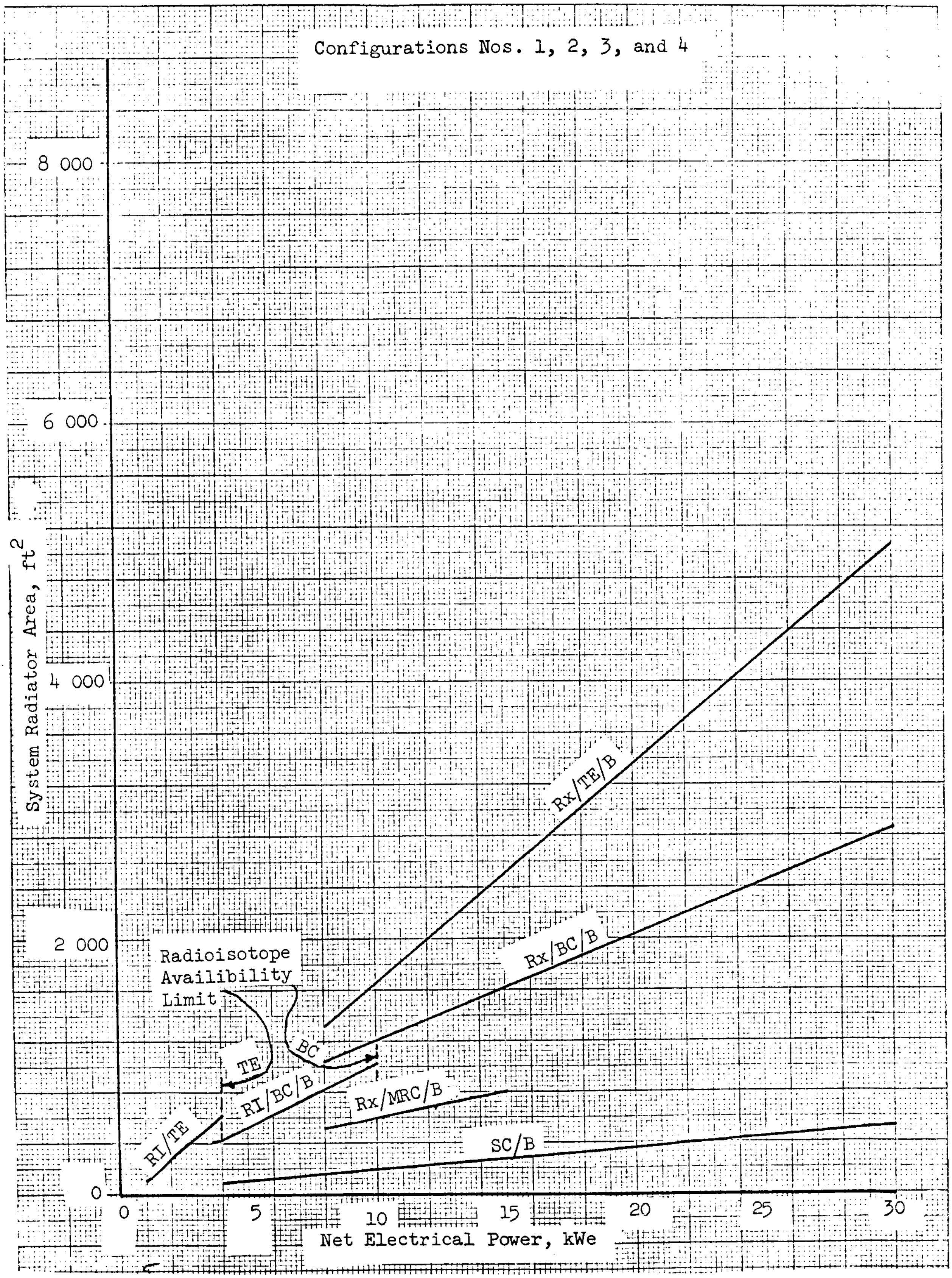


Figure 1.103 - Power system radiator area versus net electrical power

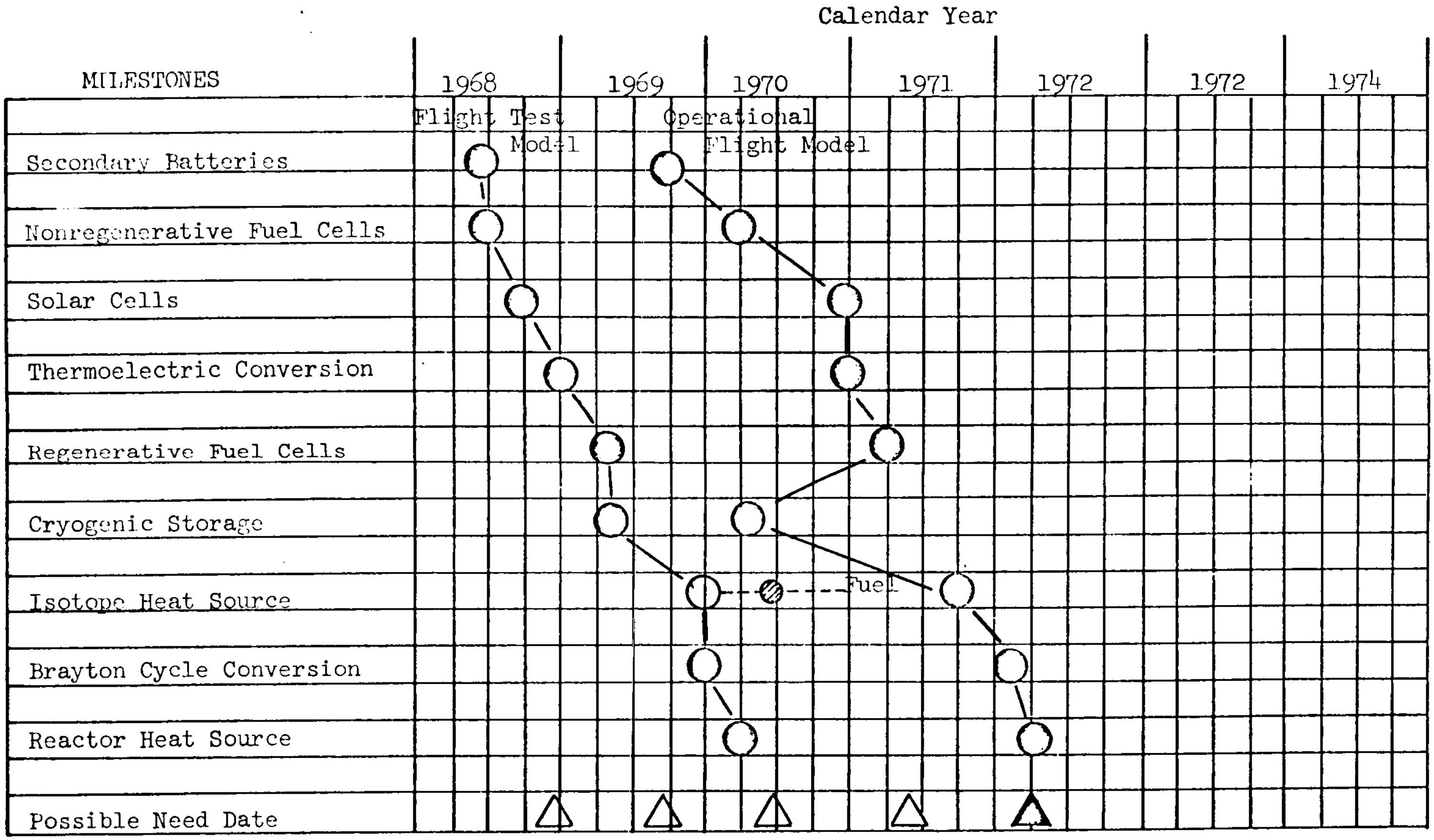


Figure 1.104 - Power system development schedule summary.

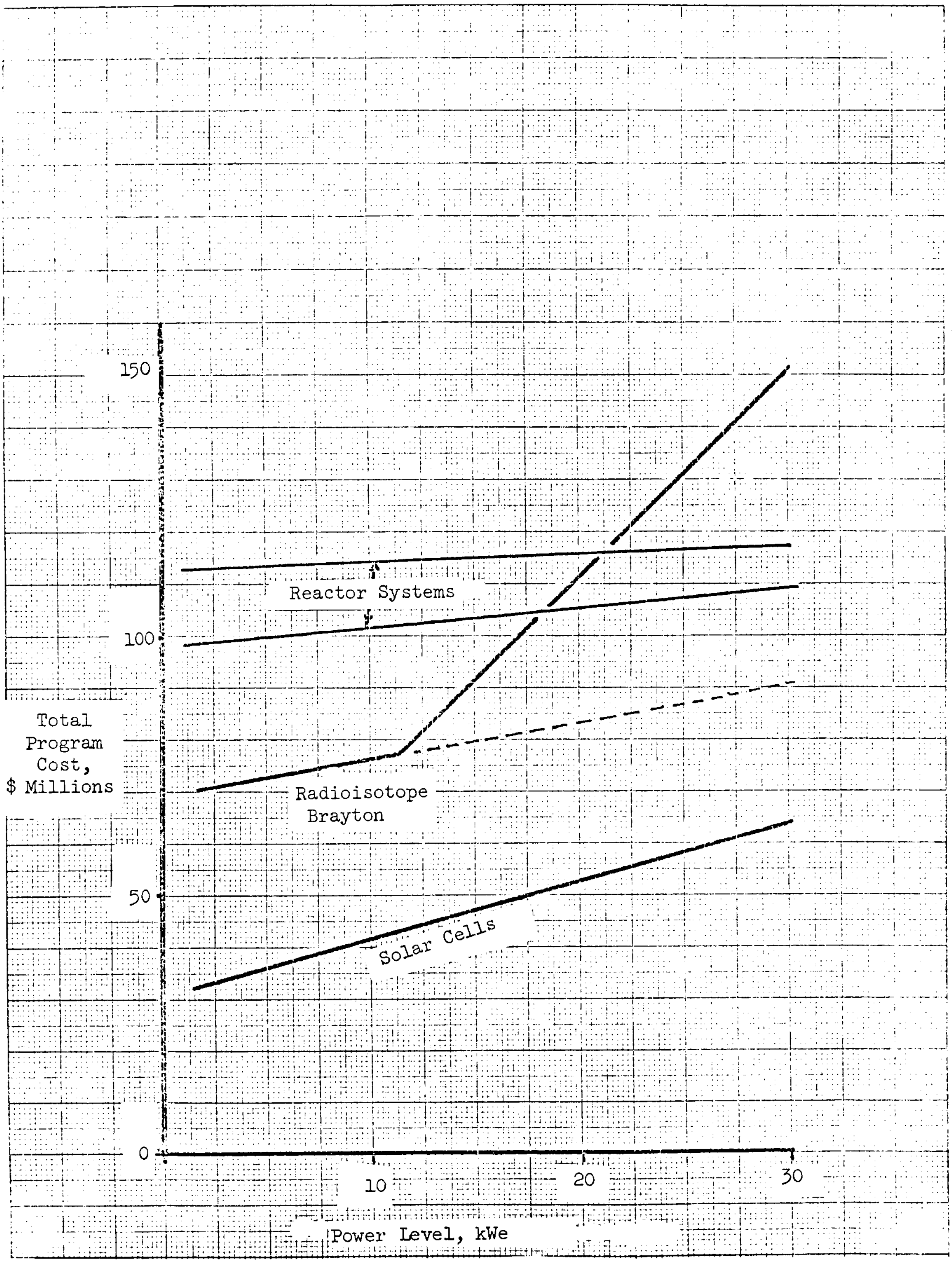


Figure 1.105. - Power system program cost versus power level

1.6 CONCLUSIONS AND RECOMMENDATIONS

For configurations and missions included in this study, solar cells and the Brayton-cycle are the only contenders worthy of further consideration as primary power systems.

Until a configuration/mission, power levels, and power split are better definitized, radioisotope and reactor Brayton-cycle systems should be pursued as well as solar cells.

Further study is required to adequately assess areas such as hybrid systems and emergency or storm cellar systems.

Further study is required to assess a 5-year life requirement on power systems.

APPENDIX A

REFERENCES AND BIBLIOGRAPHY

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APPENDIX B

REPORT BRIEF ON CYCLE LIFE TEST OF RECHARGEABLE BATTERY CELLS

Ref: (a) National Aeronautics and Space Administration Purchase Order Number W11,252B
(b) NASA ltr BRA/VBK/pad of 25 September 1961 w/BUWEPS first end FQ-1:WSK of 2 October 1961 to CO NAD Crane
(c) Preliminary Work Statement for Battery Evaluation Program of 25 August 1961

Test Assignment Brief

In compliance with references (a) and (b), evaluation of secondary spacecraft cells was begun according to the program outline of reference (c). This second annual report covers all of the cycle life test, the third phase of the evaluation program of secondary spacecraft cells, through December 31, 1965. The acceptance tests and general performance tests, the first and second phases of the evaluation program, were reported earlier.

The object of this evaluation program is to gather specific information concerning secondary spacecraft cells. Information concerning the performance characteristics and limitations, including cycle life under various electrical and environmental conditions, will be of interest to power systems designers and users. Cell weaknesses, including causes of failure of present designs, will be of interest to suppliers as a guide to product improvement.

The life cycling test was begun in December 1963.

Cells Included in Test

Only cells which had passed the acceptance tests were used in the evaluation program.

The cycle life test program began with sealed, nickel-cadmium cells of the types listed below:

Manufacturer	Rated Capacity	Number of Cells
General Electric Company	3.0 A-h	120
	12 A-h	60
Gould-National Batteries, Inc.	3.5 A-h	120
	20 A-h	60
Gulton Industries, Inc.	6.0 A-h	120
	20 A-h	60
Sonotone Corporation	5.0 A-h	120

Description of Cycle Test

Cells were arranged into packs of 5 or 10 cells. Each pack cycled with a given set of test parameters until more than half of the cells had failed, at which time the pack was considered to have failed.

Cycling test parameters included ambient temperature, charge voltage limit, percent depth of discharge, percent of recharge, and orbit period, as follows:

a. 50° C, 1.41 volts per cell limit, 15 or 25 percent depth of discharge, 160 percent recharge, and 1.5 or 3-hour orbit. All packs begun at 50° C were subsequently changed to 40° C, 1.45 volts per cell limit, with the remaining parameters unchanged.

b. 25° C, 1.49 volts per cell limit, 25 or 40 percent depth of discharge, 125 percent recharge, and 1.5 or 3-hour orbit.

c. 0° C, 1.55 volts per cell limit, 15 or 25 percent depth of discharge, 115 percent recharge, and 1.5 or 3-hour orbit.

The ampere-hour capacity of each pack was measured at approximately 88-day intervals.

Failed cells were removed from the pack at the time of failure and subjected to failure analysis.

Test Results

A total of 51 of the original 84 packs have failed. The remaining 33 packs have completed from 516.6 to 738.5 days (a maximum of 11 816 1.5-hour cycles) of continuous cycling as of December 31, 1965. The status of each pack is given in table B-1 and figs. B-1(a) through B-1(g).

It was found that 50° C was, in general, an unsatisfactory ambient temperature, for the specified currents and orbit periods, due to inefficient charge acceptance and accelerated separator deterioration.

There have been 281 cell failures as of December 31, 1965. Table B-2 shows the distribution according to test parameters and cell types.

A high percentage of cell failures was premature due to defects in manufacture or design.

Ampere-hour capacities changed with time in a manner which was strongly dependent on test parameters and cell type.

For those packs which had completed 264 or more days of cycling, average initial capacities and average capacities after 264 days of cycling are listed below in terms of percent of rated capacity.

	0° C	25° C	40° C*
Average initial capacity (percent of rated capacity)	104.0	117.9	63.8
Average capacity after 264 days (percent of rated capacity)	96.2	65.5	46.7
(Percent of initial capacity)	92.6	55.4	79.9

*The measurement of initial capacity at 40° C was made after the cells had been cycled at 50° C.

Certain packs appear to have exhibited the "memory effect."

Cells Added To The Cycle Life Test Program

Cells using conventional charge control methods.-

Nickel-cadmium types:

a. Gulton 4.0 A-h (commercial), six 5-cell packs, 1.5-hour orbit period: These packs have completed from 7638 to 8136 cycles with two cell failures.

b. Gulton 5.0 A-h (NIMBUS), six 5-cell packs, 1.5-hour orbit period: These packs have completed from 3087 to 3795 cycles with one cell failure.

c. Gulton 5.6 A-h (Neoprene seal), six 5-cell packs, 1.5-hour orbit period: These packs have completed from 208 to 453 cycles with no cell failure.

d. Gulton 6.0 A-h, one 5-cell pack, 24-hour orbit period: This pack failed after 545 cycles.

e. Gulton 6.0 A-h (improved), three 5-cell packs, 1.5-hour orbit period: These packs have completed from 4697 to 4793 cycles with one cell failure.

f. Gulton 12 A-h (OGO), six 5-cell packs, 1.5-hour orbit period: These packs have completed from 4869 to 5739 cycles with eight cell failures.

g. Gulton 50 A-h, two 5-cell packs, 1.5-hour orbit period: One pack failed after 3227 cycles. The other pack failed after 1873 cycles.

h. General Electric 5.0 A-h (NIMBUS), six 5-cell packs, 1.5-hour orbit period: These packs have completed from 3142 to 3874 cycles with no cell failures.

i. General Electric 12 A-h, one 5-cell pack, 24-hour orbit period: This pack failed after 349 cycles.

j. Sonotone 3.0 A-h (triple seal), six 5-cell packs, 1.5-hour orbit period: These packs have completed from 2576 to 2890 cycles with one cell failure.

Silver-zinc types:

a. Delco-Remy 25 A-h, two 5-cell packs, 24-hour orbit period: One pack failed after 80 cycles. The other one failed after 32 cycles.

b. Delco-Remy 25 A-h, two 5-cell packs, 3-hour orbit period: Four of the five cells were still functioning after 120 cycles, at which time the pack was removed from cycling. The other pack failed after 352 cycles.

c. Delco-Remy 40 A-h, one 5-cell pack, 24-hour orbit period: Three of the five cells were still functioning after 139 cycles, at which time the pack was removed from cycling.

d. Yardney 12 A-h, one 10-cell pack, 24-hour orbit period: This pack failed after 57 cycles.

Silver-cadmium types:

a. Yardney 5.0 A-h (C-3 separator), three 5-cell packs, 24-hour orbit period: These packs have completed from 61 to 104 cycles with two pack failures.

b. Yardney 5.0 A-h (radiated separator), two 5-cell packs, 24-hour orbit period: These packs have completed from 34 to 63 cycles with one pack failure.

c. Yardney 5.0 A-h (Pellon control separator), one 5-cell pack, 24-hour orbit period: This pack has completed 63 cycles with no cell failures.

d. Yardney 12 A-h, two 10-cell packs, 24-hour orbit period: These packs failed after 210 cycles and 166 cycles, respectively.

<u>Cells using charge control methods and devices</u>.-

Auxiliary electrode:

a. Gulton 6.0 A-h (nickel-cadmium), six 5-cell packs, 1.5-hour orbit period: These packs have completed from 2785 to 4855 cycles with three cell failures (none due to the auxiliary electrode).

b. General Electric 12 A-h (nickel-cadmium), four 5-cell packs, 1.5-hour orbit period: These packs have completed from 665 to 1698 cycles before two packs were discontinued due to low capacity of the negative plates.

Stabistor:

a. Sonotone 5.0 A-h (nickel-cadmium), eight 5-cell packs, 1.5-hour orbit period: These packs have completed from 747 to 2133 cycles, with four cell failures due to high internal pressure caused by high cell voltage.

Coulometer:

a. Sonotone 5.0 A-h (nickel-cadmium), one 5-cell pack, 1.5-hour orbit period: This pack has completed 6597 cycles with no cell failures.

b. Gulton 3.6 A-h (nickel-cadmium), one 10-cell pack, 1.5-hour orbit period: This pack has completed 805 cycles, with no cell failures.

Sherfey upside-down cycling:

Gulton 3.6 A-h (nickel-cadmium), one 10-cell pack, 1.5-hour orbit period: This pack has completed 1871 cycles, with no cell failures.

Two step charge regulator:

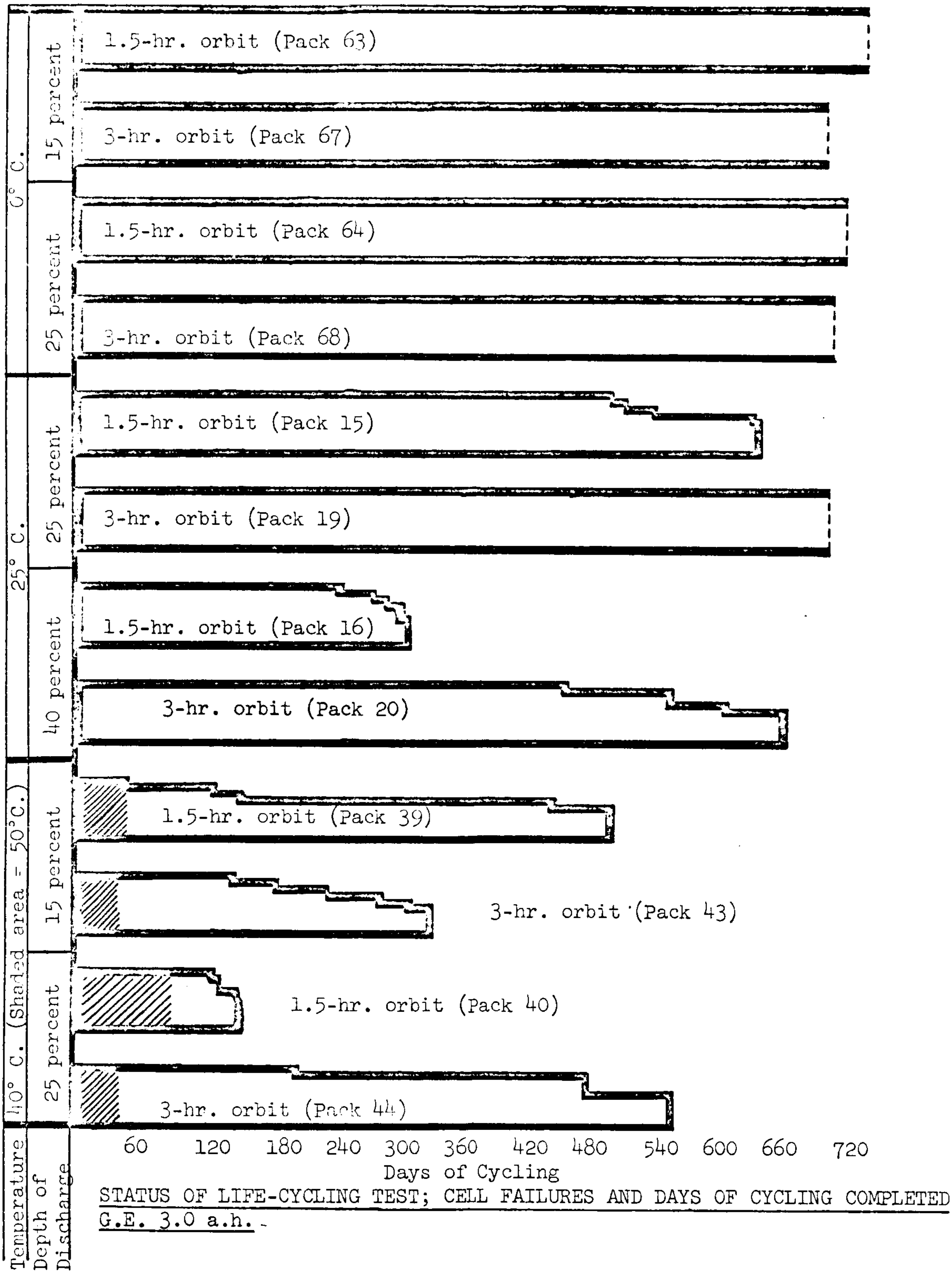
Delco-Remy 25 A-h (silver-zinc), one 10-cell pack, 24-hour orbit period: This pack has completed 19 cycles with no cell failures.

TABLE B-1. [illegible] DAYS OF CYCLING [illegible] AS OF DECEMBER 31, 196[illegible]

[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible] A-h			[illegible], [illegible] A-h			[illegible] A-h			[illegible] A-h			Gulton, [illegible] A-h			Sonotone, [illegible] A-h		
			[illegible]	[illegible]	Days of cycling	Pack number	Cycles	Days of cycling	Pack number	Cycles	Days of cycling	Pack number	Cycles	Days of cycling	Pack number	Cycles	Days of cycling	Pack number	Cycles	Days of cycling	Pack number	Cycles	Days of cycling
0° C	[illegible] percent	1.5	[illegible]	[illegible]	[illegible]	110	10 963	[illegible]	51	11 533	720.8	84	10 909	681.8	61	10 146*	634.1	101	3631*	[illegible]	49	11 191	699.4
		3	[illegible]	[illegible]	[illegible]	111	[illegible]	684.[illegible]	[illegible]	[illegible]	706.9	80	5 427	678.4	65	5 549	693.6	102	5255	[illegible]	53	5 521	690.1
	25 percent	1.5	[illegible]	11 537	721.1	124	10 730	670.[illegible]	52	11 246	702.9	98	10 422	651.4	62	10 869	679.3	115	2291*	143.2	50	11 300	706.3
		3	68	5 [illegible]	708.0	125	5 472	684.0	56	5 634	704.3	94	5 309	663.6	66	4 414*	551.8	116	5097	637.1	54	5 539	692.4
25° C	25 percent	1.5	15	10 382*	648.9	82	10 878*	679.9	3	4 751*	296.9	104	2 980*	186.3	13	4 021*	251.3	73	7763*	485.2	1	10 971	685.7
		3	19	5 636	704.5	83	5 538	692.3	7	4 173*	521.6	105	5 227	653.3	17	2 885*	360.6	74	1754*	219.3	5	5 364	670.5
	40 percent	1.5	16	5 013*	313.3	96	4 020*	251.3	4	3 164*	197.8	118	2 837*	183.6	14	2 086*	130.4	87	627*	39.2	2	6 672*	417.0
		3	20	5 349	[illegible]	97	5 002*	625.3	8	2 494*	311.8	119	1 793*	224.1	18	1 550*	193.8	88	358*	44.8	6	5 211*	651.4
[illegible] C	[illegible] per cent	1.5	39	8 109*	506.8	85	9 710*	606.9	27	4 485*	280.3	112	5 213*	325.8	37	6 064*	379.0	76	9348*	584.3	25	9 328*	583.0
		3	43	[illegible]*	332.0	86	5 289	661.1	31	2 524*	315.5	108	4 273*	534.1	41	1 689*	211.1	77	5264	658.0	[illegible]	[illegible]	[illegible]
		[illegible]	[illegible]	[illegible]*	[illegible]	[illegible]	[illegible]*	303.3	[illegible]	1 811*	113.[illegible]	126	1 569*	98.1	38	1 377*	[illegible]	[illegible]	4045*	[illegible]		[illegible]*	[illegible]
		[illegible]	[illegible]	[illegible]*	[illegible]	[illegible]	[illegible]*	[illegible]	[illegible]	[illegible]*	121.8	122	985*	122.9	42	4 133*	[illegible]	[illegible]	[illegible]*	[illegible]	[illegible]	[illegible]*	[illegible]

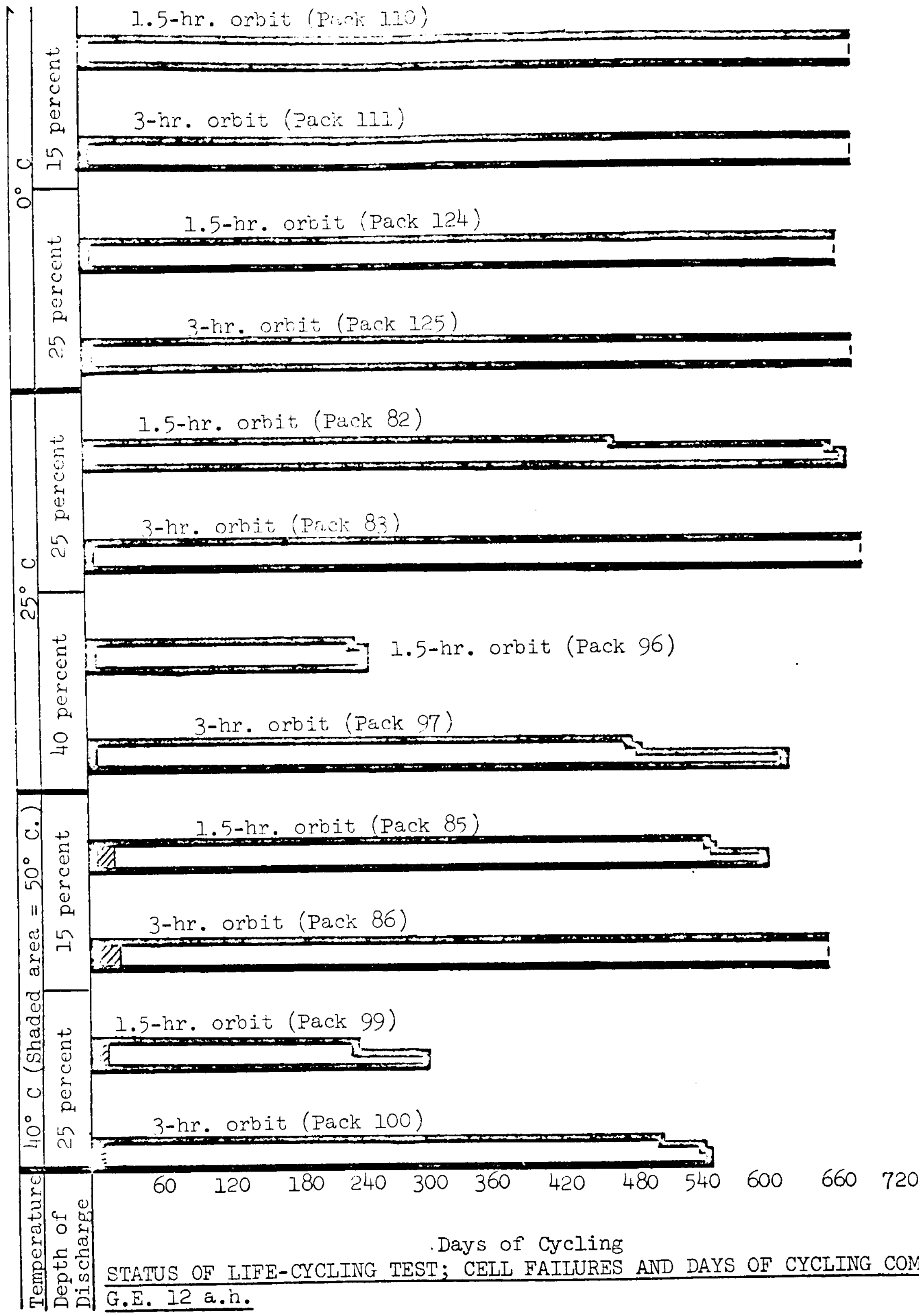
TABLE B-2. DISTRIBUTION OF CELL FAILURES ACCORDING TO TEST PARAMETERS AND CELL TYPES

Temperature, °C	Depth of discharge	Orbit period, hours	G.E., 3.0 A-h	G.E., 12 A-h	Gould, 3.5 A-h	Gould, 20 A-h	Gulton, 6.0 A-h	Gulton, 20 A-h	Sonotone, 5.0 A-h	Total failures	Total original cells
0°	15 percent	1.5	0	0	0	0	6	3	2	11	55
		3.0	0	0	0	0	1	1	0	2	55
	25 percent	1.5	0	0	0	2	4	3	0	9	55
		3.0	0	0	5	0	6	0	0	11	55
25°	25 percent	1.5	6	3	7	3	6	3	3	31	55
		3.0	0	0	6	1	6	3	2	18	55
	40 percent	1.5	6	3	6	3	6	5	6	35	55
		3.0	4	3	7	3	6	4	6	33	55
50/40°	15 percent	1.5	1/5	0/3	0/6	0/3	1/5	0/3	0/6	33	55
		3.0	0/6	0/0	0/7	0/3	0/6	0/0	0/4	26	55
	25 percent	1.5	0/7	0/3	0/10	1/2	2/4	0/3	0/6	38	55
		3.0	0/6	0/3	0/6	0/3	1/5	0/4	0/6	34	55
Total failures			41	18	60	24	65	32	41	281	
Total original cells			120	60	120	60	120	60	120		660



(a) G.E. 3.0 A-h

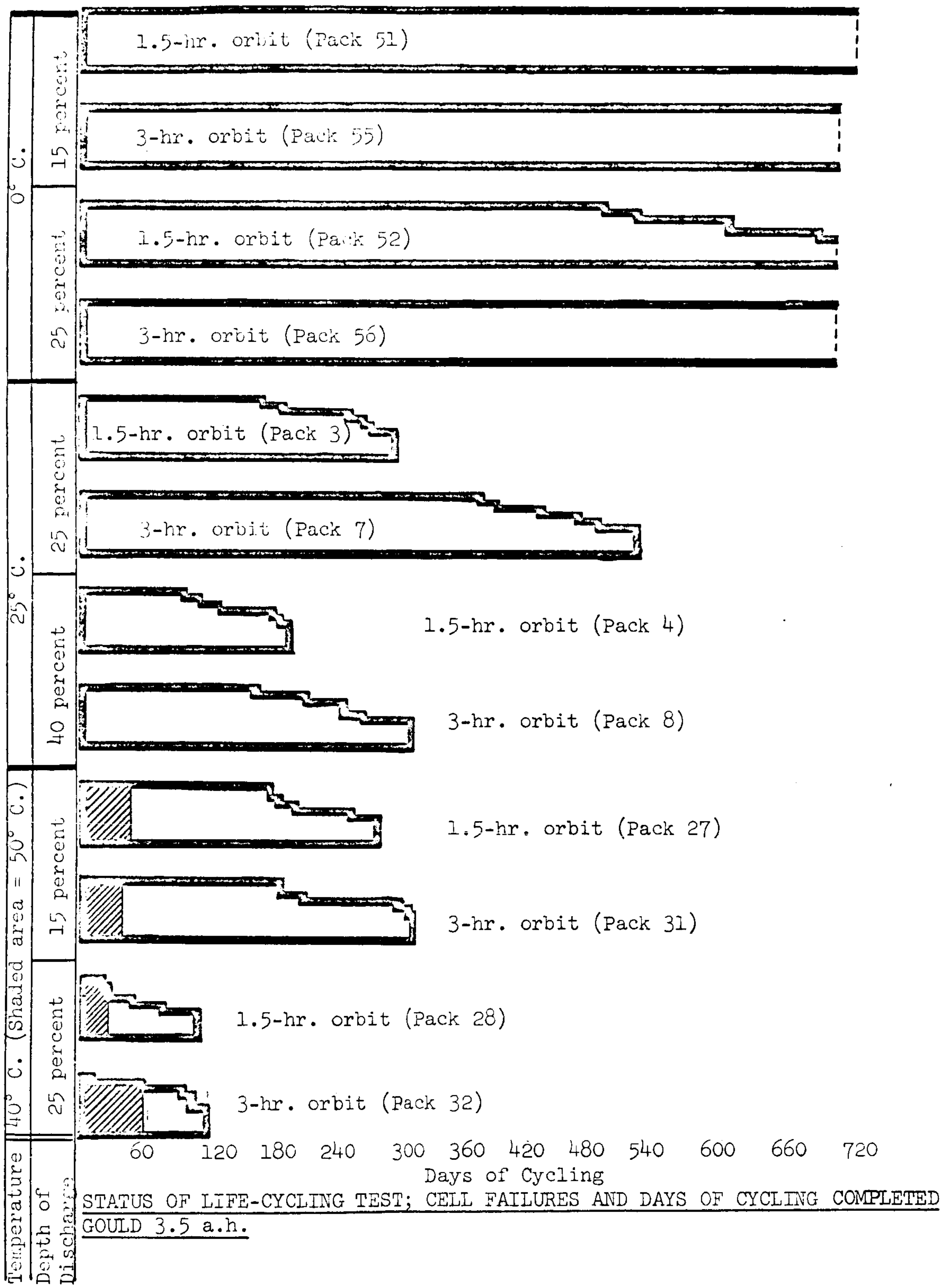
Figure B-1. - Status of life-cycling test; cell failures and days of cycling completed.



STATUS OF LIFE-CYCLING TEST; CELL FAILURES AND DAYS OF CYCLING COMPLETED
G.E. 12 a.h.

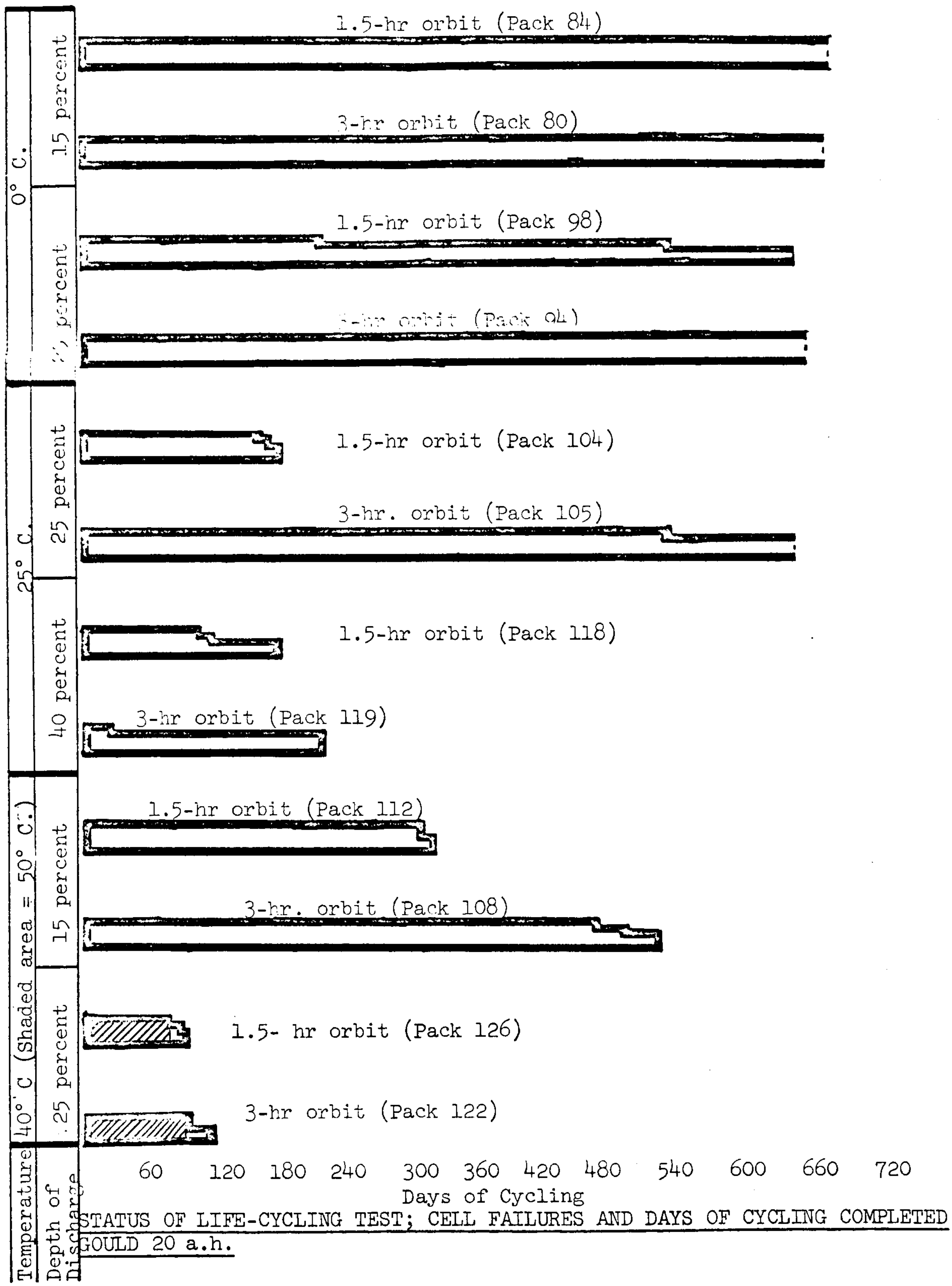
(b) G. E. 12 A-h

Figure B-1. - Status of life-cycling test; cell failures and days of cycling completed.



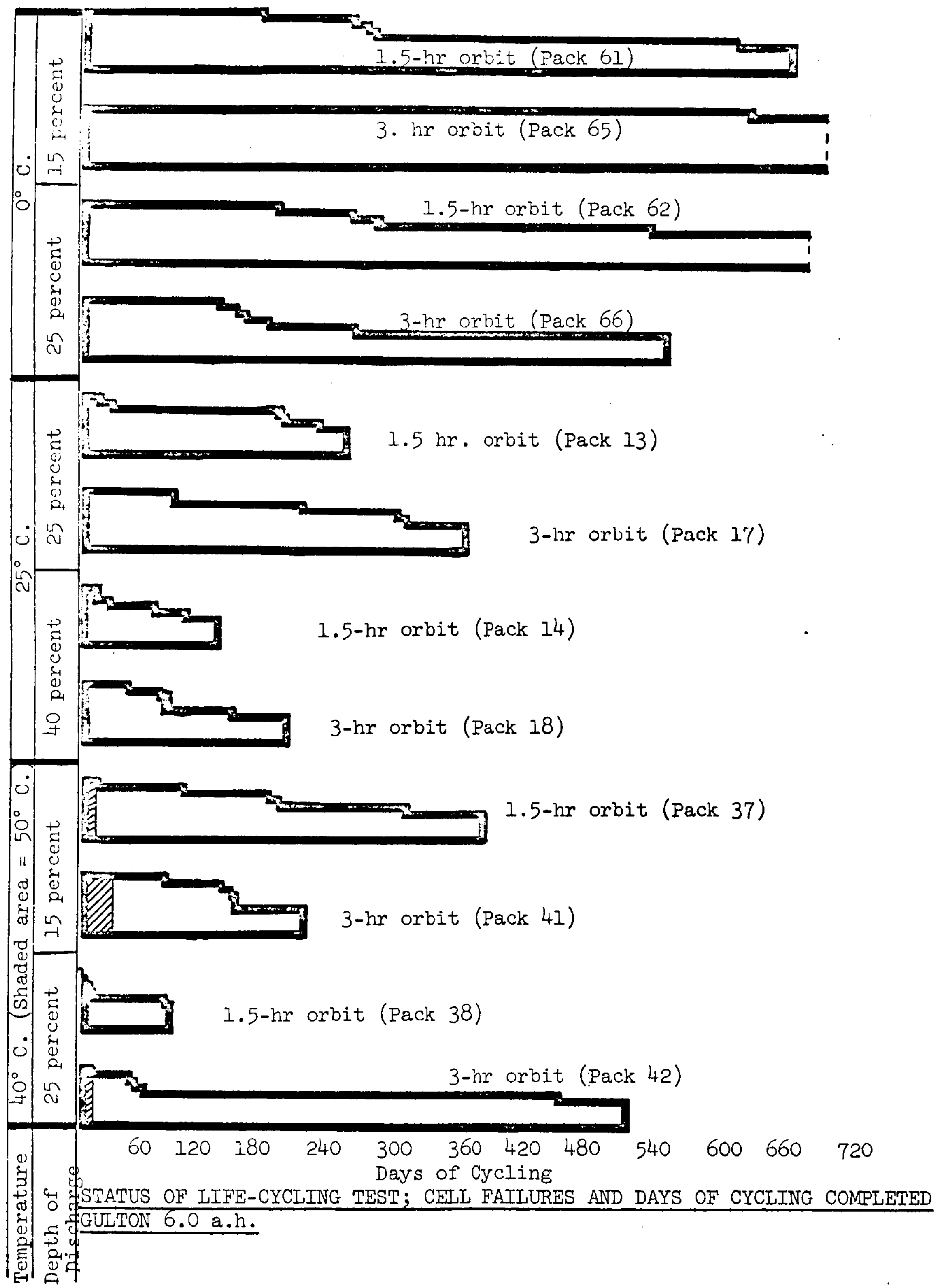
(c) Gould 3.5 A-h

Figure B-1. - Status of life-cycling test; cell failures and days of cycling completed.



