CONTRACT NAS9-10925

FINAL REPORT

Pre-Phase A Study for an Analysis of a Reusable Space Tug

VOLUME 5

SUBSYSTEMS

Space Division

North American Rockwell

NAS9-10925

CR-114939

SD 71-292-

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MARCH 22, 1971

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FOREWORD

This volume presents a subsystems analysis of the Prephase A Study for an Analysis of a Reusable Space Tug. This study was conducted by the Space Division of North American Rockwell Corporation, Seal Beach, California, for the National Aeronautics and Space Administration, Manned Spacecraft Center, Houston, Texas. The effort was performed under Contract NAS9-10925. The six volumes comprising this final report include:

Volume 1.	Management Summary	SD 71-292-1
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Volume 3.	Mission and Operations Analysis	SD 71-292-3
Volume 4.	Spacecraft Concepts and Systems Design	SD 71-292-4
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ACKNOWLEDGMENTS

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ABSTRACT

Subsystem trade studies were conducted to select feasible approaches and to define operational and performance requirements for the reusable space tug. The study areas included environmental control and life support, active thermal control, guidance and navigation, communications and data management, electrical power, and auxiliary control propulsion. In most areas, equipment selections were drawn from descriptions of similar equipment used in Apollo, Skylab, Earth Orbital Space Station, or Earth Orbital Shuttle. In many cases, to more optimally fit specific tug requirements, modifications to the selected equipment were described.

Of primary emphasis was the feasibility of the concepts of reusability, multimission capability, space-basing, and autonomy from excessive external support. Implementation approaches to these concepts included functional modularity, common propellant tankage, and integrated electronics; each of which also required a feasibility analysis.

A baseline set of subsystems was established at the study midterm, including the description of selected equipment for inclusion in intelligence, propulsion, and crew modules. When dividing equipment between modules, consideration was given to mission functional requirements, modular interface complexity, and cost in terms of development economy and minimum inert weight.

The intelligence module contains astrionics (guidance, navigation, communications, and data management), electrical power, active thermal control, and auxiliary control propulsion. Main engines and propellant are stored in the propulsion module. The crew module includes all of the environmental control and life support equipment as well as the crew interface equipment, such as the manual controls, displays, and access to the computer and communications equipment.

To minimize inert weight, manned and unmanned, space-operated and lunar landing versions of the tug differ significantly. Each version, however, utilizes a nucleus of basic equipment to which is added internal packages and external kits to afford mission capability. The added equipment not only increases the number and capacity of functions but in many cases also enhances mission success and crew safety by increasing component redundancy.

Another cross-sectional analysis of subsystems led to the definition of autonomy levels and the differences between space and ground basing.

ABBREVIATIONS

The following abbreviations are used in this document:

ACS	Auxiliary control subsystem
ACPS	Auxiliary control propulsion subsystem
APCU	Auxiliary propulsion conditioning unit
ATC	Active thermal control subsystem
CAM	Cargo module
CIS	Chemical interorbital shuttle
CM	Crew module
COMM	Communications
C/O	Checkout
CSM	Apollo Command and service module
EC/LSS	Environmental control and life support system
EO	Earth orbit
EOS	Earth-orbital shuttle (two-stage reusable)
EOSS	Earth-orbital space station
EPS	Electrical power subsystem
FO	Fail operational
FS	Fail safe
G&N	Guidance and navigation subsystem
IM	Intelligence module

IPP	NASA integrated program plan
IOC	Initial operational capability (date)
IMU	Inertial measurement
IRU	Inertial reference unit
LEO	Low earth orbit
LG	Landing gear kit
LM	Lunar module
LSB	Lunar surface base
MEP	Master executive program
MK	Manipulator kit
OLS	Orbiting lunar station
OMS	Orbital maneuvering system
OPD	Orbital propellant depot
PM	Propulsion module
RNS	Reusable nuclear shuttle
RST	Reusable space tug
TDRSS	Tracking and data relay satellite system
TS	Tank set
T/W	Thrust to weight

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1.0 INTRODUCTION

1.1 OBJECTIVES

The tug subsystems perform functions which are generally the same as those accomplished by existing upper stages and are critical to successful and safe operation. The primary objective of the subsystems analysis task was to define candidate subsystems, perform the basic tradeoffs, and select and further define the characteristics of the best subsystems for the tug. These characteristics include relevant weight, volume, and power effects on vehicle sizing; operating details to the extent of mission influence; and redundancy, reliability, and replacement requirements which affect cost. The criteria for selecting subsystems are identical with those of the entire tug: a low-cost, flexible, long-lived, highly reliable space system with a high degree of reusability and commonality with other Integrated Program Plan elements. Parametric data, where applicable, were generated to determine trends and regions of requirement compliance. Design point data were developed for use in vehicle performance studies. Key issues explored during the subsystems study include:

1. The feasibility of multipurpose reusable subsystems
2. The feasibility of modular vehicle design from the standpoint of subsystems
3. The feasibility of autonomous operation
4. Potential problems in the use of common cryogenic tankage for main propulsion, auxiliary propulsion, electrical power, and crew oxygen
5. Potential problems associated with docking: unrestricted lighting conditions, compatibility with the requirements of other vehicles, and cargo handling
6. Mission resupply requirements
7. The feasibility of onboard active refrigeration/reliquifaction cycling of cryogenics

8. Potential problems associated with lunar landing: unrestricted lighting conditions and unmanned landing

9. Subsystem requirements for rescue operations

In pursuing these objectives, five reasons dictated more than cursory studies of the subsystems:

1. In several areas, particularly guidance, navigation, and communication, the choice of components affected the vehicle operational sequence and criticality.

2. Component choice is moderately influenced by exterior envelope constraints. Since the tug must fit within the earth orbital shuttle cargo bay, auxiliary control jets, antennae, navigation instruments, and outgassing requirements are affected. Solar cell arrays (if they should be chosen), landing gear, manipulators, and other exterior structures must have erectable features.

3. Subsystem costing data may be more confidently prepared if the approximate technology period, the complexity, and typical previous program use of equipment are established.

4. Small variations in component weight and power produce proportional variations in the structure, cooling system, power reactants, and propellants. The result is a compounding effect which significantly influences vehicle gross weight.

5. It is necessary to assess the cost of space-based over ground-based vehicles, manned over unmanned vehicles, and reusable over expendable vehicles in order to fully justify these aspects. In the subsystems area, these costs are closely related to the weight and technology level; their accurate assessment is therefore influenced by decisions made at the component level.

The influence of operational sequence, criticality (redundancy), and envelope considerations form the major requirement sources and receive most of the attention in this section. Discussions of costs are treated in a separate section.

As a rough estimate of the magnitude of vehicle gross-weight sensitivity to component-weight variations, an increase of 1 pound (0.5 kilograms) in subsystems equipment causes an increase of approximately 8.5 pounds (3.8 kilograms) in vehicle gross weight for a geosynchronous mission. For the selected equipment, including redundancy effects, an average power increase of one watt during a 7-day mission causes an increase of:

0.410 pounds (0.185 kilograms) of electrical power equipment

0.225 pounds (0.102 kilograms) of active thermal control equipment

0.176 pounds (0.080 kilograms) of reactants

0.608 pounds (0.275 kilograms) of vehicle structure

6.080 pounds (2.750 kilograms) of propellant for main propulsion

Adding these increments, it is found that the one watt average increase over a 7-day mission costs approximately 7.5 pounds (3.4 kilograms). The power sensitivity is therefore 88 percent of the weight sensitivity in a user subsystem. In the usual case, an increase in component weight will cause an increase in power demand, which causes the two sensitivities to be additive. It should also be noted that although the subsystem weights estimated for the tug may, for example, be only 5 percent in error from final design figures, they could, at the same time, easily represent a gross weight uncertainty of 2000 pounds (900 kilograms) if the errors are in power-consuming components.

On this basis, the somewhat detailed attention paid to subsystems during the Reusable Space Tug study is justified.

1.2 SUBSYSTEM REQUIREMENTS

The preliminary subsystems requirements were derived from mission requirements and interface groundrules. These are listed in Table 1-1 and are cross-referenced with specific RFP statements (Reference 1-1). These requirements assume a ten-mission, space-based vehicle with primary automatic operation and backup manual operation. Common oxygen/hydrogen storage for main propulsion, auxiliary propulsion, electrical power (since fuel cells are recommended), and crew oxygen are of principal interest. Guidelines for equipment reliability and space maintenance were undefined.

The first item in the table, which deals with launch vehicle options, was reduced in scope shortly after the contract effort began, to include only internal earth orbital shuttle launches. The resulting intelligence module (IM) contains no compromises to permit use as an alternative to the Saturn-V instrument unit (IU). Although this subject was not pursued in depth, it appears that the IM contains excess capability over that of the IU. The IM could be used effectively as an IU during launch and, with ground support, could be reprogrammed in orbit for subsequent space-based missions.

The fifth item in the table specifies reliability considerations. The original interpretation was a fixed redundancy in all equipment areas. A

Table 1-1. Preliminary Subsystems Requirements

Requirements/Driver	Subsystems Interpretation	Subsystem: Environmental Control and Life Support	Guidance, Navigation, and Control	Communications and Data Management	Electrical Power	Auxiliary Control and Auxiliary Propellant
1. Launch - The space tug may be launched from earth on the Saturn, Saturn derivatives, and possibly smaller vehicles such as Titan with or without use of the tug's propulsion, or as an internal or external payload on the earth orbital shuttle (EOS). Limited consideration shall be given to using the intelligence module (IM) to replace the functions of the Saturn-V (S-V) instrument unit (IU) when the space tug is flown as an S-V fourth stage. (3. 1b, (3. 1b, 4. 1. 1)*	Launched by EOS, S-V, S-V derivatives, and smaller vehicles. IM may be used as an alternate to the S-V IU.	No interfaces.	No interfaces; alternative is to use IM as IU.	External antenna size constraints, radio link interface compatibility with launch vehicles and ground. Alternative is to use IM as IU.	No interfaces; alternative.	ACS jet propulsion constraint, propellant venting constraint during launch. Alternative is to use IM as IU.
2. Space based capability - Many earth orbit missions will begin at and return to the space base (270 nmi al, 55° inclin). Other inclinations and altitudes will be considered, such as 28 1/2° and 200 nmi alt. The space tug will be refuelable in earth or lunar orbit. All communication systems are to be compatible with the Manned space flight network, deep space network and available communication satellite systems and all hardware elements of the IPP. The space tug will incorporate neuter docking devices compatible with all IPP hardware elements. Minimum interfaces shall be required between the payload and the spacecraft to reduce complexity and increase the flexibility of the kinds of payloads to be transported; however, consideration should be given to how the space tug communications and power subsystems could support the payload. (3. 1c, d, v, x, y)*	Space based, active or passive docking capability at any time with all IPP elements. Space and ground communications interface compatibility. Minimize interfaces with ground, payloads, and other vehicles.	Cabin atmosphere compatible with other IPP elements.	Automatic docking capability with manual override. Remote pilot docking capability. No sun angle constraints on docking. Docking visual aids required.	Voice and data link compatible with payload, other IPP elements, MSFN, DSN, and CSS. Television link required. Sensor measurements of attitude, range and rates Command receiver system.	Supports payload requirements.	ACS acceleration resolution for docking.
3. Operation - The space tug shall be capable of manned or automated flight. Remote control from the earth, earth-orbiting and lunar-orbiting stations/bases shall be considered. The space tug design shall minimize the necessity for ground support during flight. Autonomy is the design objective. (3. 1g, h)*	Manned or automated flight, autonomous or semi-autonomous operation.	Automatic onboard monitoring and checkout.	Automatic onboard monitoring and checkout. Guidance sequencer-programmer with backup remote and keyboard entry.	Automatic onboard monitoring and checkout. Systems status data transmission. Data processing equipment.	Automatic onboard monitoring and checkout.	Automatic onboard monitoring and checkout.

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Table 1-1. Preliminary Subsystems Requirements (Cont)

Requirements/Driver	Subsystems Interpretation	Subsystem: Environmental Control and Life Support	Subsystem: Guidance, Navigation, and Control	Subsystem: Communications and Data Management	Subsystem: Electrical Power	Subsystem: Auxiliary Control and Auxiliary Propellant
4. <u>Reusability and operating life</u> - The design shall be capable of maintaining a quiescent status for long periods (180 days) in earth and lunar orbit, docked to other vehicles and free flying. Quiescent periods up to 30 days or more may be required on the lunar surface. Quiescent capability shall exist before use, between uses, and after use. The space tug shall be capable of going from this quiescent status to a fully operational mode within approximately 2 hours. Refrigeration or other techniques may be considered to achieve quiescent requirements. The space tug shall be reusable at least ten times by refueling, replacement of consumables, and a minimum of refurbishment. Return to earth for major refurbishment should be considered. Replacement of components or subsystems in space should also be considered. The space tug shall have a reusable lifetime goal of 3 years or longer: i.e., the space tug shall be reusable by replacing consumables for a period of 3 years or longer after earth launch. (3.1i, j, k)*	Reusable at least 10 times over a 3-yr period with quiescent periods up to 180 days in space or 30 days on lunar surface. Space replacement of components or subsystems an alternative to ten-mission reliability.	Space refurbishment. Space expendable resupply. Space maintenance or adequate equipment life.	Space maintenance or adequate equipment life.	Space maintenance or adequate equipment life.	Space maintenance or adequate equipment life. Rechargeable fuel tanks.	Space maintenance or adequate equipment life. Rechargeable propellant tanks.
5. <u>Reliability</u> - It shall be a design objective to maximize crew safety and the probability of fulfilling all space tug functions and objectives. Subsystems identified as necessary for crew survival will be designed such that no single failure or credible combination of failures will result in a loss of life. Functions identified as necessary for safe continuation of planned space operations shall be designed for long life with minimum replacement, with single failure points prevented or controlled. (3.1l)*	Multiple redundancy as required.	Critical equipment redundant, spare parts stocked onboard for crew-maintained non-critical equipment. Margins for emergency expendables and consumables also stocked onboard.	Unmanned reliability equivalent to man-rating for docking and separation.	Links capable of remote tug operation as backup mode.	Fail-safe mode required for emergency rescue period.	Attitude-hold mode required for emergency rescue period.

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Table 1-1. Preliminary Subsystems Requirements (Cont)

Requirements/Driver	Subsystems Interpretation	Subsystem: Environmental Control and Life Support	Subsystem: Guidance, Navigation, and Control	Subsystem: Communications and Data Management	Subsystem: Electrical Power	Subsystem: Auxiliary Control and Auxiliary Propellant
6. Manned provisions - The space tug crew compartment will contain a flight station for control of all flight maneuvers. The crew compartment will serve as the primary crew living quarters and a base of operations for the lunar surface missions. It will also serve as a crew transport system when used with the nuclear shuttle. The crew compartment will contain an airlock for crew EVA. (3.1s, t, u)*	Crew module serves as control station, crew transport, and lunar surface base.	2 to 6 man crew. Periods of occupancy from a few hours to long duration orbital missions, or 14 to 28 days on lunar surface plus 14 days contingency plus additional flight time.	None.	None.	None.	None.
7. Intelligence module provisions - The feasibility of a separate IM containing electrical power generation, all astrionics, and an attitude control system shall be investigated. This IM might be used for all phases of the space tug operations and for astrionics, guidance and attitude functions on other hardware elements such as the orbiting lunar station (OLS), nuclear shuttle, and large payloads operating remotely in earth or lunar orbit. (4.3)*	Common basic subsystems for all applications.	None.	Guidance and control equipment in IM, with input/output to crew module.	C&DM equipment in IM with input/output to crew module.	Basic power source with add-on capability.	Constraints on number of jets vs location. Gaseous O_2/H_2 temporary storage.
8. Lunar surface provisions - The space tug will conduct lunar surface missions beginning at and returning to a near polar OLS at 60 n mi altitude. Some missions may be conducted from other orbits with the nuclear shuttle and without the OLS. Lunar surface mission crew size is between 3 and 6 crewmen. The space tug will be flyable normally by one crewman but capable of automated operations. Space tug operations shall not be constrained by lunar lighting conditions. The nominal 28 day manned lunar landing mission vehicle consists of a propulsion module, a separate or integral intelligence module, a crew compartment, an estimated 5000 lb of mobility aids and an estimated 5000 lb of scientific equipment all of which must be returned to lunar orbit in case of abort at touchdown. Mobility aids may include lunar flyers, ground effect machines and wheeled rovers. The scientific equipment and mobility	Lunar landing capability from any orbital altitude, inclination, or lighting conditions, with 0 to 6 crewmen. Landing capability at any latitude and longitude. Surface stays will be 28 days duration plus 14 days contingency, landing and returning a 10,000 lb payload plus crew or 14 days duration plus 14 days contingency.	Minimum atmosphere leakage losses. Maximum crew requirements are 28 plus 14 day crew provisions, plus additional in-flight provisions.	Automatic landing capability w/manual override. Remote pilot landing capability. No sun angle constraints on landing.	Landing sensor measurements of attitude, range, and rates. Selenographic landing point location equipment. Obstacle avoidance equipment.	Electrical power system supplies surface operation needs.	Conditioning unit charges accumulators on surface.

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Table 1-1. Preliminary Subsystems Requirements (Cont)

Requirements/Driver	Subsystems Interpretation	Subsystem: Environmental Control and Life Support	Subsystem: Guidance Navigation, and Control	Subsystem: Communications and Data Management	Subsystem: Electrical Power	Subsystem: Auxiliary Control and Auxiliary Propellant
8. Lunar surface provisions - (Cont) aids are described in the attached data package. Landings will occur at all lunar latitudes and longitudes with appropriate time phasing. Lunar landing mission duration is 28 days plus 14 days contingency capability or 14 days plus 14 days contingency. Both cases are to be investigated in this study. (3. 1m, n, o, p, q, r)*						
9. Attitude reference capability - The space tug attitude reference system shall be capable of extensive maneuvering on any axis without loss of attitude reference. (3. 1w)*	Three-axis rotation and translation auxiliary propulsion. Three-axis attitude reference system with complete freedom.	None.	3-axis rotation and translation. 3-axis attitude reference system with unlimited rotational freedom.	None.	None.	3-axis rotation and translation auxiliary propulsion.
10. Common LOX/LH_2 - The primary propulsion system will be LOX/LH_2. While existing or planned engines may be considered, it is intended that the space tug requirements design the engine within the constraints of existing and projected technology. No constraints are to be placed on the space tug design because of existing or planned engine designs. A new LOX/LH_2 engine will be acceptable as part of the pre preliminary design. The advantages and disadvantages of appropriate propellants other than LOX/LH_2 will be documented and summarized for comparison purposes. Auxiliary propulsion systems will emphasize LOX/LH_2; however, consideration and study of other propellants will be included. Feasibility of using common LOX/LH_2 tanks for both the primary propulsion system, auxiliary propulsion system, and for electrical power generating and life support functions shall be determined by conducting a tradeoff analysis between the common tankage arrangement and a more conventional, separate-tanks arrangement. The advantages of using the LOX/LH_2 for long-term survival in case of a contingency situation shall be determined. (3. 1z, 4. 3)*	Common LOX/LH_2 tankage for main propulsion, ACS, EPS, and crew atmosphere is a primary study item.	Oxygen for crew use stored in main tanks.	Autopilot design includes effects of 90 to 110-millisecond minimum jet pulse duration.	None.	LOX/LH_2 fuel stored in main tanks.	ACS jets use LOX/LH_2 propellants. Cryogenic LOX/LH_2 heated, pressurized, and converted to gas by conditioning unit.

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later interpretation made use of knowledge of the relative criticality of equipment and also past reliability of similar components. The resulting selective redundancy is shown in equipment lists to be discussed later.

In each following section devoted to a subsystem, more specific requirements are listed. Additionally, groundrules and assumed requirements are discussed. Where applicable, each of the sections displays parametric and design-point data as another type of requirement.

2.0 ENVIRONMENTAL CONTROL AND LIFE SUPPORT SUBSYSTEM

2.1 REQUIREMENTS

Studies in the EC/LSS area encompassed all of the requirements for habitation that significantly affected tug design. Most of these items are described in Table 2-1. In addition, related criteria for CM volume, atmospheric purity, furniture, and crew functions were investigated. All of the requirements having general application were drawn from EOSS documentation. Since the tug requirements differed significantly from those of EOSS, the resulting equipment selections do not directly correspond.

Each item in the requirements list was drawn from a current source. Crew size and mission duration are determined from operations analyses and mission planning. The workspace volume data has been surveyed in Reference 2-1 and is summarized in Figure 2-1. Requirements 4 through 11 and 14 through 21 enjoy industry-wide agreement, but are tailored to be somewhat austere for tug usage. Requirements 12 and 13 indicate leakage rates below those of Apollo, and predict advanced technology.

It was assumed that all solid biological waste, and internally used expendables and consumables, are returned to the tug base; however, future studies may show appreciable payload increase by periodic space dumping during the mission. It was also assumed that, in minor cases, space maintenance could be conducted within the confines of the pressurized CM. This ground-rule leads to the storage of spare equipment aboard the tug.

Only two sets of EVA life support systems (suit, PLSS, and OPS) are stored for space missions, and four for landing missions. The practice of not supplying full crew EVA capability has precedence in nearly all previous space flights, current EOSS philosophy, and is similar to the case of parachutes for airline passengers.

A more detailed list of requirements is given in the EC/LSS study report, Appendix A. The appendix also includes a complete list of the contaminants allowable for the tug atmosphere and a breakdown of consumption rates.

Table 2-1. Environmental Control and Life Support Subsystem Requirements

Requirement Area	Space Missions	Lunar Landing Missions	Rescue Missions
1. Crew size, men	2-6	2-4	≤12
2. Mission duration, days	7	45	1
3. Workspace volume, ft^3/man (m^3/man)	70-130 (1.98-3.69)	190-430 (5.38-12.2)	50 (1.42)
4. Atmospheric temperature, °F (°K)	65-75 (291-296)	65-75 (291-296)	65-75 (291-296)
5. Atmospheric ventilation rate, ft/min (cm/sec)	15-100 (7.6-50.8)	15-100 (7.6-50.8)	15-100 (7.6-50.8)
6. Oxygen atmospheric pressure, PSIA (n/cm^2)	3.5 (2.41)	3.5 (2.41)	3.5 (2.41)
7. Nitrogen atmospheric pressure — docked, PSIA (n/cm^2) undocked, PSIA (n/cm^2)	11.2 (7.70) 1.5 (1.0)	11.2 (7.70) 1.5 (1.0)	11.2 (7.70) 1.5 (1.0)
8. Carbon dioxide concentration, mm hg (n/m^2)	5-7.6 (700-1010)	5-7.6 (700-1010)	5-7.6 (700-1010)
9. Food supply rate, lb/man-day (kg/man-day)	1.68 (0.76)	1.68 (0.76)	0-1.68 (0-0.76)
10. Oxygen consumption rate, lb/man-day (kg/man-day)	1.84 (0.84)	1.84 (0.84)	1.84 (0.84)
11. EVA oxygen consumption rate, lb/man-hr (kg/man-hr)	0.4 (0.18)	0.4 (0.18)	0.4 (0.18)
12. Oxygen leakage rate, lb/day (kg/day)	0.362 (0.164)	0.362 (0.164)	0.362 (0.164)
13. Nitrogen leakage rate, lb/day (kg/day)	0.316 (0.143)	0.316 (0.143)	0.316 (0.143)
14. Water metabolic consumption rate, lb/man-day (kg/man-day)	6.13 (2.78)	6.13 (2.78)	6.13 (2.78)
15. Wash water consumption rate, lb/man-day (kg/man-day)	4.0 (1.8)	4.0 (1.8)	0-4.0 (0-1.8)
16. EVA water consumption rate, lb/man-hr (lb/man-hr)	1.4 (0.63)	1.4 (0.63)	1.4 (0.63)
17. Water leakage rate, lb/day (kg/day)	0.3 (0.14)	0.3 (0.14)	0.3 (0.14)
18. Metabolic heat production, BTU/man-hr (w/man)	430-496 (126-145)	430-496 (126-145)	430-496 (126-145)
19. Carbon dioxide production, lb/man-day (kg/man-day)	1.98-3.0 (0.90-1.4)	1.98-3.0 (0.90-1.4)	1.98-3.0 (0.90-1.4)
20. Urine production, lb/man-day (kg/man-day)	2.67-4.61 (1.21-2.09)	2.67-4.61 (1.21-2.09)	2.67-4.61 (1.21-2.09)
21. Feces production, lb/man-day (kg/man-day)	0.38 (0.17)	0.38 (0.17)	0.38 (0.17)

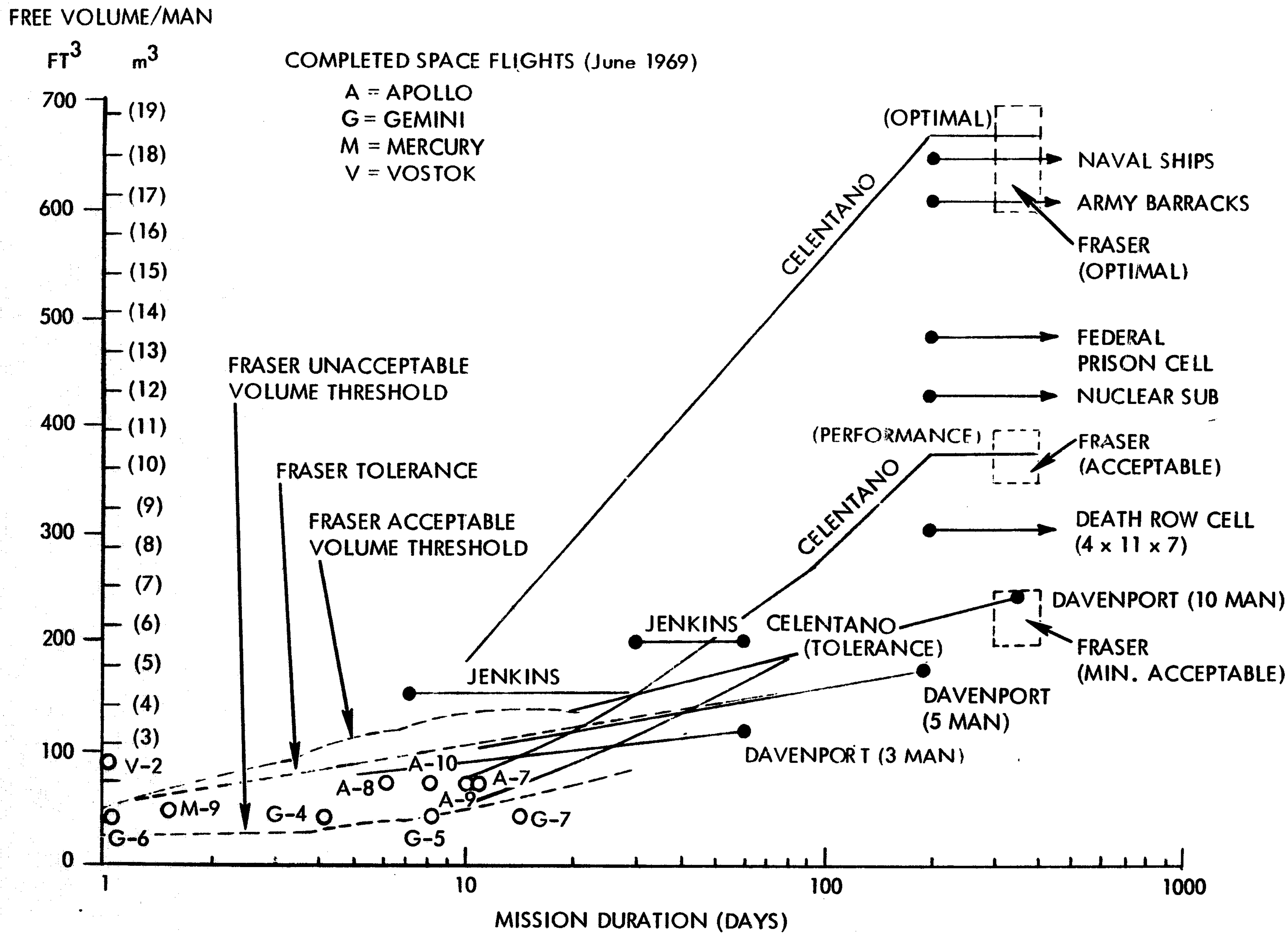

Figure 2-1. Living Space Requirements for Long-Duration Missions

2.2 CANDIDATE EQUIPMENT

The principal consideration in selecting subsystem components is commonality with other IPP elements, especially the EOS and EOSS. Both EOS and EOSS requirements are somewhat in opposition to those of tug, however. The EOS is a ground-based vehicle of short mission duration which must operate in space and atmosphere. The EOSS is a non-accelerating vehicle of exceedingly long mission duration and has a larger crew. Moreover, from a crew standpoint, the EOS is somewhat akin to an airline in its functions, EOSS caters to scientific endeavors, and the tug may be likened to a safari vehicle. All of the EC/LSS configuration drivers for tug, however, are more nearly like Apollo. Thus, an economical tug EC/LSS would intuitively use Apollo components to a large extent, although a minimum weight tug EC/LSS might result from scaled-down EOS and EOSS equipment to take advantage of advanced technology. There is, therefore, a strong incentive to draw from Apollo equipment at the cost of IPP element commonality.

The scope of the EC/LSS study included the areas of interest shown in Figure 2-2. All of the lower five areas in the figure involved principal trade studies.

Widespread use of regenerative (closed) systems for the IPP elements proved to be another factor influencing commonality. The results of the trade studies indicate that nonregenerative (open) systems, of the type used in Apollo and Skylab, are lighter and require less electrical power. Figure 2-3 summarizes a comparatively extensive analysis of principal trades and illustrates the weight trend. If the weight penalty of electrical power is included, the crossover points move further to the right, indicating even longer missions are necessary to justify closed systems.

2.3 RECOMMENDATIONS

After considering the relative merits of equipment commonality, advanced technology, minimum weight and power, and all of the detailed requirements, the equipment was selected. These selections are shown in Table 2-2. A detailed discussion of the equipment is given in Appendix A.

The schematic diagram of the atmospheric purification system using the catalytic oxidation/sorption process appears in Figure 2-4. This system uses electrical power to provide heat for catalytic action. Future studies may find that heat from the thermal loop or a radioisotope source may be less costly. Two other schematic diagrams involve EC/LSS systems: the thermal loop and the oxygen supply system, both of which are discussed in later sections.

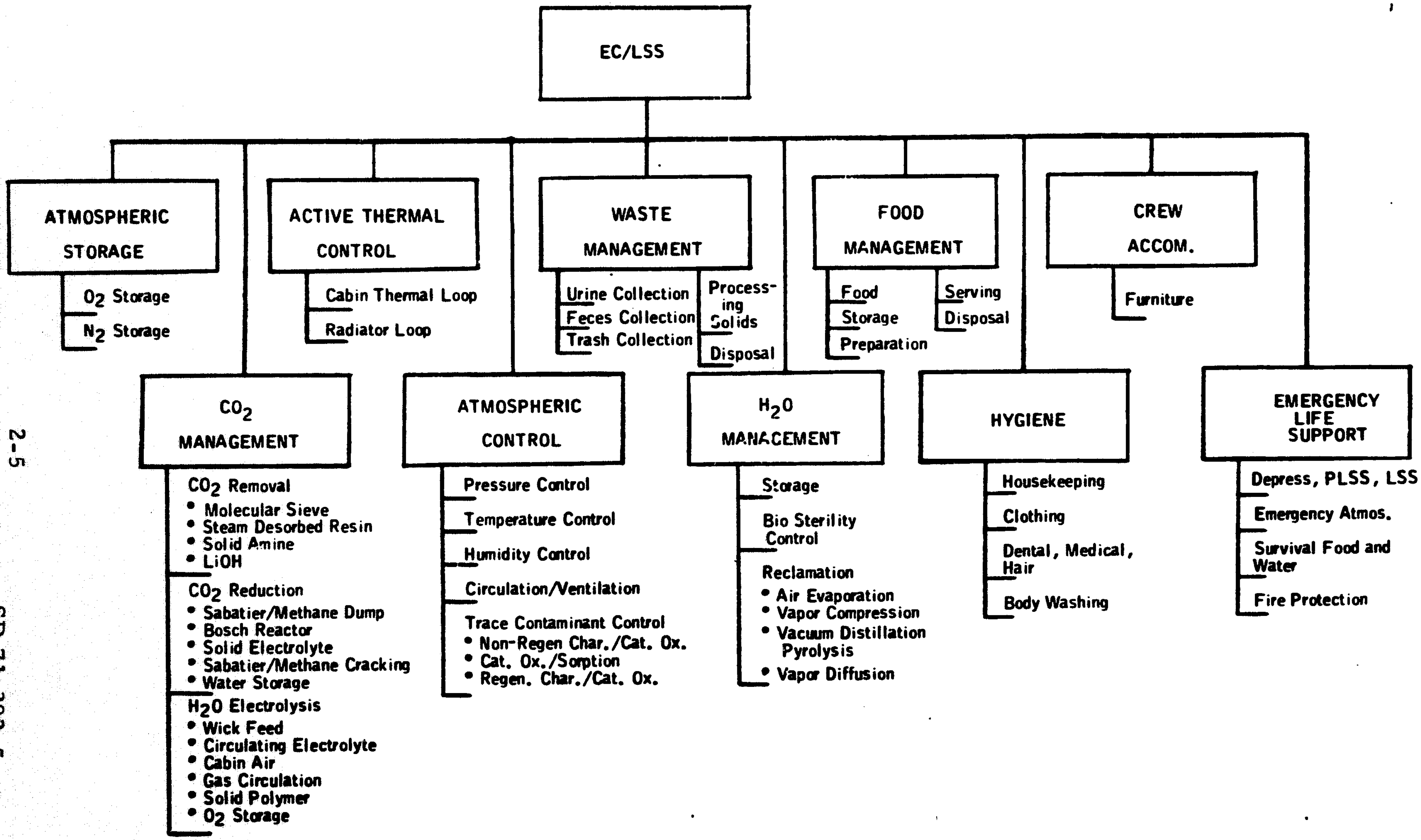

Figure 2-2. ECLSS Candidate Trade Study Tree

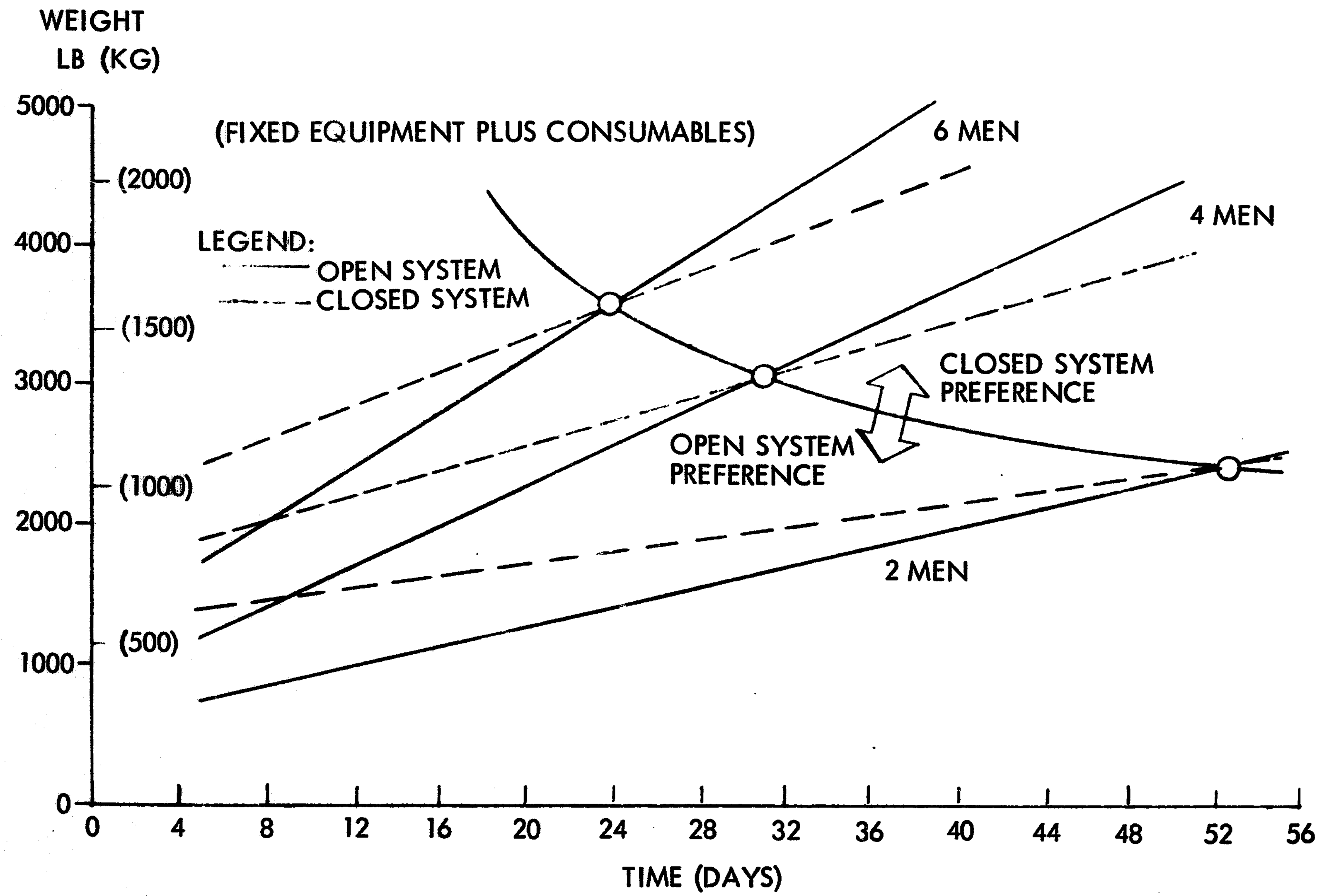

Figure 2-3. ECLSS Weight Tradeoff - Open Versus Closed Systems

Table 2-2. ECLSS Equipment Selections

Function	Method and Source	Fixed Equipment	Spare Equipment	Consumables and Expendables
Housekeeping	(EOSS extrap)	None	None	Cleaners, trash bags, charcoal filters
Furniture	(EOSS extrap)	Bunks, control seats, cabinets, Dri John, exper station, food prep station	None	None
Food management	Space: Dehydrated and freeze-dried, Lander: Dehydrated and frozen (Skylab)	Drink gun, trays, chiller oven, freezer, controls brackets, plumbing	Drink nozzle, controls, support	Dehydrated, freeze-dried, or frozen food, utensils, soap
Water management	Potable accum (Skylab)	Potable tank, chiller, heater, PLSS recharge, purif ampules, fire ext, controls, plumbing	Purif ampules, quick disc hoses	Water from fuel cells
Waste management	Solid storage, liquid dump (Skylab)	Tank, fecal and urine collector, processor, dump nozzle, controls, brackets, plumbing	Storage bags, quick disc hoses	Chemical and bacterial filters
Temperature and humidity control	Fans, heat exchanger and condenser (Apollo/Skylab)	Central, personal hyg and humidity heat exchangers; fans, controls, plumbing	Fans, filters, and controls	Filters
Atmospheric purification	Catalytic Ox/Sorption (Skylab, EOSS Extrap)	Filters, regen unit, char beds, controls, brackets, plumbing	Filters, controls	Particle and molecular filters catalytic burner charcoal
Atmosphere pressure control	(Skylab)	Partial and total pressure regulators and controls, dump valves	Controls and valves	None
Atmosphere circulation and CO_2 removal	Fans, lithium hydroxide (Apollo/Skylab)	Ducts, outlets, fans, controls, brackets, valves	Fans, controls, filters	Filters, lithium hydroxide canisters
Atmosphere thermal loop	Water-glycol (Apollo/Skylab)	Pumps, HX, accums, controls, waterfill, intercool, brackets, plumbing	Pumps, valves, controls	None
EVA life support	PLSS and ops (Apollo/Skylab extrap)	OX accum, PLSS, ops, suits, battery charger, hoses, disc, cont, plumbing	Quick dis-connect hoses, controls	Oxygen from IM
Emergency life support	High pressure OX (EOS, Apollo/Skylab)	IVA station, controls, regs, disc, hoses, plumb, tank, brackets	Quick disconnect hoses, controls	Oxygen from IM
Nitrogen storage	Crew module tank (Skylab)	Tank, regulator, controls, plumbing	None	Nitrogen
Crew support	(EOSS extrap)	None	None	Clothing, bedding, towels, soap, medikits
Interior lighting	(Apollo)	Fixtures, bulbs	None	None

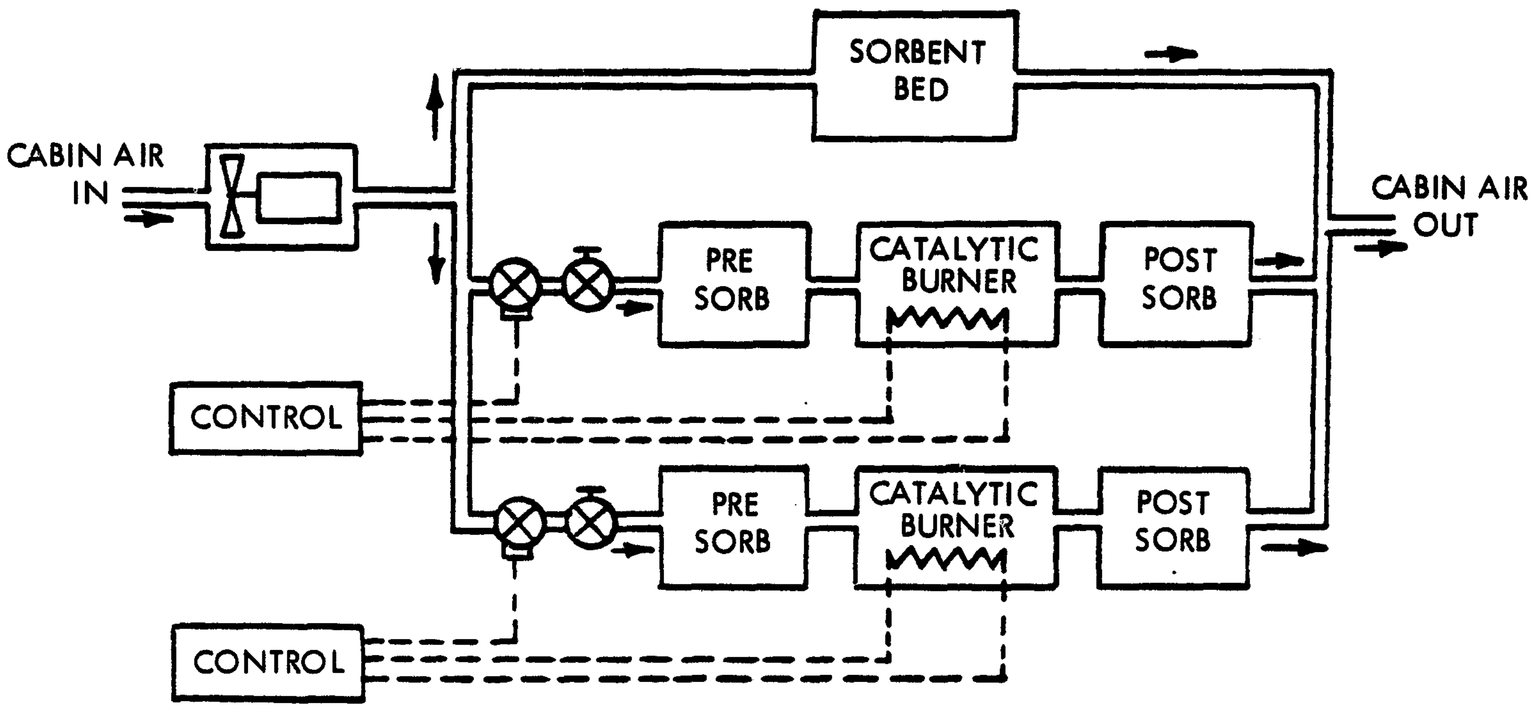

Figure 2-4. Catalytic Oxidation/Sorption Atmospheric Purification System

Sufficient data was generated for the subsystem to permit parametric presentation. Table 2-3 shows the parametric data specific to tug requirements. Discrimination is made between fixed equipment, spare equipment for maintenance tasks during the mission, and expendables and consumables. It also reflects the difference between space missions and lunar landing missions. The numerical entries in the table may be used as coefficients in the general expression

$$A = a + bM + cD + dMD \tag{2-1}$$

where

A = total weight, volume or power

M = number of crew members

D = mission span in days

a, b, c, d = coefficient entries from the appropriate columns of the table.

Table 2-3. Environmental Control and Life Support Subsystem Parametric Data

Function	Mission	Fixed Equipment Weight lb (kg)	Fixed Equipment Weight lb/M (kg/M)	Fixed Equipment Weight lb/D (kg/D)	Fixed Equipment Volume ft^3 (m^3)	Fixed Equipment Volume ft^3/M (m^3/M)	Fixed Equipment Volume ft^3/D (m^3/D)	Fixed Equipment Power w	Fixed Equipment Power w/M	Spare Equipment Weight lb (kg)	Spare Equipment Weight lb/M (kg/M)	Spare Equipment Volume ft^3 (m^3)	Spare Equipment Volume ft^3/M (m^3/M)	Expendables and Consumables Weight lb (kg)	Expendables and Consumables Weight lb/M (kg/M)	Expendables and Consumables Weight lb/D (kg/D)	Expendables and Consumables Weight lb/M-D (kg/M-D)	Expendables and Consumables Volume ft^3 (m^3)	Expendables and Consumables Volume ft^3/M (m^3/M)	Expendables and Consumables Volume ft^3/M-D (m^3/M-D)
Furniture	Space	277.0 (125.6)	–	–	16.32 (0.46)	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–
	Lunar landing	465.0 (210.9)	–	–	23.78 (0.67)	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–
Food management	Space	9.0 (4.1)	3.0 (1.36)	–	0.90 (0.03)	0.30 (0.01)	–	–	–	5.7 (2.6)	0.90 (0.41)	0.57 (0.02)	0.090 (0.003)	–	–	–	4.390 (1.991)	–	–	0.1990 (0.0056)
	Lunar landing	84.0 (38.1)	3.00 (1.36)	–	8.40 (0.24)	0.30 (0.01)	–	80.0	–	5.7 (2.6)	0.90 (0.41)	0.57 (0.02)	0.090 (0.003)	–	–	–	3.830 (1.737)	–	–	0.1740 (0.0049)
Water management	All	10.0 (4.5)	–	–	1.82 (0.05)	–	–	50.0	–	–	–	–	–	–	–	–	9.000* (4.082)	–§	–	–
Waste management	All	–	12.10 (5.49)	–	–	0.64 (0.02)	–	–	–	–	3.63 (1.65)	–	0.398 (0.011)	2.5 (1.1)	6.14 (2.79)	–	0.016 (0.007)	0.275 (0.008)	0.674 (0.019)	0.0018 (0.0001)
Temperature and humidity control	All	–	37.20 (16.87)	–	–	5.50 (0.16)	–	–	41.6	–	11.16 (5.06)	–	0.269 (0.008)	0.5 (0.2)	2.70 (1.22)	–	–	0.012 (0.000)	0.065 (0.002)	–
Atmospheric purification	All	34.0 (15.4)	6.67 (3.03)	–	1.91 (0.05)	0.38 (0.01)	–	15.0	22.2	4.0 (1.8)	–	0.12 (0.00)	0	5.0 (2.3)	–	–	0.040 (0.018)	0.150 (0.004)	–	0.0012 (0.000)
Atmospheric pressure control	All	31.0 (14.1)	–	–	1.57 (0.04)	–	–	30.0	–	9.3 (4.2)	–	0.53 (0.02)	–	–	–	–	–	–	–	–
Atmospheric circulation and CO_2 removal	Space	–	15.00 (6.80)	–	–	2.24 (0.06)	–	–	25.0	–	4.95 (2.25)	–	0.875 (0.025)	–	–	–	4.500 (2.041)	–	–	0.1550 (0.0044)
	Lunar landing	–	15.00 (6.80)	–	–	2.24 (0.06)	–	–	25.0	–	4.95 (2.25)	–	0.875 (0.025)	–	–	–	3.680 (1.669)	–	–	0.1265 (0.0036)
Atmospheric thermal loop	All	–	7.70 (3.49)	–	–	0.83 (0.02)	–	150.0	25.0	–	2.31 (1.05)	–	0.238 (0.007)	–	–	–	–	–	–	–

Table 2-3. Environmental Control and Life Support Subsystem Parametric Data (Cont)

Function	Mission	Fixed Equipment								Spare Equipment				Expendables and Consumables						
		Weight			Volume			Power		Weight		Volume		Weight				Volume		
		lb (kg)	lb/M (kg/M)	lb/D (kg/D)	ft^3 (m^3)	ft^3/M (m^3/M)	ft^3/D (m^3/D)	w	w/M	lb (kg)	lb/M (kg/M)	ft^3 (m^3)	ft^3/M (m^3/M)	lb (kg)	lb/M (kg/M)	lb/D (kg/D)	lb/M-D (kg/M-D)	ft^3 (m^3)	ft^3/M (m^3/M)	ft^3/M-D (m^3/M-D)
EVA life support	Space	360.0 (163.3)	—	—	14.40 (0.41)	—	—	—	—	29.4 (13.3)	—	1.17 (0.03)	—	—¶	—	—	—	—¶	—	—
	Lunar landing	720.0 (326.6)	—	—	28.80 (0.82)	—	—	—	—	59.0 (26.8)	—	2.36 (0.07)	—	—¶	—	—	—	—¶	—	—
Emergency life support	All	7.4 (3.4)	10.30 (4.67)	—	6.64 (0.10)	0.23 (0.01)	—	—	—	—	3.09 (1.40)	—	0.069 (0.002)	99.0† (44.9)	—	0.362 (0.164)	1.840 (0.835)	—§	—	—
Nitrogen storage	Space	6.0 (2.7)	—	0.15 (0.07)	0.13 (0.00)	—	0.0030 (0.0001)	—	—	—	—	—	—	14.8 (6.7)	—	0.380 (0.172)	—	—§	—	—
	Lunar landing	5.0 (2.3)	—	0.13 (0.06)	0.10 (0.00)	—	0.0025 (0.0001)	—	—	—	—	—	—	12.4 (5.6)	—	0.316 (0.143)	—	—§	—	—
Crew support	All	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	0.915 (0.415)	—	—	0.0416 (0.0012)
Interior lighting	All	—	0.67 (0.30)	—	—	0.30 (0.01)	—	—	33.3	—	—	—	—	—	—	—	—	—	—	—
Total space mission EC/LSS		734.4 (333.1)	92.64 (42.02)	0.15 (0.07)	43.69 (1.24)	10.42 (0.30)	0.0030 (0.0001)	245.0	147.1	48.4 (22.0)	26.04 (11.81)	2.39 (0.07)	1.939 (0.055)	22.8 (10.3)	8.84 (4.01)	0.380 (0.172)	9.361 (4.473)	0.437 (0.012)	0.739 (0.021)	0.3986 (0.0113)
Total lunar landing mission EC/LSS		1,356.4 (615.2)	92.64 (42.02)	0.13 (0.06)	73.02 (2.07)	10.42 (0.30)	0.0025 (0.0001)	325.0	147.1	78.0 (35.4)	26.04 (11.81)	3.58 (0.10)	1.939 (0.055)	20.4 (9.3)	8.84 (4.01)	0.316 (0.143)	8.481 (3.847)	0.437 (0.012)	0.739 (0.021)	0.3451 (0.0098)

Notes:

*Water requirements for reference only (supplied by IM fuel cells), not included in totals.

§Water, oxygen, and nitrogen volumes accountable to tank fixed equipment.

¶Oxygen requirements included under emergency life support.

†Oxygen requirements for reference only (supplied by PM tanks), not included in totals.

In the case of space missions, assumed to span seven days or less, contingency expendables and consumables are included in the coefficients when D is the nominal mission duration. The data for the longer lunar landing missions does not include contingency expendables and consumables, hence the variable D must be the sum of nominal and contingency periods to arrive at the correct value. Including fixed, spare, expendable and consumable weights, the totals for space and lunar landing are respectively

$$806.6 + 127.5(M) + 0.530(D) + 9.861(M\text{-}D) \text{ Lbs} \tag{2-2}$$

and

$$1454.8 + 127.5(M) + 0.346(D) + 8.481(M\text{-}D) \text{ Lbs} \tag{2-3}$$

To these weights must be added the oxygen and water requirements as noted in Table 2-3.

The volumetric adequacy of the CM was assessed by testing the parametric volume data of Table 2-3 against the living space criteria in Figure 2-1. Fixed equipment, spare equipment, and expendables and consumables for all subsystems represented in the CM were subtracted from the inner volume (less the airlock) of the module. All of the EC/LSS volume was variable with crew size and mission duration. The results of the analysis are shown in Figure 2-5, where rectangular areas are bounded horizontally by crew size and vertically by approximate mission duration. A 15-foot (4.6-meter) diameter crew module appears to be more than adequate for space missions with a six-man crew, and as marginal as Apollo under emergency 12-man crew occupation. For a lunar landing mission, a four-man crew has adequate workspace for the longest anticipated mission. Both the 22-foot (6.7-meter) diameter and the 12-foot (3.7-meter) diameter, two level, CM's are more than adequate under normal staffing.

To determine the characteristics of the tug from the parametric data, the operating point of all fixed equipment for the most constraining case was found—either a six-man crew for a seven-day mission or a four-man crew for a 45-day mission. Once the standard fixed equipment values were established, they were used for any size crew and any mission. In a few areas it was found that the accommodation for lunar landing imposed too much penalty on space missions. In these cases it is recommended that EC/LSS equipment be added for the landing missions. This equipment consists of an oven, a freezer, two additional EVA life support units (in the case of a four-man crew), and a general rearrangement of furniture to provide more storage space and an experiment control station.

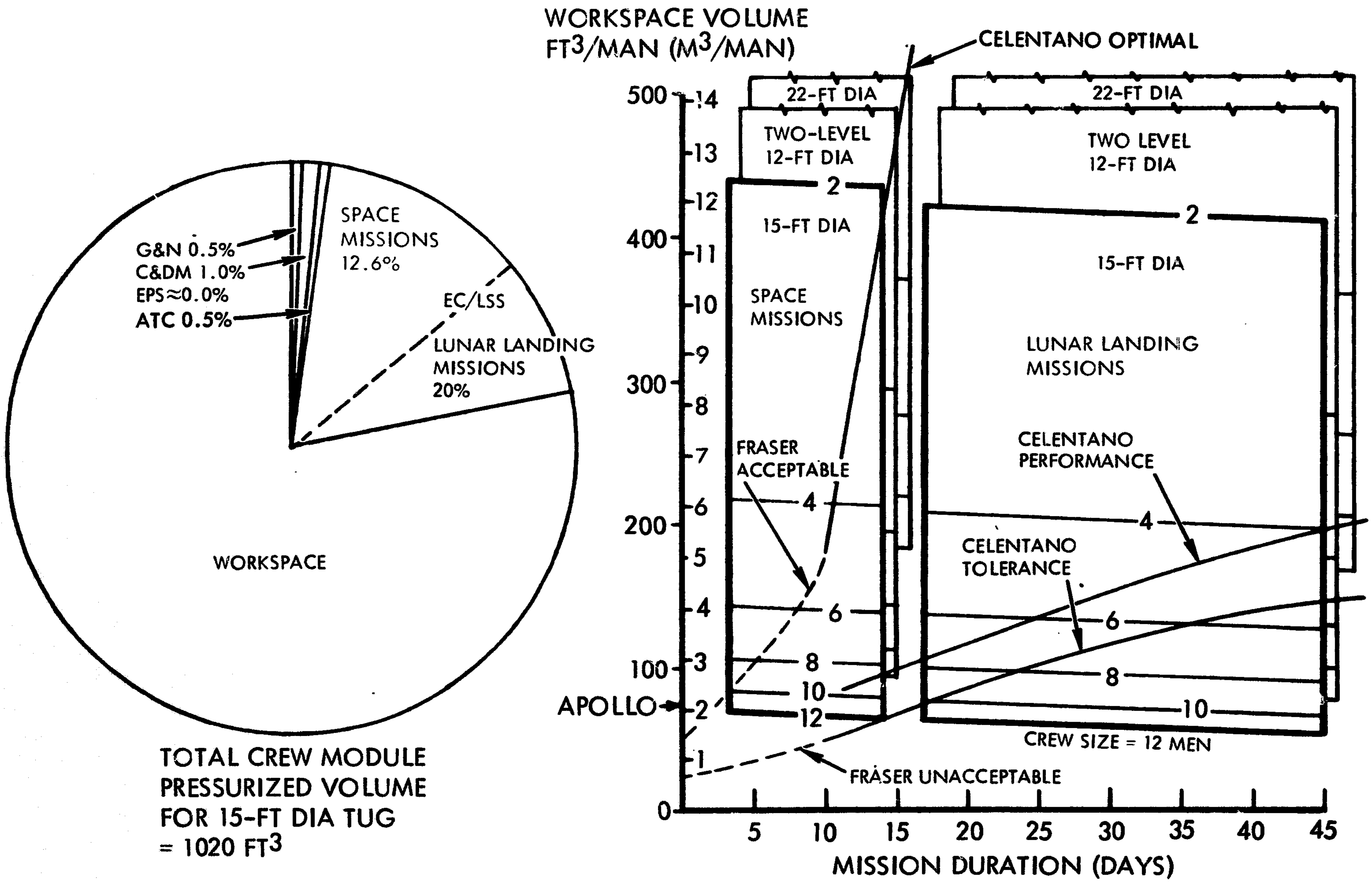

Figure 2-5. Crew Module Volume Allocations

Once the fixed equipment is sized, the quantities of spare equipment, expendables, and consumables are tailored to the specific mission. These data are shown in Tables 2-4 through 2-7, which represent the recommended equipment for EC/LSS.

An evaluation of the EC/LSS was made to show the delta penalty to the baseline requirements if a 12-man, 24-hour rescue mission were adopted. The candidate subsystems affected were: CO_2 management, atmosphere and water storage, radiator capability to remove the increased metabolic load, and the capability of the thermal loop to maintain a tolerable cabin environment. Table 2-8 summarizes the evaluation.

Results indicate that 28 pounds (12.17 kilograms) of LiOH is required (assuming a CO_2 partial pressure level below a maximum design value of 7.6 mm HG). Spare weight of LiOH for the six-man, seven-day mission exceeded the required amount and therefore no delta penalty was assessed. The gas flow rate to be processed for CO_2 removal had to be increased, requiring the use of the secondary circulation loop. Operation of this loop required a power increase of 150 watts. Oxygen metabolic requirement doubled. Delta weight of O_2 required is 22.0 pounds (10 kilograms). There is no water penalty for this mission since the fuel cell generation rate exceeds drinking requirements of the crew. The radiator system has built-in redundancy capable of removing the increased metabolic load of 12 men. The active thermal loop pumping rate must be increased to reject the additional heat load from the cabin. The power penalty to operate additional pumps is 50 watts. The total delta penalty to the EC/LSS results in 22 pounds (10 kilograms) weight and 300 watts of electrical power.

2.4 CONCLUSIONS

An EC/LSS for tug may be composed almost entirely of Apollo and Skylab-type open system equipment. The limited use of EOSS-type equipment is recommended; however, the regenerative systems of EOS and EOSS are too heavy and require too much electrical power to benefit tug.

Table 2-4. Environmental Control and Life Support Subsystem Fixed Equipment

Description	Source	Weight		Volume		Power (w)		
		lb	(kg)	ft^3	(m^3)	Two Men	Four Men	Six Men
Crew Module Space Mission Fixed Equipment								
Furniture: bunks, control seats, food prep, cab, Dri John	EOSS extrap	277.0	(126.0)	16.32	(0.46)	0	0	0
Food mgt: drink gun, trays, chiller, controls, brackets, plumbing	Skylab	27.0	(12.0)	2.70	(0.08)	0	0	0
Water mgt: potable tank, chiller, heater, purif ampules, fire ext, cont, plumbing	Skylab	10.0	(5.0)	1.82	(0.05)	50	50	50
Waste mgt: tank, collector, processor, dump nozzles, cont, brack, plumbing	Skylab	73.0	(33.0)	3.84	(0.10)	0	0	0
Temperature and himidity cont: fans, central hx, pers hyg hx, humid. hx, cont, plumbing	Apollo/Skylab	225.0	(102.0)	33.40	(0.95)	250	250	250
Atmos purif: filters, regen unit, char beds, controls, brackets, plumbing	EOSS extrap	74.0	(34.0)	4.15	(0.12)	59	104	148
Atmos press cont: partial and total press regs and controls, dump valves	Skylab	31.0	(14.0)	1.57	(0.04)	30	30	30
Atmos circ: ducts, outlets, fans, controls, brackets, valves	Apollo/Skylab	91.0	(41.0)	13.58	(0.38)	50	100	150
Atmos therm loop: pumps, hx, accums, cont, waterfill, brackets, plumbing	Apollo/Skylab	46.0	(21.0)	4.95	(0.14)	200	250	300
EVA life support: Ox accum, PLSS, ops, suits, batt charger, hoses, disc, cont, plumbing	Apollo/Skylab	360.0	(163.0)	14.40	(0.41)	0	0	0
Emerg life support: IVA station, controls, regs, disc, hoses, plumbing, tank	Apollo/EOS	69.4	(31.5)	7.98	(0.23)	0	0	0
Nitrogen store: tank, regulator, controls, plumbing	Skylab	6.2	(2.8)	0.31	(0.01)	0	0	0
Interior lighting: fixtures, bulbs (1200 watt capability)	Estimate	4.0	(2.0)	1.60	(0.05)	200	200	200
Total CM EC/LSS space mission fixed equipment	--	1,293.6	(587.3)	106.62	(3.02)	839	984	1,128
Crew Module Lunar Landing Additional Fixed Equipment								
Furniture: lockers, experiment station	EOSS extrap	188.0	(85.0)	7.46	(0.21)	0	0	-
Food mgt: oven, freezer	Skylab extrap	75.0	(34.0)	9.00	(0.25)	80	80	-
EVA life support: PLSS, ops, suits	Apollo/Skylab	360.0	(163.0)	14.40	(0.41)	0	0	0
Total CM EC/LSS lunar landing mission addl fixed equipment	--	623.0	(282.0)	30.86	(0.87)	80	80	0

Table 2-5. Environmental Control and Life Support Subsystem Spare Equipment

Description	Source	Two Men				Four Men				Six Men			
		Weight		Volume		Weight		Volume		Weight		Volume	
		lb	(kg)	ft^3	(m^3)	lb	(kg)	ft^3	(m^3)	lb	(kg)	ft^3	(m^3)
Crew Module Space Mission Spare Equipment													
Food management: drink nozzle, controls, support	Skylab	7.5	3.4	0.75	0.0212	9.3	4.2	0.93	0.0263	11.1	5.03	1.11	0.0314
Water management: storage bags, quick disconnect hoses	Skylab	7.3	3.3	0.80	0.0226	14.5	6.6	1.59	0.0450	21.8	9.89	2.39	0.0677
Temperature and humidity controls: fans, filters, controls	Apollo/Skylab	22.3	10.11	0.50	0.0142	44.6	20.2	1.08	0.0306	67.0	30.39	1.61	0.0456
Atmosphere purifier: filters, controls	EOSS extrap	4.0	1.8	0.12	0.0034	4.0	1.8	0.12	0.0034	4.0	1.8	0.12	0.0034
Atmosphere press. cont: controls, valves	Skylab	9.3	4.2	0.53	0.0150	9.3	4.2	0.53	0.0150	9.3	4.2	0.53	0.0150
Atmosphere circ: fans, controls, filters	Apollo/Skylab	9.9	4.5	1.75	0.0496	19.8	8.98	3.50	0.0991	29.7	13.47	5.25	0.1487
Atmosphere therm loop: pumps, valves, controls	Apollo/Skylab	4.6	2.08	0.48	0.0136	9.2	4.17	0.95	0.0269	13.9	6.30	1.43	0.0405
EVA life support: quick disconnect hoses, controls	Apollo/Skylab	29.4	13.3	1.17	0.0331	29.4	13.34	1.17	0.0331	29.4	13.33	1.17	0.0331
Emergency life support: quick disconnect hoses, controls	Apollo/EOS	6.2	2.8	0.14	0.0040	12.4	5.62	0.28	0.0079	18.5	8.39	0.41	0.0116
Total CM EC/LSS space mission spare equipment	--	100.5	45.49	6.28	0.1767	152.5	69.11	10.15	0.2874	204.7	92.80	14.02	0.3970
Crew Module Lunar Landing Additional Spare Equipment													
EVA life support: quick disconnect hoses, controls	Apollo/Skylab	30.0		1.20		30.0		1.20		30.0		1.20	

Table 2-6. Environmental Control and Life Support Expendables and Consumables

Description	Source	Space Missions												Lunar Landing Missions							
		2-Man/7-Day				4-Man/7-Day				6-Man/7-Day				4-Man/31-Day				4-Man/45-Day			
		Weight		Volume		Weight		Volume		Weight		Volume		Weight		Volume		Weight		Volume	
		lb	(kg)	ft^3	(m^3)	lb	(kg)	ft^3	(m^3)	lb	(kg)	ft^3	(m^3)	lb	(kg)	ft^3	(m^3)	lb	(kg)	ft^3	(m^3)
Crew Module Expend. and Consumables																					
Housekeeping: cleaners, trash bags, char filters	EOSS extrap	6.6	3.0	0.30	0.0085	13.1	5.94	0.60	0.017	19.7	8.93	0.90	0.0255	55.0	24.95	2.50	0.0708	82.0	37.2	3.73	0.1056
Fqod mgt: dehyd/froz food, utensils, soap	Skylab	61.5	27.9	2.79	0.0790	123.0	55.79	5.57	0.1577	184.0	83.46	8.38	0.2373	450.0	204.11	20.40	0.5777	664.0	301.2	30.20	0.8552
Waste mgt: chemical and bacterial filters	Skylab	15.0	6.8	1.65	0.0467	27.5	12.47	3.02	0.0855	40.0	18.14	4.40	0.1246	29.0	13.15	3.20	0.0906	30.0	13.6	3.30	0.0934
Temp and humid. cont: filters	Apollo/ Skylab	5.9	2.68	0.14	0.0040	11.3	5.12	0.27	0.0076	16.7	7.57	0.40	0.0113	11.3	5.12	0.27	0.0076	11.3	5.1	0.27	0.0076
Atmos purif: partial and mobile filters, char	EOSS extrap	5.6	2.54	0.17	0.0048	6.1	2.77	0.18	0.0051	6.7	3.04	0.20	0.0057	10.0	4.53	0.30	0.0085	12.2	5.5	0.37	0.0105
Atmos circ: LiOh	Apollo/ Skylab	63.0	28.58	2.17	0.0614	126.0	57.1	4.35	0.1232	189.0	85.73	6.50	0.1841	456.0	206.84	15.70	0.4446	662.0	300.3	22.80	0.6456
Crew support: clothes, bedding, towels, soap, medikits	EOSS extrap	12.8	5.81	0.58	0.0164	25.5	11.57	1.14	0.0323	38.3	17.37	1.74	0.0493	108.0	48.99	4.90	0.1388	149.0	72.1	7.20	0.2039
Nitrogen (volume included in tank)	Est.	3.4	1.54	--		3.4	1.54	--		3.4	1.54	--		11.0	4.99	--		15.4	7.0		
Total CM EC/LSS expend. and con.	--	173.8	72.85	7.80	0.2208	335.9	152.3	15.13	0.4284	497.8	225.78	22.52	0.6378	1,130.3	512.68	47.27	1.3385	1,635.9	742.0	67.87	1.9218

Table 2-7. Environmental Control and Life Support Subsystem Oxygen Weight Summary

Function	Mission			
	6-Man/7-Day		4-Man/45-Day	
	lb	(kg)	lb	(kg)
Initial pressurization	21.6	9.8	21.6	9.8
Normal metabolic (1.84 M-D)	77.4	35.1	331.0	150.0
Leakage (0.362 D)	2.5	1.1	16.3	7.4
Emergency, IVA and EVA	77.0	34.9	77.0	34.9
Total crew oxygen	178.5	80.9	445.9	202.1

Table 2-8. 12-Man, 24-Hour Rescue Mission EC/LSS Delta Penalty to Baseline Requirements

	Weight lb (kg)	Power Watts
CO_2 Management		
Requires primary and secondary loop		
No delta LiOH required. Use spare expendables for 6-man, 7-day mission. CO_2 partial press below 7.6 mmHg	None	---
Power required by secondary fans	---	150
Metabolic requirements for 6 men at 136 watts/man		
Oxygen—12 men at 1.84 (0.83 kg) per man day	22.0 (10.0)	
Water—from fuel cells	None	
Radiator redundancy capable of removing increase metabolic load	None	---
Active thermal loop power requirement increase	---	150
Total delta penalty	22.0 (10.0)	300

3.0 GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

3.1. SYSTEM GUIDELINES

Guidelines in the request for proposal for this study that particularly affect the GN&C system concept are listed below, paraphrased in some cases:

1. The tug shall be capable of manned or automated flight. Remote control from the earth, earth orbiting, and lunar orbiting stations or bases shall be considered.

2. The space tug design shall minimize the necessity for ground support during flight. Autonomy is the design objective.

3. No single failure or credible combination of failures will result in loss of life. It shall be a design objective to maximize crew safety and probability of mission success.

4. Tugs will conduct lunar surface missions from, and returning to, a lunar orbiting space station (LOSS) in a 60 n. mi/111.1 km near polar orbit. Some missions may be conducted from other orbits with a reusable nuclear shuttle (RNS) and without a LOSS.

5. The tug normally will be flyable by one crewman, but capable of automated operation for lunar surface missions.

6. Tug operations shall not be constrained by lunar lighting conditions.

7. The tug crew compartment will contain a flight station for control of all flight maneuvers.

8. All communication systems will be compatible with the MSFN, DSN, available Comsat systems, and all hardware elements of the integrated program plan (IPP).