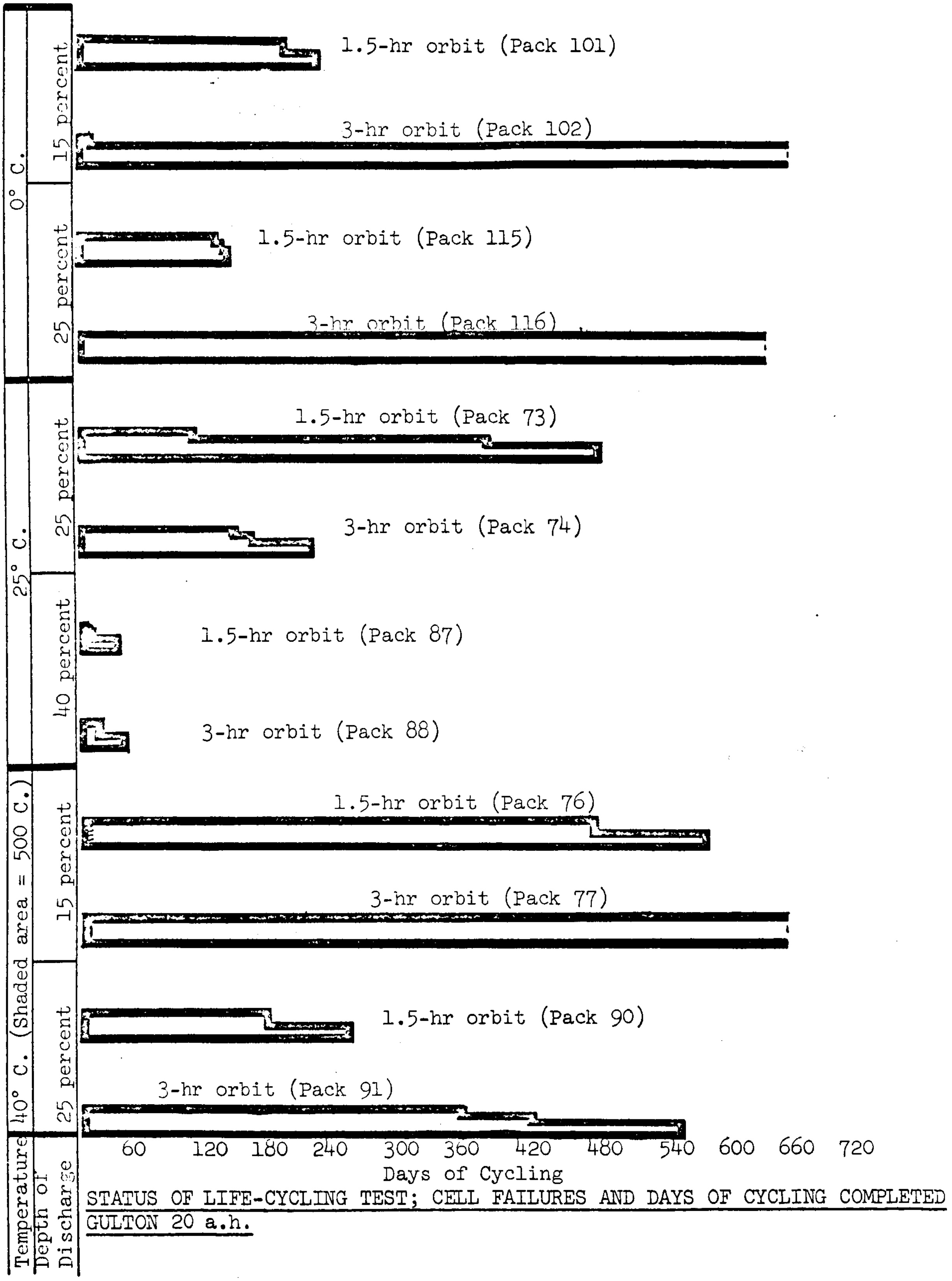
(d) Gould 20 A-h

Figure B-1. - Status of life-cycling test; cell failures and days of cycling completed.



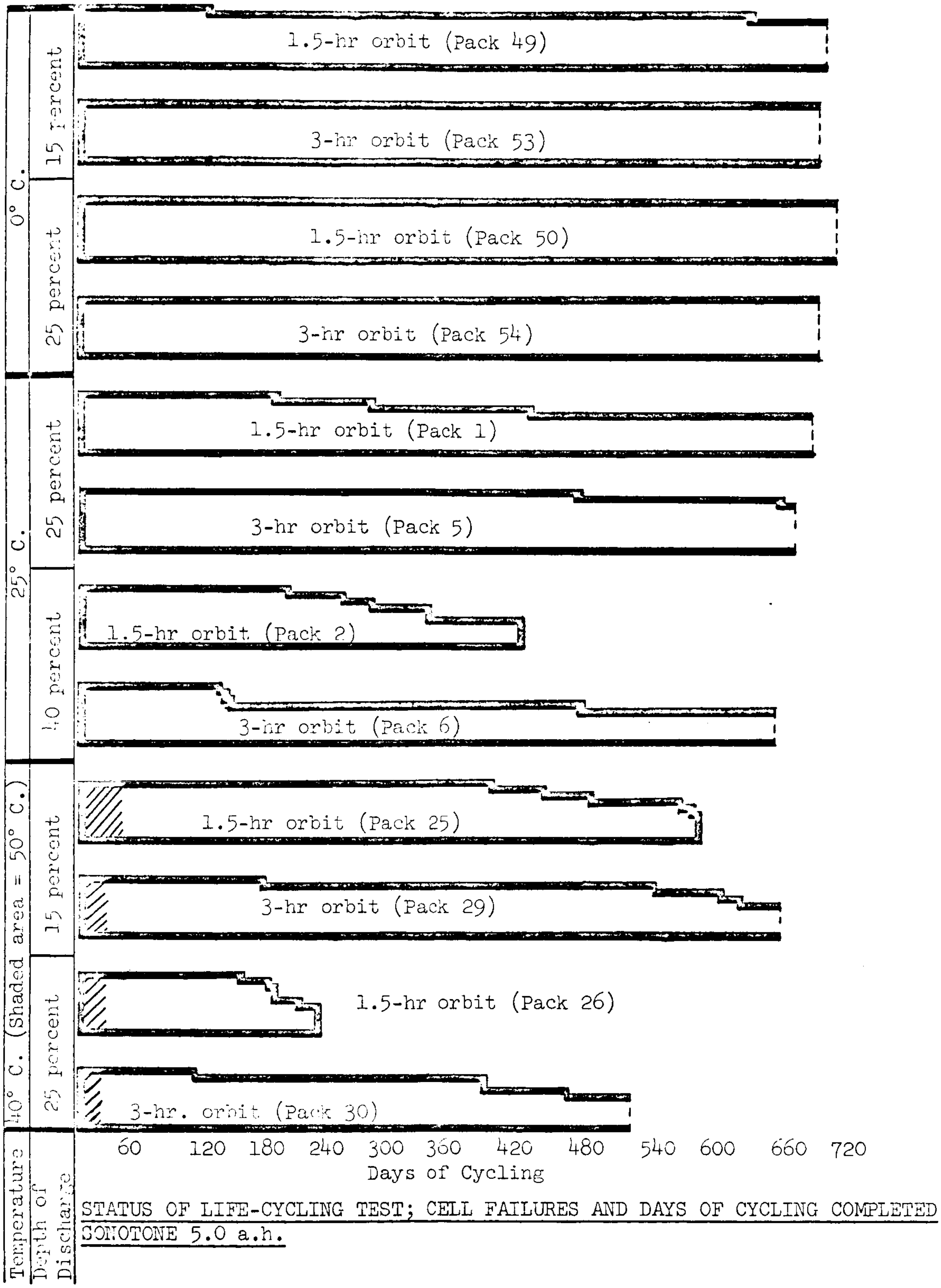
(e) Gulton 6.0 A-h

Figure B-1. - Status of life-cycling test; cell failures and days of cycling completed.



(f) Gulton 20 A-h

Figure B-1. - Status of life-cycling test; cell failures and days of cycling completed.



(g) Sonotone 5.0 A-h

Figure B-1. - Status of life-cycling test; cell failures and days of cycling completed.

APPENDIX C

DETAILED TABULATION OF FUEL CELL SUBSYSTEM WEIGHTS

MARS

Peaking Power Systems

Solar.- F/C regeneration: In order to calculate the energy and power requirements for reactant regeneration, the three power profiles for the Mars mission are used. It is seen that the most stringent duty cycle occurs between days 281 and 450. This profile, then, is chosen as the design case for reactant regeneration. Assuming that the primary solar cell system supplies all fuel cell parasitic power between peaks (fuel cells must be kept hot continually throughout the mission, since they are put on load approximately every 10 hr), the energy required for reactant regeneration can be calculated as follows for a 3-day period.

Useful energy requirement at bus = 14.75 kWh

Since two FCA's are required to handle this power requirement at a total load time of 14.5 hr, the parasitic energy requirement (at 100 watts parasitic power per FCA) during the time the fuel cells are on load is 14.5 hr × 0.2 kW or 2.9 kWh.

Using an overall distribution and power conditioning efficiency of 0.8, the total energy required for reactant regeneration is:

$$\frac{(14.75\text{kWh} + 2.9\text{kWh})}{0.8} = 22.1\text{kWh}$$

Then, using a 50 percent reactant regeneration efficiency and a 90 percent controller efficiency, the total energy required of a solar cell primary EPS is $\frac{22.1\text{kW}}{0.45}$, or 49.2kWh.

Since total fuel cell load time during the 3-day period is 14.5 hr, the time available for reactant regeneration is 57.5 hr. Hence, the power level required of the solar cell primary EPS is $\frac{49.2\text{ kWh}}{57.5\text{ hr}}$ or 857 watts.

But since the solar cell primary EPS supplies all fuel cell parasitic power between peaks, the total solar cell power requirement between peaks is $\left(857 + \frac{200}{0.8}\right)$ or 1107 watts.

However, according to a previous groundrule, 1000 watts of solar cell power is available between peaks. Therefore, the "delta" solar cell system weight penalty for reactant regeneration is $\left(107 \text{ watts} \times 0.38 \frac{\text{lb}}{\text{watt}}\right)$ or 41 lb.

Other more severe and less severe cases are possible for calculating the energy and power level for reactant regeneration, but this design case is considered adequate.

This particular case illustrates the manner in which the reactant regeneration calculations are made for all regenerative fuel cell system configurations in the study. Subsequent calculations are not illustrated in as much detail.

Weight summary, lb	Nominal mission power, kWe 5	10	15
7/9/13 FCA's at 200 lb	1400	1800	2600
Regeneration equipment package	100	100	100
"Delta" solar cell system penalty	41	133	266
Total system weight	1541	2033	2966

F/C non-regeneration:

a. Nominal mission power, 5 kWe

7 FCA's at 200 lb = 1400 lb

Reactants:

$$\begin{array}{ll} kWh_{net} & = 937 \\ kWh_{parasitic} & = 1575 \\ \hline kWh_{total} & = 2512 \end{array}$$

$$\text{Reactant weight} = \frac{kWh_{total} \times SRC\left(\frac{lb}{kWh}\right)}{\text{distribution and power conditioning}}$$

$$= \frac{2512 \times 0.8}{0.8} = 2512 \text{ lb}$$

Reactant tankage: 1368 lb

$$\left[2512 \text{ lb reactant} \times 0.545 \frac{\text{lb tank } (H_2 + O_2)}{\text{lb reactant}}\right]$$

Water tankage credit is 10 percent of water weight, that is, 251 lb.

Hence, total system weight (1400 + 1368 - 251) = 2517 lb

b. Nominal mission power, 10 kWe

9 FCA's at 200 lb = 1800 lb

Reactants:

$$\begin{array}{ll} kWh_{net} & = 1405 \\ kWh_{parasitic} & = 2081 \\ \hline kWh_{total} & = 3486 \end{array}$$

Therefore: $$\text{reactant weight} = \left(\frac{3486 \times 0.8 \text{ lb/kWh}}{0.8 \text{ distribution and power conditioning efficiency}}\right) = 3486 \text{ lb}$$

Tankage (H_2 and O_2) = 1900 lb

Therefore, total system weight = 3351 lb

c. Nominal mission power, 15 kWe

13 FCA's at 200 lb			= 2600 lb
Reactants:	kWh_{net}	= 1874	
	$kWh_{parasitic}$	= 2911	
	kWh_{total}	= 4785	
Therefore:	reactant weight		= 4785 lb
Tankage			= 2610 lb
Total system weight			= 4731 lb

Nuclear.-

F/C regeneration: The peaking requirements of Mars mission days 145 to 154 are the most stringent. Hence, this profile is used as the design case.

Two possibilities exist: (1) the fuel cell system supplies all its own parasitic power between peaks, and (2) the nuclear primary system supplies all fuel cell parasitic power between peaks.

For case (1), the energy output of the regenerative fuel cells during any 3-day period between days 145 and 154 is

$$\left[\frac{(\text{Net kWh}) + (\text{F/C parasitic during peaks}) + (\text{F/C parasitic between peaks})}{(\text{Distribution and power conditioning efficiency})}\right]$$

$$\text{or } \frac{(3\text{ kWh} \times 9 + 10\text{ kWh}) + (19\text{ hr} \times 0.2\text{ kW}) + (53\text{ hr} \times 0.2\text{ kW})}{(0.80)}$$

or 64.3 kWh

Hence the required energy output of the nuclear primary EPS for reactant regeneration is $\left(\frac{64.3}{0.45}\right)$ or 143 kWh. With 53 hr for charging between peaks, the average charge level is 2700 watts.

For case (2), that is, for all F/C parasitic power between peaks being supplied by the nuclear primary system, the energy output of the regenerative fuel cells during a 3-day period is

$$\frac{(\text{Net kWh}) + (\text{F/C parasitic during peaks})}{(\text{Distribution and power conditioning efficiency})}$$

or $$[(37 \text{ kWh}) + (19 \text{ hr} \times 0.2 \text{ kW})]/\ 0.8$$

or 51.1 kWh

For the same charge period as case (1), 53 hr, the net nuclear primary system EPS charge level is $\left(\frac{51.1}{0.45 \times 53}\right)$ or 2150 watts. But since the nuclear system supplies the 2-FCA F/C parasitic power between peaks at 100 watts per FCA, the total nuclear system charge level is 2350 watts.

Hence case (2) is chosen as the design case.

Weight summary, lb	Nominal mission power, kWe 5		10		15	
	R/I	Rx	R/I	Rx	R/I	Rx
11/11/22 FCA's at 200 lb	2200	2200	2200	2200	4400	4400
Regeneration equipment package	100	100	100	100	100	100
"Delta" nuclear system penalty	2350	1175	3470	1735	4530	2265
Total system weight	4650	3475	5770	4035	9030	6765

F/C non-regeneration:

a. Nominal mission power, 5 kWe

11 FCA's at 200 lb		= 2200 lb
Reactants:		
kWh_{net}	= 3105	
$kWh_{parasitic}$	= 2981	
kWh total	$= 6086 \times \frac{0.8}{0.8}$	= 6086 lb
Reactant tankage:	6086×0.545	= 3320 lb

Water tankage credit	= 609 lb
Therefore: total system weight	= 4911 lb

b. Nominal mission power, 10 kWe

11 FCA's at 200 lb		= 2200 lb
Reactants:		
kWh_{net}	= 4701	
$kWh_{parasitic}$	= 2981	
kWh_{total}	$= 7682 \times \frac{0.8}{0.8}$	= 7682 lb
Reactant tankage		= 4180 lb
Water tankage credit		= 768 lb
Therefore: total system weight		= 5612 lb

c. Nominal mission power, 15 kWe

22 FCA's at 200 lb		= 4400 lb
Reactants:		
kWh_{net}	= 6210	
$kWh_{parasitic}$	= 5394	
kWh_{total}	$= 11\ 604 \times \frac{0.8}{0.8}$	= 11 604 lb
Reactant tankage		= 6350 lb
Water tankage credit		= 1160 lb
Therefore: total system weight		= 9590 lb

Auxiliary Power Systems

Storm cellar EPS (non-regenerative F/C system).-

Assumed duty cycle: three solar events, 10 days each

Average power 2.5 kW → 720 hr × 2.5 kW = 1800 kWh
Peak Power 3.1 kW

<u>Weight Summary</u>

2 FCA's at 200 lb		= 400 lb
Reactants:		
kWh_{net}	= 1800	
$kWh_{parasitic}$	= 115	
kWh_{total}	$= 1915 \times \frac{0.8}{0.8}$	= 1915 lb
Reactant tankage		= 1043 lb
Water tankage credit		= 192 lb
Therefore: total system weight		= 1251 lb

Encounter experiments system
(needed only with a nuclear primary system).-

10-day duration assumed

Minimum power	1200 watts	
Average power	1500 watts	→ 354 kWh
Maximum power	2000 watts	

Weight Summary

2 FCA's at 200 lb = 400 lb

Reactants: $kWh_{net} = 354$

$kWh_{parasitic} = 39$

$kWh_{total} = 393 \times \frac{0.8}{0.8} =$ = 393 lb

Reactant tankage = 215 lb

Water tankage credit = 39 lb

Therefore: total system weight = 576 lb

Midcourse power system.-

6 midcourse corrections, 36 hr at 800 watts each

	w/nuclear primary EPS	w/solar primary EPS
Total load time, hr	220	72
Total cycles	6	2
Total energy, kWh	175	58
Energy per cycle, kWh	29	29

W/nuclear primary EPS:

1 FCA = 200 lb

Reactants: $kWh_{net} = 175$

$kWh_{parasitic} = 18$

$kWh_{total} = 193 \times \frac{0.8}{0.8} = 193$ lb

Reactant tankage	= 105 lb
Water tankage credit	= 19 lb
Therefore: total system weight	= 286 lb

W/solar primary EPS:

1 FCA = 200 lb

Reactants: $kWh_{net} = 58$

$kWh_{parisitic} = 8$

$kWh_{total} = 64 \times \frac{0.8}{0.8} = 64$ lb

Reactant tankage	= 35 lb
Water tankage credit	= 6 lb
Therefore: total system weight	= 229 lb

Single auxiliary EPS for storm cellar encounter experiments, and midcourse power.-

	w/nuclear primary EPS	w/solar primary EPS
Total load time, hr	1300	800
Total energy, kWh	2330	1860
Average power, kW	2.5	2.5
Maximum power, kW	3.1	3.1

W/nuclear primary EPS:

2 FCA's at 200 lb = 400 lb

Reactants: $kWh_{net} = 2330$

$kWh_{parasitic} = 210$

$kWh_{total} = 2540 \times \frac{0.8}{0.8} = 2540$ lb

Reactant tankage	= 1382 lb
Water tankage credit	= 254 lb
Therefore: total system weight	= 1528 lb

W/solar primary EPS:

2 FCA's at 200 lb			= 400 lb
Reactants:	kWh_{net}	= 1860	
	$kWh_{parasitic}$	= 130	
	kWh_{total}	$= 1990 \times \frac{0.8}{0.8}$	= 1990 lb
Reactant tankage			= 1090 lb
Water tankage credit			= 199 lb
Therefore:	total system weight		= 1291 lb

EARTH ORBIT

Peaking Power Systems

Low Earth orbit (power profile is the same for a nuclear and solar primary EPS).-

F/C regeneration:

Design case for reactant regeneration (3-day period) -

Useful energy requirement at bus	=
$(kWh_{net} + kWh_{3\text{-FCA parasitic}})$	=
$(21.25 + 0.3 \text{ kW} \times 25 \text{ hr})$	= 28.75 kWh

Using an overall distribution and power conditioning efficiency of 0.80, the total energy required for reactant regeneration is (28.75/0.8) or 35.9 kWh.

For a 50 percent reactant regeneration efficiency and a 90 percent controller efficiency, the total energy required of a primary solar or nuclear EPS is 79.8 kWh.

With 46 hr charge time available in the 3-day period, the average charge level required of a solar or nuclear EPS is 1735 watts.

a. Nominal mission power, 5 kWe
(assume that primary EPS supplies F/C parasitic power between peaks)

6-mo Resupp/2-yr optn.	Solar	R/I	Rx
2/7 sets of 3 FCA's	1200/4200	1200/4200	1200/4200
Equipment package, regeneration	100/ 100	100/ 100	100/ 100
"Delta" primary system weight penalty	342/342	735/735	368/368
Total system weight	1642/4642	2035/5035	1668/4668

b. Nominal mission power, 10 kWe
(assume that primary EPS supplies F/C parasitic power between peaks)

6-mo Resupp/2-yr optn.	Solar	R/I	Rx
2/7 sets of 4 FCA's	1600/5600	1600/5600	1600/5600
Equipment package, regeneration	100/100	100/100	100/100
"Delta" primary system weight penalty	468/468	735/735	368/368
Total system weight	2168/6168	2435/6435	2068/6068

2. F/C non-regeneration (solar OR nuclear):

6-mo Resupp/2-yr optn.

a. Nominal mission power 5 kWe

2/7 sets of 3 FCA's = 1200/4200 lb

Reactants: kWh_{net} = 2160/8640

$kWh_{parasitic}$ = 126/504

kWh_{total} $= (2286/9144) \times \frac{0.8}{0.8} \approx 2286/9144$ lb

Reactant tankage = 1250/4980 lb

Water tankage credit = 229/914 lb

Total system weight = 2221/8266 lb

b. Nominal mission power, 10 kWe

6-mo. Resupp/2-yr optn.

2/7 sets of 4 FCA's = 1600/5600 lb

Reactants: kWh_{net} = 3240/12 950

$kWh_{parasitic}$ = 163/672

kWh_{total} $= (3403/13\ 622) \times \frac{0.8}{0.8} = 3403/13\ 622$ lb

Reactant tankage = 1855/7430 lb

Water tankage credit = 340/1362

Total system weight = 3115/11 668 lb

Synchronous Earth Orbit

Solar.-

F/C regeneration:

a. nominal mission power, 5 kWe
(assume that primary EPS supplies F/C parasitic power between peaks)

6-mo Resupp/2-yr optn.

2/8 sets of 4 FCA's	= 1600/6400 lb
Equipment package, regeneration	= 100/100 lb
"Delta" primary system weight penalty	= 239/352 lb
Total system weight	= 1739/6852 lb

b. Nominal mission power, 10 kWe

2/8 sets of 6 FCA's	= 2400/9600 lb
Equipment package, regenation	= 100/100 lb
"Delta" primary system weight penalty	= 352/352 lb
Total system weight	= 2853/10 052 lb

F/C non-regeneration:

a. Nominal mission power, 5 kWe

2/8 sets of 4 FCA's = 1600/6400 lb

Reactants: kWh_{net} = 1960/7840

$kWh_{parasitic}$ = 243/540

kWh_{total} $=(2203/8380) \times \frac{0.8}{0.8}$ = 2203/8380 lb

Reactant tankage	= 1245/4575 lb
Water tankage credit	= 220/838
Total system weight	= 2625/10 137 lb

b. Nominal mission power, 10 kWe

2/8 sets of 6 FCA's		= 2400/9600 lb
Reactants:	kWh_{net} = 2940/11 750	
	$kWh_{parasitic}$ = 333/1332	
	kWh_{total} $=(3273/13\ 082) \times \frac{0.8}{0.8}$	= 3273/13 082 lb
Reactant tankage		= 1780/7130 lb
Water tankage credit		= 327/1308
Total system weight		= 3852/15 422 lb

Nuclear.-

$\boxed{\text{6-mo Resupp}/\text{2-yr optn.}}$

F/C regeneration:

a. Nominal mission power, 5 kWe

	R/I	Rx
2/8 sets of 1 FCA	= 400/1600	400/1600
Equipment package, regeneration	= 100/100	100/100
"Delta" primary system penalty	= 1895/1895	950/950
Total system weight	= 2395/3595	1450/2650

b. Nominal mission power, 10 kWe

	R/I	Rx
2/8 sets of 1 FCA	= 400/1600	400/1600
Equipment package, regeneration	= 100/100	100/100
"Delta" primary system penalty	= 2840/2840	1420/1420
Total system weight	= 3340/4540	1920/3120

2. F/C non-regeneration:

a. Nominal mission power, 5 kWe

			R/I and Rx
2/8 sets of 1 FCA			= 400/1600 lb
Reactants:	kWh_{net}	= 935/3740	
	$kWh_{parasitic}$	= 432/1680	
	kWh_{total}	$= (1367/5420) \times \frac{0.8}{0.8}$	= 1367/5420 lb
Reactant tankage			= 745/2960
Water tankage credit			= 137/542
Total system weight			= 1008/4018 lb

b. Nominal mission power, 10 kWe

			R/I and Rx
2/8 sets of 1 FCA			= 400/1600 lb
Reactants:	kWh_{net}	= 1402/5610	
	$kWh_{parasitic}$	= 432/1680	
	kWh_{total}	$=(1834/7290) \times \frac{0.8}{0.8}$	= 1834/7290 lb
Reactant tankage			= 1000/3980 lb
Water tankage credit			= 183/729
Total system weight			= 1217/4851 lb

PRELIMINARY TECHNICAL DATA

FOR EARTH ORBITING SPACE STATION

VOLUME III

SYSTEMS

SECTION 2.0

ENVIRONMENT CONTROL/LIFE SUPPORT SYSTEM

CREW SYSTEMS DIVISION

ENGINEERING AND DEVELOPMENT DIRECTORATE

MANNED SPACECRAFT CENTER

2.0 ENVIRONMENTAL CONTROL/LIFE SUPPORT SYSTEM

2.1 INTRODUCTION

The Environmental Control/Life Support System (EC/LSS) described herein is designed for an artificial gravity space station with a large zero gravity hub. The hub and the rotating module will be independently sealed, except during crew transfer, to minimize rotating seal leakage. Hence, two independent systems will be required.

It appears that total crew time will be divided about evenly between the hub and the rotating module. The two systems can, therefore, be identical with some exceptions such as radiator panels, cold plates, etc. For the nine man crew considered here, each EC/LSS has been sized for six men to accommodate more efficiently periods of unequal crew distribution. However, the cabin is sufficiently large that overloading for several hours will not cause a serious CO_2 problem.

This discussion incorporates the Sabatier CO_2 reduction process into the EC/LSS. Although CO_2 reduction is not included in the recommended system at this time, the process is shown here to illustrate its integration into the overall system. It is not included in the weight summaries elsewhere in this report.

EC/LSS guidelines are presented in Table 2.1. Estimated weights and power requirements are shown in Table 2.2. These represent the total system. Figure 2.1 is a schematic diagram of a single EC/LSS.

The EC/LSS discussed in this report is equally applicable to zero gravity or artificial gravity space stations. The latter has been used since it presents constraints not involved in a zero gravity station, whereas the hub of the artificial gravity station has all the problems of the zero gravity configuration.

2.2 SYSTEM DESCRIPTION

The vehicle environmental control system provides a conditioned, shirtsleeve atmosphere for the crew, thermal control of all electrical equipment, and a closed cycle water supply system.

A cryogenic subcritical oxygen and nitrogen storage system will provide the leakage needs, CO_2 dissociation inefficiency, the experimental requirements, airlock repressurizations, backpack recharges, and sufficient stores for a laboratory pressurization. Additional oxygen and nitrogen reserves are provided to allow for boil-off during prelaunch and postlaunch standby. A gaseous accumulator tank is provided in both the oxygen and nitrogen supply subsystems as a surge tank to prevent spasmodic system

TABLE 2.1

EC/LSS GUIDELINES

Orbital altitude	260 n.mi.
Resupply interval	3 - 6 months
Module emissivity	0.24
Module absorptivity	0.22
Radiator emissivity	0.9
Radiator absorptivity	0.2
Mission duration	5 years
Electrical power:	15 kw average 20 kw peak
Crew size	9
Metabolic heat	11,200 Btu/man-day
O_2 consumption	1.84 lb/man-day
CO_2 production	2.12 lb/man-day
Drinking water	6.07 lb/man-day
Water of oxidation	0.337 lb/man-day
Urine water	3.08 lb/man-day
Fecal water	0.22 lb/man-day
Wash water	26.4 lb/man-day
Cabin pressure	14.7 psia (21/79:O_2/N_2)
CO_2 partial pressure	7.6 mm Hg maximum
Cabin temperature	75 $\pm$ 5°F
Leakage rate	18 lb/day

TABLE 2.2

EC/LSS WEIGHT AND POWER REQUIREMENTS

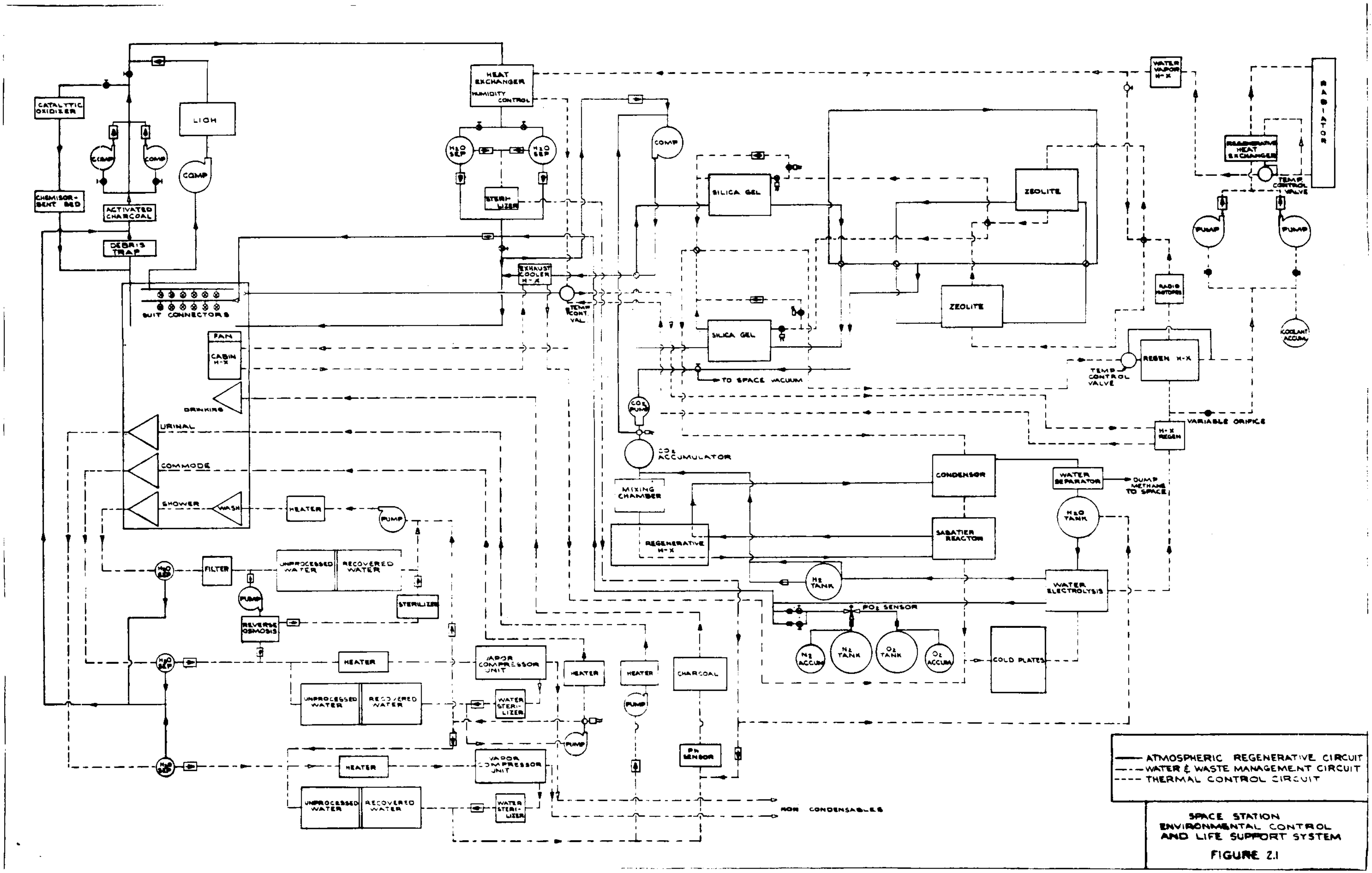
	Weight, Pound	Power, Watts
Atmospheric Regeneration Loop	175	200
Carbon Dioxide Removal	420	300
Carbon Dioxide Reduction	220	1100
Cabin Circulation Loop	225	200
Coolant Loop	4050	1000
Water Supply System	750	200
Solid Waste Management	440	100
Subtotal	6280	
Hydrogen and Tank	650	
Other Expendables	140	
Total	7070	3100
Total without CO_2 Reduction	6110	2000

9-man crew

Artificial gravity

3-month resupply + 50% contingency

Electrical power: 15 kw average, 20 kw peak



SPACE STATION ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEM

FIGURE 2.1

demands from being reflected in the cryogenic heater control system.

Reclamation systems will be provided for reclaiming metabolic oxygen from carbon dioxide and water. The carbon dioxide will be reduced by reacting it with hydrogen to form methane and water (commonly called the "Sabatier reaction"). This water plus a sufficient amount of the humidity loop condensate is electrolyzed to oxygen (by a capillary matrix - KOH, electrolysis cell) and supplied to the cabin atmosphere for metabolic consumption. The hydrogen formed during electrolysis is returned to the reactor where it again reacts with carbon dioxide. The methane produced during the reaction is separated from the water of reaction and discarded to space.

Crew comfort will be maintained by two atmospheric loops - an atmospheric regeneration circuit and a cabin atmospheric thermal circuit. The regenerative circuit loop supplies the conditioned atmosphere to the crew compartment while the cabin loop circulates the cabin atmosphere and controls the crew compartment temperature.

The regenerative circuit loop will provide particulate removal, noxious gas removal, humidity control and carbon dioxide removal. A debris trap will provide aerosol and particulate removal. Noxious gases will be controlled by acid impregnated activated charcoal and a radioisotope heated catalytic burner. Humidity control will be accomplished by condensing and collecting perspired and respired water vapor from the atmosphere utilizing a liquid coolant heat exchanger and a centrifugal water separator. The partial pressure of carbon dioxide is controlled by passing a portion of the cabin atmosphere through a regenerable solid adsorption subsystem. The device removes carbon dioxide from the atmosphere by adsorbing carbon dioxide on molecular sieves. After adsorbing carbon dioxide from the gas stream, the molecular sieve bed is regenerated through the application of heat from a radioisotope heat source. The released CO_2 is pumped into a surge tank for utilization by the oxygen reclamation subsystem. Sufficient circulation through the regenerative circuit loop will be provided by a constant flow blower. This additional blower is included in the molecular sieve circuit to provide the necessary pressure rise without an excessive power penalty.

The cabin circulation loop will consist of a protective screen, blower and liquid coolant heat exchanger with an integral wick water separator. The protective screen protects the cabin blower from damage by particulate material in the atmosphere; the blower circulates the required cabin flow; the liquid coolant heat exchanger maintains the cabin temperature within limits; and, the wick prevents condensed moisture from being introduced into the cabin during normal thermal transients or transients induced by a control failure. Several individual cabin circulation units, similar to the above, may be required to maintain adequate circulation in separate laboratory compartments. This can be better defined after more specific vehicle designs are available.

Thermal control of the vehicle will be accomplished through dissipation of the internal heat load with compensations for the varying external thermal environment. The heat rejection system will consist of a space radiator and a coolant circulation loop. The liquid coolant will circulate through the necessary environmental control system components and the electrical equipment coldplates to provide heat dissipation. The coolant flow will be maintained at a fixed rate by one of two constant speed pumps.

Water will be stored in positive expulsion tanks. Sufficient quantity for metabolic and personal hygiene purposes will be provided at all times. A portion of the water collected in the humidity control water separators will be made available to the water electrolysis unit to supplement the oxygen reclamation system. The remainder of the humidity condensate and all the wash water and urine are purified by a water reclamation system. The humidity condensate is purified by passing it through a water sterilizer consisting of a silver ion exchange bed. The wash water is filtered, regenerated through a reverse osmosis process, and passed through another silver ion exchange sterilizer before being made available for reuse. The solid waste processing system utilizes a wash and rinse cycle. This water is mixed with the fecal material after defecation. The water from the mixture is then reclaimed through a vapor compression process, sterilized by a silver ion exchange bed, and recycled to the flush water supply. Urine is collected separately and reclaimed through another vapor compression unit. It is then sterilized by a silver ion exchange bed before being made available for drinking.

This system is presented schematically in Figure 2.1.

The oxygen and water subsystems should be located so that they will be easily accessible for continuous servicing by the crew. The carbon dioxide removal and reduction systems, water separators, blowers, and pumps will be more subject to failure than the static components. They should be located in positions where repair can be affected (i.e., the location should be chosen to reflect accessibility as well as environmental parameters).

2. SYSTEM OPERATION

2.3.1 ATMOSPHERIC REGENERATION CIRCUIT

The cabin gaseous environment is revitalized by the atmospheric regeneration circuit through CO_2 absorption, noxious and toxic gas removal, filtering, water vapor control, and thermal dissipation. This circuit utilizes a blower system, condensor-heat exchanger, mechanical water separator, contaminant removal circuit, the CO_2 management circuit and filters. Makeup for leakage and O_2 reclamation system inefficiencies is also provided through the regeneration circuit from cryogenic stores.

Secondly, the atmospheric regeneration circuit purifies the directional gas flow from the commode, urinal, and shower by inputting this flow upstream of the contaminant control circuit.

A third, but important, function of the atmospheric regeneration circuit is its emergency crew support facilities. For this purpose, a separate gas compressor, LiOH CO_2 removal subsystem, and six sets of suit connectors were added. This provision is not included for life support subsystem failures, since redundancy and maintainability are innate features of this EC/LSS design, but in the event of a catastrophe such as loss of cabin pressurization.

The atmospheric regeneration circuit contains two compressors, a condenser, mechanical water separator, and lithium hydroxide for emergency CO_2 removal. The compressors are of the single-stage, centrifugal type. A condenser is a non-wick type, plate-fin heat exchanger. The gas/liquid separation is effected by a mechanical centrifugal-type separator. The LiOH is packaged in easily managed amounts to facilitate changing canisters under the emergency mode.

2.3.2 ATMOSPHERIC TRACE CONTAMINANT CONTROL CIRCUIT

Removal of odors and trace contaminants from the cabin atmosphere is accomplished in the atmospheric regeneration circuit. The atmospheric flow is directed through an absorpbion bed where most of the odors and trace contaminants are absorbed. Compounds which are not readily absorbed (e.g., hydrogen, carbon monoxide, and methane) are controlled by subsequently directing a small portion of the atmospheric flow through a catalytic oxidizer by diverting around the circuit blowers using their normal pressure rise for power. A post-chemisorbent bed then removes contaminants from the oxidizer effluent and the flow is returned to the atmospheric regeneration circuit, upstream of the main absorption bed.

The trace contaminant control system consists of a large absorbent bed in the main gas stream of the atmospheric regeneration circuit and a low flow, high temperature, catalytic oxidizer with a post-chemisorbent bed.

The large absorbent bed, which is rechargeable during the mission, uses acid impregnated activated charcoal as the absorbent.

The catalytic oxidizer is sized for 3 CFM flow and has integral heat exchanger to reduce the thermal power requirement which is supplied by a radioisotope heater.

The chemisorbent bed, which is rechargeable during the mission, handles the total effluent of the catalytic oxidizer.

2.3.3 CO_2 COLLECTION/O_2 RECLAMATION CIRCUIT

The CO_2 management system includes the CO_2 removal, transfer and reduction systems, as well as the necessary integration equipment. The CO_2 removal system is a four bed regenerable solid absorption system which utilizes silica gel as a desiccant and molecular sieves (or zeolites) for CO_2 removal. Flow enters the removal system from the atmospheric regeneration circuit and is directed through one of the alternate silica gel beds. Here, the air is dried to a few parts per million H_2O to allow removal of the CO_2 without contaminating the molecular sieve which preferentially absorbs water. From the silica gel bed the process air is routed to one of the molecular sieve beds for removal of the CO_2. The silica gel and molecular sieve beds have integral heat exchangers. During absorption cycles, low temperature transport fluid is circulated through these beds to remove the heat of adsorption and provide a favorable equilibrium condition. Upon leaving the molecular sieve bed, the flow returns through the other silica gel bed where the moisture removed during the absorption cycle is driven off. During desorption, high temperature transport fluid from the radioisotope heater is circulated through the bed; however, only until the moisture has been removed, then the transport fluid is diverted around the bed. After picking up the moisture from the desorbing silica gel bed, the process gas is returned to the atmospheric regeneration circuit. The transport fluid returns through a regenerable heat exchanger, preheating the flow to the radioisotope heater, and continues to the space radiator.

The off-line molecular sieve bed is desorbed of its CO_2 by the addition of heat from the transport fluid circulating through the bed, and the CO_2 is pumped into an accumulator which serves as feed source for the CO_2 reduction system.

The CO_2 reduction system consists of a Sabatier reactor, a hydrogen storage tank, and an electrolysis cell and the necessary integration equipment. The CO_2 from the accumulator is mixed with hydrogen from the electrolysis cell and the storage tank and fed through a regenerative heat exchanger to the Sabatier reactor.

In the reactor, CO_2 and hydrogen react in the presence of the catalyst to form methane and water. The reaction is exothermic and has an unfavorable equilibrium shift with increase in temperature. Therefore, the reactor is constructed with an integral heat exchanger to control the reaction temperature to 500°F. The product gas is fed counter current to the feed gas through the regenerative heat exchanger and then to a condenser and a gas/liquid separator. The gas is dumped overboard and the water removed by the separator is directed to the electrolysis cell by the overall system control where it is electrolyzed to hydrogen and oxygen.

Since some hydrogen is dumped overboard in the form of methane, cryogenic hydrogen is stored in a state for makeup.