9. The tug attitude reference shall be capable of large maneuvers about any axis without loss of attitude reference.

Guideline (3.) is interpreted to mean at least fail operational/fail safe (FO/FS) tolerance. In some instances degraded modes of operation after failure have been considered.

An additional guideline adopted in selecting system components was that, where feasible, they should be identical or similar to components of the space shuttle or earth orbiting space station GN&C systems.

3.2. GN&C REQUIREMENTS

The GN&C functional requirements are briefly described here. The choice of equipment to perform the various functions was influenced by the guidelines listed in Section 3.1, particularly items 1 and 2 (automatic and autonomous operation) and 3. Performance requirements for the various functions have not been formally defined since, in many cases, the required performance is really a tradeoff among system weight, complexity, and cost versus additional propellant for midcourse corrections and perhaps some operational restrictions.

3.2.1 Attitude Reference

The tug must have an attitude reference system that permits unlimited maneuvering without loss of attitude knowledge (guideline 9). The system must be capable of automatic initialization (alignment) in earth and lunar orbit, cislunar space, and on the lunar surface. The attitude reference must be continuously available, and the accuracy must be compatible with navigation sensor pointing accuracy requirements and thrust maneuver control requirements.

3.2.2 Navigation

The tug shall be capable of maintaining a state vector (position and velocity) estimate with accuracy compatible with mission trajectory requirements in earth and lunar orbit and in cislunar space. Autonomous navigation capability is desired wherever feasible (guideline 2).

3.2.3 Guidance

The tug GN&C system must provide guidance for all tug thrusting maneuvers. These maneuvers include:

1. Earth and lunar orbit change maneuvers, including rendezvous transfer orbit insertions and transfer orbit midcourse corrections.

2. LEO to geosynchronous Hohmann transfer orbit initiation (≈8000 fps/2438 mps), midcourse corrections, and geosynchronous orbit circularization (≈5000 fps/1524 mps).

3. Translunar injection (TLI, ≈10,000 fps), lunar orbit insertion and transearth injection (LOI and TEI, ≈3,000 fps/914 mps each), and earth orbit insertion (≈10,000 fps/3048 mps).

4. Retrobraking from lunar orbit (≈5000 fps/1524 mps) lunar landing terminal guidance, hovering translation maneuvers, and touchdown.

5. Ascent from the lunar surface (≈6,000 fps/1828 mps).

6. Transplanetary injection (11,000 - 15,000 fps/3353-4572 mps).

7. Rendezvous, docking port acquisition (target fly-around), terminal approach, and docking.

It is evident from the list that the term guidance encompasses two types of functions. The first is targeting or the selection and computation of an orbital velocity change (generally not unique) at a specified time that will produce desired terminal conditions at some later time (e.g., rendezvous transfer computation). The second function is the computation and issuance of thrust on, off, direction and throttling commands during the burn. There are significant qualitative differences in the guidance required for the several types of maneuvers. Most of the thrust maneuvers listed will be made with the main propulsion engines operating at full thrust, and the only guidance objective during the burn is to achieve the desired vector velocity change in a reasonably efficient manner. However, a few maneuvers require control of both the final velocity and position, by thrust direction control and either throttling (lunar landings) or reaction jet pulsing (docking approach). Another distinct type of thrust control problem that has received little attention is control of RNS thrust during the reactor cool-down cycle. The requirement for the tug intelligence module (IM) to perform as an instrument unit (IU) for the RNS could well impact the IM computer requirements.

The maneuvers demanding the highest accuracy are the transplanetary, translunar, and transearth insertion maneuvers. The LEO to geosynchronous transfer orbit insertion is sensitive to velocity magnitude errors (approximately ten n mi/18.5 kilometer) increase in apogee altitude per foot per second), but is relatively insensitive to small delta-V direction errors if the transfer is nearly a Hohmann transfer.

3.2.4 Lunar Landings

For pinpoint lunar landings to be achieved, sensors are required to locate and/or track the landing site, provide accurate altitude and ground velocity data, and detect surface obstacles that must be avoided for landing. Capability to redesignate the landing site during the terminal phases is considered necessary. The sensing techniques for landing site locating and obstacle detection are severely limited by the desire for all-lighting landing capability (guideline 6). Obstacle detection techniques are limited by the cloud of surface dust kicked up below 200 to 300 feet (61 to 91 meter) altitude and that obscures the landing site to visible or near-visible spectrum sensors. After the tug reaches a hover above the landing site, hovering translation maneuvers must be executed to avoid surface obstacles.

3.2.5 Rendezvous and Docking

Autonomous docking of unmanned tugs will require docking sensors capable of a few inches position accuracy and a few tenths of a foot per second velocity accuracy at docking range. Autonomous rendezvous will require sensors capable of acquiring the rendezvous target at sufficient range to allow maximum (three sigma) transfer orbit errors to be detected and corrected in order to achieve gross rendezvous. After gross rendezvous (≈1000 foot/305 meter separation), the tug may have to execute an automatic target fly-around maneuver to approach the target docking port.

3.2.6 Autopilot Functions

The spacecraft autopilot must accept pilot or computer guidance commands, plus attitude and rate data, process these through a stabilization filter, and issue reaction jet firing signals, main engine on-off, throttling, and engine gimbaling signals.

3.2.7 Controls and Displays

Manual controls and displays must allow the pilot to exercise effective spacecraft control during all flight phases and maneuvers and provide spacecraft trajectory information.

3.2.8 Requirements Summary

The GN&C requirements described are briefly summarized in Table 3-1.

Table 3-1. GN&C Requirements

Attitude Reference
All attitude capability
Realignment in space
Continuously available reference
Lunar surface realignment
Accuracy compatible with G&N requirements
Navigation
Earth orbit, lunar orbit, cislunar space
Accuracy compatible with mission requirements
Autonomous navigation desired
Guidance
Thrust maneuver targeting
Thrust direction commands
V's up to 15,000 fps/4572 mps
Lunar Landings
Landing site locating or tracking
Manual landing site redesignation (on board or remote)
Landing guidance
Surface obstacle detection
Hovering translation maneuvers
Rendezvous and Docking
Target acquisition and tracking
Transfer orbit error correction
Target fly-around maneuvers
Automatic docking
Autopilot Functions
Automatic and manual attitude control
Thrust maneuver control
Controls and Displays
Provide spacecraft trajectory information
Allow manual spacecraft control at any time
All Functions/Mission Phases
Automatic operation

3.3 SYSTEM SELECTION CONSIDERATIONS

This section discusses the technical factors and techniques or sensor types considered in system concept selection.

3.3.1 Navigation Sensors

Navigation sensors and techniques must be considered separately for earth orbit, cislunar space, and lunar orbit since each region of operation presents its own opportunities and limitations on sensing methods. The desire for autonomous and automatic operation is dominant in considering navigation techniques.

Earth Orbit Navigation

Earth-orbit-navigation techniques will be divided into two categories: (1) autonomous methods, in which only passive on-board optical sensors are used, and (2) semi-autonomous methods that do not depend on ground tracking of the spacecraft, but that utilize radio signals from ground sources or navigation satellites.

Autonomous methods that have been suggested for space navigation and that were investigated are:

1. Star occultation measurements
2. Automatic known or unknown landmark tracking
3. Horizon tracker measurements of local vertical
4. Manual known landmark tracking
5. Semiautonomous methods

The techniques will be discussed in the order listed.

Star Occultation Measurements. Measurement of the occultation times of known stars has several disadvantages: First, relatively little information is obtained from each occultation measurement; information obtained about the orbit plane orientation is quite weak. Second, the accuracy is limited since occultation is a gradual dim-out as the star, seen from the spacecraft, sinks into the atmosphere. Third, the requirement to acquire a new known star for each measurement is operationally undesirable. Comparison of star occultation measurements with horizon tracking indicates that a horizon tracker can provide more information more frequently and with probably better accuracy than star occulation measurements. Consequently, star occultation measurements are unattractive for earth orbit navigation.

Automatic Known Landmark Tracking. Automatic tracking of known and unknown landmarks has received considerable study since landmark tracking is potentially capable of providing navigation accuracies of a few tenths of a mile. Compared with horizon sensing, landmark tracking can provide much better navigation accuracy for basically two reasons:

1. The angular accuracy of landmark optical line of sight measurements is potentially much better than horizon tracker local vertical measurements since a horizon tracker must estimate the horizon from atmospheric luminence, which diminishes with altitude.

2. The position error corresponding to a given angular measurement error is much less for landmark tracking since the spacecraft landmark line-of-sight distance in low earth orbit is only a few hundred miles, whereas with horizon tracking the position error for each observation is the local vertical angular error multiplied by the spacecraft distance from the earth's center.

However, a major problem with earth landmark tracking is the varying cloud cover over landmarks. Reference 3-1 describes a mechanization for automatic known landmark tracking that uses an image correlation technique. Reportedly, tests have indicated the capability of successful image matching and tracking with a 90 percent cloud cover over the landmark area. Estimated navigation accuracy, from reference 3-2, with a landmark tracker and two star trackers is on the order of 0.1 to 0.2 nautical miles (0.18 to 0.36 km) after one orbit with one milliradian landmark tracker pointing accuracy. However, the reference assumes six landmarks per orbit at 60 degree intervals, which would place three of them on the night side of the earth. Limited tracking capability in moonlight is reported to be possible, but the number and distribution of landmarks assumed seems clearly optimistic.

The weight, volume, and power of an automatic known landmark tracker is estimated in reference 3-2 to be 30 pounds (13.6 kilogram), 0.67 feet^3 (.019 meter^3) and 40 watts power consumption. If the contention of successful tracking with 90 percent cloud cover can be substantiated, automatic known-landmark tracking should be reconsidered for the space tug. It is not recommended now because of the questionable feasibility of the technique.

<u>Automatic Unknown Landmark Tracking.</u> Automatic unknown-landmark tracking has been extensively investigated by Autonetics with its ULTRA (Unknown Landmark tracking) concept in a feasibility study for the USAF. Less information about the orbit is obtained from an unknown landmark than a known landmark, but the cloud cover problem is somewhat alleviated since cloud free areas may be selected. Autonetics studies indicate that three to five cloud-free landmarks can be tracked on each orbit on the sunlit side. Navigation accuracy of the system is classified, but is substantially better than that achievable with a horizon tracker. However, the ULTRA concept requires attitude references (star trackers) with tracking and relative orientation accuracies on the order of 0.001 degree. To provide the star trackers adequate fields of view in low earth orbits when the landmark tracker was looking earthward, and maintain alignment between the sensors to 0.001 degree, would require an extremely rigid temperature-controlled navigation base spanning nearly half of the circumference of the IM. Alternately, an inertial measurement unit (IMU) with zero-g drift rates on the order of 0.001 degree/hour or better would be required. In addition to these drawbacks, the ULTRA concept has not been proved in an actual space test, to our knowledge. It is concluded that the very high sensor accuracy requirements for unknown landmark tracking do not make this technique a reasonable choice for space tug navigation.

<u>Horizon Tracking</u>. The remaining technique available for automatic autonomous earth orbit navigation is horizon tracking. The navigation accuracy is an order of magnitude poorer than potentially achievable with known landmark tracking, but horizon tracking has one distinct advantage: it is a proved technique that works.

Several estimates of navigation accuracy with horizon tracking are available in the literature. Reference 3-2 estimates RMS (one sigma) errors after one orbit or more of tracking as 1.0 to 1.7 nautical miles/1.8 to 3.15 kilometers (6000 to 10,000 feet/1829 to 3048 meters) and 8 to 12 fps/2.4 to 3.7 mps. Reference 3-3 estimates 10,000 to 20,000 feet/3048 to 6096 meters RMS position errors. Reference 3-4 considers both random errors (0.6 milliradian RMS) and star tracker/horizon tracker alignment uncertainties (0.15 milliradian RMS), and error correlation between sequential measurements due to sensed horizon altitude variations resulting from large-scale weather patterns, etc. The reference estimates one-sigma position errors after one orbit of tracking of approximately 3800 feet/1158 meters down range, 800 feet/244 meters in altitude, and 600 feet/183 meters cross track. The corresponding velocity errors are approximately 1.5 fps/.46 mps along track, 0.7 fps/.2 mps radially, and 0.6 fps/1.8 mps cross track. The error parameters chosen for the analysis of Reference 3-4 are considered optimistic. The navigation errors were estimated for a 240-nautical mile/444-kilometer altitude. However, a sign significant fact shown by the data is that almost all of the position error in LEO is in the down-range direction.

The crucial question in evaluating horizon tracking or any other technique for space tug navigation is not the navigation accuracy PER SE, since no accuracy specifications have been set, but the penalties arising from the navigation errors. The principal penalties to consider are:

1. Increased delta-V requirements

2. Increased acquisition range requirements for rendezvous sensors

For these penalties to be evaluated, three types of missions will be considered: low earth orbit transfers, low earth orbit to geosynchronous orbit transfer, and transplanetary injection. The one-sigma errors of Reference 3-4 will be multiplied by 3 1/3 to give a total one-sigma position error of approximately 13,100 feet/3993 meters. The one-sigma errors were increased by a factor of 3 1/3 to account for a lower altitude orbit (100 nautical miles instead of 240 nautical miles), and a lower observation rate than was assumed in Reference 3-4. Thus, the three-sigma errors are ten times the one-sigma errors of Reference 3-4. The delta-V penalty and orbit transfer terminal position errors will then be grossly estimated for the three missions using three-sigma errors. The error parameters assumed are 0.6 milliradian RMS combined random instrument errors of the horizon tracker and attitude reference star trackers, one-kilometer (0.54 nautical miles) sensed horizon altitude variation, which give a total RMS error of 1.2 milliradians at 100 nautical miles/185 kilometers altitude and one measurement every three minutes (versus one observation per minute assumed in the reference). Thus, the assumed three-sigma navigation errors in LEO are:

δx = 38,000 feet/11582 meter (down range)

δy = 8,000 feet/2438 meter (altitude)

δz = 6,000 feet/1829 meter (cross range)

$\delta \dot{x}$ = 15.0 fps/4.6 mps

$\delta \dot{y}$ = 7.0 fps/2.13 mps

$\delta \dot{z}$ = 6.0 fps/1.8 mps

The magnitude of the delta-V penalties for low earth orbit transfers caused by these navigation errors can be roughly assessed by considering a 6000-feet/1829 meter cross-range position error at rendezvous. To remove this error within 30 degrees of orbital travel (7 1/2 minutes) would require a 14 fps/4.3 mps cross-range impulse after detection of the position error and an opposite impulse at rendezvous. Velocity penalties of this magnitude are acceptable. The maximum altitude errors at rendezvous in LEO due to the assumed in-plane velocity errors at transfer initiation are about 8.4 nautical miles/15.5 kilometers due to $\delta\dot{x}$ (180 degrees transfer) and 1.0 nautical miles/1.8 kilometer due to $\delta\dot{y}$ (90 or 270 degrees transfer). The sensor requirements to detect the rendezvous target with an 8.4 nautical miles/16 kilometer altitude error are a 33-nautical miles/61-kilometer detection range with a ±15 degree scan field of view. Thus, both the velocity penalties and rendezvous requirements for low earth orbit transfers are reasonable for the navigation errors assumed.

The second mission type if LEO to geosynchronous orbit transfers. The apogee position uncertainties at the transfer midpoint (≈9600 nautical miles/178 kilometer altitude) were roughly evaluated, assuming the navigation errors listed before at transfer initiation, and nine horizon tracker observations from 90 degrees beyond perigee to the transfer midpoint, approximately 136 degrees beyond perigee. The horizon tracker errors during the transfer were those mentioned before, and the time interval for the nine observations is about 56 minutes. The estimated three-sigma apogee position errors obtained from the rather approximate analysis performed are 40 nautical miles/74 kilometers downrange position, 77 nautical miles/142 kilometers altitude and 5.2 nautical miles/9.6 kilometers cross track. The downrange and cross-track errors are quite acceptable, but the 77-nautical miles/142 kilometers altitude error would require, for example, a detection range of 300 nautical miles/555 kilometers for a rendezvous sensor with a ±15 degree field of view. A recomputation indicates, though, that if the apogee altitude estimate (and transfer orbit correction) are delayed until near apogee, the three-sigma uncertainty can be reduced to 20 or 30 nautical miles (37 or 55 kilometer).

The third type of mission evaluated is transplanetary injection from a nominally circular orbit. It was assumed for simplicity that the orbit period is perfectly known, so that (using the same notation for the navigation errors as before),

$$\sqrt{\frac{r}{\mu}}\delta\dot{x} = \frac{-\delta y}{r} \qquad (3\text{-}1)$$

where

r = circular orbit radius.

It was also assumed that the RMS values of δz and $\delta \dot{z}$ are related by

$$\sqrt{\frac{r}{\mu}}\sigma(\dot{z}) = \frac{\sigma(z)}{r} \qquad (3\text{-}2)$$

Also, let

$$u = \sqrt{\frac{r}{\mu}}V \qquad (3\text{-}3)$$

$$V_\infty = \sqrt{V^2 - 2\frac{\mu}{r}} \qquad (3\text{-}4)$$

where

V = velocity after the insertion burn

V_∞ = asymptotic earth escape velocity

The insertion burn was assumed to be impulsive, coplanar, and in a fixed inertial direction, with no errors. Finally, we define an x, y, z coordinate system with x along the asymptotic velocity direction, y normal to x in the orbit plane, and z normal to the orbit plane. The asymptotic velocity errors are

$$\delta\bar{V}_\infty = \left[\bar{1}_x \frac{u-1}{\sqrt{u^2-2}} + \bar{1}_y \frac{2-u}{u^2-1}\right]\delta\dot{x} + \bar{1}_y\, u\frac{\sqrt{u^2-2}}{u^2-1}\delta\dot{y} - \bar{1}_y \frac{V_\infty}{u+1}\frac{\delta x}{r} + \bar{1}_z V_\infty \frac{\delta z}{r} \qquad (3\text{-}5)$$

The z term in this expression is the root sum square (RSS) addition of the out-of-plane velocity errors due to δz and $\delta \dot{z}$.

Evaluating the above expression for the horizon tracker three-sigma navigation errors previously given, for V_∞'s up to 23,000 fps (7010 mps), and taking the root sum square of the several terms, gives total velocity errors of 21 fps (6.4 mps) or less. Translunar insertions would require corrections of similar magnitude. The velocity errors are quite acceptable from the standpoint of delta-V penalty. However, determination of the V_∞ velocity vector errors to satisfactory accuracy probably would require radio ground tracking.

Horizon tracker navigation errors and their effects have been discussed at some length because, at present, horizon tracking appears to be the only reasonable prospect for fully autonomous, automatic earth orbit navigation. The delta-V penalties introduced by the navigation errors are quite reasonable. For LEO to geosynchronous orbit rendezvous transfers the navigation accuracy is marginal and may require either a long range rendezvous sensor (radar) on the tug, imposing a substantial weight penalty, or more accurate semi-autonomous or non-autonomous navigation techniques before or during the orbit transfer. However, estimation of the critical apogee altitude uncertainty requires a more thorough and exact analysis than was possible here. Horizon tracker earth orbit navigation accuracy before translunar or transplanetary insertion is satisfactory, provided that the resulting velocity errors after insertion can be accurately determined by other means.

Although automatic landmark tracking techniques offer the potential of substantially increased navigation accuracy, they have not been proved in space. Therefore, horizon tracking was selected as the baseline technique for autonomous earth orbit navigation. Should automatic landmark tracking be proved in space, it should be reconsidered for use on the space tug.

Manual Landmark Tracking. On manned space tugs, manual landmark tracking has been selected as a second mode of earth orbit navigation for several reasons: First, it can provide improved position (but not velocity) accuracy. Second, landmark tracking provides a backup mode of navigation in the event of horizon tracker failure. Third, a manual telescope, as discussed later, is useful for manned lunar missions.

Landmark tracking navigation accuracy has been estimated in several reports, including Reference 3-2, in the range of 0.1 to 0.2 nautical miles (0.18 to 0.36 kilometers), one sigma, assumes one milliradian RMS sighting errors. Apollo experience with landmark tracking has been somewhat disappointing. Simulations with the NR mission evaluation simulator have encountered state vector convergence problems. Thus, although the technique potentially offers greatly increased navigation accuracy, experience casts some doubts about its usefulness.

Semiautonomous Methods. These methods use signals from ground radio sources or navigation satellites. Existing ground radio sources such as broadcast AM or FM and aircraft navigation aids were initially considered. Broadcast AM and long-range aircraft navaids such as loran do not penetrate the ionosphere. Higher-frequency sources such as broadcast FM or VORTAC generally have a limited reception range and are radiated in a horizontal pattern so that little energy goes into space. Thus, the feasible alternatives seem to be either ground radio beacons or navigation satellites.

The EOS team has recently completed a tradeoff of navigation techniques for the EOS and selected a network of ground radio beacons to provide range and one-way doppler data. This replaces an earlier selection of manual landmark tracking for EOS navigation. Assuming that the ground beacon system is actually built, it will also be used by the space tug. The shuttle beacon network will not be adequate in itself for tug navigation though, since relatively few beacons are planned and up to three orbits may elapse between beacon contacts. Thus, either a much more extensive beacon network would be required for space tug navigation or the EOS beacon network would be used only to update the on-board navigation measurements whenever beacons were within range.

Several estimates of LEO navigation accuracy with ground beacons are available in the literature; for example the navigation survey or Reference 3-5. This reference estimates, for one-way doppler ground beacons, and three beacons available per orbit, average RMS position errors of 1000 feet (305 meters) building up to 1800 feet (549 meters) before acquisition of the next beacon.

Navigation satellite systems, both existing and proposed, are also discussed in Reference 3-5. These range in altitude from the Navy TRANSIT satellites at 600 nautical miles/1111 kilometers altitude, to proposed geosynchronous systems such as the NASA TDRSN and the AF system 621B. Estimated navigation accuracies with these systems are in the range of a few hundred to a few thousand feet. These navigation satellite systems would be an alternative to the proposed EOS ground beacon network for intermittent (updating) or continuous space tug navigation.

If a ground beacon or navigation satellite system is deployed, the space tug will utilize it, since none of the possibilities for fully autonomous navigation is entirely satisfactory with respect to accuracy, crew involvement, or proved feasibility. If the radio link operates at S band, little if any additional equipment will be required on the tug since it will have S-band communications equipment for data transfer.

Earth Orbit Navigation Summary. The selected earth orbit navigation techniques for the space tug are horizon tracking and the proposed EOS ground radio system, with perhaps an expanded beacon network. On manned tugs these will be augmented by manual landmark tracking, if required. If a NAVSAT system is deployed, it will also be utilized. The selected earth orbit navigation techniques are depicted in Figure 3-1.

3.3.1.2 Cislunar Navigation

Cislunar space can be divided into near earth and near moon regions for purposes of discussion. Navigation techniques for the near earth region would be applicable not only to lunar missions, but to the earth escape phase of planetary missions.

Automatic, autonomous methods of near earth navigation are essentially limited to horizon tracking. (Horizon trackers are available that operate from low earth orbit to synchronous altitude.) The accuracy of current horizon trackers, even at synchronous altitude, is inadequate for planetary missions and of questionable adequacy for lunar missions. If currently available horizon trackers were used, a separate earth tracker would be required for sightings beyond synchronous altitude.

In near moon space, a horizon tracker that determined the geometric center of the moon from tracking the bright lunar limb appears feasible

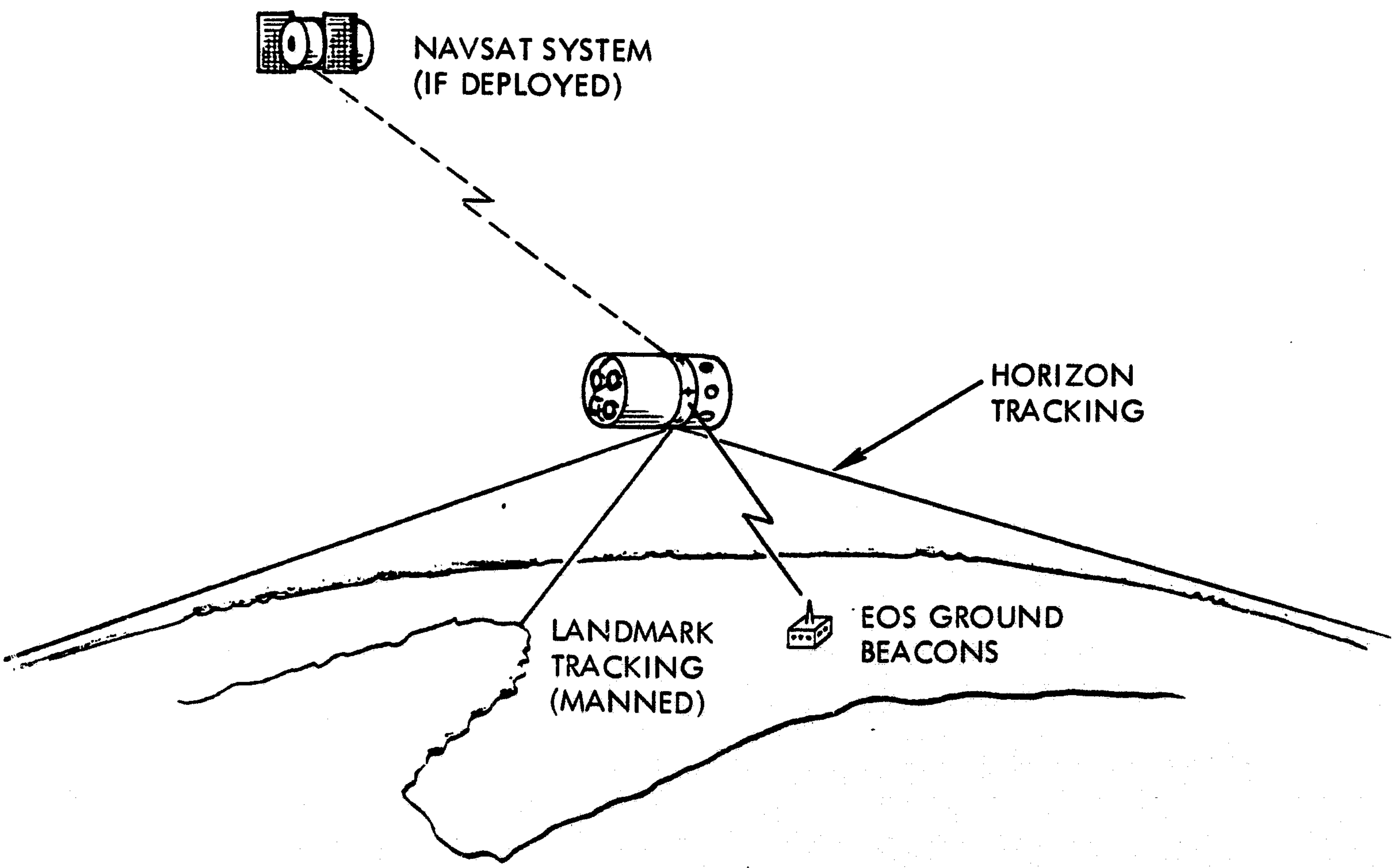

Figure 3-1. Earth Orbit Navigation Methods Selected for Space Tug

and would probably provide adequate navigation accuracy, but with some penalty in midcourse correction delta-V requirements.

A lunar and planetary horizon sensor is now being investigated by Barnes, Inc.

On manned missions, a manual sextant or telescope to take horizon sightings can provide adequate navigation accuracy in either near earth or near moon space. Sextant sightings taken on Apollo 8 during translunar flight predicted the perilune altitude within 1.8 nautical miles (3.3 kilometers) nearly 20 hours before LOI. Sextant sightings of the earth for earth return will not produce as good perigee accuracy because of the uncertainty in altitude of the atmospheric horizon read by the navigator. The mean horizon altitude may vary from one navigator to another on the order of 5.4 nautical miles (10 kilometers) and the horizon fuzziness increases the random sighting errors. On Apollo 8, the random sighting errors were about 10,000 feet (3 kilometers) in horizon altitude. These errors will still allow a safe return from the moon and insertion into earth orbit, but may be inadequate for translunar navigation.

The recommended equipment complement for cislunar navigation is, for the present, an earth horizon tracker and, for manned tugs, a manual telescope. Both sensors will require star trackers for basic attitude reference. Unmanned missions will require ground radio tracking by the DSN or MSFN, and such tracking is desirable for manned missions as it provide a much more accurate state vector than on-board sightings. While dependence on ground tracking does not conform to the autonomous operation guideline, it is felt that, realistically, lunar missions will not be conducted without use of the most accurate navigation techniques available.

A telescope, rather than a sextant, is recommended for the tug to save weight. Alignment of the telescope base (in the CM) to the star trackers (in the IM) can be determined by sighting the telescope on known stars while the star trackers are locked onto known stars. To reduce telescope-star tracker alignment errors during sightings, it may be necessary (as is currently done on Apollo) to hold the spacecraft in the sighting attitude for 20 minutes or so before the sightings are made to achieve thermal equilibrium.

Horizon sightings will require the telescope to be equipped with a rotating optical reticle that can be automatically or manually driven to place a horizon line in the reticle tangent to the horizon. This defines the point on the earth or moon limb on which the sighting is taken. If the reticle is automatically driven, it will allow the G&N computer to choose the point on the limb for making the sighting(s) that produces the greatest reduction in state vector uncertainty. This procedure appears to be operationally simpler than the current Apollo technique for making star-horizon sextant sightings.

Autonomous automatic cislunar navigation capability could be provided by adding an earth tracker for operation beyond geosynchronous altitude and a lunar limb tracker. The penalties would be increased IM weight (on the order of 50 pounds/23 kilograms), development cost, and midcourse correction delta V. The cislunar navigation accuracy and delta-V penalties from using such sensors have not been determined.

Lunar Orbit Navigation

Initial lunar orbit navigation will occur in a nominally 60-nautical miles (111-kilometers) orbit prior to descent orbit initiation (DOI). Lunar orbit navigation could be accomplished by MSFN tracking from earth by on-board terrain correlation, by landmark tracking, or by a horizon sensor.

MSFN tracking produces accurate orbital elements along the radio line of sight, but much poorer accuracy normal to the line of sight. Thus, for polar orbits, if the orbit plane is near the 0 to 180-degree meridan, the in-plane orbital elements will be well defined, but the orbit plane inclination and node will be relatively poorly defined. Conversely, an orbit near the 90 to 270-degree meridian will have the orbit plane well defined, but the in-plane orbit elements will be poorly known. References 3-6 and 3-7 indicate that three-sigma orbit determination errors with MSFN tracking for Apollo-type orbits are 3000 to 6000 feet/914 to 1830 meters down range, 500 feet/152 meters in altitude, and 10,000 feet/3048 meters in cross-range position. Navigation accuracies for 0 to 180 degrees longitude polar orbits should be comparable.

For on-board navigation, two lunar terrain matching techniques were considered. The first is radar terrain contour correlation. This does not appear feasible, when reasonable radar beamwidths are considered, at 60 nautical miles (111 kilometers) altitude. The second technique is optical correlation of the terrain pattern with stored images. Experimental studies at Boeing have indicated that lunar surface features wash out completely for high sun angles and when viewed within about 20 degrees of the sun line. Thus, it is estimated that optical image correlation of lunar surface features would not be feasible within 20 to 30 degrees of the lunar noon. The method would also require accurate reference images obtained under similar lighting conditions. In summary, terrain matching techniques do not appear feasible for lunar orbit navigation.

The other on-board lunar navigation technique considered is landmark tracking. On manned tugs this would be accomplished with the manual telescope already discussed. On unmanned tugs, a gimbaled TV camera is proposed. It would be remotely viewed and pointed via uplinked commands from either an orbiting lunar station (OLS) or from earth if there is no OLS. The TV camera is also used during terminal descent prior to landing to locate the landing site. To facilitate TV landmark tracking, the computer would keep the camera pointing at the estimated landmark location, based on the last received pointing commands. Telemetered pointing update commands would change the computed landmark position. This pointing mechanization is similar to that of the LM landing point designator (LPD) system.

Since the TV camera would be part of the lunar landing kit, perhaps mounted on part of the landing gear, significant alignment uncertainties with IM-mounted star trackers must be assumed. But even assuming 0.6-degree maximum pointing errors (alignment and tracking errors), the down-range and cross-range position errors relative to a landmark directly below the spacecraft would be only about 3800 feet (1158 meters), which is acceptable for navigation prior to landing. The position fix error when the manual telescope is used should be less than 0.1 degrees, or 750 feet (229 meters).

The limitations due to lighting conditions mentioned for optical image correlation would roughly apply also for landmark tracking. In addition, horizon sightings could be made with the manual telescope near the terminators on the dark side of the moon. These limitations indicate that MSFN tracking for one or two orbits prior to DOI will be desirable or necessary.

Landmark tracking after DOI prior to powered descent initiation (PDI) can be used to update the spacecraft position. If a landmark near perilune is chosen, the much lower altitude would permit more accurate position fixes. For example, if a 0.6-degree error with the remotely pointed TV camera and 50,000-feet (15240 meters) altitude are assumed, the position error would be only 500 feet (152 meters).

Regardless of the navigation method used, lunar orbit navigation accuracy is limited by uncertainties in the lunar gravitational field.

Reference 3-8 indicates that orbit propagation uncertainties may be as large (three sigma) as 1 nautical mile (1.85 kilometer) in cross range and 0.5 nautical mile (0.93 kilometer) down range after one orbit.

To summarize, the recommended navigation methods for lunar orbit navigation are landmark tracking with either a remotely pointed TV camera (unmanned) or manual telescope (manned) and MSFN tracking prior to landing.

Navigation Summary

Potential navigation techniques for space tugs have been discussed at length because it is a significant problem. For earth orbit navigation, the recommended techniques are horizon tracking and the proposed EOS ground radio beacons. Horizon tracking navigation alone is acceptable for most missions, but is marginal for LEO to geosynchronous transfers, even with a midcourse correction based on horizon tracking data taken during the transfer.

For unmanned tug cislunar navigation, MSFN or DSN tracking is recommended. Autonomous navigation probably could be achieved by adding an earth tracker for use beyond geosynchronous altitude and a moon limb tracker. The navigation accuracy, estimated weight, and delta-V penalties have not been evaluated; but the estimated weight penalty of 50 pounds (22.7 kilograms) is unattractive.

On manned tugs, a manual telescope was selected for landmark tracking in earth and lunar orbits and horizon sightings in cislunar space. The cislunar navigation accuracy with the telescope should be acceptable, but not as good as MSFN tracking.

In unmanned lunar orbits, landmark tracking can be performed with a remotely pointed TV camera. However, MSFN tracking probably will be required, on both unmanned and manned tugs, prior to landing.

3.3.2 Attitude Reference Sensors

The need for star trackers to determine the spacecraft attitude during use of the on-board navigation sensors (horizon tracker, manual telescope, and remotely pointed TV camera) has been mentioned. Star trackers can provide an inertial attitude reference in earth orbit, cislunar space, and lunar orbit. For IMU realignment on the lunar surface, local vertical can be determined by accelerometers and azimuth by a star tracker. Gimbaled star trackers were selected to provide adequate fields of view in space with minimum vehicle attitude maneuvering. A combined tracker and gimbal angle resolver accuracy of 20 to 30 arc seconds is obtainable. Two star trackers are needed for FO/FS failure tolerance and simultaneous operation of the two trackers. This will allow attitude hold in coasting flight with the IMU shut down. Failure of one tracker will still allow IMU realignment by sequentially acquiring two known stars. For FO/FO/FS failure tolerance, three star trackers are required.

Another approach, selected for the EOS, is a nongimbaled star pattern mapper. With +3 star magnitude detector sensitivity and a 34- x 34-degree field of view, two stars are always visible. A resolution of 30 arc seconds would be obtainable.

In addition to the star trackers, four wide-angle sun sensors, mounted 90 degrees apart around the IM, were selected. Each sun sensor will have 90 to 100 degrees angular field of view in roll and 180 degrees field of view longitudinally. To guard against the sun sensors seeing the attitude control jet exhaust plumes, it is suggested they be mounted on top of the ACS pentads.

The sun sensors will provide a quick attitude reference to facilitate star acquisition and will allow an attitude hold mode of operation with either end of the spacecraft pointed toward the sun to minimize propellant boiloff with low electrical power consumption. In earth orbit, a horizon tracker can also be used to facilitate initial attitude acquisition.

Equipment Location

To minimize alignment errors between the star trackers and horizon tracker(s), all should be mounted on a common navigation base with the IMU. However, since the earth subtends an angle of about 155 degrees in a 100-nautical mile (0.185-kilometer) orbit, mounting the star trackers near the horizon tracker would severely restrict the star tracker fields of view. Therefore, the equipment layout tentatively recommended is to place the star trackers and IMU on a common navigation base and the horizon tracker(s) on the other side of the IM. The alignment errors resulting from this arrangement must be evaluated by an analysis of the potential deflections of the IM structure.

3.3.3 IMU Configuration

The principal considerations in selecting a candidate IMU were that it meet the failure tolerance requirements and, to minimize development cost, that it be the same IMU as some other element of the IPP. Contemporary IMU's generally have adequate performance (accuracy) for space tug requirements. Hence no evaluation of IMU errors was attempted for this initial study phase. A possible exception to the adequate accuracy statement is retrobraking from lunar orbit for landing with a low thrust/weight ratio spacecraft, where accelerometer biases can cause position errors on the order of one nautical mile (1.85 kilometer), three sigma (Reference 3-9). However, calibration of the accelerometer biases during orbital coast can reduce these errors by a factor of three, which produces acceptable position errors.

The NR design for the earth orbiting space station selected a strapdown Hexad IMU (six gyros and six accelerometers with input axes normal to the faces of a regular dodecahedron). The configuration allows continuous comparison of the gyro and accelerometer outputs during operation to detect failures and permits complete attitude reference and velocity measurement with three gyros and/or three accelerometers failed. It thus provides FO/FO/FS failure tolerance. The strapdown configuration allows replacement of individual gyros and accelerometers and eliminates the relatively high failure-rate slip rings necessary with an all-attitude stable platform.

The NR shuttle (EOS) also originally selected a strapdown Hexad IMU. However, this decision recently has been changed, and four gimbal platforms (to provide FO/FO/FS redundancy) are now proposed. Low-cost platforms, such as the Kearfott KT-70 and KT-84 models with the simple gyroflex gyros, are being investigated. Primary reason for the reselection of the IMU was development cost of the proposed strapdown Hexad unit (which was not the same IMU as that proposed for the space station).

The IMU selected for the space tug is the proposed strapdown Hexad IMU for the space station, manufactured by Hamilton Standard. The Hexad IMU configuration can be reduced to a pentad configuration (five gyros and five accelerometers) if only FO/FS failure tolerance is required. The input axis configuration is no longer completely symmetrical, as is the Hexad arrangement, but would still allow complete IMU operation with any two gyros and/or two accelerometers failed. A pentad configuration has been assumed for estimating weight and power with FO/FS failure tolerance.

Another IMU considered for the tug is the Micron IMU being developed by Autonetics. This unit uses two electrostatically suspended, 1-cm diameter, spinning balls to measure altitude and acceleration, and an MOS integrated circuit computer for preprocessing the inertial data. The physical characteristics of the IMU are approximately 3 pounds (1.4 kilogram) 12 watts, and 0.05 cubic feet/0.0014 cubic meter. The design goal for accuracy is 0.01 degrees/hour, and the MTBF (with AF derating) is 2000 hours. The low acceleration levels experienced in space would allow the electrostatic levitation voltage to be lowered, with a resultant reduction in operating power.

A potential shortcoming of the micron IMU is the angular pickoff accuracy, which may be on the order of one milliradian. This would require the star trackers to be used for attitude reference during navigation sightings.

The Micron IMU was not selected for the space tug because it is still under development. However, the order of magnitude or more reduction in weight, volume, and power, compared to the Hexad IMU, makes it extremely attractive. Should initial tests, within the next two or three years, prove out the concept, it should be considered for the tug.

3.3.4 Computer Requirements

GN&C computations will be performed by the communications and data management system computer complex, and the weight, volume, and power requirements are included in the C&DM computer estimates. Computer organization for FO/FS and FO/FO/FS failure tolerance are discussed in the C&DM system section.

GN&C computations will be required for navigation data processing, guidance, attitude control, thrust vector control (including throttling), alphanumeric crew displays, accepting pilot commands and inputs, and overall mission sequencing and control. The requirement for automatic, autonomous tug operation will require that all GN&C operations be scheduled in a master executive program (MEP). These operations will occur not only at specified times, but also on the basis of flight variables such as range, altitude, prior sensor acquisition, etc. It is believed that the diversity of constraints and objectives for space tug missions will require a separate MEP to be assembled for each mission. Each MEP will be composed of standard building blocks, and the assembly can be semi-automated with a program compiler. The MEP will be loaded into the tug computer memory directly or via uplink from the ground in the case of tugs already in space.

The memory requirements for tug GN&C operations were grossly estimated by extrapolation from the Apollo LM computer memory (approximately 38, 000 words). The major new requirements for the tug are processing for alphanumeric displays and the MEP. The estimated memory requirement for tug GN&C function is 50, 000 words.

Word length and computational speed were not evaluated. A 24-bit word length would be adequate for all GN&C computations except navigation, which would require double-precision operations. A 32-bit word length probably would eliminate the need for double-precision computations, especially if the Apollo W-matrix technique were used, but would require many split word operations for efficient memory use.

Autonomous cislunar navigation and midcourse guidance (possible only on manned missions with the selected navigation sensor complement) imposes substantial on-board computation and memory requirements. On Apollo, the spacecraft state vector can be updated in cislunar space but midcourse guidance capability is not included. Autonomous G&N computation techniques for the space tug will require detailed investigation and may increase the G&N computer memory requirements above the estimated 50,000 words.

3.3.5 Lunar Landing Sensors and Guidance

Special sensors are required during the terminal phases of lunar landing to:

1. Measure altitude and velocity relative to the lunar surface.
2. Locate the intended landing site.
3. Detect surface obstacles before touchdown.

The sensors for performing these three functions are discussed in the following subsections. Lunar landing guidance is discussed in the final subsection.

Landing Radar

The Apollo LM landing radar was selected to measure altitude and vertical and lateral velocity components relative to the lunar surface. A second choice would be a modification of the landing radar being developed by Ryan for the Viking Mars lander. This would save a few pounds, but not enough is known about the Viking radar or the modifications that might be needed to adopt it for lunar landings.

Laser radars have also been suggested for landing radar systems, but present several potential drawbacks: First, no such system is currently under development to our knowledge. Second, the feasibility of obtaining laser doppler returns from the porous lunar surface appears questionable. Third, power limitations might limit the region of operation to less than the 30,000-feet (9144 meter) altitude capability of the present LM landing radar. Fourth, a laser radar would be blinded by dust during the last few hundred feet of descent and probably would be susceptible to false returns as this region was approached. Laser landing radars do not appear attractive.

Landing Site Locating Sensor

The spacecraft location relative to the intended landing site must be determined during the terminal phases of landing to remove accumulated navigation errors. The potential sensing methods for landings at new sites are basically the same as those mentioned for lunar orbit navigation: terrain contour matching or optical image correlation and optical tracking of the landing site. For follow-on landings at previously reached sites, the rendezvous radar can track a beacon or reflector at the site.

Terrain contour matching, at altitudes below about 30,000 feet (9144 meter), probably would be feasible, although some lunar mare areas may not contain enough surface features to allow a contour match. Contour measurement could be performed by the landing radar altimeter beam. Contour matching would be independent of lighting conditions, but would require an accurate contour map of the approach corridor to the landing site for each mission.

Terrain image correlation would also be feasible under suitable lighting conditions, but would require a reference image of the approach corridor under similar lighting and would require new hardware development.

Landing site tracking could be accomplished on manned missions with the manual telescope and on unmanned missions with the remotely pointed television camera discussed under lunar orbit navigation. The TV camera would be controlled from an OLS or from earth. An early simulation study (Reference 3-10) indicates that the earth-moon round-trip delay of 2.6 seconds is not a significant problem. Reference 3-9 estimated that a 500 by 500 element TV picture would provide sufficient field of view and resolution for identifying the landing site. With a one-second frame time, the required transmission bandwidth would be approximately 0.8 MHz. The principal drawbacks to optical landing site tracking are the lighting conditions required during landing and, for unmanned tugs, dependence on communications with earth or an OLS during landing.

Despite these restrictions, landing site tracking with the manual telescope or remotely pointed TV camera is the recommended technique for landing site identification for landings at new sites. The principal advantage is the capability for visual inspection of the landing site during the terminal guidance phase and redesignation to a nearby site if necessary. This selection is predicated on the opinion that landing at new sites without visual inspection during final approach is not a realistic mission requirement. Follow-on landings at night for resupply or rescue can still be accomplished by tracking a beacon or reflector at the landing site, as mentioned.

Pilot or teleoperator workload and TV transmission delay effects can be minimized by automatically pointing the telescope or TV camera at the computed landing site location and using manual pointing commands only to update the computer location. This is similar to the LM landing point designator mechanization.

Reference 3-9 estimates the required field of view for landing site tracking to be 21 x 21 degrees. This would allow the manual telescope to have a magnification of approximately 3X and still retain the field of view needed to assure landing site acquisition.

Surface Obstacle Detection

The final sensing function before touchdown is to determine if the selected site is smooth enough to permit a safe landing. The problem is compounded by the dust cloud stirred up by the rocket exhaust during the final few hundred feet or descent, which blocks direct visual inspection of the site.

Reference 3-9 discusses two techniques for surface obstacle detection. The first is a mapping radar that would scan a surface area about 50 feet (15 meter) square directly under the spacecraft from 100 to 200 feet (30.4 - 60.8 meter) altitude. Data processing of the radar range (altitude) returns would detect any abrupt elevation changes (rocks or small craters) greater than one to two feet (0.304 to 0.608 meter). Lateral resolution would be on the order of two to three feet (0.608 to 0.914 meter). The obstacle detection radar would be a new development.

The second technique for obstacle detection proposed in the reference is to freeze the landing site tracking TV picture at 500 to 1000 feet (152 to 304 meter) altitude, before the landing site is obscured by dust. The frozen TV picture would then be continuously displayed on the teleoperator's screen or in the CM. This could be accomplished with a simple video-tape loop.

The spacecraft position in the picture would be computed from its position at the time of picture freeze (based on spacecraft altitude, altitude and TV camera pointing angle at the instant of picture freeze), and the spacecraft velocity since picture freeze, from IMU acceleration and landing radar velocity measurements. The computed spacecraft position also would be displayed in the frozen TV picture. The static picture of the lunar surface with the current spacecraft position superimposed would provide the teleoperator or pilot with sufficient information to maneuver the spacecraft to a safe touchdown area.

Resolution of surface features in the frozen TV picture, assuming a 500-line picture resolution and a low sun angle, would be more than adequate to detect dangerous rocks or craters. A potential problem with the concept is the spacecraft position uncertainty, which accrues in the minute or so of time from picture freeze to touchdown due to lateral velocity uncertainties. On the Apollo LM, the three-sigma velocity error at touchdown is estimated to be 0.8 fps (0.24 ms). Thus, for example, if the time from picture freeze to touchdown were 90 seconds (a reasonable value), the spacecraft 3-sigma position uncertainty in the frozen picture would be about 75 feet (23 meter). This uncertainty might be reduced by modifying the landing radar to provide more accurate lateral velocity measurements at low altitudes.

Figure 3-2 is a sketch of a frozen television picture display. The three-sigma spacecraft lateral position uncertainty circle, about the computed spacecraft position, is superimposed on the frozen picture of the lunar surface. Altitude, altitude rate (h dot), and the hover time remaining are also shown in the display.

The selected obstacle detection technique for the space tug is the frozen, or instant replay, TV picture concept just described. The selection is made on the basis of excellent surface feature resolution and minimum requirements for new hardware development. The restrictions on sun angle during landing inherent in the concept is consistent with the lighting restrictions already assumed for optical tracking of the landing site.

Failure Tolerance

Fail operational/fail safe (FO/FS) tolerance will require two landing radars and two gimbaled landing TV cameras (unmanned) or one landing TV camera (manned). On manned tugs, landing site tracking is normally done with the manual telescope, although the landing TV would be used as backup for landing site tracking if the telescope failed. If both landing

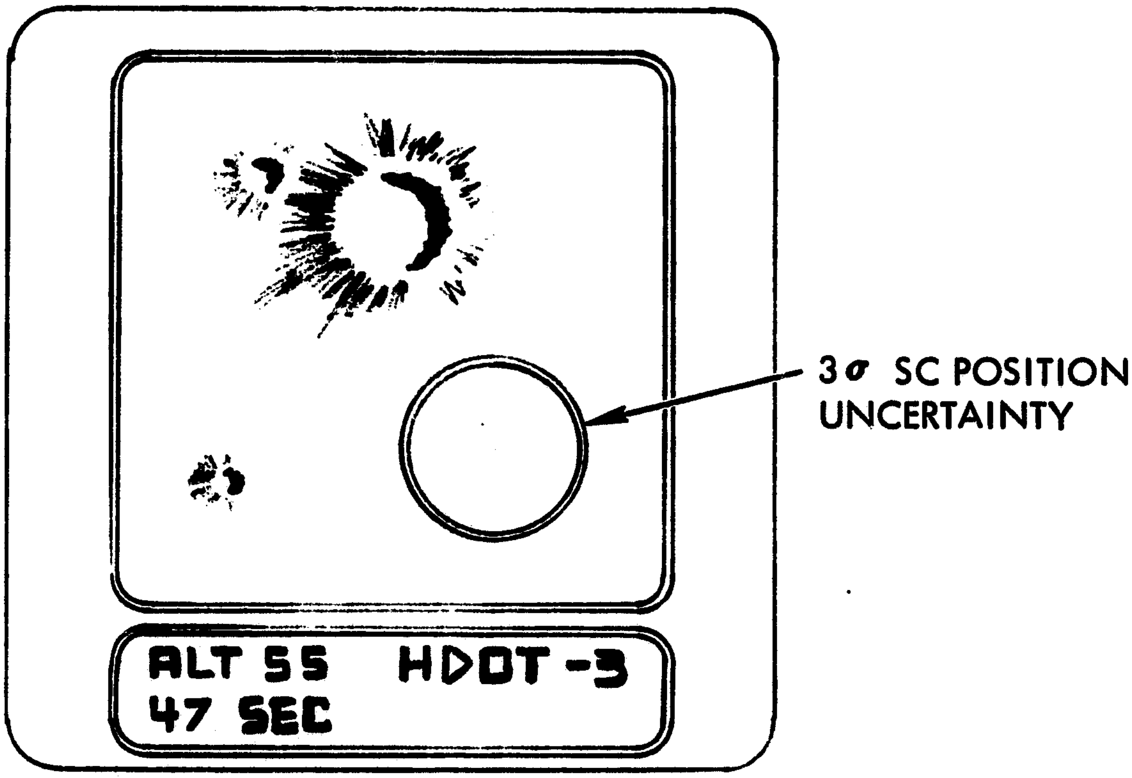

Figure 3-2. Frozen Television Picture for Lunar Landing Surface Obstacle

radars fail before landing or the landing TV camera and the manual telescope both fail before the landing site is reached, the fail safe mode of operation is to abort the landing. For landing TV failures during the surface obstacle detection phase (during the final near-vertical descent to touchdown), pilot inspection of the landing site prior to visual blackout by the rocket-generated dust cloud is considered to be a fail operational mode of landing. If the pilot decides that a landing based on prior visual inspection of the landing area is unsafe, the landing must be aborted. Two facilitate landing site visual inspection, outward leaning windows for the two crewmen are recommended. For FO/FO/FS failure tolerance, one additional landing radar and landing TV camera must be added in each case.

Landing Terminal Guidance

Selection of specific guidance laws is beyond the scope of this initial study phase, but the terminal guidance equations developed in Reference 3-9 are recommended for consideration for the space tug. Terminal guidance commences after the intended landing site is acquired with the telescope or landing TV camera, within a few miles of the landing site and at a velocity of a few hundred feet per second.

The nominal trajectory computed by the guidance equations of Reference 3-9 is an initial thrust-off coast, followed by a final constant thrust arc to the terminal point at a constant spacecraft angular rate.

At the start of terminal guidance, after landing site acquisition and the first trajectory computation, maximum thrust is maintained for one guidance cycle. The thrust direction is defined by extrapolating the constant attitude rate of the final thrust arc backwards in time to the initial time. The resulting initial thrust direction is near optimal for reducing the duration of the final thrust arc.

At the next guidance cycle, a new nominal trajectory is computed, based on the present spacecraft velocity and position relative to the landing site. The total delta V of the new nominal trajectory, including the delta V expended during the preceding guidance cycle, is then compared with the final thrust arc delta V of the previously computed nominal trajectory. If the computed delta V has decreased, maximum thrust is maintained for another guidance cycle. Otherwise, the engine is throttled back to minimum thrust.

During the minimum thrust coast, the thrust direction is again defined by extrapolating the computed angular rate of the final thrust arc backwards to the current time. At the end of the coast, the engine is throttled up to the preselected thrust level and, if the terminal point is not changed, the constant-angular-rate fixed-thrust program is flown to the terminus. If the landing site is redesignated during the final thrust arc, a new constant-rate thrust program at a modified constant thrust (within the engine capabilities) to reach the new landing site is computed. Thus, the nominal thrust level selected for the final thrusting arc is a compromise between minimum propellant consumption (maximum thrust) and landing site uprange redesignation capability.

The thrust direction at the terminal point is monitored during each guidance cycle to avoid unsafe flight path angles at the terminal point. If the estimated final thrust direction is too far from vertical, maximum thrust in the vertical direction is commanded until the situation is corrected.

The guidance equations and logic were verified in a three-dimensional digital simulation. One trajectory from the simulation is shown in Figure 3-3. The figure shows most of the final thrust period (after the

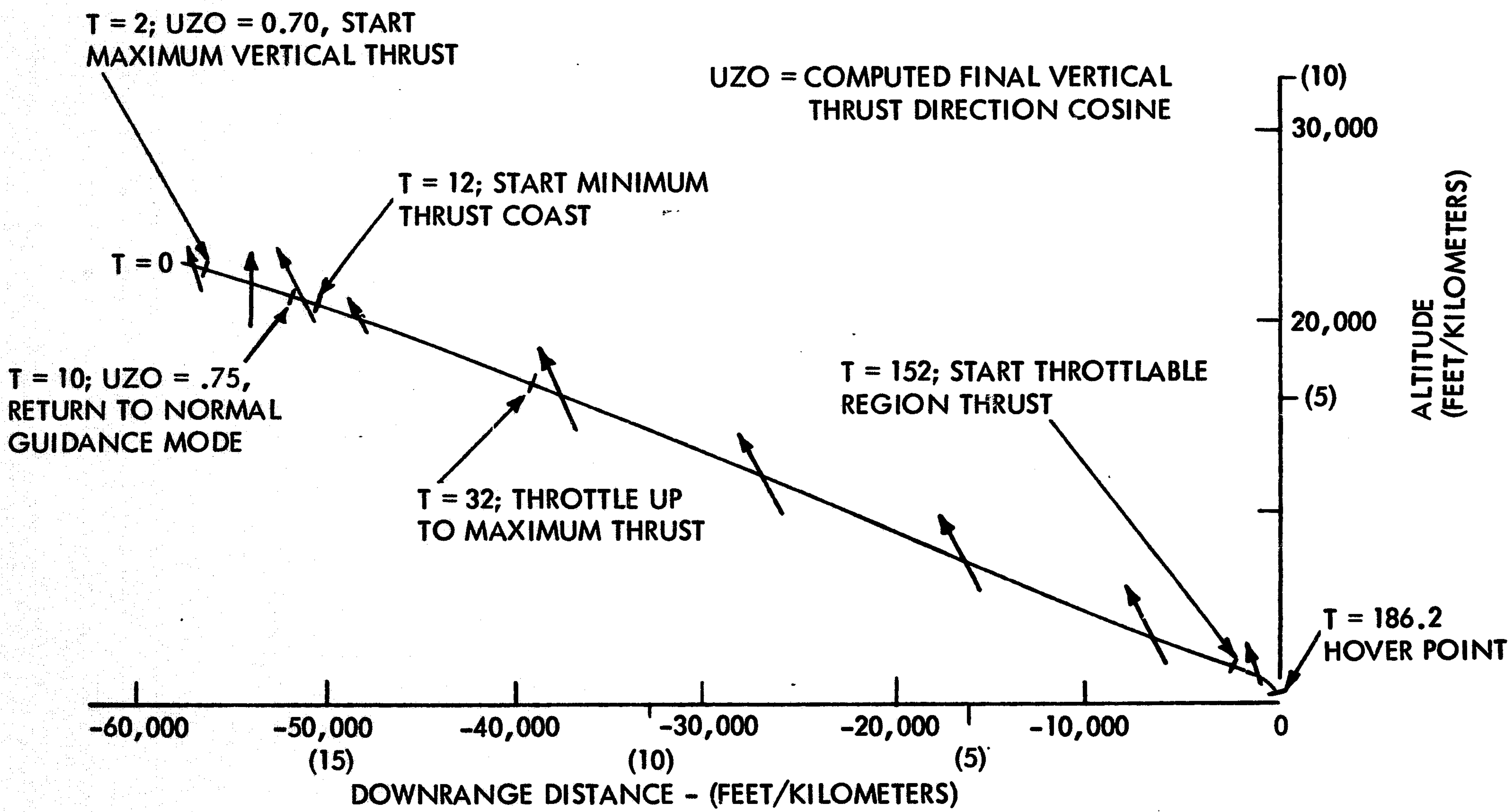

Figure 3-3. Terminal Guidance Vertical Thrust Mode

minimum thrust coast) at maximum thrust, with a short part-throttle arc at the end. This occurs because the vehicle studied in the simulation used the LM descent engine (DPS), and the DPS throttling restrictions (continuously throttleable only below 67 percent of maximum thrust) necessitated this thrust schedule to retain limited uprange landing site redesignation capability.

The maximum-minimum-constant thrust sequence of the guidance scheme is the same as an optimal (minimum delta V) trajectory for landing (for the same thrust levels), but the constant-attitude-rate thrust-direction program differs somewhat from the optimal bilinear tangent program, and the guidance logic extends the initial maximum thrust period beyond the optimum throttle-down point. However, analysis shows that the constant attitude rate thrust direction program requires less than two fps more delta V for landing than the optimal thrust direction program, and examination of the simulation results shows that the try-it-and-see throttle-down guidance logic also produces less than two fps delta V penalty for a two-second guidance cycle. This latter penalty could be reduced by simple modifications to the guidance logic. Thus, the constant angular rate guidance scheme of Reference 3-9 provides near optimal performance.

However, the constant attitude rate guidance equations are far simpler to solve than determining the optimal bilinear tangent thrust program parameters. In the digital simulation, the guidance equations and logic required only about 700 words of computer memory, compared to the 3000 words of memory estimated in Reference 3-2 for a true optimal guidance scheme. (The constant rate guidance memory requirements appear to be also somewhat less than the memory requirements for the comparable LM guidance program, although a precise comparison is difficult.)

Although constant attitude rate guidance was evaluated only for lunar landings, the near optimal performance and modest memory requirements make the concept attractive for other thrust maneuvers as well. Principal modifications required would be the inclusion of, approximately, central gravity field effects and modifications of the end point constraints to allow, for example, the final down-range position to be unconstrained.

3.3.6 Rendezvous and Docking Sensors

The selected sensor for rendezvous and docking is the ITT laser rendezvous and docking radar. This system was chosen because it is in an advanced state of development and is the only system known that can provide relative position and velocity measurements for automatic docking. Also it has been chosen for the NR space station.

Two hundred docking simulations with the first generation ITT laser rendezvous and docking system produced RMS errors of 0.03 feet (0.009 meter) in the lateral position and 0.01 degree is pitch and yaw angle at docking. All runs resulted in satisfactory alignment for docking.

Laser Acquisition Considerations

The principal drawbacks to the ITT laser system are the requirement for a cooperative target and the limited acquisition range of 65 nautical mile (120 kilometer). This range is adequate for LEO rendezvous, but is marginal for geosynchronous orbit rendezvous with the estimated apogee altitude errors of 20 to 30 nautical mile (36 to 48 kilometer) (three sigma) using only horizon tracking data, and the ±15-degree search coverage of the active vehicle laser radar. System improvements to increase the laser detection range to about 85 to 90 nautical miles (157 to 167 kilometer) have been investigated, but this would still provide marginal acquisition capability. Even if the geosynchronous mission is to a still operating Comsat, Navsat, or TDRS, most of the LEO to geosynchronous orbit transfer will be outside of the usual antenna beamwidth of such satellites. Thus little navigation assistance from the target will be available during the transfer.

Either ground tracking of the tug or contact with one of the EOS ground beacons before or during the transfer probably would reduce the apogee altitude uncertainty to an acceptable value for laser radar acquisition of the target. However, range limitations of the EOS ground beacons, which are unknown, probably would limit beacon contacts to the first 45 to 60 degrees of the transfer (600 to 1000 nautical mile/1111 to 1852 kilometer altitude).

It may be reasonable to assume the availability of ground tracking for this phase since many of the unmanned satellites currently use this technique for deployment and stationkeeping. Current NR studies being conducted under contract to GSFC have shown that the accuracy of this technique is sufficient to obtain target acquisition.

On autonomous lunar missions (which can only be manned missions with the selected navigation sensor complement), some difficulty may also be experienced in locating an OLS with the laser radar after lunar orbit insertion. However, landmark tracking and lunar horizon sightings for one or two orbits should refine the orbit parameters sufficiently to permit laser acquisition if the OLS orbit ephemeris is known and it is within detection range. Thus, the primary target acquisition problem is for geosynchronous transfers.

Three approaches were considered for aiding target acquisition at geosynchronous transfer apogee:

1. Gimbal the active radar to obtain wider search angle coverage.

2. Place flashing light beacons on the target vehicle and acquire the target with the tug star trackers.

3. Add a microwave rendezvous radar on the tug and a transponder on the target to obtain longer range acquisition capability.

The first approach appears feasible if the relative velocity between the vehicle is small so that several minutes are available for acquisition. (Target acquisition for the 30 degree by 30 degree search field of view of the non-gimbaled unit may take up to 150 seconds.) For geosynchronous orbits, this would require that target acquisition be started after the tug circularization burn at apogee, since the tug-to-target closing rate before circularization is approximately 0.8 nmi/second (1.48 kilometer/sec), which would allow insufficient acquisition time.

The second acquisition technique, tracking a flashing light beacon on the target with a star tracker, has been adopted for the shuttle. This is to be used on conjunction with the ground beacon interrogator on the shuttle and a transponder on the target to obtain range and range rate. (The EOS originally also selected the ITT laser radar for rendezvous and docking, but adoption of manual docking as the normal mode of operation allowed other rendezvous-sensing techniques to be considered.)

The detection range of visible beacons is, at this point, undetermined. The shuttle-evaluation estimated detection ranges up to 200 nmi (370 kilometer). Limited data on Apollo experience in seeing the flashing light on the LM is given in Reference 3-11. Maximum detection ranges on the order of 100 nmi (185 kilometer) with the bare eye or with the Apollo scanning telescope were recorded. With the 28-power Apollo sextant, and a sunlit LM against a sky background, visual detection occurred at ranges up to 400 nmi (740 kilometer). Star tracker sensitivities generally allow detection of +2 to +3 magnitude stars, or no better than the human eye. This limited data are inconclusive, but indicate that star tracker detection of a flashing light beacon at 100 nmi (185 kilometer) or more is feasible, which would be adequate for acquisition with a gimbaled star tracker. When used in conjunction with the ground beacon interrogator, or a simple VHF or S-band ranging scheme, sufficient information for rendezvous would be obtained.

The third method for improved acquisition is the addition of a rendezvous radar. This method is less attractive than the other techniques discussed because of the requirement for additional equipment on the tug, but is definitely feasible. Reference 3-12 describes an advanced rendezvous radar design capable of operation at range rates up to those encountered in geosynchronous orbit rendezvous. The most significant system parameters are:

Detection range:	0 to 250 nmi (463 kilometer)
Maximum range rate:	0 to ±4900 fps (1493 mps)
Angular coverage:	$\pm 15^{\circ}$, ax. and el.
Range accuracy:	0.0014 percent or 1.2 feet/(0.36 m)
Range rate accuracy:	1 percent or 1 fps/0.3 mps
Angle tracking accuracy:	2 milliradians (3σ)
Acquisition time:	14.7 seconds
Weight:	
Radar and antenna:	30 lb (13.5 kilogram)
Transponder and antenna:	23 lb (10.4 kilogram)
Power:	
Radar	69 Watts
Transponder	32 Watts
Radar antenna:	24 x 24 inches (61 x 61 cm)
Frequency:	X band

A fixed interferometer array antenna provides the excellent angle resolution. Pseudo-random noise code ranging and separate transmitting and receiving elements provide ranging capability to near zero range.

This radar would solve the geosynchronous orbit rendezvous acquisition problem for a moderate-weight penalty. The measurement accuracies and minimum range also appear adequate for docking, but this capability is questionable and is not claimed for the system in the reference.

The geosynchronous orbit target acquisition problem with the rendezvous laser radar has been examined in some detail because it appears to be a significant problem for geosynchronous missions, and several potential solutions have been discussed. At this time, no recommendation is made as to which solution is preferable. Analysis of the alternative solutions and better estimates of geosynchronous transfer apogee position errors are required before a choice is made.

Noncooperative Rendezvous

Reference 3-12 also investigated a noncooperative rendezvous radar. For a 30-nmi (55-kilometer) detection range against a 5-square-meter (50 square feet) target, the estimated system characteristics are 94 pounds (43 kilograms) and an average transmitted power of 132 watts. Estimated measurement accuracies and range-rate capability are similar to those for the cooperative radar. Several problem areas of noncooperative radar design are mentioned in the reference, including return signal strength fluctuations with target aspect angle of up to 50 db. Because of the weight penalty and unresolved problems of noncooperative rendezvous radars, only cooperative rendezvous capability is proposed for the space tug. If a noncooperative rendezvous capability is required for some tug missions, a suitable radar must be developed and carried along as part of the payload.

Additional Docking Sensors

A nongimbaled television camera was also selected for monitoring automatic docking of the IM or PM, by the crew or on the ground. It was assumed that crew module docking could be monitored by direct vision by one crewman. Provisions for manual takeover of docking, by the pilot or ground monitor, will also be incorporated. Each docking port will also have a contact sensor or sensors to indicate when docking has been completed.

Failure Tolerance

For fail operational/fail safe (FO/FS) tolerance, two laser radars are specified. If both radars fail, then fail safe operation can be obtained, depending on the range to target and other factors when the radar failure occurs, by either aborting the rendezvous or completing the rendezvous, using a star tracker and radio ranging, and manually docking with the TV camera or by direct vision. For FO/FO/FS tolerance, three radars are required, with the same rationale for fail safe operation after the third radar failure.

Selected G&N Techniques/Sensors Summary

The conclusions of this section are summarized in Table 3-2.

Table 3-2. Selected Guidance and Navigation Techniques and Sensors

Function	Selected Technique
Attitude reference	• Star trackers • Acquisition sun sensors
Navigation	
Earth orbit	• Horizon tracker • Landmark tracking (manned)
Lunar orbit	• MSFN tracking • Landmark tracking with telescope (manned) or remotely pointed TV (unmanned)
Cislunar space	• MSFN tracking • Horizon sightings (manned)
Inertial measurement unit	• Strapdown Pentad IMU used on space station
Lunar landing	
Altitude and velocity	• LM landing radar
Landing site tracking	• Telescope (manned) or remotely pointed TV (unmanned)
Surface obstacle detection	• Frozen TV picture
Follow-on landings	• Surface beacon tracking
Lighting conditions	• Low sun angle required
Rendezvous and docking	ITT laser radar (Docking television camera monitoring and remote manual docking)

3.4 CREW MODULE GN&C EQUIPMENT

Although crew module G&N equipments were briefly mentioned in several places in the preceding section, this section more clearly defines the displays, controls, etc., needed in the CM. The concept of the space tug G&N is to perform most mission phases without crew assistance, but the crew can enhance the automatic capability and in a few phases crew participation or ground monitoring is required.

3.4.1 Crew GN&C Functions

The most frequently performed crew GN&C task will be taking navigation sightings with the manual telescope. Sightings will be taken, in various mission phases, on earth or lunar landmarks, earth or lunar horizon, and lunar landing sites. The proposed technique for taking horizon sightings with the telescope is described in the navigation discussions of the previous section. Telescope sightings on rendezvous targets may also be made to provide target acquisition beyond the range of the rendezvous laser radar.

The telescope will be designed, as on Apollo, to be automatically pointed by the computer to facilitate target acquisition, or manually pointed by the navigator with suitable manual controls. Hence, a more accurate name for the instrument is semi-automatic telescope. To facilitate rendezvous target or lunar landing site acquisition, and to increase the accuracy of navigation sightings, a 2X to 3X magnification is recommended for the telescope.

The crew will manually control the tug prior to lunar landings to select the touchdown point and may also manually control docking if the laser rendezvous and docking radars fail. The manual controls for docking will enable the pilot to manually control spacecraft attitude or perform small delta-V maneuvers with the ACS thrusters during any mission phase. Capability will also be provided, as on Apollo, for manual takeovers during main engine thrust maneuvers.

The crew will also control, through switch settings or computer keyboard inputs, the G&N system configuration. This includes equipment on-off control and spacecraft control modes, deadbands, etc. In addition, as part of the status monitoring of all systems, the crew will monitor the G&N system status via appropriate displays.

An additional important function will be performed by the crew on manned quick-response missions in which there is insufficient time to compile a master executive program (MEP) before the mission. On these missions, the crew will replace the MEP by calling up specific computer programs and routines through the computer entry keyboard, as is done on Apollo spacecraft. The capability will also exist, of course, for manually entering data into the computer via the keyboard.