2.3.4 CABIN ATMOSPHERIC THERMAL CIRCUIT

The cabin atmospheric thermal circuit maintains a reasonably temperatured environment for the crewman while dissipating the environmental heat leak, non-coldplated electronics and atmospherically cooled experiments. This is accomplished with two high flow blowers in conjunction with a plate fin/integral wick heat exchanger. Although the system is designed to preclude heat exchanger condensation, the wicks should retain cabin atmospheric water vapor which might condense during abnormal system transients, at least, until the situation is corrected and the water reevaporates. A heating mode is also included in the cabin heat exchanger for a high negative environmental heat leak condition when it is associated with low internal thermal loads.

2.3.5 THERMAL CONTROL CIRCUIT

Thermal control is provided by a single coolant circuit which serves the atmospheric regeneration circuit, cabin cooling, CO_2 collection/O_2 reclamation subsystems, and the electronic componentry. Heat rejection is accomplished with a space radiator. The coolant selected for this application is FC - 75. The wide operating temperature range, non-toxic and non-flamable fluid nature, and the fluid's materials compatibility most strongly influenced the selection. However, the potential fluid freezing because of the low temperatures induced by the regenerator at minimum heat load in a favorable environment could not be assessed absolutely; and, thus, a fluid selection change might result from a detailed systems evaluation. This systems evaluation could demonstrate that the potential freezing problem of FC - 75 can be relaxed by incorporating an in-line heater upstream of th the radiator system.

The heat rejection system outlet fluid stream (45°F) is divided and directed to the components requiring the coldest fluid. One leg goes to the humidity control heat exchanger of the atmospheric regeneration circuit, and then to the cabin heat exchanger. Since the coolant outlet temperature is bounded by the cabin atmospheric temperatures under maximum load ($80^{\circ}F$), a minimum coolant flow is established. The second leg is directed to the molecular sieve/silica gel beds, Sabatier condenser, Sabatier reactor, and then rejoins the coolant from the cabin heat exchanger. Since the CO_2 adsorption efficiency is adversly affected by increasing temperature, a coolant outlet restriction of $60^{\circ}F$ is required. The coolant flow necessary to satisfy this condition is the minimum flow for this leg; thus the temperature requirements for the condenser and reactor are met by a series coolant arrangement with this flow rate.

The total coolant flow is next routed through the coldplates, the water electrolysis cell, and cabin reheater. The established minimum flow meets, with margin, the temperature requirements of this equipment.

The last requirement of the thermal control loop is the heating of the molecular sieve desorption beds; therefore, this component is located in the circuit in a high temperature location. However, since the temperature requirements for desorption ($350^{\circ}F$) exceed the available coolant temperature, an auxiliary heater is included. The heater requirements are minimized by dividing the coolant and thus heating only a small portion of the total flow. Also by recovering most of the high temperature of this coolant in a regenerative heat exchanger before mixing with the main coolant stream the heater requirements are further reduced.

The radiator consists of parallel tubes mounted on a thin aluminum panel (.02 inches) such that flow is circumferentially routed along the cylindrical surface of the space station. Prior to entering the radiator panels, the coolant is automatically divided into two symmetrical legs which feed the radiator fluid manifold system. The automatic flow distribution is accomplished with a proportioning valve which prevents undesired flow maldistribution between panels and increases heat rejection capability during periods when the environmental sink temperatures for the two panels differ. An inlet fluid manifold system is provided such that all radiating tubes receive approximately equal flow during high heat load operation. Although the radiator is sized to reject the maximum nominal EC/LSS heat load, short duration loads could exceed the radiator rejection capability. As a result, a water evaporator is included downstream of the regenerative heat exchanger to provide supplemental cooling capability; however, this evaporator is sized to cool the entire system heat load under emergency low-load conditions. The outlet fluid manifold configuration collects flow from the individual tube passages and joins the "mixed" coolant from the second panel. The total system flow then is directed to the regenerative heat exchanger, which attenuates radiator performance for low load operation by reducing the inlet radiator temperature. This is accomplished in the regenerator by cooling the inlet flow with the cold radiator outlet flow. The net effect of this process is to lower the average radiator temperature and thus reduce heat rejection. Valve stagnation is also employed to aid low load operation by reducing radiator fin effectiveness. (If the requirement for wide heat load resolution is relaxed, a combination bypass-valve stagnation design could replace the selected regenerator-valve stagnation system and thus save some system weight.)

The radiator panels are isolated from the space station structure in order to minimize the thermal interface. Radiator plumbing lines, valves and fluid manifolds on the underside of the panels are also insulated with superinsulation to reduce heat gains or losses. Redundant (or secondary) tube passages are provided on each radiator panel and will preferably be mounted directly beneath the primary tube passages which are exposed to the space environment. This will provide two flow systems with each taking maximum advantage of the radiating area. A suitable fabrication technique will be employed so that radiator transient response capability is not degraded with this extra set of tubes. To assure maximum utilization of the total radiating area, independently controlled isolation valves are provided for each of the panels in the primary and secondary radiator circuits. This feature allows use of the primary circuit on one panel and the secondary circuit on the other panel.

The surface coatings utilized for the radiator have a maximum solar absorptivity of 0.20 and an emissivity of at least 0.90. These coating values will be maintained throughout the entire operational life of the radiator system since heat rejection capability is very sensitive to coating properties in the worst environment conditions.

2.3.6 WATER AND WASTE MANAGEMENT CIRCUIT

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

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

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

The humidity condensate is only sterilized before storage since no chemical impurities will be present in this water except those that are absorbed from the atmospheric regeneration circuit gas. This water is used as drinking water; and, since it is free from process system anomalies, it is electrolized for metabolic oxygen consumption as supplement to the Sabatier condensate.

These three major subsystems areas - the wash water, feces, urine - are further described in the three succeding paragraphs.

The wash water loop utilizes, as the basic processing system, a Reverse Osmosis Unit. Water used for sponge baths, showering, washing clothes, etc., is pumped into a holding tank for subsequent treatment. The osmosis unit separates the bulk of the detergent, dirt and particulate matter from this water resulting in reclamed water and a concentrated process stream. The reverse osmosis concentrate is pumped to the fecal water loop for final solids removal. Water processed by the reversed osmosis treatment is pumped through a sterilizer to a storage tank.

The fecal water loop utilizes as the basic processing system, a Vapor Compression Vacuum Distillation Unit. Prior to defecation, a small quantity of water is injected into the blender portion of the commode. Subsequently, feces are blended into a slurry and pumped to a vapor compression apparatus. Here, water and other volatiles are evaporated and recondensed. The water is separated, pumped through a sterilizer and into a recovery tank and waste gases are vented overboard. Residual solid waste from the feces and wash water is periodically removed and stored.

The urine loop utilizes, as the basic processing system, a second vapor compression unit. The urine and rinse water introduced into the urine system are processed by filtration, vapor compression and bactericidal techniques like those discussed above; and in fact, the apparatus is separate but identical. Unprocessed dilute urine is held in pressurized tanks until treatment; after which the resulting potable water is available for drinking. However, additional filtration for silver ion removal will be imposed on any water drawn for drinking from this tank. The fecal matter and urine are directed into the receptacle with gas jets utilizing cabin gas. This directional flow is purified and all odors are removed by injecting the gas into the atmospheric regenerative curcuit, upstream of the contaminant removal system.

The commode consists of a seat with a restraint belt, a pump-blender, sampling unit, volume measuring unit, and air and water heaters. The seat has a fecal transport opening with an annular section immediately below the opening which contains water and air jets around the periphery for washing and drying the anal area.

The pump-blender consists of a stationary housing containing an electrically driven rotating plate with a cylindrical coarse mesh screen in the center and series of blender blades located on the periphery. The rotating screen breaks up the stool which is then finely blended with the flush water by the blender blades. A discharge port is located at the bottom of the housing and a septum sealed sampling port on the side.

The sampling port receives a syringe for sample collection. Sampling is achieved by withdrawing the plunger. A turbine-type flow meter measures the volume of slurry pumped to the holding tank.

The urine collection unit consists of a urinal, a phase separator, a volume measuring unit, a sampline unit and air and water heaters. The urinal has a diaphram-type splash shield.

The phase separator consists of a stationary housing containing an electrically driven vane type impeller which centrifugally separates the liquid collected from the transport air. A discharge port is located at the bottom of the housing and a septum-sealed sampling port on the side. The sampling port receives a syringe for sample collection. Sampling is achieved by withdrawing the plunger. A turbine-type flowmeter measures the volume of urine pumped to the holding tank.

Two identical vapor compression units are provided to permit separate processing of urine and fecal waters. The vapor compression (V-C) units include three concentric cylinders consisting of boiler, condenser, and drier. A compressor is used to raise the temperature of the vapor produced in the boiler and to transfer it to the condenser where the vapor condenses transferring its heat of condensation across the common boiler condenser wall to evaporate more liquid. Automatic purge of non-condensables to vacuum is accomplished to maintain the condenser operating pressure. Liquid orientation is achieved by rotating the V-C unit. The boiler has a circumferential wiper blade which is automatically actuated to push the concentrated feed from the boiler surface to the drying chamber where the additional water is reclaimed and the solid residue is stored.

The reverse osmosis system consists of multiple membranes bonded to both sides of porous support plates assembled in series. The waste water is channeled between the plates. The product passes through the membrane into the porous membrane support plates, and is collected at the periphery of the plates. Rate of feed to the system at the required pressure is provided by a positive displacement pump. A flow control valve is used to bypass a portion of the concentrate.

The water sterilization units introduce a bactericidal agent into the effluent of the water processing units to effect a positive kill and prevent subsequent bacterial growth in the storage tanks. The sterilization units consist of cylindrical tubes containing silver chloride dispersed in a matrix of glass beads. Silver ion is eluted into the recovered water killing bacteria present.

Constant volume tanks are used in each of the three loops in the water and waste management subsystems. The tanks are identical in design and consist of two hemispherical sections separated by a pair of flexible bladders which separate the unprocessed and processed liquids, allowing variations in volume of either liquid to occur while still maintaining a constant total volume. Bladder pressurization is used to transfer the materials to the processing equipment.

PRELIMINARY TECHNICAL DATA

FOR EARTH ORBITING SPACE STATION

VOLUME III

SYSTEMS

SECTION 3.0

INSTRUMENTATION

INSTRUMENTATION & ELECTRONIC SYSTEMS DIVISION

ENGINEERING & DEVELOPMENT DIRECTORATE

MANNED SPACECRAFT CENTER

3.0 INSTRUMENTATION

The Instrumentation Subsystem consists of several major components. These are: Measurement systems, signal conditioning systems, displays and controls, caution and warning systems, timing, lighting system and the power distribution system.

3.1 Measurement System

The function of the measurement system is to sense all physical stimulus for which measurement is required, and to provide a repeatable, proportionate electrical signal which is functionally related to the variable and which can be used for indication or control or can be recorded or transmitted. Some of the measurements will be engineering measurements used to evaluate the space station or its subsystems. Other measurements will be used to indicate proper operation or status of subsystems.

The measurement system will consist of transducers to measure such parameters as temperature, pressure, flow, quantities, position, events, etc. Although the space station would require subsystems not presently used on manned vehicles, it is felt that the measurements would reduce to similar quantities. Thus, uniqueness would be manifested more in application than in physical variable.

The problems as presently forecasted are similar to those inherent in other systems; namely, those related to long life and reliability. In addition, absolute calibration schemes must be devised. New installation techniques must be developed which will allow replacement of sensors while not disrupting system operation.

Changes in size of the Space Station will primarily influence the ranges of the physical variables to be measured. A slight increase in quantity of measurements will result for larger vehicles. The resupply time will not influence the measurement system generally. However, mission time will affect calibration requirements.

3.2 Signal Conditioning System

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

The signal conditioning system will consist primarily of microminiaturized DC amplifiers. In addition, some pulse shaping, attenuators, frequency to dc, and resistance to dc converters will be required.

A few of the conditioners may possibly have to be conventional, notably phase sensitive demodulators.

A micro-miniaturized system is not presently operational and additional developmental work is required. Except for devising replacement techniques, the remaining problems in the signal conditioning system are related to reliability required for the proposed mission time.

There is no foreseeable effect on the signal conditioning equipment of space station resupply time. The effects of vehicle size would be reflected in the required number of measurements, but the difference would not be appreciable.

3.3 Displays and Controls System

The display and controls (D&C) subsystem will provide a centralized station designed to: (1) monitor the condition or status of the operational subsystems; and (2) control or alter appropriate variables as required.

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

Television monitors will be required to monitor remote systems, and to display proceedings during: (1) rendezvous with the resupply vehicle; (2) subsequent cargo disposition; and (3) EVA. Special displays will be required to monitor degradations, contaminations, etc., which will occur by virtue of extended mission times.

New designs are required which will permit servicing or replacement of components without disruption of subsystem operation. Standardizations should evolve which permit the direct interchange of various subassemblies or components when required by emergency conditions.

The size and complexity of the D&C will be only slightly affected by the resupply cycle, primarily with regard to ranges as related to status of consumables. The crew complement would also have a slight effect on the D&C, primarily due to the assumed difference in size of the station. For example, doubling the crew size should result in only a small percentage increase in D&C components.

3.4 Caution and Warning System

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

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

Speakers and display lights will be required in living and recreation areas to alert the crew to conditions which may require corrective action or other response. In a "suited-up" condition, such as during EVA, alarm tones shall be audible over headsets. Displays will be required on the main console which monitors the condition of the astronaut maneuvering unit (AMU) during EVA.

The C&W subsystem interfaces with all other subsystems and the final configuration is dependent on the mission complexity. Even so, the C&W hardware should be basic and would differ from Apollo primarily in magnitude and in types of systems monitored. It is felt, for example, that unique subsystems will evolve such as would be required for handling cargo. However, the majority of the problems will be centered around providing the necessary reliability and implementing the repair-or-replace concept.

Whether the resupply cycle is three months or six months should not alter the size or complexity of the C&W system. The basic system will not be affected by crew size. However, crew size will influence the quantity and types of remote indicators.

3.5 Central Time and Frequency Standard and Associated Equipment

This subsystem will provide the Space Station with a highly accurate time reference. Two types of time will be necessary. The first will be station master time for use by the on-board navigation and guidance subsystem and to provide other general station timekeeping. A second function will be to provide on-board experiments with time and/or interval measurements as needed.

This subsystem requires a highly accurate and reliable standard time reference standard. This time reference will be utilized in station keeping, experiment control and data annotation. To achieve this a master frequency standard will be needed with a flexible and multipurpose electronic system which will provide multiple inputs and outputs to the various units as required. A flexible programmer will be needed to activate and deactivate various on-board systems at the proper time or location.

This central timing subsystem will utilize an extremely precise rubidium frequency standard as the fundamental time oscillator. This oscillator will be variously employed to count out an interval, operate equipment, or furnish mission time as needed.

Present development status indicates that there will be no major problems of a fundamental nature in this timing subsystem. Spacecraft qualified time standards are scheduled to be available by 1969. This requirement implies extending the basic capabilities to cover more readout units, and more experimental control functions, but does not involve development per se.

3.6 Space Station Lighting

The light environment in the Space Station must be controlled to a comfortable and constant level that will allow visual acuity for controlling and operating the Space Station. The sunlight in the vicinity of the earth produces 13,600 foot candles (FC) of light during earth orbit; the suns light will be reflected by clouds covering the earth, the maximum cloud coverage would reflect 85% of the 13,600 FC or 11,650 FC. Ninety percent of the reflected light striking the Space Station perpendicular to the windows would produce 9,900 FC inside the Space Station. The light levels above are maximum and would occur only during certain attitudes. The average light levels would be less and would be controlled by the attitude of the Space Station. Therefore, the control of light entering the Space Station windows becomes a necessity and a system of light shades and filters similar to that used on the Apollo control module would be used.

To accomplish the internal lighting of the manned Space Station the following brightness values are recommended as guidelines:

Illumination Brightness Values:

Component or Area	Design
Control and Display Panel	20 FC Normal 40 FC Max
Work Areas	20 FC Normal 40 FC Max
Caution/Warning Lights	150 FC
Lighted Push Buttons	150 FC
Indicator Lights	150 FC
Ward Room	10 to 15 FC
Corridors	5 FC
Crew Sleeping Areas	10 FC Max to 0 Min
Battery Room, Storage Areas	5 FC Min to 15 FC Max

Lighting in the above areas would be accomplished by means of electroluminescente panels supplemented by incandescente lamps where required. An auxiliary emergency lighting system would be provided in all areas of the Space Station, this system would be tied into the emergency battery system and would provide illumination intensities of approximately 5 foot candles.

Additional lighting will be provided in the controls and displays area by means of flood lights directed on the console. The flood lighting system will consist of the following components: Flood light fixture, circuit breakers and dimming controls. A flood light fixture will contain two flourescent type lamps, a lense to diffuse the light, and a mount. 28 VDC will be converted to AC at 5 to 10,000 CPS inside the fixture.

Lighting controls will be placed in the area of usage or duty station on the Space Station. The lighting controls will consist of a rotary dimming switch for the primary light and a toggle switch for the secondary light.

The Manned Space Station will have an external light system consistence of the following:

1. Docking lights (running lights)
2. Rendezvous Beacon Light
3. Portable lighting

The docking lights will be similar to the running lights on an aircraft. There will be two red, two green and 4 amber lights, strategically located on the Space Station, the light intensity will be 1.4 FC and will operate on 28 VDC.

The rendezvous beacon light will be similar to the recovery beacon light used on the Apollo C/M and shall operate from the Space Station DC power. The flashing light emits a blue-white strobe once every four seconds or 15 times per minute.

External portable lighting would be in the form of battery powered lamps to be used in the inspection of shadowed areas of the Space Station on EVA.

The lighting systems herein described can be obtained with out any development or design time by using and enlarging upon the existing operational Apollo lighting systems. Total lighting system weight would be approximately 200 pounds including wiring, controls and lighting assemblies. All lighting systems would operate from 28 volts DC and would require approximately 500 watts.

3.7 Power Distribution System

The power distribution system for the Manned Space Station will provide the capability of monitoring, distributing, and controlling the electrical power required for the vehicle to perform its mission. The power distribution system will consist of twc major subsystems, AC subsystem and DC power subsystem. Both will be designed to provide for the maximum electrical power requirement of the Space Station plus a 50 percent overload for emergency or contingency. The electrical system will provide power for the following:

Lighting:	Both internal and external in operating and emergency modes
Power:	Operate station life support systems
	Operate station communications
	Operate station instrumentation
	Operate station data system
	Provide experiment monitor power
	Provide power for experiments
Emergency Power:	Provide emergency power for life boat launch, communication, etc.

Space Station AC power will be provided throughout the mission by eight solid state static inverters, five of the inverters will be in use at all times with three inverters for back-up. The AC power subsystem will comprise approximately 45 percent of the total space station power requirement. Two inverters, operating at 76.5 percent efficiency, could normally provide for all AC loads for the space station housekeeping requirements and three inverters would provide the power required for the experiment.

Each inverter can produce 115/200 volt, 3 phase, 400 cycle power at a maximum output of 1250 volt amps. Voltage regulation, current limiting during overload, and automatic inverter/bus disconnect in the event of over-voltage or extreme overload is also provided in the inverter and its control circuits.

Basic AC distribution would be accomplished with a four wire system via two redundant buses, space station AC loads would be powered by either bus as selected at the station power distribution panel.

DC power distribution would be accomplished with a two wire system via a series of interconnected buses consisting of the following:

1. Two redundant main DC buses powered by the electrical power source and auxiliary batteries.

2. Two battery buses, each powered by its individual auxiliary battery.

3. A non-essential bus powered through either main DC bus

4. A battery relay bus powered by the auxiliary batteries

through the individual battery buses and isolation diodes.

5. Pyro buses powered by the pyrotechnic batteries and divorced from the main electrical power system.

All critical loads will be connected directly to both main DC buses through isolation diodes, or can be transferred from one bus to the other by bus selection switches.

Any failures in the system would in most cases be compensated for by redistribution of the bus loads. The auxiliary battery power would be manually connected to both main DC buses upon detection of an under voltage condition.

The power distribution system would have one common grounding point on the Space Station structure. All negative DC buses and the AC neutral buses would be connected to this point, this would primarily eliminate ground loop effects.

Sensing and control of the DC power subsystem would consist of DC sensing circuits provided to detect under voltage, overload and reverse current conditions, and to alert the Space Station personnel. Overload sensing circuits would be used to protect the electrical power source and reverse current sensing would be provided to reveal reverse current flow resulting from the electrical power source failure. In instances when overload or reverse current occurs, disconnect motor switches would be automatically activated by the sensing circuitry, under voltage and electrical power source disconnect would be indicated by illumination of a caution lamp on the station power distribution panel.

AC sensing circuits would be provided to reveal inverter malfunctions and provide a warning indication. Disconnect motor switches would be automatically activated when an overvoltage or overload occurs. Caution indicators would be provided on the power distribution panel to indicate overvoltage and overload as well as undervoltage.

The Space Station power distribution panel would contain such switches, meters, gauges, indicators, lights, and annunciators as necessary for monitoring and controlling all facets of the power distribution system.

The power distribution system will use space qualified components as are now in use in the Apollo program. These items consist of:

1. Solid State Inverters
2. Batteries (Auxiliary Power)
3. AC Control Boxes
4. DC Control Boxes
5. Fuse and Breaker Panel
6. Motor Switch Assemblies

7. Relay Assemblies
8. Electrical Controls and Displays
9. Wiring Harnesses (Connector and Cables)
10. R.G.S. Sequencer (other sequencers as may be required)
11. Battery Charger

All of the systems are operational. But product improvement and system reconfiguration will be required for use in a Manned Space Station. Resupply of the power distribution would be required on a yearly basis and would consist of replacing the following:

1. Auxiliary Batteries
2. AC Inverter
3. Battery Charger Components
4. Control and Display Actuators and Inductors

Of special importance is the requirement for charging batteries in the event solar cells are used for an electric energy source. Since the space station will be in sunlight about 50% of the time, batteries must be used to power the station during the portions of the orbits which are in darkness. This, in effect, doubles the power requirement for the solar cells since it must charge batteries as well as supply power to the connected load during the sunlit portion of the trajectory.

3.8 INSTRUMENTATION SYSTEM WEIGHTS, VOLUMES AND POWER CONSUMPTIONS

Instrumentation system weights, volumes and power consumptions are presented in Table 3.1.

INSTRUMENTATION WEIGHT, VOLUME, AND POWER

TABLE 3.1

EQUIPMENT		WEIGHT (lb.)	VOLUME (in 3)	POWER (watts)
A.	INSTRUMENTATION SUBSYSTEM			
I.	629 - Measurement Transducers	80	650	80
II.	270 - Signal Conditioners	20	300	50
III.	1 - Display and Control System	240	8300	263
IV.	1 - Caution and Warning System	15	350	18
V.	1 - Timing Equipment	35	822	38
VI.	4 - Event Timers	48	1400	96
VII.	1 - Lighting System	50	1000	800
VIII.	Electrical Power Distribution and Sequencing (for assumed 15 KW load)			
	Station Wiring*	3300	10,000	
	5 - Inverters	240	6,000	
	3 - Voltage Regulators	75	1,400	
	3 - Battery Chargers	350	8,650	= 85%
	Batteries (15 KW)	1100	25,000	
	Synchronizer	12	760	
	Controls	100	1,700	
	Sequencer	40	850	
	Sub Total	5,217	54,360	
Instrumentation Totals		5,705	67,182	1345

*Based on Apollo CSM weight of 220 lb/connected kw.

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SECTION 4.0

COMMUNICATIONS

INSTRUMENTATION & ELECTRONIC SYSTEMS DIVISION

ENGINEERING & DEVELOPMENT DIRECTORATE

MANNED SPACECRAFT CENTER

4.0 COMMUNICATION SYSTEMS

4.1 General Considerations

The basic Space Station Telecommunication and Tracking System would be similar in design to the Apollo system. Additions will be required such as a modified Apollo S-band transponder for use as an alternate radar, a wideband S-band transmitter to transmit photogrammetric information, a hard copy printer for receiving non-critical information, a data fax transmitter for transmitting handwritten experiment notes and possibly a TV receiver and monitor for entertainment of crew. An efficient data management system will also be required.

The Unified S-band System would use a baseband combination of pseudorandom ranging code (PRN), telemetry, astronaut voice and bio-medical data phase modulating a coherent replica of the uplink carrier for Space Station to ground communications and tracking. The Apollo type FM transmitter can be used to transmit TV or high rate experimental data (500 KC analog or 1 megabit digital) on a time-shared basis. The Apollo high power amplifiers would be changed from vacuum tube to solid state devices to provide high reliability over long periods of time.

If an additional transmission facility is required for high data rate experimental data and high resolution TV signals, a separate solid state 20 watt FM transmitter would be employed. This is recommended for reliability and ease of tailoring the system to fit the mission. The number of FM channels would be determined by the experimental data load.

Circuit margin calculations show that omni antennas can be used acceptably for the 260 N. M. orbit. The use of the LM high gain antenna is recommended for use in synchronous orbit; however, the use of a directional antenna will present an attitude stabilization constraint on the Space Station due to the requirement of pointing the high gain antenna towards the earth. Location of the antenna on the counter rotating hub of the Space Station will require either location of the S-band equipment in the hub or slip rings.

With both 260 N. M and synchronous orbits, the circuit margins would be entirely adequate to increase the Apollo PM 51.2 kbs TM channel to 100 kbs or greater. This may necessitate changing the present TM and voice subcarriers, however. At both 260 N. M. and synchronous altitude, the TV channel circuit margin would support commercial quality TV.

Ground to Space Station communication would combine command voice, and PRN signals phase modulating the uplink S-band carrier.

A standard LM configuration Telecommunications and Tracking System is recommended for the supply vehicle. The addition of a mode to the Space Station S-band transponder permitting inverse ratio transmit and receive operation will permit spacecraft-to-spacecraft communications plus a backup mode to the rendezvous radar. The primary rendezvous radar would be an updated Gemini radar with one transponder on the spacecraft. This radar is preferred to the Apollo radar because of its simplicity and because a steerable radar antenna will not be required on the Space Station mission.

Uplink TV for entertainment of the crew at synchronous altitude would be comparatively simple to implement utilizing the present S-band ground stations and a separate TV receiver. TV reception by a 260 N. M. orbiting Space Station is not considered practical for entertainment purposes due to numerous interruptions in handover between grouns stations and nonoverlapping ground station coverage. This difficulty can be overcome by relay from a synchronous orbit satellite. However, the use of such a relay imposes a major antenna problem on the Space Station. This problem is one of keeping a four foot parabolic antenna aligned with the satellite while the Space Station is spinning on an axis aligned with the sun. Also, a steerable 30 foot antenna will be required for the satellite. Planning for the ATS4 Communication Satellite incorporates such an antenna.

4.2 Circuit Margins

Circuit margin calculations for the down link S-band FM mode are shown in Table 4.1. This mode is of particular importance since it is the mode with the least margin. Calculations show the link to be adequate for transmission of commercial broadcast quality TV.

Table 4.2 shows calculations for the Satellite-to-Space Station entertainment TV link. An x-band frequency is recommended due to the benefit obtained from the extra gain of the transmitting antenna.

It can be seen from the TV circuit margin calculations that a video bandwidth of only 2 MC is transmitted. While this is only one half the bandwidth transmitted by commercial TV stations, it should be adequate for entertainment.

4.3 Antennas

The rotating Space Station imposes a particular problem for S-band antennas. To minimize masking, the placing of two omni antennas on separate booms extending five feet from the crew compartment is recommended. The antennas must be automatically switched in accordance with which antenna is receiving the greatest signal strength from the uplink. However, S-band reception is accomplished by means of a narrow band phase lock loop and switching of the antennas may produce transients which cause the loop to break phase lock and consequently interrupt its operation. This area must have further investigation.

TABLE 4.1

Circuit Margin Calculations for FM (Down TV) Mode - S-band

Altitude = 300 n.m.

Maximum slant range from ground station to S/C, 5° elev. angle = 1200 n.m.

Frequency = 2300 M.c

B = F. M. Modulation Index = 3

Video B. W. = 4 Mc

Transmitter Power (20 watts)	13 dbw
Spacecraft Transmission losses	-5
Spacecraft Omni Antenna Gain	-3
Ground Station Transmission losses	-1
Ground Station Antenna Gain (30')	44
Space loss	-167
Ground Station Noise Spectral Density	208 dbw/cps
AM Equivalent B. W. (2 x 4 mc)	-69 db
FM Gain ($3B^2$)	14
Resulting Video RMS Signal to RMS Noise Ratio	34 db

FM Threshold Tests:

Noise Bandwidth = 2 (B+1) (B.W.) = 2 (3+1) (4Mc) = 32 MC	-75
Minimum Signal to Noise Ratio (-)	6
Circuit Margin	+8 db

TABLE 4.2

Circuit Margin Calculations

for

Synchronous Satellite to Spacecraft for Entertainment TV

Altitude = 300 N. M.

Range from Satellite to S/C at earth's link = 23,000 N. M.

Frequency = 10 Kmc

B = Modulation Index = 3

Video BW = 2 Mc

Transmitter Power (10 watts)	10 dbw
Satellite Transmission losses	-3
Satellite Antenna Gain (30')	56
Space loss	-205
Spacecraft Antenna Gain (4' antenna)	39
Spacecraft Noise Spectral Density	190 dbw /cps
Spacecraft Transmission Losses	-3
AM Equivalent B. W. (2 x 2 MC = 4 MC)	-66
FM Gain ($3B^2$)	14
Resulting Video RMS Signal to RMS Noise Ratio	32 db

FM Threshold Test:

Noise Bandwidth = 2 (B+1) (B.W.) = 16 MC	-72
Minimum Signal to Noise Ratio (-)	10
Circuit Margin =	+2 db

Table 4.2 shows that a four foot parabolic is necessary for reception of entertainment TV from a satellite. The antenna tracking system developed for Apollo can probably be modified to enable its uses in tracking a station emitting an FM signal. A beacom must also be incorporated with this antenna to enable the communications satellite 30 foot antenna to track the space station.

4.4 Extravehicular Activity: Communications

Voice and biomed will be handled by the standard Apollo EVA communications system. By use of demodulators in the Space Station, the biomed data can be placed on PCM or examined on biomed console displays.

EVA TV is generated by a modified Apollo TV camera and transmitted to the Space Station. The camera has been fully qualified for such use. The EVA TV transmitter is presently under development with support funds and an in-house development effort is underway to provide a prototype receiver.

The EVA type of communications system can also be used for communication with the rotating hub.

4.5 Data Management

Infrequent earth-space station contact in a 60° orbit makes a very efficient data management system (DMS) necessary. The DMS will consist of the equipment necessary to receive experiment and housekeeping sensor outputs and efficiently process, sort, select, format, program, route, control and/or display these data.

The amount and diversity of activities to be performed by a space station makes the use of standard telemetry techniques impractical as well as inefficient. For example, Apollo with a three-man crew, requires a transmission capacity of 51.2 kbs. The space station with a 9 to 27-man crew will obviously require a significant increase in the number of channels to be measured. The DMS must provide an integrated relationship of all activities to perform routine functions, handle tremendous quantities of data, perform rapid calculations and operations, and monitor, interpret and control the system, simultaneously. These functions are beyond the capability of unaided crew personnel. The DMS must provide operation where man and machine will be brought together for man's positive override control.

The data requirements will change as a function of mission phase, i.e. launch, drifting orbit, abort, etc. The function of the DMS will provide only pertinent data dependent on mission phase and crew safety for display to the crew.

Priorities will be established regarding criticality of various types of housekeeping information. Several orders of priority must be established to deal with the occurrence of malfunctions that will initiate either manual or automatic change in the sequence of operations.

Prior to transmission of data to outside sources, the establishment of data format is accomplished. This includes assembling the data in correct sequence, as well as into predetermined record lengths with identification, time, and redundancy coding, if required. Priorities of data will be selected for transmission, eliminating the transmission of channels yielding little or no information.

Figure 4.1 is a functional flow diagram of a typical data management system.

4.5.1 Data Acquisition Unit (DAC) - The DAC will have the following capabilities:

a. Accept analog, digital serial and discrete inputs.

b. Digitize the analog inputs.

c. Output a PCM data train for backup S/C to ground data.

d. Accept a request from the Digital Processor and Controller (DPC) and grant valid data to the DPC.

e. The DAC must be programable so that relative mission phase data need only be sampled.

4.5.2 Digital Processor and Controller (DPC) - The DPC will have the following capabilities:

a. Accept data from the DAC and the Data Conditioning Unit.

b. Perform data compression for transmission to the ground network.

c. Perform data reduction on required channels for display to the crew.

d. Provide for priority interrupt on critical crew safety items.

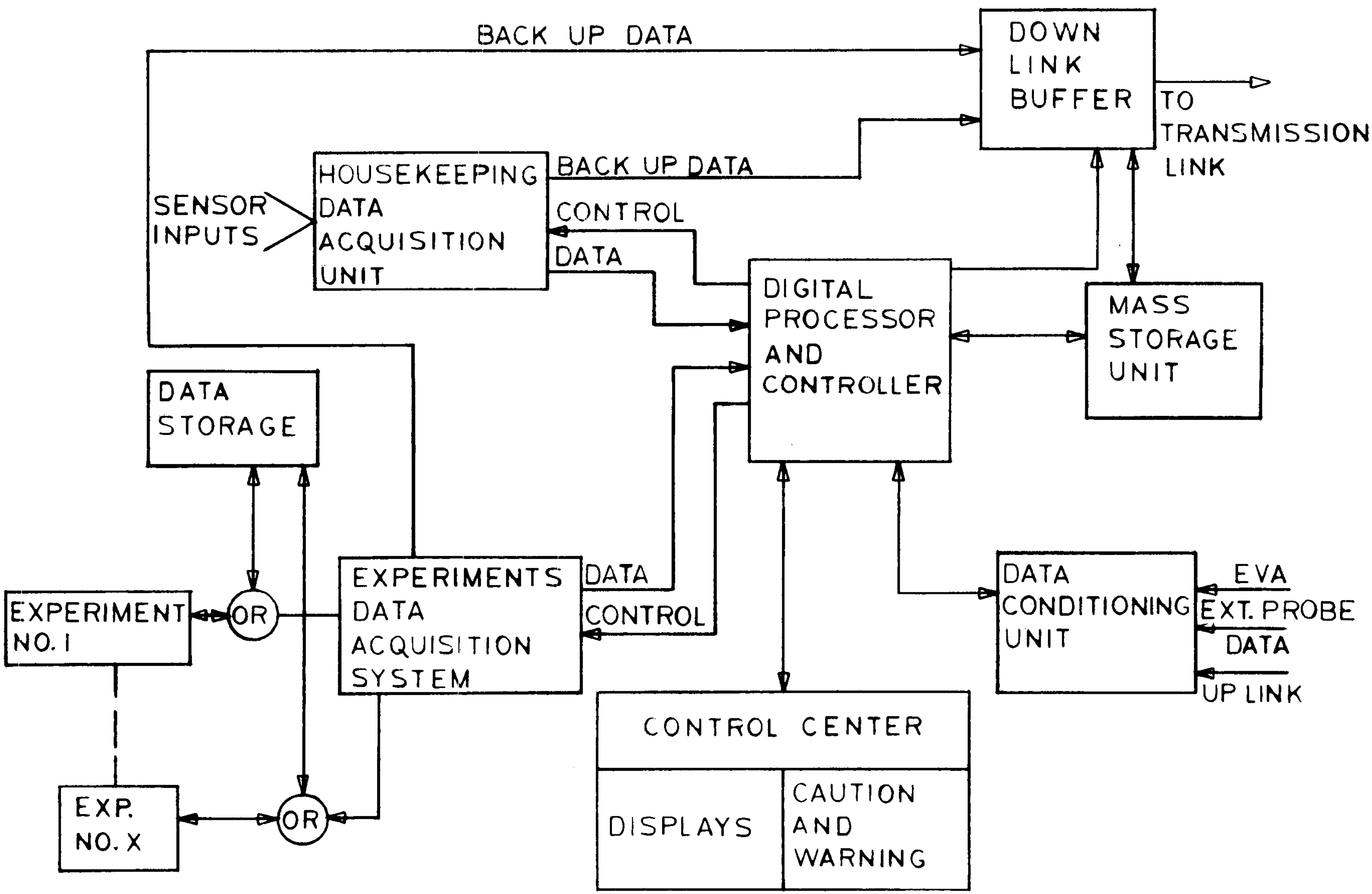
e. Experiment control.

4.5.3 Control Center - The Control Center will provide a central location for crew control of the DMS and information presentation from the DMS to the crew for decision.

4.5.4 Data Conditioning Unit - The Data Conditioning Unit will have the following capabilities:

a. Regeneration and Bit Synchronization from external probe type PCM data.

FIGURE 4.1 FUNCTIONAL FLOW DIAGRAM MANAGEMENT SYSTEM



b. Data conditioning of biomed EVA data.

c. Up-data link interface.

4.5.5 Downlink Buffer (DLB) - The Downlink Buffer will transfer synch. data from the DPC to syncnronous data for transmission to ground network. It will also provide a backup transmission mode by time-sharing of the transmission link with experimental and housekeeping data from DAC.

No significant technological advancements are foreseen to develop the DMS. However, a lead time required for the finished product is estimated around $4\frac{1}{2}$ years at a cost of approximately 17 to 25 million dollars dependent on reliability requirements.

4.6 Data Storage System

The data storage subsystem on the manned Space Station is of primary importance to the applications of such a facility. Two basic types of data will be collected and handled. The first of these will be of a station housekeeping and monitoring nature. The second type of data will pertain to experiments and/or scientific data. This scientific data will be stored primarily in the form of either photographic film or magnetic tape. The photographic film, which at the present time allows the highest data density, will be used and analyzed ont he ground after recovery by the resupply vehicle and will not be discussed here because the specific experimental envelope determines the quantity and quality of photographic data required.

The magnetic tape data storage requirement is to provide a highly reliable, adaptive, and flexible on-board facility. Magnetic tape is an attractive storage media because it can be easily examined and verified during playback by scientist-experimentor on-board the Space Station to help him with his work, data can be easily and rapidly retrieved through an appropriate down data link in those cases where the station application requires it, e.g. meteorology, and advanced recorder state-of-the allows wide variations in type and format of data. The most demanding magnetic tape storage requirements is the scientific data where very high bandwidth information is anticipated from selected sensors. The Space Station recorder subsystem could be designed to accomodate analog signal information bandwidths up to at least 5 Mc. The total data storage capacity of the station will depend on the allocation of weight and volume for this purpose as determined by the experiments, the data down-link system, and the resupply plan adopted.

Some of the scientific experiments, e. g. biomedical, have low information bandwidths but require very high resolution to detect a change of significance. This requires a different type of recording hardware and associated instrumentation and in general, is best handled by sampling, converting, and digitally recording the data directly.

Some experiments produce data intermediate of the two systems described so a third magnetic tape recorder capability of a more conventional nature will be included to collect this data. The main function of this facility will be to augment the data management system to optimize the storage volume available or the telemetry down-link bandwidth.

The data recording subsystem will consist of the following units:

4.6.1. Video Bandwidth Recorders: Two operating high bandwidth recorders will be needed. One to accumulate data and a second to monitor, verify, manage, and optimize the on-board data to achieve maximum utility from the on-board storage allotment or the down-data link. There will be one complete spare recorder which can be used in the event one of the operational units fail. It will be physically stored adjacent to the operational units and readily available for use.

4.6.2 Multichannel, Variable Speed, Wide Bandwidth Recorders: The same equipment complement as discussed above will be used for these units.

4.6.3. Digital Recorder: The same equipment complement as discussed above will be used for these units.

4.6.4. Portable Recorders: In connection with Extra-Vehicular Activity (EVA) and peripheral scientific data requirements a portable tape recorder complement will be included in the data storage subsystem. The portable tape recorder system will consist of two separate but compatible instruments. These instruments are:

(a) A multichannel miniature, portable, cartridge loaded record unit which may be carried by the astronaut during EVA or utilized by the experimenter as a peripheral recorder for experimental data.

(b) A reproduce unit which will be located in the vehicle. This unit will be utilized for the reproduction of the data from the record unit. The portable recorder system will record and reproduce digital and analog data with DC capability, long time record capability, portability and cartridge loading.

There are no outstanding problems currently anticipated which would cause difficulty in the 1973 time period. The most important advantage would derive from developing low power consumption recorder systems of the type described.

The anticipated operational life of the recorder units is one year at which time they should be replaced. In the development and design to meet the stringent space station reliability requirements, these recorders should utilize the same spooling, power racking, and accessory fitments were possible.

The experimental envelope and the resupply cycle have great impact on the data storage system. The number of crew members has no particular significance.

4.7 Television

The television subsystem will consist of TV monitors, cameras, video recorders, and use of the spacecraft S-band transmitter and receiver on the wide band RF transmission system. The equipment descriptions outlined below apply to qualified hardware available by 1969. For example, hardware development contracts for the Apollo camera transmitter, video recorder, and video monitor are in the terminal stages at this time. The Apollo camera qualification test will be complete by December 1966. High resolution camera for in-cabin applications are available, requiring only space qualification.

4.7.1 Camera

Two TV camera types shall be used to provide complete video data coverage. These are a modified Apollo camera and a high resolution camera. The modified Apollo camera can be used on EVA, providing a high resolution low frame rate picture or it can be used with the standard Apollo format. This camera is completely space qualified for the lunar surface environment. A transmitter pack will be added to the camera to provide a completely portable handhold unit unrestricted by cables, etc. The unit will be capable of 8 hours operation on one battery charge. This slow scan camera will primarily be used for EVA video coverage. In addition, its low bandwidth requirements provide excellent use for a general video link to earth not requiring high resolution or fast motion rendition. The camera will monitor EVA activity to provide the space station manager cognizance of any emergency conditions.

The high resolution camera will be used to obtain high resolution pictures inside the space station of experiments requiring visual observation. In addition, the crew can be monitored while performing duties in remote cabins or through windows while performing extravehicular activities.

The high resolution cameras combined with the video monitors form a closed circuit television system providing an additional crew safety factor and experimental operation efficiency. The system will provide the space station manager as well as earth monitors with a video presentation of selected areas throughtout the station.

Cameras and monitors shall be strategically located in the space station. The complete system shall be designed to operate at standard broadcast television rates. Therefore, the monitors will be used as the up-data video link for educational training and recreational viewing. The cameras will be used for down link video for ground controllers and public viewing. All cameras and monitors shall be semi-portable in that hookup to the closed circuit system can be accomplished at various locations throughout the space station.

The modified Apollo camera is 200 in.^{3}, 12 lbs. and operates at 10 watts. The high resolution camera is 200 in.^{3}, 10 lbs. and operates at 10 watts. Their specifications are:

TV Camera (instrument high resolution)			Apollo
Frame rate 30 cps			10 or .625 F/S
Lines/frame 525			320 or 1280
Interlace Ratio 2/1			1/1
Video Bandwidth 5 Mc			500 kc
System Resolution:	horizontal	400 TV lines	220 or 500
	vertical	350 TV lines	220 or 500
Camera S/N 35 db minimum			35 db

4.7.2. Receiver

The spacecraft video receiver is used to receive video in the spacecraft when the camera is used during EVA. The received video can then be transmitted on to earth, recorded on the video recorders, or displayed on the spacecraft TV monitor. The receiver is 20 in.^{3}, 2 lbs. and operates at 4 watts.

4.7.3. Recorder

The wideband video recorder capable of up to 4 mc response will be used to record video phenomena when the spacecraft is out of range of tracking stations and dump the stored data when the spacecraft is over a tracking statinn. The system is capable of an 8:1 record playback ratio for video information recorded up to 500 kc.

The recorder is a two-speed device with 4 hours of record capability for up to 500 kc video and $\frac{1}{2}$ hour record capability for up to 4 mc video. The system is 850 in^3, 26 lbs. and operates at 50 watts. Its electrical specifications are:

a. Bandwidth - Dc to 4 mc

b. Record Capability - 30 min. @ bandwidths of 500 kc to 4 mc; 4 hrs. @ bandwidths below 500 kc.

c. S/N - 35 db.

4.7.4. TV Monitor

The spacecraft video monitor can be used to assist the astronaut in the operation of television and scanning type instrumentation such as spatially scanning IR and UV sensors. In addition, the monitor can be used as an up-data video link to present training information, etc., from the ground to the astronauts as well as recreational TV programs.

The monitor is 600 in.^{3}, weights 20 lbs. and operates at 20 watts power maximum.

Those system specifications of specific interest and which describe the operational capabilities of the video monitor include operation in two modes, either as an image display (TV) or as a signal display (A-scope) and include maximum-minimum limits on certain parameters as listed below:

a. Ambient Illumination — 1 to 2 ft. L

b. Gray Scale — 6 uniform steps minimum

c. Resolution

   TV lines — 1000 at 20% mod

   Spot Size — 1200 vertical lines

d. Deflection

   X - 0.5 cps to 20 Kc, 1% linearity

   Y - (image) - 0.625 cps to 60 cps, 2% linearity

   Y - (signal) - dc to 1 mcps, 1% linearity

e. Display storage - 1.6 sec. min.

f. Power - 20 watts maximum

g. Weight - 20 lbs. maximum

h. Volume - 600 cu. in. (maximum)

The video monitor as presently designed will display TV signals of either the Block I or Block II Apollo format and standard EIA television. The signal display mode provides a device capable of operation as a laboratory oscilloscope for waveform monitoring.

4.8 COMMUNICATION SYSTEMS WEIGHTS, VOLUMES, AND POWER CONSUMPTIONS

Communication systems weights, volumes, and power consumptions are presented in Table 4.3.

Table 4.3

EQUIPMENT WEIGHTS, VOLUMES, AND POWER CONSUMPTION

EQUIPMENT	WEIGHT (lb.)	VOLUME (in 3)	POWER (watts)
A. COMMUNICATION SYSTEMS			
I. RF SYSTEMS			
1 - Unified S-band Transponder	30	1420	38
1 - Inverse Ratio S-band Transponder	30	1420	24*
1 - Dual S-band Power Amplifier	32	768	180
1 - 20 watt S-band Wide Band Transmission System	40	1500	75
4 - VHF Transceivers (including 2 for hub rotating interface)	56	1728	122*
1 - Up-Data Link Receiver	20	918	15
1 - TV Receiver	10	500	15
2 - S-band Omni Antennas and Supporting Booms	10	100**	
1 - Omni Switching System	10		10
1 - X-band Four Foot Parabolic Antenna (TV Reception)	40	9000**	50
3 - EVA and Rotating Hub VHF Antenna	6	2000**	
1 - Radar Transponder	10	300	50
1 - Radar Antenna	4	50**	
Sub Total	298	19,704	579

Table 4.3 (Cont'd)

EQUIPMENT	WEIGHT (lb.)	VOLUME (in 3)	POWER (watts)
II TERMINAL EQUIPMENT			
1 - Data Fax Set	10	300	15
1 - Hard Copy Printer	8	280	4
Sub Total	18	580	19
III DATA MANAGEMENT SYSTEM			
2 - Data Acquisition Unit	30	800	30
1 - Digital Processor and Controller	60	1700	160
1 - Data Conditioning Unit	27	500	14
1 - Down Link Buffer	23	300	10
1 - Control Center	20	2000	100
Sub Total	160	5300	314
IV DATA STORAGE			
2 - Digital Recorders	200	4000	170*
2 - Video Bandwidth Recorders	120	2400	150*
2 - Wide Bandwidth Analog Recorders	200	4000	170*
Portable Recording Systems consisting of: 2 - Record Systems; 2 - Reproduce Systems; 2 - Battery Packs; 6 - Electronic Modules; 40 - Tape Cartridges	84	1000	100*
Sub Total	604	11,400	590

Table 4.3 (Cont'd)

EQUIPMENT	WEIGHT (lb.)	VOLUME (in 3)	POWER (watts)
V. AUDIO AND PREMODULATION PROCESSING			
Audio Center for 9 men	24	750	50
2 - Premodulation Processors	50	1200	20*
Microphones and Headsets (9 men)	4	60	
Sub Total	78	2010	70
VI TELEVISION			
4 - Slow Scan EVA Camera and Transmitter	48	800**	40*
6 - High Resolution Camera	12	168	42*
8 - Video Monitor	160	4800	160*
2 - Video Recorder	52	1700	100*
4 - Video Receiver	8	80	16*
Misc. Cabling/switches, etc.	50	1000	--
Sub Total	330	8548	358
Communication and Tracking Systems Totals	1488	47,542	1930

*Includes duplicate units which may not be operated simultaneously.

**Located outside the spacecraft.

NOTE: A demand factor must be applied to all communication and tracking subsystem loads as well as to all other subsystem loads.

PRELIMINARY TECHNICAL DATA

FOR EARTH ORBITING SPACE STATION

VOLUME III

SYSTEMS

SECTION 5.0

CRYOGENIC STORAGE

PROPULSION AND POWER DIVISION

ENGINEERING AND DEVELOPMENT DIRECTORATE

MANNED SPACECRAFT CENTER

5.0 CRYOGENIC STORAGE SYSTEM

5.1 SUMMARY

This paper presents the results of a study to define the methods of cryogenic storage for Manned Earth Orbital Space Station and Mars Flyby Missions. State-of-the-art thermal performance is reviewed and compared with the requirements to accomplish the Space Station and Mars Flyby Missions. Where the present insulation technology fails to satisfy the long term storage requirements, alternatives are considered. The thermodynamic advantages of subcritical and high pressure supercritical storage are presented.

The study indicates that the degree of improvements expected in static insulation concepts are not expected to be sufficient to meet the long term cryogenic storage requirements if current design environment temperatures are maintained. It is shown, however, that by lowering the vessel environmental temperature ($0^{o}F$ to minus $100^{o}F$ range), present insulation techniques will satisfy the thermal performance requirements for the Manned Space Station as well as the Mars Flyby Mission.

The diluent gases considered are nitrogen, neon and helium. Nitrogen will result in the highest diluent gas weight penalty and helium, due to its low molecular weight, will result in the lowest weight penalty. Neon, with a liquid density of 77 pounds per cubic foot, possibly could be used in a vessel designed for oxygen which has a density of 71 pounds per cubic foot. Neon is not presently available in the quantities required and the liquid would probably be significantly more expensive than either helium or nitrogen.

5.2 INTRODUCTION

Fuel cell reactants, metabolic oxygen and diluent gases may be stored at high pressures and ambient temperatures, or cryogenically in the subcritical or supercritical state. The latter approaches result in the lightest system weights unless the mission time is quite short. Supercritical storage has been successfully used in the Gemini and Apollo programs to store both hydrogen and oxygen.

Fluid flow from any storage system results in energy removal from the system. For constant pressure operation the energy, which is removed with fluid flow, must be replaced. The amount of energy which is removed from a cryogenic storage system is strongly dependent upon the storage pressure. Where missions impose long term non-venting storage requirements, the storage pressure should be optimized with the thermal protection system.

5.3 MISSION CONSIDERATIONS

5.3.1 Design Reference Missions

Crew size, mission objective, and mission duration are envisioned to vary widely. It is therefore necessary to consider in detail several design reference missions. These design reference missions are considered to encompass the full range of cryogenic fluid requirements.

The design reference missions considered in this study are as follows:

24 Man Space Station	6 Months and 24 Months
9 Man Space Station	6 Months and 24 Months
5 Man Mars Flyby	700 Days
3 Man Earth Synchronous	6 Months and 24 Months

Depending upon whether or not a re-supply of the spacecraft(s) is required, the mission durations are anticipated to be six (6) or twenty-four (24) months for the Earth Synchronous or Space Station missions. The Mars Flyby mission is anticipated to require 700 days.

5.3.2 Fluid Requirements

Figures 5.2 through 5.9 show oxygen and diluent gas requirements as a function of mission time for the design reference missions considered. The diluent gases considered are nitrogen, neon, and helium. The fluid requirements were determined from the anticipated leakage rates (suits, cabin, and plumbing) and metabolic oxygen usage. For this study, nitrogen leakage is assumed to be 50 percent of total leakage and Ne and He leakage rates are based on the ratio of their molecular weights to the molecular weight of nitrogen. The system weight can be reduced significantly if spacecraft leakage rates are controlled more closely. The leakage rates considered are as follows:

Mission	Fluid, Pounds Per Day			
	O_2	N_2	Ne	He
3 Man Earth Synchronous	7	7	4.4	0.875
5 Man Mars Flyby	9	9	5.62	1.125
Space Station	27	27	16.9	3.375

Metabolic oxygen rate, 2 lb/man-day

5.3.3 System Sizing

Figure 5.25 shows O_2, N_2, H_2, He and Ne fluid weights as a function of outer shell diameter for varying L/D ratios. The data for developing

Fig. 5.25 is based upon a 95 percent fluid fill density and a 1.25 inch annulus between the pressure vessel and vacuum shroud.

Figure 5. 26 shows pressure vessel surface area as a function of outer shell diameter and L/D ratios. Data for developing Figure 5.26 is also based upon a 1.25 inch annulus.

5.3.4 Resupply

In-flight resupply can be accomplished by replacing depleted vessels with filled vessels from a logistics vehicle, or by fluid transfer from the logistics vehicle to the spacecraft. The latter method may permit the use of lower cost vessels for logistics and perhaps would be economically advantageous.

Two possibilities to facilitate transferring liquids in-flight are:

- Artificial gravity
- Positive expulsion

A major problem in cryogen transfer thus far is venting due to rapid boiling of liquid during transfer. If the liquid-vapor interface is not controlled, all the liquid could be exhausted through relief valves. This problem can be handled by expanding a portion of the supply liquid through a valve and using the refrigeration effect to subcool the receiving system and transfer lines sufficiently to lower the vapor pressure and thus eliminate boiling which is the source of the problem. The necessary components for this scheme have been developed and flown in MSC experiment No. 13. The arrangement is shown schematically in Figure 5.10. Positive expulsion schemes are shown schematically in Figure 5.11.

5.3.5 Quantity Gauging

Present methods of quantity gauging (QG) can be applied to the supercritical storage system. Quantity gauging system accuracy requirements should be critically scrutinized for large space station applications. Some less accurate, less expensive means of QG may be acceptable for these applications. MSC experiment No. 13, with prior ground tests, proved the matrix capacitance gauging approach to be acceptable for small systems. The dynamics and weight characteristics of this approach should be evaluated for application to the larger sizes.

5.4 STATE OF THE ART

Cryogenic oxygen and hydrogen stored at supercritical pressures has been successfully used in the Gemini and Apollo programs and is in the latter stages of development for a Lunar Module (LM) helium pressurization system.

This method of storage was selected for Gemini and Apollo due to low weight and relatively low development risk. The single phase supercritical fluid minimizes pressure control problems, and fluid quantity is gauged with a capacitance probe.

Internal pressurization heater configurations differ for the Gemini and Apollo systems. Concentric spherical heaters were used in the Gemini vessels while electric fan-heaters are used in the Apollo vessels. The fan-heater provides faster response and better thermal equilibration than the static spherical heaters and thus should prove more advantageous from a pressure control standpoint; however, motor problems still exist in the Apollo program and further development will be required for long duration missions.

5.4.1 Insulation

The Gemini cryogenic vessels are insulated with aluminized mylar radiation shields which are suspended in a vacuum annulus between the pressure vessel and outer vacuum shroud. The inner vessel is supported with compressed fiberglas pads. Six different sized cryogenic vessels were developed for the Gemini program to supply oxygen and hydrogen for two and fourteen day missions. Thermal requirements were based on non-venting standby or minimum flow.

The Apollo Block I hydrogen and oxygen and Block II oxygen vessels are insulated with load bearing insulation made up of alternate layers of aluminum foil and dexiglas paper spacer material. The Apollo Block II hydrogen vessels have a somewhat improved insulation and support scheme. The pressure vessel is supported by three (3) straps of alternate layers of foil and spacer material. The radiation shields in the Block II hydrogen systems are aluminized H-film. Both the Block I and Block II systems have vapor cooled shields suspended within the annular space.

Several prototype systems have been developed by the Bendix Corporation under contract to MSC, Houston. These systems all have discrete radiation shields suspended within the annular vacuum space. The shields are supported with teflon snap-spacers and the pressure vessel is supported with glass-filled teflon bumpers.

The cryogenic LM helium system uses an insulation and support scheme similar to that used in Gemini vessels. The performance appears to be satisfactory for the LM requirement.

The thermal performance of average vessels from both the Gemini and Apollo programs has been marginal with several unable to meet specification requirements.

Vessels using discrete shields, developed by MSC under R and D contracts, have demonstrated many advantages over systems containing laminar insulations. Assembly time is considerably less, the units have a clean annulus, better vacuum life characteristics are exhibited, and thermal performance is equal to or better than those systems using laminar insulations. Nominal performance data from all the systems discussed is shown in Figure 5.1.

Since introduction of the multiple layer radiation shielded insulation concept, the most significant advancement in thermal protection schemes has been the introduction of vapor-cooled shields. No other really significant improvements in insulation performance have been noted.

The performance trend indicates gradual improvements primarily in the area of insulation application. It is believed that the potential for further major breakthroughs in insulation thermal performance for cryogenic vessels has diminished to an extent that such developments should not be anticipated for application to any new cryogenic program starting in the next year or two.

5.4.2 Materials and Fabrication Techniques

Materials for the Gemini and Apollo cryogenic pressure vessels have been Inconel 718 for oxygen, and titanium 5 AL-2.5 Sn for hydrogen. The Gemini pressure vessels are spheres made of hydroformed or deep drawn hemispheres. The Apollo pressure vessels are spheres fabricated from forged and machined hemispheres. Forging and machining have proved to be expensive processes of fabrication and the titanium forgings presently require more than one year lead time. The forged and machine hemispheres are not considered to be of better quality than hemispheres made to the same dimensions from rolled sheet stock by hydroforming, spinning, deep drawing or hydraulic bulge forming. In fact, any of the forming processes which start with rolled sheet stock and introduce additional material cold working during hemisphere forming, should produce as good or better pressure vessel for dewars compared with any other fabrication process.

MSC has funded the development of pressure vessels with Arde', Inc., and the Bendix Corporation. The Arde' process requires a preformed pressure vessel of 301 stainless steel which can be fabricated from any of the above mentioned processes. The pressure vessel is then pressurized and cold worked with liquid nitrogen (LN_2) at minus 320°F. The additional cold working with LN_2 produces a material strength to density ratio which is slightly better than titanium 5 AL 2.5 Sn. The material is compatible with all cryogens as well as N_2O_4. The Arde' process should not be confused with fabrication processes in general. It is an additional step to a completed pressure vessel which is fabricated from 301 stainless steel. This material, together with the Arde' process, should be considered as a strong candidate in any future spacecraft cryogenic tankage program.

5.5 SYSTEM THERMODYNAMICS

Selection of the fluid thermodynamic state for a cryogenic system is influenced by thermal performance values which can be expected, flow rate, and required mission duration. The choice of a storage state will be strongly affected by such considerations as weight, quantity measurement, and fluid orientation. For this study it was assumed that solutions will be developed for problems arising from the choice of a storage state selected.

5.5.1 Pressure

The thermodynamic state for cryogenic systems is established by the storage pressure chosen. For systems considered herein, pressures are optimized for long duration requirements. For oxygen and nitrogen subcritical storage permits a higher specific heat input value than supercritical storage (Figs. 5.12 & 5.13). A lower pressure limit of 150 psia has been assumed for subcritical storage.

For hydrogen and helium supercritical storage permits a higher specific heat input value than subcritical. Maximum pressures of 600 psia and 1000 psia have been assumed for hydrogen and helium, respectively. Subcritical neon (150 psia) was assumed for this study, however, supercritical neon may prove to be weight optimum due to suspected higher specific heat input values above the critical pressure.

It should be noted that none of the pressures selected are weight optimum. Further study will be done, especially on hydrogen, helium and neon, to optimize system pressure from thermodynamic and weight standpoints.

The pressures selected for this study are as follows:

Oxygen	150 psia
Hydrogen	600 psia
Helium	1000 psia
Neon	150 psia
Nitrogen	150 psia

5.5.2 Thermal Protection

No significant and timely thermal performance improvements are anticipated for use in a new program. Therefore, the best values demonstrated to date are used in this study.

The thermal performance requirements can be obtained from Figures 5.15 thru 5.19 for oxygen, hydrogen, nitrogen, neon, and helium, respectively. These curves show the mission life limits for the above fluids for various ratios of heat leak to initial stored mass (Q/M). It should be noted that these curves are straight lines on the logarithmic plots as presented.

The heat leak is a function of the thermal properties of the insulation, the temperature difference and the area. For a given design, the area (which is a function of the diameter) is the only variable. Therefore $Q = K_1 f(D^2)$ (1)

where Q = heat leak, Btu/hr
K_1 = constant
D = pressure vessel dia., ft.

The fluid mass is a function of the density and the volume, but for a given fluid, only the volume is a variable. Since the volume is a function of the diameter the following equation applies.

$$M = K_2 f(D^3) \tag{2}$$

where M = Mass, Lb
K_2 = Constant

Then from equations (1) and (2)

$$\frac{Q}{M} = K_3 f\left(\frac{1}{D}\right) \tag{3}$$

where $K_3 = K_1/K_2$

Time is related as follows:

$$T = \frac{M}{W} \tag{4}$$

where T = Time, Hrs
W = Flow Rate, Lb/hr

$$\text{but } W = \frac{Q}{K_4} \tag{5}$$

where K_4 = specific heat input, Btu/lb

$$\text{Then, } T = f\left(\frac{KM}{Q}\right) \tag{6}$$

or conversely from equation (6)

$$\frac{Q}{M} = K_4 f\left(\frac{1}{T}\right) \tag{7}$$

The above analysis shows that the plot of Q/M as a function of time will be asymptotic to both axes of a linear plot or that it will result in a straight line on a logarithmic plot. The following assumptions must be noted when using these curves:

1. The flow rate was assumed to be a constant for both supercritical and subcritical storage.

2. Since these curves were derived for constant flow rate, a contingency factor is required in order to allow for variable flow requirements.

With the exception of the above limitations, these curves are applicable to any cryogenic for the fluids and missions considered herein.

The use of the Q/M curves is twofold: First, if a vessel is available that will store a given mass and has a given heat leak, the mission(s) that this vessel will satisfy can be determined. Secondly, if a mission is known and the mass requirements are known, the vessel allowable heat leak can be calculated.

Figures 5.20 through 5.24 show, for oxygen, hydrogen, nitrogen, neon, and helium, the ratio of heat leak to area (A/A) as a function of mission time. These values are developed by taking the values of Q/M from Figures 5.15 thru 5.19 and multiplying them by mass/area terms found in Figures 5.25 & 5.26. It should be noted that the M/A ratio is a function of the vessel geometry. Therefore, for each curve presented in Figures 5.15 thru 5.19 , a family of curves will result. Each curve will represent a vessel diameter and/or a length-to-diameter (L/D) ratio.

These curves will establish design, A/A valves for a specific mission time and storage state. The data presented in Figures 5.20 through 5.24 for the determination of the Q/A ratios are valid only for the thermodynamic storage state picked for that particular fluid.

To determine which technical areas are open for improvement, the basic heat transfer equation must be examined. Since the equation is a combination of conductive and radiative terms, it is as follows:

$$Q_L = \frac{KA}{\Delta X}(T_A - T_f) + F_{af}AGE(T_A{}^4 - T_f{}^4)$$

Where:

Q_L = Total heat leak, $^{BTU}/_{HR}$

K = Thermal conductivity, $^{BTU}/_{HR}$ - ^{O}R-FT

ΔX = Insulation thickness, Ft

T_A = Temperature of the environment, ^{O}R

T_f = Fluid temperatire, ^{O}R

F_{AF}= View factor from the environment to the tank

E = Emissivity

σ = Stefan-Boltzmann constant, $BTU/HR\text{-}FT^2\text{-}{}^oR^4$

A = Surface area of the tank, FT^2

In the above equation, the only terms that are not constants for a given system are the thermal conductivity, insulation thickness, surface emissivity and the ambient or outer shell temperature. Each of these variable terms must be examined in order to find an avenue of improvement. Since the spacecraft volume is usually very limited, the insulation thickness must be kept to a minimum. However, this volume constraint does not limit the thermal protection available from radiative type insulation. The following equation approximately show the effect of additional shields on the radiation heat transfer.

$$Q_N = Q_o \left(\frac{1}{N+1}\right)$$

Where, Q_o = Heat leak with no shields

N = Number of shields

Q_N = Heat leak with "N" shields

The equation indicates that as "N" becomes large, effect of additional shields and/or layers becomes small.

Finally, since the thermal conductivities and surface emissivities are physical properties, significant improvements cannot be expected. For example, in December 1963, the Bendix Corporation determined that electro-deposited silver on Inconel has the best surface available with an emissivity of .008, and to date no better surfaces have been found. A similar lack of progress has been noted in the development of low thermal conductivity materials. This indicates that some other means of insulation improvement must be found.

In view of the above considerations, the obvious variable left for consideration is the ambient or outer shell temperature. Radiation can best be controlled by temperature variation since it is a function of the fourth power of the temperature. Presently there are cryogenic systems developed that limit the total conductive heat transfer to 10 percent to 15 percent of the total. Therefore, control of the radiation is mandatory. This control can be done best by refrigerating the outer shell to a given intermediate temperature.

As noted above, for a given environmental temperature, an improvement in the overall heat leak may be accomplished by a reduction in the conductive heat transfer. The launch environment requires that the pressure vessel be well supported with respect to the spacecraft. However, the dynamics of normal spaceflight place a considerably smaller requirement on the supporting structures. Retractable annular support schemes are therefore being considered to further reduce conductive heat transfer.

5.5.3 Refrigeration

Refrigeration to an intermediate temperature appears attractive as an inexpensive means of positively controlling vessel thermal performance with present state-of-the-art insulation schemes.

The capacity and temperature levels of the refrigeration equipment required to chill the outer surface of the cryogenic storage vessels are primarily dependent on (1) the allowable heat leak per unit area of tank surface and (2) the magnitude of the surface temperature decrease necessary to achieve the allowable specific heat leak.

Figure 5.27 shows nominal state-of-the-art heat leak per square foot as a function of outer shell temperature. In assessing refrigeration requirements, entry into Figure 5.27 with a required QL/A valve from Figures 5.20 thru 5.24 will yield a required outer shell temperature. If the required outer shell temperature is below 70°F, a refrigeration penalty has been assigned to the system weight which is discussed later.

Sufficient test data on feasible refrigeration systems capable of satisfying the necessary requirements is not presently available to allow precise sizing of the unit. However, the data available on heat pump systems (Reference 1) appears reasonable and was extrapolated down to the range of temperatures under consideration. Figure 5.28 shows the refrigeration requirement, as a function of required outer shell temperature, based on a spacecraft environment temperature of 70°F. Entry into Figure 5.18 with an environmental temperature extracted from Fig. 5.27 will produce the required refrigeration load to maintain the desired outer shell temperature. Fig. 5.29 gives total refrigeration weight as a function of outer shell temperature and refrigeration load.

5.6 CONFIGURATION SELECTION

Configuration selection involves spacecraft constraints and the Cryogenic Gas Storage System (CGSS) dewar(s) size, weight, thermal performance and cryogen quantity.

Figures 5.30 through 5.35 present wet system weights as a function of mission duration for the design reference missions considered. The weights are shown for oxygen and three diluent gases with varying vessel sizes. The data in Figs. 5.30 thru 5.35 is based on a 50-50 mixture of oxygen and nitrogen. The requireu neon and helium quantities are reduced considerably due to their lower molecular weights.

Figure 5.36 illustrates the ratio of fluid weight to wet system weight as a function of fluid weight for oxygen, nitrogen and neon. Figure 5.37 shows the same parameters for hydrogen and helium.

All systems dry weights are based on Inconel 718 pressure vessels, aluminum outer shells and two discrete aluminum radiation shields. An accessory and mounting weight of 10 percent is added.

It is seen from Figures 5.36 and 5.37 that with increasing fluid weights the ratio approaches a constant with the same dewar length to diameter (L/D) ration, thus illustrating that the CGSS wet weights are essentially independent of the number of dewars and dewar size. It should be noted that this is valid only when the stored quantities per dewar are above the following minimum requirements: Oxygen, nitrogen, and neon - 1,200 pounds each; and hydrogen and helium - 100 pounds each.

Data for the figures depicting the six (6) month missions indicate that present dewar technology can support these missions without refrigeration.

Figures depicting the 700 and 730 day missions include the weight of the refrigeration system which is required to maintain a low temperature environment. The refrigeration system weight and power consumption can both be reduced by locating the CGSS dewars on the dark side of the spacecraft and isolating them from heat sources, thus passively lowering the environmental temperature.

5.6.1 Use of AAP Vessels

Bay I of the Apollo Service Module can accept cylindrical tankage 146 inches long by 41.5 inches diameter. Two equally sized tanks in this Bay would be 73 inches x 41.5 inches each. Assuming pressure vessel inside dimensions of 70 inches x 38.5 inches, the volume is 38.5 cubic feet per vessel. This volume will accommodate the following usable cryogen quantity:

Oxygen	2,600 lbs
Hydrogen	163 lbs
Nitrogen	1,850 lbs
Neon	2,750 lbs
Helium	285 lbs

The AAP dewar is suitable both from size and thermal standpoints for the storage of oxygen, nitrogen, neon, hydrogen and helium for all 6 month missions and the Mars Flyby Mission.

Refrigeration is required for all the cryogens considered for the Mars Flyby Mission and helium storage for the 6 month missions.

Table 5.1 illustrates the utilization of AAP dewars for oxygen storage as a function of the various missions.

Table 5.2 illustrates the utilization of AAP dewars for hydrogen storage as a function of power levels and mission duration.

5.7 DEVELOPMENT PROGRAM

5.7.1 Schedule

The estimated development schedule for the space station cryogenic storage system is given in figure 38. The cryogenic system schedule shows the points at which a nominal 90-day system design could be frozen and qualification and early flight unit production implemented to satisfy early program needs.

5.7.2 Costs

The total program for the cryogenic storage system costs approximately $35 million and includes a recurring cost of $11 million based on the following reference configuration:

1. 24-Man Space Station
2. 6-Month Resupply (4 shipsets)
3. Each shipset consists of five 65 cubic feet capacity tanks: three for oxygen and two for the diluent gas (N_2, He, or Ne).

A preliminary cost-effectiveness study was performed to assess the impact of tank size variations upon flight system cost. The cost model in Figure 5.39 was used as the basis for this study. While detailed costs will depend somewhat on the L/D of the tanks (in addition to obvious factors such as materials and fabrication techniques), it was assumed that production cost is determined by diameter only.

Figure 5.40 shows tank cost, as a function of fluid quantity stored, for oxygen, nitrogen, neon, helium, and hydrogen. The effect of fluid density is readily apparent. This plot is based on single-tank storage of these fluids.

Since the fluid quantities involved in this study are large, multiple-tank storage will undoubtedly be used. The relative cost of dividing the fluid inventories among several vessels is shown in Figure 5.41. These curves define quantitatively the general contention that one large tank is generally less expensive than two small ones for storing a given amount of fluid.

Figure 5.42 defines the cost trends for storing the particular fluid quantities required for the space station. The storage model assumed for this figure is based on a 6 month resupply interval; the costs shown include one shipset for the initial launch and one for each of three resupply flights. The cost of these four shipsets is plotted as a function of the number of tanks into which the required fluid quantity is divided.

Of the three possible diluent gases investigated, neon is the most cost-effective (e.g., tank cost for the required quantity of neon (3400 lbs) is less than that of the required quantity (610 lbs) of the second-ranked helium). Nitrogen storage is the most expensive.

The data presented are valid for trending purposes but do contain the assumption that no commonality in size is dictated for both oxygen and diluent gas tanks.

For design and production simplicity, however, a multipurpose tank capable of oxygen or diluent gas storage would be desirable. The economy of being able to use one set of dimensional tooling and assembly fixtures in the manufacture of tanks for both cryogens is especially attractive in the light of the high capital outlay required for quality tooling.

Figure 5.43 shows the total production cost for the reference four ship-set package as a function of the size of such a multipurpose tank. The size range for "AAP size" tanks is spotted on the figure for comparison. The total program cost for the cryogenic system is plotted in Figure 5.44 as a function of the number of flights.

The shortage of both time and accurate cost data did not permit a complete cost-size optimization for the multipurpose tank concept. It is felt that a particular size does exist which results in minimum program cost, as illustrated qualitatively in Fig. 5.45. The production cost pattern from Fig. 5.43 is repeated here as a function of tank size. Also shown are the cost trends for other significant cost categories which depend directly or indirectly on the selected size. It is known that certain non-recurring costs such as tooling and handling rigs increase significantly with tank size. Other costs, however, such as those associated with system plumbing and servicing, tend to increase with the number of tanks in the system and exhibit little dependence on the size of the dewars. Costs in the latter category will be higher for systems using several small tanks. Thermal protection costs, both non-recurring and recurring, are also dependent on tank size. More study is required before a cost minimum can be identified with a particular multipurpose tank size, but it is felt that the optimum size will be in the 50 to 80 cubic foot range. It is of particular importance to determine whether the presently specified AAP tank size is in a practical range in order to prevent a substantial investment in tooling destined for very early obsolescence.

5.8 CONCLUSIONS

5.8.1 Storage Condition

The methods of subcritical storage should be carefully studied and an early flight experiment should be considered for selected approaches. The subcritical mode of operation should permit alternate withdrawal of either liquid or vapor.

5.8.2 Diluent Gases

The possibility of using neon and helium as diluent gases should be critically reviewed due to the significant potential weight savings of these gases over nitrogen. If these gases are considered acceptable, an indication should be expressed to industry suppliers of NASA's interest in the availability of LNe in the desired quantities to support space station flights.

5.8.3 Refrigeration

A separate study of this particular area should be performed in order to trade off penalties associated with venting, shadow shielding, vehicle orientation and intermediate refrigeration, or any combination of the above.

5.8.4 Potential for AAP Sized Dewars

The AAP sized dewars (limited in size by Apollo SM bay envelope) can be used for the space station or Mars Flyby requirements. Subcritical storage of oxygen and nitrogen would be required. Refrigeration would be required for six months helium storage, and all AAP vessels would have to be refrigerated if used for a Mars Flyby Mission.

5.8.5 Life Limited Components

The obvious life limited components are equilibration motors, valves, and quick disconnects. An active equilibration system is desired, and motor development must be undertaken to advance life beyond present Apollo capability.

The present Apollo relief valves are cycle life limited due to internal spring friction. However, the cycle life is probably unrealistic for the application since the Apollo systems are designed to prevent relief valve operation; and, in fact, the relief valve should not be required to cycle unless a failure occurs in the thermal protection system or unless flow from the vessel is reduced below the design minimum.

References: Heat Pump Study

1. Investigation and Analysis of the Application of a Heat Pump in Thermal Control Systems for a Manned Spacecraft, General Dynamics/Convair, San Diego, California, May 1965, Final Report, Contract NAS 9-3523, Report No. GD/C-65-120

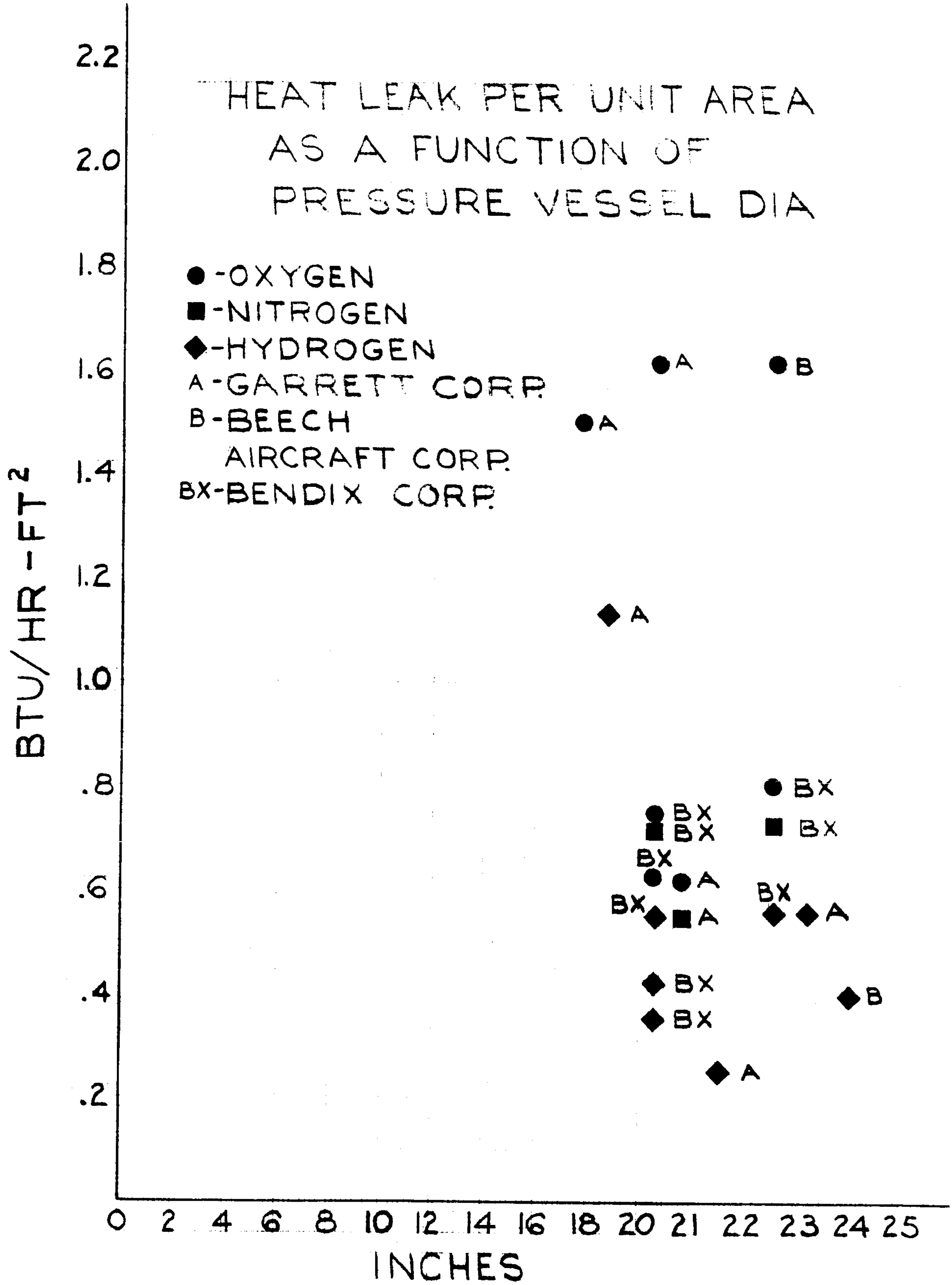

HEAT LEAK PER UNIT AREA
AS A FUNCTION OF
PRESSURE VESSEL DIA
● -OXYGEN
■ -NITROGEN
◆ -HYDROGEN
A -GARRETT CORP.
B -BEECH
AIRCRAFT CORP.
BX-BENDIX CORP.
BTU/HR - FT²
INCHES
2.2
2.0
1.8
1.6
1.4
1.2
1.0
.8
.6
.4
.2
0 2 4 6 8 10 12 14 16 18 20 21 22 23 24 25

FIGURE 5.1

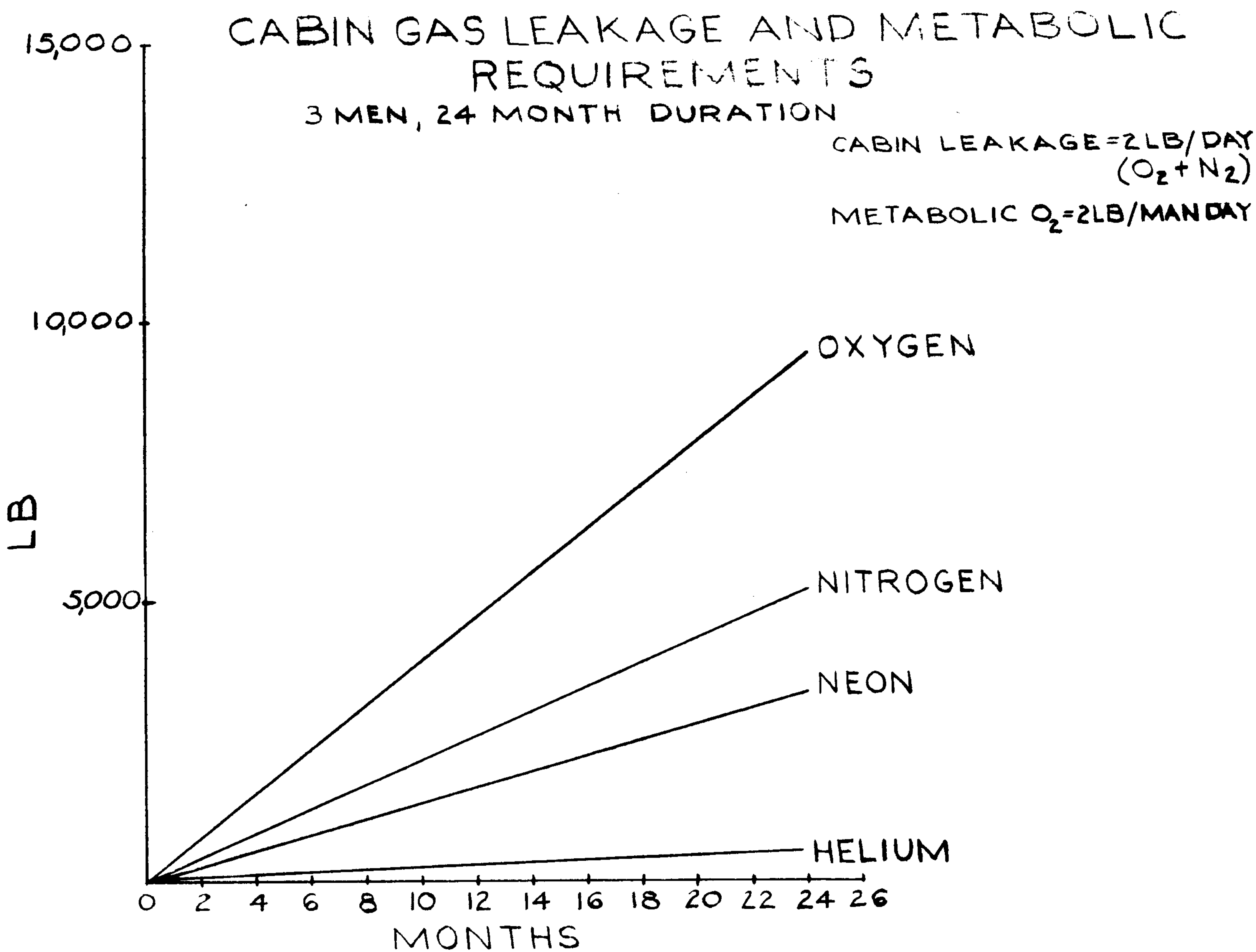
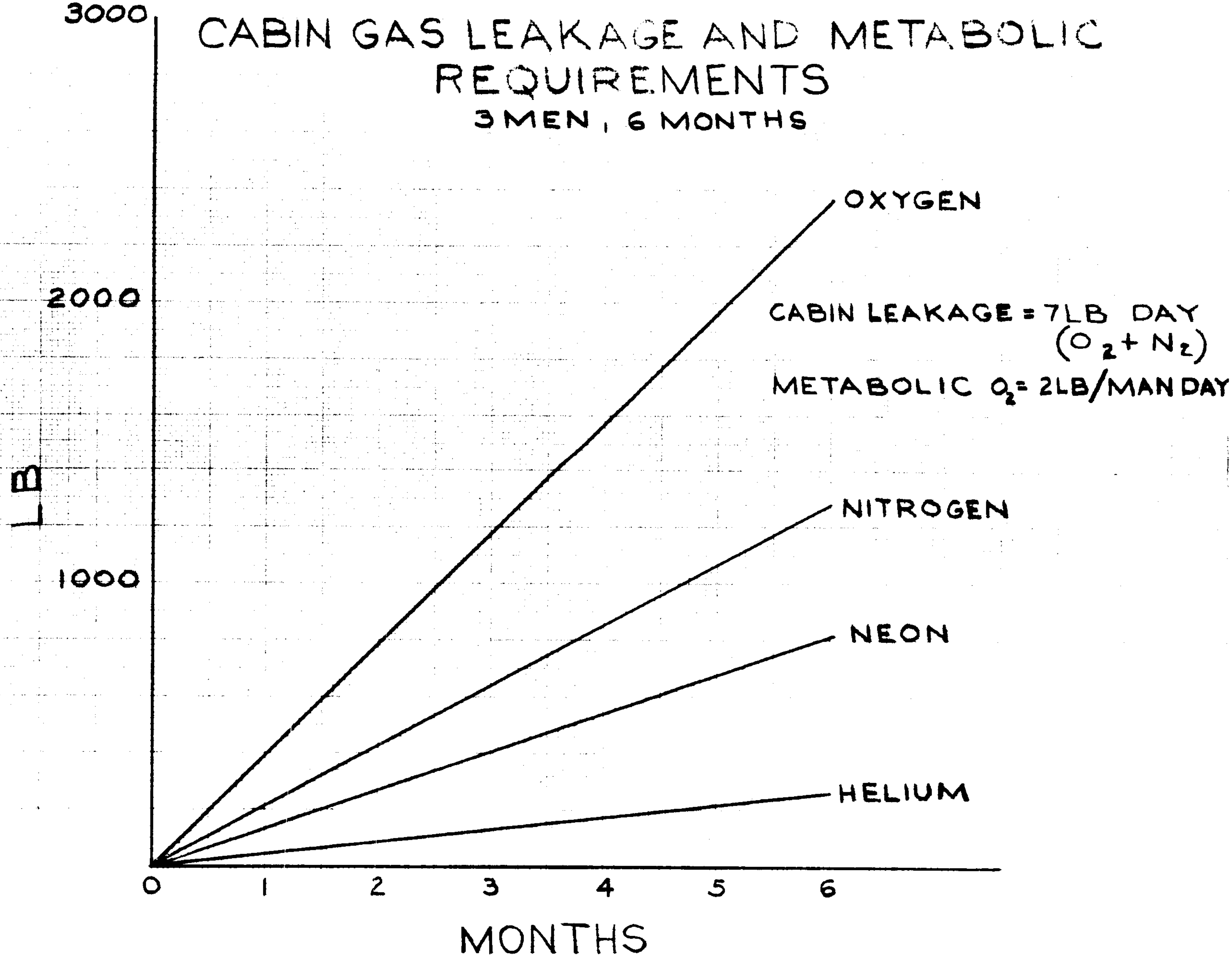