3.4.2 Crew Controls and Displays

Manual Controls

The manual controls required in the CM are:

1. Two or three rotation (attitude) controllers

2. Two translation (ACS delta-V) controllers, semi-automatic telescope pointing Horizon reticle rotation controls and "mark" button

3. One main engine throttle control

One rotation controller will be portable for spacecraft control during navigation sightings or docking. The rotation controllers not only control spacecraft attitude through the ACS jets, but also control the main engine gimbals after manual takeovers of thrust maneuvers and during hovering translation maneuvers before lunar landings. Two or three rotation controllers provide FO/FS or FO/FO/FS tolerance during lunar landings, which is the critical mission phase for manual control.

The translation controllers will include a T-handle, as on Apollo, to initiate manual takeovers of main engine thrusting maneuvers in space, and perhaps initiate lunar landing aborts. Two translation controllers are required for failure tolerance during occasional cargo transfer operations, etc., in which docking must be manually controlled.

Automatic engine throttling to control spacecraft velocity and altitude will be the normal mode of operation for the terminal phases of lunar landings, but a manual throttle is also provided to enable manual control of altitude during hovering translation maneuvers to clear crater rims or other surface obstacles that the landing radar altimeter would not sense soon enough.

The semi-automatic telescope is used not only for navigation sightings from orbit, but for landing-site tracking during lunar landings. During this phase, the landing television camera(s) will be slaved to the telescope line of sight. In the event ot telescope failure, landing site tracking can be accomplished with the TV camera(s). This provides FO/FS or FO/FO/FS tolerance for landing site tracking.

GN&C System Configuration Controls

The GN&C configuration will be controlled by appropriate switches on two panels: an ACS mode select panel and a GN&C power control panel. The ACS mode control switches and spacecraft manual controls will be wired into a manual control electronics unit also mounted in the CM.

Computer Entry Keyboards

Two keyboards will be provided for entering or requesting data from the computer, calling up particular programs, etc. These units will be adjacent to the all-purpose displays described below and are included in the communications and data management system (C&DMS) weight, volume, and power tabulations.

GN&C Displays

The primary GN&C displays will be provided by a light emitting diode (LED) matrix alphanumeric display and a three-color cathode ray tube (CRT) display. Both of these displays are considered part of the C&DMS.

The CRT display will be mounted at the pilot's station and will provide:

1. Alphanumeric data from the computer

2. Graphic displays (computer generated) of spacecraft attitude and rates, velocities, range, altitude and other flight variables

3. Landing TV and docking TV pictures

The alphanumeric displays replace the display portion of the Apollo display-keyboard (DSKY). The most frequent graphic display will be an attitude display replacing the Apollo flight director attitude indicator (FDAI). During the landing and docking operations, flight variables (velocities, altitude, range) will be displayed in addition to the television picture. After the TV picture is frozen before lunar landings, the spacecraft horizontal position will be indicated on the static TV picture.

The alphanumeric display will be mounted at the second crewman's (navigator's) station. The alphanumeric and CRT displays will be mounted adjacent to keyboards so that the desired displays can be requested.

An FDAI was also considered for inclusion, but was rejected because of the weight and power penalty, lack of all-attitude capability, and because it could be replaced by an equivalent CRT display.

During lunar landings, if lateral hovering maneuvers were required to reach a safe touchdown site, failure of the CRT display would mean loss of both lateral velocity and position indications and might force a landing abort. If pilot or navigator visual inspection indicated no surface obstacles to be avoided, the landing could proceed under automatic control with the CRT failed. During all other mission phases, a CRT failure is not crucial since manual control is a backup mode only.

In addition to the primary CRT and alphanumeric displays, various GN&C system caution and warning lights may be desirable. These have not yet been defined.

3.4.3 Backup Attitude/Velocity Control

A rate/attitude strapdown gyro triad and longitudinal integrating accelerometer were included in the CM equipment to provide a simple backup attitude and velocity control system. The electronics for these sensors would be part of the electronics unit for manual controls and ACS mode switching. The gyro outputs would be displayed by attitude error needles. The accelerometer output would be displayed on a digital delta-V readout that could be manually preset. Thrusting maneuvers could be accomplished almost entirely independently of IM GN&C equipment by manually aligning the spacecraft to a desired inertial orientation from telescope sightings to known stars (an awkward but feasible procedure), setting the delta-V meter, and manually controlling the spacecraft attitude during the burn. The sensor concept is, of course, similar to the Apollo stabilization and control system (SCS) and entry monitor system (EMS) delta-V meter. It is recommended that the rate gyro triad and accelerometer be mounted on a common navigation base with the telescope.

3.4.4 CM-IM Relative Equipment Location

In Section 3.3.2, placement of the horizon tracker(s) on the opposite side of the IM from the star trackers was recommended to provide maximum fields of view for the star trackers. Since the telescope is used for earth landmark tracking, it must be on the same side of the spacecraft as the horizon tracker(s). The telescope base-star tracker base alignment will be flight-calibrated by sightings on known stars. Since the fields of view of the telescope and star trackers are mutually exclusive, the alignment will be determined by telescope sightings on known stars other than those seen by the star trackers.

3.5 SELECTED GN&C EQUIPMENT

The GN&C system components selected for the space tug are briefly described here and the system weight, volume, and power requirements are tabulated. Where suitable components are not known to exist or existing equipment would require substantial modification, specific hardware has not been selected, and the weight, volume and power have been estimated. For several components, such as the IMU, the hardware finally chosen probably will be identical to that chosen for the EOS or space station, so that the present selection must be tentative. The required redundancy level for the FO/FS or FO/FO/FS tolerance is in most cases obvious or has been discussed in Section 3.3, but where appropriate, it is mentioned here.

In general, the equipment selected meets minimum requirements of the space tug and only proved sensor types have been chosen. This selection philosophy was adopted to minimize weight (a problem area) and development cost. However, because this philosophy was followed, the design objective of autonomous operation has not been achieved for several mission phases, and other guidelines (e.g., unrestricted lighting conditions for lunar landings) have not been met. Satisfying these design objectives or guidelines would require, in most instances, development of entirely new sensors. In some cases, the sensor feasibility is questionable. A further comment about the selection approach is that advanced sensors now under development, such as the Autonetics Micron IMU, would provide significant reductions in weight and power if their expectations are borne out.

Since GN&C equipment is split among the various tug modules and varies somewhat with the vehicle configuration and type of mission, the equipment complements have been tabulated separately by module (IM, CM, and PM) and by function (basic equipment, docking, and lunar landings). Total GN&C weight volume and power is then tabulated for various types of missions.

Upon request, Honeywell Aerospace Division submitted recommendations for the tug GN&C description based on subsystem requirements generated early in the study. The HI data took advantage of much more detailed studies performed for the EOS program and provided NR with valuable insight. Appendix C contains the complete HI final report.

3.5.1 Intelligence Module Basic GN&C Equipment

The basic equipment complement is common to all missions and does not include add-on's such as the rendezvous and docking sensors. The IM basic G&N equipment consists of an IMU with associated preprocessor, gimbaled star trackers for attitude reference, sun sensors for attitude acquisition and sun pointing attitude hold, horizon tracker(s) for navigation measurements up to geosynchronous altitude, and a navigation sensor base for the star trackers and IMU. For adequate fields of view for the star trackers while making horizon tracker measurements in low earth orbit, the horizon tracker(s) is mounted on the opposite side of the IM from the navigation base and sensors. If further study shows that horizon-tracker-to-navigation-base alignment uncertainties resulting from structure deformations cause unacceptable navigation errors, then an additional star tracker mounted on a common base with the horizon tracker(s) will be required to determine the alignment in orbit.

The IM will also contain the electronics for controlling the ACS reaction jets (ACS driver amplifiers) and the main engine gimbals and throttling (engine gimbal amplifiers). In addition to the equipment just listed, the GN&C system will use the computers, central timing unit, data bus, and signal conditioning units, which are part of the communications and data management system.

The selected IMU is the Hamilton Standard strapdown Hexad IMU and associated preprocessor proposed for the EOSS. This unit, with six nonorthogonally mounted gyros and accelerometers, provides FO/FO/FS tolerance. For FO/FS tolerance, a modified concept of this IMU with only five gyros and accelerometers was selected. The IMU preprocessor compares the redundant gyro and accelerometer measurements to detect instrument failure and includes logic to disregard failed instrument outputs.

No specific star tracker was selected, but representative weight, volume, and power was estimated for a gimbaled star tracker with ±60 degrees gimbal freedom about two axes. The estimated weight of 17 pounds (7.7 kilograms) is considered to be slightly conservative. Star trackers typically have RMS tracking accuracies of 15 arc-seconds or better, which is more than adequate for space tug requirements. The limiting factor on tracking accuracy may well be the gimbal angle readout resolution, typically 20 to 40 arc-seconds.

The acquisition sensor concept was modeled on the surveyor acquisition sun sensors. The weight, volume, and power were estimated somewhat larger than for the surveyor acquisition sensors to be conservative.

The selected horizon tracker is the Quantics Industries edge tracker model IVA selected for the NR space station and EOS designs. Each unit is composed of four tracking heads and electronics. Estimated local vertical direction accuracy is 0.05 degrees. Prototypes have been built, but have not yet been flown in space. The horizon tracker electronics are dual redundant, and the unit can operate with only three tracking heads operative. Hence, only one tracker is needed for FO/FS tolerance, and two for

FO/FO/FS capability. The range of operation is from 80 to 25,000 nautical miles (148 to 46,400 kilometers) altitude. The navigation base (for the IMU and star trackers) weight and volume are estimates, with Apollo data used as a basepoint.

Weight, volume and power estimates for the ACS driver amplifiers and main engine gimbal amplifiers were supplied by Honeywell, based on proposed EOS equipment. One engine-gimbal amplifier assembly with internal redundancy is required for FO/FS tolerance. Two assemblies with less internal redundancy provide FO/FO/FS capability. In some cases, the fail safe mode of operation may be with an ACS reaction jet or one main engine inoperative.

Most GN&C signals, commands, and data within the IM will be carried on the data bus as part of the C&DM system. Some signals and commands to the reaction jets and main engines will be carried on separate wires. An allowance of 15 percent of the IM component weight has been included in the IM GN&C basic equipment weight estimate for wiring.

The weight volume and power estimates for the IM GN&C basic equipment is tabulated in Table 3-3.

3.5.2 Crew Module Basic GN&C Equipment

The crew module basic GN&C equipment consists of rotation and translation controllers to manually control the spacecraft with the ACS reaction jets; a triad of attitude/rate gyros and a longitudinal integrating accelerometer mounted on the telescope base to provide a backup attitude control and manual thrust maneuver control system; a semi-automatic telescope with pointing controls and base for navigation sightings and lunar landing site tracking; manual control electronics; an ACS control mode selection panel; and a GN&C power control panel. The rotation controllers are also used to manually control the main engine gimbals, through the manual control electronics unit, during manual thrust maneuvers and hovering translation maneuvers before lunar landings. The functions of the various items of equipment are more fully described in Sections 3.3 and 3.4.

In addition to the components listed, the crew will use for GN&C operations the alphanumeric and multi-format graphics displays and the two keyboards for requesting displays or entering data into the computer. The displays and keyboards are part of the C&DM system.

A firm requirement for the accelerometer and gyro triad does not exist since these signals are generated by the primary system. The elements were added to the system to minimize the functional involvement of manual control with the primary system.

Table 3-3. Guidance, Navigation and Control Subsystem Basic Equipment

Description	Source	Unit Characteristics: Weight lb	Weight (kg)	Volume ft^3	Volume (m^3)	Pwr (w)	Code	Total Characteristics: Qty	Weight lb	Weight (kg)	Volume pt^3	Volume (m^3)	Pwr (w)
Intelligence Module Basic Equipment													
Inertial measuring unit and processor	HS pentad	61	28	1.25	0.035	178	I	1	61	28	1.25	0.035	178
Gimbaled star tracker	Estimate	17	8	0.20	0.006	10	CP	2	34	15	0.40	0.011	10
Acquisition sun sensors and electronics	HAC	1.5	1	0.01	0.000	1	CP	4	6	3	0.04	0.001	4
Horizon/earth tracker (edge tracker)	QI	45	20	0.44	0.012	38	CP	1	45	20	0.44	0.012	38
Navigation sensor base	Apollo	17	8	0.80	0.023	0	T	1	17	8	0.80	0.023	0
ACS driver amplifiers	HI	20	9	0.31	0.009	16	T	2	40	18	0.62	0.018	16
Main engine gimbal amplifiers (4 engines)	HI	--		--		--	O	1	25	11	0.36	0.010	58
Signal distribution wiring (0.15 x weight)	Estimate	--		--		--	--		36	16	0.07	0.002	0
Total IM GN&C basic equipment	--	--		--		--	--		264	119	3.98	0.112	304
Crew Module Basic Equipment													
Rotation control	Apollo	12.5	6	0.12	0.003	12	L, D	2	25	11	0.24	0.007	24
Translation control	Apollo	9	4	0.10	0.003	20	L, D	2	18	8	0.20	0.006	40
Rate gyro triad (mounted on telescope base)	Estimate	9	4	0.10	0.003	15	L, D	1	9	4	0.10	0.003	15
Integrating accelerometer (mounted on telescope base)	Estimate	2	1	0.05	0.001	10	L, D	1	2	1	0.05	0.001	10
Semi-automatic telescope, controls, and base	EOS	38	17	1.28	0.036	15	T	1	38.	17	1.28	0.036	15
Manual control electronics	Apollo	16	7	0.40	0.011	50	T	1	16	7	0.40	0.011	50
ACS mode select panel	EOS est	10	5	0.30	0.008	0	T	1	10	5	0.30	0.008	0
GN&C power control panel	EOS est	10	5	0.30	0.008	0	T	1	10	5	0.30	0.008	0
Signal distribution wiring (0.3 x weight)	Estimate	--		--		--	--	-	36	16	0.07	0.002	0
Total CM GN&C basic equipment	--	--		--		--	--	-	164	74	2.94	0.082	154

In general, crew intervention in tug GN&C functions is a backup mode of operation. Hence, no redundancy is required for most of the crew module GN&C equipment. The major exception to this is manned lunar landings, where manual control of hovering translation maneuvers before touchdown is the primary mode. Hence, two rotation controllers are required to allow for failure of one controller before a lunar landing.

The weight, volume, and power estimates for the rotation and translation controllers and the manual control electronics are Apollo values. The weight, volume, and power assumed for the attitude/rate gyro triad and the integrating accelerometer are estimates based on contemporary instruments. The semi-automatic telescope weight, volume, and power were estimated based on Apollo data and EOS estimates (for a telescope-sextant combination). The ACS control mode selection panel and GN&C power control panel estimated weights are EOS estimates. Thirty percent of the CM GN&C equipment weight was allowed for signal wiring in the CM.

The CM basic GN&C equipment weight, volume, and power are tabulated in Table 3-3.

3.5.3 Rendezvous and Docking GN&C Equipment

Sensors required for rendezvous and docking are a rendezvous and docking radar mounted on the docking face for target acquisition, tracking, and automatic docking; a docking television camera, also mounted on the docking face, for crew or ground monitoring of automatic docking and manual completion of docking if necessary; and contact sensors to indicate when latching of the docking mechanism has been completed.

Equipment required on target vehicles consists of corner reflectors or a laser radar transponder mounted on the docking face, and visual docking aids, consisting of a distinctive pattern, standoff cross, etc. on the docking face to provide visual cues for manual docking. Noncooperative rendezvous can be accomplished only by adding a skin-tracking rendezvous radar to the tug as part of the payload.

The selected rendezvous and docking radar is the ITT scanning laser radar system. This unit is the only known system that can provide accurate enough measurements for automatic docking. It has sufficient target detection range for rendezvous most of the time, and it has also been selected for the EOS and space station. The system is currently in an advanced state of development.

The principal system characteristics are given in Table 3-4. In the rendezvous and docking discussion of Section 3.3, it was pointed out that the laser radar has marginal detection range and field of view for geosynchronous

Table 3-4. Scanning Laser Radar System Characteristics

Maximum range	65 nautical miles (120 kilometers)
Minimum range	0
Range accuracy	±0.02% or ±0.33 feet (10 cm), whichever is greater
Range rate	1-16, 400 fps (0-5 kmps)
Range rate accuracy	±1.0% or ±0.16 fps (±0.5 cmps) whichever is greater
Angular field of view without gimbals (by piezoelectric beam steering)	30 x 30 degrees
Angle accuracy	±0.02 degree
Angular rate	
Acquisition mode	0-0.4 degrees/sec
Track mode	0-10 degrees/sec
Angular rate accuracy	±1.0% or ±0.01 degree/sec, whichever is greater
Acquisition scan time	1 to 150 seconds
Weight	35 pounds (16 Kg)
Volume	0.86 ft^3 (0.024 m^3)
Power	30 watts

orbit rendezvous. Several methods of alleviating the problem were considered, including use of a star tracker to acquire a flashing-light beacon on the target.* Tracking of the rendezvous target with a star tracker, in conjunction with radio ranging, could also be used as a backup rendezvous technique if all laser radars failed. If further investigation verifies that target acquisition and tracking by a star tracker is feasible, then the required redundancy level of the laser radars could be reduced by one. It would also require, of course, that targets be equipped with a flashing light beacon. For the present, though, it is assumed that two laser radars are required for FO/FS and three are required for FO/FO/FS tolerance. Rendezvous and docking radars are required on the IM and CM docking faces, but not on the PM docking face since it will dock only to an OPD, which itself has a laser radar.

The docking television camera weight, volume, and power estimates were based on the Lear Siegler, Inc. (LSI) missileborne television system. This vidicon unit has 700-line resolution and 10 shades of grayness. The field of view may be chosen at will by proper selection of camera lens, which is mounted with a standard 16-mm "C" mount.

Since manual docking is a backup (FS) mode of operation, only one camera is required on each docking face. The CM docking face does not require a TV camera since it is assumed that the pilot will have direct vision through a window in the docking face. One translation controller in the CM should be portable so that it can be attached to a mount by the docking window.

The contact sensors weight was estimated at 2 pounds (0.907 kilograms) per sensor, with four sensors required for each docking port. Rendezvous and docking sensor weight, volume, and power estimates for the CM, IM, and PM are shown in Table 3-5.

3.5.4 Lunar Landing GN&C Equipment

Additional G&C equipment required in the CM for manned lunar landings are an engine throttle control for complete manual takeover of the spacecraft (the normal mode is manual control of lateral motion only, with automatic altitude and descent rate control) and a landing television picture memory for displaying the frozen TV picture after the landing site is obscured by dust. The weight, volume, and power of these components were estimated, and a 30-percent weight factor was allowed for signal wiring.

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*This method has been adopted for EOS (shuttle) rendezvous.

Table 3-5. Guidance, Navigation and Control Subsystem Rendezvous and Docking Equipment

Description	Source	Unit Characteristics						Qty	Total Characteristics				
		Weight		Volume		Pwr (w)	Code		Weight		Volume		Pwr (w)
		lb	(kg)	ft^3	(m^3)				lb	(kg)	ft^3	(m^3)	
Crew Module Docking Equipment													
Laser rendezvous and docking radar	ITT	35	15.9	0.86	0.0243	30	D	2	70	31.8	1.72	0.0487	30
Contact sensor	Est	2	0.9	0		0		4	8	3.6	0		0
Signal distribution wiring (0.2 x weight)	Est	--	--	--	--	--	-	-	15	6.8	0.03	0.0008	0
Total CM docking GN&C equipment	--	--	--	--	--	--	-	-	93	42.2	1.75	0.0495	30
Intelligence Module Docking Equipment													
Laser rendezvous and docking radar	ITT	35	15.9	0.86	0.0243	30	D	2	70	31.8	1.72	0.0487	30
Docking TV camera	LSI	10	4.5	0.20	0.0056	30	D	1	10	4.5	0.20	0.0056	30
Contact sensor	Est	2	0.9	0	--	0	-	4	8	3.6	0	--	0
Signal distribution wiring (0.2 x weight)	Est	--	--	--	--	--	-	-	17	7.7	0.03	0.0008	0
Total IM docking GN&C equipment	--	--	--	--	--	--	-	-	105	47.6	1.95	0.0551	60
Propulsion Module Docking Equipment													
Docking TV camera	LSI	10	4.5	0.20	0.0056	20	D	1	10	4.5	0.20	0.0056	30
Contact sensor	Est	2	0.9	0	--	0	-	4	8	3.6	0	--	0
Signal distribution wiring (0.2 x weight)	Est	--	--	--	--	--	-	-	8	3.6	0	--	0
Total PM docking GN&C equipment	--	--	--	--	--	--	-	-	26	11.7	0.20	0.0056	30

Lunar landing sensors consist of a landing radar to measure altitude and three velocity components and a gimbaled television camera for landing site acquisition and tracking and generation of the static picture of the landing site for surface obstacle avoidance. These sensors will be mounted at the lower end of the spacecraft or on the landing gear.

The Apollo LM landing radar (or possibly a modification thereof) was selected for the landing radar, and Apollo values were used for the weight, volume, and power estimates. Two or three landing radars are required for FO/FS or FO/FO/FS tolerance.

The landing television camera (and associated pointing system) weight, volume, and power were estimated without specific units in mind. Two or three landing TV systems are required for FO/FS or FO/FO/FS capability on unmanned landings. On manned landings, one less TV system is needed since the normal mode of operation is to track the landing site through the telescope, and pilot direct vision of the touchdown area can be substituted for the frozen picture display. A 15-percent weight factor was allowed for landing sensor signal wiring.

The lunar landing equipment weight, volume, and operating power estimates for the CM and for manned and unmanned lunar landing kits are given in Table 3-6.

3.5.5 Total GN&C Subsystem

The total space tug GN&C subsystem weight, volume, and operating power for unmanned and manned space missions and unmanned and manned lunar landing missions are shown in Table 3-7. A schematic of the system is shown in Figure 3-4.

3.6 SPACE TUG GUIDELINE IMPLICATIONS

Several of the guidelines and design objectives given in the space tug RFP introduce significant feasibility questions or development cost or substantial weight penalties for the GN&C subsystem. The equipment redundancy required for FO/FS or FO/FO/FS tolerance of course introduces a weight penalty on all subsystems and will not be considered here as it is discussed elsewhere. The guidelines and design objectives affecting primarily the G&N system are discussed briefly in this section.

3.6.1 Autonomous Earth Orbit Navigation

All elements of the NASA IPP (space station, EOS, space tug, RNS), as well as proposed Air Force spacecraft, have a requirement or design objective of autonomous operation. Fully autonomous earth orbit navigation

Table 3-6. Guidance, Navigation and Control Subsystem Lunar Landing Equipment

Description	Source	Unit Characteristics: Weight lb	Unit Weight (kg)	Unit Volume ft^3	Unit Volume (m^3)	Unit Pwr (w)	Code	Total Characteristics: Qty	Total Weight lb	Total Weight (kg)	Total Volume ft^3	Total Volume (m^3)	Total Pwr (w)
Crew Module Lunar Landing Equipment													
Main propulsion throttle control	Est	6	2.72	0.06	0.0017	6	L	2	12	5.44	0.12	0.0034	12
Landing television graphics memory	Est	10	4.54	0.10	0.0028	50	L	1	10	4.54	0.10	0.0028	50
Signal distribution wiring (0.3 x weight)	Est	--	--	--	--	--	-	-	5	2.27	0.01	0.0003	0
Total CM lunar landing GN&C equipment	--	--	--	--	--	--	-	-	27	12.25	0.23	0.0065	62
Lunar Landing Kit, Unmanned													
Landing radar	Apollo LM	43	19.5	2.00	0.0566	135	L	2	86	39.0	4.00	0.1133	135
Gimballed television camera	Est	30	13.6	0.40	0.0113	60	L	2	60	27.2	0.80	0.0226	120
Signal distribution wiring (0.15 x weight)	Est	--	--	--	--	--	-	-	22	10.0	0.04	0.0011	0
Total unmanned lunar landing kit GN&C equipment	--	--	--	--	--	--	-	-	168	76.2	4.84	0.1370	255
Lunar Landing Kit, Manned													
Landing radar	Apollo LM	43	19.5	2.00	0.0566	135	L	2	86	39.0	4.00	0.1133	135
Gimballed television camera	Est	30	13.6	0.40	0.0113	60	L	1	30	13.6	0.40	0.0113	60
Signal distribution wiring (0.15 x weight)	Est	--	--	--	--	--	-	-	17	7.7	0.03	0.0008	0
Total manned lunar landing kit GN&C equipment	--	--	--	--	--	--	-	-	133	60.3	4.43	0.1254	195

Table 3-7. Guidance, Navigation and Control Subsystem Totals

System Description	Weight		Volume		Power
	lb	(kg)	ft^3	(m^3)	w
Unmanned space missions					
Intelligence module basic equipment	264	120	3.98	0.1127	304
Intelligence module docking equipment	105	476	1.95	0.0552	60
Propulsion module docking equipment	21	10	.20	0.0057	30
Total unmanned space mission GN&C equipment	390	606	6.13	0.1736	394
Manned space missions					
Crew module basic equipment	164	74	2.94	0.0832	154
Crew module docking equipment	93	42	1.75	0.0495	30
Intelligence module basic equipment	264	119	3.98	0.1127	304
Propulsion module docking equipment	21	10	0.20	0.0057	30
Total manned space mission GN&C equipment	542	245	8.87	0.2511	518
Unmanned lunar landing missions					
Unmanned space mission equipment	390	177	6.13	0.1735	394
Unmanned lunar landing kit	168	76	4.84	0.1371	255
Total unmanned lunar landing mission GN&C equipment	558	253	10.98	0.3106	649
Manned lunar landing missions					
Manned space mission equipment	542	246	8.87	0.2512	518
Crew module lunar landing equipment	27	12	0.23	0.0065	62
Manned lunar landing kit	133	60	4.43	0.1254	195
Total manned lunar landing mission GN&C equipment	702	318	13.53	0.3831	775

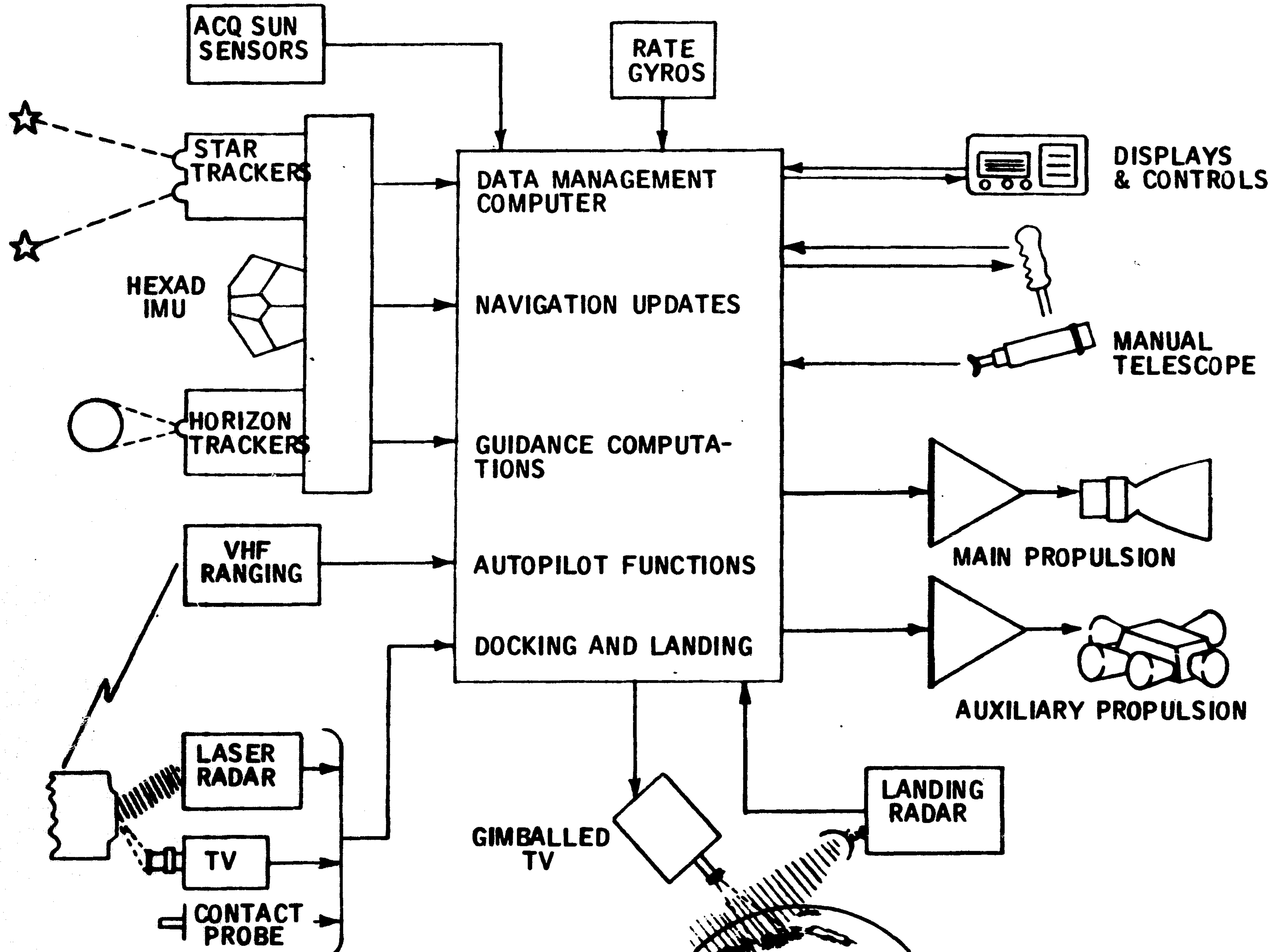


Figure 3-4. Guidance, Navigation and Control Subsystem Schematic

methods, using only on-board sensors, either provide marginal or inadequate navigation accuracy for some missions or involve unproved sensing techniques (e.g., automatic known or unknown landmark tracking). The cost to develop and test new techniques would be substantial, and the techniques might be shown to be unworkable.

The alternative approach, for earth orbit navigation, is to deploy a ground beacon or navigation satellite system to provide semi-autonomous navigation capability. The ground beacon system recently selected for the EOS appears to be a good choice, but all IPP vehicles and missions should be considered in selecting the system. To summarize, external aids for earth orbit navigation appear to be highly desirable or necessary. The selection and deployment of an orbital navigation aid system is a national space program decision.

3.6.2 Autonomous Navigation for Earth Escape, Cislunar Space and Lunar Orbit

Autonomous cislunar navigation of unmanned tugs would require the addition of an earth tracker for earth sensing above synchronous altitudes (or modification of the horizon tracker) and a lunar limb tracker. These sensors do not exist to our knowledge. Thus, both a weight penalty and development cost would be incurred in providing autonomous cislunar navigation capability.

Autonomous unmanned lunar orbit navigation could be achieved with a network of lunar surface radio beacons or by automatic lunar terrain image correlation. Either approach would be costly.

Lunar or planetary missions require determination of the trajectory after escape orbit insertion. On-board horizon trackers, earth sensors, and lunar limb sensors could provide adequate navigation for lunar missions with some delta-V penalty, but probably not for planetary missions. Radio beacons or navigation satellites for earth orbit navigation have too short a detection range or, in the case of synchronous satellites, have antennas pointed at the earth. (A steerable high-gain antenna on a synchronous satellite would be a possibility.) None of the potential autonomous or semi-autonomous methods would be as accurate as MSFN or DSN ground tracking.

The potential development costs, weight, and delta-V penalties, and accuracy degradation of autonomous or semi-autonomous navigation techniques for lunar missions and transplanetary injection should be weighed against the cost of maintaining some MSFN or DSN stations before autonomous or semi-autonomous navigation methods are adopted.

3.6.3 Unrestricted Lighting for Lunar Landings

Unrestricted lighting conditions for landings at new sites would require development of a lunar terrain contour matching radar and an obstacle-detection radar. These probably could be combined into one radar, but a new development would be required. Follow-on landings for resupply or rescue under almost all lighting conditions can be achieved by tracking a visible light beacon at the landing site with a star tracker. Radar tracking of a transponder at the landing site would permit follow-on landings under all lighting conditions, but would require addition of a tracking radar to the tug equipment.

A terrain contour/obstacle detection radar has not been specified for the tug because of the weight penalty and because a requirement to land at a new site without visual inspection of the site before landing is considered unrealistic.

3.6.4 Mission Operational Limitations

Although not specifically in conflict with any RFP guidelines or design objectives, three operational limitations are worth mentioning. The first two deal with rendezvous.

Optical rendezvous sensors (laser radar or star tracker tracking a light beacon on the target) will require avoiding rendezvous or docking approach directions within about 20 degrees of the sun. The second restriction is that only cooperative rendezvous capability is included in the basic tug capability. Non-cooperative rendezvous will require addition of a skin-tracking radar, weighing about 100 pounds (45 kilograms), to the payload.

The third limitation is the requirement to assemble an MEP for unmanned missions, as discussed under computer requirements in Section 3.3. Although a standard MEP for many missions may be feasible, with only specific targeting data read into the computer for individual missions, the variety of potential constraints and objectives for tug missions may well dictate the assembly of special MEP's for some missions. Thus, tug missions will still require pre-mission planning.

3.6.5 Summary

The weight and delta-V penalties, feasibility questions, and sensor development costs to obtain autonomous navigation and unrestricted lighting lunar landing capabilities have been discussed. These should be carefully weighed against the cost of performing ground tracking of the spacecraft during some mission phases and against the operational limitation of restricting lunar landing lighting conditions. The feasibility of autonomous lunar missions and lunar landings at new sites without visual inspection of the landing site should be evaluated.

3.7 FUTURE STUDIES

The most significant studies for refining the space tug GN&C subsystem concept revolve around weight reduction, autonomous navigation accuracy, and additional sensor studies. These considerations have been mentioned in various places in the previous sections and are summarized in the following paragraphs.

3.7.1 Weight Reduction

GN&C system weight can be reduced primarily by substitution of lighter weight sensors and by reducing the required equipment redundancy. A primary candidate for sensor weight reduction is the IMU, where substitution of the Micron IMU, if it is proved on tests, would save 60 to 65 pounds (27 to 29 kilograms). Equipment redundancy can be reduced by accepting FO/FS tolerance. Secondary methods for performing G&N functions can be investigated.

3.7.2 Autonomous Navigation

Studies of navigation accuracy with autonomous or semi-autonomous methods are needed to evaluate these methods, especially for LEO to geosynchronous transfers and in cislunar space. The computer requirements for autonomous cislunar guidance and navigation should also be estimated in order that the feasibility of autonomous cislunar space operations can be evaluated.

3.7.3 Additional Sensor Studies

The feasibility and potential detection range of visible light beacon tracking by a star tracker during rendezvous should be evaluated. If the method is feasible, this might allow removal of one laser radar from each docking face.

Radar sensors for lunar terrain contour matching and surface obstacle detection should be more closely investigated. This is contingent, though, on whether or not a new-site landing without visual inspection of the landing site is considered to be a realistic mission requirement. If the frozen television picture concept for landing-obstacle detection is adopted, the errors in the estimated spacecraft position relative to the frozen picture should be evaluated.

4.0 COMMUNICATIONS AND DATA MANAGEMENT SUBSYSTEM

4.1 REQUIREMENTS

The communications and data management subsystem provides the capability for acquisition, processing, storage, and both internal and external exchange of information related to tug checkout, monitoring, and operations. The subsystem supports all mission concepts without reconfiguration within individual modules, with the exception of software. FO-FO-FS is a design goal, with FO-FS a minimum requirement.

4.2 COMMUNICATIONS

Communications parameters establishing requirements for transmitter power, antennas, receivers, and operational limitations include the information rates and quality (required signal-to-noise ratio, or error rates) and the geometry of the links. Basic requirements for the tug links are shown in Table 4-1.

Consideration of the broad variety of potential tug missions has led to establishment of four classes of information. A low data rate capability includes commands, tracking information, gross status measurements, and voice. This link would be capable of data rates up to ≈4000 bits per second.

Moderate rates would be required for remote monitoring of vehicle subsystem response to commands or status during critical periods, checkout routines, and dumps of stored data. Rates are assumed up to 50,000 bits per second.

Television capability equivalent to that on Apollo provides a general video monitoring capability for overall situation assessment. The link capacity also could be used for high speed data dumps or for simultaneous transmission of several channels of data over the link.

The high resolution link requirement is primarily for visual support to a remote operator. This applies when the tug is remotely controlled and when the manned tug is controlling another vehicle. Again, the link capacity is available for multiplex transmissions or backup to a lower capacity mode.

Table 4-1 indicates the probable requirements for these classes of links for tugs in low or geosynchronous earth orbit, in lunar orbit, or on

Table 4-1. Tug Communications Requirements

From \ To	Low Earth Orbit Tug	Other Low Earth Orbit Elements	MSFN	TDRS	Geosync Tug	Lunar Orbit Tug	Lunar Surface Tug	Lunar DRS
Low Earth Orbit Tug	AB-D	ABCD	ABCD	AB--	----	----	----	----
Other Low Earth Orbit Elements	AB-D	N/A	N/A	N/A	A---	AB--	AB--	N/A
MSFN	ABCD	N/A	N/A	N/A	A---	AB--	AB--	N/A
TDRS	AB--	N/A	N/A	N/A	----	----	----	N/A
Geosync Tug	A---	AB--	ABCD	----	AB-D	----	----	N/A
Lunar Orbit Tug	----	AB--	ABC-	----	----	AB-D	AB-D	A---
Lunar Surface Tug	----	ABC-	ABCD	----	----	AB-D	N/A	AB--
Lunar DRS	----	N/A	N/A	N/A	----	A---	AB--	N/A

A Low data rates - commands, tracking, status, voice - up to ≈4000 bits per second
B Moderate data rates - SS response, checkout, data dump - ≈50,000 bits per second
C Apollo type TV - general video information ≈500,000 hertz baseband... Power equivalent 3,000,000 bits per second data
D High resolution (broadcast) Low frame rate TV - visual monitor for control... Power equivalent to ≈10,000,000 bits per second data

the lunar surface. Future studies, with detailed analysis of the missions and with firm specifications for the performance of interfacing elements, could alter the specific rates, but the changes should not be significant in terms of the equipment requirements. Exceptions would be in cases where a requirement develops for a higher capacity on specific links, or if commercial quality color TV were to be required at the maximum ranges. No such requirements are foreseen now.

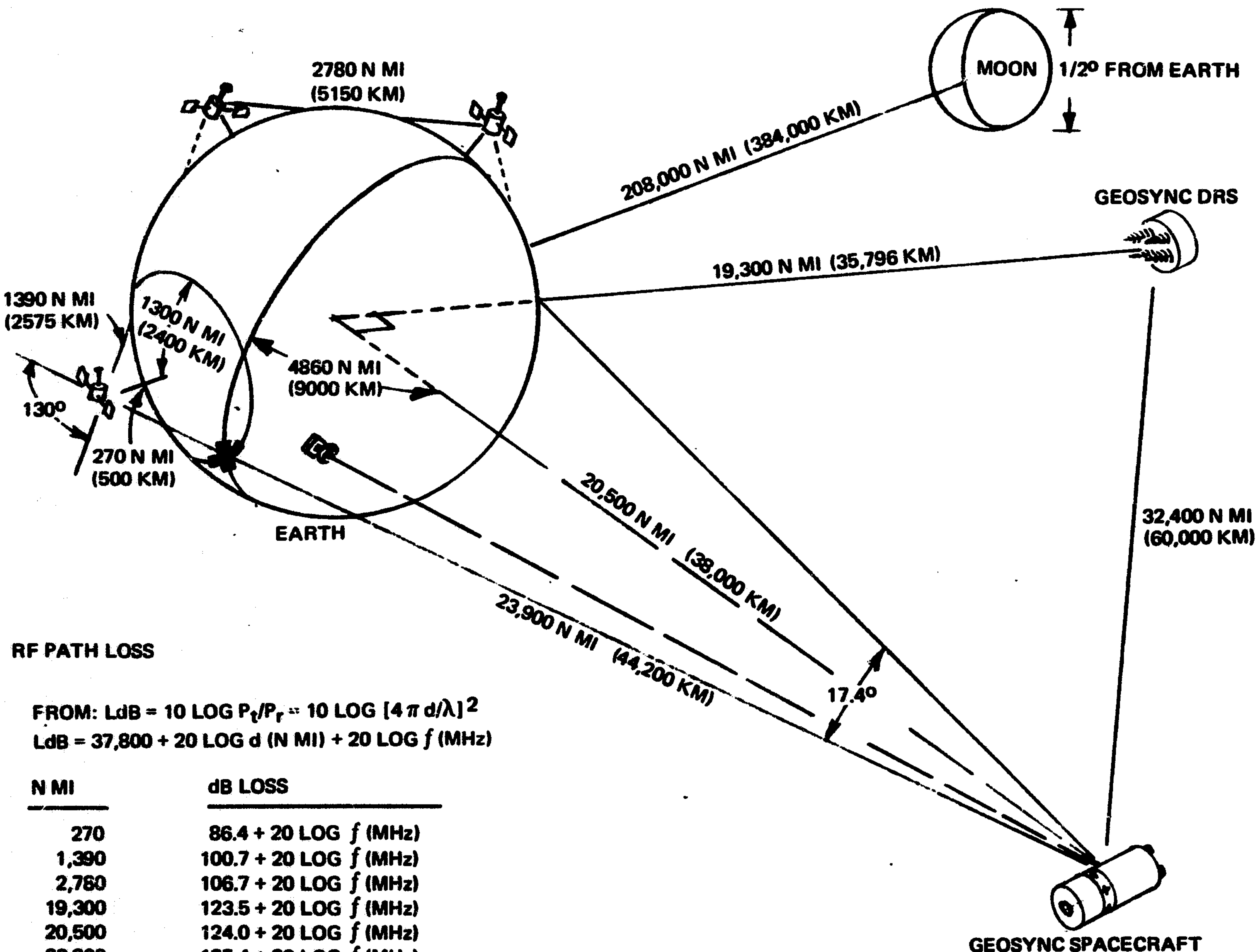
Figure 4-1 indicates the geometry involved between elements operating in the vicinity of earth. Ranges are of interest in determining the RF path losses, as indicated on the chart. Angles are of importance in that antenna coverage, gain, and pointing requirements are related. Line of sight restrictions may also be seen. Figure 4-2 indicates similar parameters for operations in the vicinity of the moon. The potential lunar satellite parameters shown are considered likely candidates for such a system. The closest libration point satellite would be at something like ten times the altitude, and would require somewhat more detailed studies to establish reasonable operating parameters.

4.3 DATA MANAGEMENT

Data management functional requirements are outlined in Table 4-2. The major requirements which affect the data management are the rates, information storage requirements, and subsystem interfaces. The requirements are based on a general purpose computer, multiprocessor data management approach. The weight and power advantage for aerospace vehicles requiring flexibility and long operational life is great, and studies such as that for the EOS and EOSS show feasibility. In addition, a similar approach offers a potential for commonality in design, operations, and in actual hardware.

Table 4-3 presents an estimate of the tug operational memory and mass memory requirements. The operational memory includes the rapid access (nanoseconds) programs required for essentially real time control of the subsystems, including data management. Mass memory includes programs which are accessible in milliseconds. It is supplemental to the operational memory and stores long term operational programs, alternatives, data for long term analysis, and any special mission information.

Word length (40 bits) is for sizing only. Figure 4-3 shows the EOSS sizing with tug requirements indicated by arrows. It should be noted that the original effort in this study separated the G&C computer and the data management computer, with an interface between data busses. Similarity of data handling resulted in processors and memories integrated into the data management subsystem, with the exception of a few dedicated



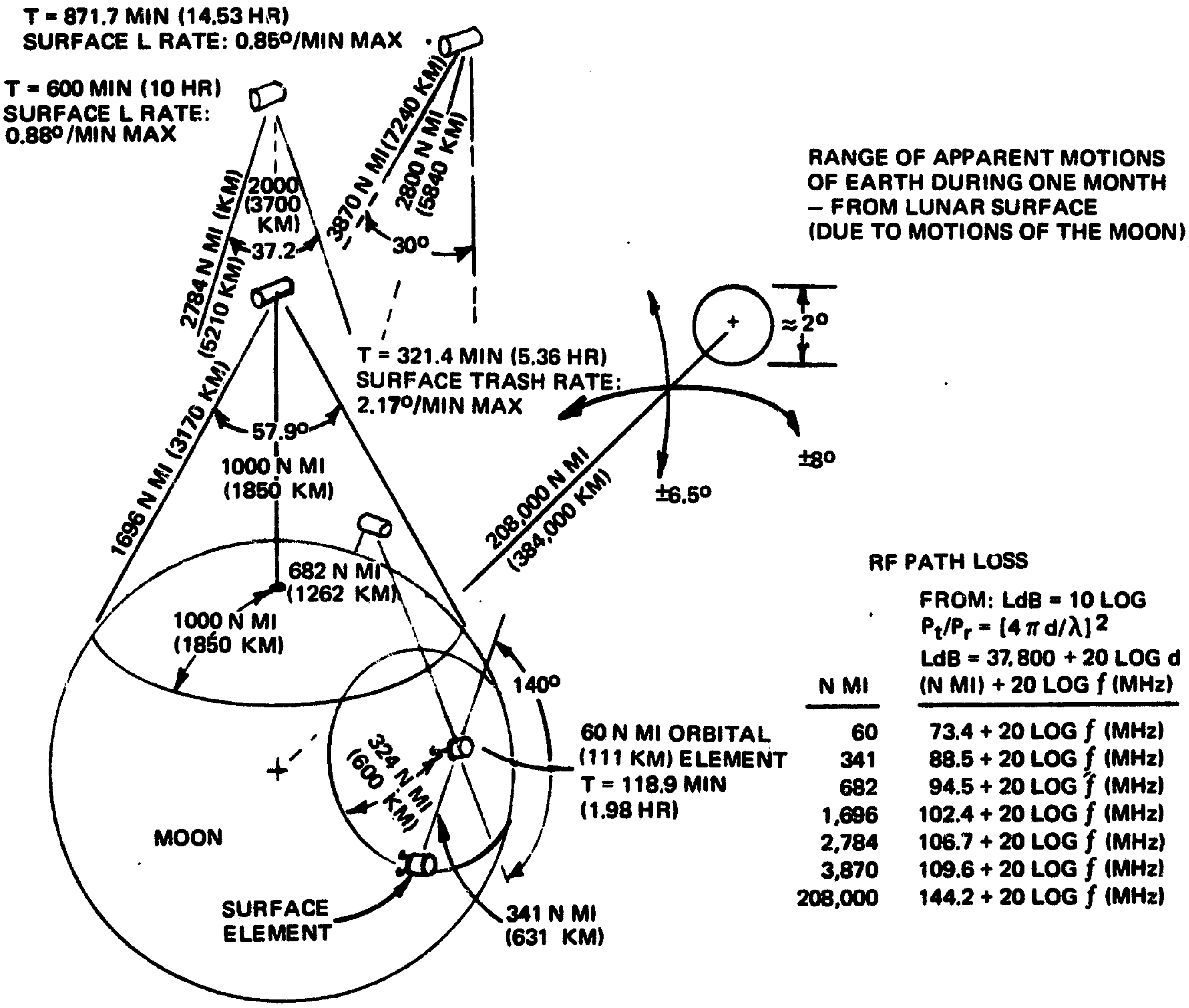
RF PATH LOSS

FROM: $L_{dB} = 10 \text{ LOG } P_t/P_r = 10 \text{ LOG } [4\pi d/\lambda]^2$

$L_{dB} = 37{,}800 + 20 \text{ LOG } d \text{ (N MI)} + 20 \text{ LOG } f \text{ (MHz)}$

N MI	dB LOSS
270	86.4 + 20 LOG f (MHz)
1,390	100.7 + 20 LOG f (MHz)
2,760	106.7 + 20 LOG f (MHz)
19,300	123.5 + 20 LOG f (MHz)
20,500	124.0 + 20 LOG f (MHz)
23,900	125.4 + 20 LOG f (MHz)
32,400	128.0 + 20 LOG f (MHz)
208,000	144.2 + 20 LOG f (MHz)

Figure 4-1. Link Geometry - Earth Elements



RF PATH LOSS

FROM: $LdB = 10 \text{ LOG } P_t/P_r = [4\pi d/\lambda]^2$

$LdB = 37.800 + 20 \text{ LOG } d \text{ (N MI)} + 20 \text{ LOG } f \text{ (MHz)}$

N MI	
60	73.4 + 20 LOG f (MHz)
341	88.5 + 20 LOG f (MHz)
682	94.5 + 20 LOG f (MHz)
1,696	102.4 + 20 LOG f (MHz)
2,784	106.7 + 20 LOG f (MHz)
3,870	109.6 + 20 LOG f (MHz)
208,000	144.2 + 20 LOG f (MHz)

Figure 4-2. Link Geometry - Lunar Elements

Table 4-2. Data Management Functional Requirements

Internal information exchange	Digital, audio, video
Stored information	Command response programs Subsystem performance limits Manned operation data Manned procedural data Emergency responses
Information acquisition External	 Ranging Commands and updates Manned verif., video, voice EVA biomed. T/M
Internal	Subsystems status. G&N computer parameters TV and manned inputs
Information processing	Status: Caution and Warning, C/O, ops, maint. Program selection and display Transmission or storage formatting Trans mode selection and antenna pointing Navigational updating
Manned control and display	Remote command and control Subsystem control, C/O and override Local and external TV display Program readout and status indication Caution and warning

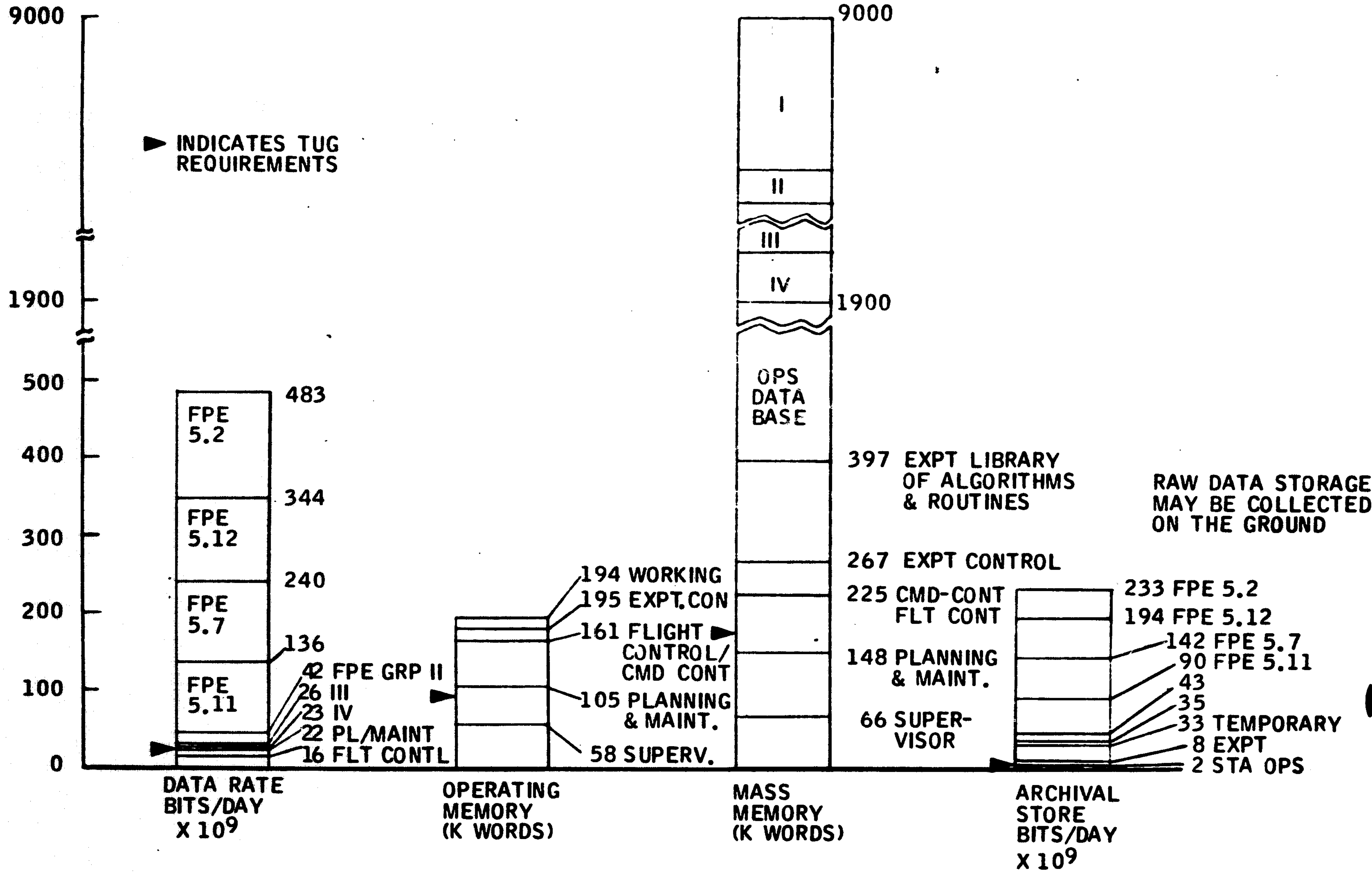


Figure 4-3. Comparison Between EOSS and Tug Computer Requirements

Table 4-3. Computer Memory Requirements

Functional Category	Operational Memory (40 bit words)	Mass Memory (40 bit words)
Supervisory (Overall subsystem control)	27,000	8,000
Flight Oper. (guidance, command and control)		
1. Mission plans	10,000	32,000
2. Orbital operations	5,000	65,000
3. Command	16,000	20,000
Onboard C/O, and monitor and alarm	7,000	17,000
Operations management	28,000	14,000
(Ant. pointing, cmd execution, SS operation, communications control)	94,000	156,000

operations required continuously by the guidance and control subsystem. These computing units are identified as pre-processor and are included in the G&C subsystem.

4.4 SELECTIONS

Configuration selections are outlined in Table 4-4 for the communications and data management subsystem. Heavy reliance is placed on EOS and EOSS trade studies. Since basic functional requirements are similar, the majority of trade parameters such as weight, power, size and complexity are valid, with major differences being in the sizing as indicated in Figure 4-3.

4.4.1 Link Frequencies

Frequency selection can be optimized in terms of power, bandwidth, coverage, and antenna requirements. However, the frequency spectrum is heavily assigned and utilized, with both international and government agreement required for operation in specific frequency ranges. The driving factor

Table 4-4. Communications and Data Management Selection

Key Areas	Criteria	Selections
Link frequencies	Allocations Compatibility	IPP elements: S-band TDRS link: S-band Local voice: VHF
Antennas	Minimum impact on structure Ant. dev. stage	Near-high rate/cislun-lo rate: Omnis Geosync to MSFN; TDRS, LDRS: 2 ft (.61m) parab Lun surface to MSFN: 6 ft (1.8m) parab (surface deployable)
Tracking	Compatibility Weight	>1000 Ft: Integ with comm <1000 Ft: G&N (video-laser)
Multiplexing	From EOSS trades - system weight, EMI, complexity	Video: FDM Voice: FDM Digital: TDM
Processing	Flexibility EOSS trades	General purpose multiprocessor I/O controller: MOS-LSI
Storage	EOSS trades Speed Weight, volume	Operating memory: Plated wire Mass memory: Plated wire Archival memory: Tape
Data acq/output	EOS/EOSS trades	Remote acquisition - local commutation/digitization

for tug frequencies will be compatibility with interfacing programs. Earth and moon programs will apparently still be largely using S-band frequencies in the vicinity of 2.3 GHz. The TDRS link may be either S or K_u (around 13.5 GHz) band. The use of K_u-band will require equipment for both bands, but will offer a distinct advantage in the antenna requirements should a need develop for TV or equivalent data rate relay through the satellite. S-band is preferred to avoid the need for an isolated system for TDRS use only.

VHF is expected to be used less in the tug time period. However, there is roughly a 100 to 1 power advantage over S-band due to lower path loss when antennas on both transmitting and receiving elements are required to be near omni-directional. The advantages of omni coverage are that transmissions can be to several receivers in different directions at the same time, and acquisition and tracking will not be required to maintain communications. These conditions would apply to voice links associated with the crew module.

4.4.2 Antennas

Omni-directional antennas will provide adequate gain for low data rate links, and for high data rates at shorter ranges. Structural limitations favor surface or near-surface mounted antennas. Apollo data indicates that four S-band omni-directional antennas, equally spaced, will be required. Locations avoiding RF masking by RCS plumes are desirable. VHF omni-antennas will be needed on the CM only. Two VHF antennas on opposite sides of the CM should provide adequate coverage.

Directional antennas will be required for longer ranges and for relative angle measurements. Basic choices are between arrays and parabolas. The arrays are attractive if surface mounting could be achieved. However, coverage could not include angles near the longitudinal axis, and some vehicle design problems exist in using a large surface area for an antenna array.

Arrays extended on a boom may compete favorably with the conventional parabolic design in the next few years. Small apparent advantages and development complexity lead to a choice of conventional parabolic designs at this time.

4.4.3 Tracking

Tracking requirements include the capability to transpond a ranging signal from the MSFN or other space elements such as the EOSS, and to perform ranging on cooperative targets as well as determine relative bearings to the target. Combining these functions with the communications links may be achieved with little penalty in power and weight, compared to an independent tracking system. There is a potential capability at distances closing to 1000 feet (305 meters). Closer approaches will be supported by the G&C subsystem with a laser radar or by a video camera assigned to G&C.

A possible requirement exists for higher accuracy tracking in the range from 1000 (305 meters) to about 20 miles (37 kilometers), depending on the detailed operational parameters of the vehicles involved. Alternatives are to provide an approach radar as a separate installation, or to use the G&C laser radar at the extended range. In the tug time period, the extension of the laser range appears to be feasible with cooperative targets. A skin track requirement (non-cooperating targets), not presently foreseen, would force addition of a separate radar. Current estimates for such a radar are 120 pounds and 400 watts for a 20-mile (37-kilometers) range, plus redundant units.

4.4.4 Multiplexing

EOS/EOSS trades have been performed on both quantity and type of multiplexing. Choices are possibly more straightforward in the tug. Video and voice are basically analog signals which lend themselves to frequency division multiplexing (FDM), unless a very large number of channels are involved. The tug will have a small number of such channels and will not approach the complexity required to consider time division multiplex (TDM). The digital signals are inherently TDM. FDM of the digital signals would only be a real consideration for data rates exceeding those currently estimated for the EOSS. The required tug rates will be considerably lower.

4.4.5 Processing

The general purpose multiprocessor approach selected in EOSS studies is also suitable for the tug. This distribution of the processing allows a large amount of backup to critical functions with minimum weight, and permits great flexibility in the missions over a long time period.

4.4.6 Storage

EOSS trade information indicates plated wire as a choice for operating and mass memories. Early studies had rejected plated wire because of the limitation of the memory stack size. Present memories of this type are limited in size due to manufacturing problems. The high precision required for magnetic plating on the memory wires currently limits production runs to wires with a maximum length of about 9 inches (22.8 centimeters). The result is that available production modules are usually about

10^5 bits in size. To achieve reliable larger storage, a large amount of supporting electronics is required to combine additional memory stacks with a total weight and power (and cost) penalty overriding the attractive features of the basic plated wire memory.

Further investigation indicated that progress is continuing in the plated wire manufacturing techniques, and production lengths are increasing. Modules with 10^7 bit capacity are expected to be available in the near future. The plated wire then becomes a choice based on overall speed, weight, size, and inherent reliability.

It should be noted that scratchpad memories within the processor probably will be solid state to achieve the high speed needed. However, the capacities are small enough so that the higher power and volume per word will not be a significant factor.

Archival memory consists of long-term records of data samples, alternate programs for mass memory, collections of unprocessed data for later transmission to the MSFN, or other information for which access is not time critical. Tape machines now are the best approach to this type of storage. A great deal of investigation by industry and various agencies in several electro-optical storage methods is being conducted. Development is at an early stage and potential utilization should be reviewed at a later date.

4.4.7 Data Acquisition/Output

EOS/EOSS studies indicate factors in thousands of pounds for trades between wire runs from subsystems to the computer against remote acquisition units with local commutation and digitization for subsystems inputs and outputs. The shorter runs in tug will decrease the factors considerably. However, weight factors in hundreds of pounds will still be considerable, and the physical signal interconnections, a traditional problem in aerospace vehicles, is made much simpler.

4.5 SUBSYSTEM CONFIGURATION

4.5.1 Functional Interfaces

A preliminary functional diagram of the tug communications and data management (C&DM) subsystem is shown in Figure 4-4. At this general functional level, a strong similarity exists with both the EOS and EOSS configurations. The duration of space operations, flexibility, lack of atmospheric operational requirements, and to some extent the stress on

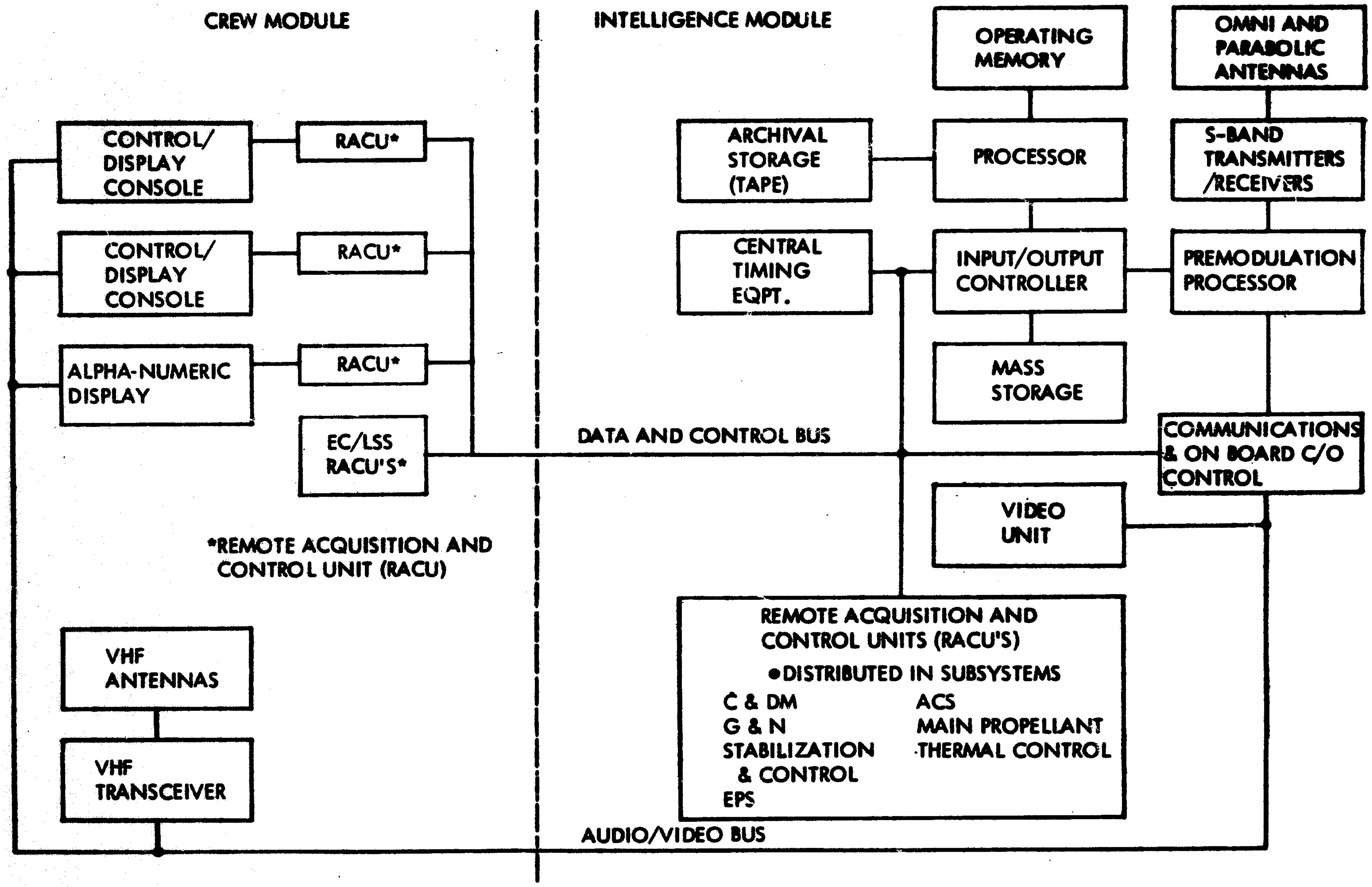


Figure 4-4. Preliminary Communications and Data Management Functional Diagram

unmanned capability has resulted in a tendency to lean toward the EOSS studies. However, the use of EOS components is not precluded by the functional arrangement.

Current estimates indicate that the major portion of the crew-related subsystem elements can be located in the CM, resulting in very little penalty on components in the IM for manned operation capability. In general, the CM interface with the IM data management elements would be similar to the interface with communications during remotely controlled operations. Primary differences would be in the duty cycle requirements and in the additional load due to the EC/LSS. The effects of adding a crew module are expected to be primarily in the software.

4.5.2 Physical Configuration

Communications Equipment

Required communications capacity has been categorized as low and moderate data rates, Apollo type TV, and high resolution low frame rate TV. To establish a value for communications equipment physical characteristics, the link gain required for these categories must be established for the links indicated in Table 4-1.

The required ratio of total signal power to noise density (C/No) in a receiver output is a function of the type of modulation, bandwidth, and the required quality of the output information. Table 4-5 shows values that are assumed for the four classes of links. Links will be calculated for the 46 db-Hz voice-low data value. Moderate data rates will require an additional 10.7 db link gain, Apollo TV 18 db more than moderate data, and high resolution TV 5 db more than the Apollo quality link.

A nominal receiving system temperature of 500 K is assumed for space elements. Current MSFN station values for G/T, 19.6 db for a 30-foot (9.1-meter) station and 28.6 db for an 85-foot (26 meter) station, are used for MSFN values. G/T for a TDRS is assumed to be 0 db, and for a LDRS as indicated in Figure 4-2, -14 db. Path losses are shown in Figure 4-5 as a function of range. Table 4-6 establishes a baseline, assuming low orbit vehicles to use omni antennas (G/T = -27 db) and not including any system losses.

A next iteration includes additional 5 db losses, allowing for modulation, feed and multiplexing, and antenna losses as well as some time degradation. An exact value would require more detailed system designs. An additional consideration is that about 17 dbW represents a practical

Table 4-5. Data Link Classes

Class	C/N_o	Description
Voice - low data rates	46 db-Hz	High quality FM voice, 4 Kb/s coherent PSK, 10^{-5} P_E
Moderate data rates	56.7 db-Hz	50 Kb/s coherent PSK
Apollo television	74.7 db-Hz	FM, 500 KHz TV
High resolution television	79.7 db-Hz	≅10 Mb/s coherent PSK

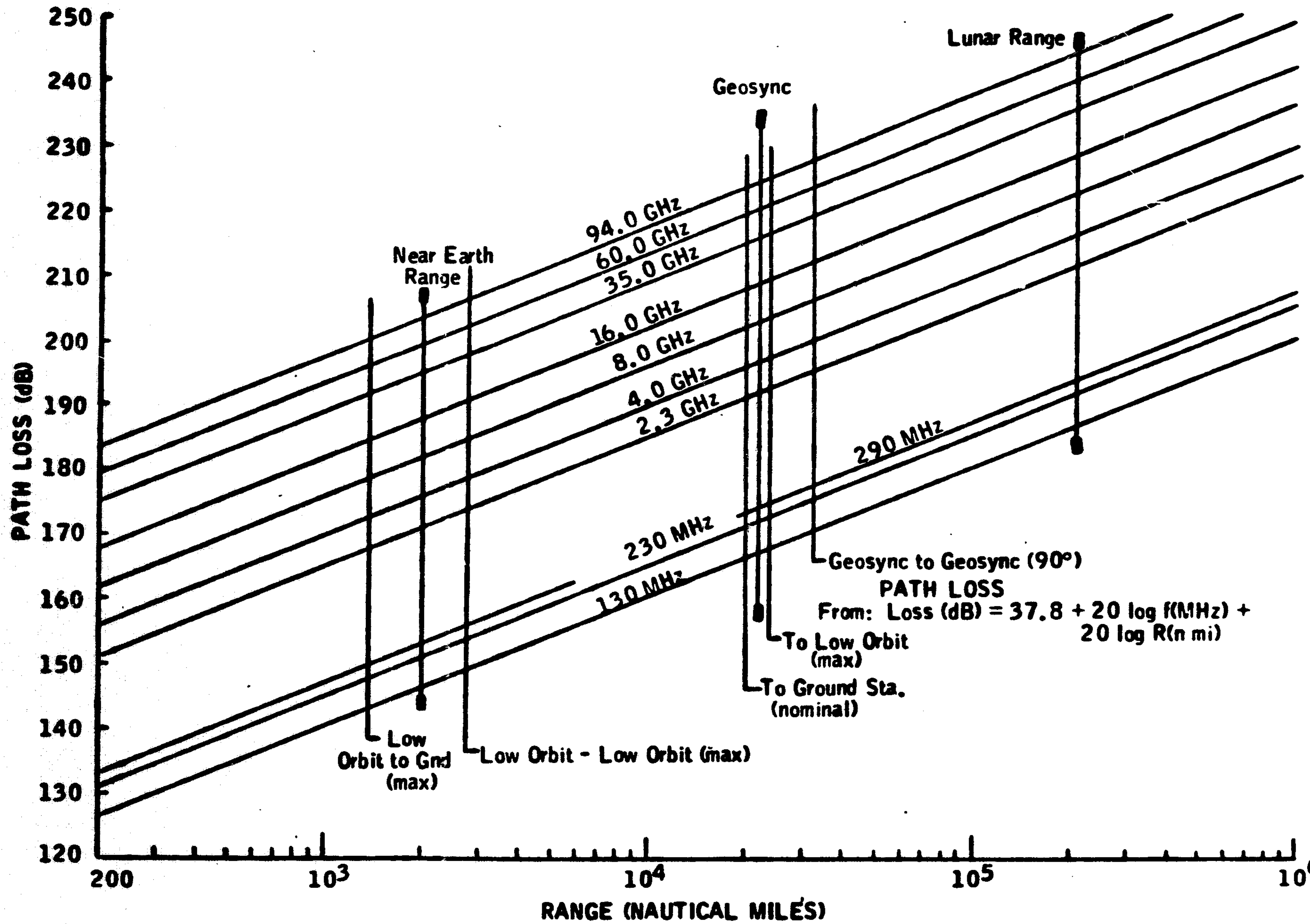


Figure 4-5. Path Loss

Table 4-6. Link Margins

Link	Link Parameters (228.6 dbw/°K-Hz Loss - G/T - Boltz Noise)	[C/No + Reqd]	=	EIRP Required: EIRP Lo-Data	or	dBW Mod Data	or	Apollo TV	or	TV
Low Earth Orbit To:										
Low orbit (max range)	173.9- (-27)-228.6	+ 46		18.3		29				
MSFN (30 ft)	167.9- 19.6-228.6	+ 46		-34.3		-23.6		-5.6		-0.6
TDRS	192.6- 0.0-228.6	+ 46		10.0		20.7				
Low orbit (Nominal Range) (270 nmi)	153.6- (-27)-228.6	+ 46		-0.2		10.5		28.5		33.5
Geosync Orbit To:										
Low orbit	192.6- (-27)-228.6	+ 46		37.0		47.7				
MSFN	Same as low orbit db more path	+ 24.7		-9.6		1.1		19.1		24.1
Geosync	Same as nominal low-orbit to low-orbit			-0.2		10.5				33.5
			=		or		or		or	
Lunar Orbit To:										
Earth orbit	211.4- (-27)-228.6	+ 46		55.8		66.5				
MSFN (85 ft)	211.4- 28.6-228.6	+ 46		0.2		10.9		28.9		
Lunar orbit (max)	161.7- (-27)-228.6	+ 46		6.1		16.8				34.8
LDRS	176.8- (-14)-228.6	+ 46		8.2						
Lunar surface	155.7- (-27)-228.6	+ 46		0.1		10.8		28.8		33.8
Lunar Surface To:										
Earth Orbit	Same as lunar orbit			55.8		66.5		84.5		
MSFN	Same as lunar orbit			0.2		10.9		28.9		33.9
Lunar orbit	Same as orbit to surface			0.1		10.8				33.8
LDRS	Same as lunar orbit			8.2		18.9				

upper limit of power for long life S-band space transmitters. Higher powers appear reasonable only if severe weight, redundancy, and cooling problems are accepted.

Table 4-7 includes the above factors, indicating power in excess of 17 dbW as required antenna gain.

Inspection of the table shows that operation with omni-directional antennas will require a maximum of 35 watts RF in near-earth operations, and 40 watts RF in the vicinity of the moon. Modes requiring greater power exceed the practical limit of transmitters in tug applications. For example, the next lowest requirements would require transmitter power as follows to operate with omni-directional antennas.

Low earth orbit	213 watts RF
Geosynchronous orbit	256 watts RF
Lunar orbit	152 watts RF
Lunar surface	2450 watts RF

These powers are excessive, and establish the possible omni-directional antenna operational modes as those listed in the table.

Gain and beamwidth parameters for a standard parabolic antenna are shown in Figure 4-6. Gain requirements from the right hand column of Table 4-7 are illustrated in Figure 4-7, with the gains of 2-, 4-, and 8-foot (.61-, 1.22-, and 2.43-meter) parabolic antenna indicated.

The chart indicates that a 2-foot (.61-meter) antenna will be adequate for the majority of links, with little improvement realized by an increase to an 8-foot (2.43-meter) antenna. Problems arise with links from geosynchronous altitude or lunar distances to low earth orbital elements. For these specific cases, directional antennas, possibly combined with low noise receivers, will be assumed as a requirement for the earth orbital elements. Requirements from the lunar surface may be supplemented with a deployable 8- to 10-foot (2.44- to 3.05-meter) antenna to be set out after landing. The additional gain will allow the MSFN links to operate in a minimum power mode.

The resultant tug communication configuration consists of a transponder capable of operation at 1-, 5-, and 40-watt outputs, a 2-foot (.61-meter) parabolic antenna and 4-foot (1.22-meter) omni antenna. Nominal operation probably would be in the 5-watt mode.

Table 4-7. Gain Requirements

Link	Omni Modes		High Gain	
	Description	Required RF Power Watts	Description	Required Antenna Gain db
Low earth orbit to:				
MSFN (30 foot)	All	3 max	None	--
TDRS	Low data	32	Mod data	10
Low earth orbit (nominal range)	Low/mod data	3.5/35	Apollo TV	16.5
			TV	21.5
Low earth orbit (max range)	None	--	Low data	6.3
			Mod data	17
Geosync orbit to:				
Low earth orbit	None	--	Low data	25
			Mod data	37.5
MSFN (30 foot)	Low/mod data	3	Apollo TV	7.1
			TV	12.1
Geosync (nominal range)	Low/mod data	3.5/35	TV	21.5
Lunar orbit to:				
Earth orbit	None	--	Low data	43.8
			Mod data	54.5
MSFN (85 foot)	Low/mod data	4/40	Apollo TV	16.9
Lunar orbit (max R)	Low data	12	Mod data	4.8
LDRS	Low data	28	None	--
Lunar surface	Low/mod data	4/40	TV	21.8
Lunar surface to:				
Earth orbit	None	--	Low data	43.8
			Mod data	54.5
			Apollo TV	72.5
MSFN (85 foot)	Low/mod data	4/40	Apollo TV	16.9
			TV	21.9
Lunar orbit	Low/mod data	4/40	TV	21.8
LDRS	Low data	28	Mod data	16.9

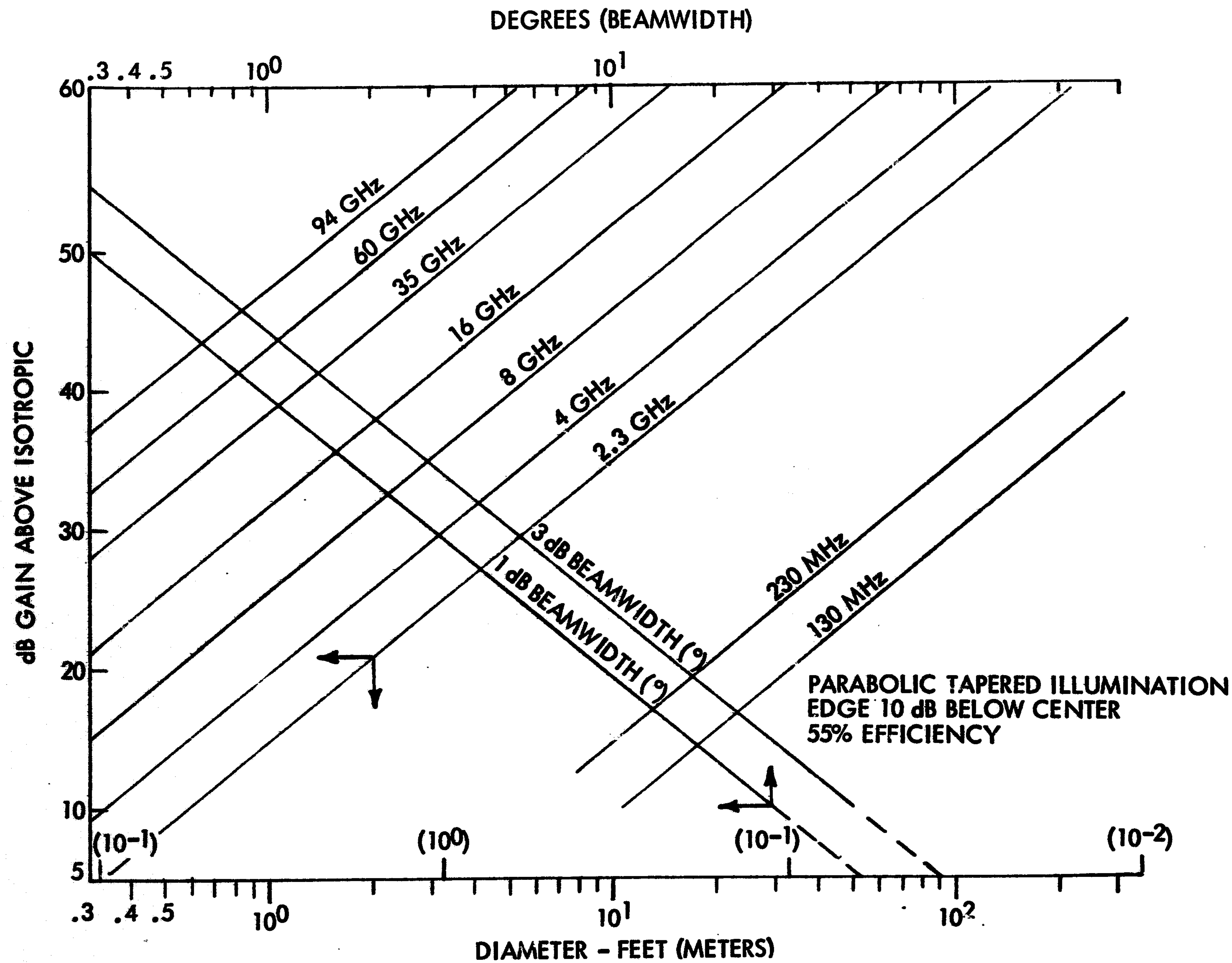


Figure 4-6. Parabolic Antenna Requirements

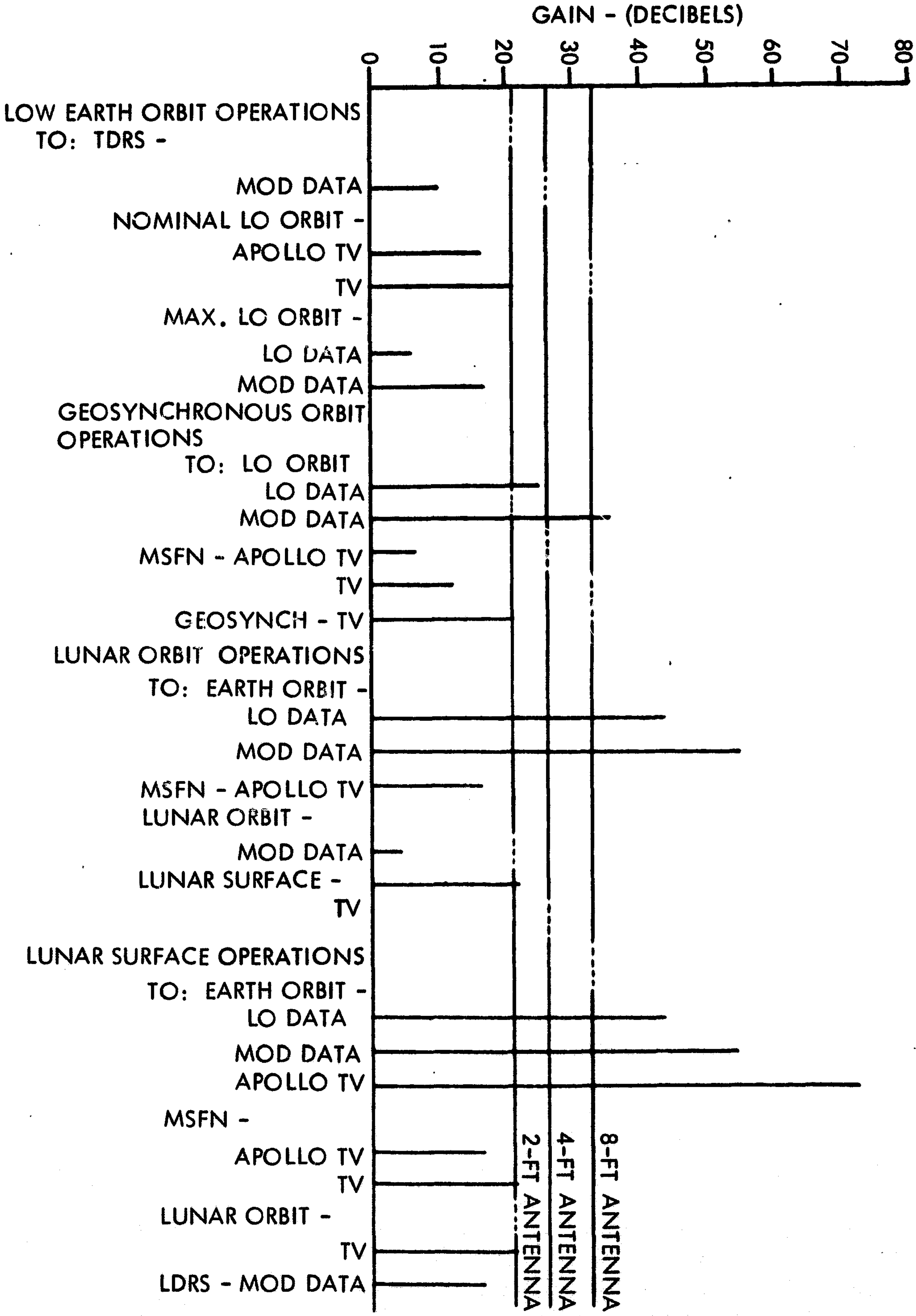


Figure 4-7. Link Antenna Gain Requirements

A consideration in specifying antennas is the detailed flight operational modes. Any location of a single antenna will result in a fairly large area blocked to the antenna. Operational problems could add a requirement for a second high-gain antenna. An additional consideration is potential loss of signals due to blockage or reflections when two or more vehicles are operating near each other. Reflection problems may be reduced to some extent by requiring circular polarization on all antennas.

The consideration of voice operations with the addition of a crew module leads to the VHF requirement. The 18 db advantage in path loss over S-band allows wide voice coverage with little penalty in weight and power. An Apollo VHF unit is assumed suitable for this function.

Data Management

The major factors in sizing the data handling portions of the subsystem include the total capacity of the memories, the operating rates, and the redundancy required. Components identified in EOSS studies provide somewhat simpler extrapolation to the tug requirements than the current EOS configuration, and therefore were used for physical configuration estimates.

EOS studies utilized core memories as a baseline, with evaluation in progress of plated wire. The plated wire essentially represents an order of magnitude reduction in the power required. Figure 4-8 shows the weight and power requirements of a memory consisting of four 32,000 32-bit word modules from information supplied by Spacetac, Inc. The estimated tug access rates, based on the plated wire speeds, are indicated for operational and mass memories.

The present EOS concept also integrates the input-output function into the processor. This approach may satisfy the tug requirements also. However, a deeper study into the detailed performance requirements and subsystem interfaces is needed to evaluate the potential direct use of EOS hardware.

Memory requirements were extrapolated from similar functions in the EOS/EOSS studies. Redundance estimates are based on assumed simultaneous performance of critical computations with comparison of results and self-testing capability in the overall processing system. The operating memory would include redundant access to the memory stacks, plus spare storage in the stacks. The mass memory would include complete redundancy of storage on all critical programs. An additional area for backup storage of programs is in the archival tape memory, although the primary use anticipated is for the storage of operational history for dump to an interrogating station on demand.

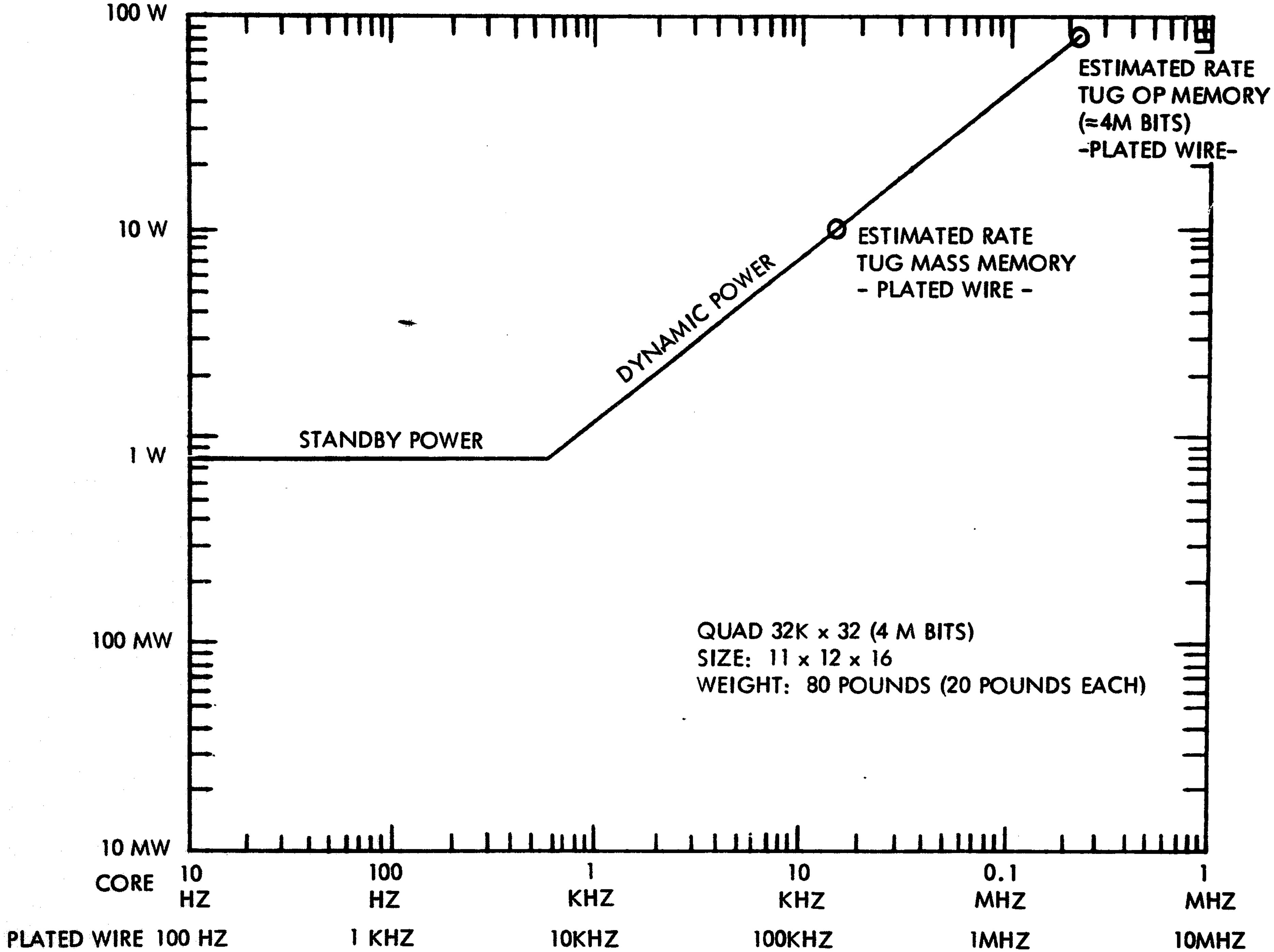


Figure 4-8. Power Required—Core Vs Plated Wire Memories

Potential reductions in the processing requirements could be made by assuming re-programming for each mission phase through the communication link, or by finding methods for reducing operating rates. There is a possibility of designing the system to operate at a wide variety of access rates, with the speed dependent on the operations actually being performed. While this adds considerable complexity to the design, reference to the variation in power with speed shown in Figure 4-8 indicates that a major reduction in average power might be obtained.

The displays and controls are extrapolated from the components identified in the EOSS studies. The major units are suitable for a wide variety of missions with both format and content adjustable by software changes. Direct control and display elements would be restricted to caution and warning, and backup to critical areas. In general, the approach would be to treat the crew module as a remote control station, with the exception of the critical backup mentioned above. Basic components include color TV displays formatted by the processor and by the video bus, light emitting diode (LED) alphanumeric displays, keyboards, hand controllers, switches, and lights. While EOSS assemblies were used for estimations of size, weight and power, the actual installations would be tailored specifically for tug use.

Equipment Summary

The C&DM equipment for the IM is listed in Table 4-8, and that for the CM in Table 4-9. As indicated in the tables, EOSS studies are a major source of physical characteristics. The major effects which are readily apparent of assuming EOS hardware would be in the memory power requirements, integration of the I/O units into the processors, and use of a larger number of the smaller EOS interface units to replace RACU's. The footage requirements for wiring assumes basic wire and cable runs will be near walls, with reasonably direct access to crew and subsystem areas. Weights are based on current Apollo RF and video coax, and on triple redundant data busses with weights equivalent to standard twisted shielded pair line.

Table 4-8. Communications and Data Management Subsystem IM Equipment

Description	Source	Unit Characteristics						Total Characteristics					
		Weight		Volume		Pwr			Weight		Volume		Pwr
		lbs	(kg)	ft^3	(m^3)	(w)	Code	Quantity	lb	(kg)	ft^3	(m^3)	(w)
S-band transmitter/receiver (10-lb spare package)	EOSS extrap EOSS extrap	25	0.113	0.70	0.0198	30	C, CC C, CC	2	50	22.7	1.40	0.0396	30
Omni-antenna	Apollo	7.5	3.4	0.25	0.0071	0	C, CC	4	30	13.6	1.00	0.0283	0
2-ft parabolic steerable antenna	Estimate	20	9.07	1.50	0.0424	15	C, CC	2	40	18.14	3.00	0.0849	15
Comm switching and checkout control (dual)	EOSS extrap	40	18.14	1.31	0.0371	25	T	1	40	18.14	1.31	0.0371	25
Premodulator processor (dual)	EOSS	50	22.68	1.17	0.0331	50	-	1.5	75	34.02	1.75	0.0495	50
MOS-LSI input/output controller	EOSS	10	4.54	0.26	0.0074	25	T	3	30	13.61	0.78	0.0221	75
MOSS-LSI processor	EOSS	10	4.54	0.26	0.0074	30	T	4	40	18.14	1.04	0.0294	120
Plated-wire operational memory (787.5 K bits each)	EOSS extrap	37.5	17.01	0.495	0.0140	20	T	4	150	68.03	1.98	0.0561	80
Tape archival storage memory	Jmpr Apollo	40	18.14	0.88	0.0249	40	I	1	40	18.14	0.88	0.0249	40
Plated-wire mass storage memory (1680 K bits each)	EOSS extrap	20	9.07	0.26	0.0074	5	T	5	100	45.36	1.36	0.0385	25
Central timing unit (dual)	EOSS	18	8.16	.44	0.0124	30	T	1	18	8.16	0.44	0.0125	30
Video unit	EOSS	18	8.16	0.23	0.0651	12	CC	2	36	16.32	0.46	0.0130	12
Remote acquisition and control units: 256 channel	EOSS	6	2.7	0.04	0.0011	8	T	7	42	19.05	0.28	0.0079	56
128 channel	(est	3	1.4	0.02	0.0006	4	T	1	3	1.36	0.02	0.0006	4
64 channel	number)	1.5	0.7	0.01	0.0003	2	T	3	4.5	2.04	0.03	0.0008	6
Signal dist wiring and data bus: heavy coax (RF) 70 ft		52	23.6										
light coax (video/RF) 148 ft	Est	18	8.2	3.25	0.0920	0	-	-	255	115.66	3.25	0.0920	0
Shielded twisted pair (data bus) 925 ft		185	84.0										
Total IM comm and data management									953.5	432.47	18.98	0.5372	568

Table 4-9. Communications and Data Management CM Equipment

Description	Source	Unit Weight lb	Unit Weight (kg)	Unit Volume ft^3	Unit Volume (m^3)	Pwr (w)	Code	Quantity	Total Weight lb	Total Weight (kg)	Total Volume ft^3	Total Volume (m^3)	Total Pwr (w)
VHF transmitter/receiver (dual)	Apollo	12	5.4	0.22	0.0062	20	C	1	12	5.4	0.22	0.0062	20
VHF antenna	Apollo	10	4.5	0.40	0.0113	0	C	2	20	9.1	0.80	0.0226	0
Commander's console (color TV, audio, I/O keyboard, manual controls connector, status lights)	EOSS	50	22.7	3.00	0.0849	85	T	2	100	45.4	6.00	0.1699	85
Light emitting diode alphanumeric display and electronics	EOSS	30	13.6	0.90	0.0255	25	T	1	30	13.6	0.90	0.0255	25
Remote acquisition and control units: 256 channel	EOSS (Est)	6	2.7	0.04	0.0011	8	T	2	12	5.4	0.08	0.0023	16
128 channel		3	1.4	0.02	0.0006	4	T	1	3	1.4	0.02	0.0006	4
64 channel		1.5	0.7	0.01	0.0003	2	T	2	3	1.4	0.02	0.0006	4
Signal dist wiring and data bus:													
Heavy coax (RF) - 44 ft	Est	33	15.0	2.62	0.0742	0	-	-	219	99.3	2.62	0.0742	0
Light coax (video/RF) - 80 ft		10	4.5										
Shielded twisted pair (data bus) - 800 ft		176	79.8										
Total CM comm and data mgt									399	181.0	10.66	0.3019	154

5.0 ELECTRICAL POWER SUBSYSTEM

5.1 REQUIREMENTS

The electrical power subsystem must generate, condition, and distribute electrical power to the using subsystems throughout the tug mission cycle. Internal energy sources are required to deliver power during all free-flying mission phases, including dormant periods up to 180 days for unmanned missions. During maintenance periods or while the tug is attached to the propellant depot, electrical power is assumed to be available from the depot. The following basic requirements are considered applicable:

1. Provide continuous electrical power to all subsystems during active mission cycle phases ranging nominally from 10 to 15 days and as high as 45 days. Quiescent space time periods for unmanned missions (minimum power levels) may range up to 180 days.

2. No added power capacity will be planned for power transfer to other vehicles or for experiments. Available power to payloads other than tug subsystems will be limited to that available due to some subsystems being turned off (e.g., lunar surface operations).

3. Minimum ground checkout and space-based servicing/maintenance will be required to meet a 3-year life span or 10 missions (reusability).

4. Fail-operational is required after first and second failures, and fail-safe is required after third failure.

5. Provide emergency power to essential loads for a minimum of 24 hours for manned missions and 4 hours for unmanned missions.

6. Provide for modular design for equipment removal and replacement or commonality between manned and unmanned configurations.

7. Minimize or eliminate transients caused by equipment failure.

This study presents parametric trade data and rationale necessary to establish a preferred EPS configuration.

5.2 LOAD PROFILE ANALYSIS

The EPS must supply a 24-hour average power level per mission day, which is a function of the type of mission or crew size. Typical power level requirements for various tug subsystems are listed in Table 5-1 for both unmanned and manned missions (selected missions for 2-, 4- and 6-man crews). An emergency operation power level also is assumed. A more detailed breakdown by subsystem component is contained in the separate subsystem sections. A typical load profile is shown in Figure 5-1. The power profile is based on the following assumptions:

1. All subsystems not operating at full power simultaneously
2. Subsystems redundancy adequate to meet fail operational, fail-safe requirements
3. Average power levels were obtained by estimates given in the "Code" column of each subsystem component list. A key to the codes is given in Table 5-2.

5.3 ENERGY SOURCES

Solar, electrochemical, and nuclear energy sources are available as potential power generation energy source candidates for the tug. Because of unique mission requirements (i.e., long space dormant periods), a combination of these energy sources may be required to meet the electrical power requirements. Energy source selections are influenced by many factors which are discussed in the following paragraphs. Some of the parameters may appear favorable from the EPS subsystem viewpoint but may have an undesirable impact upon the overall tug mission operational capabilities. Evaluation of the EPS configuration considers adverse effects or constraints upon tug missions as presently envisioned. Candidate energy sources are shown in Figure 5-2.

Solar energy, available only during the light orbital periods, can be converted to electricity by using solar cells or collectors. Collectors were eliminated because of their large size requirements and because of the associated stowage, deployment, orientation, and pointing accuracy problems.

Use of solar energy in conjunction with solar cells emerged as a strong energy source candidate, because it requires no consumable reactants and technology for its conversion is well advanced by means of solar cells.

Table 5-1. Anticipated Tug Subsystem Average Power Levels

Subsystem	Manned Missions					Unmanned Missions		
	6-Man 7 Days	4-Man 45 Days	4-Man 7 Days	2-Man 7 Days	Emergency	Space 7-Days	Lunar Landing 45-Days	Emergency
Crew Module						-	-	-
Environment control and life support	1128	1064	984	839	450			
Guidance and navigation	154	154	154	154	100			
Manned docking equipment	30	30	30	30	-			
Communication and data management	154	154	154	154	50			
Thermal control	88	88	88	88	70			
Electrical equip and losses	25	25	25	25	15			
Total crew module (watts)	1579	1515	1435	1260		-	-	-
Intelligence module								
Guidance and navigation	331	331	331	331	200	331	331	200
Unmanned docking equipment	-	-	-	-	-	60	60	-
Communication and data management	568	568	568	568	70	568	568	70
Thermal control	105	105	105	105	50	62	62	50
Electrical equipment and losses	270	270	270	270	80	230	230	80
Total intelligence module (watts)	1274	1274	1274	1274		1251	1251	400
Propulsion module								
Docking equipment	30	30	30	30	-	30	30	-
G and N lunar landing equipment	-	62	-	-	-	-	279	-
Total tug vehicle (watts)	2883	2881	2739	2564	1085	1281	1560	400

Note: Peak transient and propulsion heater loads are not included. Heater loads as high as 3600 watts additional for durations less than 20 minutes can be expected.

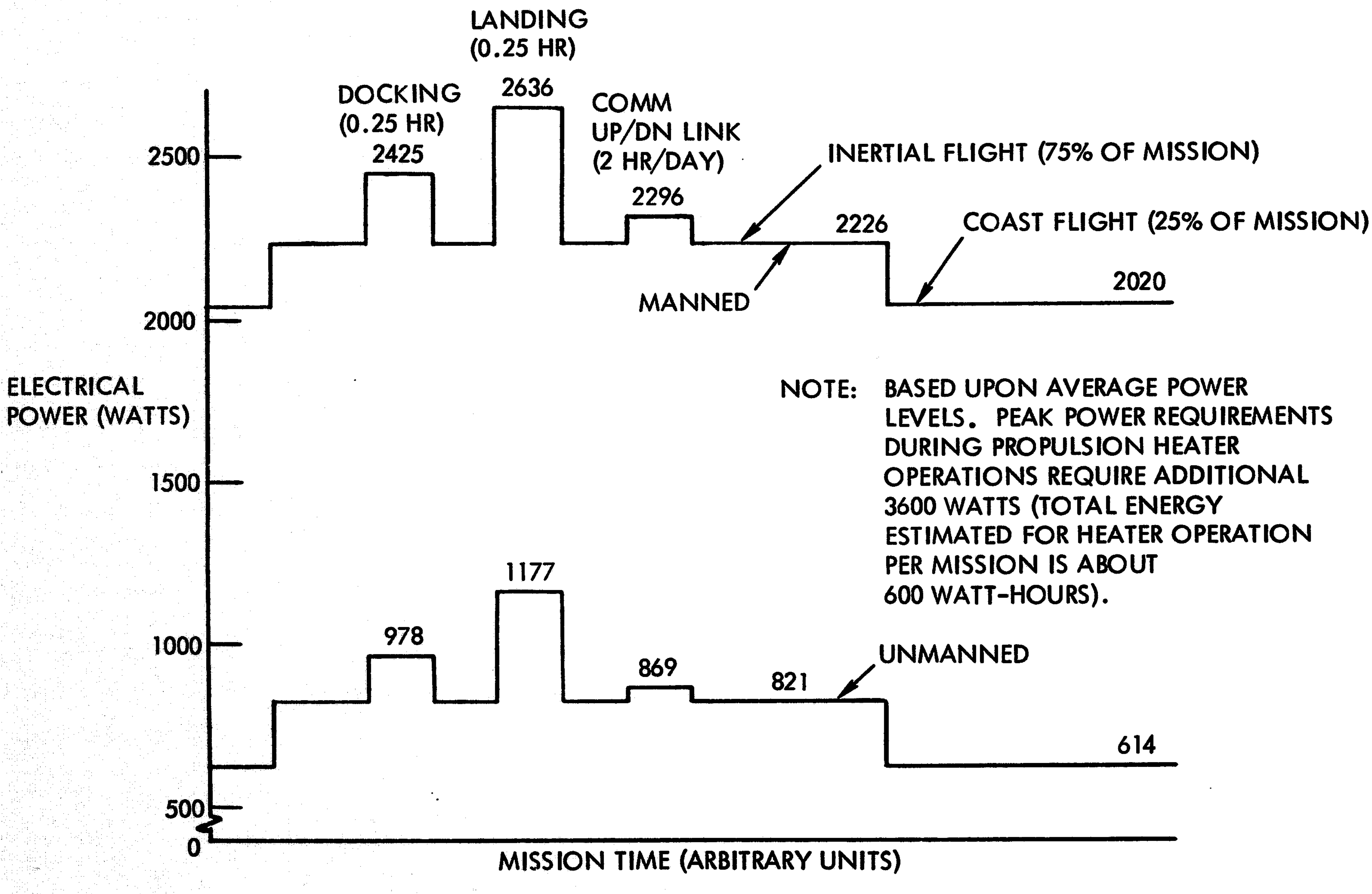


Figure 5-1. Tug Power Profile Summary

Table 5-2. Average Power Timeline Codes and Assumptions

Codes	Descriptions	Time Duration (Hr)
L	Landing operation	0.25
D	Docking operation	0.25
R	Rendezvous operation	1.00
I	Inertial flight	75 percent of mission
CP	Coast phase	25 percent of mission
C	Communication up/down link	2.00 per day
CC	Communication checkout	1.00 each dock, separation, land
T	Total mission	100 percent of mission
O	Negligible amount of mission	0 percent of mission

Solar array systems have a number of drawbacks; i.e., impact of cell equilibrium temperature, radiation degradation, power requirement for solar array orientation if employed, mission operational constraints in docking or other vehicle operations, drag penalty in low earth orbits, etc. Thus, not only must the solar array panel be sized for end-of-the-mission performance but also for storage of electrical energy required for the dark period of operation. These factors impose weight and cost penalties on solar array use. On the other hand, solar array systems have demonstrated high reliability for space applications.

The electrochemical energy sources considered for the tug are fuel cells, APU's and batteries. In some instances, it is difficult to isolate the chemical energy sources and the associated conversion functions (i.e., batteries). For this reason, the electrochemical candidates will be treated as a direct electrical energy source.

Fuel cells are high thermal efficient devices for the direct conversion of chemical energy into electricity. Two reactants (fuel and an oxidizer) supplied to the fuel cell are consumed in an electrochemical reaction producing electricity, water, and heat. The hydrogen-oxygen fuel cell, because of its high thermal efficiency and water production capability, becomes the most practical means of electrical power generation for

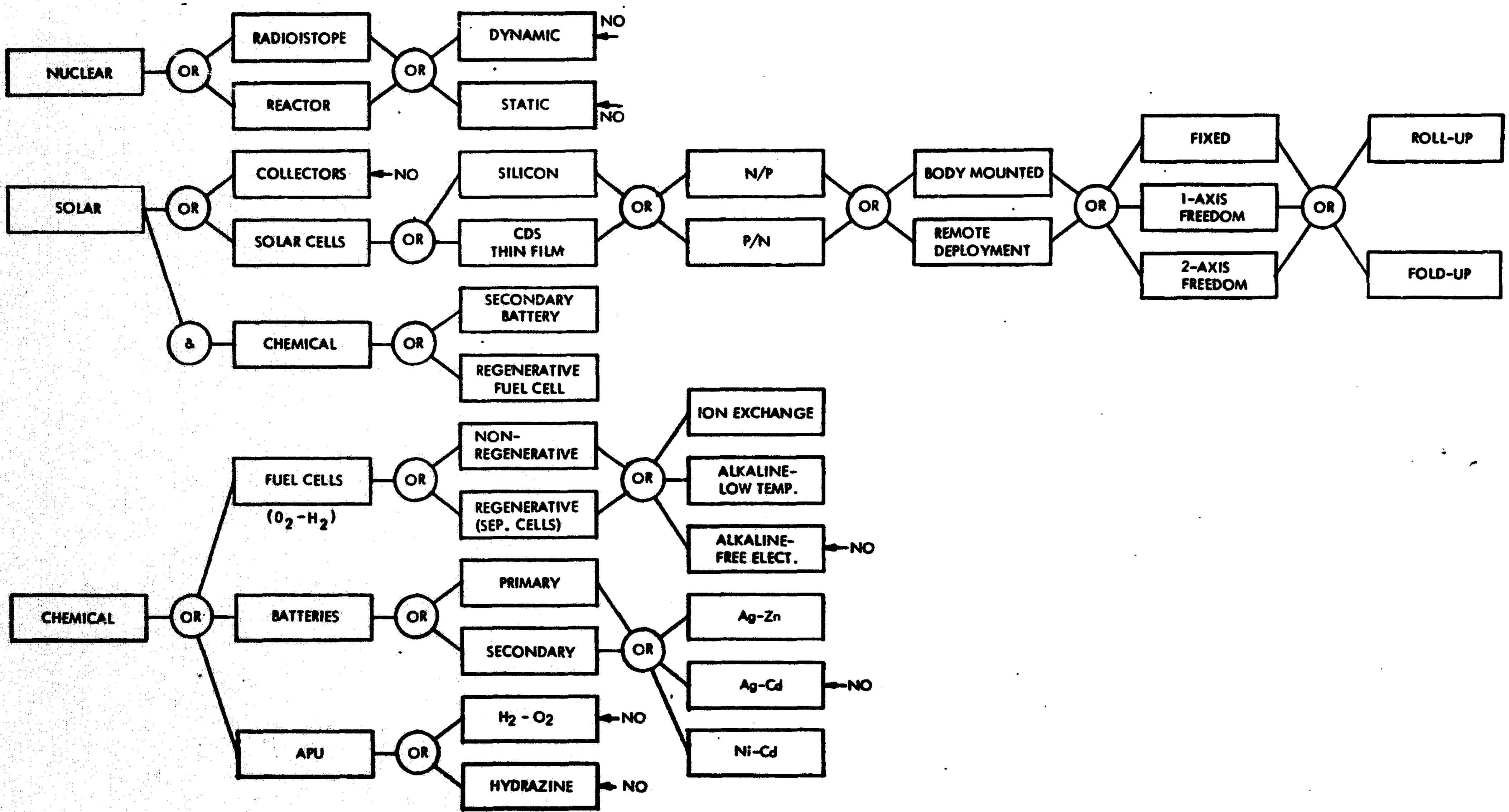


Figure 5-2. Primary Electrical Power Source Trade Tree

manned earth-orbiting applications and other relatively short-duration missions where the water byproduct is required to fulfill mission requirements (water conversion efficiency equals approximately 98 percent; hence, some of the reactant fuel penalty is affected by the water credit). If the fuel weight is chargeable to the EPS (byproduct water considered a waste in this case), then the fuel cells are less attractive as a means of electrical power generation. Solar power sources by themselves (i.e., without energy storage) are not candidates because of their requirement for light; thus, fuel cells may be used with solar energy systems for supplying eclipse power. In this case, a portion of the water byproduct may be used to meet manned mission water requirements and the balance dumped overboard. The combination of fuel cells and electrolysis cells (fuel cells operating in a back-to-back mode) becomes a strong candidate for supplying eclipse power (where water credit is not needed) because of the elimination of the consumable reactants fuel penalty.

Secondary alkaline batteries are potential energy sources for both the storage and generation of electrical power when used with solar array systems and as peak power batteries employed to optimize selection of the fuel cells and nuclear power sources. In addition, batteries can function as an emergency power source or primary source for relatively low power levels and short-mission durations. Application of batteries for the tug primary energy source have been eliminated because of the initial and resupply weight penalties needed to meet the tug energy requirements (power levels and mission durations).

APU's were eliminated on the basis of noncompetivity with fuel cells in reactant consumption rate and expected lifetimes. The tug loads would not demand APU performance characteristics (i.e., high peak loads for short durations with a relatively low average power level).

Radioisotope heat-energy source concepts are advanced to the stage that both static and dynamic energy conversion methods are applicable to the power range required for the tug. Fuels that may be considered are: Co-60, Sr-90, Po-210, and Pu-238, with the last fuel being the primary contender. Shadow shielding and source separation distances were evaluated in an attempt to reduce shielding requirements (weight). Safety aspects (i.e., shielding, controlled reentry and recovery, and end-of-life disposal) present design problems resulting in high weight penalties for the heat source; however, as with all large isotope heat sources, the major disadvantages are cost and lack of assurance of availability.

Other nuclear (i.e., reactors) are not considered competitive for tug applications based upon availability, cost, and safety (mission constraints) requirements because several alternate power systems may be used that are more applicable for the required power levels. Trade study data for the leading energy source candidates are presented in the following paragraphs.

5.3.1 Solar Photovoltaic Concepts

A number of solar array configurations have been considered in previous NR studies. Several different design approaches are possible, four of which are:

1. General Electric rollup solar array
2. Hughes flexible rollup solar array
3. Boeing large area solar array
4. ATM array

The General Electric concept for the rollup array features two-array substrates that are deployed by a single Bi-stem deployable boom. The array structural stiffness is provided by tension in the two substrates. In the deployed state, the array has a single-point attachment at the vehicle interface. A calculated power-to-weight ratio of 32.3 w/lb (71.3 w/kg) for the basic 250-square foot (23 square meter) array module is made possible by the use of beryllium for the storage drums and the leading edge member.

The Hughes flexible rollup array (1.5 kw) concept deploys two substrates from a common drum with four Bi-stem rods mounted in two actuators. Development of this array, which has a power-to-weight ratio of 21 w/lb (46 w/kg), is expected to be completed with a flight demonstration in the third quarter of 1971 (under Air Force Contract F33615-68-C-1676). In addition to the basic rollup array, the Contract includes the associated two-axis orientation and control system along with the necessary power conditioning, energy storage, and instrumentation.

The Boeing large area solar array is a 1250 square-foot (116-square-meter) (gross module area) folding panel array, which has a power-to-weight ratio of 21 w/lb (46 w/kg). This concept utilizes a beryllium framework panel with the solar cells mounted on a fiberglass tape substrate, which is stretched on this frame.

The ATM array is composed of four wings with a total module area of 1230 square feet/114 square meters and has a power-to-weight ratio of 2.8 w/lb (6.2 w/kg). This array, scheduled to be flown on the NASA Skylab Program, consists of aluminum honeycomb panels that are hinged to each other for deployment by a scissor-type mechanism.

Further development of solar array technology is presently being conducted by Lockheed under NASA contract for large area (space station-type) applications.

A qualitative comparison of the first four concepts was made to determine the configuration best suited for a tug mission. A three-point rating system based on a set of mission-related performance areas was used to rank each concept. The performance factors are discussed with supporting rationale with results shown in Table 5-3.

Maximum Productivity

This generalized performance area has been divided into two specific measures of productivity (i.e., power-to-weight ratio and power-to-stowed-volume ratio). The power-to-weight ratio is a function of the basic array design if it is assumed that the solar cell and cover glass are the same for each approach. The GE rollup array ranks "best", because the system has the highest power-to-weight ratio (30 watts per pound/66 watts per kilogram). Both the Hughes rollup and the Boeing foldout arrays have a power-to-weight ratio of approximately 21 watts per pound (46 watts per kilogram). The ATM array is the obvious choice for the last place because of the 2.76 watts-per-pound (6.08 watts-per-kilogram) ratio. The scaling up or down of the existing concepts to the sizes required for the tug should not change the relative ranking of these concepts. Also, the structural modifications required to withstand the acceleration modes of operation should have similar weight penalties for all solar-array approaches.

Table 5-3. Solar Array Comparison Matrix

Performance Factor	GE Rollup	Hughes Rollup	Boeing Foldout	ATM Array
Maximum productivity				
Power-to-weight ratio	1	2	2	3
Power-to-volume ratio	1	1	3	3
Cost ($/kw)	2	2	3	1
Minimum constraint imposed on operation	1	1	3	3
Minimum degradation and resupply	2	2	2	1
Maximum ability to achieve primary power operation (reliability)	1	2	2	1
Code: 1 = best; 2 = good; 3 = worst				

The stowed volume of the array also is an important measure of the performance of each concept. Both rollup array approaches have been ranked "best" because of the inherent compactness of these designs (approximately 300 watts/ft^3 or 10,590 watts/m^3). By comparison, both fold-out panel approaches have large stowed volume requirements (approximately 60 watts/ft^3 or 2118 watts/m^3).

Cost

A quantitative comparison of array system costs would be difficult to determine; however, the relative ranking of each concept can be estimated based on present development status and inherent hardware costs associated with each design. The ATM array is ranked lowest in cost because of its advance state of development and relatively low hardware cost. Both rollup array approaches are presently in the same state of development (i. e., design qualification of a prototype model). Both of these systems should have approximately the same hardware costs. The Boeing foldout array must be ranked as the highest in program cost because of the lack of present funding and the high hardware costs associated with the formed beryllium framework.

Minimum Constraint Imposed on Operations

Both rollup array approaches are ranked "best" in this category because of their capability for repeated orbital extension and reaction cycles. In addition, these rollup arrays can be retracted into a relatively small package for removal and replacement in orbit. The mechanical interface with the station is relatively simple for either rollup array approach. On the other hand, the folding panel array configurations are not capable of orbital retraction without major redesign of the linkages.

Minimum Degradation and Resupply

All candidates should rank approximately the same in this category, except for a small advantage associated with the ATM array with its honeycomb substrate. This heavy substrate will afford backside shielding against radiation damage as well as increased thermal mass to increase the low temperature during eclipse periods. This reduced operating temperature excursion may reduce thermal cycling damage to the solar cell interconnections during lunar orbit operation.

Maximum Ability to Achieve Primary Power Operation

This performance factor can be related to the inherent reliability of the array system. Because the reliability of the solar cells and associated interconnection should be the same for all candidates, the reliability of the deployment operation is the factor that must be assessed to determine the relative ability to achieve primary power operation. The GE rollup array was ranked "best," because deployment is accomplished with one deployable boom in one actuator. The Hughes rollup array requires four deployable booms in two actuators. The Boeing foldout array is a multiple fold system with flipout side panels. The ATM array also was ranked "best" because of the flight experience that will be logged prior to the possible application of the array for tug.

Array Configuration Conclusion

Based on this qualitative comparison, it is concluded that the two rollup concepts have many distinct advantages over the foldout panel approaches. It is more difficult to select between the two rollup concepts because of the performance similarities between these approaches. However, the GE single deployable boom configuration has the following additional advantages over the Hughes two-boom concept:

1. It requires no synchronization of boom deployment/retraction.

2. It has a single hard-point attachment to the supporting structure in the deployed condition.

3. It is potentially a lighter-weight approach because less structure is required to deploy the same solar cell area.

Figure 5-3 shows the relative solar array size and system weights as a function of power level for both earth orbit and lunar orbit operation.

5.3.2 Energy Storage Candidates

All candidate EPS's require storage of electrical energy to meet electrical peak, eclipse or emergency power requirements. The trade tree for the energy storage concepts is shown in Figure 5-4 and discussed in the following paragraphs. Two candidates considered were secondary batteries and regenerative fuel cells (separate fuel and electrolysis cells) for possible application in conjunction with solar array primary system. The regenerative fuel-cell system is eliminated because more than adequate reactants would be available on the tug (boiloff from propulsion system requirements).

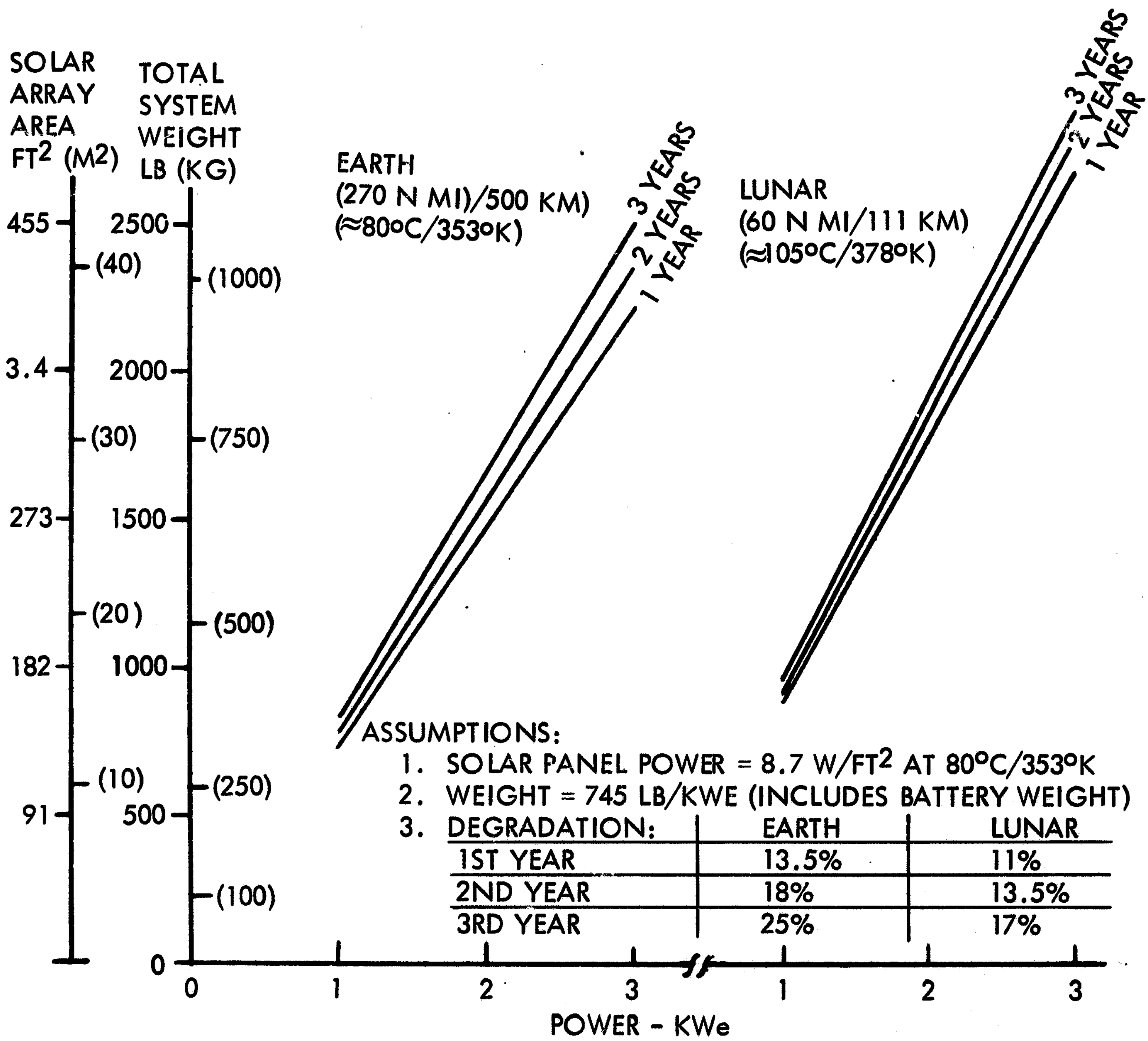


Figure 5-3. Rollup Solar Array Characteristics

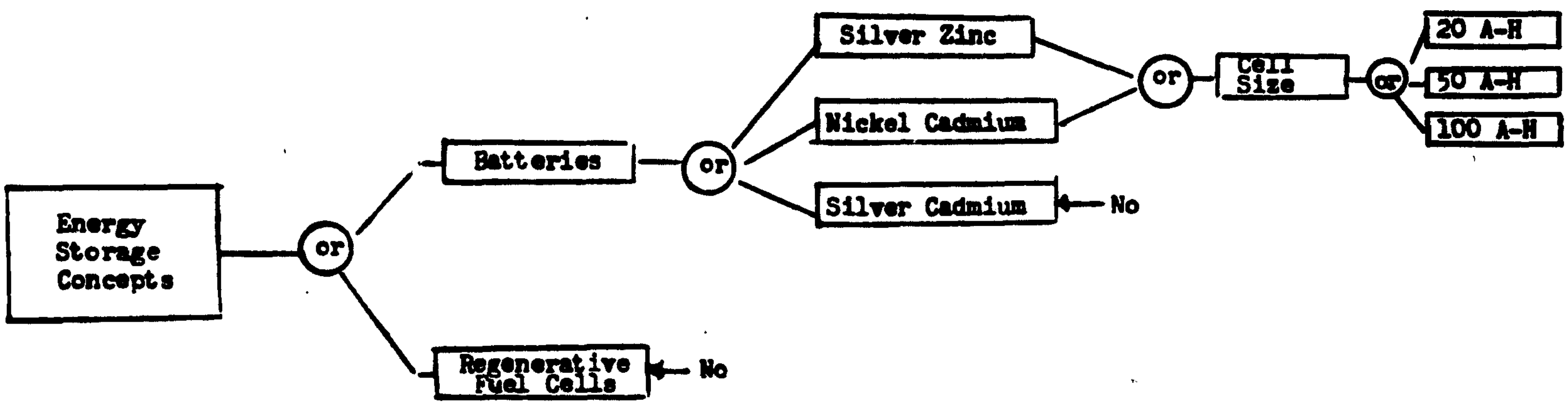


Figure 5-4. Energy Storage Concepts

The requirement of long reliable life for a high number of charge and discharge cycles (25 to 30 thousand) is the primary factor favoring a nickel-cadmium battery for use with solar arrays. Silver-zinc batteries may be considered for peaking power requirements. However, silver-zinc batteries have a short wet stand life and a lower charge-discharge cycle capability at the discharge depth required for application. Silver-zinc battery capability is expected to be increased to at least a one-year operation with a few hundred cycles charge-discharge capability at 15 percent DOD. Although lighter than nickel-cadmium batteries (from two to five times lighter), the silver-zinc batteries have a shorter life than nickel-cadmium batteries (≈two to three years) and this may impose a greater logistics and maintenance problem. The silver-cadmium batteries are excluded from consideration for large-scale application because of their excessively high cost. Such batteries are limited to applications where minimum magnetic field interference with experiments is a requirement. Long battery-life requirements are best met by operating at low temperatures ($\leq$ 32 F/273 K) and at shallow depth of discharge ($<$ 15 percent). Thermal control of the battery employs a coldplate with three separate cooling loops with the excess heat rejected to the radiator. Loss of any one cooling loop will in no way degrade battery performance, whereas loss of any two loops will result in some performance degradation.

Figure 5-5 shows the battery cycle life and total energy characteristics for nickel-cadmium, silver-cadmium, and silver-zinc batteries. Two lines are plotted in the figure for each type of battery. The lower line predicts the number of discharge-charge cycles to failure versus the percent depth of discharge in each cycle. A 20-percent discharge of a 10-kwh battery yields 2 kwh, for example, and a total energy output over its life of 2 kwh times the number of cycles, which provides a point on the upper curve. Peaks on the energy curves for each battery correspond to depths of discharge which produce optimum life. Table 5-4 lists representative values for available nickel-cadmium secondary power sources.

5.3.3 Radioisotope Power Systems

Radioisotope heat sources decay exponentially in time at a rate that is not affected by external forces (i.e., for any particular radioisotope the decay rate is a constant). As a result of this property, the heat generation from the decay of radioisotopes cannot be turned off. This feature makes them a very reliable heat source. On the other hand, they continually constitute a hazard that must be designed for during all handling phases. The manner in which safety constraints are met may result in large effects on the radioisotope heat source weight.

Table 5-4. Nickel-Cadmium Battery Summary

Battery Capacity (Ampere Hours)	Total Packaged Weight (24 Cells lb/kg)	Physical Dimensions (in/cm)
12	34/15.4	6 x 8 x 11.8 (1.5 x 20.3 x 30)
20	56/25.4	7.5 x 7.5 x 13.1 (19 x 19 x 33.3)
50	132/59.9	19.9 x 11.9 x 6.4 (50.5 x 30.2 x 16)
100	255/115.7	21.7 x 16.6 x 8.4 (55 x 42 x 21.3)

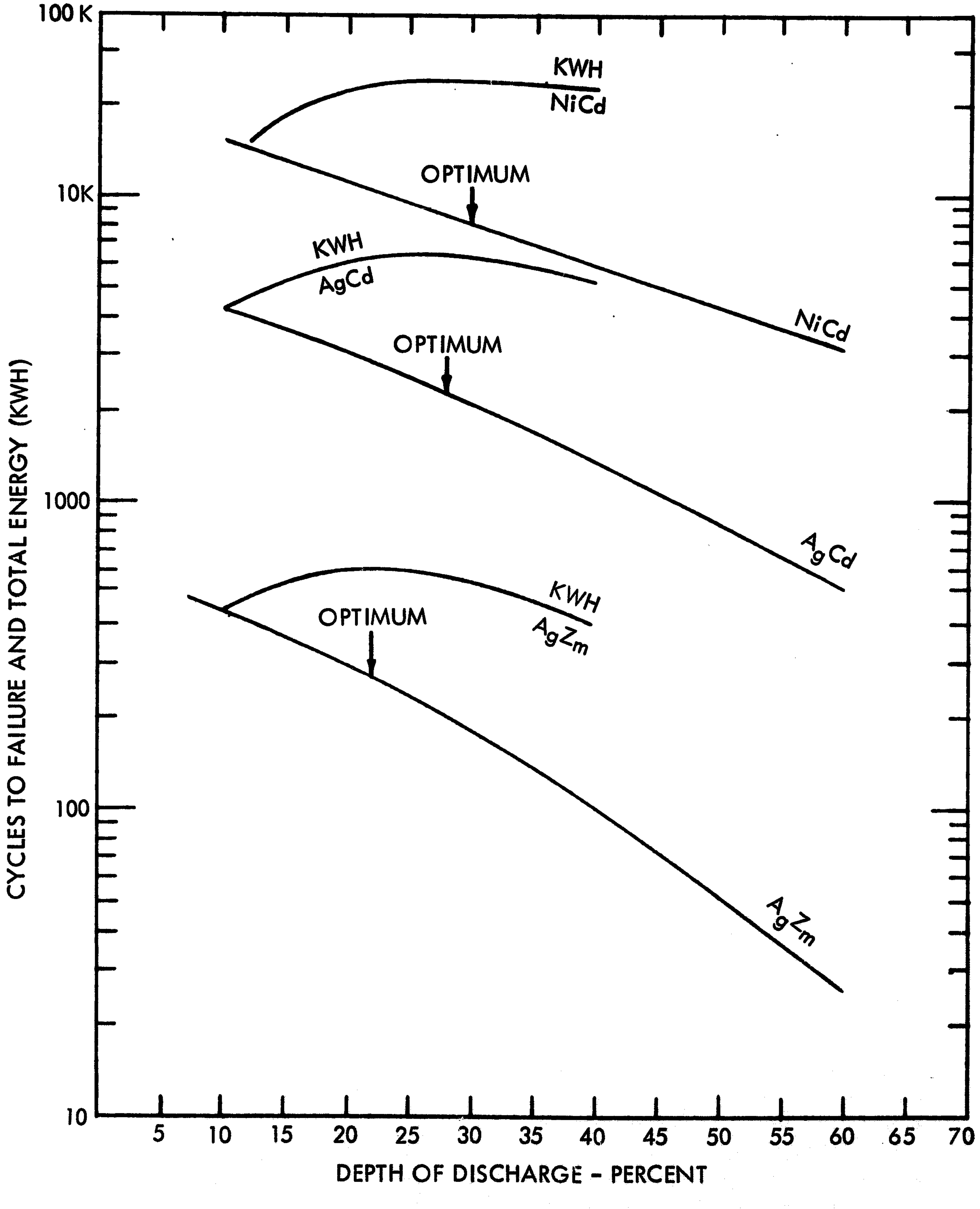


Figure 5-5. Battery Cycle Life and Total Energy Characteristics—Energy Based on 10-Kwh Battery

The selection of an isotope for a particular application is based on considerations related to the isotope half-life, power density, availability, and safety. For unmanned tug applications, safety is primarily concerned with protection of the general public and requires that the isotope heat source be controlled during all phases of use: fabrication, shipment, ground handling, prelaunch, launch, preorbit, operation, rendezvous, and end-of-life disposal.

The following properties are desirable for radioisotope fuels to be used in space power systems: long half-life, high-power density, readily available, minimum shielding required, low biological hazard, fuel form compatible with encapsulation materials, high melting temperature, and low cost.

Radioisotopes with power densities sufficiently high to warrant consideration for space power application can be categorized as alpha, beta, or gamma emitters. The gamma emitters have been eliminated from consideration for manned space applications because of the executive shield weights that would be associated with their use. The beta emitters, with the exception of Pm-147 and Pm-170, also require excessive shielding. Of the alpha emitters, Po-210, Pu-238, Cm-242, and Cm-244, have received the most attention for use in space power systems.

No radioisotope completely satisfies the previously mentioned list of desirable requirements. Further evaluation narrows the six radioisotopes down to Po-210 (half life of 138 days) for short-duration missions and Pu-238 (half life 86 years) for long-duration missions. Therefore, the AEC has concentrated on these two isotopic fuels for their heat source development effort. One of the drawbacks of using Pu-238 is its cost ($540 per thermal watt) and availability. For unmanned missions where the payload has a high tolerance for nuclear radiation, perhaps some of the lower-cost gamma and beta emitters should be reconsidered. For example, Cm-244 (an alpha emitter) with a half life of 18 years, has an estimated price of $64 per thermal watt. Co-60, a heavy gamma emitter with 5.3 years half life, has an estimated future price of $16 per thermal watt. However, it is assumed that the use of gamma emitters would be excluded because ground personnel and the general public must be protected from the consequences of handling, launch, or reentry accidents for unmanned space applications and system commonality with manned mission applications. For this reason, a Pu-238 heat source is used to show representative radioisotope power system weights for the tug.

All the radioisotope power systems considered to date for unmanned applications are based on static power conversion devices (thermoelectric or thermionic). The current operational systems, SNAP 3, SNAP 19, and SNAP 27, use thermoelectric converters. SNAP 27 is the largest of these systems with a 63-watt electrical output. Current state-of-the-art thermoelectric devices convert thermal energy to electricity with efficiencies varying from 5 to 9 percent. The higher efficiency is obtained by cascading SiGe and PbTe materials. The weights shown on Figure 5-6 are based on the use of cascaded thermoelectric converters. The hot junction of the converter is thermally coupled to the isotope heat source by a circulating liquid metal. Another liquid metal loop removes heat from the converter and rejects the waste heat to space by means of a radiator. The heat source weight includes an allowance for a controlled system for intact reentry and recovery of the isotope fuel capsules. This feature accounts for 75 percent of the heat source weight and over 40 percent of the total power system weight.

The cost of Pu-238 and the direct relationship between the weight necessary for safety and the quantity of radioisotope required places emphasis upon the importance of power conversion efficiency. Column I of Table 5-5 lists weights and other characteristics for a radioisotope power system based on an organic Rankine power conversion system. Total weight is reduced approximately 50 percent over the thermoelectric conversion system shown in Column II of the table. Also, in order to minimize Pu-238 inventory, the system is sized for average orbit power (allows for power conditioning losses). Peak power is obtained by a secondary battery assist.

5.3.4 Fuel-Cell Concepts

A number of fuel cell types considered in previous NR studies are capable of space shutdown and restart. Two of the leading fuel-cell candidates types for space power application are: (1) ion exchange membrane (solid polymer electrolyte) - General Electric, and (2) Alkaline low temperature matrix - Pratt and Whitney, Allis-Chalmers.

The General Electric Company and Pratt and Whitney Aircraft are presently under NASA contract to further develop applicable fuel cell technology for the earth-orbiting shuttle program. Although 2000 to 3000 hours of operating life can presently be obtained, a potential of 10,000 hours of fuel-cell operating life appears feasible. Table 5-6 compares projected characteristics for the ion exchange membrane and low-temperature absestos matrix units for a projected 2-kilowatt unit with a 5000-hour life. No clear choice between the two fuel-cell design approaches should be made at this time. Figure 5-7 shows the reactant supply pressure effect for the ion exchange membrane unit on cell voltage versus current. Typical 34 to 40 such cells are required in series per fuel-cell stack to provide the required voltage regulation (minimum voltage requirement considerations).

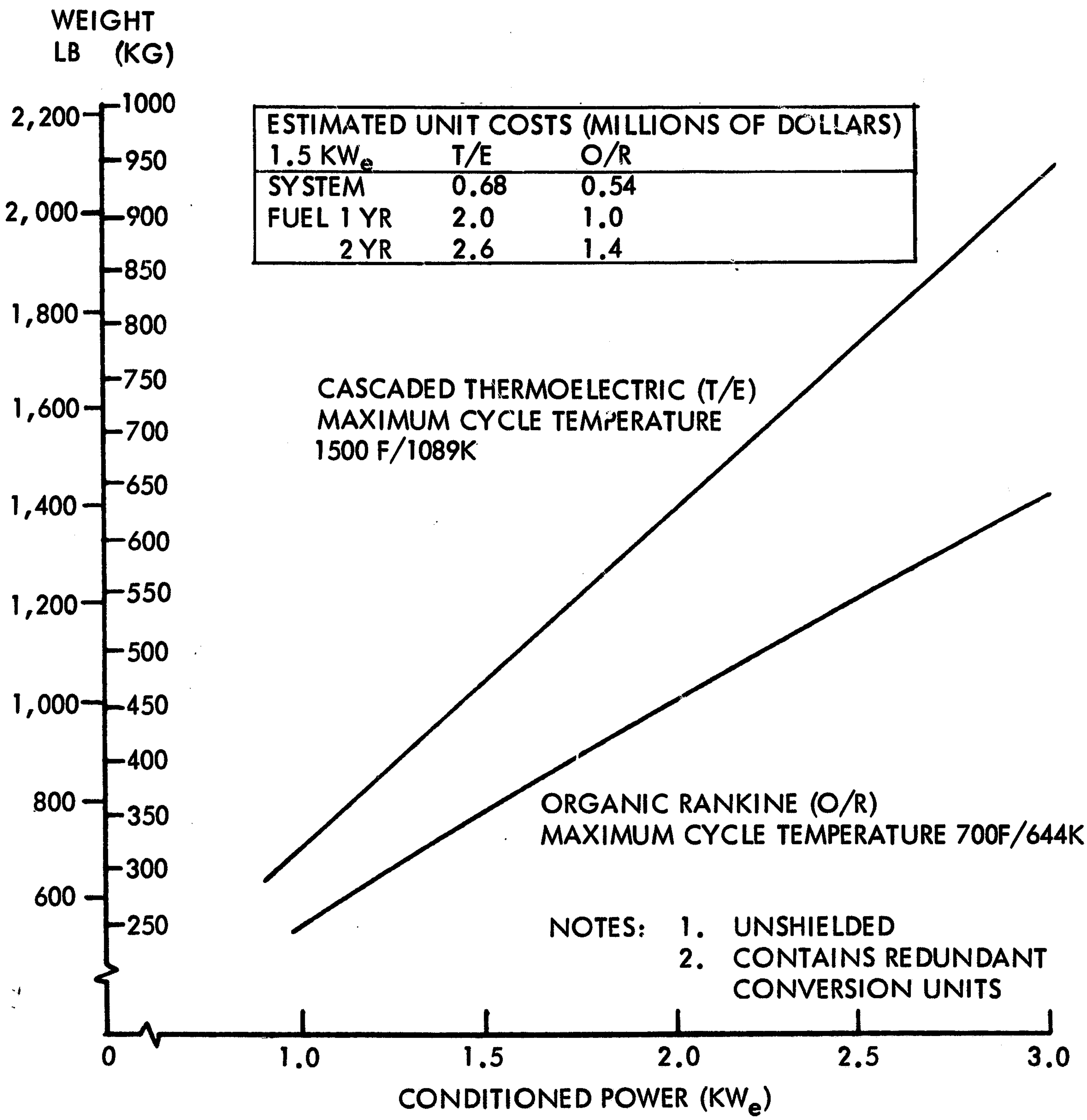


Figure 5-6. PU-238 Radioisotope Power Systems Comparisons

Table 5-5. Pu-238 Fueled Radioisotope Power Systems
(One-Year Life)

Power Conversion	I Organic Rankine		II Thermo-Electric	
Net Electrical Power Output, kw_e	2.36		2.36	
Heat source power, kw_T (Beginning of life)	16.4		28.6	
Maximum cycle temperature (F/K)	700/644		1450/1061	
Radiator area (ft^2/m^2)	127/11.8		198/18.4	
Average radiator temperature (F/K)	310/428		305/425	
Overall thermal effect percent (1)	-		60	
Weight (lb/kg)				
Power conversion subsystem	110	50	400 (2)	181
Heat rejection subsystem	185	83.9	290	132
Boiler and NaK Lines	100	45.4	-	
Start system and organic inventory	73	33	-	
Auxiliary NaK heat rejection	64	29	112	51
Power conditioning and control	220	100	220	100
Structure	68	31	102	46
Heat source CIR (3)	984	446	1720	780
Secondary batteries	200	91	200	91
Total (lb)	2004	909.3	3044	1381

(1) Not including PC and D
(2) Includes NaK lines, pumps, etc.
(3) CIR = controlled intact reentry

Table 5-6. Comparison of Projected Fuel Cells (2 KW Rating)

Item	General Electric Ion Exchange	Pratt and Whitney Low Temp Matrix
Water removal	Wick	Pump
Weight (lb/kg)	60/27	80/36
Volume (ft^3/m^3)	4/0.113	7/0.198
Operating temp ($^oF/K$)	140-180/333-356	180/356
Parasitic power (w)	50	80
Specific reactant Consumption (lb/kwh)/(kg/kwh)	0.98/0.44	0.84/0.38
Voltage degradation (MV/cell-hr)	0	5
Waste heat (Btu/kwh)/(W_t/W_e)	2400/0.703	2100/0.615
Reactant purging Frequency and quantity rating	2nd	1st
Tolerance air, CO and CO_2 Life degradation, system inerting required and reactant supply integration rating	1st	2nd
Supply integration rating		
Low-temperature tolerance		
Storage and quiescent mode rating	2nd	1st

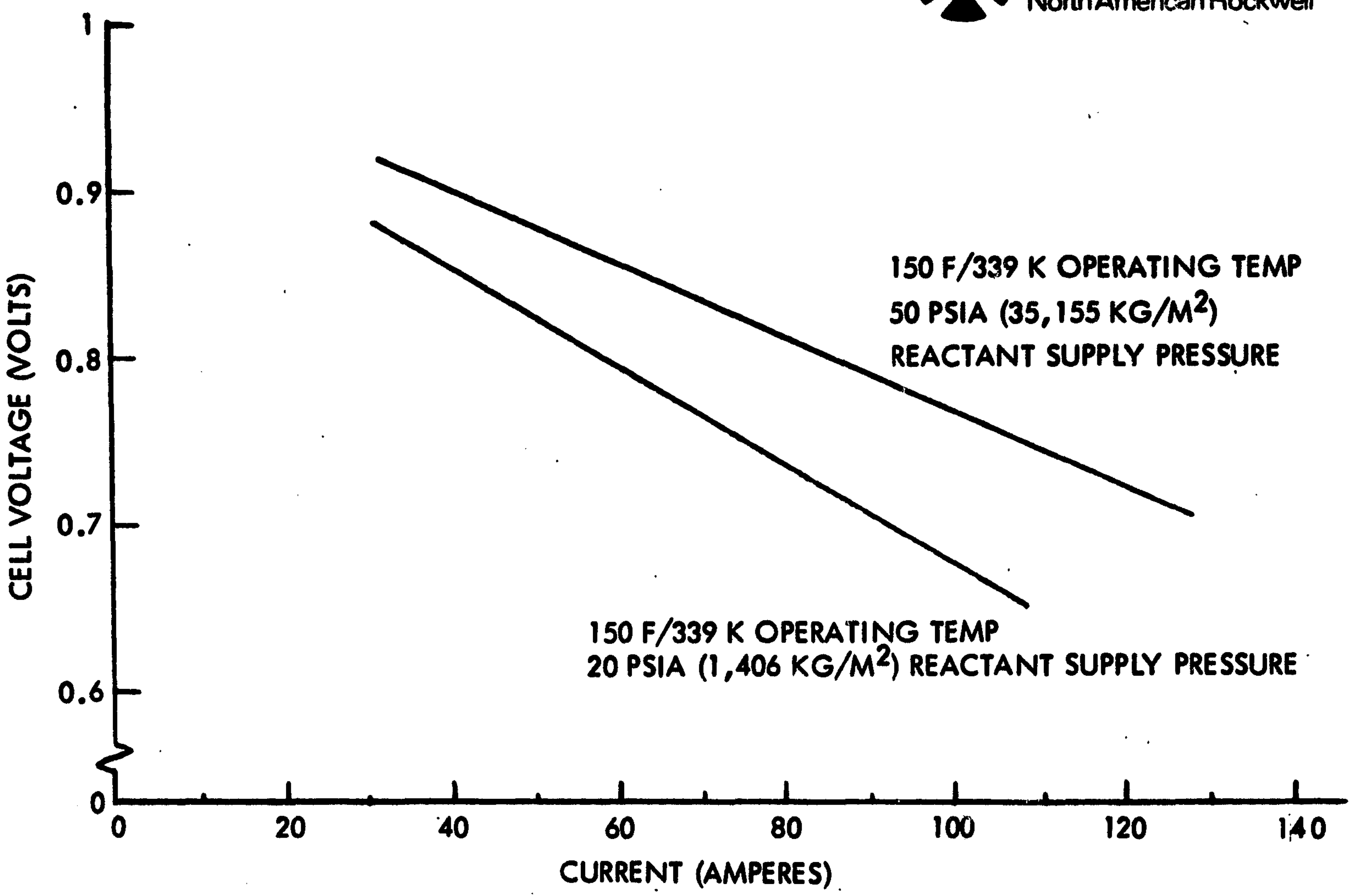


Figure 5-7. Typical Ion Exchange Single-Cell Performance (Cell Active Area $0.7 Ft^2/0.065 M^2$)

The number of stacks per fuel cell module will affect the module weight, redundancy, and transient performance capability. A three-stack-per-module configuration may be optimum for tug applications.