CABIN GAS LEAKAGE AND METABOLIC REQUIREMENTS
3 MEN, 24 MONTH DURATION
CABIN LEAKAGE = 2 LB/DAY ($O_2 + N_2$)
METABOLIC O_2 = 2 LB/MAN DAY
15,000
10,000
5,000
LB
OXYGEN
NITROGEN
NEON
HELIUM
0 2 4 6 8 10 12 14 16 18 20 22 24 26
MONTHS

FIGURE 5.2



CABIN GAS LEAKAGE AND METABOLIC REQUIREMENTS
3 MEN, 6 MONTHS
3000
2000
1000
LB
OXYGEN
NITROGEN
NEON
HELIUM
CABIN LEAKAGE = 7LB DAY (O_2+N_2)
METABOLIC O_2= 2LB/MAN DAY
0
1
2
3
4
5
6
MONTHS

FIGURE 5.3

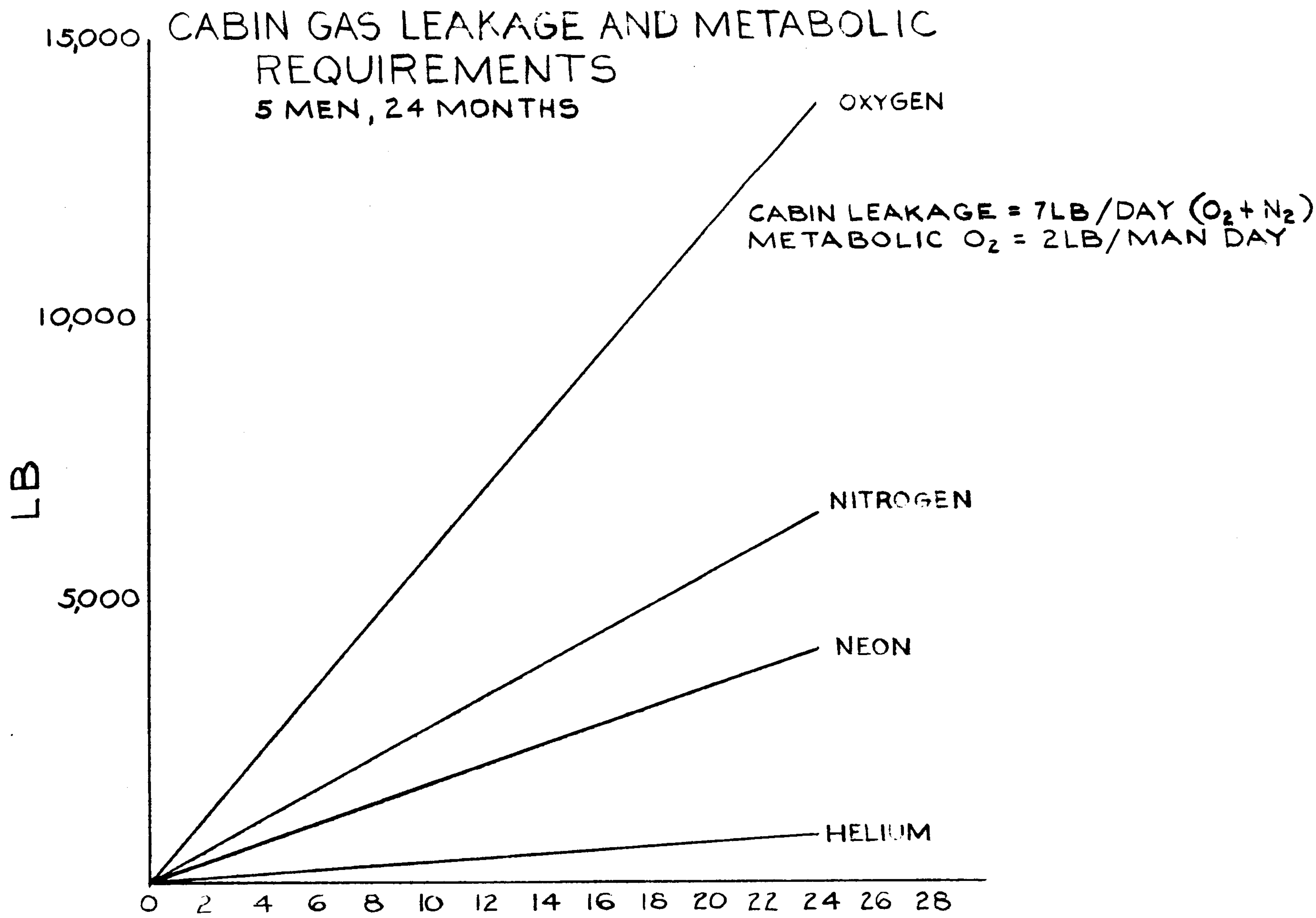

CABIN GAS LEAKAGE AND METABOLIC REQUIREMENTS
5 MEN, 24 MONTHS
CABIN LEAKAGE = 7LB/DAY ($O_2 + N_2$)
METABOLIC O_2 = 2LB/MAN DAY
OXYGEN
NITROGEN
NEON
HELIUM
LB
15,000
10,000
5,000
0
2
4
6
8
10
12
14
16
18
20
22
24
26
28
MONTHS

FIGURE 54

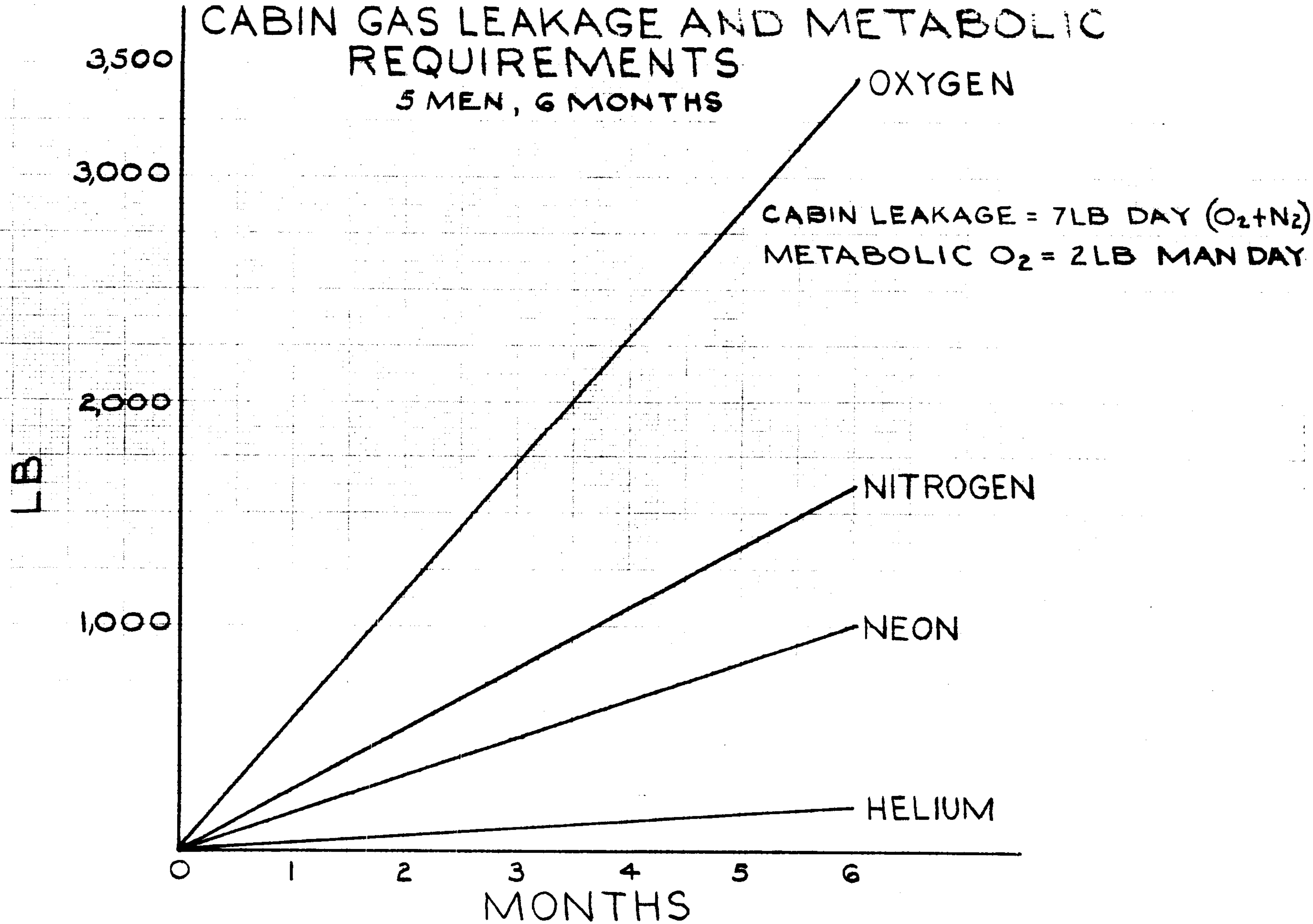

CABIN GAS LEAKAGE AND METABOLIC
REQUIREMENTS
5 MEN, 6 MONTHS
CABIN LEAKAGE = 7LB DAY (O_2+N_2)
METABOLIC O_2 = 2LB MAN DAY
OXYGEN
NITROGEN
NEON
HELIUM
3,500
3,000
2,000
1,000
LB
0
1
2
3
4
5
6
MONTHS

FIGURE 5.5

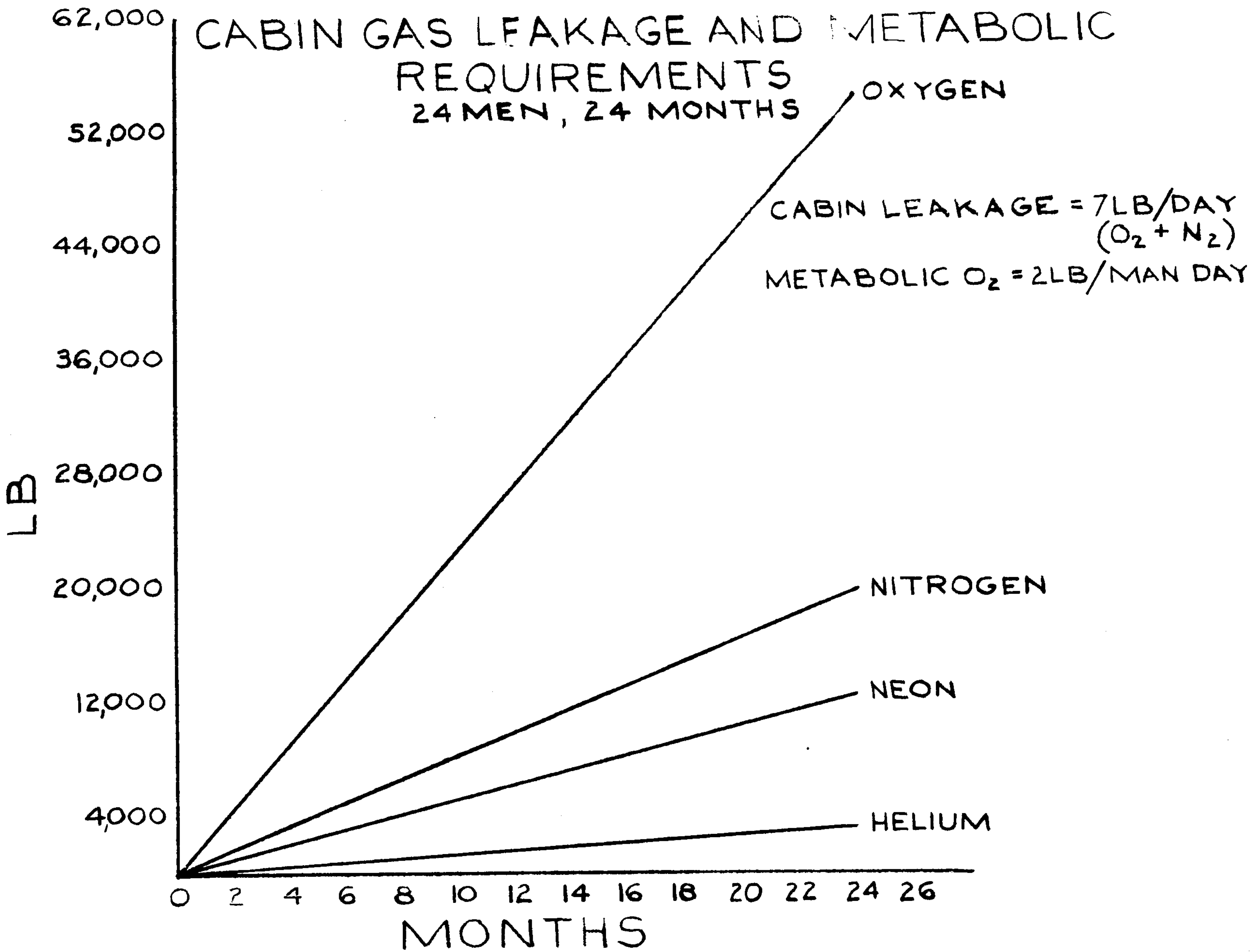
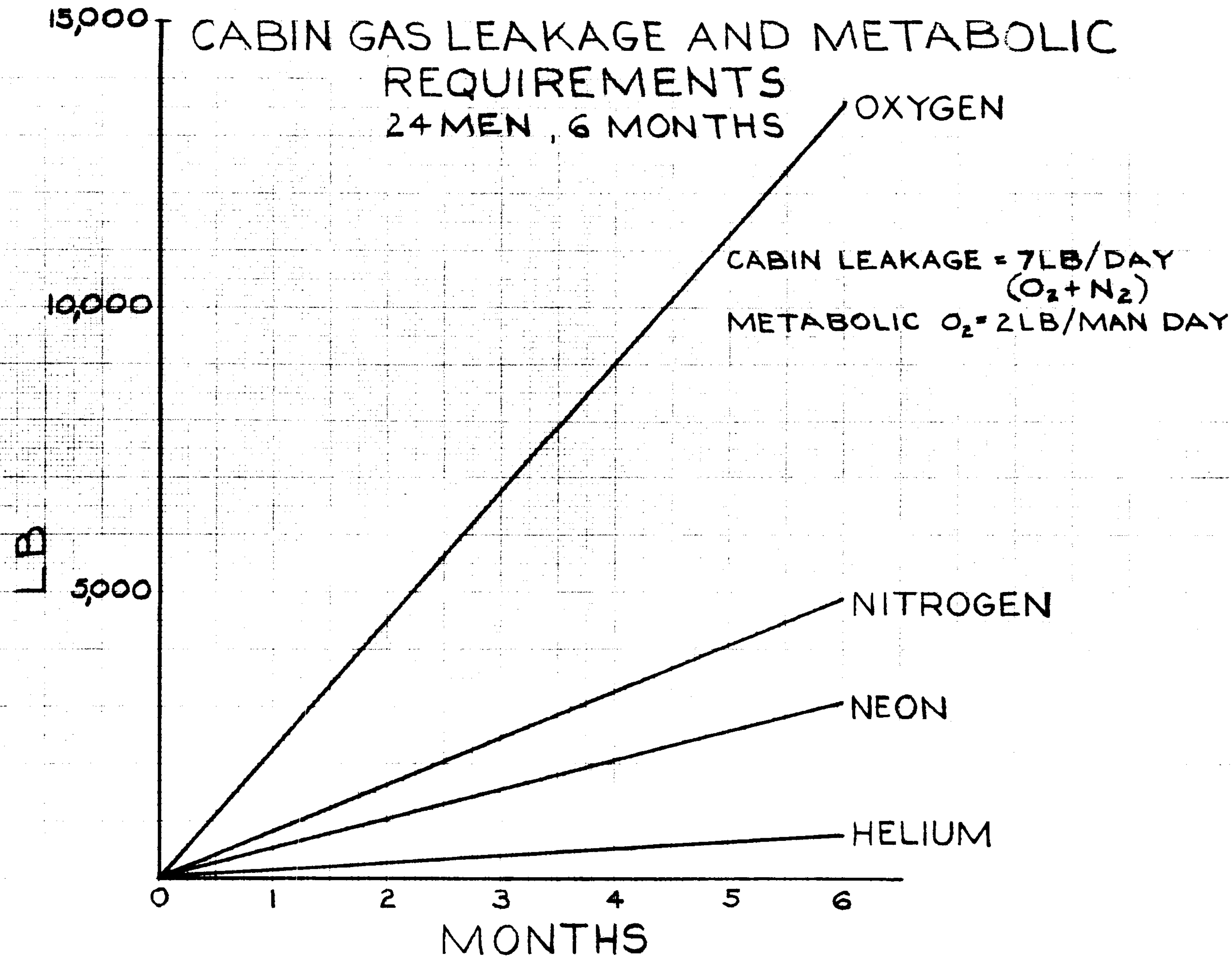

CABIN GAS LEAKAGE AND METABOLIC
REQUIREMENTS
24 MEN, 24 MONTHS
OXYGEN
CABIN LEAKAGE = 7LB/DAY
(O_2 + N_2)
METABOLIC O_2 = 2LB/MAN DAY
NITROGEN
NEON
HELIUM
LB
62,000
52,000
44,000
36,000
28,000
20,000
12,000
4000
0
2
4
6
8
10
12
14
16
18
20
22
24
26
MONTHS

FIGURE 5.6


CABIN GAS LEAKAGE AND METABOLIC REQUIREMENTS
24 MEN, 6 MONTHS
OXYGEN
CABIN LEAKAGE = 7LB/DAY ($O_2 + N_2$)
METABOLIC O_2 = 2LB/MAN DAY
NITROGEN
NEON
HELIUM
15,000
10,000
5,000
LB
0
1
2
3
4
5
6
MONTHS
FIGURE 5.7

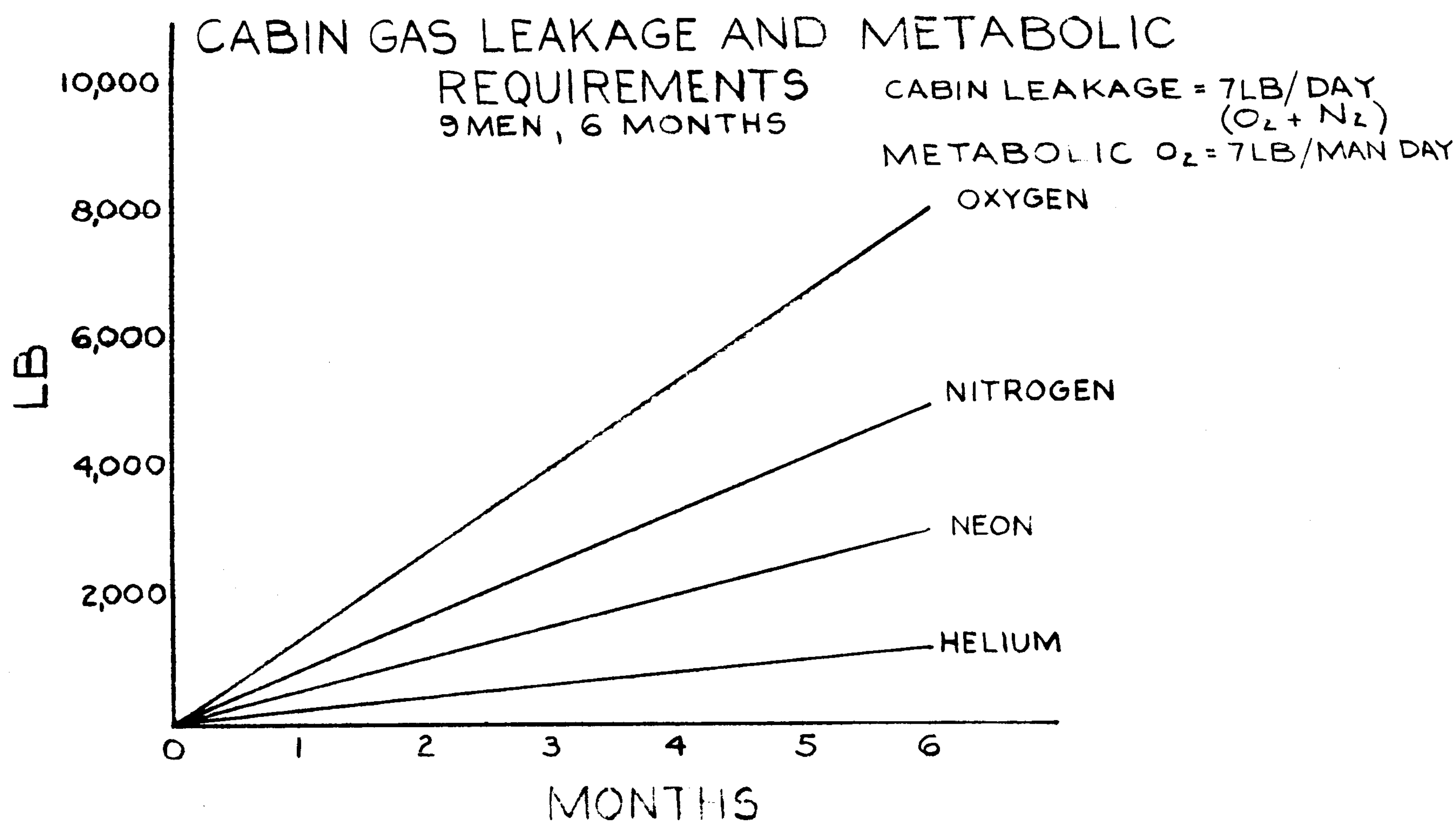

CABIN GAS LEAKAGE AND METABOLIC REQUIREMENTS
9 MEN, 6 MONTHS
CABIN LEAKAGE = 7LB/DAY (O_2 + N_2)
METABOLIC O_2 = 7LB/MAN DAY
OXYGEN
NITROGEN
NEON
HELIUM
LB
10,000
8,000
6,000
4,000
2,000
0
1
2
3
4
5
6
MONTHS

FIGURE 5.8

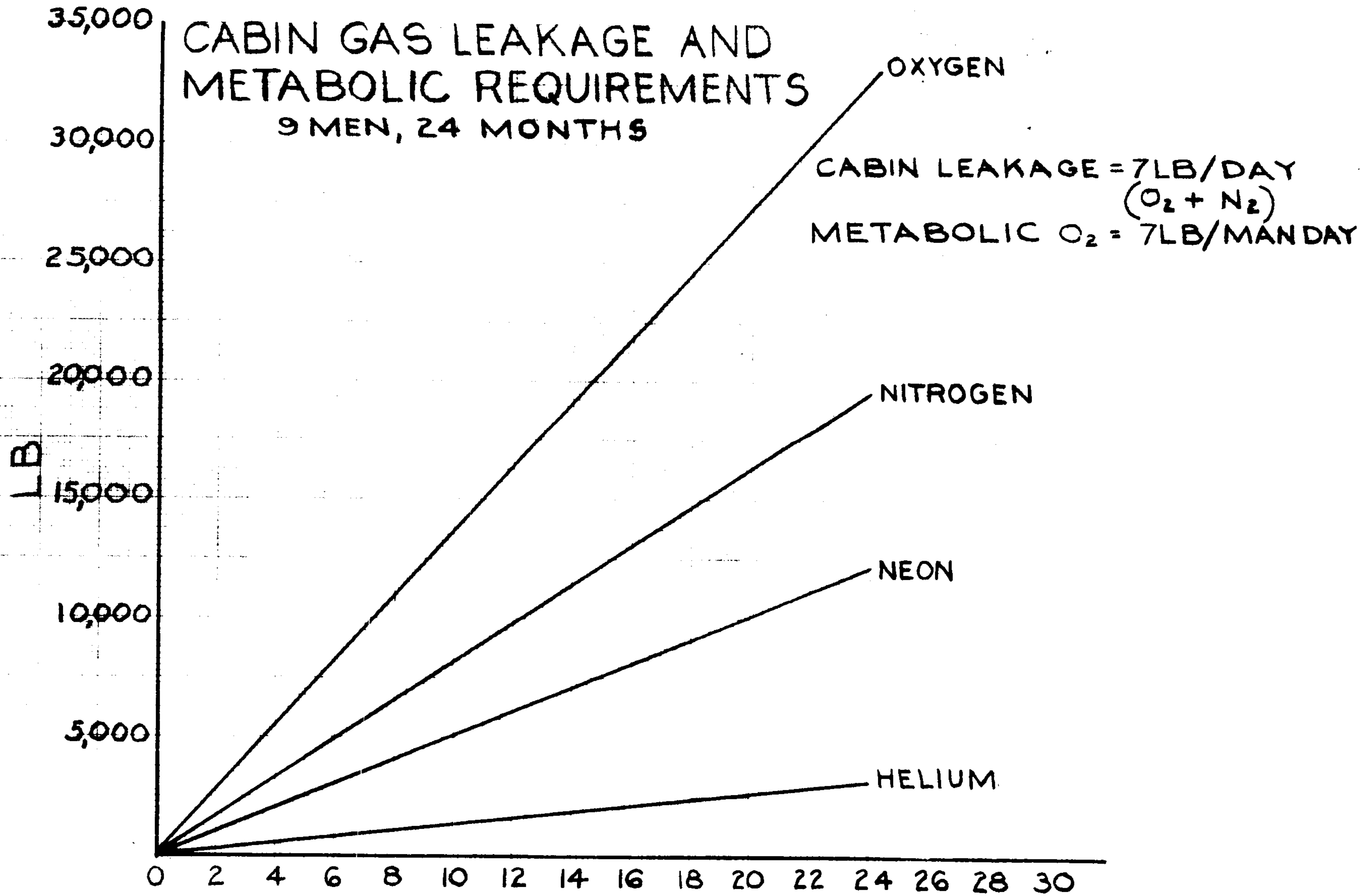

CABIN GAS LEAKAGE AND
METABOLIC REQUIREMENTS
9 MEN, 24 MONTHS
CABIN LEAKAGE = 7LB/DAY
(O_2 + N_2)
METABOLIC O_2 = 7LB/MAN DAY
OXYGEN
NITROGEN
NEON
HELIUM
LB
35,000
30,000
25,000
20,000
15,000
10,000
5,000
0
2
4
6
8
10
12
14
16
18
20
22
24
26
28
30
MONTHS

FIGURE 5.9

SCHEMATIC ARRANGEMENT
IN FLIGHT CRYOGEN TRANSFER

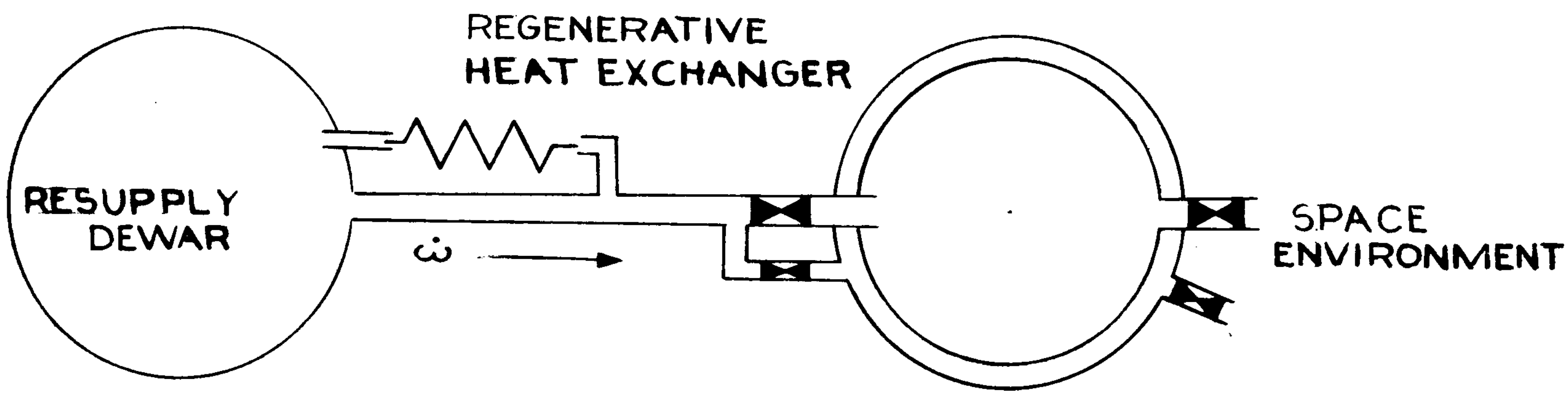


FIGURE 5.10

LIQUID TRANSFER CONCEPTS

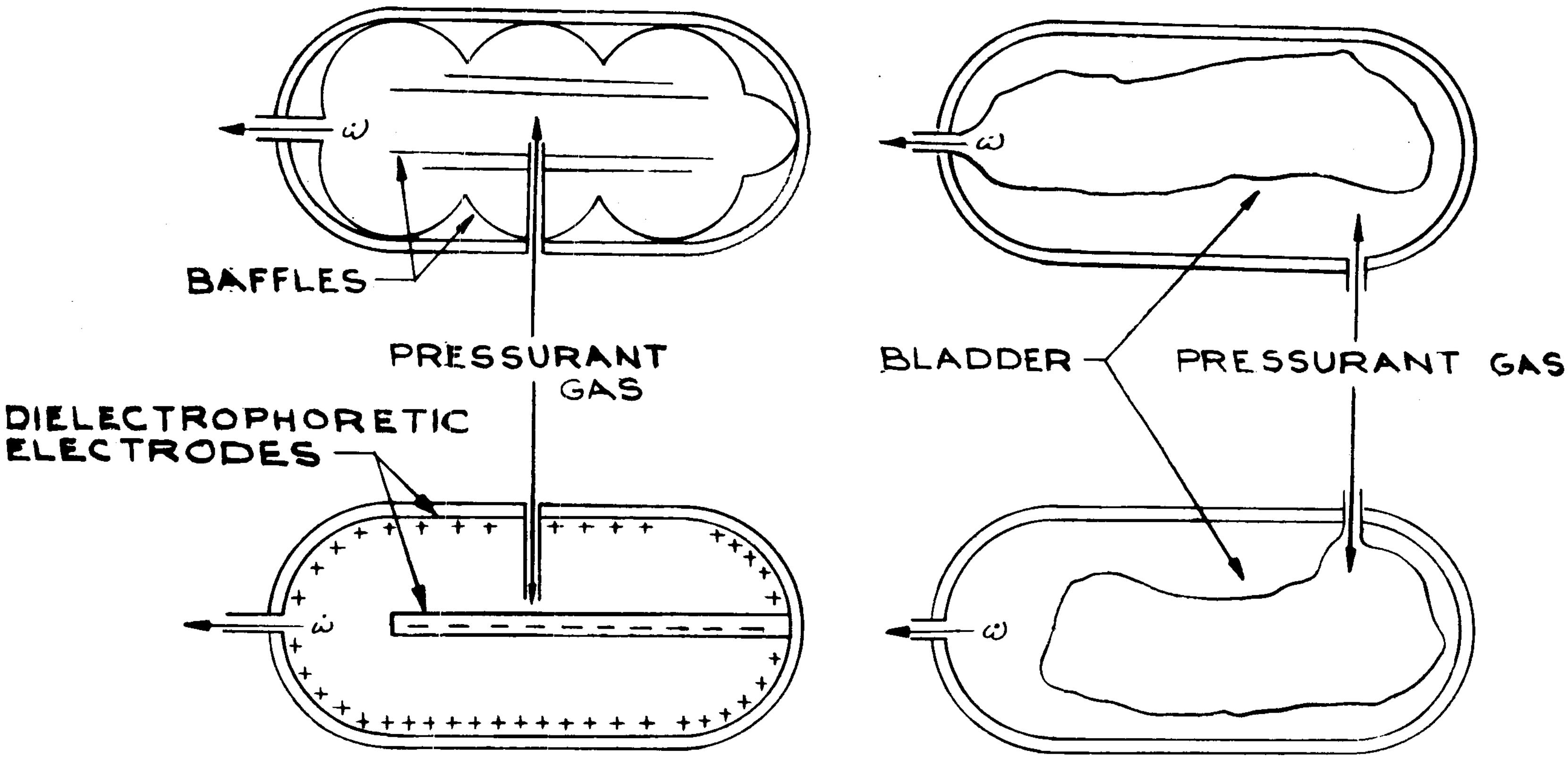


FIGURE 5.11

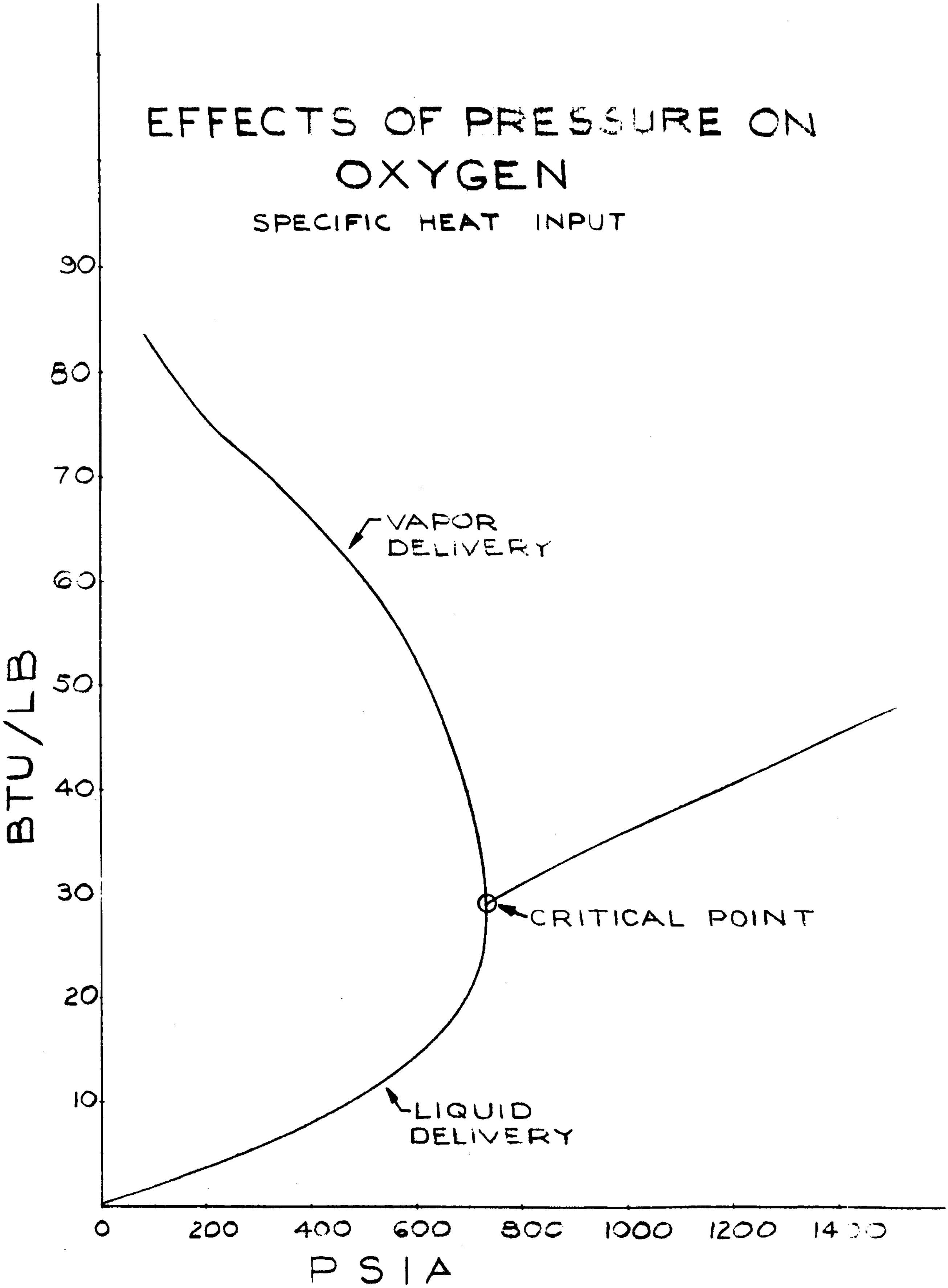

EFFECTS OF PRESSURE ON
OXYGEN
SPECIFIC HEAT INPUT
BTU/LB
90
80
70
60
50
40
30
20
10
VAPOR
DELIVERY
CRITICAL POINT
LIQUID
DELIVERY
0
200
400
600
800
1000
1200
1400
PSIA

FIGURE 5.12

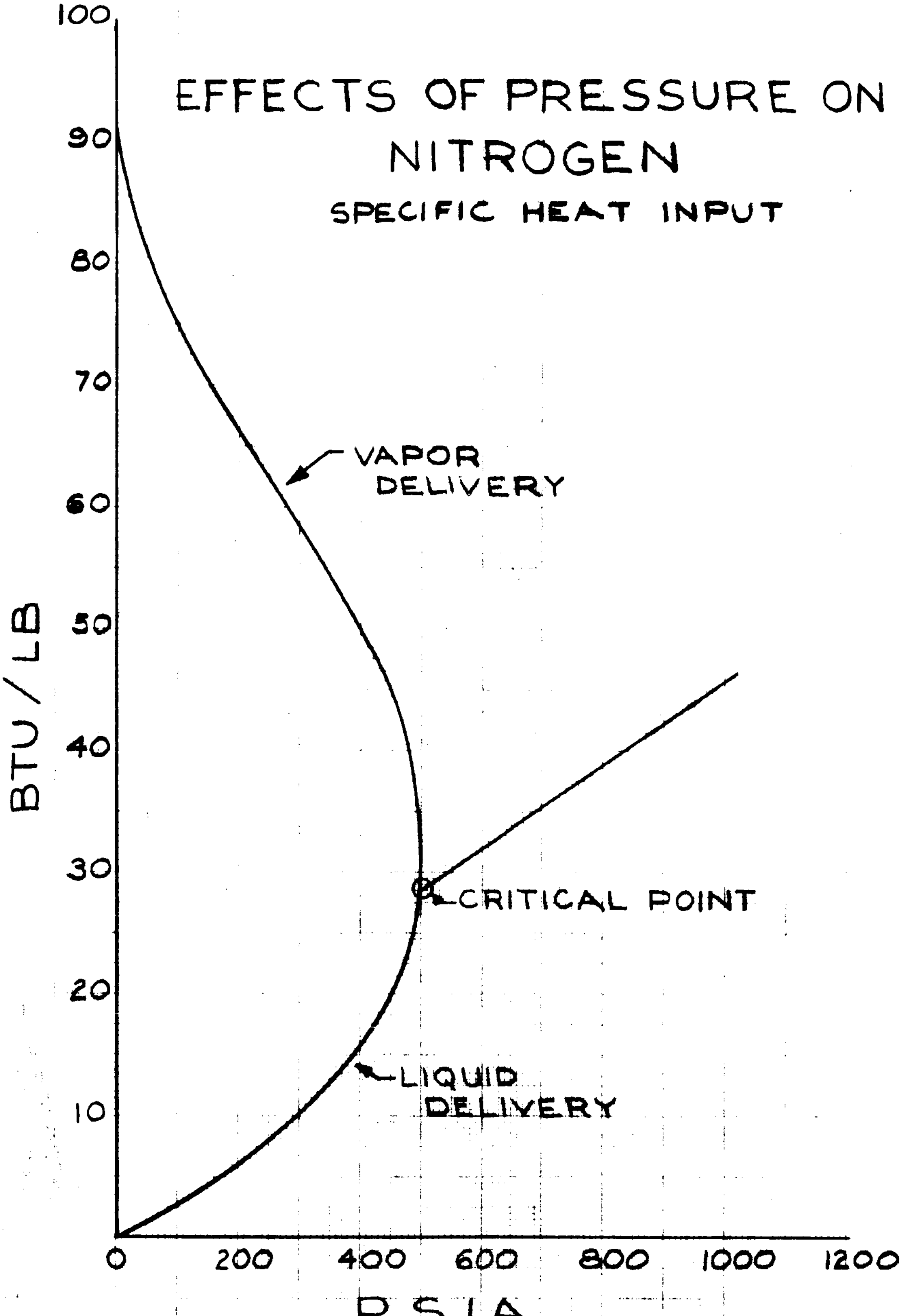

EFFECTS OF PRESSURE ON NITROGEN
SPECIFIC HEAT INPUT
100
90
80
70
60
50
40
30
20
10
BTU / LB
VAPOR DELIVERY
CRITICAL POINT
LIQUID DELIVERY
0
200
400
600
800
1000
1200
PSIA

FIGURE -5.13

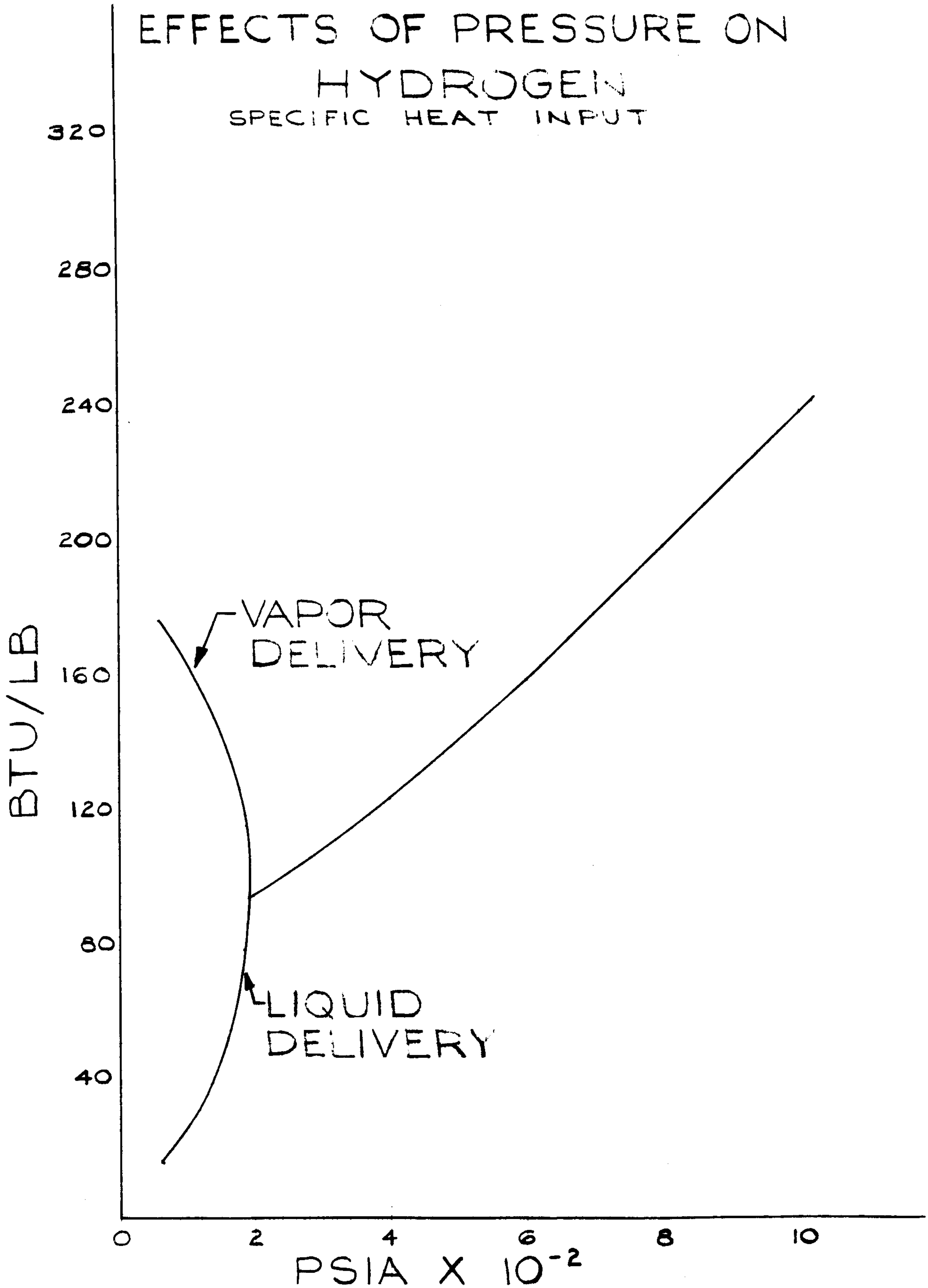

EFFECTS OF PRESSURE ON
HYDROGEN
SPECIFIC HEAT INPUT
320
280
240
200
160
120
80
40
BTU/LB
VAPOR
DELIVERY
LIQUID
DELIVERY
0
2
4
6
8
10
PSIA X 10^{-2}