The reactant (H_2 and O_2) supply for fuel-cell operation is assumed to be available from the tug propulsion subsystem supply. Filtering of the propellant grade reactants will improve fuel-cell performance and minimize purging requirements. Figure 5-8 shows the energy requirements, reactant consumption, and water production rates for various average power load levels as a function of mission cycle duration. Figure 5-9 depicts the heat rejection requirements for fuel-cell electrical power output. Because fuel cell operation will be in the range of 140 to 180 F/333 to 356 K, heat rejection may be integrated with the tug thermal heat rejection loop without significant weight penalty.

5.4 POWER CONDITIONING AND DISTRIBUTION

The basic function of power conditioning equipment is to modify the electrical characteristics of the power source to some desired form or values. The power generation concept recommended for the tug is primarily a direct current (dc) system at a 28-volt nominal value. Static

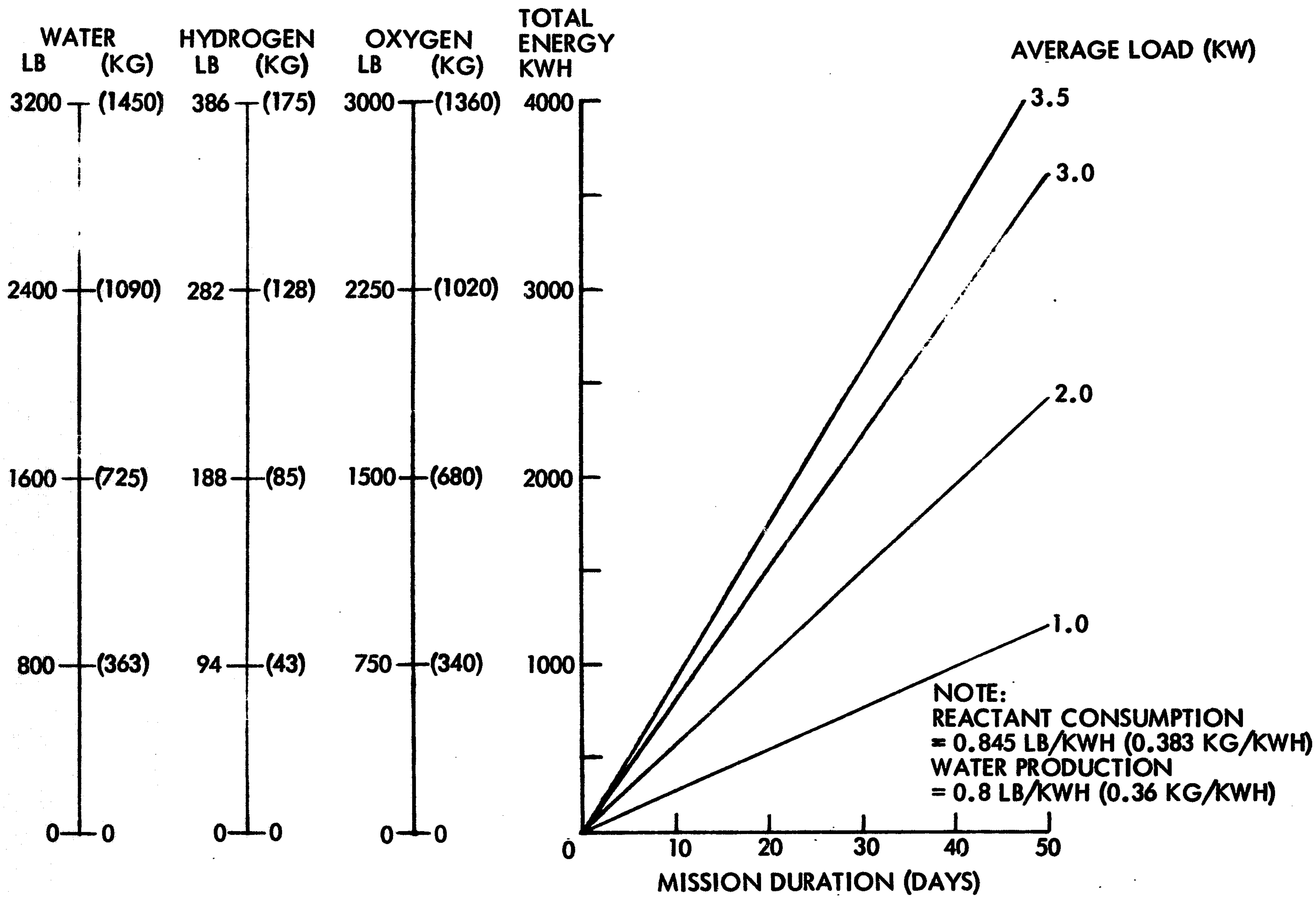


Figure 5-8. Total Electrical Energy, Reactant Consumption, and H_2O Production as Function of Load and Mission Duration

Space Division
North American Rockwell

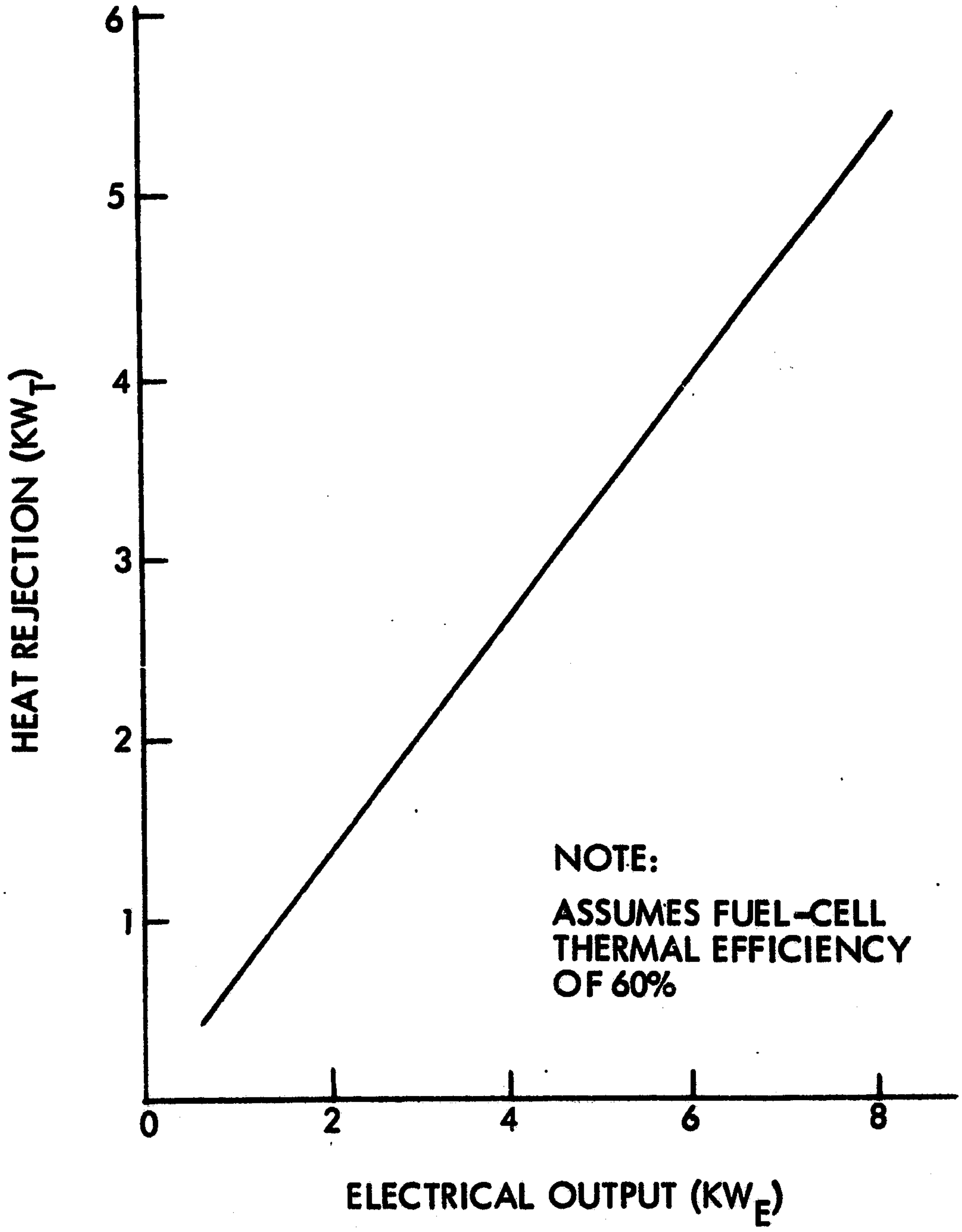


Figure 5-9. Fuel-Cell Heat Rejection as Function of Load

inverters will be used to provide a centralized primary alternating current (ac) power source rated at 115 volts and 400 Hertz. Trend data indicate power conditioning equipment has about a 30 lb/KVA (13.4 kg/KVA) weight-to-power density. Requirements for voltage regulators and battery chargers will be influenced by the primary energy source selection. For example, a higher charge rate would be required for a solar array system with limited time in the light period per orbit (for earth or lunar orbit phases). Because fuel-cell system operation is relatively independent of vehicle orbiting requirements, a longer period or trickle charging could be utilized for battery recharge. Selection of the type and rating of power conditioning equipment must be established following firmer definition of the tug load requirements.

The power distribution and protection system consists of the equipment or components necessary to control and distribute conditioned electrical power to the using subsystems. Solid-state switching devices (SSSD) may be used to provide remove control (by computer) for controlling power to the using loads and to provide wire protection in event of equipment malfunctions. These devices may be used in place of conventional circuit-breakers for load control from the busses to the load equipment.

Wiring will be sized to allow a voltage line drop not to exceed the load equipment voltage and power requirements. Insulated wires will be nonflammable, nontoxic, and, where applicable, twisted or shielded to minimize electromagnetic interference and cross-talk. Typical per unit weights for various sized harness wires are listed in Table 5-7. An additional 10 percent of total harness weight is normally required for connectors.

Flat wire may find limited usage in tug applications. Although flat wire may be more flexible (bending, etc.) for harness routing and requires less volume, it is not suitable for twisting. Twisting of conductors causes cancellation of induced magnetic fields. Also, the flat-wire conductor thickness becomes excessive when shielded for general purpose wiring. Therefore, flat wiring would not be recommended for general purpose wiring but could be used for data management type signals and would alleviate possible connector space problems encountered with coaxial cables.

5.5 RECOMMENDED ELECTRICAL POWER SUBSYSTEM

A primary power system utilizing fuel cells and consumable H_2 and O_2 reactants is recommended for the tug vehicle. The primary driver for this selection is that the reactants will be available on the vehicle for the propulsion subsystem. As a result, reactant tankage and contingency (excess reactant supply) penalties can be minimized for the power system

Table 5-7. Typical Teflon Insulated Wire Weights

Wire Size Gauge No.	Type	Unit Weight	
		Lb/Ft	Kg/M
24	Single	0.0021	0.0031
24	Shielded	0.0044	0.0065
22	Single	0.0035	0.0052
22	Two-cond./twisted	0.0072	0.0107
22	Two-cond./shielded	0.028	0.0416
20	Single	0.005	0.0074
20	Two-cond./twisted	0.011	0.0164
20	Two-cond./shielded	0.036	0.0536
16	Single	0.01	0.0148
16	Two-cond./twisted	0.021	0.0312
16	Two-cond./shielded	0.055	0.0818
12	Two-cond./twisted	0.05	0.074
10	Single	0.035	0.052
8	Single	0.068	0.101
8	Two-cond./twisted	0.14	0.21
4	Two-cond./twisted	0.38	0.56

through common usage. The reactant demand varies with the tug energy requirements as shown in Figure 5-8. The reactant amounts would be insignificant compared to the propulsion subsystem requirements. Other prime considerations for this choice include:

1. Greater mission operational flexibility (less constraints in docking, etc.)

2. Greater suitability for short-term mission cycles

3. Available source of water for manned operations

4. Commonality with other NASA program developments (cost of fuel-cell development borne by EOS program)

5. Greater growth capability to meet increased power demands, because abundant reactant supply would be available

6. Fuel-cell heat rejection can be integrated with the tug thermal or environmental subsystems requirements.

The need for another primary energy source (i.e., solar array) is not considered a requirement, unless planetary missions or long dormant space operations (without reactant availability) become dominant factors.

Figure 5-10 shows the EPS subsystem configurations for unmanned and manned missions. Modular concepts will be utilized where possible to provide commonality and minimum impact for changes to meet new load requirements. Three fuel cells will provide the voltage regulation and redundancy capability for the main power subsystem for unmanned missions. Three 3-phase, 400-Hertz inverters are included for alternating current (ac) loads in the otherwise direct current (dc) power subsystem configuration. Three secondary peaking batteries will supplement the fuel cells for peak loads and emergency power requirements. Power will be distributed to decentralized power control centers throughout the vehicle with due regard for electromagnetic compatibility considerations. Solid-state switching devices and circuit-breakers will be used for switching and control where practical. Cooling of solid-state conversion equipment and batteries will be effected through mounting on cold plates. Fuel-cell cooling will be effected through the tug heat rejection radiator loops. Where practical, the EPS components will be located in the intelligence module (IM).

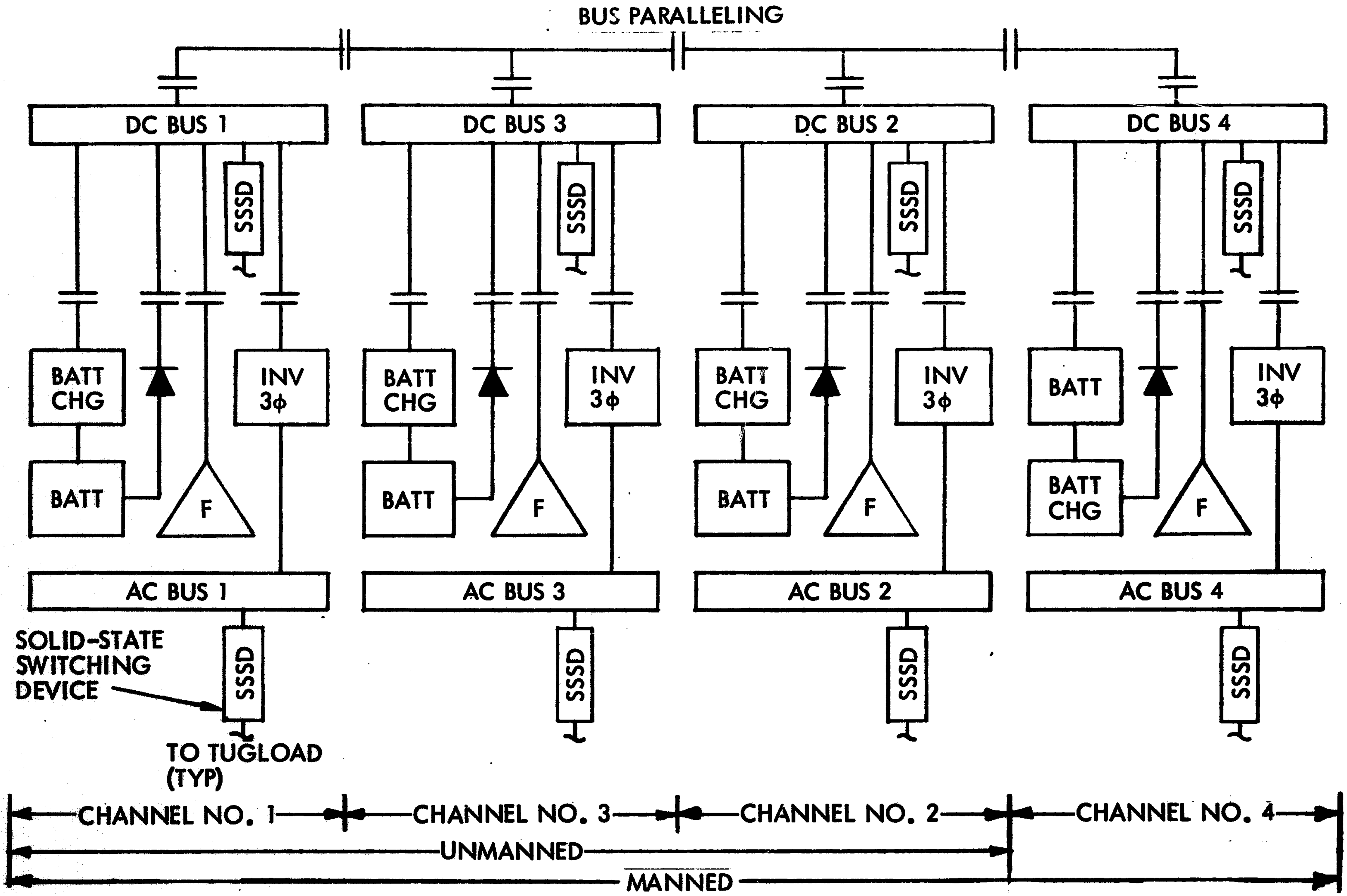


Figure 5-10. Tug EPS Schematic

For manned mission, two additional fuel cells and other components duplicating the basic unmanned EPS configuration will be utilized to meet mission requirements. The same rated type components are planned; thus, the unmanned IM EPS configuration can be converted to a manned EPS configuration by providing additional components in a modular form. Table 5-8 provides weight and volume estimates for the major components.

Table 5-8. Electrical Power Subsystem Equipment

Description	Source	Unit Characteristics						Total Characteristics					
		Weight		Volume		Power			Weight		Volume		Power
		lb	(kg)	ft^3	(m^3)	(Watts)	Code	Qty	lb	(kg)	ft^3	(m^3)	(Watts)
Crew module equipment													
Power distribution wiring	Estimate	-		-		-	-	-	50	22.6	0.50	0.014	25
Intelligence module basic equipment													
Fuel cell (1.33 kw)	EOS extrapolation	66.7	30.2	1.33	0.0376	-	-	3	200	90.7	4.00	0.113	-
Battery (26.7 AH Nicd)	EOS extrapolation	53.5	24.3	0.18	0.005	-	C, CC	3	160	72.6	0.54	0.015	-
Battery charger	EOS extrapolation	5.3	2.4	0.07	0.002	30	C, CC	3	16	7.3	0.22	0.006	90
Inverter (167 va)	EOS extrapolation	5.3	2.4	0.12	0.0034	16.7	T	3	16	7.3	0.36	0.010	50
Power controller	Estimate	4.67	2.1	0.45	0.013	23.3	T	3	14	6.4	1.33	0.0376	70
Power distribution wiring	Estimate	-		-		-	-	-	100	45.4	1.00	0.0283	20
TOTAL IM EPS basic equipment	-	-		-		-	-	-	506	207.4	7.45	0.210	230
Intelligence module added manned equipment													
Fuel cell (1.33 kw)	EOS extrapolation	66.7	30.2	1.33	0.0376	-	-	1	66.7	30.2	1.33	0.0376	-
Inverter (167 va)	EOS extrapolation	5.3		0.12	0.0034	16.7	C, CC	1	5.3	2.4	0.12	0.0034	16.7
Power controller	Estimate	4.67	2.1	0.45	0.0127	23.3	T	1	4.7	2.1	0.45	0.0127	23.3
Power distribution wiring	Estimate	-		-		-	-	-	50	22.6	0.50	0.0141	0
Total IM EPS added manned equipment	-	-		-		-	-	-	126.7	57.5	2.40	0.0678	40

6.0 ACTIVE THERMAL CONTROL SUBSYSTEM

6.1 REQUIREMENTS

The active thermal control subsystem transports heat from the interior to the exterior of the vehicle, complementing passive thermal control provisions, to maintain thermal equilibrium between preset bounds. Ideally, the passive system minimizes solar absorptivity, while maximizing infrared emissivity, to reduce the size of the active system. Both heat sinks and sources are used by the active system to maintain component temperatures within required limits.

The structure of part of the tug is dominated by thermal insulation requirements of the cryogenic propellant. Elsewhere, the vehicle envelope is rather loosely defined. The major internal heat producers, electronics, power sources, and crew size, vary with mission class. Wide variations in mission duration and thermal environment will be encountered by the tug. Under these circumstances, requirements must be supplemented by assumption in selection of a subsystem approach from the candidates. The requirements and assumptions are listed in Table 6-1.

The environmental control and life support subsystem (EC/LSS) will provide cabin atmosphere thermal control by fluid loop, but will transfer heat via a heat exchanger to the active thermal control subsystem for removal.

Thermal control requirements imposed on mission operations will be minimized. This includes special vehicle orientation requirements such as the barbecue rotation mode, orienting an area of the vehicle surface toward deep space to establish an efficient heat sink, and the practice of presenting a minimum, highly insulated part of the vehicle to the sun for prolonged periods.

6.2 CANDIDATE EQUIPMENT

All candidate active thermal control methods either result in heat loss by radiation or by mass expulsion. Each type must have a circulating system that removes heat from generators. Radiation systems require a suitable heat sink, such as deep space or fluids expended for other purposes. Typical radiation systems include space radiators of the Apollo type, heat pipes, propellant heat sinks, and propellant boiloff heat sinks. Mass expulsion systems include liquid boilers and ablative surfaces. The

Table 6-1. Active Thermal Control Subsystem Requirements

1.	Space mission crew size	0 - 6 men
2.	Lunar landing mission crew size	0 - 4 men
3.	Metabolic heat production rate	430 - 496 Btu/man-hour (126-145 W/man-hour)
4.	Space mission duration	$\leq$7 days
5.	Lunar landing mission duration	$\leq$45 days
6.	Space quiescent period	$\leq$180 days
7.	Lunar surface quiescent period	$\leq$30 days
8.	Vehicle total use span	$\leq$3 years
9.	Fuel cell heat production rate	0.667 W_{heat}/W_e
10.	Battery heat production rate	0.25 W_{heat}/W_e

candidates considered reasonable for tug include the Apollo-type radiators, a propellant boiloff heat sink, and a liquid boiler.

The circulation system may include either liquid or gas as a working fluid. A forced-gas system has the advantages of less costly manufacturing processes and may be more easily maintained in space; however, it requires more weight, power, and volume (Reference 6-1). The liquid system uses cold-plate contacts to the electronic components. The cold plates are small reservoirs in the circulation fluid lines and present a large area in common with the electronics package. Special heat-conductive grease is applied at the package/cold plate interface to facilitate heat flow.

Many of these candidates were evaluated in a trade study described in Appendix B.

6.3 RECOMMENDATIONS

Apollo-type radiators were selected as the method of heat rejection for tug on the basis of versatility and simplicity. Radiators weigh approximately 24.6 pounds per Btu/hr (38.0 Kg/w) of heat to be rejected. Water has a heat sink capacity of approximately 0.001 pound per Btu (0.001545 kg/w-hr). A vehicle generating 1 kilowatt of heat during a seven-day mission would require 84 pounds (38 kg) of radiators or 573 pounds (260 kgs) of water. Very little of the water requirement could be generated by fuel cell power sources since they produce approximately 0.00015 pounds of water per Btu (0.000232 kg/W-hr) of combined heat and electrical power. Hence, the choice of radiators for tug was clear.

The liquid cold-plate method of removing heat from electronics equipment was chosen over the forced-gas approach on the basis of a trade study described in Reference 6-1. Study results are shown in Table 6-2 and were derived for a 10-psia (7031-kg/m^2) environment. A lower pressure, as recommended for tug, would degrade the heat transfer coefficient and increase the weight, volume, and power requirements for the forced-gas system.

The arrangement of radiators on the vehicle is particularly important since in space the radiator works more effectively when pointing away from the sun. It is impractical to mount the radiators on the forward end of the vehicle, since no module is forward for all missions. Furthermore, this configuration would impose orientation requirements relative to the sun on the vehicle. Considering these facts, and recognizing the need for redundancy, the suggested configuration is four panels mounted equidistant around the periphery of the vehicle. Any two of the panels will meet the peak cooling requirement when one, or its opposite, faces the sun. Although peripheral mounting is probably the best radiator arrangement for space missions, it

Table 6-2. Comparison of Cold Plate versus Forced-Gas Requirements

Parameter	Weight		Volume		Power
	lb/watt	kg/watt	in.^{3}/watt	cm^3/watt	watts/watt
Cold plate	0.0035	0.0016	0.38	6.2	0.00063
Forced gas	0.048	0.022	3.8	62.	0.16

is unusable for daytime conditions on the lunar surface. A radiator seeing the lunar surface must itself have a surface temperature above 135 F (300 K) in order to reject any amount of heat, regardless of the relative sun angle.

It is therefore necessary to provide other radiators, which do not see the lunar surface, for the tug lander. The landing radiators could be in the form of a kit to be attached in space prior to the landing mission. They would use the available space atop the tug for rigid mounting and, in the case of an excess area, would require additional deployable panels. Because these radiators now point toward the sun in the worst case and are consequently inefficient, it is worthwhile to investigate the feasibility of heat pumps.

A heat pump is a single- or multiple-loop refrigeration system that raises the radiator surface temperature, permitting more efficient heat rejection. Unfortunately, heat pumps require relatively large amounts of electrical power to operate. The trade study described in Appendix B shows that the minimum usable radiator surface temperature with a heat pump system is the optimum, considering both weight and electrical power. Under these conditions, the heat pump transfers internal heat from the circulation loops to the top of the vehicle. It also provides a source of redundancy if a panel leaks, fails to deploy, or is otherwise unusuable. In these cases the vehicle would be cooled by raising the heat pump power to raise the surface temperature of the remaining top radiator panels.

During lunar landing missions, significant amounts of heat may be removed by utilizing the high specific heat of hydrogen boiloff gas. Approximately 1500 Btu/lb (3.5×10^6 J/kg) of waste heat may be absorbed at the predicted temperatures. It appears that the boiloff rate closely follows sun angle, peaking at high noon. If boiloff gas is used during this period, the most constraining requirement for radiators may be reduced. Assuming 500 pounds (226 kgs) of hydrogen is lost by boiloff per month on the lunar surface, and that the rate curve approximates a cosine, then the peak boiloff rate is 2.34 lb/hr (1.06 kg/hr). Effectively, therefore, the radiator heat rejection requirement is reduced by the peak boiloff rate times the best absorption capacity of the hydrogen, amounting to 3450 Btu/hr (1 kw).

The trade study reported in Appendix B provided sufficient data to establish parametric weight, volume, and power models. For the data to be used, however, the total heat load of the active system must be assessed. The following paragraphs describe the effort.

Coupling between the electrical power source and the cooling system exists because power generated to drive the cooling system also produces heat. Expressed mathematically,

$$P_T = P_S + P_C \tag{6-1}$$

where the cooling system power required is

$$P_C = C \left[P_T + H_S + H_M M\right] \tag{6-2}$$

and PT = total power generated

P_S = total subsystem power demand, including conversion and distribution equipment, but not including cooling power

P_C = cooling equipment power demand

C = constant of proportionality, containing effects of pump efficiency, temperature differences, head loss in the lines, etc;

H_S = heat of reaction in the electrical power source

H_M = crew-generated heat

M = the crew size.

All heat generated by the crew must be transported from the EC/LSS thermal loop to the radiators by the ATC. Values for crew and fuel cell heat production rates are listed in Table 6-1. A typical value of 136 watts is recommended for the crew heat production rate. Peaking battery heat production is negligible, since its rate is much lower than that of the fuel cell and occurs over relatively short time spans. The constant C is evaluated at 0.040956 by comparison with the Apollo system, which rejects heat at 5000 Btu/hr (1468 watts) using 60 watts of electrical power.

The equations become more complicated when modularity is considered. Power sources are located in the IM, while the CM contains the crew. In order for the IM penalty for manned missions to be minimized, separate cooling systems are assumed for the two modules. All of these considerations influence the cooling system power. The six separate equations are

$$P_{T_{IM}} = P_{S_{IM}} + P_{C_{IM}} \tag{6-3}$$

$$P_{T_{CM}} = P_{S_{CM}} + P_{C_{CM}} \tag{6-4}$$

$$P_{C_{IM}} = C\,(P_{T_{IM}} + H_S) \tag{6-5}$$

$$P_{C_{CM}} = C\,(P_{T_{CM}} + H_M M) \tag{6-6}$$

$$H_S = NP_T \tag{6-7}$$

$$P_T = P_{T_{IM}} + P_{T_{CM}} \tag{6-8}$$

where N is the fuel cell heat production rate. Applicable modules are denoted by subscripts. Solving for the intelligence module and crew module cooling system power,

$$P_{C_{IM}} = A_1 P_{S_{IM}} + A_2 P_{S_{CM}} + A_3 H_M M \tag{6-9}$$

$$P_{C_{CM}} = A_4 (P_{S_{CM}} + H_M M) \tag{6-10}$$

where

$$A_1 = \frac{C(1+N)}{1-C(1+N)} \tag{6-11}$$

$$A_2 = \frac{CN}{(1-C)\,(1-C-CN)} \tag{6-12}$$

$$A_3 = A_2 C \tag{6-13}$$

$$A_4 = \frac{C}{1-C} \tag{6-14}$$

Evaluating A_1 through A_4 with the values discussed, the final forms of the coolant circulation system electrical power equations are

$$P_{C_{IM}} = 0.07328 P_{S_{IM}} + 0.03057 P_{S_{CM}} + 0.1703\ M \tag{6-15}$$

$$P_{C_{CM}} = 0.04271 P_{S_{CM}} + 5.8079 M \tag{6-16}$$

where the units of all variables are watts. The subsystems power terms (P_S) are, strictly speaking, variable during the mission, but average power values give good approximations.

Once the circulation system power is established, the weight may be estimated by again scaling from Apollo data. The 60-watt, 185-pound (84-kg) system yields a ratio of 3.083 lb/watt (1.40 kg/watt) of operating power. The system weight-to-volume ratio is assumed to be 35 lb/ft^3 (560 kg/m^3).

For the space mission radiators to be sized, surface thermal coating, age, sun angle, and mounting provisions must be considered. These topics are fully discussed in Appendix B. Assuming a solar absorptivity of 0.3, an infrared emissivity of 0.9, an average sun angle of 8 degrees, and an average radiator surface temperature of 80 F (300 K), the scaling factor is 0.0277 ft^2/watt (0.00257 m^2/w) of heat transported. These values correspond to a segmented radiator, peripherally mounted, and 3500 sun hours old. Again, expressing the function mathematically,

$$A_S = K(P_T + H_S + H_M M) \tag{6-17}$$

where A_S is the radiator area,

K is the scaling factor, and the remaining terms are as defined for Equations (6-1) and (6-2). Separate equations may be shown for IM and CM:

$$A_{S_{IM}} = K[P_{T_{IM}} + H_S] \tag{6-18}$$

$$A_{S_{CM}} = K[P_{T_{CM}} + H_M M] \tag{6-19}$$

Solving for the IM and CM radiator areas with Equations (6-3), (6-4), (6-7), and (6-8),

$$A_{S_{IM}} = K(1+N)(P_{S_{IM}} + P_{C_{IM}}) + KN(P_{S_{CM}} + P_{C_{CM}}) \tag{6-20}$$

$$A_{S_{CM}} = K(P_{S_{CM}} + P_{C_{CM}}) + KH_M M \tag{6-21}$$

These equations may be further simplified by substituting for the cooling circulation system power, using Equations (6-15) and (6-16). Carrying out the substitution and evaluating coefficients,

$$A_{S_{IM}} = 0.04964\, P_{S_{IM}} + 0.02071\, P_{S_{CM}} + 0.1154M \qquad (6\text{-}22)$$

$$A_{S_{CM}} = 0.02893\, P_{S_{CM}} + 3.9349M \qquad (6\text{-}23)$$

where the units of power are in watts and area in square feet. The weight of this type of radiator is 0.2 lb/ft^2 (0.975 kg/m^2).

Lunar landing missions require a heat pump system, as previously recommended. The system would use a top-mounted, add-on assembly including a vapor compression refrigeration system and additional radiators. At 80 F (300 K), the heat pump would require an additional 0.0271 watts of operating electrical power per watt of heat transported. The hydrogen boil-off gas heat exchanger would remove approximately one kilowatt of heat from the system, however. Assuming the heat pump equipment is self-cooled, the equations are

$$P_R = R\,(P_T + H_S + H_M M - 1000) \qquad (6\text{-}24)$$

$$P_C = C\,(P_T - P_R + H_S + H_M M - 1000) \qquad (6\text{-}25)$$

$$P_T + P_R + P_C + P_S \qquad (6\text{-}26)$$

$$H_S + NP_T \qquad (6\text{-}27)$$

where P_R is the refrigeration power, and

R is the constant of proportionality for the heat pump system. Note that the heat pump is assumed to use the entire circulation system.

A procedure is followed similar to that for the circulation system power evaluation, and the heat pump equation is

$$P_R = 0.05106\, P_S + 4.166\, M - 30.6318 \qquad (6\text{-}28)$$

The heat pump equipment adds to the system weight 0.04 pounds (0.0181 kg) for every watt of heat or 1.48 lb/watt (0.67 kg/watt) of operating power. If mounted inside the tug, the equipment contributes 35 lb/ft^3 (560 kg/m^3).

Attached to the top of the tug horizontally, to avoid seeing the lunar surface, the lunar landing radiator must operate in direct sunlight. Here it is assumed the radiator is new and has a solar absorptivity of 0.018, yielding a value of 0.0683 ft^2 (0.00633 m^2) per watt of heat transported. The area equation is

$$A_R = L\,(P_T + H_S + H_M M - 1000) \qquad (6\text{-}29)$$

where A_R is the lunar landing radiator area in square feet,

L is the scaling factor, and the remaining terms were defined previously.

The evaluated top-mounted radiator equation is

$$A_R = 0.1259\,P_S + 28.027\,M - 206.079 \qquad (6\text{-}30)$$

Since the radiator may be attached in space it must have more supporting structure than the ground-installed space mission radiators. A value of 0.9 lb/ft^2 (4.4 kg/m^2) was used for the panel section which fits directly on the top of the vehicle. This panel may include up to 139 ft^2 (12.9 M^2) for a 15-foot-diameter module. If the radiator requires more area than the top of the vehicle permits, then erectable leaves may be added. These are estimated to weigh 1.9 lb/ft^2 (9.3 kg/m^2), including the erection mechanism.

The weight and volume of the hydrogen boiloff heat exchanger was estimated to be similar to an Apollo EC/LSS liquid-to-gas heat exchanger, which weighs 38 pounds (17.2 kg) and is estimated at 0.25 ft^3 (0.0071 M^3).

Table 6-3 summarizes the parametric active thermal control equipment data discussed. Schematically, the various components are related as shown in Figure 6-1.

The parametric data may be used to establish characteristics for specific missions. Once the power demand is established for all user subsystems, and the conversion and distribution equipment loads are defined, the basic requirements for ATC weight, volume, and power may be found from the table. These basic requirements must then be multiplied by the desired redundancy factors. Since the internal cooling equipment has a somewhat flexible capability, it was assumed that redundancy could be obtained by multiple systems, each a fraction of the total requirement. Thus the circulation system weight, power, and volume is not greatly affected by redundancy. Radiators may be designed to contain redundant plumbing over the same area to afford a measure of redundancy.

Table 6-3. Active Thermal Control Subsystem Parametric Data

Function	Mission	Radiator Area ft^2 (m^2)	Radiator Area ① ft^2/P$_{SIM}$ (m^2/P$_{SIM}$)	Radiator Area ② ft^2/P$_{SCM}$ (m^2/P$_{SCM}$)	Radiator Area ③ ft^2/P$_S$ (m^2/P$_S$)	Radiator Area ft^2/M (m^2/M)	Weight lb (kg)	Weight lb/ft^2 (kg/m^2)	Weight lb/W (kg/W)	Volume ft^3 (m^3)	Volume ft^3/lb (m^3/kg)	Power W	Power W/P$_{SIM}$	Power W/P$_{SCM}$	Power W/P$_S$	Power W/M
Coolant circulation system (IM)	All	-	-	-	-	-	-	-	3.083 (1.40)	-	0.0286 (0.00178	-	0.07328	0.03057	-	0.1703
Coolant circulation system (CM)	All manned	-	-	-	-	-	-	-	3.083 (1.40)	-	0.0286 (0.00178)	-	-	0.04271	-	5.8079
Fixed space radiator (IM)	All	-	0.04964 (0.00461)	0.02071 (0.00192)	-	0.1154 (0.01075)	-	0.2 (0.975)	-	-	-	-	-	-	-	-
Fixed space radiator (CM)	All manned	-	-	0.02893 (0.00192)	-	3.9349 (0.365)	-	0.2 (0.975)	-	-	-	-	-	-	-	-
Heat pump refrigeration system	Lunar Landing	-	-	-	-	-	-	-	1.48 (0.67)	-	0.0286 (0.00178)	-30.6	-	-	0.05106	4.166
Detachable space radiator (Area ≤139 ft^2, ≤12.9 m^2)	Lunar Landing	-206.1 (-19.1)	-	-	0.1259 (0.0117)	28.03 (2.61)	-	0.9 (4.4)	-	-	-	-	-	-	-	-
Erectable space radiator (Area ≥139 ft^2, ≥12.9 m^2)	Lunar Landing	-206.1 (-19.1)	-	-	0.1259 (0.0117)	28.03 (2.61)	-	1.9 (9.3)	-	-	-	-	-	-	-	-
Hydrogen boiloff gas heat exchanger	Lunar Landing	-	-	-	-	-	38.0 (17.2)	-	-	0.25 (0.0071)	-	-	-	-	-	-

Notes: ①, ② $P_{SIM} + P_{SCM} = P_S$
③ P_S is the total electrical power demand, including conversion and distribution equipment, but not including cooling power

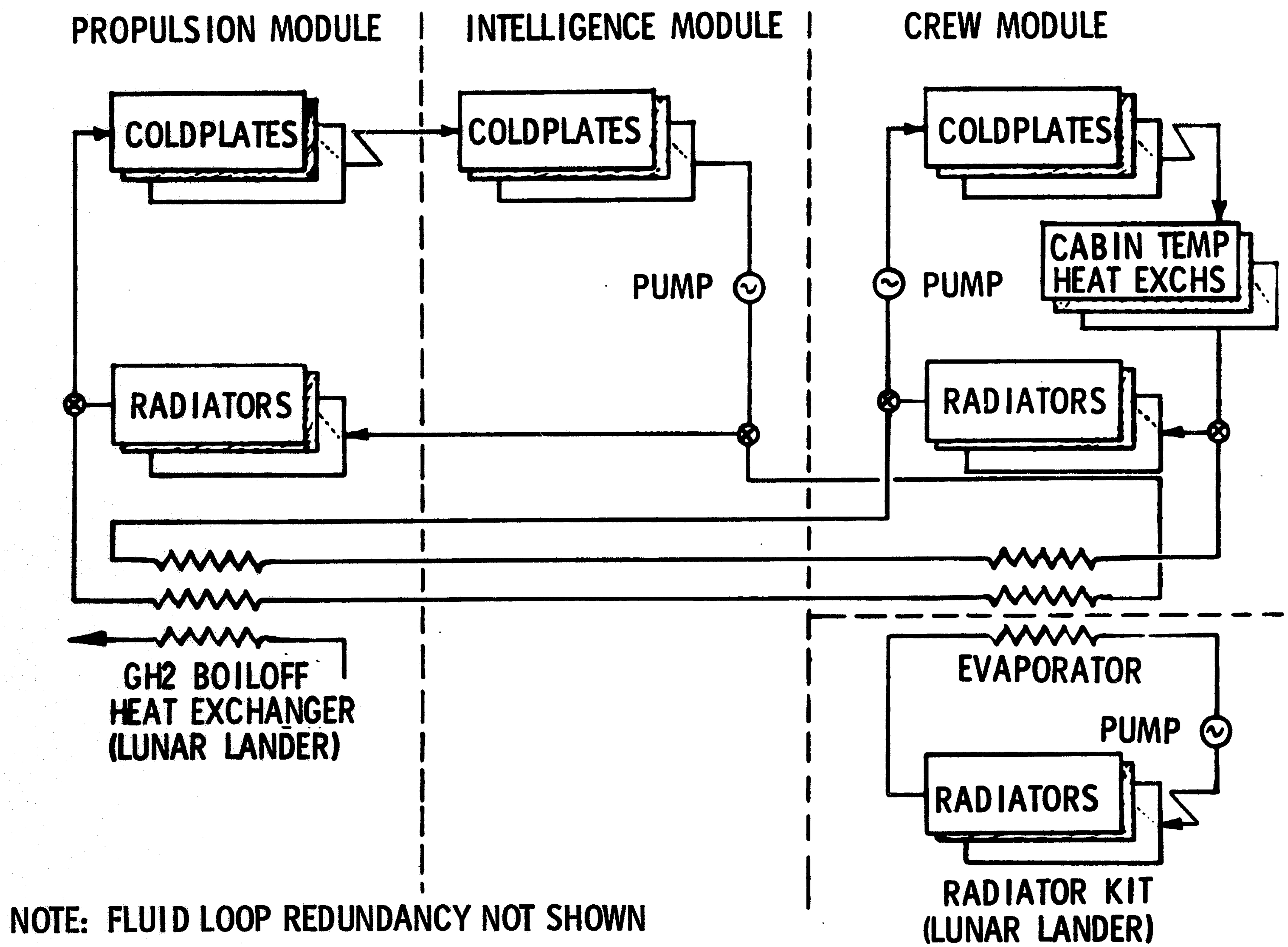


Figure 6-1. Active Thermal Control Schematic Diagram

Even so, the total area derived from Table 6-3 is doubled to ensure safety. A sketch of the configurations is shown in Figure 6-2. ATC equipment characteristics are presented in Table 6-4. Note that the intelligence module radiators are located on the propulsion module.

6.4 CONCLUSIONS

Data, both parametric and fixed, have been generated for a selected, conventional, active thermal control subsystem. The configuration shown does not appreciably affect overall vehicle operation during space missions. Lunar landing missions require that significant additions be made to the system to meet requirements in severe lunar surface environment.

Future studies should consider other active thermal control systems. These may include:

1. Boiler use of excess water generated by fuel cells
2. Integrating space and lunar landing mission radiators
3. Heat pipes for their potential weight and power savings
4. Passive cooling methods for their potential weight and power savings
5. Louver panel radiator
6. Absorption refrigeration cycle system

The last item received only preliminary treatment during the present study, as reported in Appendix B. A comparison was made between the absorption cycle and the vapor compression cycle systems, as used during lunar surface noon. Only ammonia in a water absorber was considered as a refrigerant, and a constant fin temperature for the radiator was assumed. Future studies should include consideration of other refrigerants.

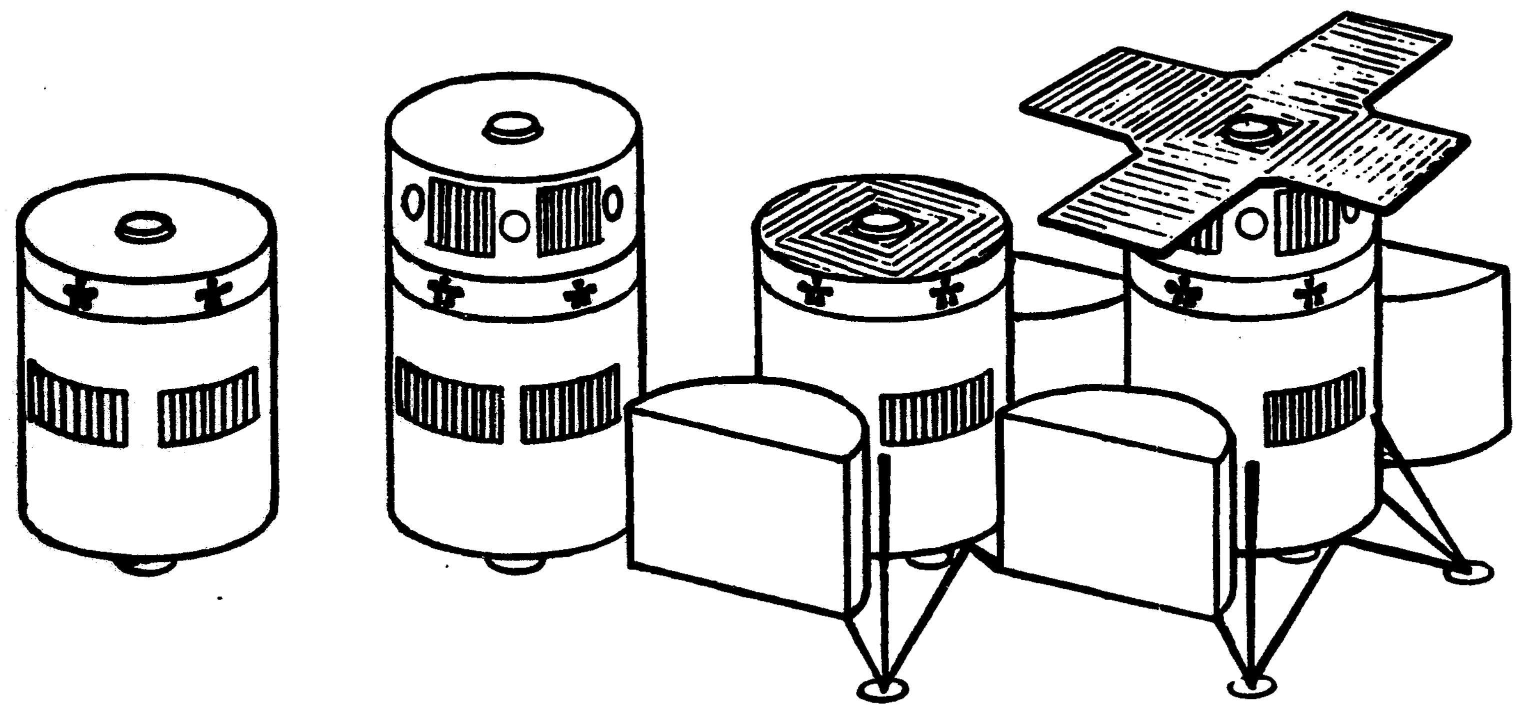


Figure 6-2. Active Thermal Control Subsystem Configurations

Table 6-4. Active Thermal Control Subsystem Equipment

Description	Source	Area ft^2	Area (m^2)	Weight lb	Weight (kg)	Volume ft^3	Volume (m^3)	Pwr (w)
Intelligence module basic equipment								
Coolant system: cold plates, pumps, dual plumbing	Apollo est	-		184	83.5	5.26	0.1489	61.6
Intelligence module added manned equipment (6 men)								
Coolant system: cold plates, pumps, dual plumbing	Apollo est	-		141	64.8	4.04	0.1144	43.4
Propulsion module basic equipment								
Space radiator (dual plumbing)	Apollo est	80.6	7.48	16.2	7.3	0		0
Propulsion module added manned equipment (6 men)								
Space radiator (dual plumbing)	Apollo est	68	6.32	13.4	6.1	0		0
Propulsion module added lunar landing equipment								
GH_2 boiloff heat exchanger (dual plumbing)	Apollo est	-		38	17.2	0.25	0.0070	0
Crew module equipment (6 men)								
Coolant system (EC/LSS separate): cold plates, pumps, dual plumbing	Apollo est	-		168	76.2	4.83	0.1368	88
Space radiator (dual plumbing)	Apollo est	118.6	11.01	23.7	10.8	0		0
Lunar landing radiator kit unmanned equipment								
Radiator panel and erection system (dual plumbing)		120	11.15	54	24.5	0		0
Coolant system: condenser, heat exchanger, plumbing		-		35.4	16.1	1.00	0.0283	24
Lunar landing radiator kit manned equipment (4 men)								
Radiator panel and erection system (dual plumbing)		228	21.28	294	133.4	0		0
Coolant system: condenser, heat exchanger, plumbing		-		137	62.1	3.41	0.0965	93

7.0 AUXILIARY CONTROL SUBSYSTEM

7.1 REQUIREMENTS

The tug auxiliary control system (ACS) is part of a stabilization system that complements the vehicle main propulsion. Upon command it holds the vehicle at a fixed inertial attitude or angular rate, maneuvers the vehicle to a desired attitude, provides the force necessary for separation from, or docking with, another vehicle, and damps disturbance torques. It may also be required to steer the vehicle during main propulsion operation. For the tug, the stabilization system may be commanded either manually or automatically. In either case, the system will include both attitude and rate feedback loops. In the conceptual phase, it is not necessary to design the stabilization system. However, estimates of jet size, quantity, configuration, and propellant usage are needed. Where possible, the estimates are derived parametrically since a broad spectrum of missions are involved.

Several questions are addressed in this section, beginning with ACS jet location on the vehicle and the number of jets to be used. Next, jet thrust levels are derived from consideration of operational requirements. Estimates of ACS propellant usage are obtained by examining individual mission elements. Finally, the jet thrust level time history and the resulting propellant usage are discussed. Simplified attitude and translation control requirements are defined to specify thrust levels within the range of Apollo spacecraft requirements.

Present concepts of the tug indicate that it will take roughly a cylindrical form consisting of specialized modules that separately enclose one or more of the major systems: propellant, oxidizer, main propulsion engines, cargo, crew, and supporting subsystems. Since not all of the modules will be included on every mission, it is desirable to concentrate the ACS on a single module. The moment arm for a jet couple is thus approximately the diameter of the cylinder. The distance between the jet station and the vehicle c.g. is an alternative moment arm, but cannot be used to provide a pure moment. A summary of the ACS requirements is given in Table 7-1.

Both forces and moments are required of the ACS to provide translational and rotational vehicle control. The most convenient arrangement is to have the jets located in clusters of four each at 90-degree intervals around the circumference of the cylinder and oriented parallel to standard body axes as in the case of the Apollo service module.

Table 7-1. Auxiliary Control Subsystem Requirements and Drivers

Requirements

1. Jet external envelope must fit within 15-foot-diameter (4.56-m) EOS cargo bay
2. ACS will provide 6-degree-of-freedom translation and rotational acceleration levels appropriate for passive or active docking and separation.
3. ACS will provide roll control during a single main-engine propulsive maneuver.
4. ACS will provide adequate rotational acceleration response about any axis for orientation maneuvers and for all disturbance damping.
5. ACS will provide adequate longitudinal acceleration for vernier midcourse velocity correction.
6. ACS will provide adequate attitude hold precision for all sensor pointing requirements.
7. No credible single points of failure without justification.
8. Ten-mission reusable life, three-year total life.

Drivers

1. Common propellant tankage for ACS and main propulsion.
2. Jets fail off only.
3. Independent translation and rotation capability for docking and separation.
4. Fail-safe condition must permit docking (tug passive).
5. Unlimited maximum surge demand for two jet operation.
6. Degraded response allowed for failure conditions.
7. Not sensitive to vehicle c.g. position.

Four clusters of four jets each provide the capability of full control with a single jet, or even an entire cluster, inoperable. However, there are other combinations of two jets failed that preclude translation in one direction.

It must be assumed at this point that the ACS will be designed such that jets only fail off. Without this design, serious loss of propellant would occur with a jet failure, even if control could be maintained.

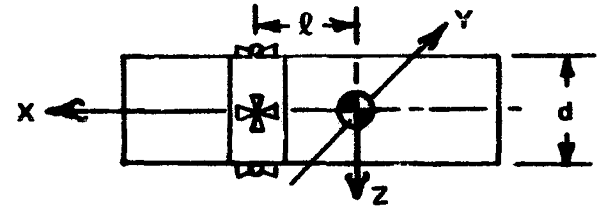
7.2 JET CONFIGURATION

A preliminary description of normal and failure mode operation is shown in Table 7-2 for a 16-jet system. Normal operation uses two to four jets to obtain pure rotations and translations. Lateral or vertical translation disturbs yaw or pitch control appreciably when the jet center is greater than a diameter from the c. g. If one jet is inoperable, a logic change to use the alternate set of roll jets is necessary to maintain full control. Lateral or vertical translation authority is approximately one-third of normal with one jet out. When two jets are inoperable, no lateral or vertical translation control is left, for the worst case, and the active docking capability is lost. A logic change is mandatory to maintain pitch or yaw stabilization, but it requires that the jet center not be coincident with the c. g. Other logic changes are recommended as shown. When three jets fail, no further complications arise.

If the 16-jet system is used on the tug, there are two requirements: The ACS must be designed so that jets cannot fail stuck on, within the multiple failure criteria, and the ACS must be located away from the c. g. to assure safe operation after the second jet failure.

The second requirement probably defeats the modularity and multi-mission tug concepts. Cargo, under certain conditions, may be added to either end of the vehicle. The IM, which should contain the jet system, may be anywhere in the module stack above the PM. The IM may also be required to operate as a free module. All three of these conditions may include times when the c. g. and jet station coincide. Therefore, the 16-jet complex at a single body station will not suffice.

In an attempt to avoid a shotgun approach to the analysis of other configurations, the following method is used. First, the question of how many jets at a single station is necessary will be attacked. The objective is to determine minimum complexity and number of jets to reach fail operational - fail operational - fail safe (FO-FO-FS) and fail operational, fail safe (FO-FS). In this case, FS means the tug is stable, but may not be capable of active docking. The rescue vehicle, however, could approach



NUMBER OF JETS FAILED	ROLL	PITCH OR YAW	X TRANSLATION	Y OR Z TRANSLATION	COMMENTS
NORMAL OPERATION					OPERATIONAL
ONE JET FAILED	(Y OR Z TRANS)		(LOGIC CHANGE)		OPERATIONAL WITH LOGIC CHANGE
TWO JETS FAILED	(LOGIC CHANGE)	(LOGIC CHANGE, Y OR Z TRANS, $\|\ell\| > 0$)	(LOGIC CHANGE)	(EITHER Y OR Z UNCONTROLLED)	LOSS OF TRANSLATION CONTROL IN ONE AXIS, FAIL-SAFE (ACTIVE DOCKING CAPABILITY LOST) CONDITIONAL ON $\|\ell\| > 0$
THREE JETS FAILED	(Y OR Z TRANS)	(LOGIC CHANGE, Y OR Z TRANS, $\|\ell\| > 0$)	($\|\ell\| > 0$)	(EITHER Y OR Z UNCONTROLLED)	FAIL-SAFE CONDITIONAL ON $\|\ell\| > 0$

Table 7-2. Auxiliary Control Jet Failure Operation

and dock with the FS tug and thereafter command maneuvers. It is important at this point to realize that one step past FS means a divergent tumbling rate, which probably precludes any rescue or salvage.

Table 7-3 shows the independent configurations necessary for rotation and translation with a single jet station at an arbitrary distance from the c.g. and no jets failed. All foreseen possibilities are shown for the stated conditions. The concepts, labeled A, B, and C are combined in Table 7-4 to show complete sets. These sets, however, suffer from weak failure tolerances, as is indicated in the right hand four columns. For example, combination AAA, which is identical to AAC, would be in the fail-safe condition after a minimum of one jet failure. Thus, the next step is to selectively add jets to increase the failure tolerance.

The FO-FS level of redundancy is reached under the conditions shown in Table 7-5 and the FO-FO-FS level of redundancy in Table 7-6. A more detailed analysis of the 20-jet pentad shown in Table 7-5 is given in Table 7-7, where it is shown that the level of redundancy is FO-FS only under a combination of worst case conditions. For all other cases, FO-FO-FS or better will be obtained.

All of the single-jet station configurations suffer from a disadvantage. Lateral translation, when the jets are away from the vehicle c.g., must be accompanied by countertorquing in pitch or yaw. The result is degraded response.

The multiple-jet-station cases are more difficult to analyze in that literally hundreds of configurations may be postulated that use 16 to 32 jet sets. Two generalized examples have been selected for analysis, however. A FO-FS generalized concept is shown in Figure 7-1, in which the two jet stations may be widely separated or they may be both on the IM. Sixteen jets are necessary if some of the jets are canted outward to serve double duty. I_{sp} losses are incurred by this approach. Figure 7-2 shows a generalized concept that uses 24 jets, with none of the jets canted.

Multiple-jet-station configurations have two distinct advantages over those of the single-jet station. They are independent of c.g. location and they do not suffer from degraded lateral translation response.

The 20-jet pentad configuration shown in Table 7-5 is recommended for the following reasons:

1. The minimum, worst case redundancy is FO-FS.

2. The configuration does not use radially pointing jets, which compound the vehicle outer envelope problem.

Table 7-3. ACS Choices Based on Minimum Requirements and a Single Jet Station

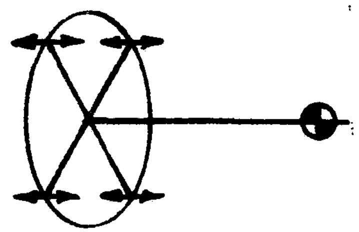
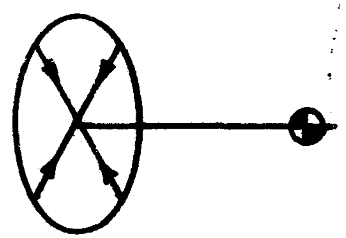
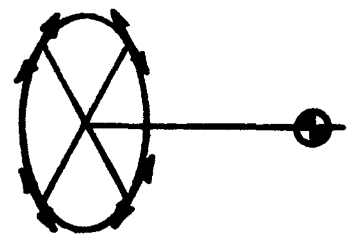
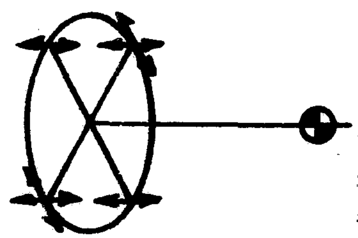
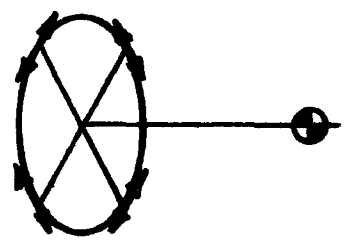
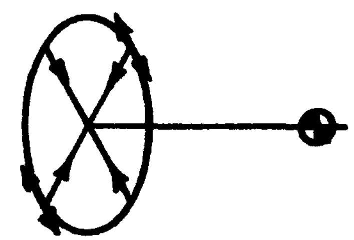
CONCEPT	A	B	C
LONGITUDINAL TRANSLATION			
TWO-AXIS LATERAL TRANSLATION			
THREE-AXIS ROTATION			

Table 7-4. Single Jet Station ACS Combinations

- WORST CASE FAILURE COMBINATIONS
- ALL FAILURES TO ZERO THRUST

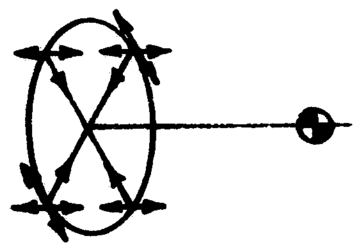
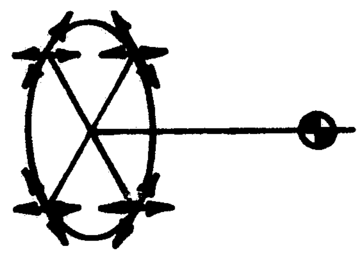
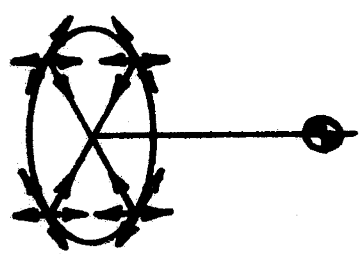
COMBINATION	GEOMETRY	NUMBER OF JETS	C.G.	FAILURES TO FAIL SAFE LONG	LAT	PITCH-YAW	ROLL
AAA, AAC		16	C.G. = Jet Station	2 - 4	1	1	1
			C.G. ≠ Jet Station	3	1	2	1
ABA, ABB		16	C.G. = Jet Station	2 - 4	2	1	3
			C.G. ≠ Jet Station	3	2	3	3
AAB, ABC		20	C.G. = Jet Station	2 - 4	3	1	3
			C.G. ≠ Jet Station	3	2	4	3

Table 7-5. Redundancy Requirements for Single Jet Station ACS
With FO-FS Redundancy

- FAIL-OPERATIONAL, FAIL-SAFE REQUIREMENT
- ALL FAILURES TO ZERO THRUST

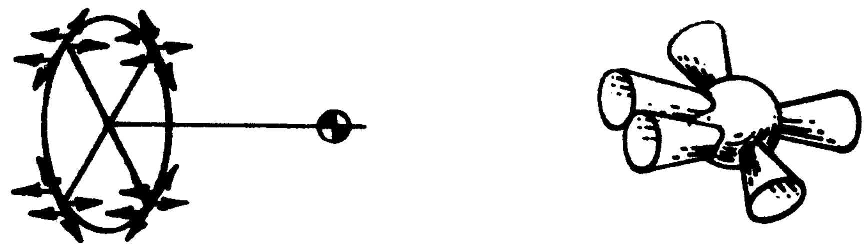
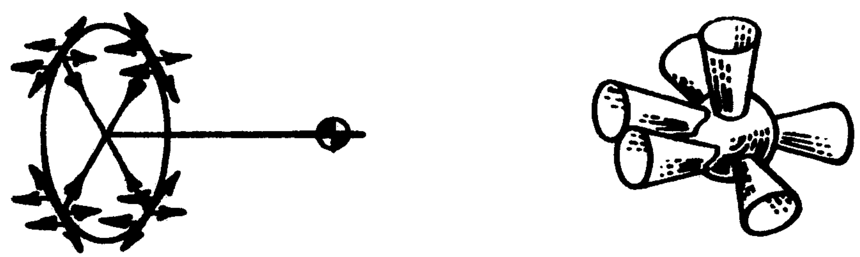
CONDITION	NUMBER OF JETS REQUIRED	GEOMETRY
ABA	20	
AAA, AAB	24	

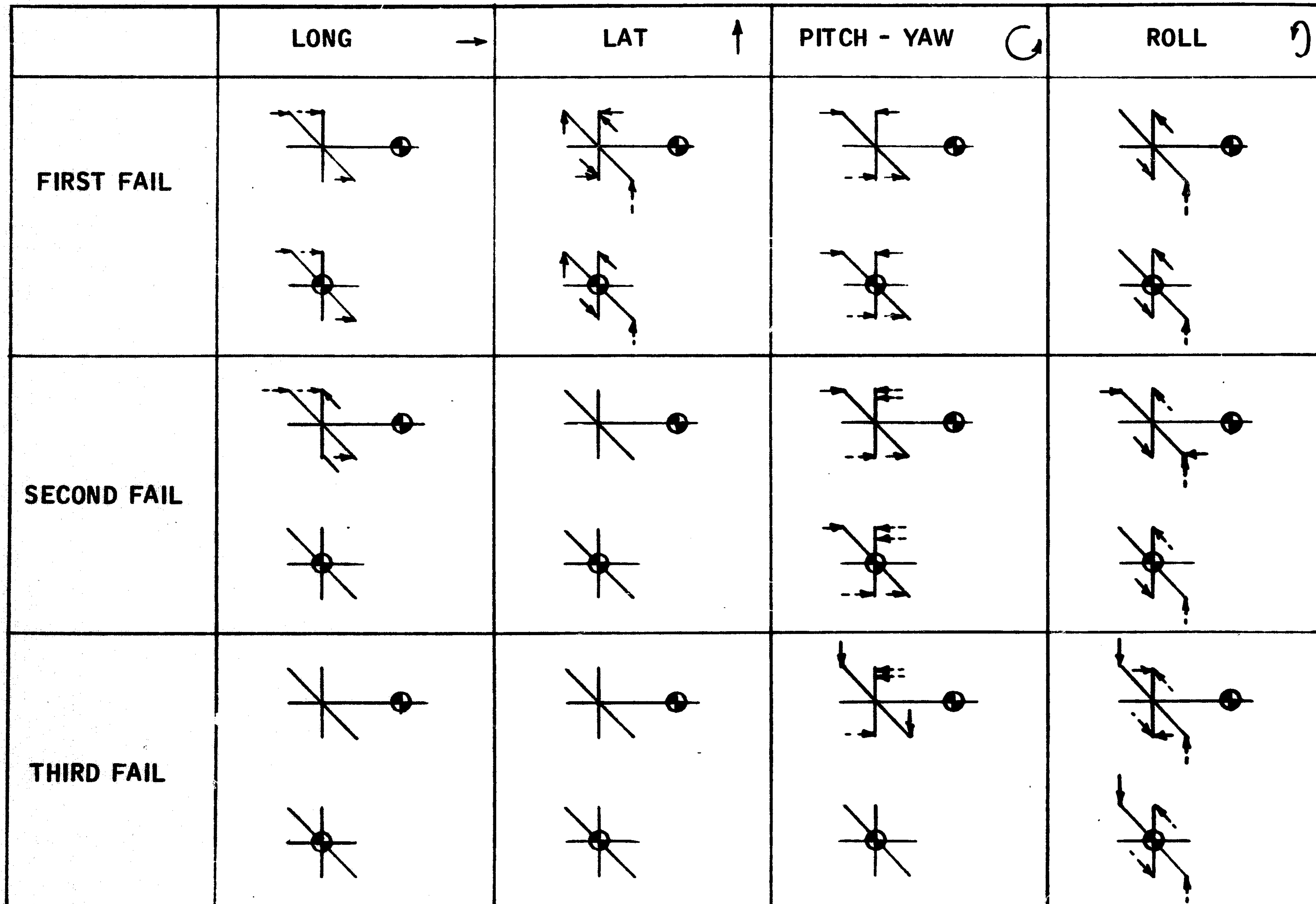
Table 7-6. Redundancy Requirements for Single Jet Station ACS With FO-FO-FS Redundancy

CONDITION	NUMBER OF JETS REQ'D	GEOMETRY
C.G. AT JET STATION	28	
C.G. NOT AT JET STATION	20	

Table 7-7. Failure Mode Analysis of a 20-Jet, Single-Station Configuration

→ INDICATES JETS USED

--→ INDICATES FAILED JETS



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NO FAILURES: - OPERATIONAL

ATTITUDE - COUPLES IN 3 DEGREES OF FREEDOM

TRANSLATION - 2 ENGINES IN EACH DIRECTION

ONE FAILURE: - OPERATIONAL

ATTITUDE - COUPLES IN 3 DEGREES OF FREEDOM

TRANSLATION - 2 ENGINES IN EACH DIRECTION

TWO FAILURES: - SAFE

ATTITUDE - SINGLE ENGINE OFF THE CG

TRANSLATION - SINGLE ENGINE WITH ATTITUDE INDUCED MOTION

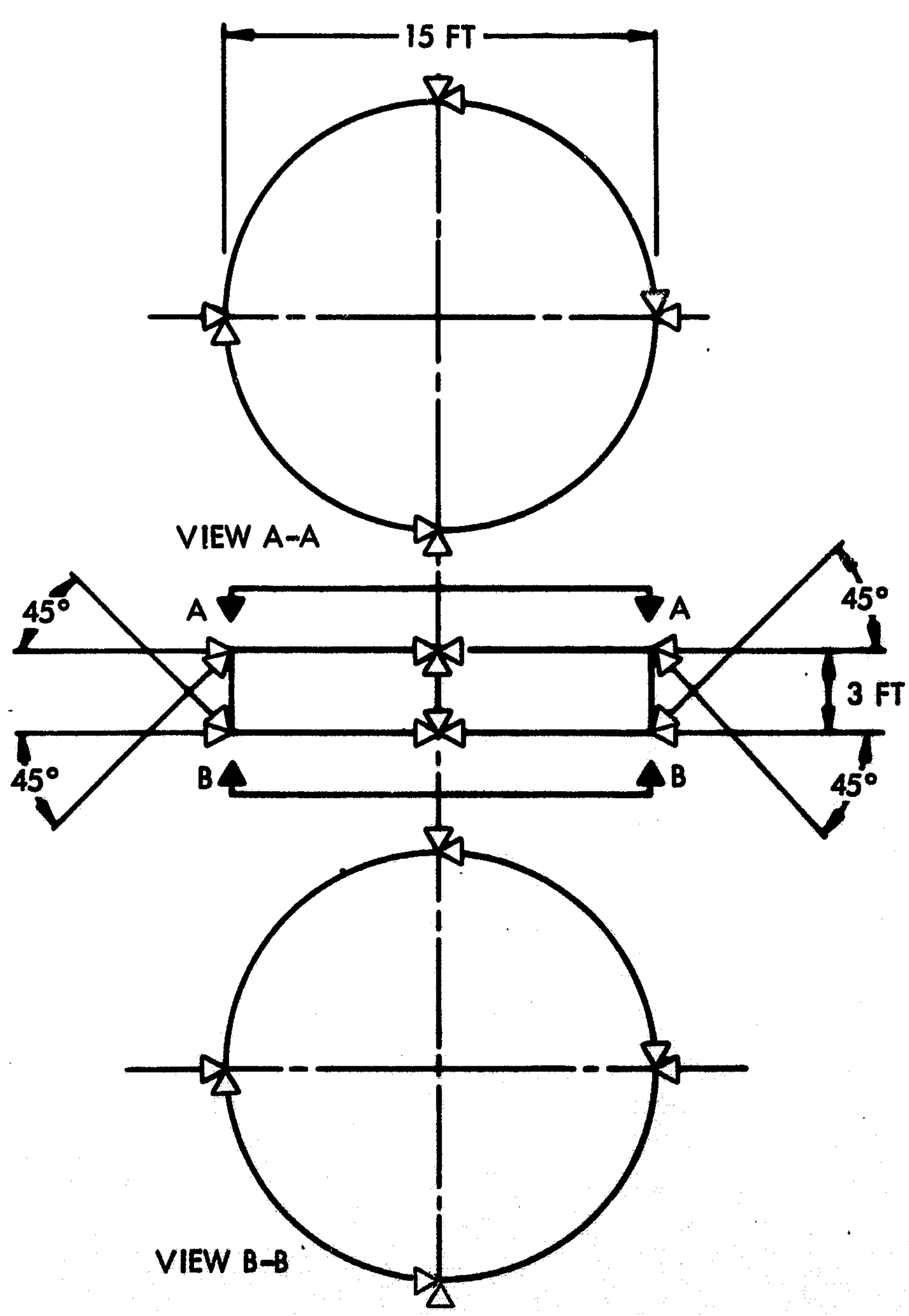
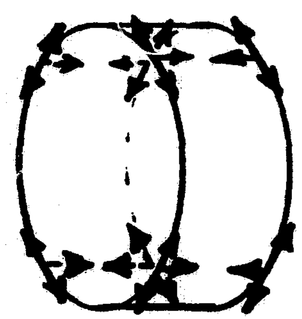


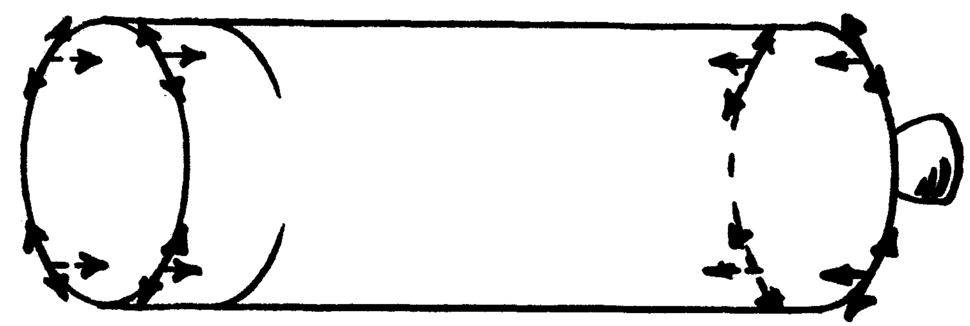
Figure 7-1. Sixteen Jet RCS—Fail Operational, Fail-Safe

- FAIL OPERATIONAL, FAIL OPERATIONAL, FAIL-SAFE ASSUMED
- UNLIMITED CG LOCATION

- MOST PROMISING SPECIFIC CASE (24 JETS):



- 8 JET TRIADS ON IM
- SHORT LINE LENGTHS
- SELF-CONTAINED IM



- FOUR JET TRIADS ON IM, FOUR ON PM
- BETTER LATERAL TRANSLATION
- MORE EFFICIENT MOMENT ARMS

Figure 7-2. Redundancy Requirements for Dual Jet Station ACS

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North American Rockwell

3. It is relatively simple, in that few jets are required and only four clusters are needed. Only one jet station is required.

4. All jets may be rated at the same thrust level.

7.3 JET THRUST LEVEL

In a rate-stabilized attitude control system, the factor dominating ACS control-moment levels is adequate authority over disturbances. If a factor of approximately seven is maintained over all prolonged distrubance torques, then the ACS will meet any other attitude control requirement for large thrust. The thrust level thus obtained may also meet requirements for sensitive control, docking, navigation sightings, etc., provided that the vehicle mass and moments of inertia do not radically change during the mission.

For assurance of adequate response during maneuvers, the jet thrust level must be of sufficient magnitude. A reasonable case of the largest pitch or yaw moment of inertia, in conjunction with the most constraining maneuver time requirement, would set the upper thrust level requirement. Unfortunately, neither the times nor the configuration yielding the largest moment of inertia (including payload) are easily identified this early in tug development. Spur-of-the-moment calculations conducted during the study produced results ranging from 123 lb/jet (547 N/jet) to 350 lb/jet (1557 N/jet). All the calculations assumed maneuver times based on intuition only. As will be shown, it is desirable to keep the maximum thrust as low as possible. The most reasonable value, therefore, appears at this time to be 200 pounds (890 newtons).

More data exist to support the requirements for minimum jet thrust. A minimum pulse should produce a rotational or translational velocity change at least half an order of magnitude less than any velocity error allowance for docking. This will be referred to later as the docking resolution requirement. Apollo easily met this requirement, with the help of a very low minimum pulse duration (approximately 0.013 seconds). If O_2/H_2 propellant is used with tug, indications are that the minimum pulse duration is nearer 0.050 seconds. The longer the pulse, the lower the thrust level has to be to achieve the same velocity change.

Another driver for a low minimum thrust is propellant usage. As will be shown, propellant increases as the square of the minimum impulse bit (thrust times minimum pulse duration) during attitude hold. Long coast periods during which attitude hold control is used could account for a major portion of ACS propellant if the jets are too large. From another standpoint, the lower the thrust level progresses, the more susceptable the control system is to the effects of disturbances. The major disturbances are strongly influenced by vehicle configuration. Since tug is designed to be somewhat variable in configuration, thrust should likewise be variable to optimize propellant consumption.

The conclusion is that a variable-thrust-level ACS is necessary to meet the varied requirements of tug, either by selection of multiple jets, use of multiple jet levels, or by throttling jets. The throttling approach is recommended for the following reasons:

1. No jet configuration change is necessary to increase or decrease the variable range.

2. Only slow response throttling is necessary; therefore, simple propellant flow rate valves may be used.

3. Throttling capability will pay for itself in propellant weight saved and is less complex than the other alternatives.

Several ACS operating regimes may be encountered during a tug mission. The requirements of each regime have been established during development of other spacecraft and are applicable to the tug. This study employs Apollo program spacecraft criteria as a basis for these requirements, since nearly all tug manned-mission elements have Apollo parallels. Although it is not necessarily true that the control forces and moments adapted for Apollo spacecraft represent the only appropriate ranges, adequate control characteristics are assured if they are used. Because the ACS jets are only part of the stabilization loops, the angular and translational accelerations they produce are time-limited by velocities. Through proper sizing of the moments and forces they produce, desirable system responses will be generated. The objective of the approach taken is, therefore, to produce tug control that responds the same as Apollo spacecraft.

Another reason for relying on Apollo requirements is to preclude the design of an inefficient system. Unnecessarily large control moments cause jets to operate in short bursts, which requires fast-acting elements in the rest of the control loop and jet operation at low I_{sp}. Weak control moments optimize I_{sp} and loop element cost but give sluggish response to commands and are unable to damp disturbances rapidly. These points are summarized in Figure 7-3.

Typical Apollo CSM and LM mass properties data used in the analysis are given in Tables 7-8 and 7-9. From the data, the rotational and translational accelerations shown in Table 7-10 were computed. These accelerations are the end product of years of intensive study in the areas of handling

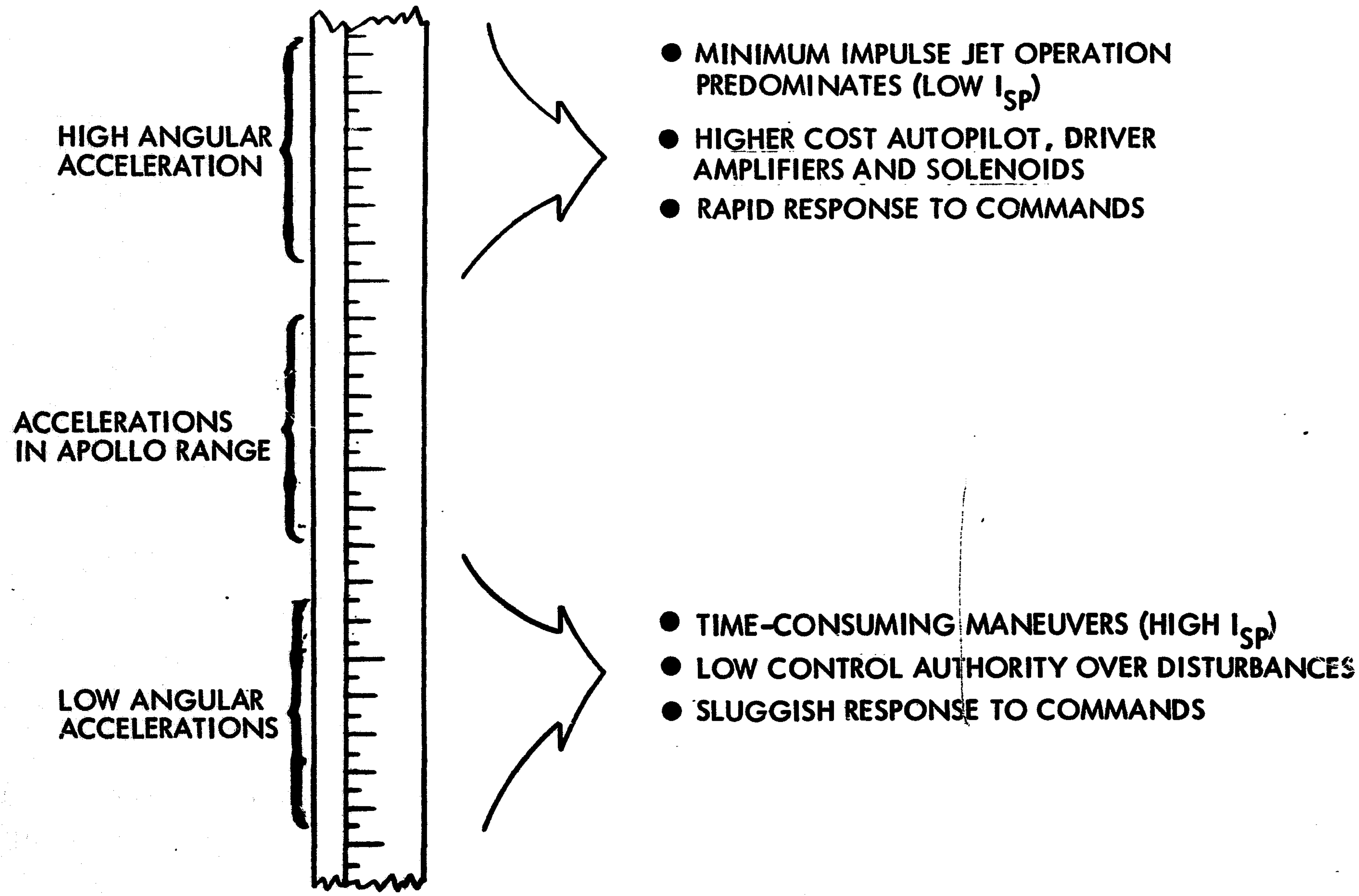


Figure 7-3. Angular Acceleration Requirement Bounds

Table 7-8. Apollo CSM Mass Properties Data

Mission Phase	W (lb/kg)	Ixx (slug-ft²/ kg-m-sec²)	Ivy (slug-ft²/ kg-m-sec²)	Izz (slug-ft²/ kg-m-sec²)
CSM separation and transposition docking	63,484/28,796	33,970/4,696	78,300/10,824	80,570/11,139
CSM/LM prior to LOI	91,200/41,276	52,200/7,216	518,000/71,613	529,000/73,134
CSM in lunar orbit	37,085/16,821	20,417/2,823	57,015/7,882	63,475/8,775
CM/SM prior to separation	25,655/11,637	14,350/1,984	47,345/6,545	48,430/6,695
Effective RCS thrust moment arm = 7.25 feet/2.21 meter) in each axis				

Table 7-9. LM Mass Properties Data

Mission Phase	W (lb/kg)	Ixx (slug-ft²/ kg-m-sec²)	Ivy (slug-ft²/ kg-m-sec²)	Izz (slug-ft²/ kg-m-sec²)
After CSM/LM separation	33,900/15,377	23,080/3,191	26,040/3,600	25,980/3,592
Touchdown	16,933/7,681	13,150/1,818	15,190/2,100	17,515/2,421
Liftoff	10,699/4,853	6,710/928	3,390/469	5,910/817
Prior to CSM/LM docking	5,634/2556	3,265/451	2,950/408	2,070/286
Effective RCS thrust moment arm = 5 feet/1.5 meter (x axis), 5.5 feet/1.7 meter (y and z axis)				

qualities, disturbance environments, and timing constraints. It may be noted in the table that the LM control moment authority is much greater than that of the CSM, since LM RCS must steer during main propulsion operation. Both stages of the LM rely on a single RCS system, which indicates that the preferred control acceleration may adequately cover a large range.

The throttling ratio may be calculated from a consideration of docking resolution. Angular rate is the most constraining of the six docking accuracy requirements listed in Table 7-10 for a small vehicle. To maintain a $\pm$ 1.0 deg/sec envelope, the control system must at least be able to control within 0.4 deg/sec. Assuming a 15-foot-diameter jet couple with a minimum pulse duration of 0.05 sec allows derivation of the maximum thrust level for rotational rate docking resolution with Apollo requirements for any vehicle moment of inertia. The minimum value of the thrust level results from considering the lowest obtainable moment of inertia. This value is approximately 2940 slug-ft^2 (406 kg-m-sec^2) for the mini-tug pitch axis. With the data used in a simple torque equation, the jet thrust level is calculated to be 27 pounds (120 newtons). The throttling ratio from the assumed 200-pound-rated (890-newtons) thrust is then 7.4 to 1. Further analysis has indicated that a throttling ratio of 5 to 1 is adequate for all other tug configurations. As a special case, the mini-tug flow-rate valves might be designed to give a larger throttle ratio.

7.4 ACS PROPELLANT USAGE ANALYSIS

For calculating the propellant used by auxiliary propulsion during a specific mission and for a defined vehicle configuration, approximate equations have been derived. These equations include many simplifying assumptions, but yield results sufficiently accurate for their purpose.

The known or assumed data pertaining to the calculations are:

n Number of jets used together

A_R Rotational acceleration

A_T Translational acceleration

d Vehicle diameter

Δt Mission segment time duration

$\Delta\theta$ Angular maneuver excursion

Table 7-10. Apollo Spacecraft RCS Acceleration Capabilities

Operation	Requirements and Assumptions	Expected Nominal Acceleration
Attitude hold and stationkeeping	Independent of acceleration	None
Disturbance settling and attitude maneuvers	Main propulsion self-stabilizing. RCS provides initial orientation and shutdown transient control.	0.16 to 1.8 deg/sec^2 (1)
	RCS attitude stabilization during main propulsion operation	2.4 to 30 deg/sec^2 (2)
Docking and separation	Axial Velocity: 0.1 to 1.0 fps (0.03 to 0.3 mps)	0.16 to 1.8 deg/sec^2 (1)
	Radial Velocity: 0 to ±0.5 fps (0 to ±0.15 mps)	0.07 to 0.50 fps^2 (3) 0.02 - 0.15 mps^2
	Angular Velocity: 0 to ±1.0 deg/sec	2.4 to 30 deg/sec (2)
	Roll Angular Position: 0 to ±10 deg	0.19 to 2.29 fps^2 (4) 0.06 - 0.70 mps^2
	Pitch and Yaw Angular Pos: 0 to ±10 deg	
	Radial Position: 0 ± 12 inches (0 ± 30 cm)	
Lunar landing and launch	Main propulsion self-stabilizing. RCS provides initial orientation and shutdown transient control.	No data
	RCS attitude stabilization and horizontal translation thrusting during main propulsion operation	3.6 to 18.7 deg/sec^2 (3) 0.38 fps^2 (4) 0.115 mps^2

(1) Apollo CSM pitch or yaw rotational bounds

(2) LM pitch or yaw rotational bounds

(3) Apollo CSM translational bounds

(4) LM translational bounds

Δt_{min} Minimum jet pulse duration

ΔV Translational velocity change

F Single ACS jet thrust level

ℓ Vehicle overall length

g Earth gravitational acceleration

I_{sp} Rated specific impulse

θdb Attitude-hold deadband half-amplitude

$\dot{\theta}_o$ Initial angular rate error

W Total vehicle weight

K ACS propellant flow constant for main propulsion roll control

X_i Distance from an arbitrary reference point on the vehicle center line to the geometrical center of the i^{th} module

The data to be derived are

I_x Roll moment of inertia

I_y Pitch or yaw moment of inertia

W_P ACS propellant weight for an operation

τ_o Minimum attitude maneuver time

τ ACS jet operating time

7.4.1 Moment of Inertia Equations

As a rough estimate of the vehicle moment of inertia, equations for a cylinder of constant density may be used:

$$I_x = \frac{Wd^2}{8g} \tag{7-1}$$

$$I_y = \frac{W}{g}\left(\frac{d^2}{16} + \frac{\ell^2}{12}\right) \tag{7-2}$$

A better approximation, however, considers the modules as separate cylinders with individual densities. The roll moment of inertia is essentially unchanged:

$$I_x = \frac{d^2}{8g}\sum W_i \tag{7-3}$$

where the W_i are the individual module weights. The pitch or yaw moment of inertia is more difficult to derive. The moment of inertia of each individual module about its own c. g. is

$$I_{y_{o_i}} = \frac{W_i}{g}\left(\frac{d^2}{16} + \frac{\ell_i^2}{12}\right) \tag{7-4}$$

Each of the centroidal moment of inertias must be translated to the total vehicle c. g., defined as

$$\bar{X} = \frac{\Sigma W_i X_i}{\Sigma W_i} \tag{7-5}$$

where X_i and X are measured from the same arbitrary reference point on the vehicle center line. The distance translated for each module is

$$D_i = \bar{X} - X_i \tag{7-6}$$

In the translation to the total vehicle c. g., the moment of inertia of each module increases accoring to the parallel axis theorem:

$$I_{y_i} = I_{y_{o_i}} + \frac{W_i}{g} D_i^2 \tag{7-7}$$

Substituting Equations (7-4), (7-5), and (7-6) into Equation (7-7) and summing for all of the modules,

$$I_y = \sum\left[\frac{W_i}{g}\left(\frac{d^2}{16} + \frac{\ell_i^2}{12}\right) + \frac{W_i}{g}\left(\frac{\Sigma W_i X_i}{\Sigma W_i} - X_i\right)^2\right] \tag{7-7}$$

Of the two major terms in parentheses, the second may be simplified. Expanding this term,

$$\frac{1}{g}\sum W_i\left(\frac{\Sigma W_i X_i}{\Sigma W_i}\right)^2 - \frac{2}{g}\sum W_i X_i\left(\frac{\Sigma W_i X_i}{\Sigma W_i}\right) + \frac{1}{g}\sum W_i X_i^2 \qquad (7\text{-}8)$$

which is the same as

$$\frac{1}{g}\frac{(\Sigma W_i X_i)^2}{\Sigma W_i} - \frac{2}{g}\frac{(\Sigma W_i X_i)^2}{\Sigma W_i} + \frac{1}{g}\sum W_i X_i^2 \qquad (7\text{-}9)$$

The first two of these terms may be combined, and the resulting form of Equation (7-9) substituted into the complete equation is

$$I_y = \sum\left[\frac{W_i}{g}\left(\frac{d^2}{16} + \frac{\ell_i^2}{12}\right) - \frac{(\Sigma W_i X_i)^2}{g\,\Sigma W_i} + \frac{\Sigma W_i X_i^2}{g}\right] \qquad (7\text{-}10)$$

which condenses to its final form:

$$I_y = \sum\left[\frac{W_i}{g}\left(\frac{d^2}{16} + \frac{\ell_i^2}{12} + X_i^2\right)\right] - \frac{(\Sigma W_i X_i)^2}{g\;\Sigma W_i} \qquad (7\text{-}11)$$

7.4.2 Lateral and Longitudinal Translation Equations

Lateral and longitudinal translation is described by a simple force consideration:

$$n\,F\,\Delta t = \frac{W}{g}\,\Delta V \qquad (7\text{-}12)$$

and the propellant relationship

$$n\,F\,\Delta t = I_{sp}\,W_p \qquad (7\text{-}13)$$

Dividing Equation (7-12) by Equation (7-13) and rearranging to obtain the final form,

$$W_p = \frac{W\,\Delta V}{g I_{sp}} \qquad (7\text{-}14)$$

7.4.3 ACS Roll Control for Main Propulsion

If the tug is designed for a single main propulsion engine, the engine is incapable of gimbaling for roll control. ACS roll control in this case probably uses an insignificant amount of propellant. Estimates of propellant usage should come from flight or test data. The form of these data is an average flow rate, K.

$$W_p = K \; \Delta t \tag{7-14}$$

7.4.4 Main Propulsion Shutoff Disturbance Damping Equations

During main propulsion engine shutoff, appreciable lateral forces are incurred, which gives rise to a net vehicle angular rate in an arbitrary direction. This rate is then damped by the ACS. Assuming a simplified logic as shown in the top sketch in Figure 7-4, the operation is completed in one cycle. To fit the case at hand, the torque equation is

$$n F \frac{d}{2} = I_y \frac{\dot{\theta}_o}{(\tau/3)} \tag{7-15}$$

where Iy was selected as a typical axis of concern. For convenience, assume a zero coast period so that Δt in Equation 7-13 may be substituted for τ in Equation 7-15. Solving for propellant in the final form,

$$W_p = \frac{6 I_y \dot{\theta}_o}{I_{sp} d} \tag{7-16}$$

7.4.5 ACS Attitude Maneuver Equations

The maneuver following a commanded change of attitude is shown in the center sketch of Figure 7-4. It may be accomplished by applying an ACS control moment for a period $\tau/2$, coasting a specified amount of time, then applying a counter moment for the same period, $\tau/2$. The minimum time for the maneuver is τ, when the coast period is zero. Integrating the ACS moment moment over its first application period yields the angular coasting rate:

$$\dot{\theta}_{max} = \frac{nFd\tau}{4 I_y} \tag{7-17}$$

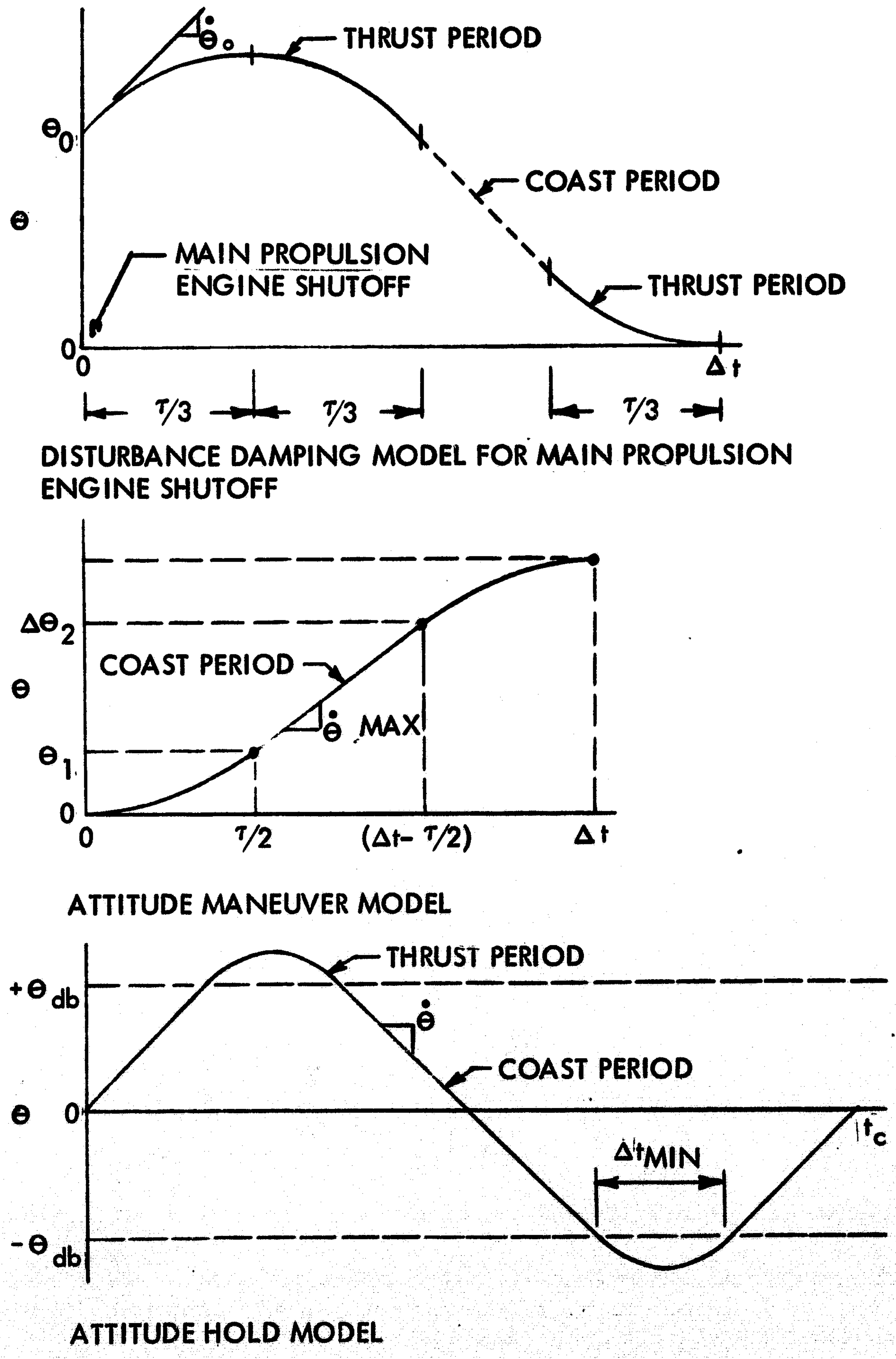


Figure 7-4. ACS Operation Details

A part of the desired attitude change is covered while the moment is applied and is described by its second integration:

$$\theta_1 = \frac{nFd\tau^2}{16\, I_y} \tag{7-18}$$

The total maneuver time, Δt, is composed of

$$\Delta t = \tau + \frac{\theta_2 - \theta_1}{\dot{\theta}_{max}} \tag{7-19}$$

and the total maneuver angle, $\Delta\theta$, includes

$$\Delta\theta = (\theta_2 - \theta_1) + 2\,\theta_1 = \theta_2 + \theta_1 \tag{7-20}$$

When the coast period is zero, $\Delta\theta = 2\theta_1$, and from Equation (7-18),

$$\tau_o^2 = \frac{8\, I_y\, \Delta\theta}{nFd} \tag{7-21}$$

which represents the minimum time to rotate the vehicle through a given angle by operating the ACS continuously. The five equations, (7-17) through (7-21) may be used to eliminate θ_1, θ_2, and $\dot{\theta}_{max}$ to yield

$$\tau = \Delta t - (\Delta t^2 - \tau_o^2)^{1/2} \tag{7-22}$$

The maneuver propellant is found by substituting Equation (7-22) into the definition of I_{sp}:

$$W_p = \frac{nF\tau}{I_{sp}} = \frac{nF}{I_{sp}}\left[\Delta t - (\Delta t^2 - \tau_o^2)^{1/2}\right] \tag{7-23}$$

7.4.6 Attitude Hold Equations

A large part of a mission is spent coasting; and, if other requirements dictate attitude maintenance, ACS must provide efficient and effective attitude hold control. Propellant usage during these periods is inversely proportional to the allowable deadband and directly proportional to the square of the number and thrust level of the jets. Minimum propellant usage, therefore, results from the relaxation of pointing requirements and the decrease of force.

The attitude hold model is shown in the bottom sketch of Figure 7-4. It ignores the effects of system lead generated with rate feedback and the effects of disturbance torques. From the sketch, the total limit cycle dura-duration is

$$t_c = 2\Delta t_{min} + 4\,\frac{\theta db}{\dot{\theta}} \tag{7-24}$$

Now integrating the ACS moment about any single axis yields the rate of drift across the deadband:

$$\dot{\theta} = \frac{nFd\,\Delta t_{min}}{4I} \tag{7-25}$$

Substituting Equation (7-25) into Equation (7-24) and factoring,

$$t_c = \frac{2}{\Delta t_{min}}\left[\Delta t^2_{min} + \frac{8\,\theta db\,I}{nFd}\right] \tag{7-26}$$

At its minimum expected value, the second term in the brackets is an order of magnitude larger than the first term in the brackets. Therefore, an approximate form of Equation (7-26) may be used:

$$t_c \cong \frac{16\,\theta db\,I}{nFd\,\Delta t_{min}}, \qquad \Delta t^2_{min} \ll \frac{8\,\theta db\,I}{nFd} \tag{7-27}$$

The total coast period is composed of p cycles, each of which is t_c in duration:

$$\Delta t = p\,t_c \tag{7-28}$$

and the total jet operating time throughout the coast period is

$$\Delta t_{on} = 2\,p\,\Delta t_{min} \tag{7-29}$$

Substituting Equations (7-27), (7-28), and (7-29) into the definition of I_{sp},

$$W_p = \frac{nF\,\Delta ton}{I_{sp}} = \frac{(nF\,\Delta t_{min})^2\,\Delta t\,d}{8\,I_{sp}\,\theta db\,I} \tag{7-30}$$

giving the final form of the equation for single-axis attitude hold. A more useful form includes three-axis attitude hold by combining moments of inertia:

$$W_p = \frac{(nF\,\Delta t_{min})^2\,\Delta t\,d}{8\,I_{sp}\,\theta db}\left(\frac{2}{I_y} + \frac{1}{I_x}\right) \tag{7-31}$$

Since attitude hold propellant is sensitive to many factors, some insight is needed for the proper use of Equations (7-30) and (7-31). The single-axis equation was plotted in a propellant flow rate versus moment of inertia field, shown in Figure 7-5. Apollo data from Table 7-8 have been included for comparison. Actual Apollo flight data, although difficult to interpret, appear to lie above the broad bars, indicating the equation yields slightly optimistic results. The ACS equations are summarized in Table 7-11.

7.4.7 ACS Propellant Computation

All of the acceleration data derived from Apollo RCS capabilities, and shown in Table 7-10, were used in a computer program together with the propellant estimation equations derived here. The program also contained logic necessary to control the ACS thrust level within the Apollo requirements. The program was developed not to produce data for mission time lines, but rather to prove the feasibility of defining ACS requirements with this approach. A flow diagram of the program is shown in Figure 7-6. Two examples of program results are shown in Figure 7-7 and 7-8. The examples were run early in the contract study and used preliminary data. In the examples no restriction on throttling ratio was exerted. For serious design, however, the throttling ratio should be held within a range that allows use of a simple, reliable flow control valve. It may be seen in both figures that the thrust was decreased by the program logic when a prolonged period of attitude hold was forseen. Equation (7-31) shows that propellant usage over these periods increases as the square of the thrust level; and, if throttling were not incurred, much larger propellant usage would result.

Although no further use of the computer program was undertaken during the study, the equations were used in the Mission Functional Analysis

Function	Equation
Roll moment of inertia	$I_x = \frac{d^2}{8g} \sum W_i$
Pitch or yaw moment of inertia	$I_y = \sum \left[\frac{W_i}{g} \left(\frac{d^2}{16} + \frac{\ell_i^2}{12} + X_i^2 \right) \right] - \frac{(\Sigma W_i X_i)^2}{g \; \Sigma W_i}$
Lateral or longitudinal translation	$W_p = \frac{W \; \Delta V}{g \; I_{sp}}$
Single main propulsion engine roll control	$W_p = K \; \Delta t$
Main propulsion shutoff disturbance damping	$W_p = \frac{6 \; I_y \; \dot{\theta}_o}{I_{sp} \; d}$
Attitude maneuver ($\Delta t \geq \tau_o$)	$W_p = \frac{nF}{I_{sp}} \left[\Delta t - \left(\Delta t^2 - \tau_o^2 \right)^{1/2} \right], \; \tau_o^2 = \frac{8 \; I_y \; \Delta\theta}{nFd}$
Attitude hold	$W_p = \frac{(nF \; \Delta t_{min})^2 \; \Delta t \; d}{8 \; I_{sp} \; \theta db} \left(\frac{2}{I_y} + \frac{1}{I_x} \right)$

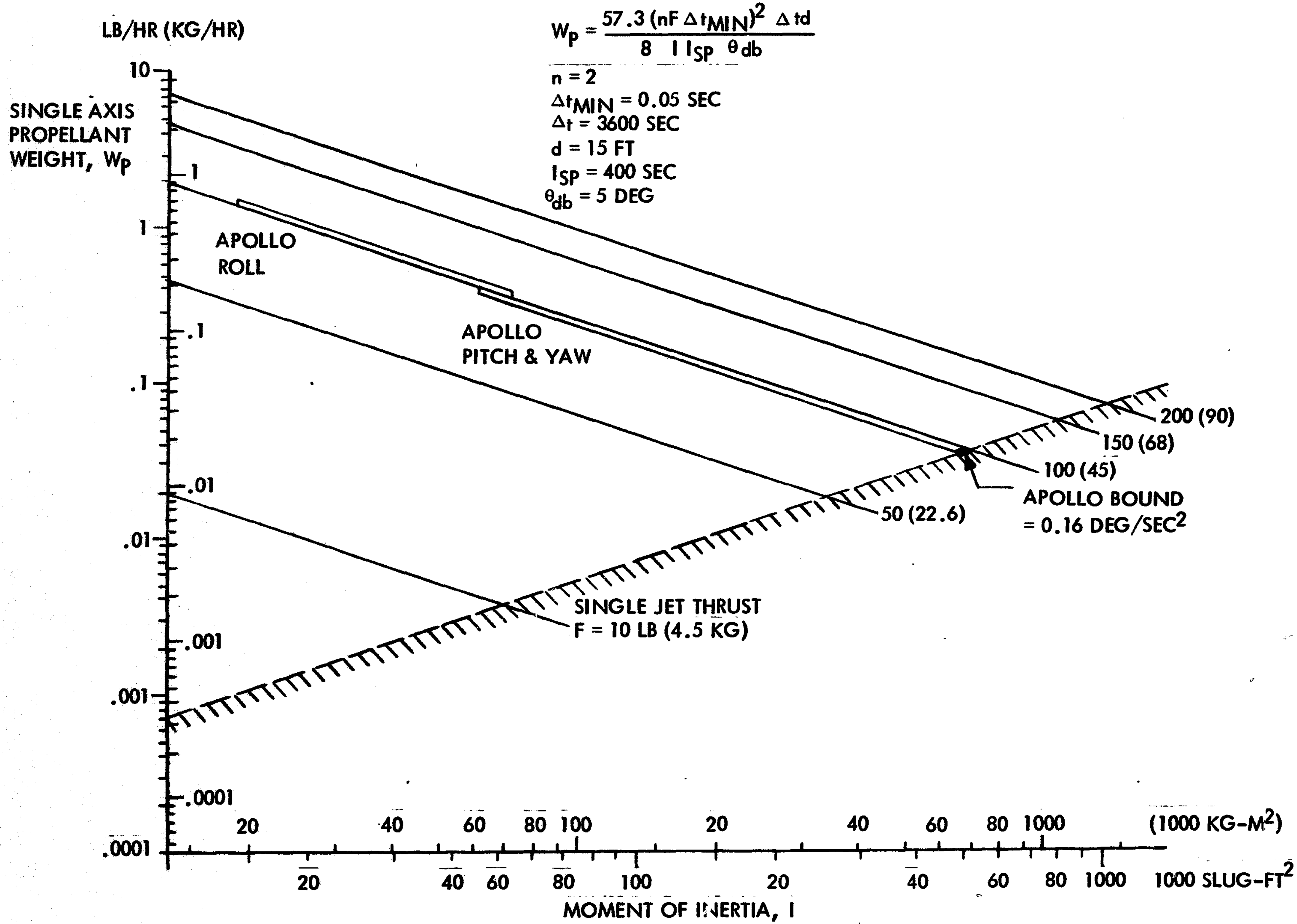


Figure 7-5. Attitude Hold Propellant

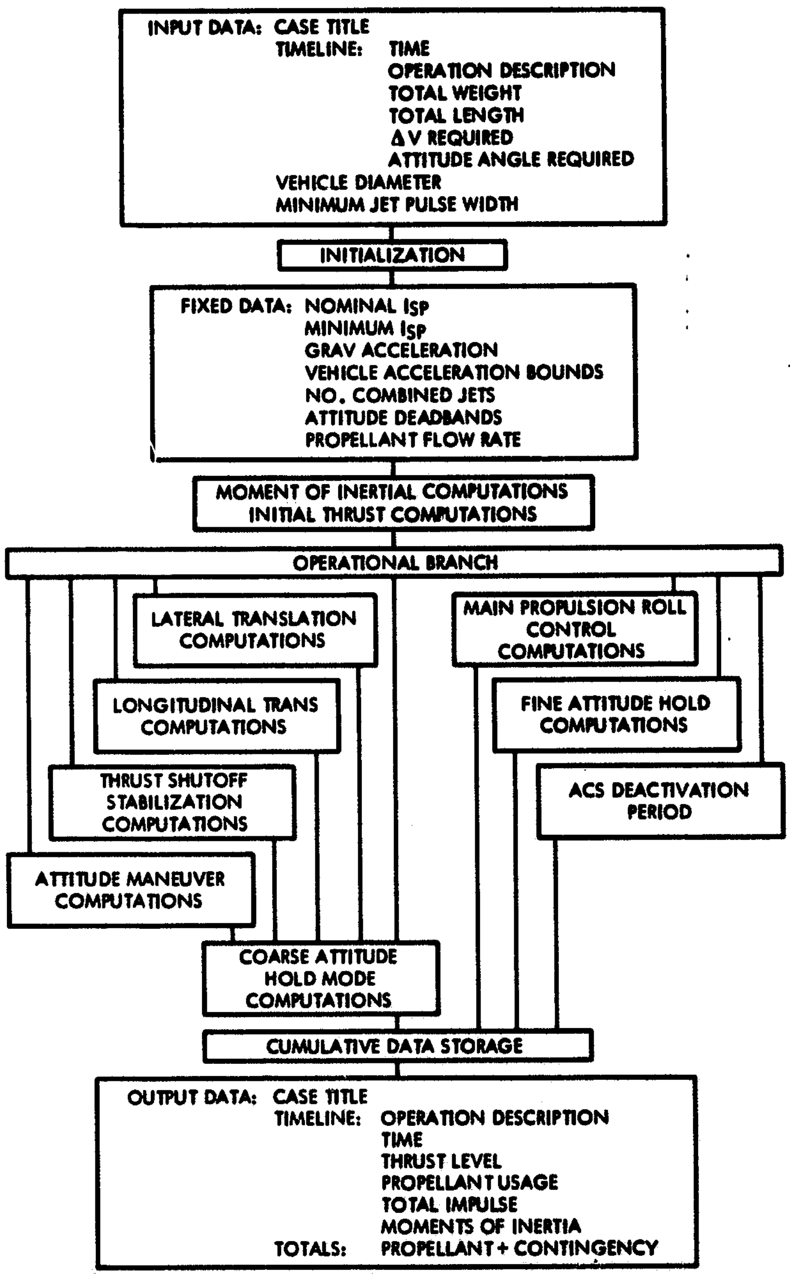


Figure 7-6. Space Tug ACS Sizing Program Functions

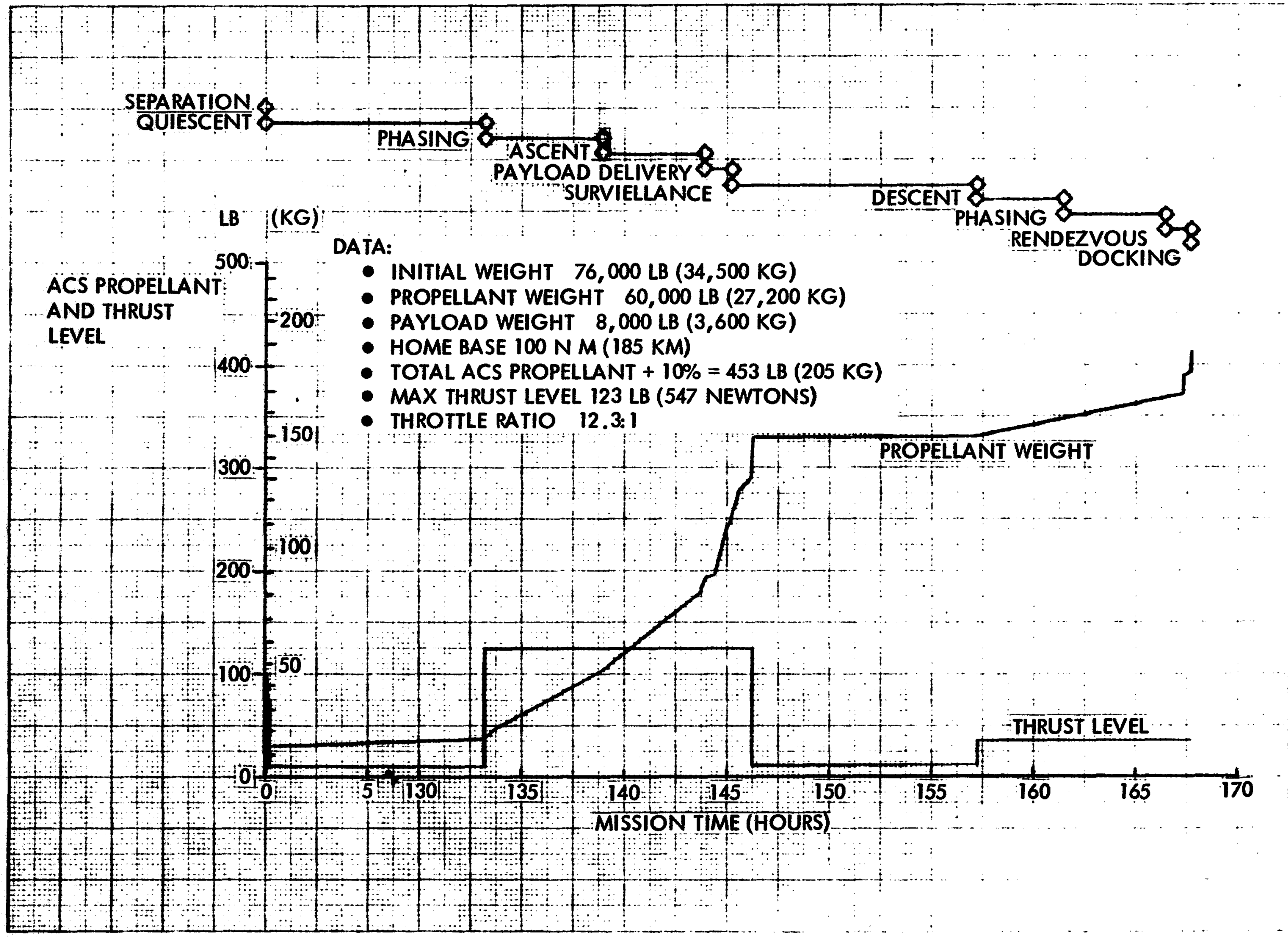


Figure 7-7. Recoverable Geosynchronous Mission

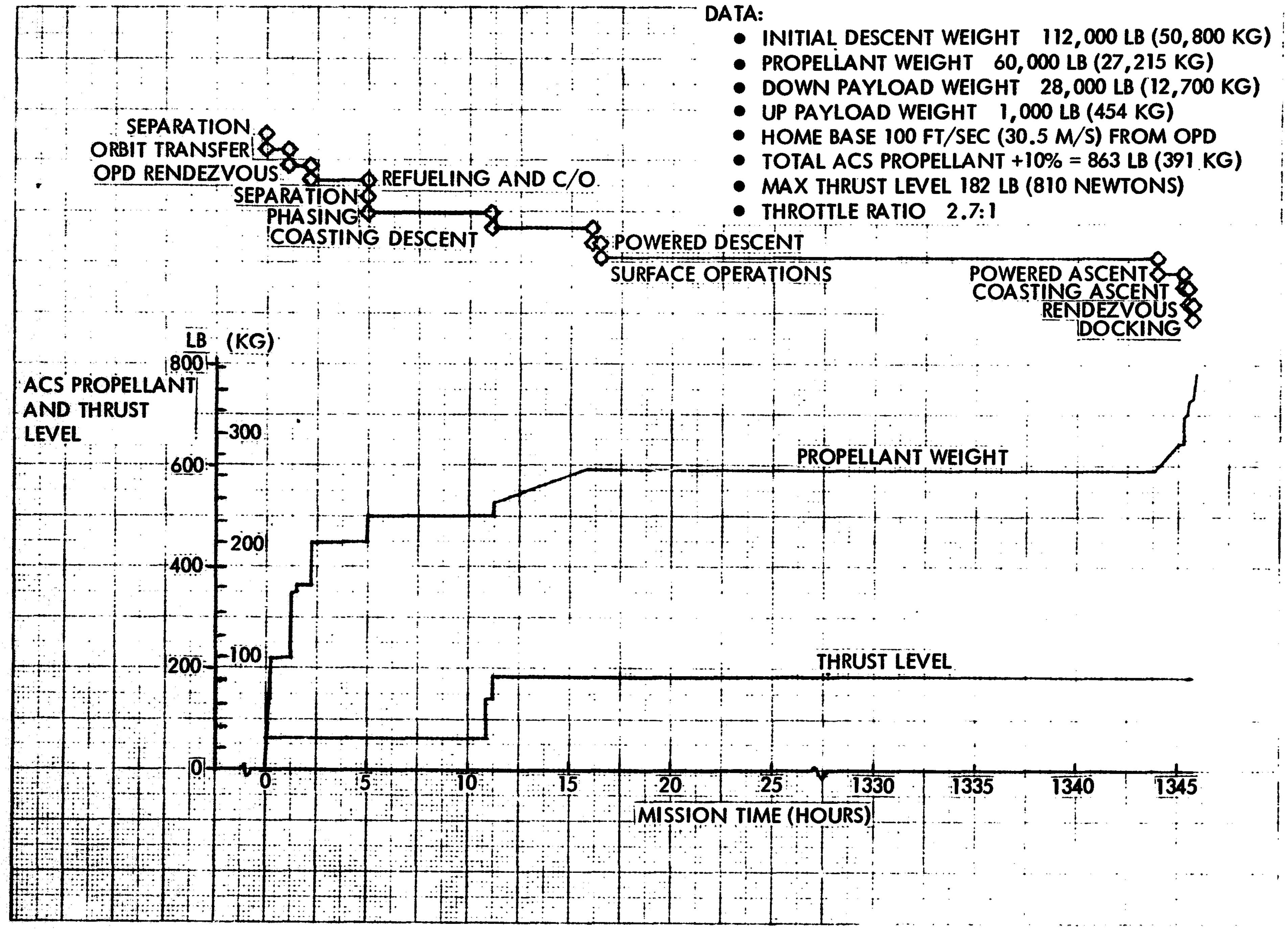


Figure 7-8. Lunar Landing Mission

Computer program, which is described in the Operations Analysis section of the report. Results of this program show the depletion of other expendables in conjunction with ACS.

7.5 RECOMMENDATIONS

An ACS jet configuration that incorporates multiple failure tolerance, without inhibiting the advantages of Tug modularity, requires 20 jets. The ACS design requires throttleable jets to utilize common tankage without reducing the tug multimission scope. The maximum jet thrust is 200 pounds (890 newtons) with a 5 to 1 throttling ratio set tentatively until refined vehicle configurations and mission descriptions are available.

Rotational and translational acceleration bounds, reflecting thrust moment and force levels, have been set by Apollo design. These, together with the approximate equations derived in this section, are sufficient to determine the ranges of total impulse and propellant required. Early indications are that the total impulse range is from 75,000 to 200,000 lb-sec (334,000 to 890,000 newton-seconds).

8.0 AUXILIARY CONTROL PROPULSION SUBSYSTEM

8.1 REQUIREMENTS

The wide range of space tug missions and payloads results in ACPS total impulse requirements that vary from 75,000 to 200,000 pound-seconds (334,000 to 890,000 newton-seconds). The functions that must be performed by the ACPS include (1) attitude control, (2) rendezvous and docking, and (3) delta-V maneuvers under the minimum capability of the main propulsion system. This range of functions, coupled with the different payloads and missions, demand thrust levels that range from 200 to 40 lb_f (890 to 178 newtons). The tug operating life and maintainability requirements differ so from those of the currently operational attitude control propulsion systems that off-the-shelf hardware is not really available, even for the earth storable liquid bipropellants. North American Rockwell experience indicates that total impulse (propellant) requirements for attitude control systems usually increase as system design progresses. For this reason, in addition to the variations in preliminary requirements, it is felt that heavy emphasis must be assigned to growth capability and mission flexibility. The preliminary space tug ACPS requirements are presented in Table 8-1.

Table 8-1. ACPS Requirements

Total impulse	75,000 to 200,000 Lb_f-sec (334,000 to 890,000 N-sec)
Maximum thrust	200 Lb_f (890 N)
Minimum thrust (for attitude hold)	40 Lb_f (178 N)
Minimum engine ON time	50 milliseconds
Number of engines	20
System life	10 missions

8.2 ACPS CANDIDATES

The candidate propellants for the space tug ACPS are earth storable liquid bipropellants and monopropellants, in addition to the cryogenic bipropellants used for the main propulsion system. In the past, spacecraft attitude control propulsion has been limited mainly to earth storable hypergolic bipropellants for larger (manned) systems and monopropellants or cold gas for the smaller (unmanned) systems. The main propulsion systems for these spacecraft operations have generally used earth storables rather than cryogenic propellants. Since the space tug will utilize cryogenic propellants for the main propulsion system, the use of these same propellants for the ACPS is desirable. This is especially advantageous for space-based operation, which will require space resupply and maintenance. Because of the impracticality of distributing liquid phase propellants to the engines through long manifolds, the ACPS thrusters must be supplied with oxygen and hydrogen in the gaseous phase. The development of this gas-gas oxygen/hydrogen attitude control propulsion technology is in progress for the EOS and the space station.

Preliminary screening of potential candidate propellants led to selection of the following candidates for more thorough evaluation.

1. Earth storable liquid bipropellant (nitrogen tetroxide/monomethylhydrazine)

2. Earth storable liquid monopropellant (hydrazine)

3. Cryogenic bipropellant (gaseous oxygen/gaseous hydrogen)

Other candidate propellants were considered initially, but were discarded prior to conceptual system design. The higher-energy earth storable bipropellant combinations, which use chlorine trifluoride or chlorine pentafluoride, offer good performance but the state of the art has not progressed to the point where the added performance is worth the added cost and risk. Further, the use of corrosive-toxic propellants complicates servicing, maintenance, and logistics with the attendant possibilities of plume damage to payloads or other vehicles, and hazards to personnel.

The cryogenic propellant combinations that use elemental fluorine were rejected because performance is not much better than can be obtained with GO_2/GH_2 and the oxidizer has the same corrosive, toxic properties of the fluorine compounds mentioned. Finally, if cryogenics are used, it is advantageous to use common propellants with the main propulsion system, which simplifies resupply and allows use of common tankage, not only for propulsion, but for life support and fuel cells.

With the candidate propellants restricted to earth storable (bipropellants or monopropellants) or cryogenics (GO_2/GH_2) the next step was conceptual design of the propulsion system, and in some instances, comparison of alternate concepts for each propellant combination.

Further evaluation led to selection of five candidate systems. They are described here and shown in Figures 8-1 through 8-5.

8.2.1 Low-Pressure GO_2/GH_2 System

The low-pressure GO_2/GH_2 system takes propellant (either liquid or gas) directly from the main propellant tank through a heat exchanger and a pressure regulator and to the engines. The system considered here does not have an accumulator, but that could be added. The major advantage of this system is that it uses low-pressure propellants from the main tanks. It does not use turbopumps or compressors. The low-pressure system has the disadvantage of being bulky and relatively heavy, since the low P_c engines become quite large if a high area ratio is used. To fit within the design envelope, the system's area ratio must be cut back to about 5, giving a specific impulse of only 375 seconds (3680 N-s/kg). Moreover, the low-pressure system is more sensitive to pressure and temperature changes than the high-pressure systems, and more accurate regulation is required for throttling. This system has been considered for the EOS, but tug is likely to have even lower main tank pressure, since it is more sensitive to residual gas weight. This implies that current low P_c thruster technology development may be inapplicable.

8.2.2 Pump-Fed GO_2/GH_2 System

The pump-fed GO_2/GH_2 system pumps liquid propellants from the main tanks to the heat exchanger, where they are converted to the gaseous state and stored in accumulators. Pressure regulators downstream of the accumulators provide propellants to the engines at a pressure of approximately 300 psia (207 n/cm^2). This inlet pressure allows an engine chamber pressure of 250 psia (173 n/cm^2), which results in a compact engine and a specific impulse of 420 seconds (4110 n-s/kg). The accumulators provide enough propellant to allow operation of four engines steady state for 20 seconds. The pumps and conditioning unit are expected to have a response of less than five seconds and a maximum flow rate that exceeds (by a small margin) the four-engine steady-state flow requirements. These design values were chosen to give a good balance between accumulator weight and turbopump short-cycling duty requirements.

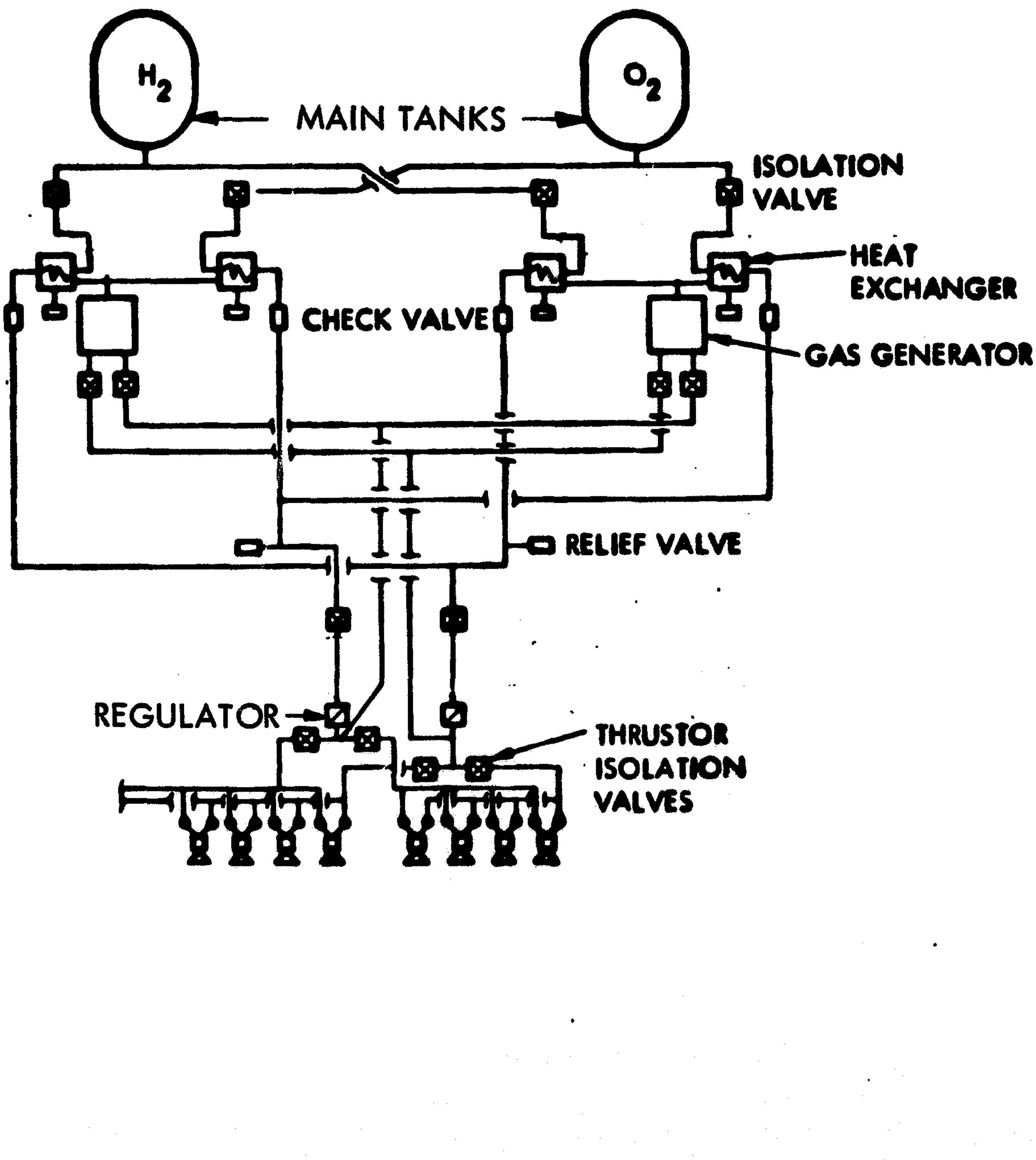


Figure 8-1. Low Pressure System CO_2/GH_2

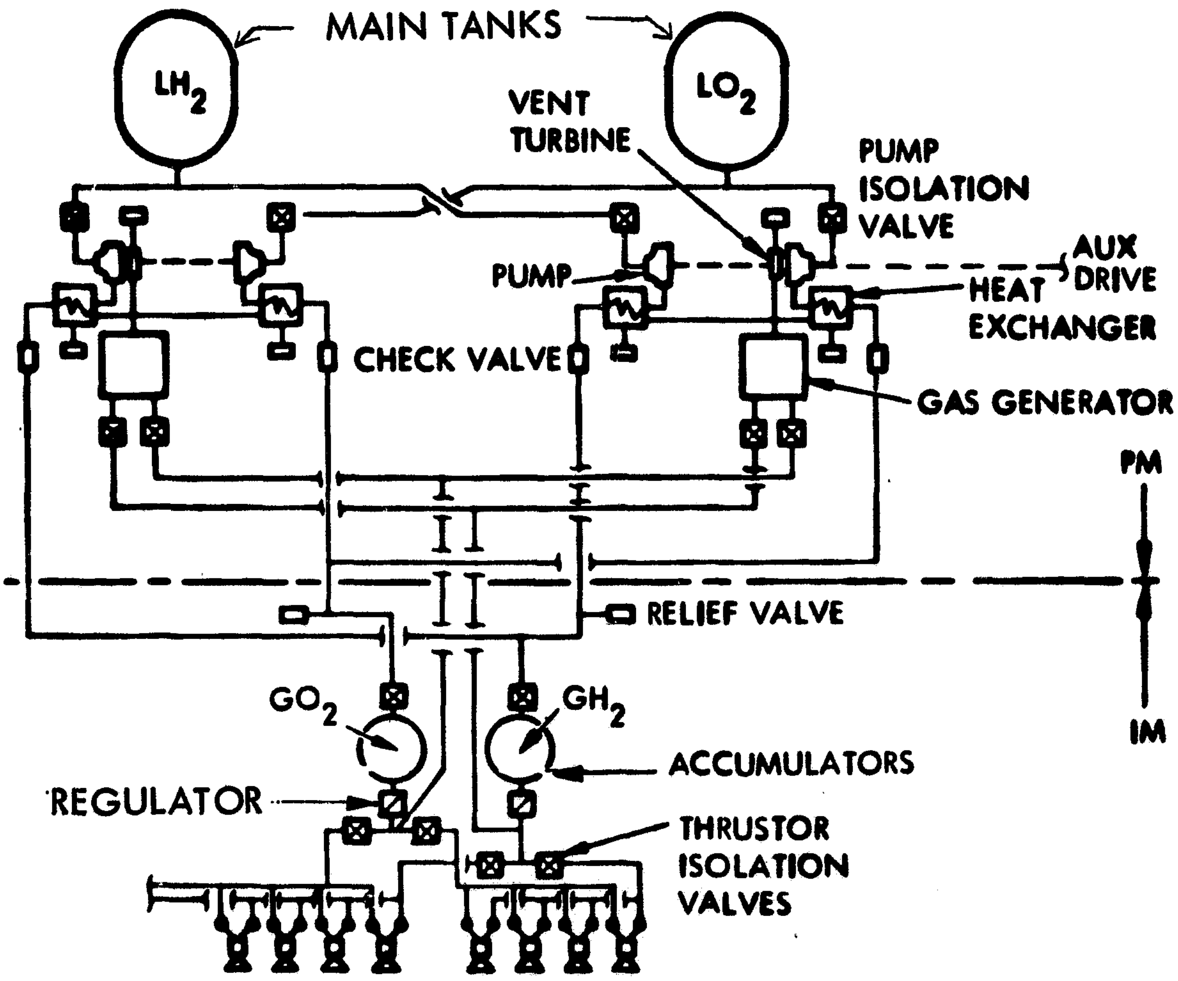


Figure 8-2. Pump-Fed System CO_2/GH_2

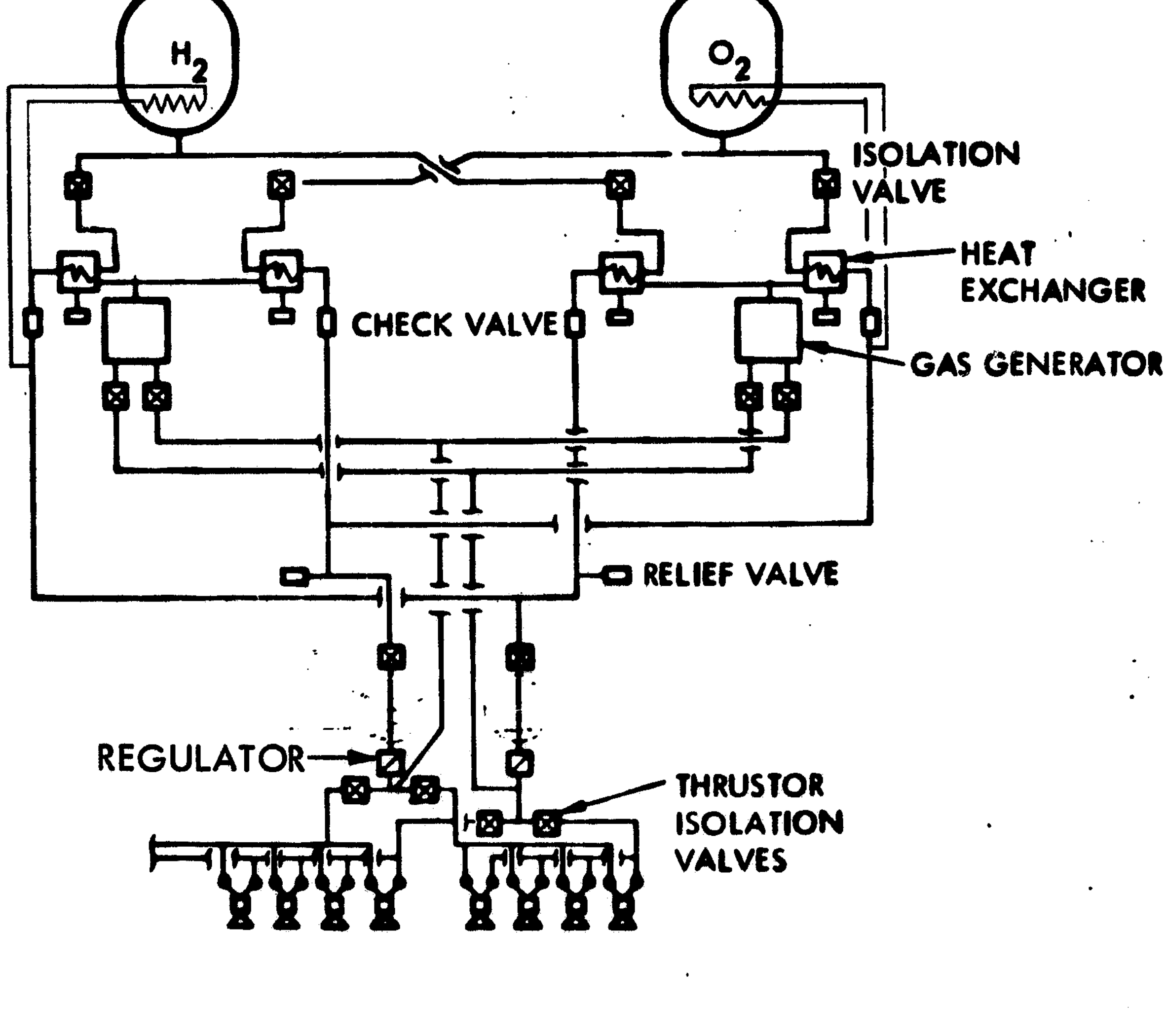


Figure 8-3. Supercritical System CO_2/GH_2

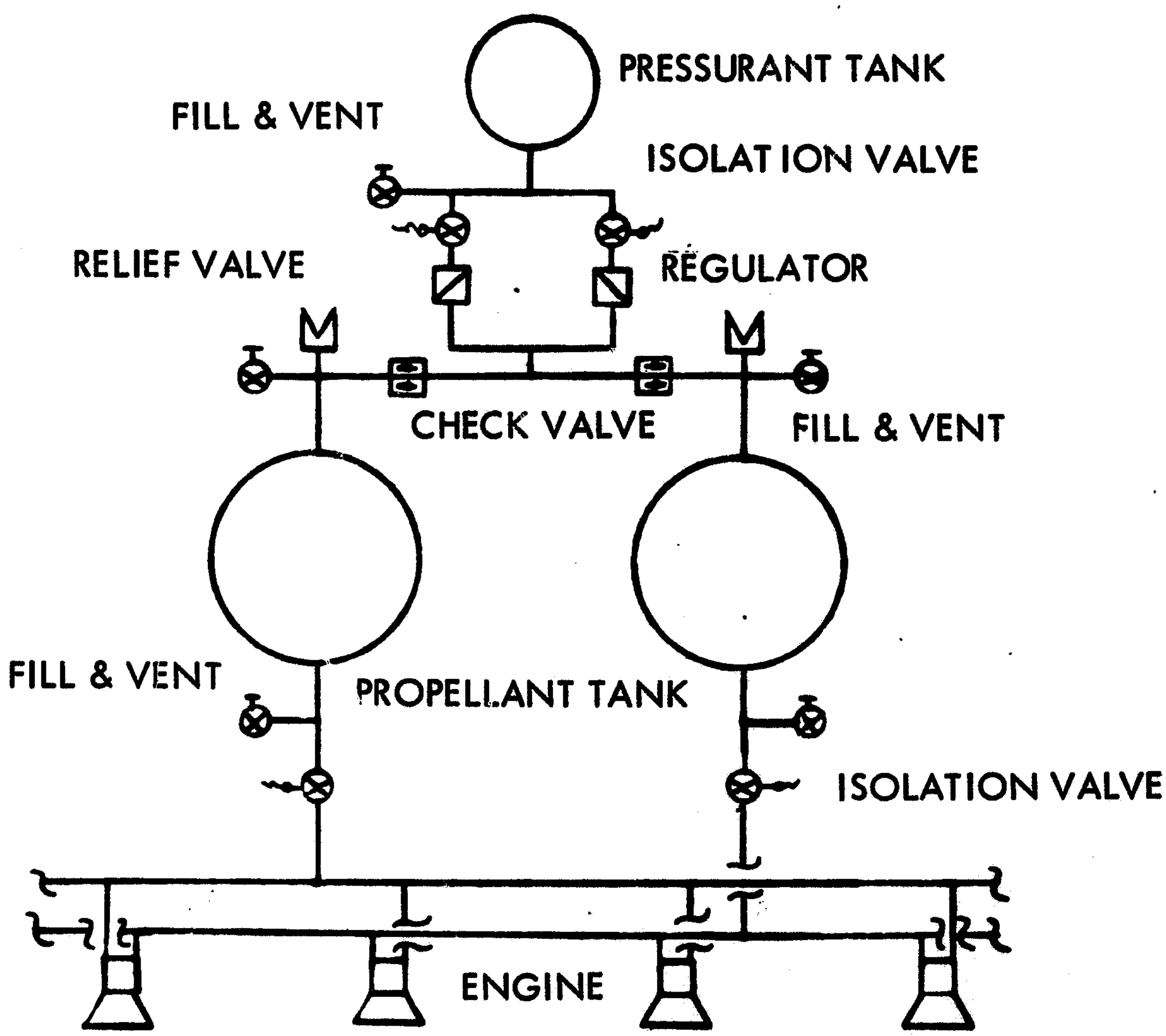


Figure 8-4. Storable Bipropellant System N_2O_4/MMH

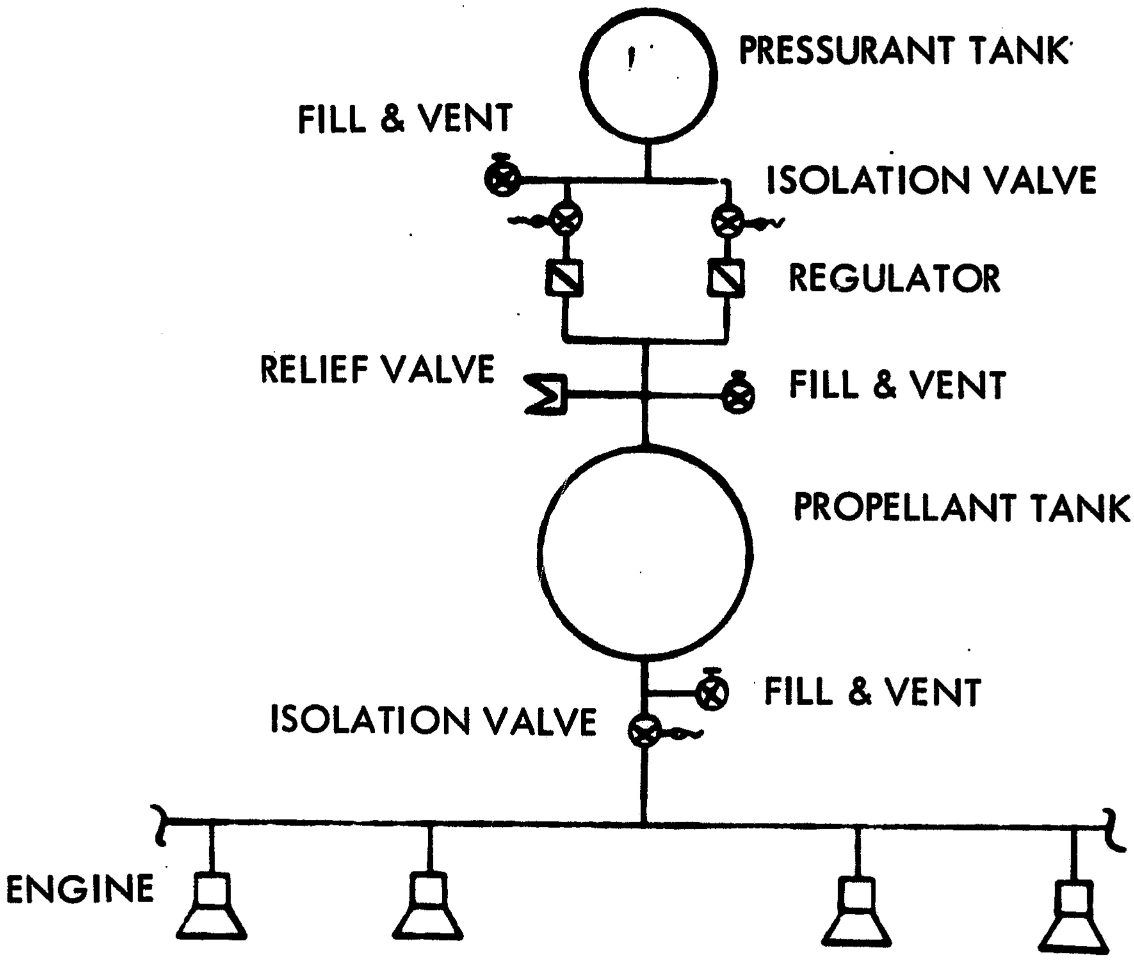


Figure 8-5. Storable Monopropellant System N_2H_4

The chief disadvantage of the pump-fed system is that use of a positive expulsion device in the main tanks is necessary to provide liquid propellants to the pumps since mixed-phase pump work would be excessive. While this system offers the best flexibility and growth potential, it requires the most apparatus of the five candidate systems considered.