FIGURE 5.14

HEAT LEAK TO FLUID MASS RATIO AS A FUNCTION OF MISSION TIME
150 PSIA OXYGEN

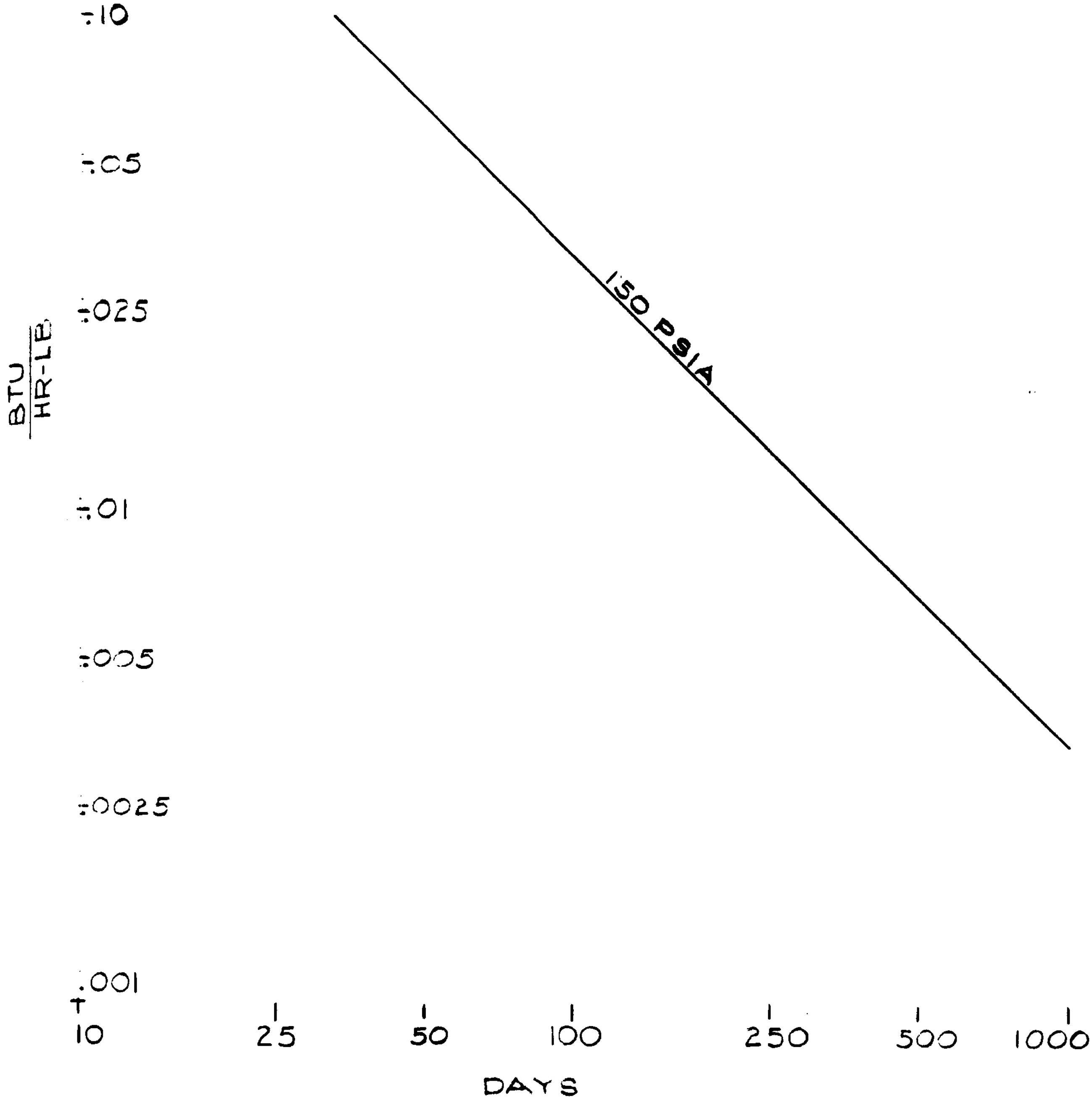


FIGURE 5.15

HEAT LEAK TO FLUID MASS RATIO AS A FUNCTION OF MISSION TIME
250 AND 600 PSIA HYDROGEN

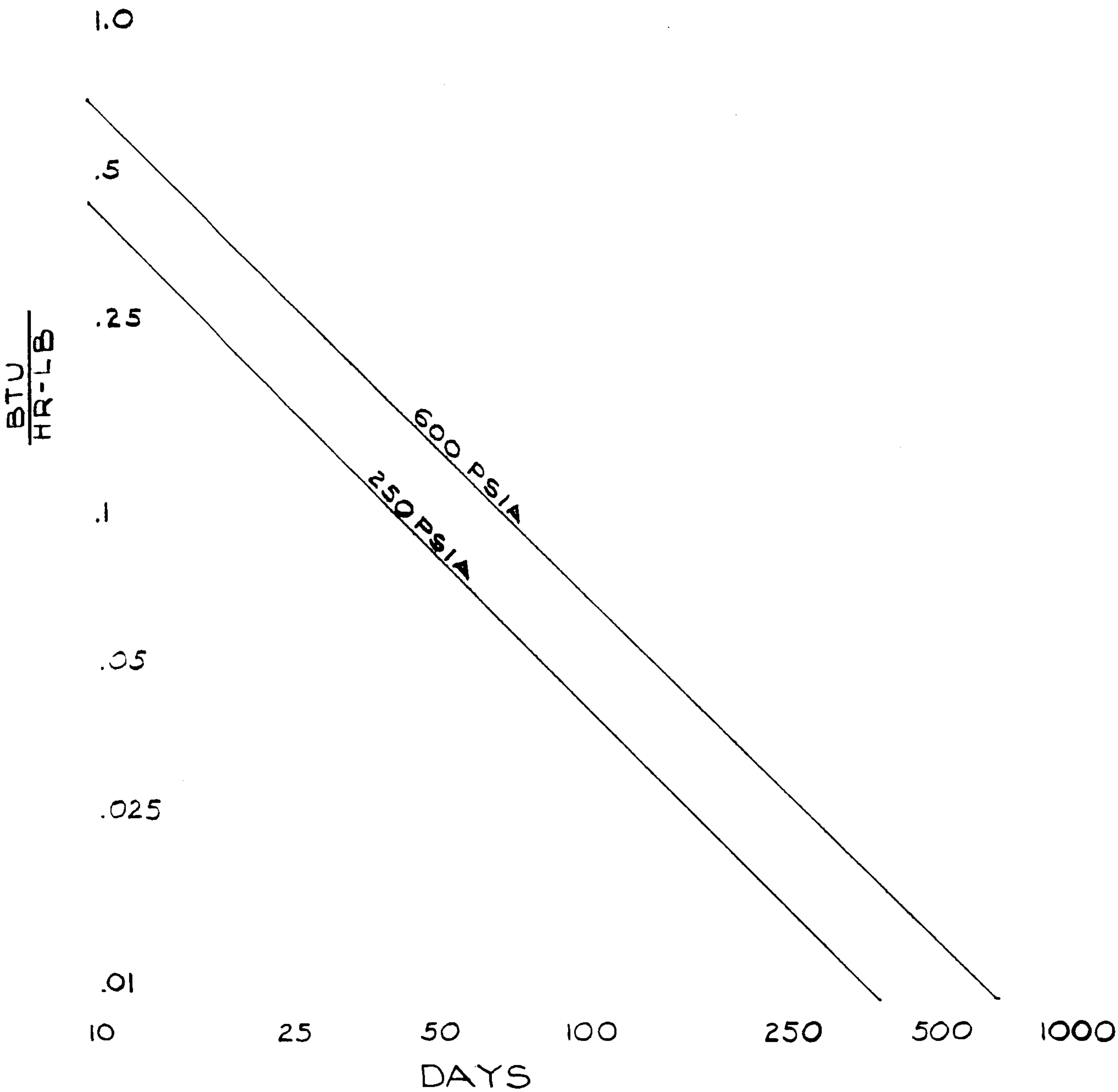


FIGURE 5.16

HEAT LEAK TO FLUID MASS RATIO AS A FUNCTION OF MISSION TIME 150 AND 1000 PSIA NITROGEN

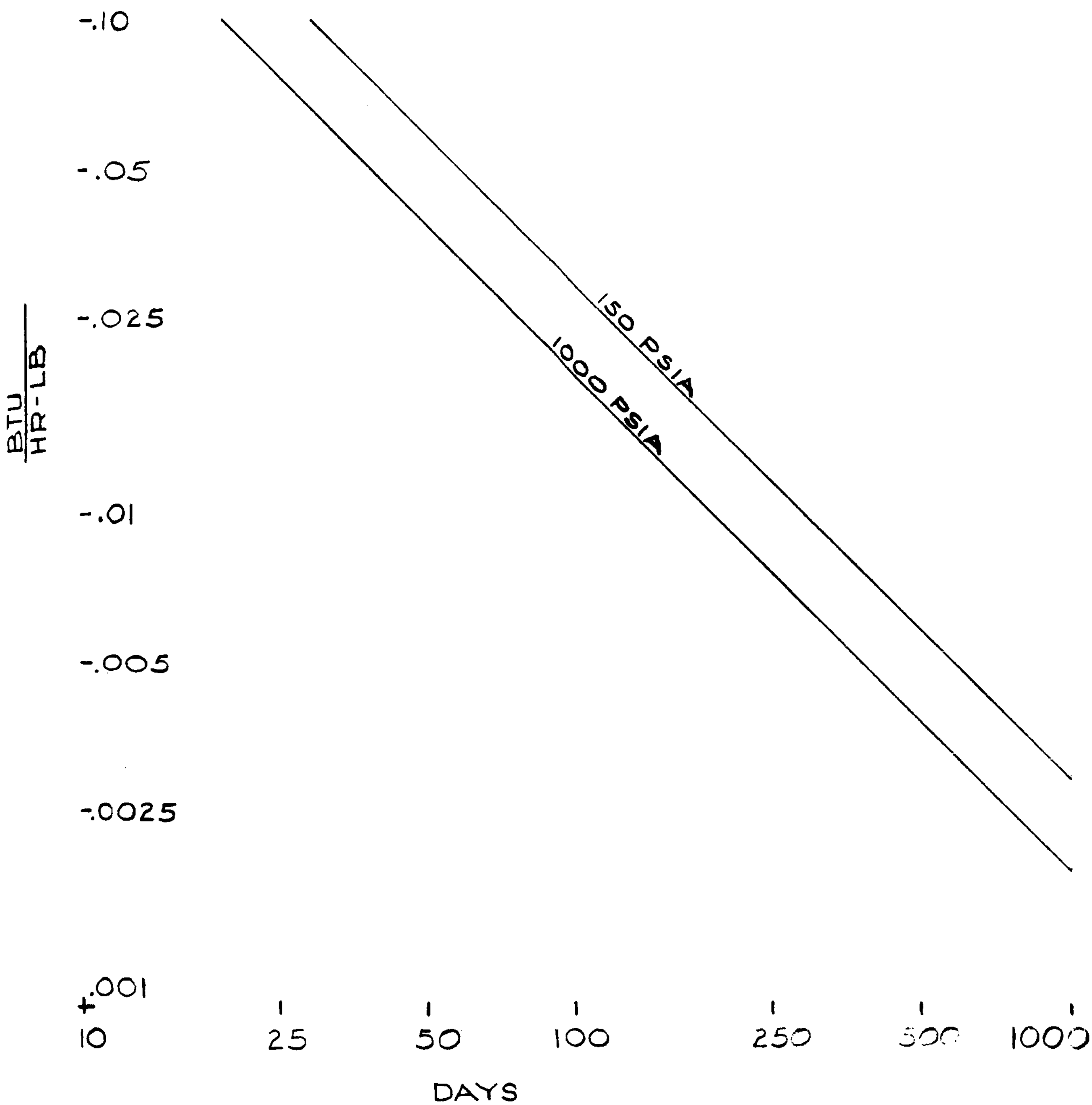


FIGURE 5.17

HEAT LEAK TO FLUID MASS RATIO
AS A FUNCTION OF MISSION TIME
150 PSIA NEON

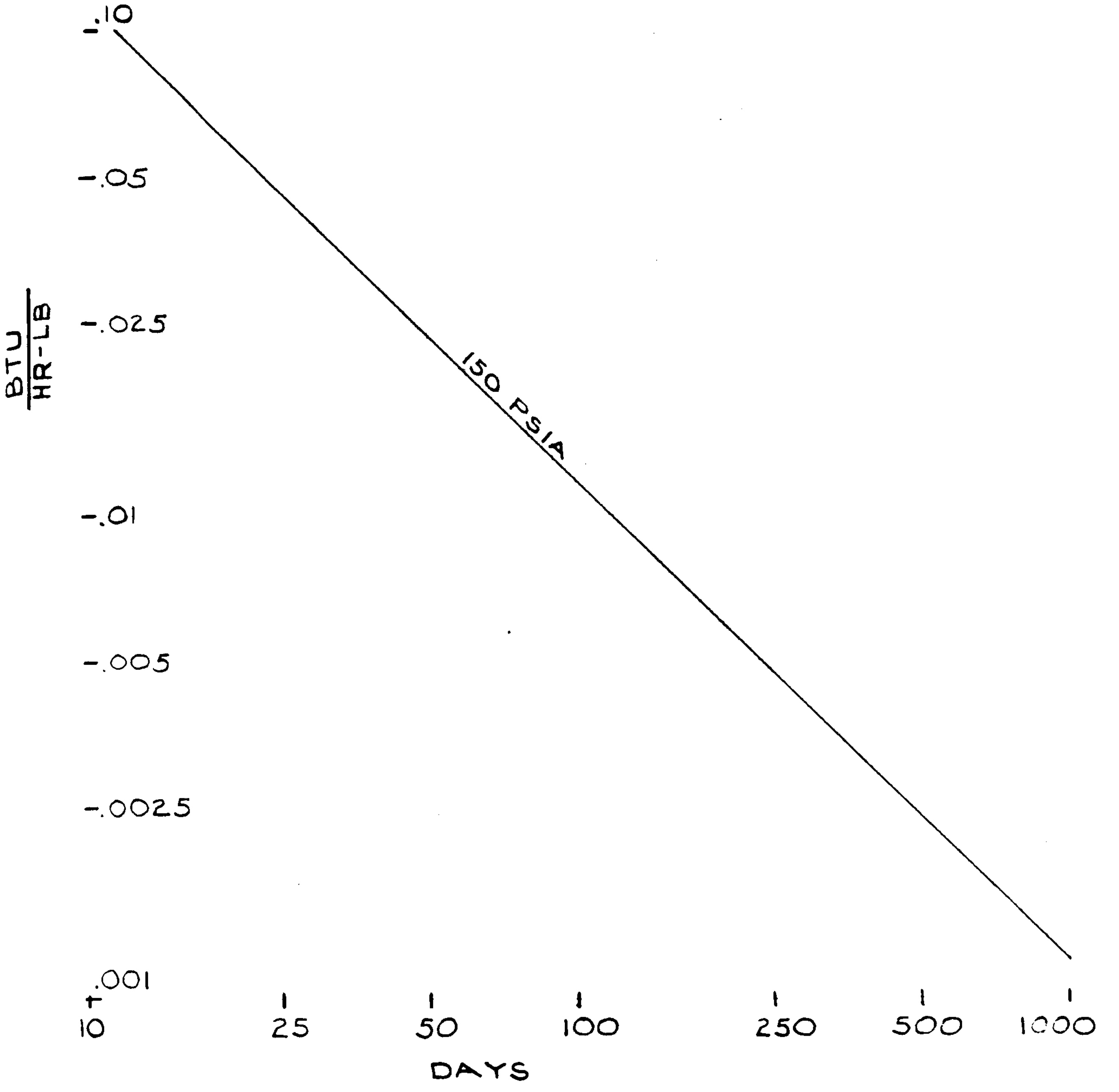


FIGURE 5.18

HEAT LEAK TO FLUID MASS AS A FUNCTION OF MISSION TIME

100 AND 1000 PSIA HELIUM

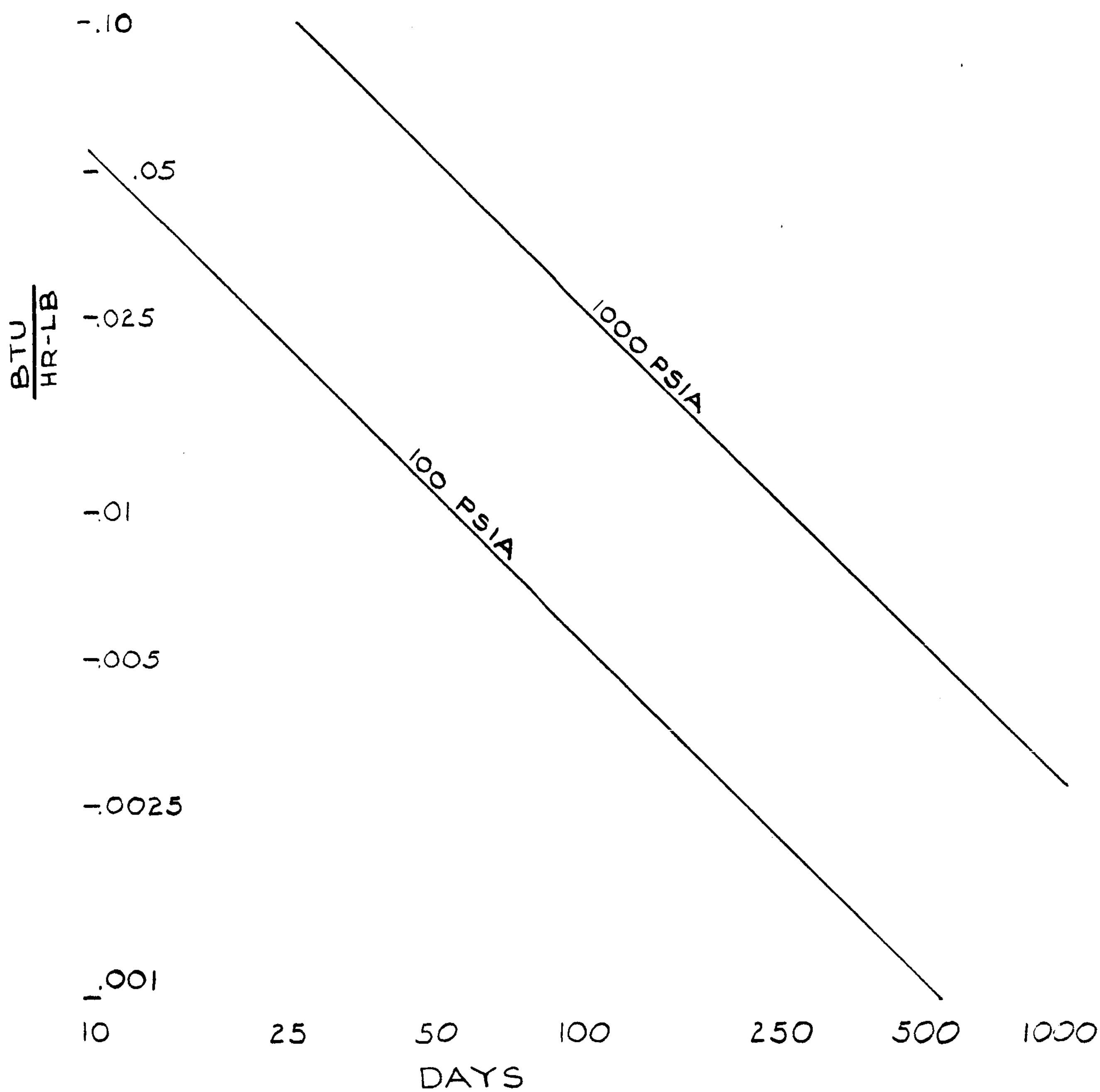


FIGURE 5.19

HEAT LEAK TO AREA RATIO AS A FUNCTION OF MISSION TIME
150 PSIA OXYGEN

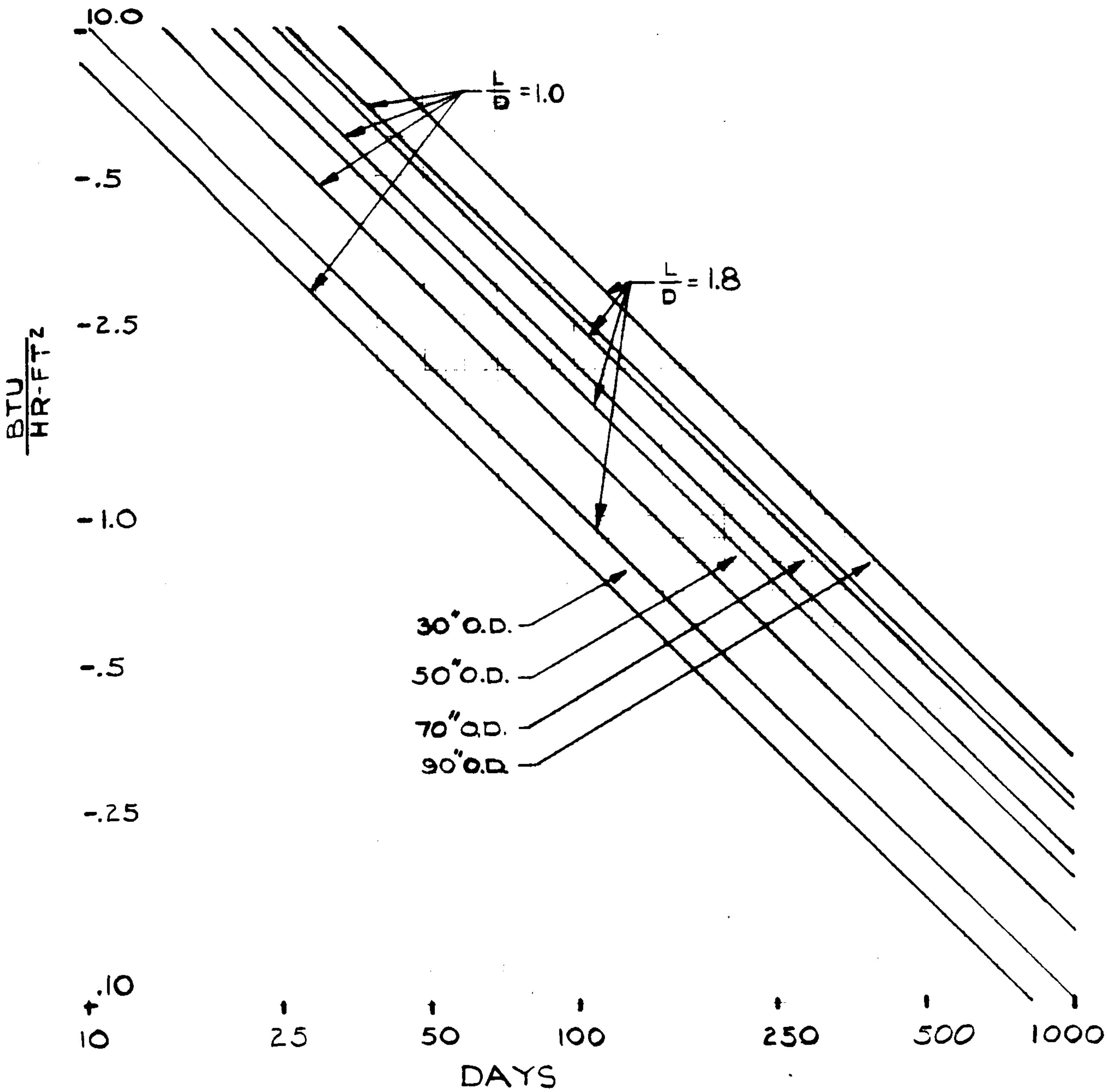


FIGURE 5.20

HEAT LEAK TO AREA RATIO AS A
FUNCTION OF MISSION TIME
600 PSIA HYDROGEN

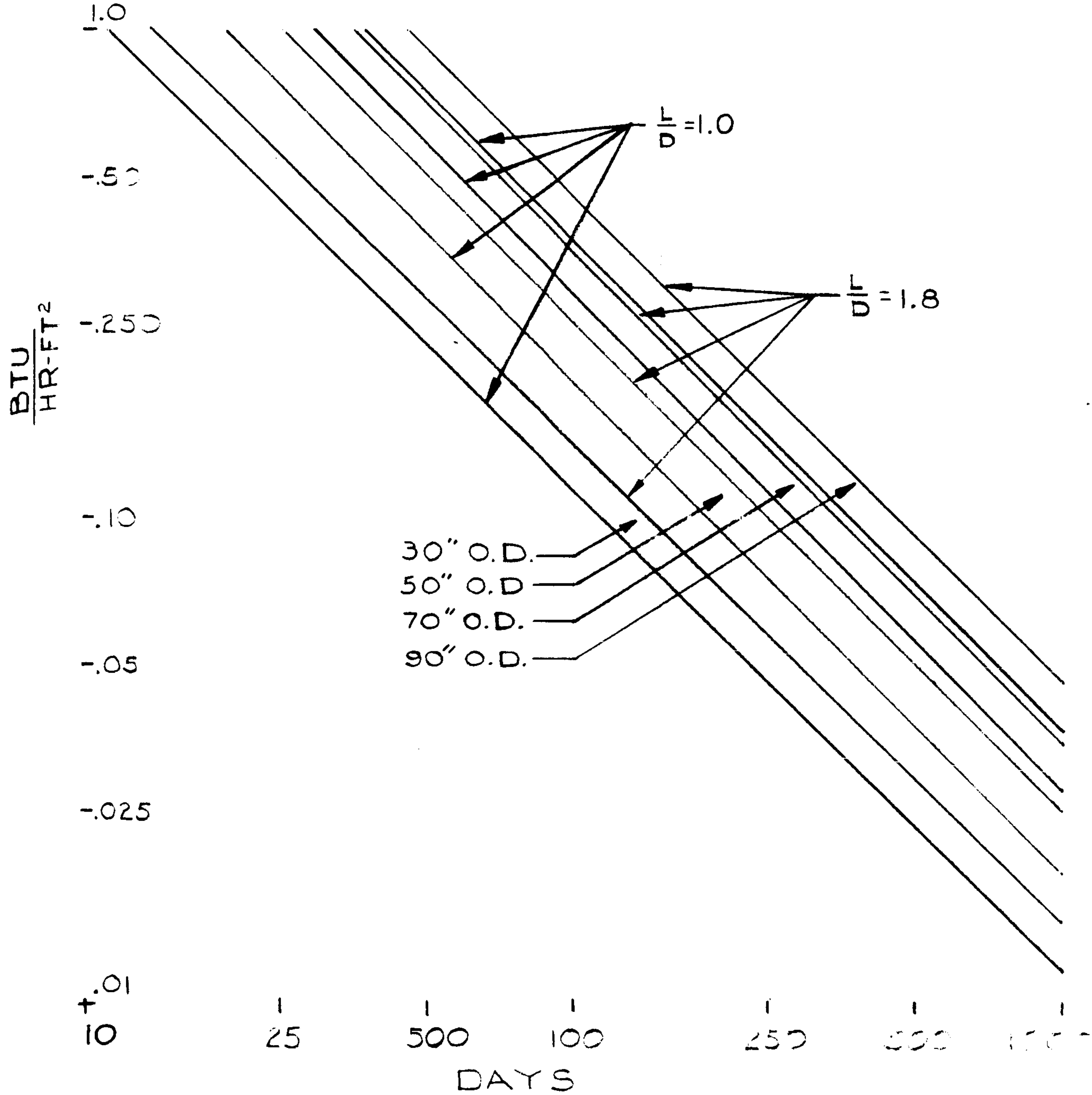


FIGURE 5.21

HEAK LEAK TO AREA RATIO AS A FUNCTION OF MISSION TIME
150 PSIA NITROGEN

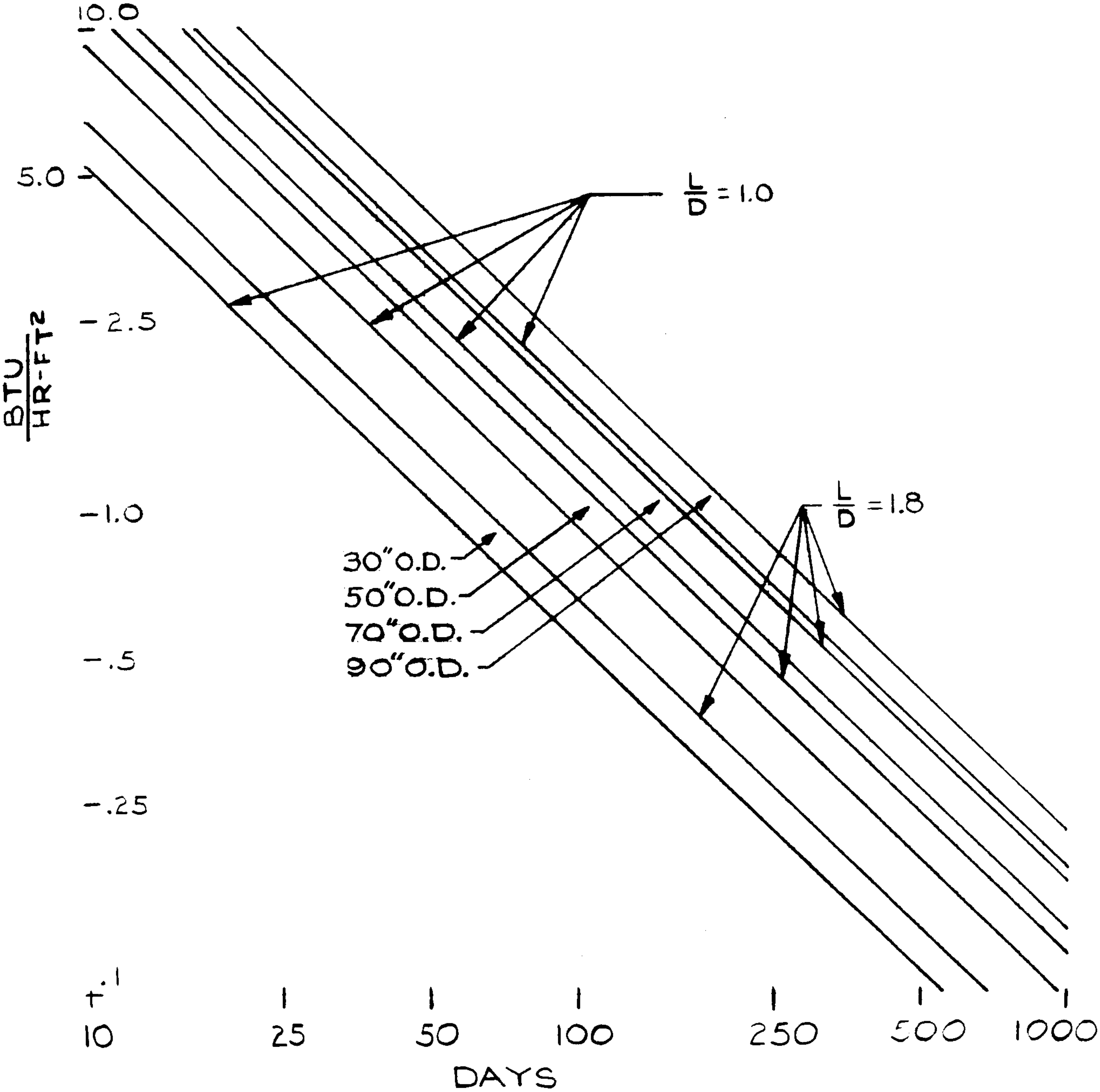


FIGURE 5.22

HEAT LEAK TO AREA RATIO AS A FUNCTION OF MISSION TIME

150 PSIA NEON

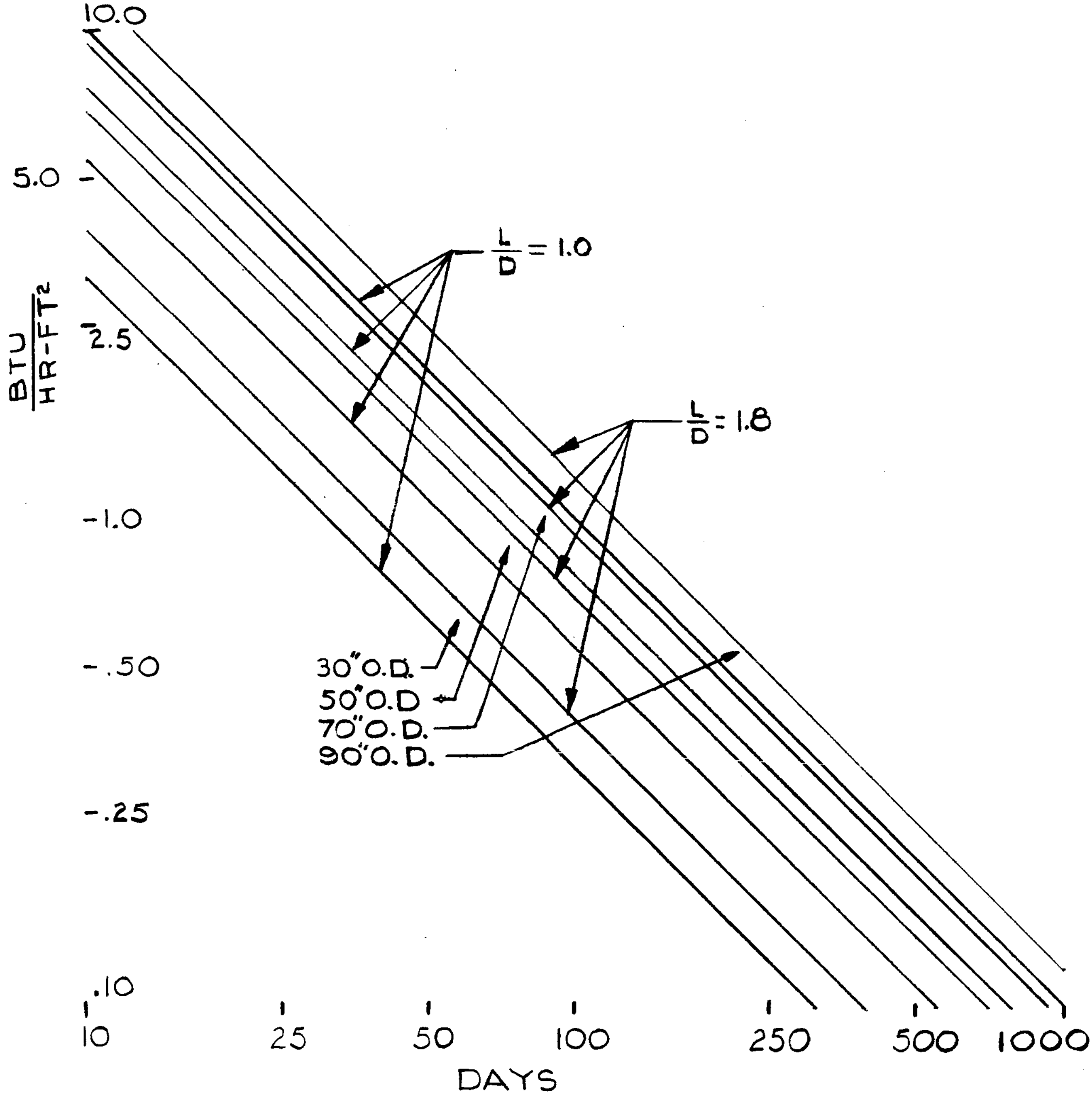


FIGURE 5.23

HEAT LEAK TO AREA RATIO AS A FUNCTION OF MISSION TIME
1000 PSIA HELIUM

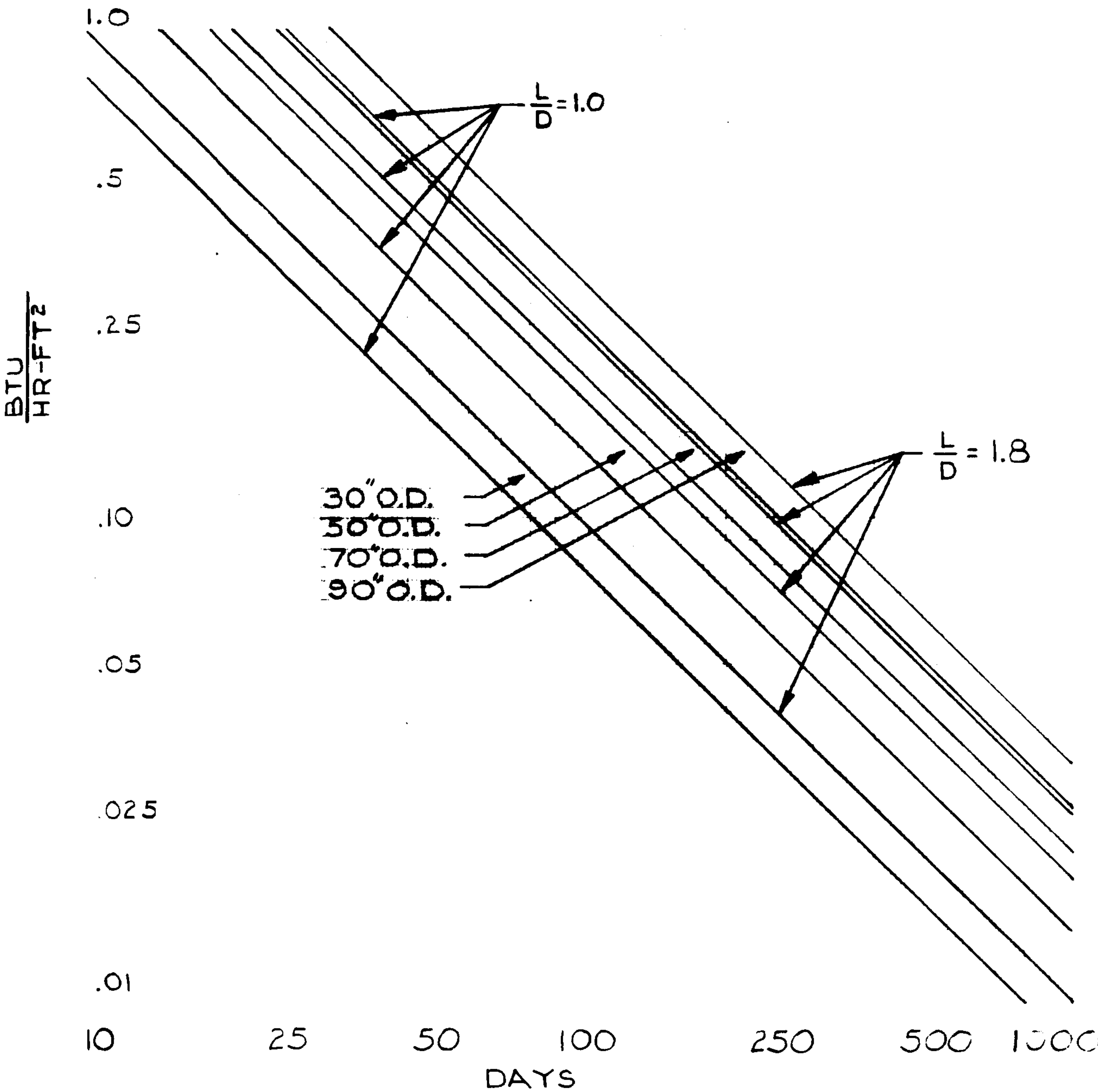
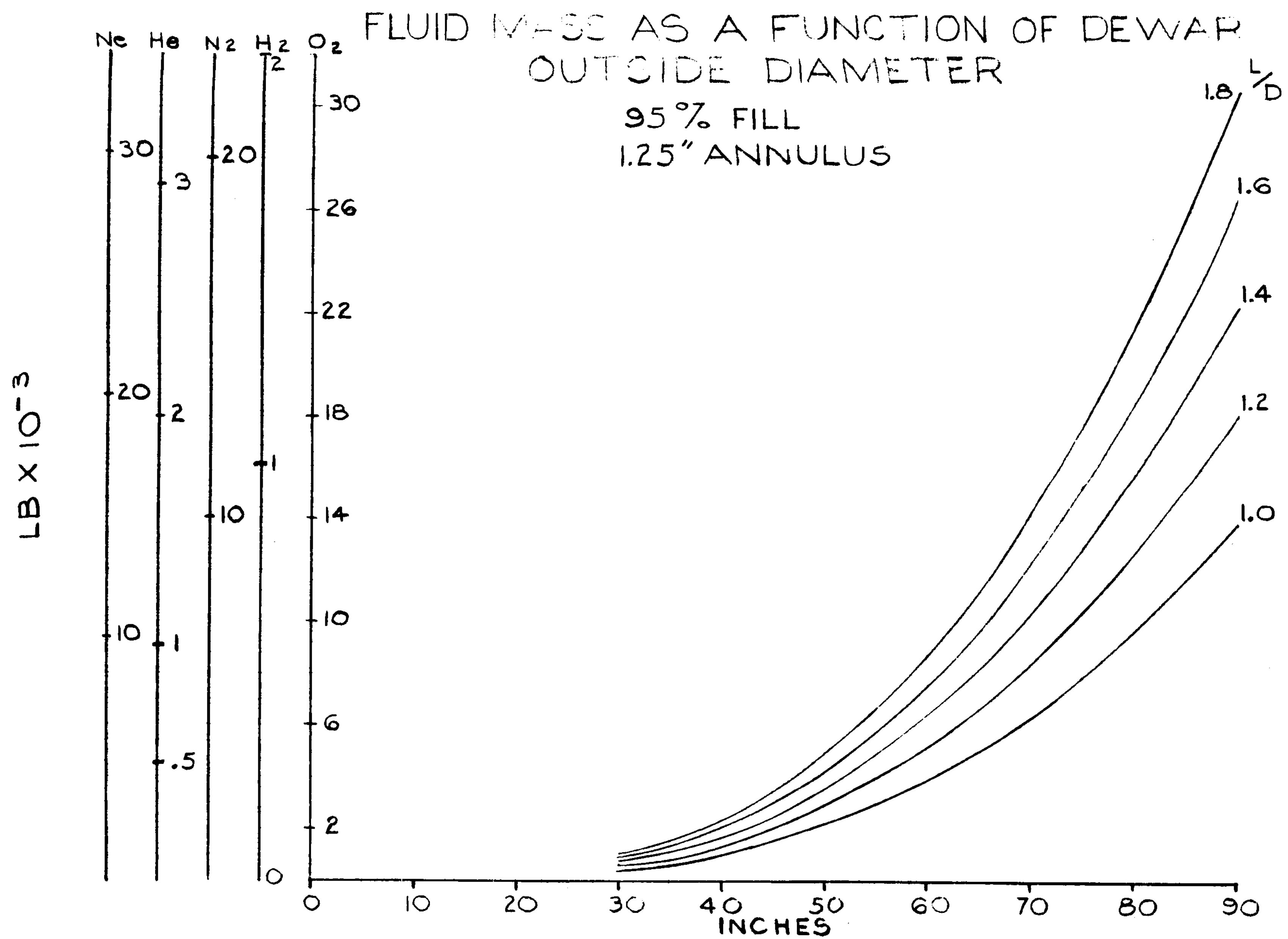
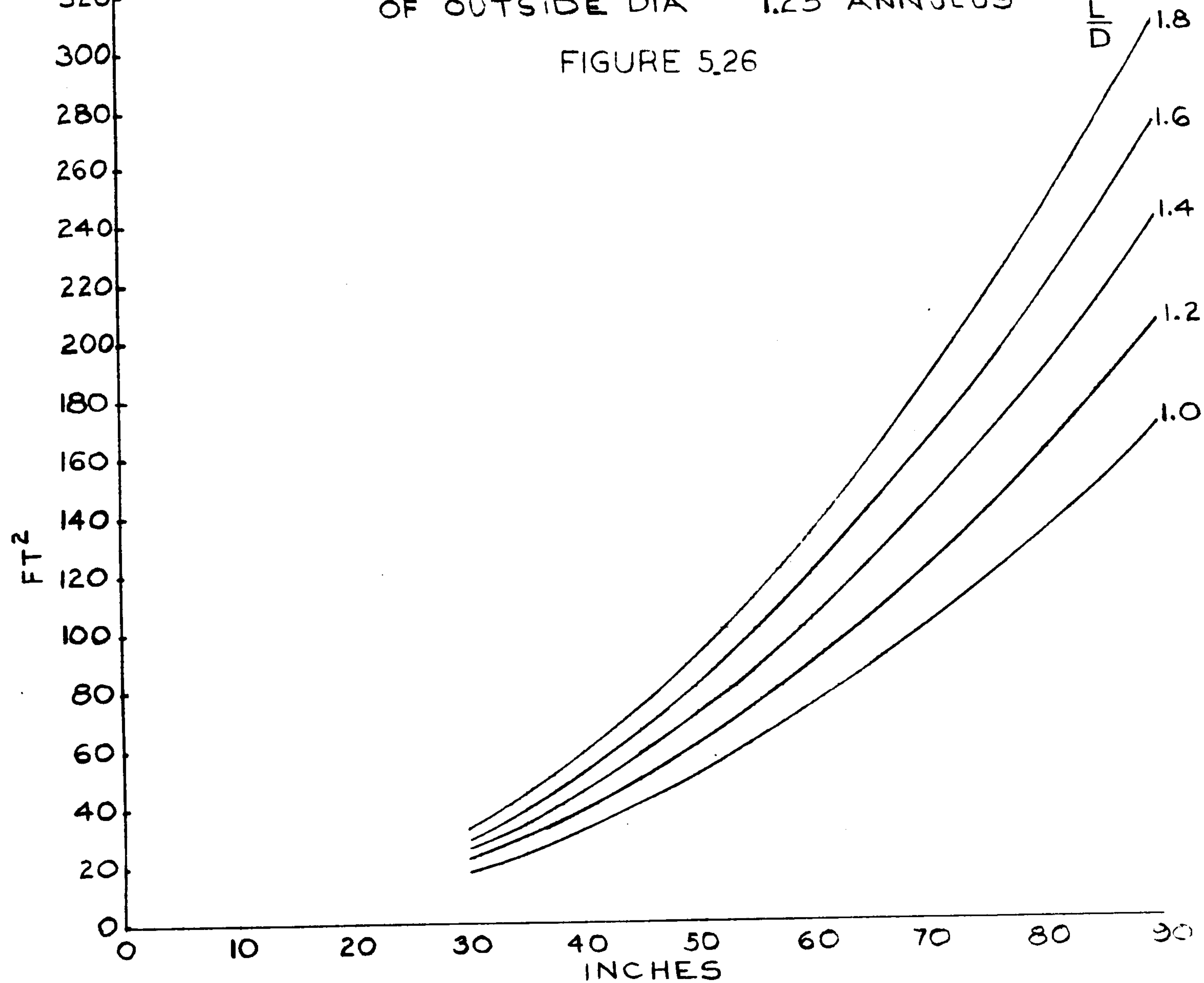


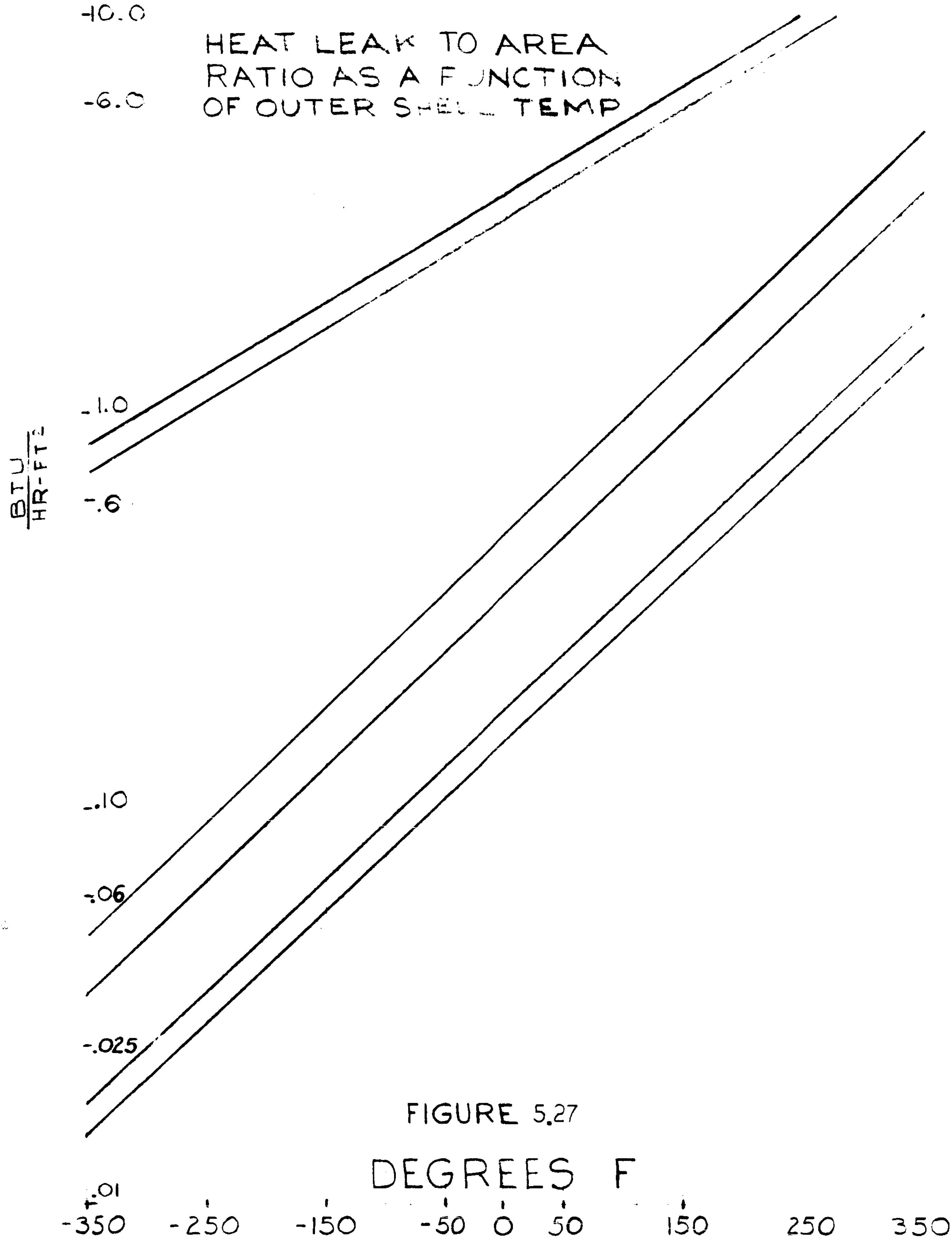
FIGURE 5.24


FLUID MASS AS A FUNCTION OF DEWAR OUTSIDE DIAMETER
95% FILL
1.25" ANNULUS
LB X 10^{-3}
Ne
He
N_2
H_2
O_2
L/D
1.8
1.6
1.4
1.2
1.0
INCHES
FIGURE 5.25

PRESSURE VESSEL SURFACE AREA AS A FUNCTION OF OUTSIDE DIA 1.25" ANNULUS

FIGURE 5.26





HEAT LEAK TO AREA RATIO AS A FUNCTION OF OUTER SHELL TEMP
BTU / HR-FT2
-10.0
-6.0
-1.0
-.6
-.10
-.06
-.025
-.01
DEGREES F
-350
-250
-150
-50
0
50
150
250
350

FIGURE 5.27

REFRIGERATION LOAD AS A FUNCTION OF OUTER SHELL TEMPERATURE

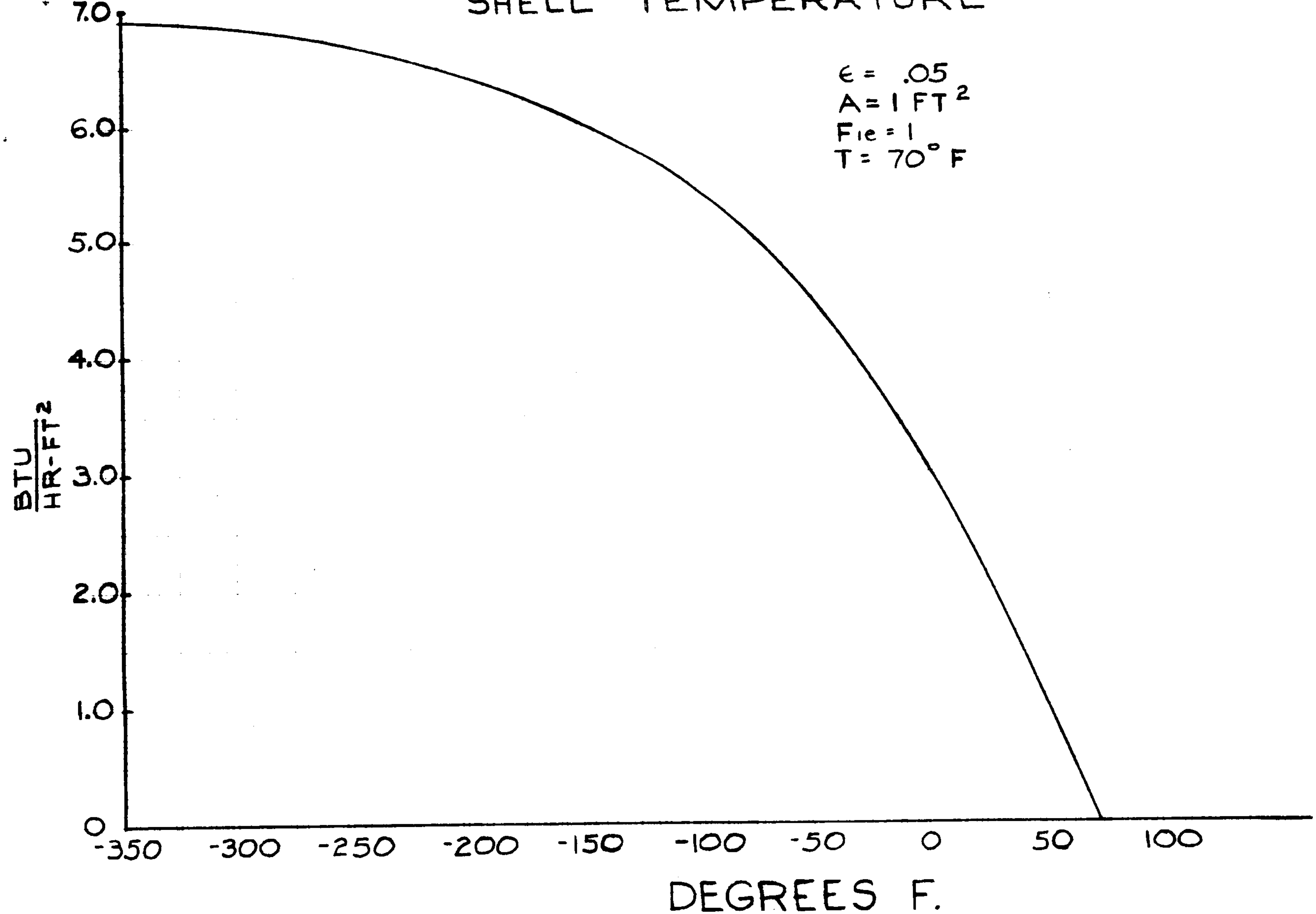


FIGURE 5.28

REFRIGERATION SYSTEM WEIGHT AS A FUNCTION OF OUTER SHELL TEMPERATURE

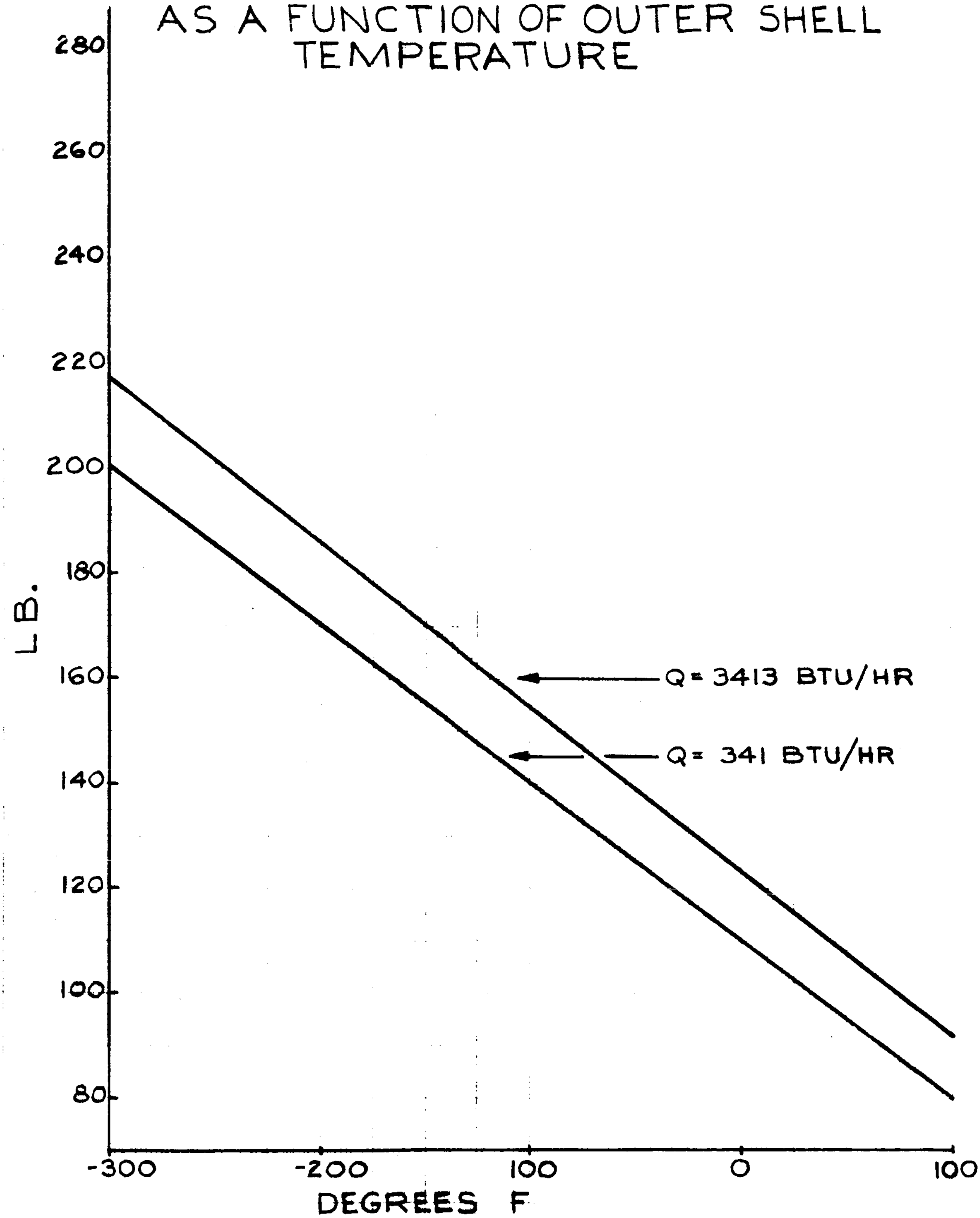


FIGURE 5.29

CRYOGENIC STORAGE SYSTEM WET WEIGHT AS A FUNCTION OF MISSION TIME - 3 MEN, 180 DAYS

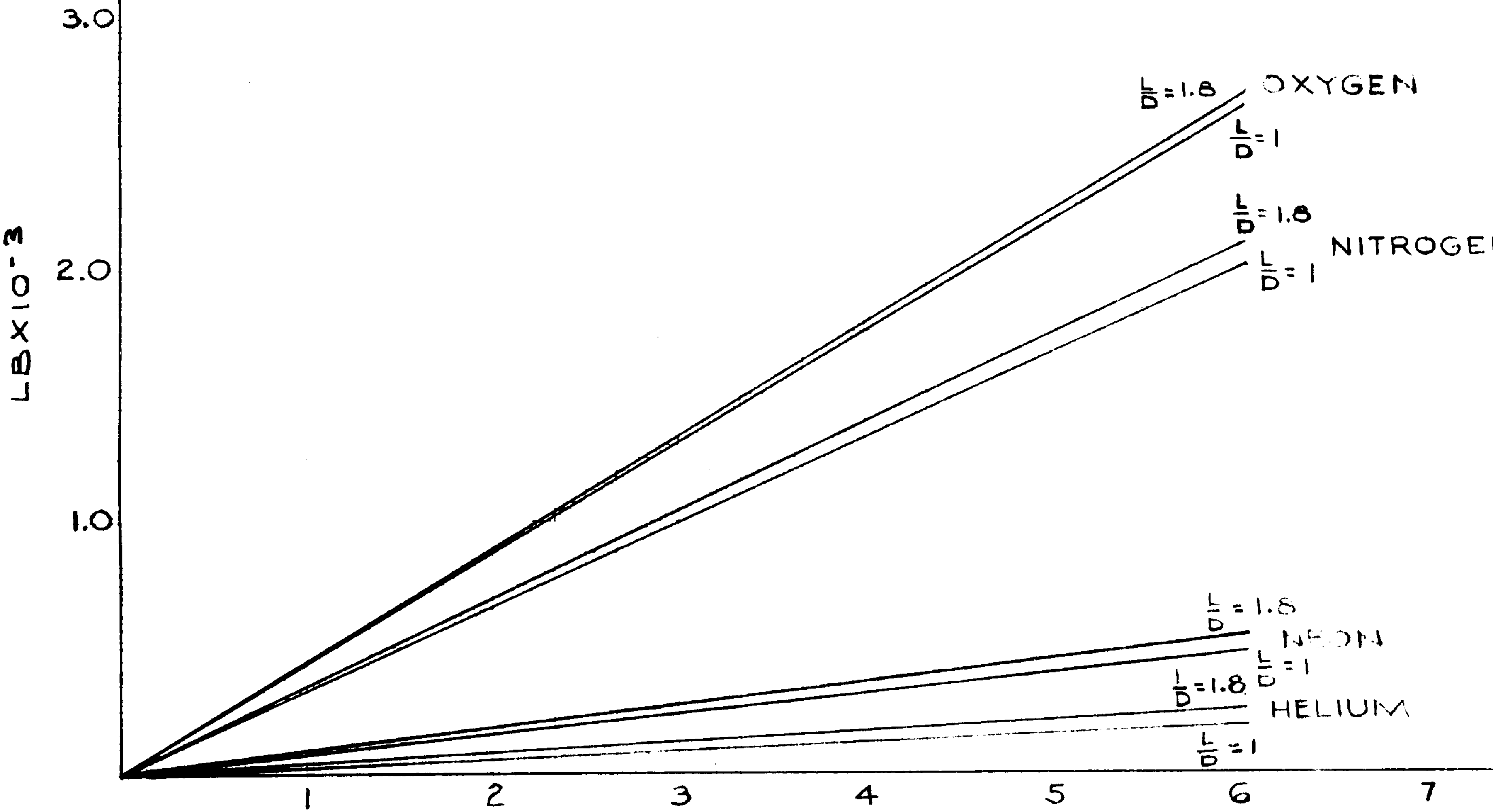


FIGURE 5.30

CRYOGENIC STORAGE SYSTEM WET WEIGHT AS A FUNCTION OF MISSION TIME - 3 MEN, 730 DAYS

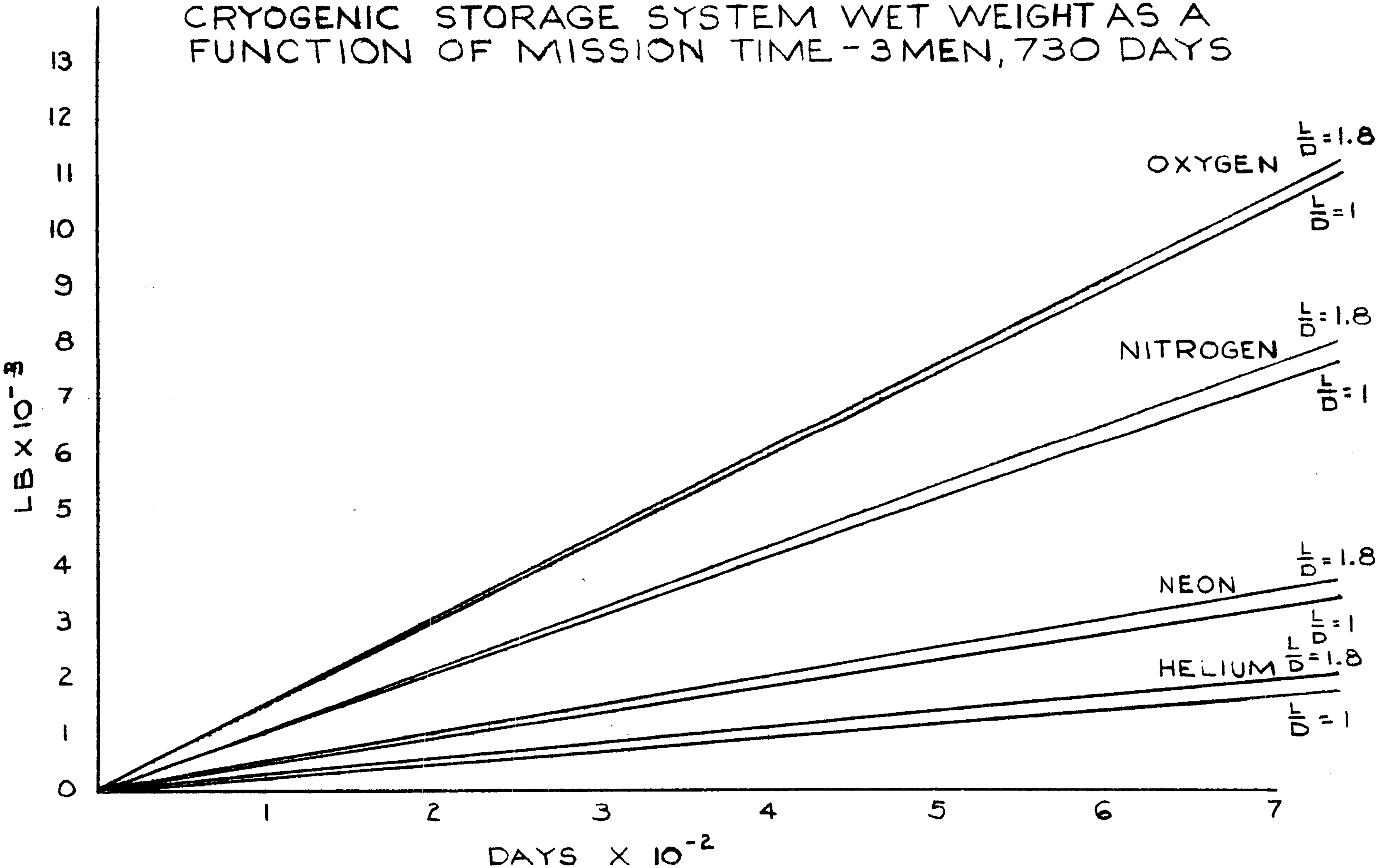


FIGURE 5.31

CRYOGENIC STORAGE SYSTEM WET WEIGHT AS A FUNCTION OF MISSION TIME - 3 MEN, 180 DAYS

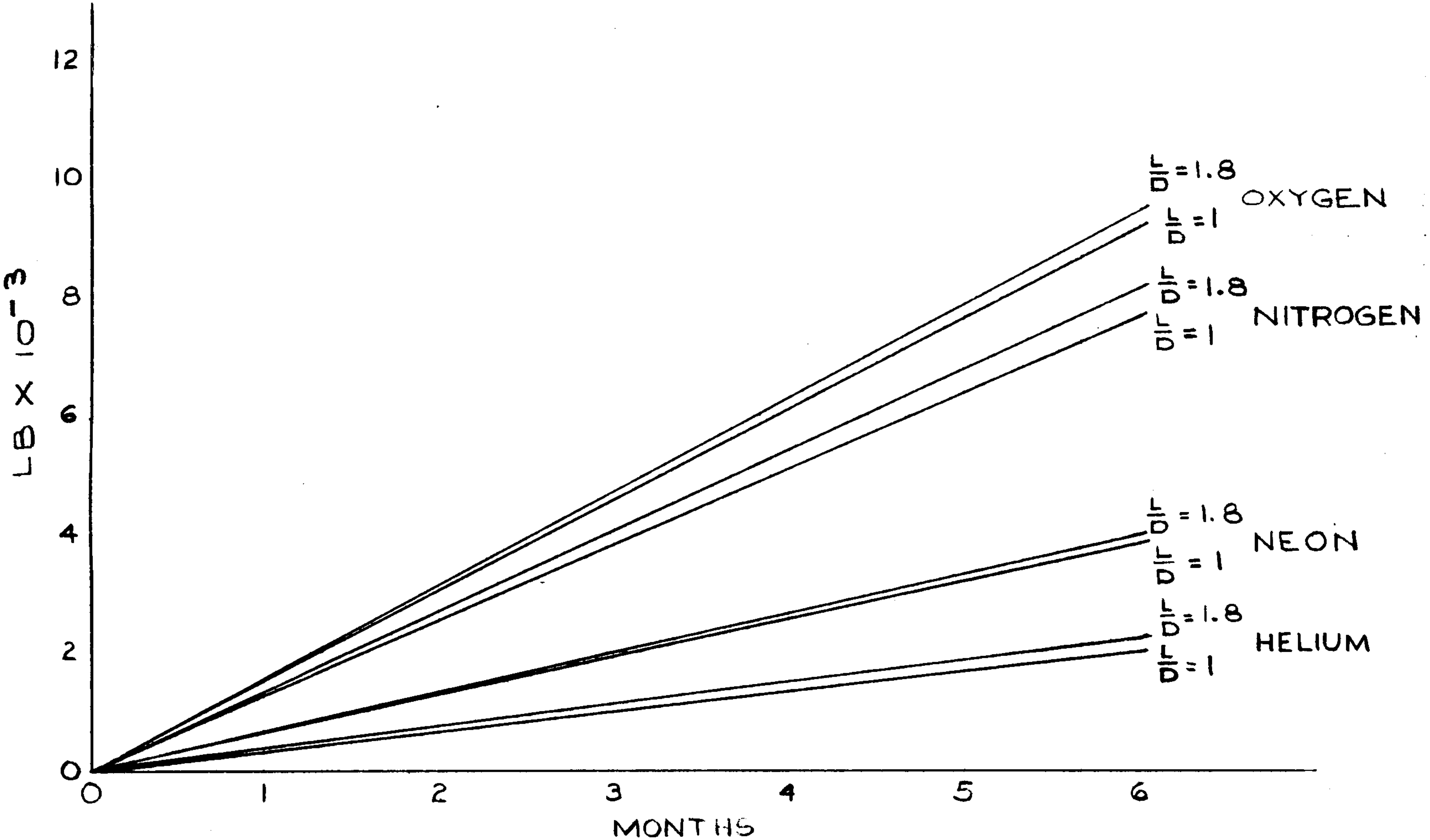


FIGURE 5.32

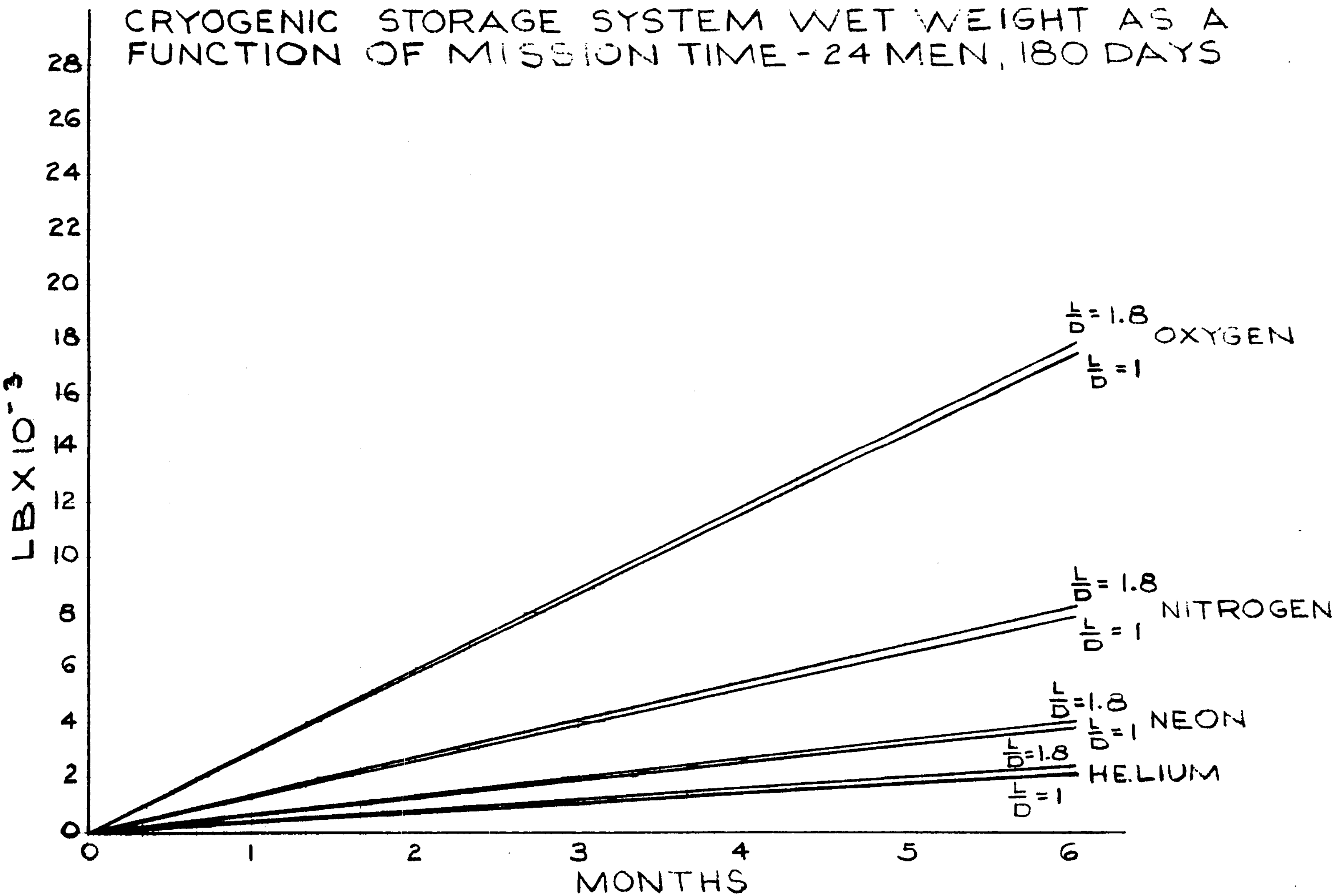

CRYOGENIC STORAGE SYSTEM WET WEIGHT AS A FUNCTION OF MISSION TIME - 24 MEN, 180 DAYS
LB X 10^-3
28
26
24
22
20
18
16
14
12
10
8
6
4
2
0
0
1
2
3
4
5
6
MONTHS
L/D = 1.8
L/D = 1
OXYGEN
L/D = 1.8
L/D = 1
NITROGEN
L/D = 1.8
L/D = 1
NEON
L/D = 1.8
L/D = 1
HELIUM

FIGURE 5.33

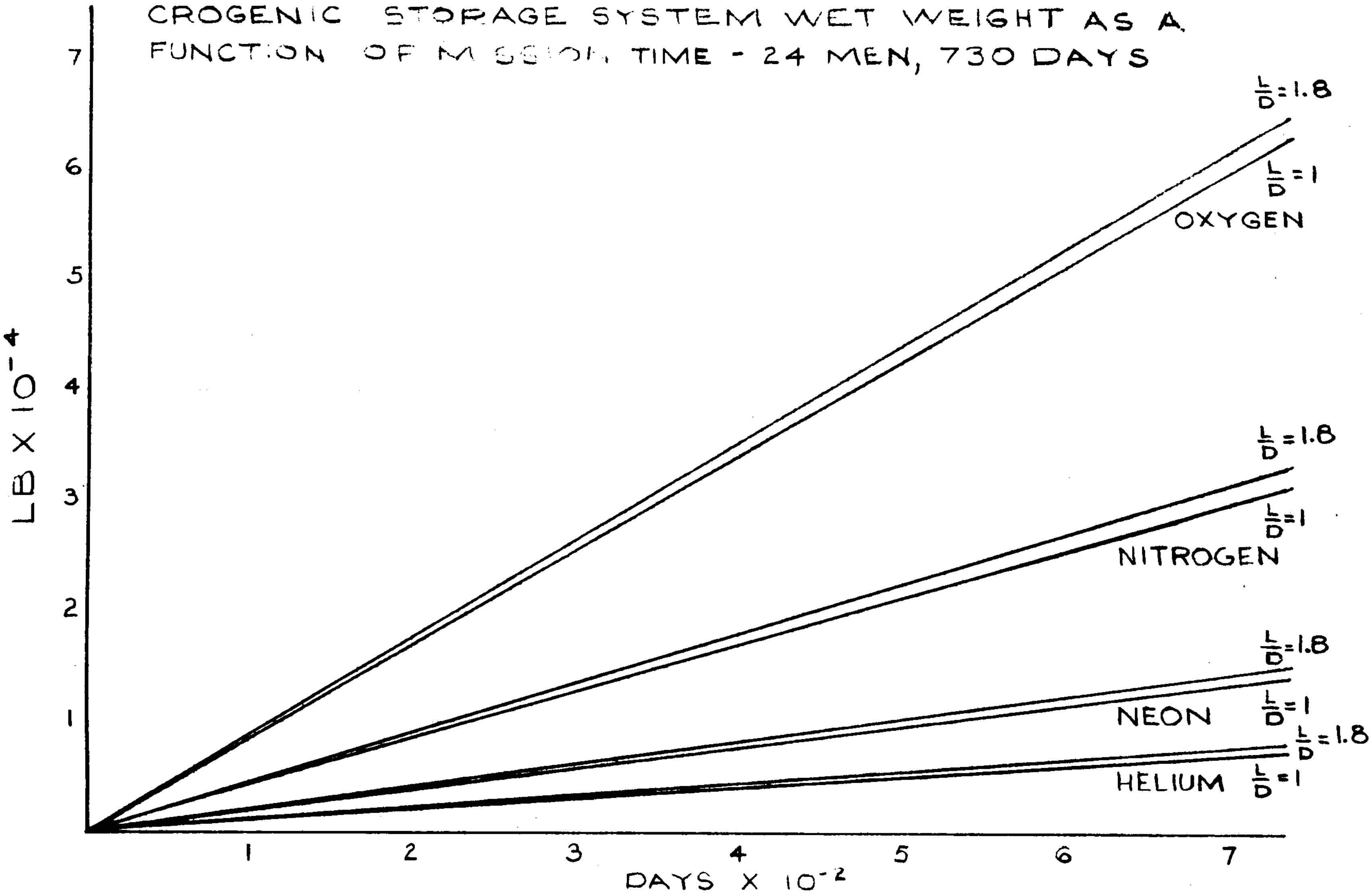
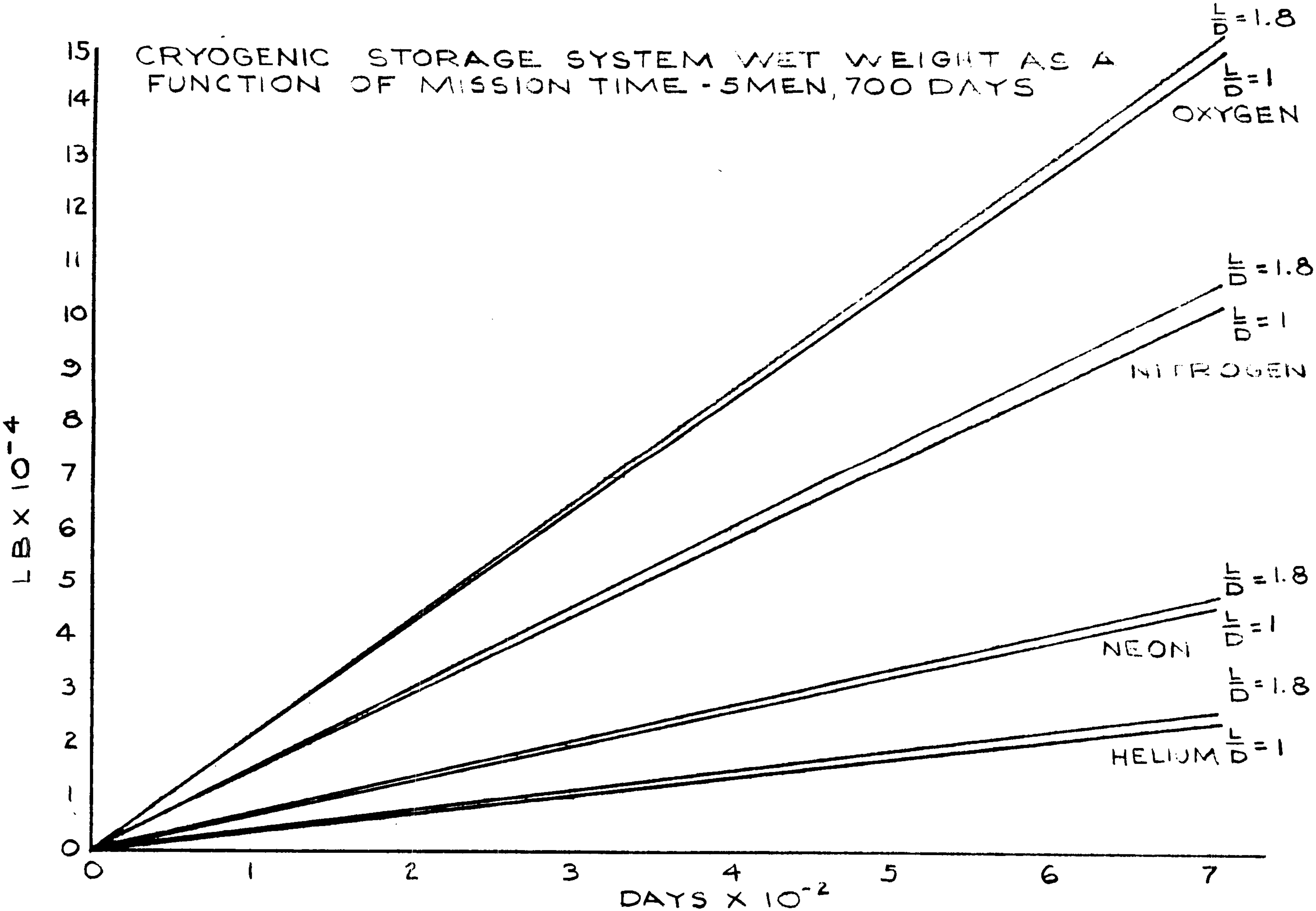
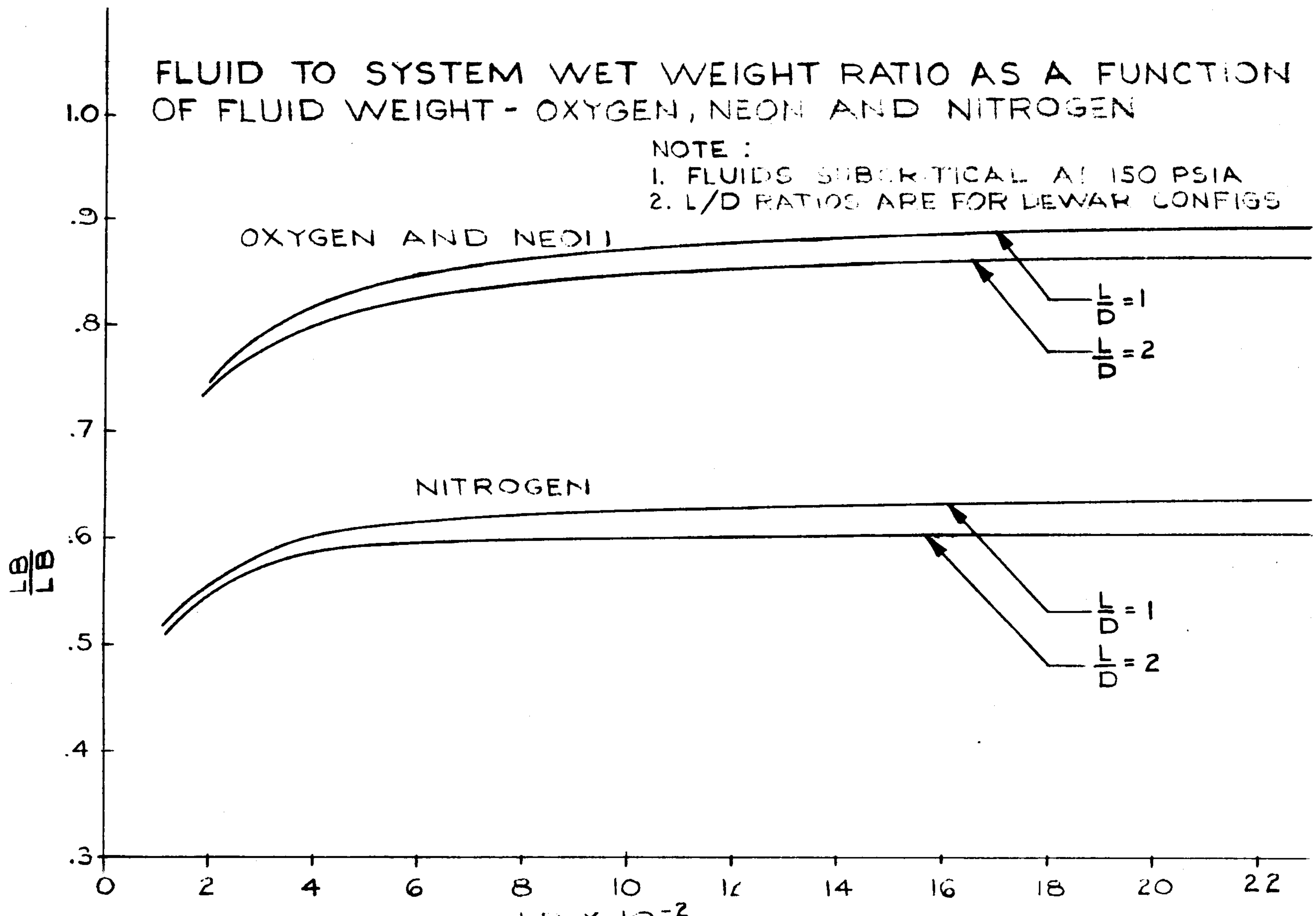