8.2.3 Supercritical GO_2/GH_2 System

This system uses independent supercritical propellant storage from which gas-phase propellants (above critical temperature) are passed through a heat exchanger and then through a pressure regulator and to the engines. This system would operate at the same chamber pressure as the pump-fed system. The supercritical storage offers the advantage of single-phase flow, which alleviates the need for an expulsion device, and a simplified conditioning unit. The major disadvantage of this system is that it is not integrated with the main tankage and is, therefore, limited in flexibility.

8.2.4 Storable N_2O_4/MMH System

The candidate storable liquid bipropellant system selected for this study uses nitrogen tetroxide (N_2O_4) and monomethyl-hydrazine (MMH). This propellant combination is hypergolic and, therefore, requires no ignition device. N_2O_4/MMH systems have been used extensively in spacecraft and have been developed to a high degree. The components developed for these systems have fairly short lives, however, and for the reusable space tug modifications would be necessary. The use of moderately toxic-corrosive propellants that are not interchangeable with the main propellants is, of course, the major disadvantage of this system. For a vehicle designed to a relatively low impulse specific mission, this system is extremely attractive, but it is not as applicable to multipurpose vehicles.

8.2.5 Storable Hydrazine Monopropellant System

Hydrazine (N_2H_4) monopropellant systems have seen extensive use since the development of the Shell 405 spontaneous catalyst in 1964. These systems have been used for numerous attitude control systems and, although performance is rather low (compared to bipropellants), they offer simplicity, low cost, ease of development, good storability, low-temperature exhaust, and relatively clean exhaust (ammonia). Use of a separate low-performance propellant that cannot be replenished from the main tanks results in loss of flexibility and complicates servicing and maintenance. As in the case of the liquid bipropellants, the hydrazine system could be attractive for design of a one-shot low-total-impulse-mission vehicle.

8.3 CANDIDATE ACPS WEIGHTS

Weight estimates for the five ACPS candidates are listed in Tables 8-2 through 8-6 for different total impulse values. Figure 8-6 depicts total system effective burnout weight as a function of total impulse for the candidate systems. Total system effective burnout weight is approximated as the system weight plus one-half of the usable propellant.

Weights presented herein are based on use of twenty 200-pound-thrust (890-newton) engines and component redundancy as necessary and practical for reliability. It should be noted that a portion of the integrated systems weight can be charged to the electrical power system and the environmental control system based on the amounts of oxygen and hydrogen that they use.

8.3.1 Candidate Trades

Although the main propulsion system trades are mostly governed by performance considerations, the primary factors involved in the choice of the ACPS concept do not include the usual performance index—impulse weight ratio. From the preceding discussion, it can be seen that there is little weight difference between the competitive concepts after the integrated systems are compensated for their support of the electrical power and environmental control systems. All the candidate systems have adequate functional capability and performance for a nominal tug mission. The primary evaluation factors are discussed in the following paragraphs:

8.3.2 EOS Compatibility

With respect to EOS compatibility, the oxygen/hydrogen systems rate much higher than the storable systems because of storable propellant toxicity, contamination, and corrosive properties. In the case of an abort, it might be necessary to dump propellants, which would be somewhat more simple with oxygen/hydrogen than with storables. It would also probably be simpler to dump from the main tanks only rather than from main tanks plus independent tanks.

8.3.3 Programmatic Factors

Space tug propulsion technology will be favored by any similarities in design choice with the EOS and space station programs. However, the EOS and space station choices are not now firm in many areas. The approach used in this study was to consider first the design alternatives irrespective of their status in order to determine their payoff. At that point, the development risk and cost of any promising technology advancement that is not certain for EOS or space station must be assessed twice, with and without the other program technology. At this time it is not possible to make those

Table 8-2. Low-Pressure ACPS Weights

Total Impulse	Metric Units (kg)				English Units (Lbm)			
	445,000 n-s		890,000 n-s		100,000 lb-sec		200,000 lb-sec	
	O_2	H_2	O_2	H_2	O_2	H_2	O_2	H_2
Propellant weight (from main tanks)	155.0	25.8	212.0	51.7	233	57	466	114
Storage tank penalty from main tanks	4.1	6.8	8.2	13.6	9	15	18	30
Conditioning unit								
Gas generators	5.5	5.5	5.5	5.5	12	12	12	12
Heat exchanger	6.8	9.1	6.8	9.1	15	20	15	20
Engine	227.0		227.0		500		500	
Lines and valves	38.1		38.1		84		84	
Total system weight	435.0		576.0		957		1271	
Effective burnout weight (Total weight - 1/2 usable propellant)	368.0		445.0		812		981	

Table 8-3. Pump-Fed ACPS Weights

Total Impulse	100,000 lb-sec (445,000 n-s)				200,000 lb-sec (890,000 n-s)			
	O_2 (lbm)	H_2	O_2 (kg)	H_2	O_2 (lbm)	H_2	O_2 (kg)	H_2
Propellant weight (from main tanks)	208	52	94.3	23.5	416	104	189.0	47.0
Storage tank penalty from main tanks	8	13	3.6	5.9	16	26	7.2	11.8
Accumulator	27	54	12.2	24.5	27	54	12.2	24.5
Insulation	3	5	1.4	2.3	3	5	1.4	2.3
Residual propellant	14	4	6.3	1.8	14	4	6.3	1.8
Conditioning unit and plumbing								
(2 ea) Turbopumps	31	45	14.0	20.0	31	45	14.0	20.0
Gas generators	12	12	5.4	5.4	12	12	5.4	5.4
Heat exchanger	15	20	6.8	9.1	15	20	6.8	9.1
Engines	260		118		260		118	
Lines and valves (in intelligence module)	22		10		22		10	
Total system weight	805		365		1086		493	
Effective burnout weight (total weight - 1/2 usable propellant)	675		306		826		375	

Table 8-4. Supercritical Storage ACPS Weights

Total Impulse	100,000 lb-sec (445,000 n-sec)				200,000 lb-sec (890,000 n-sec)			
	O_2 (lbm)	H_2	O_2 (kg)	H_2	O_2 (lbm)	H_2	O_2 (kg)	H_2
Propellant weight	225	57	102.0	26.0	450	115	204.0	52.1
Storage tank	70	175	32.0	79.0	140	350	63.5	159.0
Insulation	4	8	1.8	3.6	11	18	5.0	8.2
Heat exchanger	15	20	6.8	9.1	15	20	6.8	9.1
Gas generator	12	12	5.4	5.4	12	12	5.4	5.4
Engines	260		118		260		118	
Lines and valves	22		10		22		10	
Total system weight	880		399		1425		646	
Effective burnout weight (Total weight - 1/2 usable propellant)	750		340		1165		528	

Table 8-5. Bipropellant N_2O_4/MMH ACPS Weights

Total Impulse	75,000 lb-sec (333,616 n-s)		200,000 lb-sec (890,000 n-s)		300,000 lb-sec (1,334,000 n-s)	
	lbm	kg	lbm	kg	lbm	kg
Propellant weight	274	124.0	732	332.0	1097	498.0
Storage tanks	31	14.0	43	20.0	50	23.0
Helium storage	23	10.0	36	16.0	44	20.0
Helium	3	1.4	6	2.7	11	5.0
Lines and valves	30	13.6	30	13.6	30	13.6
Engines	190	86.2	190	86.2	190	86.2
Total system weight	551	250.0	1037	470.0	1422	645.0
Effective burnout weight (total weight - 1/2 usable propellant)	417	189.0	680	308.0	886	402.0

Table 8-6. Monopropellant Hydrazine ACPS Weights

Total Impulse	50,000 lb-sec (222,411 n-s)		150,000 lb-sec (667,233 n-s)	
	lbm	kg	lbm	kg
Propellant weight	260	118.0	775	354.0
Storage tank	12	5.4	33	14.9
Helium storage	15	6.8	45	20.4
Helium	3	1.4	5	2.3
Lines and valves	25	11.3	25	11.3
Engines	160	72.6	160	72.6
Total system weight	475	215.0	1043	473.0
Effective burnout weight (total weight - 1/2 usable propellant)	350	159.0	668	303.0

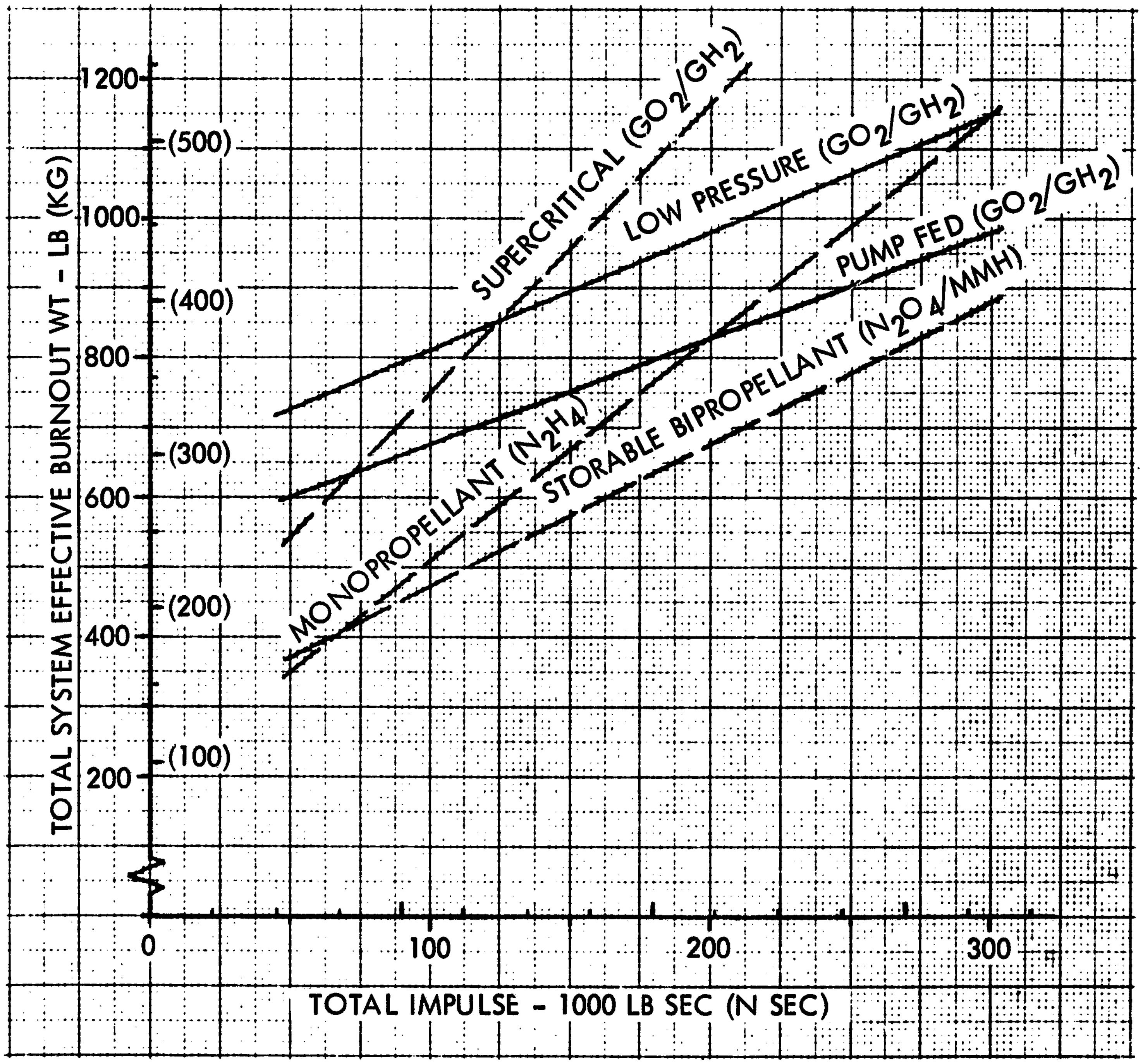


Figure 8-6. Total ACPS System Effective Burnout Weight Total Versus Impact

assessments. However, it is felt that for a Prephase A evaluation, all of the candidate systems are acceptable from a programmatic point of view. It must be noted, however, that storable system development would be considerably less expensive than the GO_2/GH_2 systems. These factors will require careful consideration and reevaluation in subsequent design phases.

8.3.4 Mission Operations Margin and Flexibility

The maximum operation margin and flexibility are obviously obtained by use of an integrated (replenishable) ACPS. This is true not only from the ACPS point of view, but also regarding life support, electrical power, and main propulsion. Growth of a system of this type is limited only by main tank propellant limitations. Of the two candidate integrated systems, the pump-fed system, although more complex, is selected as the most flexible because of better performance, compactness, and a higher propellant pressure, which allows more margin for throttling the engine thrust.

8.3.5 Maintainability, Reliability, and Operating Life

Design margin plus redundancy is the usual approach used to achieve mission success. Future long-duration missions will require drastic improvements in system capability, but the expected increase in component reliability will not be sufficient to achieve the mission requirements. Also, a point is reached where in-flight maintenance is a more effective method of meeting requirements than the use of redundancy. In the case of the space-based tug, it is expected that ACPS mission success will be achieved by a combination of hardware design, redundancy, and in-flight maintenance.

As mentioned, space tug maintenance and life requirements differ considerably from those of presently operational spacecraft. The attitude control propulsion systems are usually designed and qualified for one mission. Propellant lines and components are brazed in place, which makes component replacement difficult even on the ground. Another major problem in maintenance of an earth storable bipropellant system is the necessity to decontaminate the system before opening it to make repairs. This is necessary because the propellants are toxic and corrosive. The components are designed with extensive use of elastomeric materials, which are effective seals for relatively short missions but are not compatible with the earth storable propellants or space environment for any length of time. Because of these problems, the so called off-the-shelf components are generally not expected to be suitable for the space tug, but in the case of an expendable version an off-the-shelf storable system would be attractive. Because of the other problems associated with earth storable propellants, such as plume contamination, toxicity, interface complexity, low performance, and lack of flexibility (due to separate fixed propellant supply), an attempt to develop a system of this type with the necessary operating life and maintainability does not seem practical. The gaseous oxygen/gaseous hydrogen systems being developed for tug-associated programs will be designed with maintenance in mind and they have the major advantage of being nontoxic with clean exhaust products. This is important and possibly essential for space maintenance.

8.4 CANDIDATE COMPARISON

Table 8-7 presents relative comparisons of the five candidate systems in each of the evaluation categories. It can be seen from the table, as well as from the previous discussion, that there is no clearcut winner among the candidates. To select a baseline system it is necessary to assign priorities to the evaluation factors. In this study it is felt that maximum weight must be assigned to mission margin/flexibility, closely followed in importance by EOS compatibility and maintenance/operating life. Under these ground rules, the pump-fed GO_2/GH_2 system is the logical selection for the baseline ACPS. For an expendable tug, the evaluation factor weights would be considerably different, and the storable bipropellant would be selected.

8.5 BASELINE ACPS

The baseline pump-fed ACPS characteristics are shown in Table 8-8. The accumulator characteristics are given in Table 8-9 and general performance information is presented in Table 8-10. The accumulators were sized initially for a 60-second supply of propellant (with four engines firing), but later were reduced in size to a 20-second propellant supply after it was determined that the conditioning unit could respond within five seconds. The pressure regulators located downstream of the accumulators will be designed to regulate at a pressure of approximately 300 psia (207 n/cm^2) for normal operation with a throttling capability to allow thrust reduction to 40 pounds force (178 n) as required.

One of the major objections to the pump-fed ACPS is the need for a liquid-gas separation device. The ACPS could use a pump which would accept mixed flow, possibly a positive displacement type, but in the event of all vapor suction, the compression work is excessive.

A preliminary feasibility evaluation of a capillary device (in the main tanks) to supply propellant for the ACPS resulted in a design supplying approximately 200 pounds (90 kilograms) of propellant to the ACPS between any two, or after the last main engine burns.

An umbrella-type dutch twill retention screen of 30 by 250 mesh with pore openings of 0.00276 inch (0.07 millimeter) will assure propellant retention, but provide sufficient open area for low pressure loss during propellant flow and for rapid refilling. The retention screen is mounted on a conical

Table 8-7. Candidate ACPS Comparison

System	EOS Compatibility: Interface Complex	EOS Compatibility: Exhaust	EOS Compatibility: Prop. Hazard	Program Factors: Cost	Program Factors: Schedule	Program Factors: Commonality	Mission Margin/ Flexibility	Maint/ Operating Life	Main Tank Mod	Sep Tankage Reqd	Turbo Machinery	Gas Gen/ Heat Exch
Low-pressure GO_2/GH_2	Simple	Clean	Low	Medium	Medium	Not Presently Known	Good	Good	None Req'd	No	No	Yes
Pump-fed GO_2/GH_2	Simple	Clean	Low	Relatively High	Longest	Not Presently Known	Best	Average	Yes	No	Yes	Yes
Supercritical GO_2/GH_2	Simple	Clean	Low	Medium	Medium	Not Presently Known	Average	Above Average	No	Yes	None	Yes
Storable N_2O_4/MMH	Very Complex	Corrosive-toxic	Highest	Low	Short	None with EOS and Space Sta	Low	Poor	No	Yes	No	No
Storable N_2H_4	Complex	Relatively clean NH_3, N_2, H_2	Average	Lowest	Shortest	None with EOS and Space Sta	Poor	Low	No	Yes	No	No

Table 8-8. Baseline ACPS Characteristics

Gaseous oxygen/gaseous hydrogen
Pump-fed replenishable system
Capillary devices in main tanks for liquid feed to pumps
Accumulators provide 20 seconds propellant
Supply for four engines
Conditioning unit response 5 seconds
200-Pound (890 Newtons) thrust engines with throttling to 40 pounds (178 Newtons) thrust

Table 8-9. ACPS Accumulator Characteristics

	GO_2	GH_2
Storage pressure PSIA/(Kg/M^2)	1000.0/703.0 K	1000.0/703000.0
Storage supply temperature °R/(K)	380.0/211.0	200.0/111.0
Blowdown pressure PSIA/(Kg/M^2)	375.0/264.0 K	375.0/264.0 K
Blowdown temperature °R/(K)	460.0/256.0	242.0/134.0
Stored gas weight Lb/(Kg)	49.0/22.0	12.0/5.4
Storage volume Cu Ft (cu meters)	6.0/0.17	12.0/0.34
Max recharge/withdrawal rate lb/sec (kilograms/sec)	1.7/0.77	0.4/0.18
Residual gas weight lb (kilograms)	14.0/6.4	4.0/1.8

Table 8-10. ACPS Performance

	ACPS	Conditioner Exhaust Thrusters
Number of engines	20	4
Thrust each	200 lb to 40 lb (890 N to 178 N)	4 lb/17.8 N
Engine specific impulse	420 sec	216 sec
Net specific impulse		
Exhaust thrust cancelled	386 sec	--
Exhaust thrust used	399 sec	--
Chamber pressure	250 psia/144 K Kg/M^2	100 psia/70.3 K Kg/M^2
Area ratio	40	60
Mixture ratio (engine)	4.2	1.0
Mixture ratio - ACPS system	3.65	

frame which is advantageous for strength, refilling, and bubble purging during main engine burn. For adequate refilling of the compartment, the half cone angle should not exceed 80 degrees.

In addition to propellant retention, there is the problem of propellant acquisition and expulsion. For this function, a wheel-like propellant collector is provided within the retention compartment. The outside circumference of the wheel acts as the circumferential collector, the spokes are radial collectors and the hub is the central manifold and outlet. The wheel collector is made of thin tubular material (which has been "swiss-cheesed") wrapped with dutch twill mesh. Figure 8-7 illustrates this capillary device.

The collector is designed to provide ACPS propellant at the design flowrate of two pounds per second, and to collect propellant after gas breaks through into the compartment, but prevent gas passage into the collector.

When gas breaks through into the collector, the amount of trapped propellant is minimized by proper collector-compartment volume relationships.

The common tankage system includes a supply of cryogenics for main propulsion and a liquid-to-gas conversion system for ACS, EPS, and ECLSS. A schematic of the system, neglecting redundancy, is shown in Figure 8-8. Liquid oxygen and hydrogen are stored in the main tanks at 20 psia (13.8 Newtons per square centimeter). After conversion, the gases are stored in the accumulators at 350 to 1000 psia (240 - 689 Newtons per square centimeter). Through regulators, pressure is lowered to 300 psia (207 Newtons per square centimeter) for ACPS, 50 to 1000 psia (35 to 689 Newtons per square centimeter) for EPS, and 5.5 to 14.7 psia (3.8 to 10.1 Newtons per square centimeter) for ECLSS. In the Crew Module, an intermediate oxygen storage container is provided at the accumulator pressure.

Oxygen and hydrogen consumption by the subsystems may be calculated using the parametric data developed in the appropriate report subsections. Oxygen weight in the crew module, from Table 2-3, is

$$\left.\begin{array}{c} 99.0 \text{ Lb} + 0.362 \text{ Lb/D} + 1.840 \text{ Lb/M-D} \\ (44.9 \text{ Kg}) + (0.164 \text{ Kg/D}) + (0.835 \text{ Kg/M-D}) \end{array}\right\} \qquad (8\text{-}1)$$

where D is mission time in days and M is crew size.

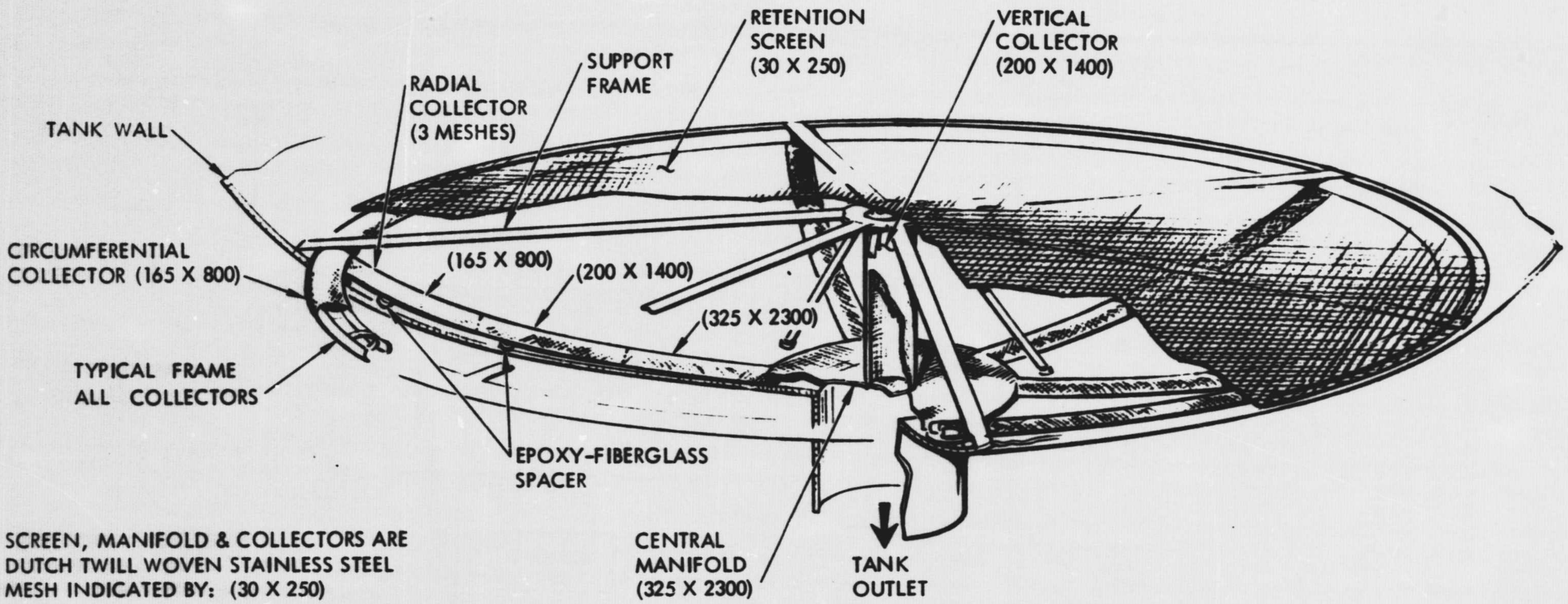


Figure 8-7. Capillary Device for ACPS Propellant Acquisition

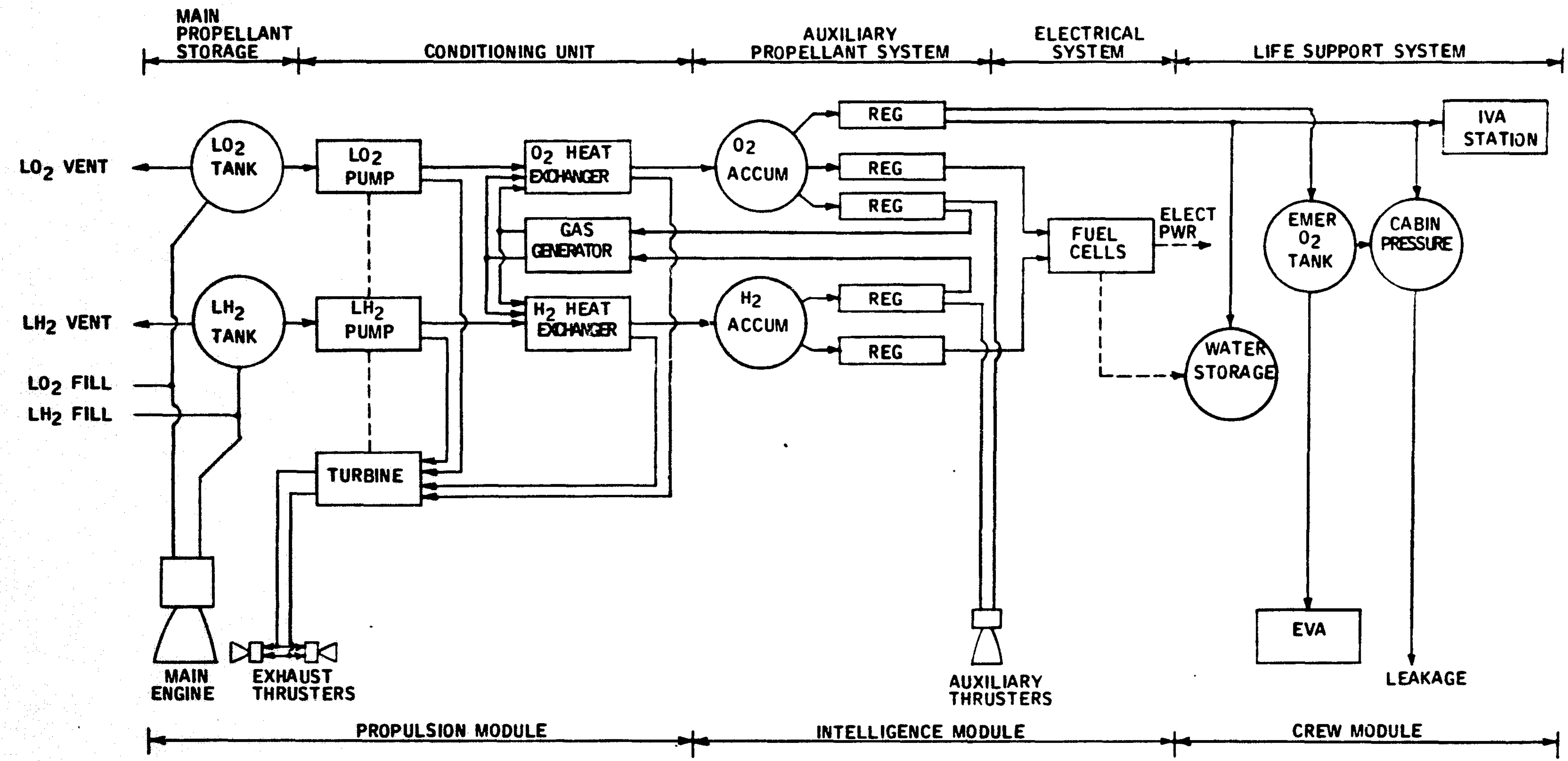


Figure 8-8. Common Tankage System

Fuel cell O_2/H_2 reactants are consumed at 0.845 pound/KWh (0.383 kilogram/KWh) (see Figure 5-8) using an 8:1 mixture ratio. ACS propellant usage may be calculated from the equations in Table 7-11 at a 4.2:1 mixture ratio. The conditioning unit consumes 8 percent of the total propellant passed to the accumulators, at a mixture ratio of 1:1.

The dependence of ACS propellant usage on mission-specific requirements does not permit the statement of a completely numerical expression of parametric oxygen and hydrogen consumption, similar to the scaling equations in other sections. Data for selected missions have been computed and are shown in Table 8-11. ACS propellant estimates in the table were obtained from Figures 7-7 and 7-8. Usable and residual propellants in the accumulator tanks were assumed to be constant throughout the mission, thus they did not enter into the calculations.

The ACS fixed equipment is summarized in Table 8-12.

Table 8-11. Oxygen/Hydrogen Weight Summary

Description	Space Missions																Lunar Landing			
	Unmanned 7 Days				6 Men 7 Days				Unmanned Concept 5 Slingshot Geosynchronous								4 Men 45 Days			
									First Stage				Second Stage							
	Oxygen		Hydrogen		Oxygen		Hydrogen		Oxygen		Hydrogen		Oxygen		Hydrogen		Oxygen		Hydrogen	
	Lb	(Kg)	Lb	(Kg)	Lb	(Kg)	Lb	(Kg)	Lb	(Kg)	Lb	(Kg)	Lb	(Kg)	Lb	(Kg)	Lb	(Kg)	Lb	(Kg)
Crew oxygen																				
Initial pressurization	-	-	-	-	22	10.0	-	-	-	-	-	-	-	-	-	-	22	10.0	-	-
Normal metabolic use	-	-	-	-	77	34.9	-	-	-	-	-	-	-	-	-	-	331	150.1	-	-
Leakage	-	-	-	-	3	1.4	-	-	-	-	-	-	-	-	-	-	16	7.3	-	-
Emergency, IVA and EVA use	-	-	-	-	77	34.9	-	-	-	-	-	-	-	-	-	-	77	34.9	-	-
Total crew oxygen	-	-	-	-	179	81.2	-	-	-	-	-	-	-	-	-	-	446	202.3	-	-
Electrical power reactants	129	58.5	16	7.2	331	150.1	42	19.1	8	3.6	1	0.45	129	58.5	16	7.3	2160	980.0	269	122
Auxiliary control propellant	362	164.2	91	41.3	362	164.2	91	41.3	72	32.7	18	8.16	362	164.2	91	41.3	690	313.0	172	78
Total subsystem usable propellant	491	222.7	107	48.5	872	395.5	133	60.4	80	36.3	19	8.61	491	222.7	107	48.6	3296	1495.3	441	200
Conditioning unit propellant	24	10.9	24	10.9	40	18.1	40	18.1	8	3.6	8	3.63	24	10.9	24	10.9	150	68.0	150	68
Total subsystem usable propellant	515	233.6	131	59.4	912	413.6	173	78.5	88	39.9	27	12.24	515	233.6	131	59.5	3446	1563.3	591	268
O_2/H_2 mixture ratio	3.93				5.30				3.26				3.93				5.83			

Table 8-12. Auxiliary Control and Propellant Subsystem Equipment

Description	Source	Unit Characteristics										
		Weight		Volume		Pwr (W)	Qty	Weight		Volume		Pwr (W)
		Lb	(Kg)	Ft^3	(m^3)			Lb	(Kg)	Ft^3	(m^3)	
Intelligence module equipment												
Engines	Bell	13.0	5.9	0.16	0.0045	0	20	260	117.9	3.20	0.0906	0
Lines and valves	Estimate	9.0	4.1	0.35	0.0099	0	2	17	7.7	0.70	0.0198	0
Accumulator tanks and insulation												
Oxygen	Estimate	-	-	-	-	0	2	30	13.6	6.00	0.1699	0
Hydrogen	Estimate	-	-	-	-	0	2	59	26.8	12.00	0.3398	0
Total IM ACPS equipment		-	-	-	-	0	0	367	166.0	21.90	0.6201	0
Propulsion module equipment												
Turbopump												
Oxygen	Aerojet & RK	15.5	7.0	0.35	0.0099	0	2	31	14.1	0.70	0.0198	0
Hydrogen	Aerojet & RK	22.5	10.2	0.35	0.0099	0	2	45	20.4	0.70	0.0198	0
Gas generator	Aerojet & RK	12.0	5.4	0.70	0.0198	0	2	24	10.9	1.40	0.0396	0
Heat exchanger												
Oxygen	Aerojet & RK	15.0	6.8	0.13	0.0037	0	2	30	13.6	0.25	0.0071	0
Hydrogen	Aerojet & RK	20.0	9.1	0.30	0.0085	0	2	40	18.1	0.60	0.0170	0
Lines and valves	Estimate	13.0	5.9	0.50	0.0141	0	2	26	11.8	1.00	0.0283	0
Total PM ACPS equipment		-	-	-	-			196	88.9	4.65	0.1316	0

9.0 RELIABILITY AND REDUNDANCY ANALYSIS

9.1 MAJOR COMPONENT REDUNDANCY RECOMMENDATIONS

Redundancy recommendations for 135 major components have been made in detail. The redundancy level varied from none to multiple depending on the particular subsystem criticality and reliability estimate. General agreement with the fail operational-fail safe redundancy level was marred by particular differences in redundancy nomenclature and configuration. Further evaluations need to be made of necessary additional, but not listed, equipment, e.g., structure, docking structure, failure detection sensors, meteoroid protection, the main propulsion system, lunar landing gear, and propellants. Table 9-1 contains a discussion of each component and an indication of the recommended redundancy based on judgment. The factor in the right hand column indicates the recommended redundancy level over the resulting interpretation of weight ratio with respect to the minimum operational unit.

Table 9-1. Subsystem Component Redundancy Recommendations

Component Description and Discussion	Redundancy Recommendation*
1. EC/LSS	
a. Crew module space mission fixed equipment	FS/1.1
(1) Furniture: Items should be designed to adequately meet requirements with minimum weight, e.g., bunks and control seats should be the same item; tables and control console should be the same item; and no chairs should be included. Potential for weight saving. Only a few items need be redundant; e.g., food preparation equipment.	
(2) Food management:	FS/1.1
(3) Water management:	FO-FS/1.5
(4) Waste management: Has the potential for producing severe emergencies. Those parts that could produce biological contamination should be made redundant. Those parts that could be bypassed by a "throw out the package" mode should be at least FS.	FS/1.5
(5) Temperature and humidity control: Failure could abort mission and lose crew.	FO-FS/1.67
(6) Atmospheric purifier: Again crew emergencies are all too possible.	FO-FS/1.67

*FS = Fail safe
FO = Fail operational
ND = Not defined
MR = Multiple-redundant
M = Margin
O = Operational

Table 9-1. Subsystem Component Redundancy Recommendations (Cont)

Component Description and Discussion	Redundancy Recommendation*
(7) Atmospheric pressure control: Crew critical function.	FO-FS/3
(8) Atmospheric circulation: Crew critical function.	FO-FS/3
(9) Atmospheric thermal loop: Moving parts should be multiple-redundant, but the heaviest parts (cold plates) need extra margin to prevent leakage as well as a backup, emergency mode. Total cold plate area need not be larger, just divided into two equal, independent loops and requiring special orientation. Tug will be sent back to Earth for cleaning following any thermal loop failure; so that there will be no necessity for multiple redundancy for continuing to succeeding missions.	FS/1.5
(10) EVA life support: Apollo-type equipment has become relatively standard. For the short time period, FS is all that is required.	FS/1.5
(11) Emergency life support: Contingency action in itself—no need for redundancy.	ND/1
(12) Nitrogen storage: Single redundancy for moving parts. Margin for tanks.	FS/1.25
(13) Interior lighting: Spare lights with fixture margin and single redundancy. Designed to be removed only when no voltage is applied.	FS/1.5

Table 9-1. Subsystem Component Redundancy Recommendations (Cont)

Component Description and Discussion	Redundancy Recommendation*
b. Crew module lunar landing additional fixed equipment	
(1) Furniture: Experiment station should not only be capable of completing missions, but also prevented from atmosphere contamination.	FO-FS/3
(2) Food management: Three smaller pieces should be used where any two would continue to succeeding missions, and any one is sufficient in emergency.	FO-FS/1.5
(3) EVA life support:	FS/1
c. Crew module space mission spare equipment [Spares need to be integrated with initial redundancy. Some spares are, indeed, necessary in all cases except emergency life support.]	
d. Crew module lunar landing additional spare equipment [The particular features which will be exercised more fully on lunar landing should be made more redundant or include more spares, e.g., temperature and humidity control and atmosphere thermal loop. In addition, special, particulate cleaning equipment needs to be added externally, in the airlock and internal to the pressure shell.]	

Table 9-1. Subsystem Component Redundancy Recommendations (Cont)

Component Description and Discussion	Redundancy Recommendation*
e. Crew module expendables and consumables - space missions	
(1) Housekeeping: Specific items need added pieces. Redundancy can be high because of low individual item weight, e.g., several extra trash bags.	MR/1.1
(2) Food management: Food can be stretched. Individual item weight is low, so that redundancy is high. Safe amount is zero, since rescue takes short time.	MR/1.1
(3) Waste management: Individual item is low, so that redundancy is high. Crew critical item..	MR/1.25
(4) Temperature and humidity control:	MR/1.25
(5) Atmospheric purification:	MR/1.25
(6) Atmospheric circulation: Margin for quantity. Crew critical.	MR/1.25
(7) Crew support: Small items, with multiple redundancy.	MR/1.1
(8) Nitrogen:	M/1.1
f. Crew module expendables and consumables - lunar landing missions	
[Same as for space missions except for items being used more on lunar surface:	
(1) Temperature and humidity control:	MR/1.5
(2) Atmospheric purification:]	MR/1.5

Table 9-1. Subsystem Component Redundancy Recommendations (Cont)

Component Description and Discussion	Redundancy Recommendation*
2. Guidance, navigation, and control subsystem	
a. Intelligence module basic equipment	
(1) Inertial measuring unit and processor: Small size makes meteoroid hit very unlikely. Gyro redundancy is adequate with Pentad. Care should also be taken to make the gyro environmental control adequately reliable (need further detail).	FO-FO-FS/2
(2) Gimbaled star tracker: Used as backup only. No redundancy necessary for reliability.	O/1
(3) Acquisition sun sensors and electronics:	O/1
(4) Horizon/earth tracker (edge tracker): Redundancy for high altitude already included in standard design of four heads.	O/1
(5) Navigation sensor base: Structure. Margin only. Environmental control needs some redundancy.	M/1. 1
(6) ACS driver amplifiers: Internally and multiple redundant.	FO-FO-FS/4
(7) Main engine gimbal amplifiers: Requires internal and multiple redundancy. Safe item only. Single engine requires two amplifiers per gimbal actuator. Four-engine configuration need have only a single amplifier per gimbal actuator. Each amplifier has selected internal redundancy.	FO-FS/3

Table 9-1. Subsystem Component Redundancy Recommendations (Cont)

Component Description and Discussion	Redundancy Recommendation*
(8) Signal distribution wiring:	FO-FS/2
b. Crew module basic equipment	
(1) Rotation control: Redundant with computer automatic and manual keyboard input rotational control as prime. Could be critical for close, delicate maneuvers.	FS/2
(2) Translation control: Redundant with computer automatic and manual keyboard input rotational control as prime. Could be critical for close, delicate maneuvers. ACS translation controls should be FS. Main engine translation controls need not be redundant.	FS/2
(3) Rate gyro triad: Backup function.	O/1
(4) Integrating acceleration:	O/1
(5) Semiautomatic telescope, controls, and base:	O/1
(6) Manual control electronics:	FS/1
(7) ACS mode select panel: Requires careful design. The correct ACS mode is very important since ACS is a critical function. Not only must access to the correct mode, but also egress from the unwanted mode be available. Judicious use of multiple mode, overriding switches could be a suitable, redundant method.	FS/1.1

Table 9-1. Subsystem Component Redundancy Recommendations (Cont)

Component Description and Discussion	Redundancy Recommendation*
(8) GN&C power control panel:	FS/1.1
(9) Signal distribution wiring:	FS/1.5
c. Crew module docking equipment	
(1) Laser rendezvous and docking radar: Not a radar. Laser on tug and corner reflector on passive docking target. Required for accurate measure of target distance and direction for efficient closing trajectory computation. Could use target or ground sources for position, or even crew controlled in a pinch. However, this is the normal docking mode and should last for the tug lifetime.	FO-FS/2
(2) Contact sensor: Required during each docking to signal initiation of latching mechanism operation. Susceptible to docking shock. Since it is light, a multiple redundant configuration would allow a voting logic and long life.	FO-FO-FS/4
(3) Signal distribution wiring:	FS/1.5
d. Intelligence module docking equipment	
(1) Laser rendezvous and docking radar:	FO-FS/2
(2) Television camera: Backup for any unplanned interference in the docking area.	FS/1
(3) Contact sensor:	FO-FO-FS/4
(4) Signal distribution wiring:	FS/1.5

Table 9-1. Subsystem Component Redundancy Recommendations (Cont)

Component Description and Discussion	Redundancy Recommendation*
e. Propulsion module docking equipment	
(1) Television camera:	FS/1
(2) Contact sensor:	FO-FO-FS/4
(3) Signal distribution wiring:	FS/1.1
f. Crew module lunar landing equipment	
(1) Main propulsion throttle control: Provide at least a separate throttle for each engine with provision for gang operation. For four engines, one throttle for each engine is sufficient. For one engine, two methods of throttling control are required, with reversion to full thrust on failure.	FO-FS/2
(2) Landing television graphics memory: Landing aid.	O/1
(3) Signal distribution wiring:	FS/2
g. Lunar landing kit - unmanned	
(1) Landing radar: Required for each landing. Internally and multiply redundant. Should adjacent radars fail, use opposite two's.	FO-FO-FS/2
(2) Gimbaled television camera: Backup for landing, but useful as viewers before and after.	FO-FS/2
(3) Signal distribution wiring:	FO-FS/2

Table 9-1. Subsystem Component Redundancy Recommendations (Cont)

Component Description and Discussion	Redundancy Recommendation*
h. Lunar landing kit - manned	
(1) Landing radar: Normal landing mode. Has crew control as backup.	FO-FO-FS/2
(2) Gimbaled television camera:	FS/1
(3) Signal distribution wiring:	FO-FS/2
3. Communications and data management subsystem	
a. Intelligence module equipment	
(1) S-band transmitter/receiver: Internal and external redundancy because this is the main source of data.	FO-FO-FS/2
(2) Omni-antenna: Operating case requires four antenna.	FO-FO-FS/1
(3) 2-Ft. Parabolic steerable antenna: Only high data rate, deep space communication means. Unmanned should have two. Manned need only have one.	FO-FS/2
(4) Communication switching and checkout control: Internally redundant.	FO-FO-FS/2
(5) Premodulation processor:	FO-FO-FS/2
(6) MOS-LSI input/output controller: Only means for controlling computers.	FO-FO-FS/3
(7) MOS-LSI processor:	FO-FO-FS/4
(8) Plated wire operational memory:	FO-FO-FS/2
(9) Tape archival storage memory:	FO-FS/1

Table 9-1. Subsystem Component Redundancy Recommendations (Cont)

Component Description and Discussion	Redundancy Recommendation*
(10) Plated wire mass storage memory:	FO-FO-FS/2
(11) Central timing unit: Internally redundant. Two sources per unit.	FO-FS/1
(12) Video unit: Has to work for unmanned operations.	FO-FS/2
(13) Remote acquisition and control units:	
256 Ch	FO-FO-FS/7
128 Ch	FO-FO-FS/2
64 Ch.	FO-FO-FS/4
(14) Signal distribution wiring and data bus: Individual pieces are normally redundant.	FS/1
b. Crew module equipment	
(1) VHF transmitter/receiver: One unit has dual performance capability.	FO-FS/1
(2) VHF antenna:	FO-FS/2
(3) Commander's console:	FO-FS/2
(4) Light-emitting diode alphanumeric display: Internally redundant display.	FO-FS/1
(5) Remote acquisition and control units: Vital to computer operations.	
256 Ch.	FO-FO-FS/3
128 Ch.	FO-FO-FS/1
64 Ch.	FO-FO-FS/2
(6) Signal distribution wiring and data bus:	FS/1

Table 9-1. Subsystem Component Redundancy Recommendations (Cont)

Component Description and Discussion	Redundancy Recommendation*
4. <u>Electrical power subsystem</u>	
a. Crew module equipment - Power distribution wiring:	FS/2
b. Intelligence module basic equipment	
(1) Fuel cell: Basic power source. Battery backup for FS.	FO-FO-FS/3
(2) Battery: Size so that any two will suffice for emergency attitude stabilization. Backup operation only.	MR
(3) Battery charger:	MR
(4) Inverter: AC power must be generated for ACS.	FO-FS/3
(5) Power controller: Internally redundant.	FO-FS/3
(6) Power distribution wiring: At least two circuits.	FS/2
c. Intelligence module added manned equipment	
(1) Fuel cell: FO-FO-FS/(3 basic +) 1 added.	FO-FO-FS/1
(2) Battery: Multiple redundant/ (3 basic +) 0 added.	MR/0
(3) Battery charger: Multiple redundant/ (3 basic +) 0 added.	MR/0
(4) Inverter: FO-FO-FS/(3 basic +) 1 added.	FO-FO-FS/1

Table 9-1. Subsystem Component Redundancy Recommendations (Cont)

Component Description and Discussion	Redundancy Recommendation*
(5) Power controller: FO-FO-FS/ (3 basic +) 1 added.	FO-FO-FS/1
(6) Power distribution wiring: At least two circuits.	FS/2
5. Active thermal control subsystem	
a. Intelligence module basic equipment - Coolant system: Vital to electrical power and electronic cooling. Moving parts need at least double redundancy, fixed cold plates must have two independent circuits.	FS/2
b. Intelligence module added manned equipment - Coolant systems: Vital.	FS/2
c. Propulsion module basic equipment	
(1) Space radiator (dual plumbing): Four independent panels, each with dual fluid paths. These eight smaller loops allow multiple redundancy through isolation valving. Leakage from any one must be detected and the flow valve closed. This will allow continuing the mission and proceeding to subsequent missions with partial system failure. The current radiator sizing at double the anticipated heat load provides for four redundant loops during normal operation with another redundant loop during emergency operation.	MR/2
d. Propulsion module added manned equipment (six man) - Space radiator:	MR/2

Table 9-1. Subsystem Component Redundancy Recommendations (Cont)

Component Description and Discussion	Redundancy Recommendation*
e. Propulsion module added lunar landing equipment - GH_2 Boiloff heat exchanger. Required for maximum cooling during lunar noon. Should also have dual loops.	FS/2
f. Crew module equipment (six man)	
(1) Coolant system (EC/LSS separate): Vital to crew module operation. Moving parts should have two circuits. Leakage inside the CM is deadly to the crew; therefore detection equipment must be sensitive and accurate. Leakage in a single loop is grounds for special maintenance (i.e., return to ground for thorough cleaning) because of possible corrosion in inaccessible corners. The second loop may allow completion of that mission and must allow return of the crew in a safe manner.	FS/2
(2) Space radiator: Four panels with two loops each.	MR/2
g. Lunar landing radiator kit unmanned equipment	
(1) Radiator panel and erection system (dual plumbing): Again four foldout panels with dual plumbing in each.	MR/2
(2) Coolant system: Closed, independent system from the IM. Electrical connection only. Required for lunar surface operations. Two loops at least.	FO-FS/2

Table 9-1. Subsystem Component Redundancy Recommendations (Cont)

Component Description and Discussion	Redundancy Recommendation*
h. Lunar landing radiator kit manned equipment	
(1) Radiator panel and erection system (dual plumbing):	MR/2
(2) Coolant system:	FO-FS/2
6. <u>Auxiliary control system</u>	
a. Intelligence module equipment	
(1) Engines: Critical system. ACS must work for mission success or retrieval success. Single mission duty cycle requires a large number of starts (on the order of 5000/mission). Designing for ten missions for no failure is difficult. Some redundancy is currently required to be able to continue missions with partial failure. Twenty engines give all the control needed in attitude and short translation. Sixteen engines will work, but require an additional axial engine for translation efficiency.	FO-FS/1.67
(2) Lines and valves: Each engine must have its own set of valves (non-redundant) with a fail-closed feature. Each engine must be interconnected with every other, so that the engine redundancy can be effective.	FO-FS/2

Table 9-1. Subsystem Component Redundancy Recommendations (Cont)

Component Description and Discussion	Redundancy Recommendation*
(3) Accumulator tanks and insulation: Oxygen. Critical component because of ACS, fuel cell, and EC/LSS supply. Provision of at least two accumulators is necessary to eliminate a single failure point. Backup is provided by direct access to conditioning unit outlet, bypassing failed accumulators.	FO-FS/2
(4) Accumulator tanks and insulation: Hydrogen. Critical equipment. Supplies ACS and electrical power. Unmanned operation could fail this entirely and still be safe, since oxygen gas in the ACS would be sufficient for ACS and batteries would supply the power. Manned operation requires at least one fuel cell to remain active for EC/LSS; e.g., backup is provided by direct access to conditioning unit outlet, bypassing failed accumulators.	FO-FS/2
b. Propulsion module equipment	
(1) Turbopumps: Oxygen and hydrogen. Two turbo pumps are enough because complete failure is still safe; i.e., the IM accumulators will supply enough O_2 and H_2 for retrieval or rescue.	FO-FS/2
(2) Gas generators:	FO-FS/2
(3) Heat exchanger: Oxygen.	FO-FS/2

Table 9-1. Subsystem Component Redundancy Recommendations (Cont)

Component Description and Discussion	Redundancy Recommendation*
(4) Lines and valves: Lines need be as redundant as the equipment they serve. Valves must be multiple-redundant, depending on function and individual reliability.	FO-FS/2
* FS = Fail safe FO = Fail operational ND = Not defined MR = Multiple-redundant M = Margin O = Operational	

10.0 PROPELLANT RELIQUEFACTION

The possible use of reliquefaction for conservation of propellants on the Space Tug has been studied. The purpose for such equipment would be to minimize boiloff over long periods for potential rescue missions, or for long term lunar base or lunar orbital storage. A study of both total and partial reliquefaction is included in this report. References 10-1 and 10-2 have been used to provide basic background on the types of power systems and reliquefiers which may be used. Reference 10-3 provides small size refrigerator scaling data on weight and power as a function of both temperature and refrigeration power to supplement data in reference 10-2.

For the purposes of this study the assumption was made that only a fuel cell power supply typical of the tug's basic power system would be considered. Both solar and nuclear power were excluded as being outside of the range of interest for the tug (too heavy, and requiring special orientation control, etc.). Such power supplies may be applicable, however, to the Space Station, orbital propellant depot, lunar base or any other interfacing system which could supply reliquefaction capability when the tug is docked.

The results show that total reliquefaction is not reasonable when constrained to the use of fuel cell power. Consumption of 5 PPH/2.3 KGPH of H_2/O_2 is required to reliquefy 1 PPH/0.45 KGPH of hydrogen and about 0.8 PPH/0.36 KGPH to reliquefy 1 PPH/0.45 KGPH of oxygen. Thus, unless other energy sources are used, total reliquefaction is not advantageous.

The results for partial reliquefaction are more favorable since some of the boiloff is used as a natural refrigerant and improves the thermal cycle. About 0.41 PPH/0.19 KGPH H_2 and 0.52 PPH/0.24 KGPH O_2 can be reliquefied from 1 PPH/0.45 KGPH boiloff of each reactant. It was assumed that fuel cell reactants were extracted from the boiloff gases and that the oxygen, not used for the fuel cell, was fully reliquefied. The added equipment weight is 120 lb/54 KG (not including extra fuel cells). When this delta is increased by the burned weight-to-propellant exchange factor of 10.4 (representing the single stage synchronous orbit mission) it overbalances the boiloff savings until break even standby times of 56 and 62 days for minimum propellant and minimum gross weight criteria respectively, are reached.

This analysis has been based on an integral propellant tank design concept for a 60 K lb/27 K kg propellant tug. Figure 10-1 shows total propellant boiloff as a function of insulation thickness. This curve is valid for low orbits or the lunar surface over the day-night average. The integral hydrogen and oxygen tanks are assumed to have boron-epoxy skirts along with NARSAM insulation on the tanks. In each case a forward and aft skirt design is assumed. For a boil-off rate of 1 PPH/0.45 KGPH each the oxygen and hydrogen tanks insulations are 1.7 in./4.44 cm and 1.3 in./ 3.5 cm thick respectively. No attempt was made to optimize this although past effort has shown that between 1.0 and 2.0 inches (2.54 and 5.1 cm) is best for a synchronous equatorial orbit, 30-day mission. Since the curve is sharply exaggerated in slope, it would appear that the boil-off of 1 PPH/ 0.45 KGPH for each propellant would be a reasonable baseline.

Figure 10-2 describes the total reliquefaction refrigeration weight and power requirements as a function of insulation thickness. The analysis has assumed a reversed Brayton cycle refrigerator rejecting its heat at 540 R/ 300 K. Reference 10-2 contains cycle performance estimates of a number of practical refrigerators although data on working units is still sparse. For 1 PPH/0.45 KGPH boil-off rate, the hydrogen and oxygen total reliquefiers require 5.5 KW and 0.8 KW respectively. Assuming the fuel cell power source, the reactant consumption is about 1 lb/hr/KW (0.45 kg/ hr/KW); thus it would require 5.5 PPH/2.5 KGPH of reactants to reliquefy 1 PPH/0.45 KGPH of hydrogen and 0.8 PPH/0.36 KGPH to reliquefy 1 PPH/ 0.45 KGPH of oxygen which is not a bargain.

Figure 10-2 also shows that in addition to the fuel cell expendables it is necessary to add in a fixed weight of 250 lb/113 kg for a hydrogen reliquefier and a weight of 35 lb/16 KE for an oxygen reliquefier. While it is unlikely that the tug will use total reliquefaction, the orbital propellant depot, the Space Station and the lunar base all may potentially incorporate large solar or nuclear power supplies that may reduce the cost of power. Considering the cost of transporting propellant into orbit (about $200/lb-$441/KE) and the added cost of transporting it to a lunar base, it is reasonable to conclude that eventually reliquefaction will be provided as an economic measure. For example, the loss of 50,000 lb/22,680 kg propellant/year from the orbital propellant depot (about 5.5 PPH/2.5 KGPH total) could cost $\$10^6$/yr. A refrigerator reliquefying 4.0 PPH/1.8 KGPH H_2 and 1.5 PPH/0.68 KGPH O_2 would require 14.5 KW of power and weigh 700 lb (650 + 50)/317 kg (295 + 23). If a nuclear or solar power supply could be obtained providing 300 lb/KW 136 kg/KW, the total system weight would be about 5000 lb (4350 + 700)/2290 kg (1973 + 317). This is off in the future but is interesting to contemplate from the vantage point of tug reusability and long term storage.

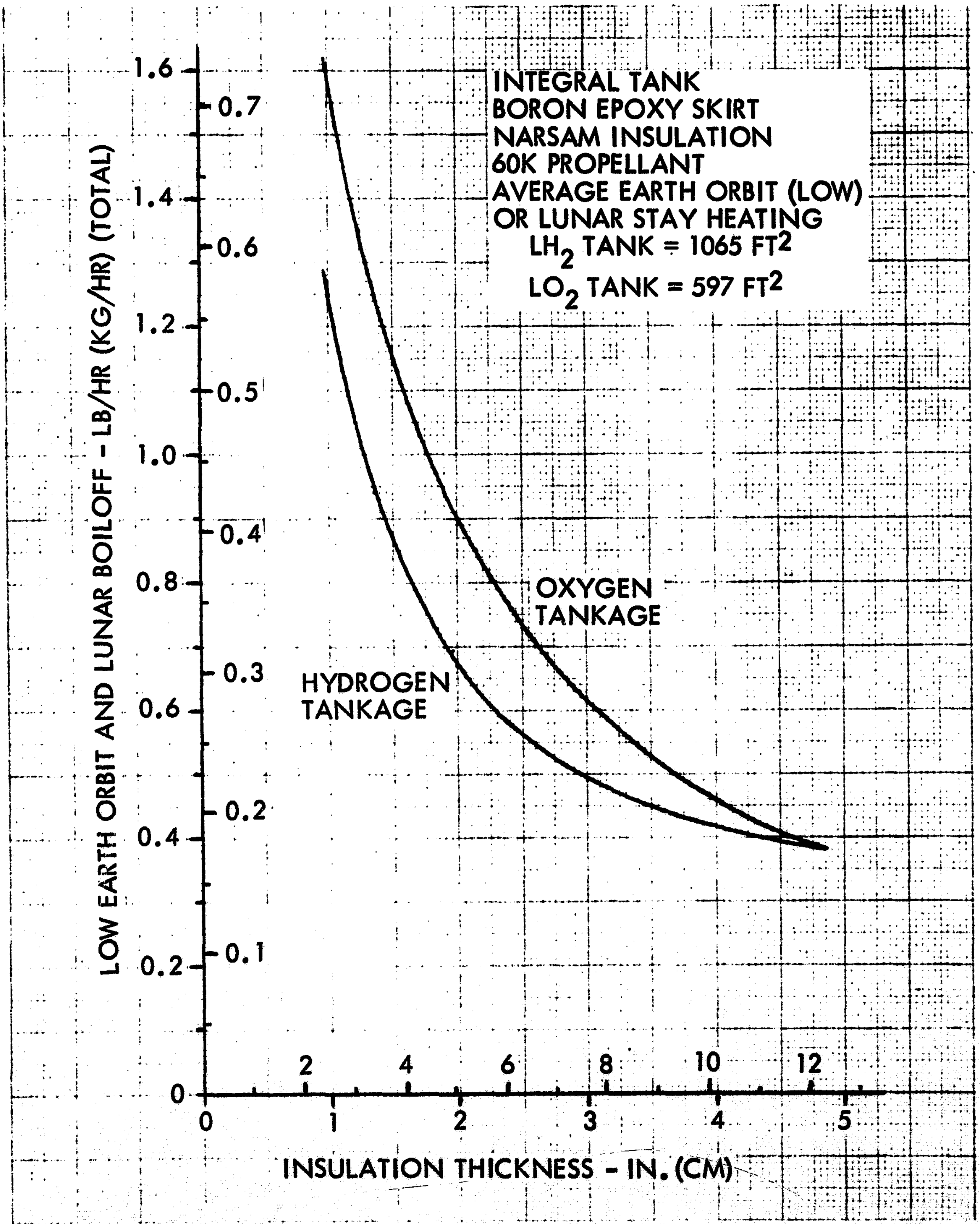
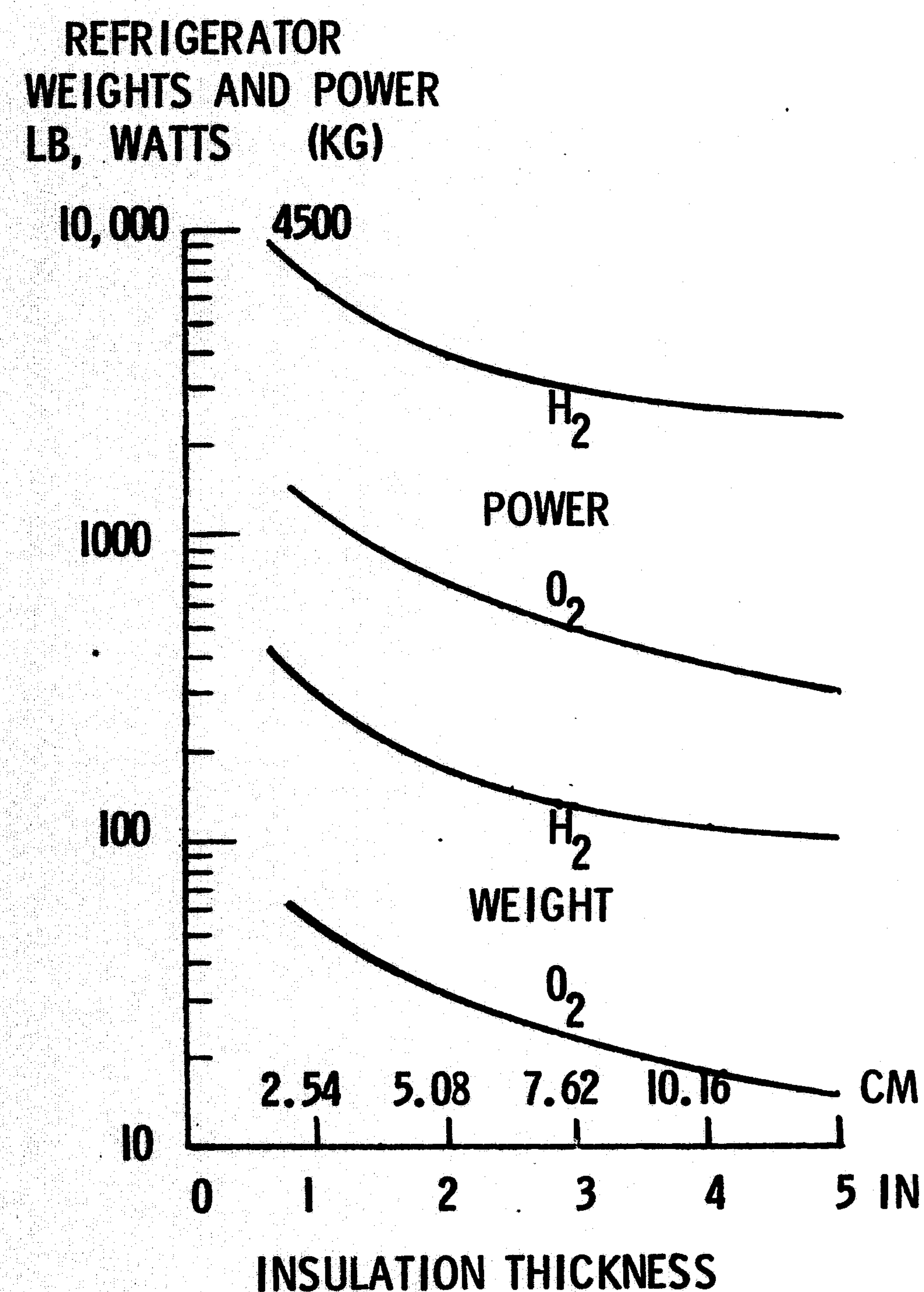


Figure 10-1. Propellant Boiloff as Function of Insulation Thickness



- REVERSE BRAYTON CYCLE REFRIG.
- REJECTION TEMPERATURE = 540°R
- INTEGRAL TANKS, 60K PROPELLANT
- NARSAM INSULATION

O_2, H_2 BOILOFF 1 LB/HR EACH			
INSULATION THICKNESS		REFRIGERATOR WEIGHT	REFRIGERATOR POWER
O_2	1.75 INCHES	35 LB	800W
H_2	1.30 INCHES	250 LB	5500W

- REFRIG WT EQUIV TO 6-DAY BOILOFF LOSS
- REFRIG PWR RQMT IMPRACTICAL FOR TUG
- MORE APPROPRIATE FOR SPACE STATION OR LUNAR BASE

Space Division
North American Rockwell

Figure 10-2. Active Refrigeration/Reliquefaction Weight and Power

The analysis of the partial reliquefier was initiated because the constraint of a fuel cell power supply requires that the hydrogen and oxygen reactants be obtained from the boil-off. The work presented in reference 10-2 shows that based on the performance of reasonable heat exchanges and turbo-machinery, including some para to ortho hydrogen conversion, that 41.3 percent of the boil-off hydrogen can be reliquefied. For the typical tug boil-off of 1 lb/hr for each propellant, the hydrogen reliquefaction requires 0.13 KW of power and the unit will weigh about 90 pounds. Balancing out the fuel cell requirements for O_2 and H_2 indicates that about 0.52 lb/hr of O_2 is reliquefied and the rest of the O_2 is consumed powering the reliquefiers. The total power requirement is about 530 watts for both systems, producing 0.41 lb/hr LH_2 and 0.52 lb/hr LO_2, for a total supply of 0.53 lb/hr of reactants to the fuel cells and a loss of about 0.45 lb/hr of hydrogen which is unrecoverable. Over a period of 30 days, a total of 1440 pounds of propellant would be vented; however, with partial reliquefaction, about 670 pounds can be recovered. The basic problem, however, is that the added 130 pounds of empty weight is magnified 10.4 times in terms of propellant so that no net saving is available. This large exchange coefficient makes it very difficult to incorporate the added fixed weight of any thermal protection scheme. It is interesting to note in Table 10-1 that for larger boil-off amounts the fraction reliquefied is increased because of the reduction in specific power. For example, if 10 lb/hr of H_2 and O_2 are to be partially reliquefied the 4.1 lb/hr of H_2 is summed with 7.5 lb/hr of O_2 for a total of 11.6 lb/hr and a total power of 3.6 KW.

The basic conclusion is that the use of reliquefaction on the Space Tug does not appear very suitable unless it can be made disposable, or a plug-in module attached to or operated by permanent bases.

Table 10-1. Partial Reliquefaction of Hydrogen

Boiloff [lb/hr (kg/hr)]	Reliquef [lb/hr (kg/hr)]	Reference Power (watts)	Specific Power $\left(\frac{\text{watts}}{\text{watts ref}}\right)$	Power (watts)	Unit Weight [16 (kg)]	Fuel Cell Reactant Consumption [lb/hr (kg/hr)]
1 (0.45)	0.41 (0.19)	23.1	5.5	130	86 (39)	0.13 (0.059)
2 (0.90)	0.82 (0.37)	42.2	4.2	194	130 (59)	0.19 (0.086)
4 (1.81)	1.64 (0.74)	92	3.3	300	200 (90)	0.30 (0.136)
10 (4.53)	4.13 (1.87)	231	2.4	500	375 (170)	0.55 (0.250)

11.0 MANIPULATORS

Firm requirements for manipulators to be used in conjunction with the tug have not yet been identified. In the operational time period most likely to be used for tug deployment (the 1980's), it seems reasonable to assume that contemporary satellites and experiment modules that require servicing will incorporate appropriate hard-docking provisions. This provision will permit reliable connection and even pressurized access to servicing areas of the satellite. However, there may be some older satellites still in service that do not have docking provisions. Unanticipated maintenance in the vicinity of a space base also could require the use of manipulators.