CROGENIC STORAGE SYSTEM WET WEIGHT AS A FUNCTION OF MISSION TIME - 24 MEN, 730 DAYS
LB X 10^{-4}
DAYS X 10^{-2}
OXYGEN
NITROGEN
NEON
HELIUM
$\frac{L}{D}=1.8$
$\frac{L}{D}=1$
1
2
3
4
5
6
7

FIGURE 5.34



CRYOGENIC STORAGE SYSTEM WET WEIGHT AS A FUNCTION OF MISSION TIME - 5MEN, 700 DAYS
LB X 10^-4
DAYS X 10^-2
L/D = 1.8
L/D = 1
OXYGEN
NITROGEN
NEON
HELIUM
0
1
2
3
4
5
6
7
8
9
10
11
12
13
14
15

FIGURE 5.35


FLUID TO SYSTEM WET WEIGHT RATIO AS A FUNCTION OF FLUID WEIGHT - OXYGEN, NEON AND NITROGEN
NOTE :
1. FLUIDS SUBCRITICAL AT 150 PSIA
2. L/D RATIOS ARE FOR DEWAR CONFIGS
OXYGEN AND NEON
NITROGEN
L/D = 1
L/D = 2
L/D = 1
L/D = 2
LB/LB
1.0
.9
.8
.7
.6
.5
.4
.3
0
2
4
6
8
10
12
14
16
18
20
22
LB x 10^-2
FIGURE 5.36

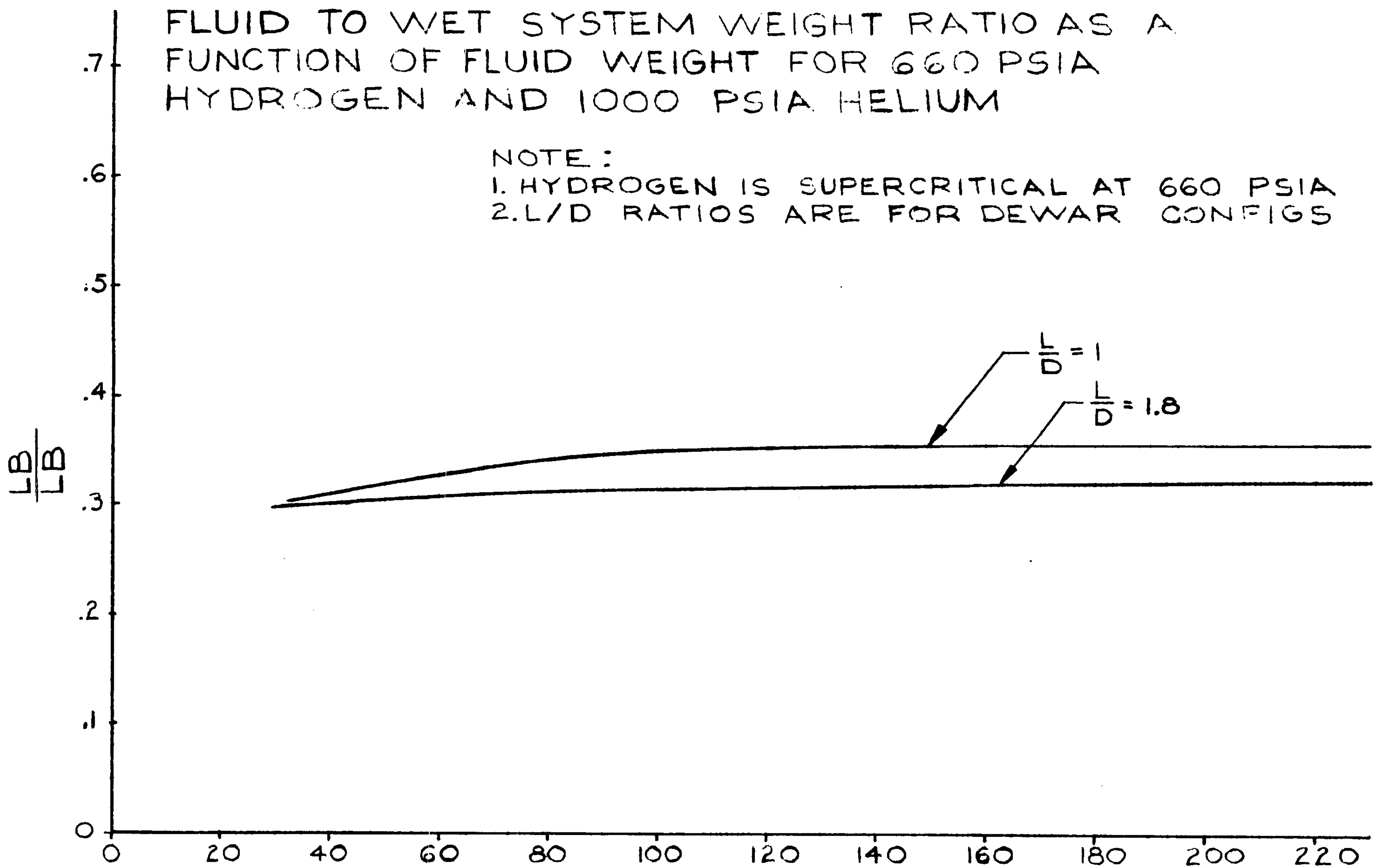

FLUID TO WET SYSTEM WEIGHT RATIO AS A
FUNCTION OF FLUID WEIGHT FOR 660 PSIA
HYDROGEN AND 1000 PSIA HELIUM
NOTE:
1. HYDROGEN IS SUPERCRITICAL AT 660 PSIA
2. L/D RATIOS ARE FOR DEWAR CONFIGS
$\frac{L}{D} = 1$
$\frac{L}{D} = 1.8$
$\frac{LB}{LB}$
.7
.6
.5
.4
.3
.2
.1
0
0
20
40
60
80
100
120
140
160
180
200
220
LB

FIGURE 5.37

CRYOGENIC SYSTEM DEVELOPMENT SCHEDULE

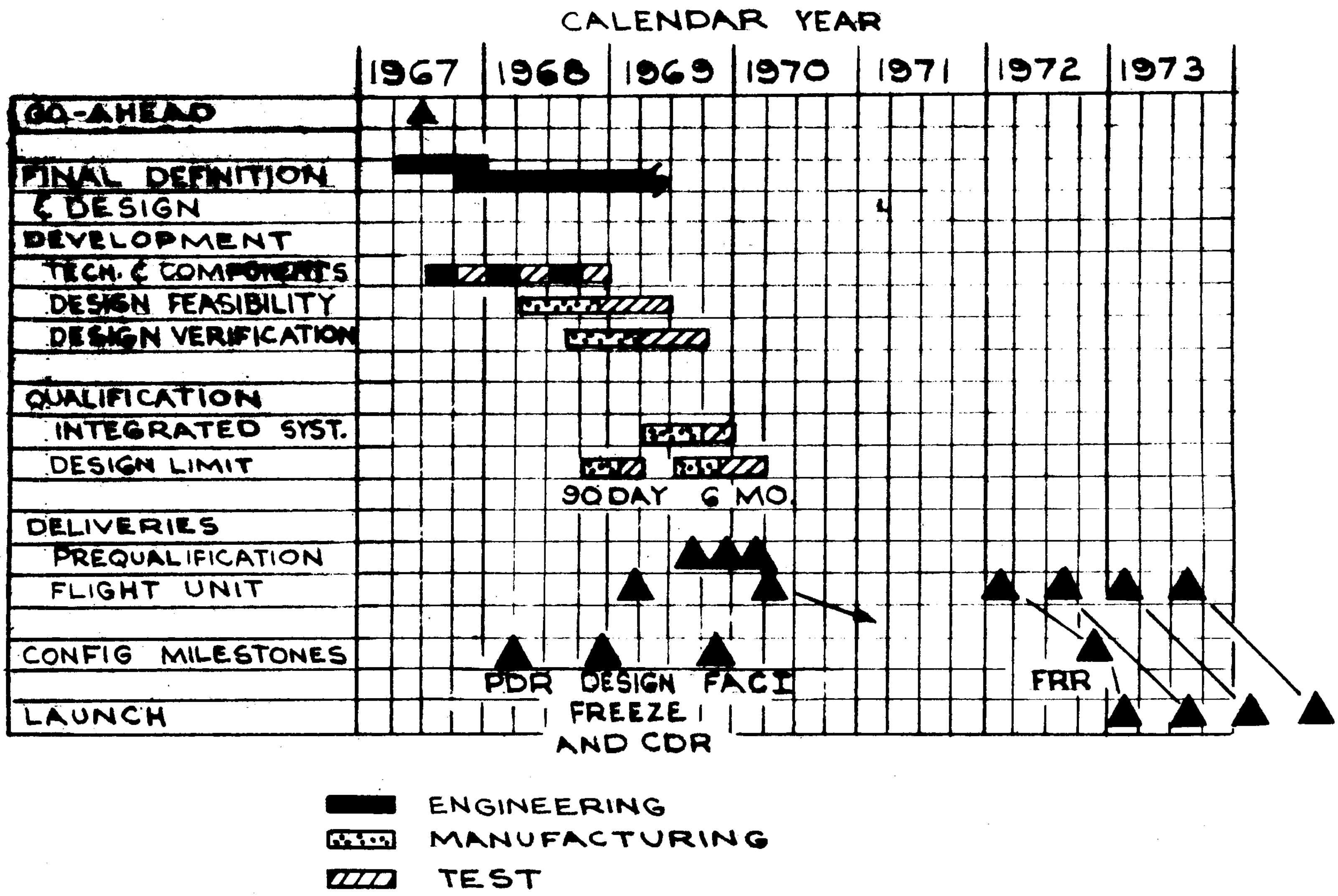


FIGURE 5.38

EFFECTS OF SIZE ON CRYOGENIC TANK PRODUCTION COST

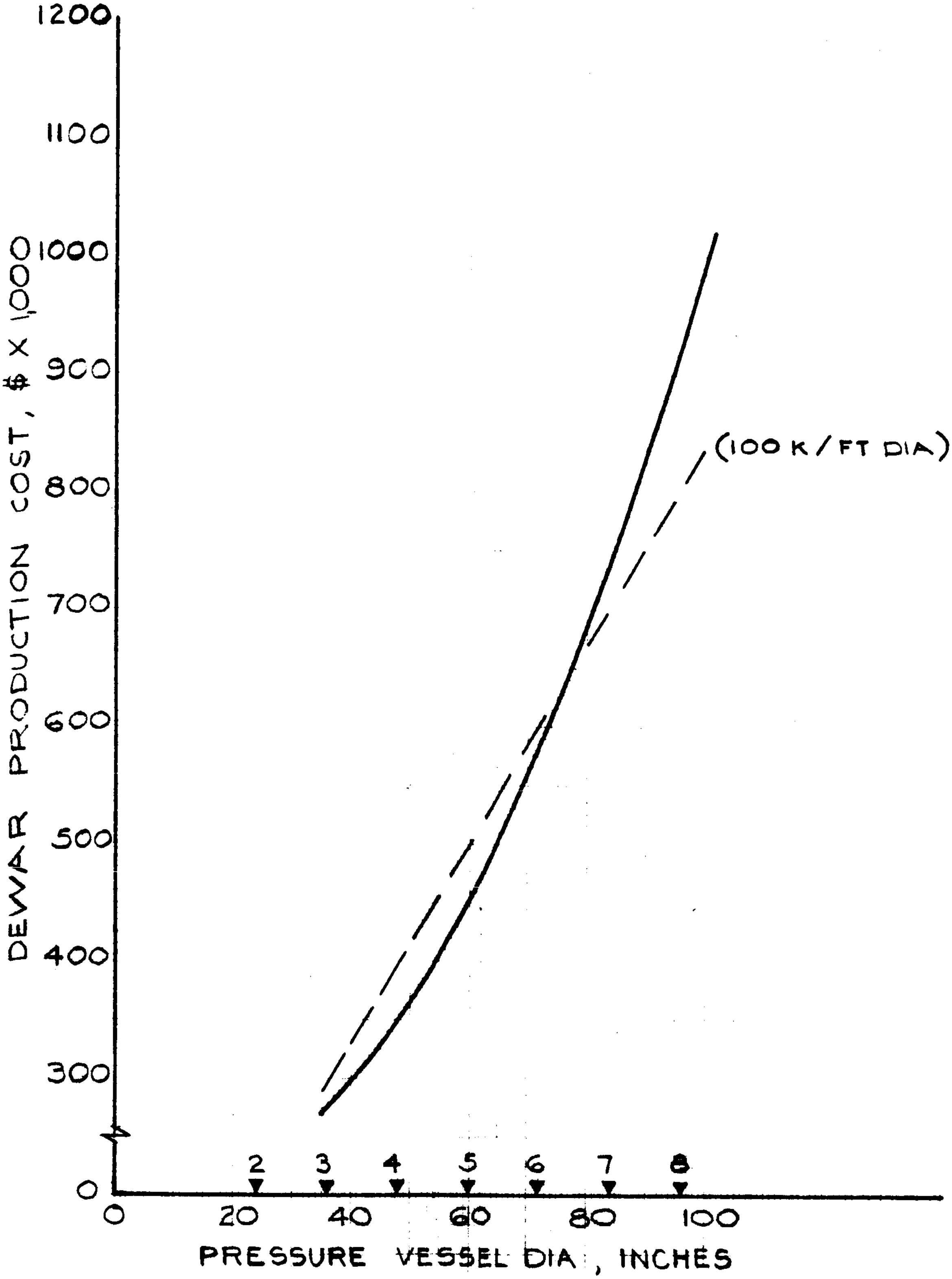


FIGURE 5.39

DEWAR COST AS A FUNCTION OF FLUID WEIGHT
SINGLE TANK CONFIG

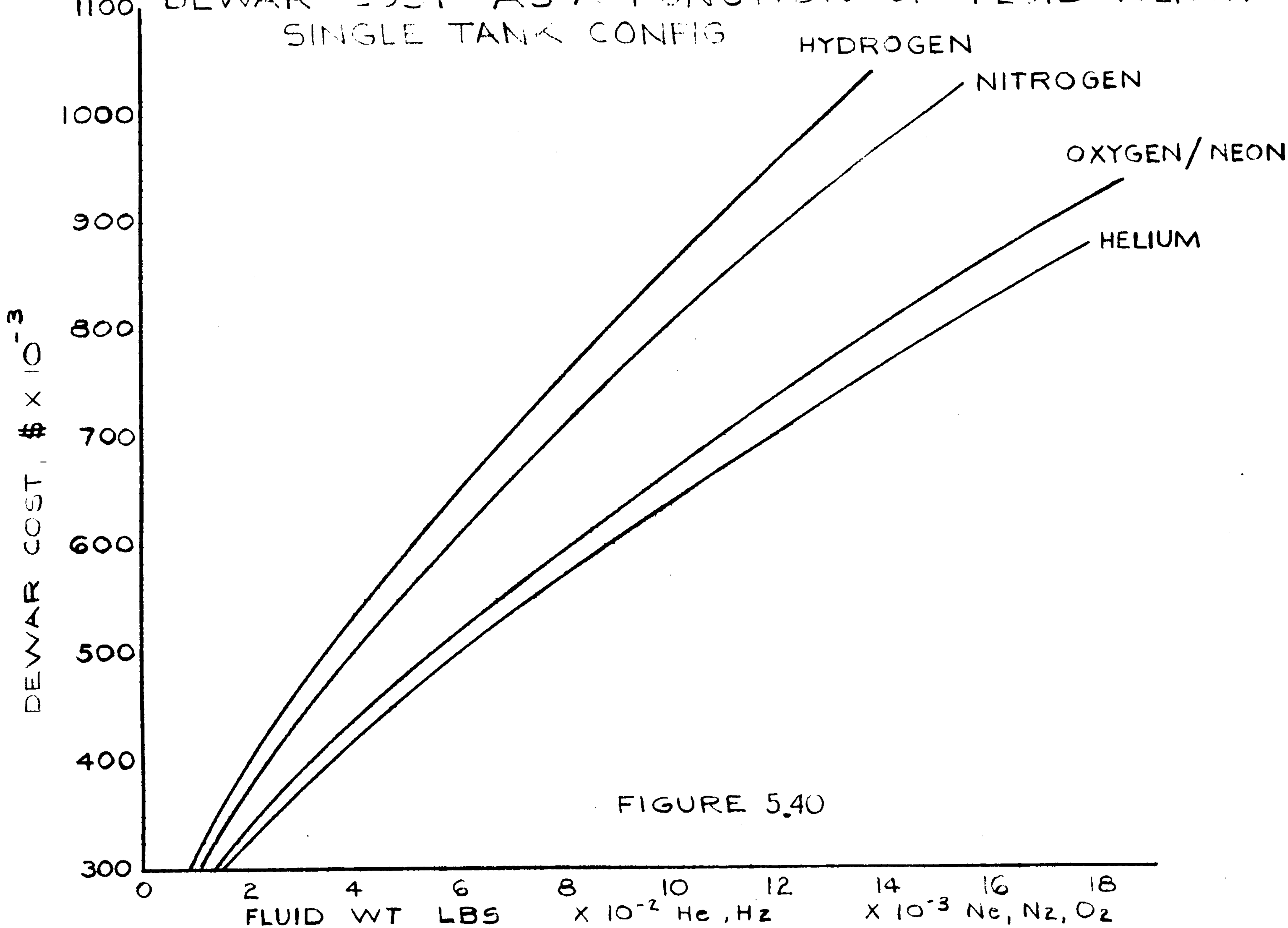


FIGURE 5.40

SHIPSET COST AS A FUNCTION OF FLUID WT FOR MULTIPLE TANK STORAGE

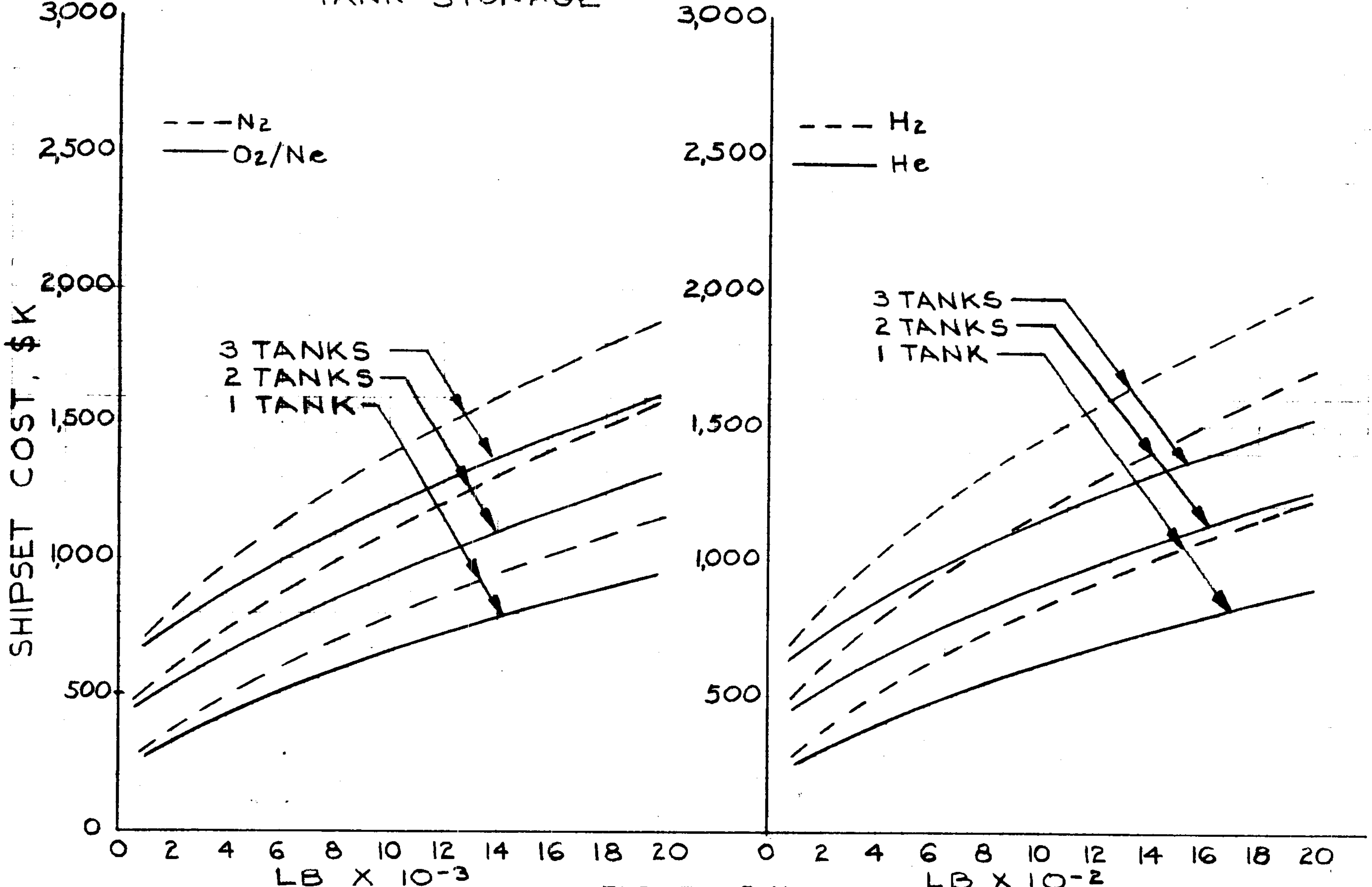


FIGURE 5.41

COST FOR FOUR SHIPSETS AS A FUNCTION OF NUMBER OF TANKS IN SHIPSET

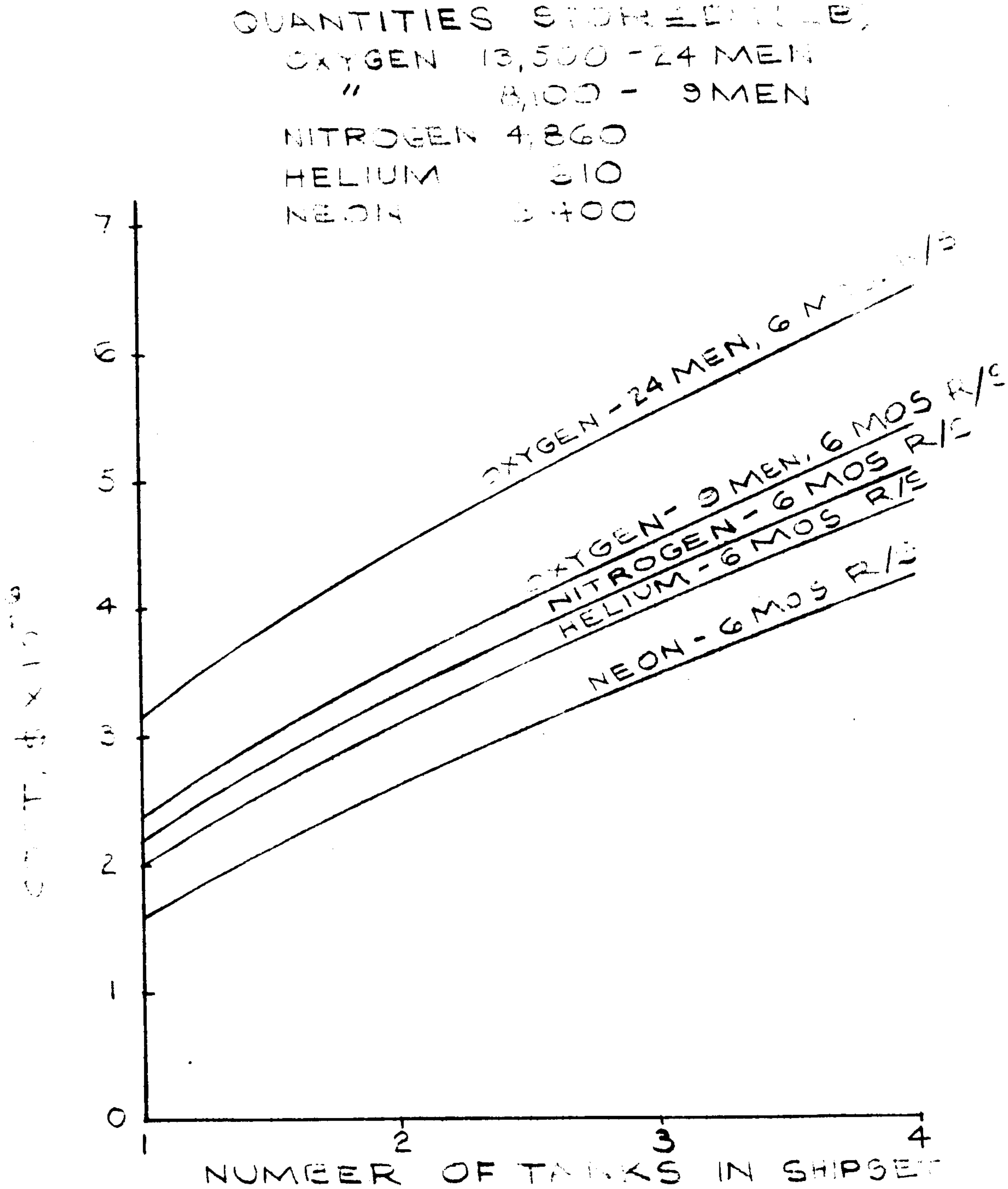


FIGURE 5.42

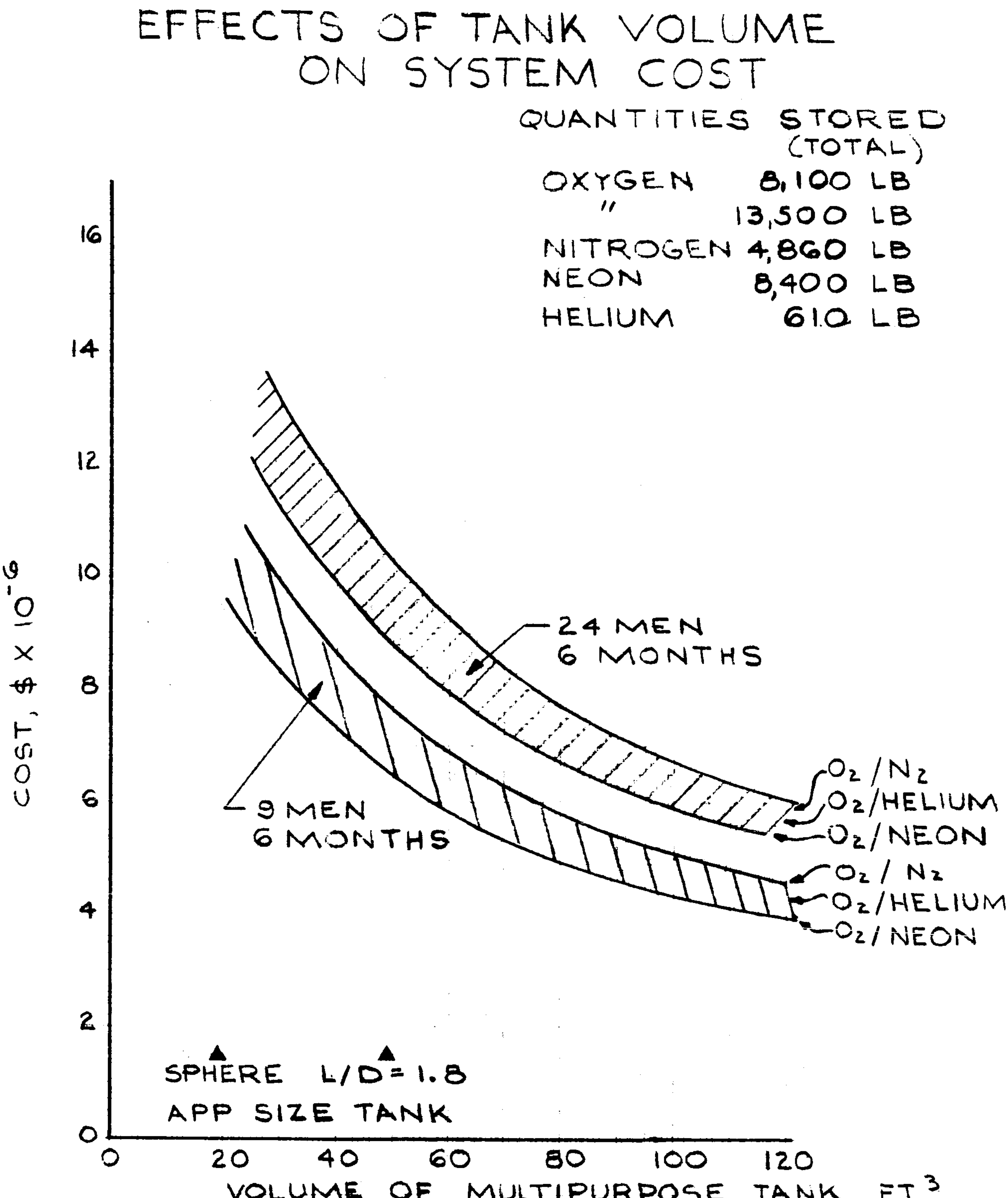

EFFECTS OF TANK VOLUME
ON SYSTEM COST
QUANTITIES STORED
(TOTAL)
OXYGEN 8,100 LB
" 13,500 LB
NITROGEN 4,860 LB
NEON 8,400 LB
HELIUM 610 LB
24 MEN
6 MONTHS
9 MEN
6 MONTHS
O2 / N2
O2 / HELIUM
O2 / NEON
O2 / N2
O2 / HELIUM
O2 / NEON
SPHERE L/D = 1.8
APP SIZE TANK
COST, $ X 10^-6
0
2
4
6
8
10
12
14
16
0
20
40
60
80
100
120
VOLUME OF MULTIPURPOSE TANK, FT^3

FIGURE 5.43

CRYOGENIC PROGRAM COST AS A FUNCTION OF NUMBER OF FLIGHTS

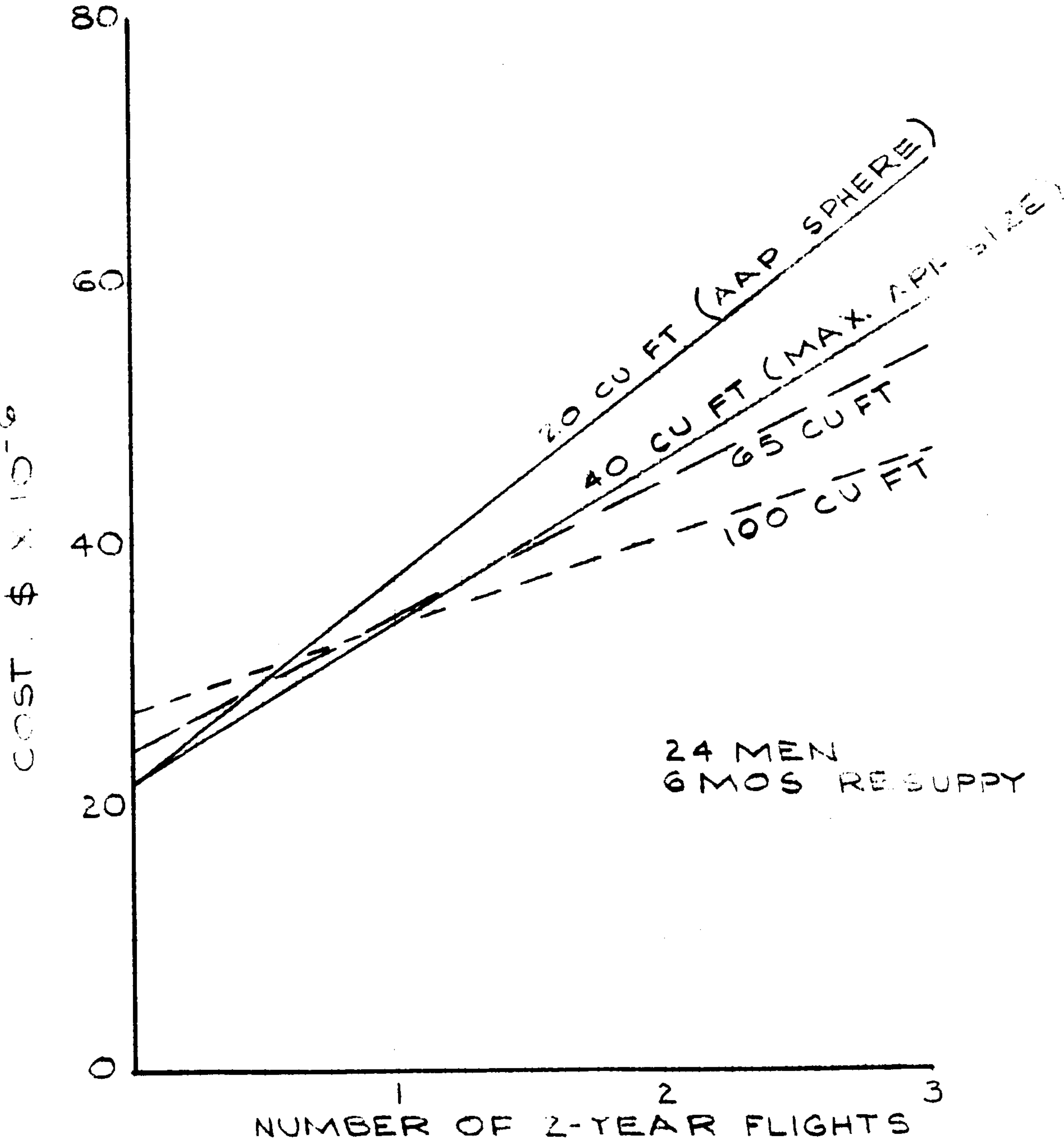


FIGURE 5.4

COST TRENDS USING MULTPURPOSE TANKS

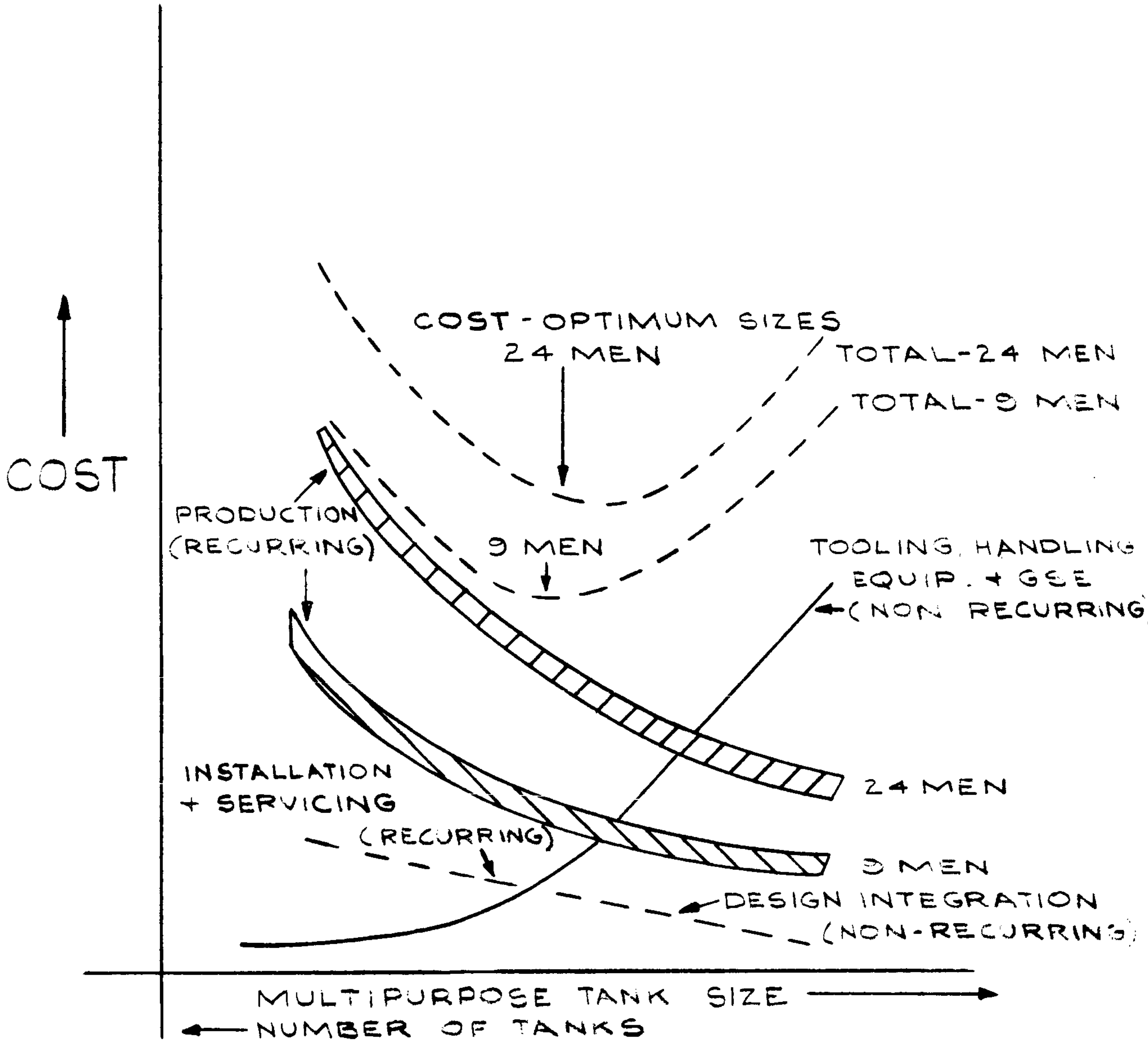


FIGURE 5.45

UTILIZATION OF AAP DEWARS FOR
OXYGEN STORAGE

TABLE 5.1

CRYOGEN: OXYGEN

OPERATING PRESSURE: 150 PSIA

Mission Type	Mission Duration Days	Resupply Period Months	Crew Size No. of Men	Cryogen Required Lbs	No. of AAP dewar (@ 2,60 lbs O_2 ea)	Dewar L/D Ratio	Dewar O.D. In.	Dewar O.L. In.
Space Station	---	6	9	8,100	(3.12) 4	1.75	41.5	73.0
Space Station	---	6	24	13,500	(5.2) 6	1.75	41.5	73.0
Earth Synchr.	---	6	3	2,340	(0.9) 1	1.75	41.5	73.0
Mars Flyby	700	---	5	13,000	(5.0) 5	1.75	41.5	73.0

TABLE 5.1 (Cont'd)

CRYOGEN: OXYGEN

OPERATING PRESSURE: 150 PSIA

System Q1/A (Max) B/hr-Ft^2	Flow Rate Average Lb/hr	System Heat Leak (Max) B/hr	Envir. Temp. (Max) ^{o}F	Refrig. System Energy B/hr	Refrig-erator Wt. Lbs	Refrig. Power Unit Wt Lbs	Dewar Wet Wt. Lbs	System Wet Wt. Lbs
.85	1.87	157.5	+75	---	---	---	3,050 each	12,200
.80	3.13	243	+68	---	---	---	3,050 each	18,300
.75	0.54	42	+61	---	---	---	3,050 each	3,050
.18	0.775	61	-140	470	100 ea.	42 ea	3,050 each	15,960

UTILIZATION OF AAP DEWARS
FOR HYDROGEN STORAGE

TABLE 5.2

CRYOGEN: HYDROGEN

OPERATING PRESSURE: 660 PSIA

Mission Type	Mission Duration Days	Resupply Period Month	Crew Size No. of Men	Cryogen Required Lbs.	Dewars AAP @ 163 lbs H_2 Ea. No.	Dewar L/D Ratio	Dewar O.D. In.	Dewar O.L. In.
2 KW Ave. Power	45	---	---	205	(1.25) 2	(1.75)	41.5	73.0
3.17 KW	45	---	---	326	(2.0) 2	"	"	"
2 KW	90	---	---	410	(2.5) 3	"	"	"
1.59 KW	90	---	---	326	(2.0) 2	"	"	"
2 KW	120	---	---	550	(3.4) 4	"	"	"
1.19 KW	120	---	---	326	(2.0) 2	"	"	"
2 KW	180	---	---	820	(5.0) 5	"	"	"
0.8 KW	180	---	---	326	(2.0) 2	"	"	"

CRYOGEN: HYDROGEN

TABLE 5.2 (Cont'd)

OPERATING PRESSURE: 660 PSIA

System Q1/A (Max) B/hr-Ft^2	Flow Rate Average Lb/hr	System Heat Leak (Max) B/hr	Envir. Temp. (Max) ^{o}F	Refrig. System Energy B/hr	Refrig-erator Wt. Lbs	Refrig. Power Unit Wt Lbs	Dewar Wet Wt. Lbs	System Wet Wt. Lbs
.25	.19	31	+50	---	---	----	510 ea	1,020
.40	.30	49	+85	---	---	---	510 ea	1,020
.166	.19	31	-15	600	115	52.8	510 ea	1,700
.197	.15	24.5	+10	341	105	30.0	510 ea	1,155
.121	.19	30	-55	1,120	135	98.5	510 ea	2,274
.145	.113	18	-35	935	120	82	510 ea	1,222
.097	.19	30	-65	1430	130	126	510 ea	2,806
.0965	.075	12	-65	570	132	55	510 ea	1,207