A brief review of nine pertinent reports on manipulators tends to substantiate the approach previously described. Matrix Research Company contributed one of the reports, which was generated for a related NASA MSFC contract (Reference 11-1). Table 11-1, which summarizes a survey of present technology, was taken from the MRC report. In addition, various in-house specialists were consulted. Although many of the topics covered by the reports related to nonspace applications of manipulators, the following appropriate considerations may be drawn from them:

1. Unless a satellite is specifically designed for physical contact, even contacts conducted with extreme care are likely to result in damage (although this is not always of great consequence). Furthermore, the tug approach to an active satellite should be planned so that the satellite is deactivated or controlled from a communication link. If such remote control is impossible, care should be taken to avoid occulting the satellites' sun-star sensors, because the unprogrammed loss of lock may initiate a search mode involving angular rates.

2. The incorporation and operation of manipulators requires appreciable electrical power and a considerable amount of specialized crew interface equipment.

3. Even simple maintenance of a satellite becomes extremely complex when all of the shutdown, removal, deployment, replacement, reactivation, and checkout operations are considered.

Table 11-1.

Manipulator	Class: Unilateral	Class: Bilateral	Slave Power: Manual	Slave Power: Electric	Slave Power: Electrohydraulic	Typical Installation	Specifications: Six Degrees	Less Than Six Degrees	Number of Joints	Reach (Inches)	Meters	Load (Pounds)	Kilogram	Tong Opening (Inches)	Centimeters	Gripping Force (Pounds)	(N)	Slave One Arm Weight (Pounds)	Kilograms	Status: Developed After 1962	Prototype	Catalog	Operational	Inactive	Remarks
ACF Industries, Inc. and Underwater Research Corporation	x				x	Trieste II		x	6	120.0	3.10	500	226.8	10.0	25.4			2390	1084.1						
American Machine and Foundry Company																									
Model S standard duty		x	x	a		Hot lab/hazardous location	x		7	97.0	2.46	10	4.5	3.2	8.1	20	89.0	500[b]	227.8			x			a) Y indexing; X optional
Model S standard duty (extended reach)		x	x	a		Hot lab/hazardous location	x		7	136.0	3.45	10	4.5	3.2	8.1	20	89.0	580[b]	263.1			x			a) Y indexing; X and Z optional
Heavy duty		x	x	a		Hot lab/hazardous location	x		7	97.0	2.46	40	18.1	3.2	8.1	60	266.9	500[b]	227.8			x			a) Y indexing; X optional
Heavy duty (extended reach)		x	x	a		Hot lab/hazardous location	x		7	136.0	3.45	40	18.1	3.2	8.1	60	266.9	580[b]	263.1			x			a) Y indexing; X and Z optional
Light duty (extended reach)		x	x	a		Hot lab/hazardous location	x		7	73.0	1.85	7	3.2	3.2	8.1	10	44.5	175[b]	79.4			x			a) Y indexing
Minimanipulator		x	x	a		Hot lab/hazardous location	x		7	36.0	0.91	5	2.3	3.0	7.6	10	44.5	16[b]	7.3			x			b) Master and slave
Argonne National Laboratory																									
Model 1		x	x			Hot laboratory	x																		
Model 4		x	x			Hot laboratory	x		7			5	2.3			10	44.5							x	
M7		x	x			Hot laboratory	x		7			5	2.3			10	44.5	30	13.6				x		
M8		x	x	a		Hot laboratory	x		7			20	9.1			20	89.0						x		a) XYZ indexing
E1		x		AC		Hot laboratory	x		7																
E2		x		AC		Hot laboratory	x		7			8	3.6					200	90.7						
E3		x		AC[b]		Hot laboratory	x		7			50	22.7	3.0	7.6	58	258.0	550	249.5				x		a) Z indexing
E4A		x		AC		Hot laboratory	x		7			50	22.7	3.0	7.6	75	333.6	375	170.1	x	x				
Casteau (France)	x				x	Diving Saucer SP-300		x		40.0	1.02	25	11.3	a				35	15.9						a) Clamshell
Central Research Laboratory Inc.																									
Model 7		x	x			Hot lab-explosive location[a]	x		7	50.0	1.27	10	4.5	3.2	8.1	10	44.5	60	27.2			x			a) Over-wall type
Model 8		x	x	a		Hot lab-explosive location	x		7	103.0	2.62	20	9.1	3.2	8.1	20	89.0	225	102.1			x			a) Y indexing; X optional
Model A		x	x	a		Hot lab-explosive location[b]	x		7	98.0	2.49	20	9.1	3.2	8.1	20	89.0	550	249.5			x			a) Y indexing; X optional b) Gastight
Model B		x[a]	x			Hot lab-canal		x	4			100	45.4	3.2	8.1	100	444.8	70	31.8			x			a) In wrist and tong motions
Model D		x	x	a		Hot lab-explosive location	x		7	103.0	2.62	100	45.4	3.2	8.1	100	444.8	400	181.4			x			a) Y indexing; X optional
Model E		x	x	a		Hot lab-explosive location	x		7	138.0	3.51	20	9.1	3.2	8.1	20	89.0	425	192.8			x			a) YZ indexing; X optional
Model F		x	x	a		Hot lab-explosive location	x		7	135.0	3.43	100	45.4	3.2	8.1	100	444.8	600	272.2			x			a) YZ indexing; X optional
Model G		x	x	a		Hot lab-explosive location	x		7	65.0	1.65	10	4.5	3.2	8.1	10	44.5	150	68.0			x			a) Y indexing; X optional
Model H		x	x	a		Hot lab-explosive location	x		7	50.0	1.27	10	4.5	3.2	8.1	10	44.5	110	49.9	x		x			a) Y indexing
Model J		x	x	a		Hot lab-explosive location[b]	x		7	135.0	3.43	20	9.1	3.2	8.1	20	89.0	650	294.8	x		x			a) YZ indexing; X optional b) Gastight
Comitato Nazionale per L'Energia Nuclear (Italy)																									
Mascot - 1		x		AC	x[b]	Hot lab-explosive location	x		7	38.0	0.97	50	22.7	3.0	7.6			1700[a]	771.1	x	x				a) Two arms and mobile base b) Lifting column and trolley
General Dynamics Corporation																									
Model 3BH	x				x	Submersibles		x	6	48.6	1.23	85	38.6	6.0	15.2	500	2224.1	465	210.9	x		x			
Model 4BH	x				x	Submersibles		x	6	57.4	1.46	85	38.6	6.0	15.2	500	2224.1	505	229.1	x		x			
Model 5BH	x				x	Submersibles		x	6	63.5	1.61	85	38.6	6.0	15.2	500	2224.1	540	244.9	x		x			
Model 6BH	x				x	Submersibles		x	7	63.0	1.60	85	38.6	6.0	15.2	500	2224.1	540	244.9	x		x			
Model 7BH	x				x	Submersibles		x	8	69.6	1.77	85	38.6	6.0	15.2	500	2224.1	575	260.8	x		x			
Model 8BH	x				x	Submersibles		x	7	74.2	1.88	85	38.6	6.0	15.2	500	2224.1	580	263.1	x		x			
Prototype	x				x	Submersibles		x	6	60.0	1.52	100	45.4	3.2	8.1	500	2224.1	150	68.0	x	x				
General Electric Company																									
Model AMD	x				x	Aluminaut		x	6	109.0	2.77	200	90.7	6.0	15.2	250	1112.1	1705[a]	773.4	x			x		a) Two arms
Hanford		x	x			Hot laboratory	x		8	48.0	1.22	20	9.1	4.0	10.2	20	89.0	100	45.4					x	
Handyman		x			x	Hot laboratory	x		10[a]	60.0	1.52	100	45.4	5.0	12.7	150	667.2	550[b]	249.5					x	a) Special hand actions b) Two arms
Model III-H	x			x		Hot laboratory		x	6	105.[a]	2.67	600	272.2	5.4	13.7	1500	6672.3					x			a) Extended
O-man	x			x		Hot laboratory	x		9	72.0	1.83	500	226.8	6.0	15.2	500	2224.1	1000	453.6				x		
Model III-H on Nerva boom						Hot laboratory			10	483.0	12.27	600	272.2	5.4	13.7	1500	6672.3							x	a) Uses Model III-H manipulator
G-man II		x	x			Hot laboratory	x		8	79.0	2.01	17	7.7	3.5	8.9	20	89.0	530	240.4					x	
General Mills, Inc.[a]																									a) General Mills Manipulator Business Sold to PAR
Model 100	x			x		Hot laboratory		x	4	18.0	0.46	40	18.1	4.0	10.2	50	222.4	400	181.4						
Model 150 (modified)[a]	x			DC		Trieste		x	6	39.0	0.99	50	22.7	2.0	5.1			110	49.9		x				a) Modified by Litton Industries
Model 150	x			DC		Alvin		x	6	63.5	1.61	50	22.7	4.0	10.2	100	444.8	450	204.1		x				
Model 300	x			x		Hot laboratory			5	37.0	0.94	65		5.0	12.7	150	667.2	175	79.4						
Model 500[a]	x			DC	x[b]	Remote underwater manipulator		x	5	180.[b]	4.57	500	226.8	6.0	15.2						x				a) Modified b) Hydraulic boom

FOLDOUT FRAME 2

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Table 11-1. Manipulators — Survey of Literature

Model 7		x	x			Hot lab-explosive location[a]	x		7	50.0	1.27	10	4.5	3.2	8.1	10	44.5	60	27.2			x			a) Over-wall type
Model 8		x	x	a		Hot lab-explosive location	x		7	103.0	2.62	20	9.1	3.2	8.1	20	89.0	225	102.1			x			a) Y indexing; X optional
Model A		x	x	a		Hot lab-explosive location[b]	x		7	98.0	2.49	20	9.1	3.2	8.1	20	89.0	550	249.5			x			a) Y indexing; X optional b) Gastight
Model B		x[a]	x			Hot lab-canal		x	4			100	45.4	3.2	8.1	100	444.8	70	31.8			x			a) In wrist and tong motions
Model D		x	x	a		Hot lab-explosive location	x		7	103.0	2.62	100	45.4	3.2	8.1	100	444.8	400	181.4			x			a) Y indexing; X optional
Model E		x	x	a		Hot lab-explosive location	x		7	138.0	3.51	20	9.1	3.2	8.1	20	89.0	425	192.8			x			a) YZ indexing; X optional
Model F		x	x	a		Hot lab-explosive location	x		7	135.0	3.43	100	45.4	3.2	8.1	100	444.8	600	272.2			x			a) YZ indexing; X optional
Model G		x	x	a		Hot lab-explosive location	x		7	65.0	1.65	10	4.5	3.2	8.1	10	44.5	150	68.0			x			a) Y indexing, X optional
Model H		x	x	a		Hot lab-explosive location[b]	x		7	50.0	1.27	10	4.5	3.2	8.1	10	44.5	110	49.9	x		x			a) Y indexing
Model J		x	x	a		Hot lab-explosive location[b]	x		7	135.0	3.43	20	9.1	3.2	8.1	20	89.0	650	294.8	x		x			a) YZ indexing; X optional b) Gastight
Comitato Nazionale per L'Energia Nuclear (Italy)																									
Mascot - I		x		AC	x[b]	Hot lab-explosive location	x		7	38.0	0.97	50	22.7	3.0	7.6			1700[a]	771.1	x	x				a) Two arms and mobile base b) Lifting column and trolley
General Dynamics Corporation																									
Model 3BH	x				x	Submersibles		x	6	48.6	1.23	85	38.6	6.0	15.2	500	2224.1	465	210.9	x		x			
Model 4BH	x				x	Submersibles		x	6	57.4	1.46	85	38.6	6.0	15.2	500	2224.1	505	229.1	x		x			
Model 5BH	x				x	Submersibles		x	6	63.5	1.61	85	38.6	6.0	15.2	500	2224.1	540	244.9	x		x			
Model 6BH	x				x	Submersibles		x	7	63.0	1.60	85	38.6	6.0	15.2	500	2224.1	540	244.9	x		x			
Model 7BH	x				x	Submersibles		x	8	69.6	1.77	85	38.6	6.0	15.2	500	2224.1	575	260.8	x		x			
Model 8BH	x				x	Submersibles		x	7	74.2	1.88	85	38.6	6.0	15.2	500	2224.1	580	263.1	x		x			
Prototype	x				x	Submersibles		x	6	60.0	1.52	100	45.4	3.2	8.1	500	2224.1	150	68.0	x	x				
General Electric Company																									
Model AMD	x				x	Aluminaut		x	6	109.0	2.77	200	90.7	6.0	15.2	250	1112.1	1705[a]	773.4	x			x		a) Two arms
Hanford		x	x			Hot laboratory	x		8	48.0	1.22	20	9.1	4.0	10.2	20	89.0	100	45.4					x	
Handyman		x			x	Hot laboratory	x		10[a]	60.0	1.52	100	45.4	5.0	12.7	150	667.2	550[b]	249.5					x	a) Special hand actions b) Two arms
Model III-H	x			x		Hot laboratory		x	6	105.[a]	2.67	600	272.2	5.4	13.7	1500	6672.3					x			a) Extended
O-man	x			x		Hot laboratory	x		9	72.0	1.83	500	226.8	6.0	15.2	500	2224.1	1000	453.6				x		
Model III-H on Nerva boom						Hot laboratory			10	483.0	12.27	600	272.2	5.4	13.7	1500	6672.3							x	a) Uses Model III-H manipulator
G-man II		x	x			Hot laboratory	x		8	79.0	2.01	17	7.7	3.5	8.9	20	89.0	530	240.4					x	
General Mills, Inc.[a]																									a) General Mills Manipulator Business Sold to PAR
Model 100	x			x		Hot laboratory		x	4	18.0	0.46	40	18.1	4.0	10.2	50	222.4	400	181.4						
Model 150 (modified)[a]	x			DC		Trieste		x	6	39.0	0.99	50	22.7	2.0	5.1			110	49.9		x				a) Modified by Litton Industries
Model 150	x			DC		Alvin		x	6	63.5	1.61	50	22.7	4.0	10.2	100	444.8	450	204.1		x				
Model 300	x			x		Hot laboratory			5	37.0	0.94	65		5.0	12.7	150	667.2	175	79.4						
Model 500[a]	x			DC	x[b]	Remote underwater manipulator		x	5	180.[b]	4.57	500	226.8	6.0	15.2						x				a) Modified b) Hydraulic boom
Model 500	x			x	x[a]	Hot laboratory		x	5	134.[a]	3.40	280	127.0	7.5	19.1	280	1245.5	750	340.2						a) Hydraulic boom
Model 550	x			x		Hot laboratory		x	7	132.5	3.37			5.0	12.7	200	889.6								
Model 700	x					Hot laboratory		x	5	114.0	2.90			3.5	8.9	1500	6672.3								
General Motors Defense Research Labs				x		Deep Ocean Work Boat (DOWB)	x			49.0	1.24	120	54.4	4.0	10.2	120	533.8			x					80 in. pounds of torque
Grumman Aircraft Engineering Co.						GSV-1 submersible			8	200.0 92.[a]	5.08 2.34	300 1120[a]	136.1 508.0								x				a) At maximum position
International Underwater Research Corporation	x				x	Recoverer I		x	6	120.0	3.05	500	226.8	8.0	20.3										
LIP Company (France)				x		Hot laboratory		x	4			4	1.8	4.9	10.2	10	44.5								
Lockheed Missile & Space Company																									
						DSRV			7	110.0	2.79			4.0	10.2	2000	8896.4								
					x	Deep quest			7	107.0	2.72	500	226.8	8.0	20.3	[illegible]	2224.1								
Naval Electronics Laboratory	x				x	Trieste II		x	4	180.0	4.57	500	226.8	18.0[a]	45.7			1330	603.3					x	a) Clamshell
North American Aviation, Inc.																									
Model 2	x				x	Submersibles	x		8	72.0	1.83	50	22.7	4.0[a]	10.2	2000[a]	8896.4	240	108.9						a) Hook terminal device
Nuclear Equipment Limited (England)																									
Model 8						Hot laboratory						20	9.1												
Model 9						Hot laboratory						96	43.5												
Programmed and Remote Systems Corporation																									
Model 1000	x[a]			AC		Hot laboratory			4	27.2	0.69	80	36.3	5.0	12.7	200	889.6	90	40.8		x				
Model 2000	x[a]			AC		Hot laboratory		x	6[b]	37.5	0.95	150	68.0	5.0	12.7	200	889.6	150	68.0		x				a) Motion and position correspondence controller available. Wrist rotation and grip on a rate basis.
Model 3000	x[a]			AC		Hot laboratory		x	7[b, c]			150	68.0	5.0	12.7	200	889.6	170	77.1		x				
Model 3500	x[a]			AC		Hot laboratory		x	6[c]	39.0	0.99	100	45.4	4.0	10.2	150	667.2	110	49.9		x				
Model 3500 underwater	x[a]			DC		Submersibles		x	6	49.0	1.24	50	22.7	4.0	10.2	100	444.8	185	83.9	x	x				
Model 6000	x[a]			AC		Hot laboratory		x	6	56.0	1.42	400	181.4	8.0	20.3	500	2224.1	270	122.5		x				b) Wrist pivot optional c) Wrist extension
Tokyo Shibaura Electric (Japan)																									
Type TP-1		x	x			Hot laboratory			5					4.9	12.4	155	689.5								
Westinghouse Electric Corporation	x				x	Deep Star		x		40.0	1.02	25	11.3	a				62	28.1	x					a) Three-lobes clamshell

From the previous considerations, the following tentative conclusions can be reached:

1. Manipulators, when needed, should be contained in an add-on kit preferably attached to a docking port on the tug. A companion kit providing the manned interfaces (controls, displays, etc.) would be placed in the crew module or other control station. All kit attachments, power, and other systems support required should be standard tug parts.

2. An alternative to attaching manipulator kits to the tug would be to develop the manipulator system as a separate system transported as a payload package to the satellite or other site for use. Thus, deployment of the system would be similar to deployment of a satellite. Systems adaptable to this philosophy are under study by several contractors in addition to NR (such as G. E.), and maximum advantages of this expertise would be taken with minimum tug impact.

3. Hard-docking provisions should be imposed as a requirement on all future space satellites expected to demonstrate long life and involving repairability. Once the tug is docked, repair may be accomplished directly within coupled pressurized compartments, with IVA, or with crew EVA if necessary.

From the point of view of tug development, use of a separate package manipulator system or provisions for hard docking would minimize the impact on the tug design requirements. Because the tug already is required to accomplish a wide spectrum of missions and operations, this design impact minimization may be desirable. From the satellite viewpoint, space repair and maintenance provisions appear to be a necessity for long economical life and high success probability; therefore, docking provisions to facilitate servicing appear justifiable. In addition, tight weight constraints on past satellites should be relieved somewhat with the advent of the space shuttle.

A separate study was conducted to apply these findings to tug constraints, and is presented in Appendix D. Figure 11-1 shows a sketch of a manipulator submodule concept. Table 11-2 presents a generalized composite mission sequence containing events pertaining to both maintenance and retrieval operations. The retrieval phase contains events showing a more rigid attachment of the satellite to the tug (in preparation for the acceleration of orbit transfer) than is necessary for maintenance tasks.

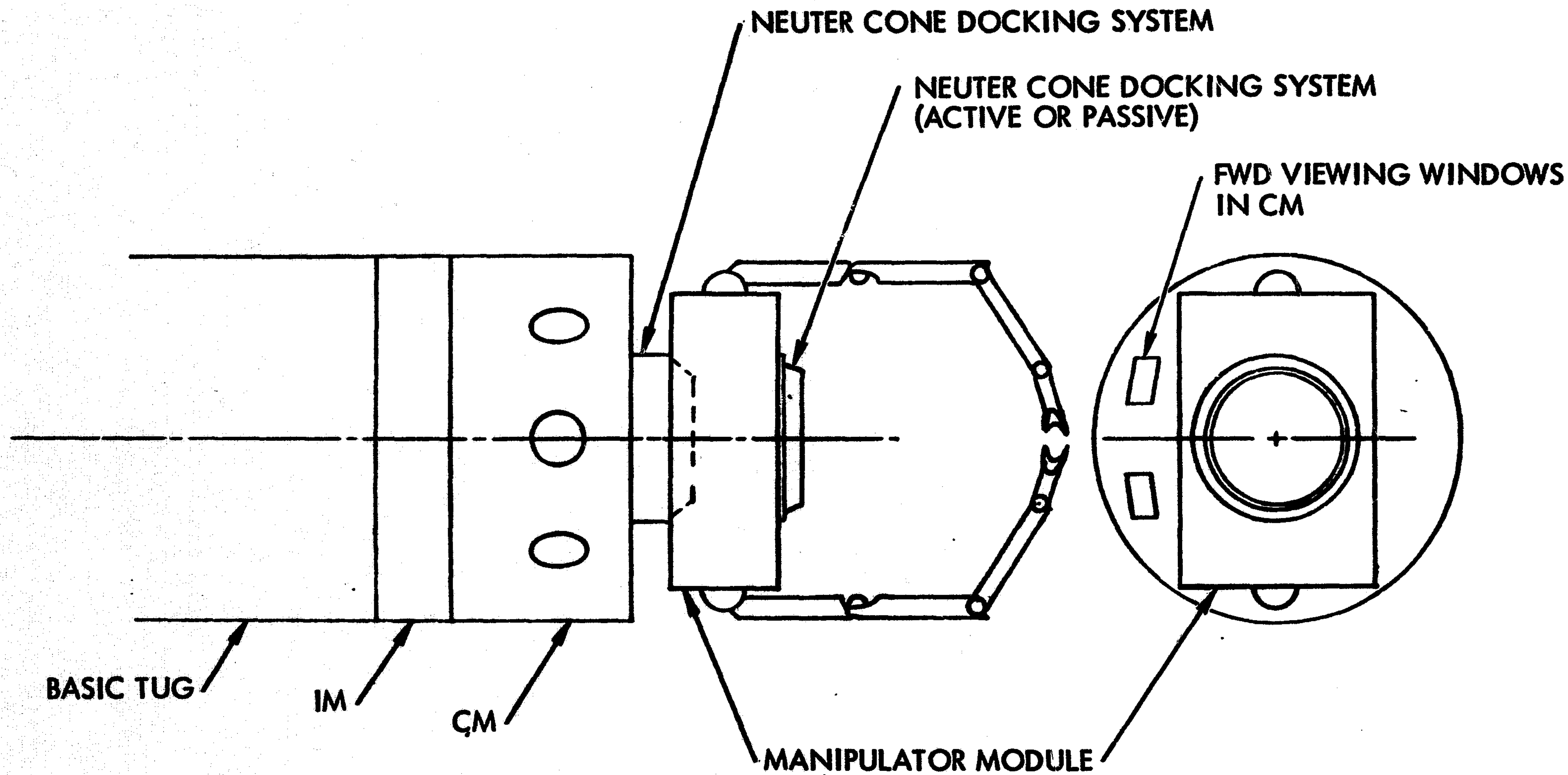


Figure 11-1. Manipulator Module Concept

Table 11-2. Satellite Maintenance, Replacement, or Retrieval Operating Sequence

Mission Phase	Phase Segment	Operation
Acquisition	Predocking	Assess satellite angular rates visually Determine hard-point contact locations Establish approach trajectory Arms extended
	Docking	Very low relative velocity at impact Satellite hard points engaged Clamps energized
	Post-docking stabilization	Deactivate satellite attitude control if necessary Damp relative angular rates Retract arms to desired position Secure tethers
Maintenance	Tug/satellite operations	Follow preplanned operational sequence
	Separation	Orient satellite attitude Reduce relative motion to within satellite bounds Release tethers and clamps, retro with ACS
Retrieval	Hard docking for propulsion	Retract arms to tug/satellite contact position Activate hard docking attachments Inspection and checkout Main propulsion burn

Table 11-3 lists estimated weight and power requirements for both the manipulator submodule and the accompanying crew module internal controls and displays for a manned tug. It should be noted that the submodule described does not utilize a full neuter docking system.

Future studies should be concerned with establishing requirements for space maintenance and the retrieval of satellites that include the impact on satellite design. These requirements should precede the development of new or modified existing manipulator systems.

Table 11-3. Estimated Manipulator Kit Description

Category	Weight (lb-kg)		Power (W)
Exterior Manipulator Kit (totals)	486	220.6	205
Manipulator arm assemblies (2)	280	127.0	40 (400 peak)
Torque and force restraining arms (2)	20	9.1	5
Tool attachments and fixtures	46	20.9	25
Manipulator/docking port attachment bracket	20	9.1	--
Control electronics	50	22.7	100
Video cameras (2)	50	22.7	35
Electrical power distribution	20	9.1	--
Crew Interface Kit (totals)	60	27.2	115
Manual controls	20	9.1	30
Binocular video viewer	20	9.1	70
Electrical power distribution	10	4.5	--
Communications and data management	10	4.5	15
Electrical Power System at 1 lb/watt	320	145.0	
Total Estimated System Weight	866	393.0	

12.0 SUBSYSTEMS SUMMARY AND CONCLUSIONS

12.1 BASELINE SUBSYSTEMS

In each of the previous sections of this report, requirements, feasibility trade studies, and considerations of redundancy led to the selection of components. The weight, volume, and power characteristics of these components may now be summed and viewed as a complete system. To provide traceability, Table 12-1 relates tables of individual subsystem and kit summaries to the totals. Tables 12-2 through 12-5 itemize the totals, where it is noted that basic PM and IM equipment is common for any space mission. Added equipment (kits) are necessary for manned and lunar landing missions.

12.1.1 Module and Kit Feasibility Analysis

The feasibility of the modular approach for the tug was investigated continuously throughout the study. It has been shown that the geosynchronous orbit mission requires very high performance, which tends to preclude propulsion module (PM) compromises for multipurpose use. However, these compromises do not appear to be substantial until considering the lunar lander version, wherein landing loads, cargo pods, landing gear attachments, and special visibility-access requirements are imposed on the configuration. The PM is generally optimized for the geosynchronous (high-energy) mission and is utilized off-loaded for other applications. Block changes, such as structure beef-up, retractable engines, and various kits, are usually required for the lunar lander version.

A single basic crew module (CM) design appears feasible for transporting up to 6 men to and from a space station, 4 men for 45-day lunar surface stay, and for 6-12 man rescue. Docking adapters on top or on bottom as needed provide crew ingress-egress for the various mission applications.

A cargo module (CAM) can be of almost any reasonable shape for use in earth-orbital missions. In general, the CAM should provide for standard docking interfaces at each end and should be pressurizable for compatibility with a space station. The special constraints of the lunar landing configuration of the tug (i.e., low cg, easy loading and surface access, and maintaining vehicle balance) result in a choice of hemipods or cylindrical halves that can be EOS-launched and readily attached laterally to tug landers low on

Table 12-1. Key to Subsystems Summaries

Subsystem/Kit Component Summaries			Totals			
			Unmanned		Manned	
	Description	Table	Space Missions (Table 12-2)	Lunar Landing (Table 12-4)	Space Missions (Table 12-3)	Lunar Landing (Table 12-5)
ECLSS	Fixed Equipment	2-4			X	X
	Spare Equipment	2-5			X	X
	Expendables and Consumables	2-6			X	X
G&N	Basic Equipment	3-3	X	X	X	X
	Rendezvous and Docking Equipment	3-5	X	X	X	X
	Lunar Landing Equipment	3-6		X		X
Comm & Data Mgt	Intelligence Module Equipment	4-8	X	X	X	X
	Crew Module Equipment	4-9			X	X
EPS	Total Electrical Power Equipment	5-8	X	X	X	X
ATC	Total Active Thermal Control Equipment	6-4	X	X	X	X
ACS	Total Auxiliary Control Equipment	8-12	X	X	X	X
	Oxygen/Hydrogen Weight Summary	8-11	X	X	X	X

Table 12-2. Unmanned Space Mission Subsystem Equipment Summary (7-Day Mission)

Description	Weight		Volume		Pwr (W)
	Lb	Kg	Ft^3	M^3	
Intelligence Module					
Guidance, navigation, and control basic equipment	296	134	4.48	0.1269	331
Unmanned docking equipment	105	48	1.95	0.0552	60
Communications and data management equipment	953	432	18.98	0.5375	568
Electrical power basic equipment	506	230	7.45	0.2110	230
Active thermal control basic equipment	184	83	5.26	0.1489	62
Auxiliary control equipment	278	126	3.90	0.1104	0
Auxiliary propellant equipment	89	40	18.00	0.5097	0
Total IM for unmanned space missions	2411	1093	60.02	1.7000	1251
Propulsion Module					
Docking equipment	21	9.5	0.20	0.0057	30
Active thermal control basic unmanned equipment	16	7.3	0		0
Auxiliary propellant equipment	196	88.9	4.65	0.1317	0
Total PM for unmanned space missions	233	105.7	4.85	0.1373	30
Total subsystem equipment for unmanned space missions	2644	1199	64.87	1.8370	1281

Table 12-3. Manned Space Mission Subsystem Equipment Summary (6-Man, 7 Days)

Description	Weight		Volume		Pwr (W)
	Lb	Kg	Ft^3	M^3	
Crew Module (6-men, 7 days)					
Environmental control and life support basic equipment	1294	587	106.62	3.019	1128
Spare equipment for ECLSS	205	93	14.02	0.397	0
Expendables and consumables for ECLSS	498	226	22.52	0.638	0
Guidance, navigation and control basic equipment	164	74	2.94	0.083	154
Manned docking equipment	93	42	1.75	0.050	30
Communications and data management equipment	399	181	10.66	0.302	154
Electrical power equipment	50	23	0.50	0.014	25
Active thermal control equipment	192	87	4.83	0.137	88
Total CM for manned space missions	2895	1313	163.84	4.640	1579
Intelligence Module					
Unmanned intelligence module less docking equipment	2306	1046	58.07	1.644	1191
Electrical power added manned equipment	127	58	2.40	0.068	40
Active thermal control added manned equipment	141	64	4.04	0.114	43
Total IM for manned space missions	2574	1168	64.51	1.826	1274
Propulsion Module					
Unmanned propulsion module	233	106	4.85	0.137	30
Active thermal control added manned equipment	13	6	0		0
Total PM for manned space missions	246	112	4.85	0.137	30
Total subsystem equipment for manned space missions	5715	2593	233.20	6.603	2883

Table 12-4. Unmanned Lunar Landing Mission Subsystem Equipment Summary

Description	Weight		Volume		Pwr (W)
	Lb	Kg	Ft^3	M^3	
Intelligence Module					
Unmanned intelligence module for space missions	2411	1093.6	60.02	1.699	1251
Propulsion Module					
Unmanned propulsion module for space missions	233	105.7	4.85	0.137	30
Active thermal control lunar landing equipment	38	17.2	0.25	0.007	0
Total PM for unmanned lunar landing missions	271	122.9	5.10	0.144	30
G&N Lunar Landing Kit (unmanned equipment)	168	76.2	4.84	0.137	255
Active Thermal Control Kit (unmanned equipment)	89	40.4	1.00	0.028	24
Total subsystem equipment for unmanned lunar landing	2939	1333.1	70.96	2.008	1560

Table 12-5. Manned Lunar Landing Mission Subsystem Equipment Summary (45 Days)

Description	Weight		Volume		Pwr (W)
	Lb	Kg	Ft^3	M^3	
Crew Module (4 men, 45 days)					
Basic crew module for space missions	2895	1313	163.84	4.639	1579
ECLSS added lunar landing fixed equipment	623	283	30.86	0.874	80
ECLSS added lunar landing spare equipment	30	14	1.20	0.034	0
ECLSS added lunar landing expendables and consumables	1138	516	45.35	1.284	0
Guidance, navigation and control added lunar landing equipment	27	12	0.23	0.006	62
Total CM for manned lunar landing missions	4713	2138	241.48	6.837	1641
Intelligence Module					
Manned intelligence module for space missions	2574	1168	64.51	1.827	1274
Propulsion Module					
Manned propulsion module for space missions	246	112	4.85	0.137	30
Active thermal control lunar landing equipment	38	17	0.25	0.007	0
Total PM for manned lunar landing missions	284	129	5.10	0.144	30
G&N Lunar Landing Kit (manned equipment)	133	60	4.43	0.125	195
Active Thermal Control Kit (manned equipment)	431	195	3.91	0.111	93
Total subsystem equipment for manned lunar landing	8135	3690	319.43	9.044	3233

opposite sides. These hemipods can serve double-duty as a single cylinder module by bolting the two halves together and adding docking adapters.

The IM can incorporate the basic subsystems required for unmanned tug flight. Add-on kits can provide the necessary additional power, thermal control, communications, etc., for longer duration and manned flight. While options exist as to the specific location of some of these subsystem elements (within PM or within IM), the differences do not appear large at this phase of the tug study.

The wide variety of missions to which the IM may be subjected indicates that a separate module is desirable. If the IM is combined with the propulsion module, little if any weight is saved, because components must still be mounted and interconnected. The structural weight as well as vehicle length would decrease slightly; however, the PM forward end must be closed out to provide support for a docking port. If the IM is combined with the PM, the missions that do not require a PM (shuttle/space station short range cargo and crew transfer and space station remote experiment module control) would be penalized. IM integration with the crew module would severely penalize all unmanned missions.

Many potential equipment divisions within the IM are possible. If the IM is to be capable of flying alone, it must have as a minimum all unmanned functions. Beyond this requirement it also may include certain manned functions. Inclusion of manned electrical power sources in the IM seems attractive, because fewer IM/CM interfaces are required, space maintenance of EPS is centralized, and crew safety may be slightly improved. The presence of auxiliary control thrusters on the IM precludes using that area for active thermal control space radiators. Therefore, the IM radiations would be mounted on the PM or on a separate skirt for the cases where the IM flies alone.

The internal IM arrangement consists of four ACS support elements (tanks, valves, electrical, and fluid lines) between which all other equipment is mounted. The ACS support elements, as well as the other equipment, are mounted to the exterior IM walls for ease of removal and to promote passive thermal control.

The preceding comments are summarized in Figures 12-1 and 12-2. In Figure 12-2, kits including radiator, landing gear, manipulators, and long range extensible antenna kits either may be eliminated or integrated into modules with further study. Basic radiators for space operation are mounted on the vehicle periphery. The lunar surface radiator is added to the top of the tug to reduce reradiation heating. The top radiator is initially folded against the vehicle sides when docked and during landing. If a 22-foot-diameter (6.7 meters) tug is landed, no deployable radiator area is required. A

EARTH ORBIT MISSIONS

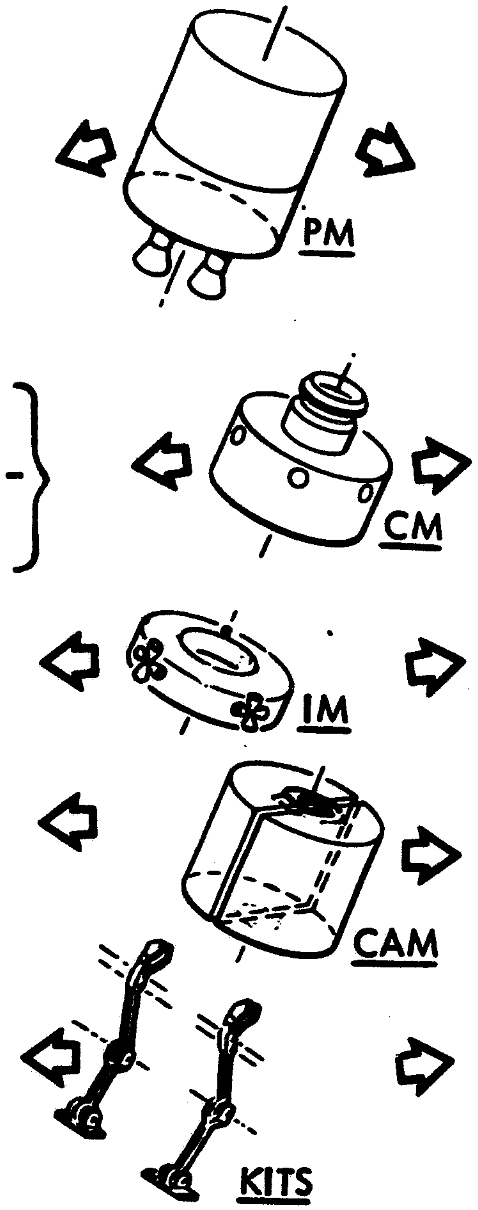
- Max. perf. in geosync mission
- Single engine preferred for efficiency
- Short configuration to save space

- 6-man crew transport
- 2-man working module
- Pressurized crew transfer
- Visibility of payload docking operations - fwd. location

- Subsystems, "brains"
- Manned or unmanned provisions
- Remote or manual control

- Easy loading at space station
- Transferable to another vehicle
- Pressurizable, double docking

- Extra provisions for manned flight
- Manipulators for repair/retrieval
- Neuter docking adapters - either end of any module



LUNAR MISSIONS

- Mod. high perf. - min. propellant logistics
- Min. scar provisions for landing missions
- Multiple engines desired for redundancy

- Provisions for 4 men, 28+ days stay
- Pressurized crew transfer
- Lunar landing visibility, surface access - low in vehicle stack

- Subsystems, "brains"
- Adaptable to manned or unmanned operations
- Additions for lunar

- Single module can be left on surface
- Easy loading at LOSS
- Easy unloading on surface
- Little effect on vehicle c.g.

- Extra subsystems req'd for manned flight
- Landing gear - min. tare wt. on vehicle

Figure 12-1. Desired Module Characteristics

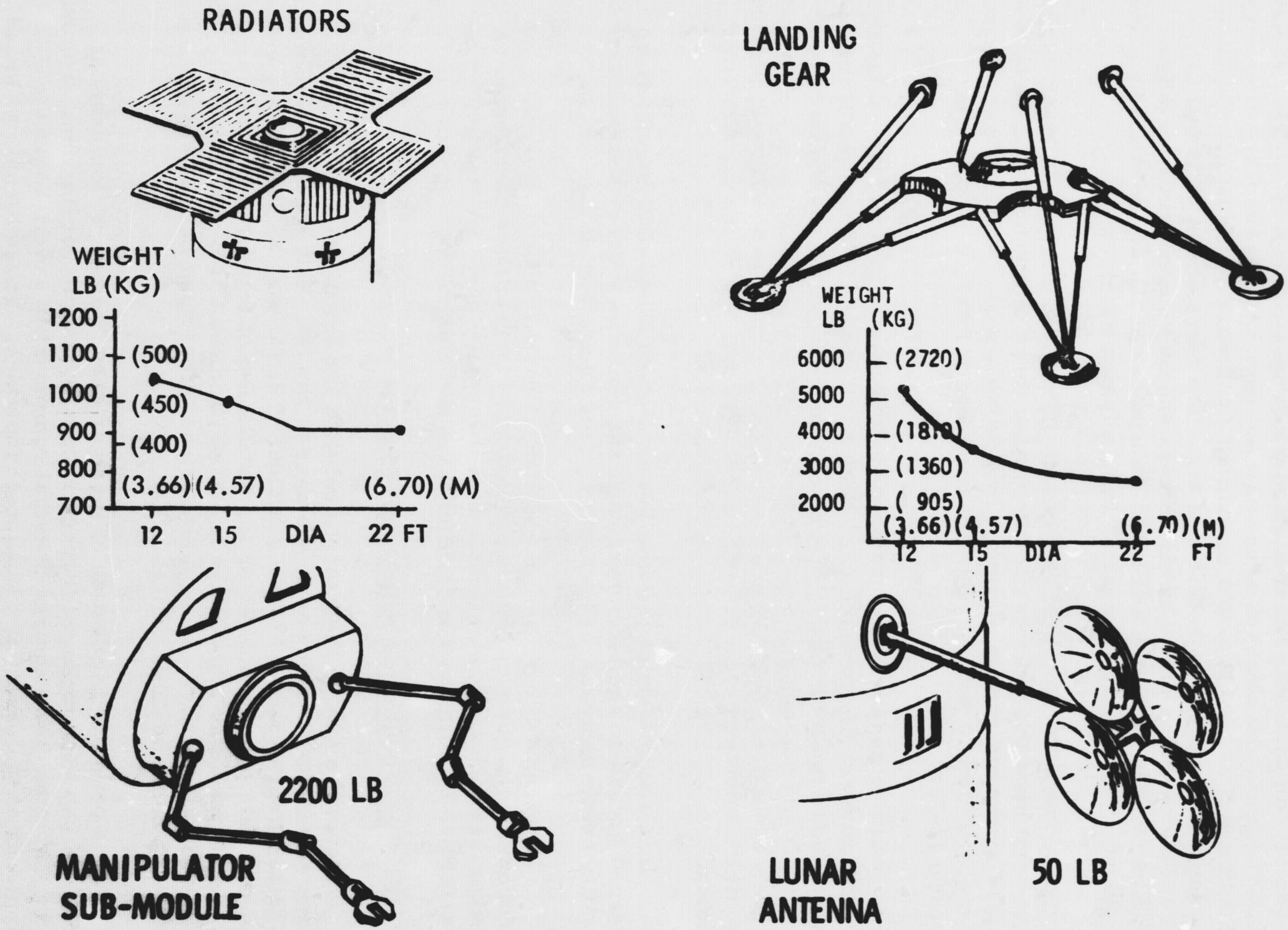


Figure 12-2. Kit Considerations

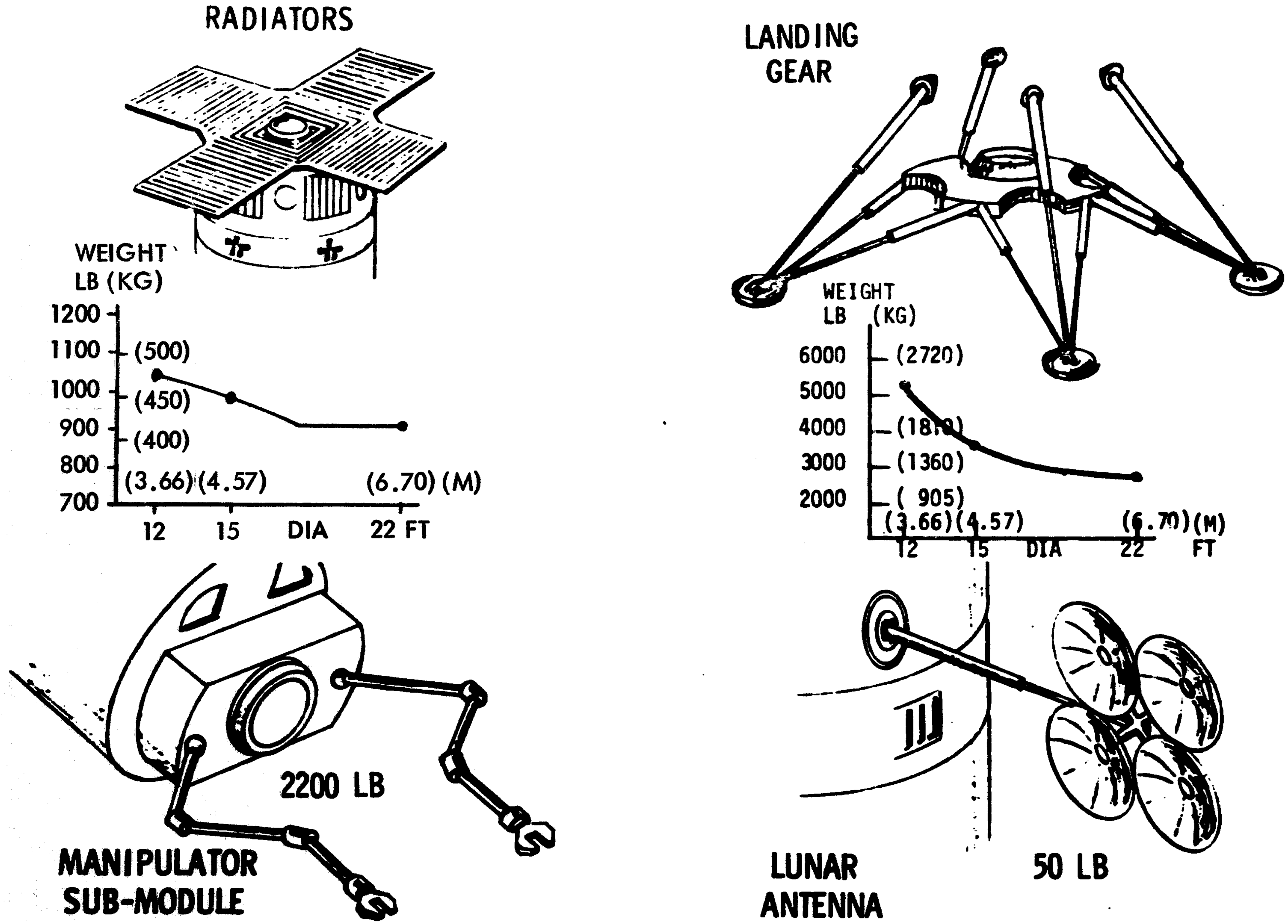


Figure 12-2. Kit Considerations

landing gear kit could be either an integral component or could be a separate assembly for each leg; either method would involve EVA to attach. A manipulator kit may be attached at the docking port. Inside the CM, if manned, an additional assembly would contain all necessary controls and displays. An antenna kit would be required if high date rates are to be transmitted from lunar orbit to earth. This antenna should be approximately 6 feet (1.8 meters) in diameter and could be attached to the tug or could be ground-deployed after landing, depending upon mission requirements.

Although the sequence of modules and kits in the vehicle stack depends somewhat upon the mission requirements, these differences will be minimized as the tug concept becomes more formalized. A preliminary description of the module and kit sequence for a space mission is shown in Figure 12-3.

Another aspect of the case for module feasibility involves assembly. An indication of the complexity of module interfaces is given in Figure 12-4 which shows the multiplicity of connections between the major modules of the tug. The number and types indicate the difficulty of making and breaking these connections reliably in space. The cryogenic propellants in particular require meticulous cleanliness, "zero" leakage, and shielding from heat shorts.

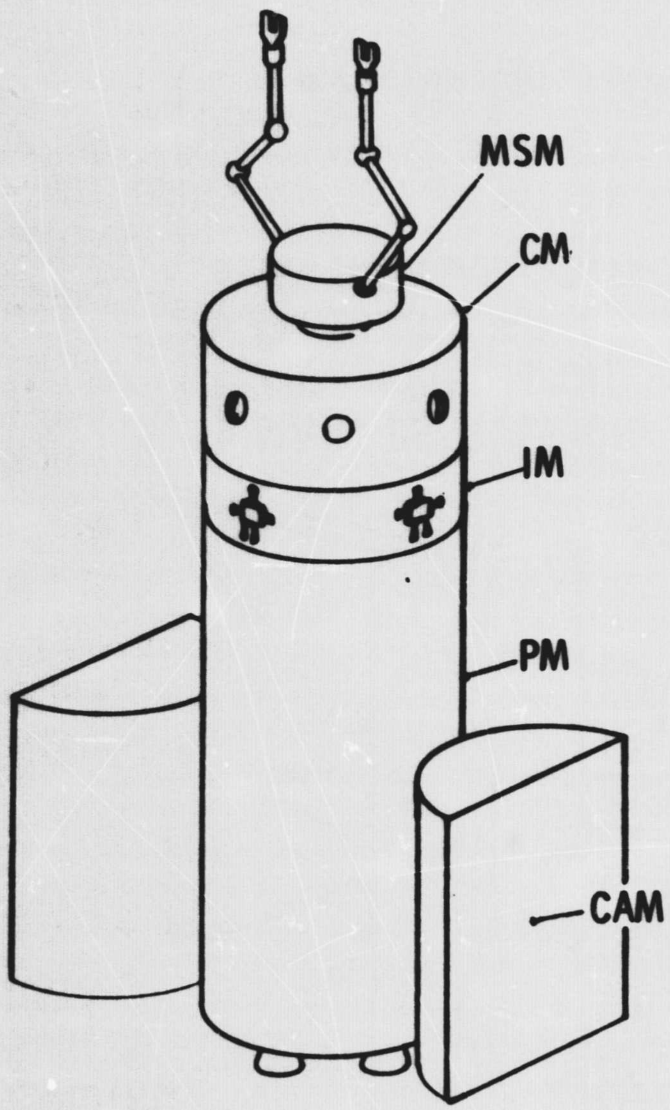
It is recommended that the PM-to-IM and IM-to-CM interfaces be considered ground-assembled only, until such time as a very capable space base facility becomes available to support these operations.

Throughout the report, many special purpose assemblies, kits, and submodules have been mentioned where their capabilities contributed to design feasibility. The description of these have been collected and summarized along with the standard modules in Table 12-6.

12.1.2 Tug Module Use With Other Vehicles

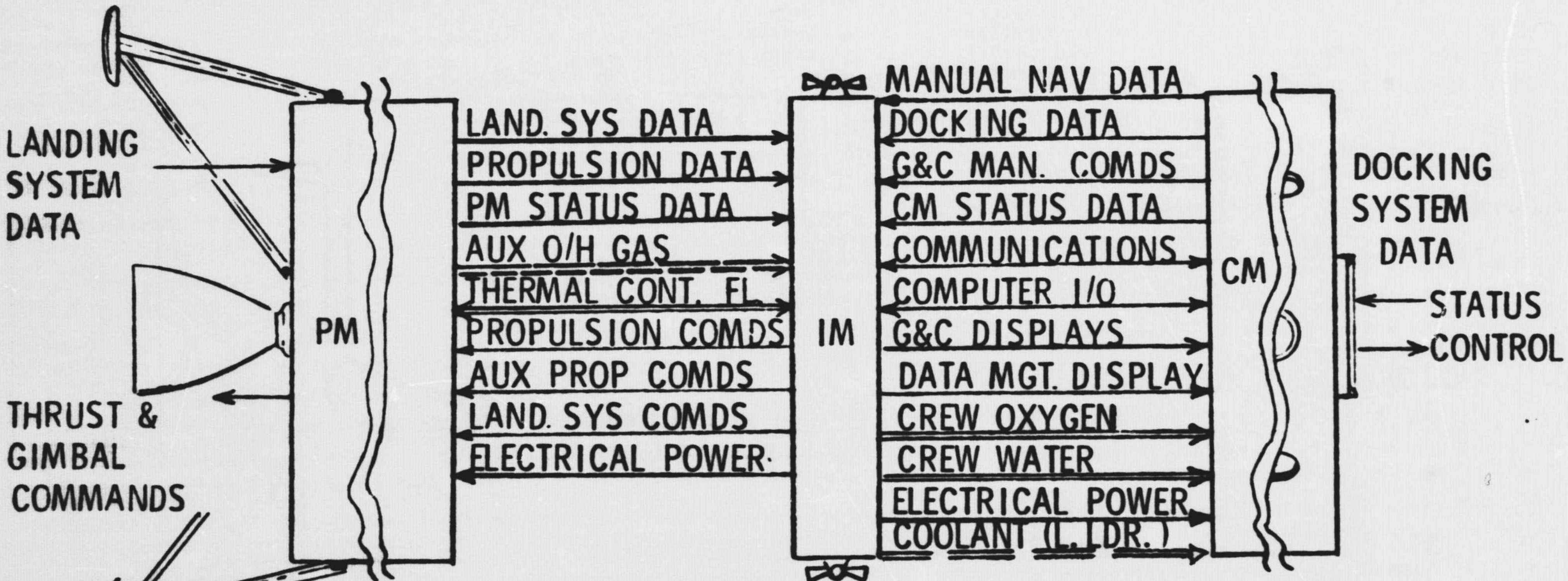
Several areas in the report deal with the use of modules in various stack sequences or alone as in the case of the IM. Tug modules also may be used as integral parts of other vehicles. The IM may be used as a stabilization and data processing center for satellites and experiment modules; the CM is adaptable as a small space station; and the PM is a suitable injection stage. In all of these cases, however, an effort must be made to ensure interface commonality.

The potential application of cislunar crew transportation using an RNS/CM has been explored through respective team coordination. To mate a tug CM to an RNS requires physical commonality and also that the RNS supply all of the essential CM support functions. These include as broad



CAPABILITIES	FUNCTIONS
• MANIPULATOR SUB-MODULE	• SAT. RETRIEVAL SERVICE • CARGO HANDLING
• 4 MEN-28 DAY LUNAR STAY 6 MEN-7 DAYS SPACE 12 MEN-1 DAY RESCUE	• EC/LSS • I/O FOR G&N, COMM • PWR DISTRIB & THERM CONT • PRESSURIZED ACCESS
• UNMANNED/MANNED FLIGHT SUPPORT AUTO & REMOTE OPS	• G&N, COMM & DATA MGT • ELECT. PWR & THERM CONT. • AUX FLT CONTROL SYS • ATT. CONTROL
• MAIN PROPULSION MAIN & AUX PROPEL.	• PROPULSION • PROPELLANT STORAGE & COND. • PWR DISTRIB & THERMAL CONT.
• CARGO HEMI-PODS CARGO TRANSPORT	• EXPENDABLES • INSTRUMENTS & TOOLS • EXPERIMENTS • MOBILITY AIDS

Figure 12-3. Current Philosophy of Module Organization



FUNCTIONS

- MAIN PROPULSION
- AUX. PROPELLANT
- LANDING GEAR
- IM THERMAL CONTROL

- MANUAL & AUTO GN&C
- COMM & DATA MGT
- ELECTRICAL POWER
- AUX PROPULSION
- AUX PROPELLANT

- LIFE SUPPORT
- DOCKING SYSTEM
- MANUAL G. N&C
- AUD & VID COMM
- DISPLAYS
- CM THERMAL CONTROL
- COMPUTER I/O

- PM-IM-CM SHOULD BE GROUND-ASSEMBLED INTERFACES

Figure 12-4. Module Functions and Interfaces

Table 12-6. Summary of Modules and Special Purpose Equipment

Module/Kit Description		Effectivity
PM	Propulsion module	All large delta-V missions
IM	Intelligence module	All tug missions
CM	Crew module	All manned missions
CAM	Cargo module	Resupply cargo missions (cylinder or hemipods)
PL	Payload	Special and general purpose cargo, satellites, experiments, etc.
-	Electrical power kit	Additional supply inserted in IM for manned missions or payload support (includes additional radiators)
-	Radiator kit	Top-mounted heat pump system for lunar landing active thermal control capability
-	Landing gear kit	Bottom-mounted deployable legs
-	Manipulator submodule	Remote control manipulator assembly attached to forward end of tug at docking port for assembly, maintenance and retrieval missions
-	CM manipulator interface assembly	On-board remote control system inside CM
-	Lunar landing G&N sensors	Landing sensor system added to PM and interior of CM
-	Lunar landing antenna	Tug-mounted or portable parabolic antenna for additional communications capability
-	Docking adapter kit	Provides capability for standard neuter docking to Apollo probe or drogue
-	G&N docking sensor assembly	Attached to foremost surface of tug and payload for active rendezvous and docking capability

categories electrical power and computer access. No major problems are foreseen in electrical power compatibility; however, it is doubtful that the RNS software is capable of supporting the statusing, command, communications, and control requirements of the CM without compromising its baseline design. Crew water and oxygen could be stored for these missions independently in the CM. A further analysis may consider the use of an IM/CM atop the RNS as an alternate concept.

With only a minimum of buffering and signal conversion equipment, the IM provides all of the functions necessary to control large launch vehicles such as the Saturn V and INT-21. The major difficulty in substituting the IM for the IU is in packaging, because different outer diameters are involved. To circumvent this problem, the standard IM could be installed higher in the launch vehicle stack or could be a part of the payload control and be used after the launch vehicle is jettisoned. Used strictly as a launch vehicle control module, the IM contains a surplus of equipment and would not be an effective replacement for the IU.

12.2 ALTERNATE SUBSYSTEM CONFIGURATION

Apart from the weight aspects of subsystems, answers to numerous other questions were posed as objectives in this report. The outcome of some of these questions includes factors that will decide the tug's feasibility. It is worthwhile, therefore, to discuss each of these factors separately. To give the answers credibility, alternative vehicle descriptions have been generated. These descriptions are confined to subsystem changes but are explained elsewhere in the report in terms of the overall vehicle and its missions.

12.2.1 Multipurpose Capability

In a sense, many spacecraft have had multipurpose capability in that they can perform small variations in a nominal mission and can carry payloads that have large functional differences. All of these spacecraft required a sizable separate design and development effort to ensure proper performance in the unique aspects of each mission. Often these additional efforts came after the initial design had flown many times. The logical next step in technology is to design a vehicle with this flexibility at the start and in such a way that the compromises for any single mission are minimized. Multipurpose capability may be a consideration completely independent of other factors, such as reusability and autonomous operation. However, it does seem necessary at this point to incorporate the concepts of modularity, common subsystem propellant tankage, and common internal data exchange for feasibility. Multipurpose tug capability also depends somewhat upon commonality with the other IPP elements (particularly support from the EOS) for economic operation.

12.2.2 Reusability

In contrast with multipurpose capability, the reusability feature has not been implemented previously. In the design of other spacecraft, gross weight decreases were traded for the needed reliability and the protection necessary for atmospheric entry without physical degradation. The increment in reliability must be added to the tug to afford reusability, but the resulting weight increment is minimized by space-basing or the protection of the EOS cargo bay during return to earth. The cost of reusability depends upon the requirements of space or ground basing, in that the technology of space maintenance and checkout lags far behind that of ground basing. Therefore, it is logical to reflect this fact in component redundancy. A ground-based version of the tug would have approximately one failure level less than that of the baseline tug.

12.2.3 Modularity

To afford a multipurpose capability, the tug may be "assembled" from building blocks. The concept partially removes the constraints of the EOS cargo bay envelope from the tug design and also permits specialized versions of the tug for different mission classes. Conversely, modularity penalizes the gross weight of any single version. A fully integrated, specialized tug would display fewer interface-oriented components. The structural cost of modularity is investigated elsewhere in the report. Intuitively, if the versatility of the tug is diminished, the justification for modularity also decreases. In particular a ground-based, less autonomous vehicle with slightly less multipurpose capability would tend to have an integrated intelligence and propulsion module.

12.2.4 Autonomous Operation

Large costs are involved in the maintenance and operation of command and control centers to support present day spacecraft. These centers are equipped for one-at-a-time missions, and their complexity will be greatly magnified if they are to support the proposed future traffic on the same basis. Any simplification of the ground station function therefore is justified provided a greater cost to the spacecraft is not incurred.

Studies in the feasibility of autonomous operation have been carried out for the tug and several related programs, and concepts have emerged. Data down-link should be condensed to short, high data rate transmissions where possible. Uplink command-control transmissions should be reduced in function and also should be condensed. One approach is to command a preprogrammed sequence of events to be carried out without ground monitoring.

Details of autonomous navigation concepts are presented in the G&N section and have been shown to be feasible. A relationship exists between orbital injection accuracy and midcourse correction propellant expenditures. An optimum condition exists when the cost of accurate navigation equipment matches the cost of midcourse correction propellant. (This problem is yet unsolved.)

Autonomy also may extend to rendezvous and docking sensors. A reasonable approach is to organize IPP elements so that the nonaccelerating spacecraft (EOSS, orbiting lunar station, lunar surface base, and OPD) furnish all rendezvous and docking sensors, allowing all accelerating vehicles (tug, CIS, RNS, EOS, OOS, etc.) to save the weight. Rendezvous and docking between two accelerating vehicles then would require special purpose sensor equipment. Another reasonable approach is to require other onboard equipment to serve a dual purpose as rendezvous and docking equipment. In this vein, the S-band communication link would yield range magnitude information, and star trackers would provide relative angular data. The feasibility of this system is now being studied for use with the EOS.

The first of these approaches has been selected as a measure of autonomy for the tug.

12.2.5 Common Cryogenic Tankage

In support of multipurpose capability, common tankage permits a more optimum tank capacity for any single mission, especially the missions where the tank weight is most important. Without the concept, separate tanks for main propulsion, auxiliary propulsion, fuel-cell reactants, and crew oxygen would include the capacity for the most demanding mission in each case plus a contingency capacity.

Common tankage provides an additional advantage in that orbital refueling problems are greatly simplified. The recommended tug concept includes a common refueling connection to enable oxygen and hydrogen to be loaded into the main propulsion tanks and to be disseminated to user subsystems upon demand.

However, this concept is not without fault. A penalty lies in the conversion of the cryogenics to gas and to increase the pressure for subsystems use. Its use by the fuel cells requires purity standards that probably do not permit helium pressurization in the main tank. An efficient method of recovering boiloff gases has not been found, even though their quantity is significant.

12.2.6 Comparison of Alternate Subsystem Configurations

The comments and conclusions drawn for multipurpose, reusable, modular, autonomous, and common tankage capabilities are summarized in Figure 12-5. One implication from the foregoing discussions is that, with proper organization, the tug capabilities may be "grown" by starting with a version that has somewhat lesser capability and gradually increasing its capability without incurring exhorbitant redesign costs. Obvious new installations when equipment is added include wiring harnesses and coolant circulation lines. Although the feasibility of the entire approach is not yet determined, it merits study, because it allows a modest beginning and permits growth to capitalize on future state-of-the-art.

Table 12-7 is included in an attempt to identify the meaning of space-basing and autonomy at the component level. The approach was to treat each vertical division as an independent characteristic; however, while an increase in autonomy means primarily the addition of different equipment, redundancy is added unavoidably by accumulating alternatives. For example, another memory module added to allow new onboard computation also provides a backup to existing memory modules. Less redundancy was assumed to be the major difference between space and ground-basing; again, the addition of redundant components sometimes provides more autonomy.

The component descriptions in Table 12-7 do not seem to produce logical vehicles balanced in their capabilities. For this reason, various mixes of capabilities were devised by selecting equipment from the baseline lists. These variations are shown in Figure 12-6. The first four bars on the left of the figure represent various baseline configurations. In the fifth case, approximately 800 pounds (360 kilograms) may be saved if all the secondary effects are considered when the requirements for precision G&N are relaxed and the rendezvous and docking sensors are removed. The next three cases show variations in autonomy for a ground-based reusable vehicle. The first of these cases shows a reduction of approximately 800 pounds (360 kilograms) by lowering redundancy levels while keeping the baseline capability. Another 250 pounds (113 kilograms) are saved in the second of the three cases by removing active docking sensors and the related equipment in other subsystem areas. A total weight of 1400 pounds (634 kilograms) results when autonomy is reduced to a safe minimum. This case retains redundancy necessary for man-rated interfaces and a high level of mission success but would require mission specialized software programming before each launch. The last case on the right of the figure was synthesized to be comparable with proposed uprated versions of existing injection stages while maintaining growth capability. As an expendable vehicle, it tends to require a shorter mission life, allowing inclusion of battery primary electrical power and a storable bipropellant ACS.

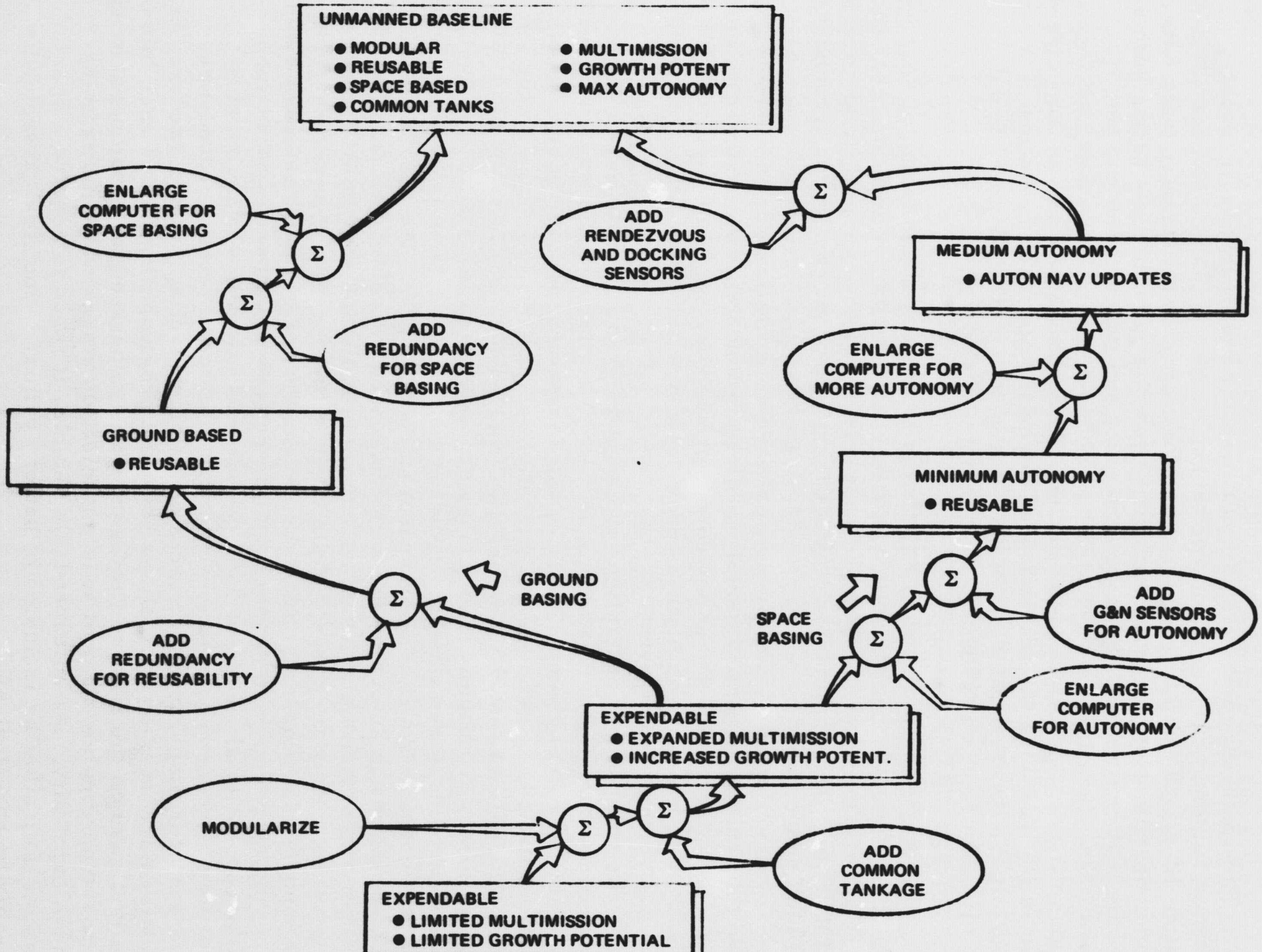


Figure 12-5. Comparison of Tug Capabilities

Table 12-7. Comparison of Tug Subsystems Capabilities

Baseline Subsystems (Unmanned)	Functionally Diminished Subsystems		
	Minimum Autonomy	Ground Based	Expendable
General Comments Modular Multimission Growth potential included Reusable Common tankage Space based Maximum autonomy	Modular Multimission Growth potential included Reusable Common tankage Space based	Modular Multimission Growth potential included Reusable Common tankage	Modular Multimission Growth potential included
Subsystem—Specific Comments			
G&N Baseline Gyro pentad IMU (1) Gimbaled star tracker (2) Sun sensors and elect. (4) Horizon/earth edge tracker (1) Navigation sensor base (1) ACS driver amplifiers (2) Main eng gimbal amps (1)	G&N — Gnd updates Delete horizon/earth tracker Simplify nav base	G&N — Less redundancy Delete one-star tracker	G&N — Minimum redundancy Delete one-star tracker Delete one ACS driv. amp assy
G&N Rend and Docking Baseline IM laser ren and dock radar (2) IM television camera (1) PM television camera (1) IM contact sensor (4) PM contact sensor (4)	G&N — Nonauton. ren and dock Delete all ren and dock sensors	G&N — Less redund, no PM sensors Delete: One laser Television cameras Two contact sensors	G&N — Min redund, no sensors Delete all ren and dock sensors
Comm and Data Mgt Baseline S-band trans/receiver (2) Omni antenna (4) 2-ft/0.61M steerable dish (2) Dual comm switching and C/O (1) Dual premod processor (1.5) MOS-LSI I/O controller (3) MOS-LSI gen purpose processor (4) 787.5 K bit operational memory (4) Tape archival storage memory (1) 1680 K bit mass storage memory (5) Dual central timing unit (1) Video unit (2) Remote ACQ and control units (11)	C&DM — Min software Delete: One I/O controller Two gen purp processors Two op memories Tape memory Five mass memories Two video units Five remote ACQ & cont un	C&DM — Less redundancy Delete: 3/5 S-B trans/rec Two omniantennas One 2-ft steerable dish 0.5 premod processor One I/O controller Two gen purp processors Two op memories Tape memory Five mass memories Video unit Five remote ACQ & cont un	C&DM — Min redund, auto disposal Delete: 3/5 S-B trans/rec Two omniantennas One 2-ft/0.6M steerable dish 0.5 premod processor One I/O controller Two gen purp processors Two op memories Tape memory Five mass memories Video unit Five remote ACQ & cont un
Electrical Power Baseline 1.33 Kw O_2/H_2 fuel cell (3) 26.7 AH NiCd battery (3) Battery charger (3) 167 VA inverter (3) Power controller (3)	EPS — Less pwr reqd Changed to reflect less demand	EPS — Less wh, less redund Changed to reflect less demand Delete: One fuel cell Two batteries Two battery chargers One inverter One pwr controller	EPS — Batt primary, min redund Changed to reflect less demand Change fuel cells to batteries (3) Delete: Three battery chargers One inverter Two pwr controllers
Active Thermal Control Baseline IM coolant system (1) PM space radiator (4)	ATC — Less heat produced Changed to reflect less heat	ATC — Less heat produced Changed to reflect less heat	ATC — Less heat produced Changed to reflect less heat
Auxiliary Control Baseline 200-lb/91 Kg O_2/H_2 engines (20) Lines and valves (2 sets)	ACS — No change	ACS — No change	ACS — 100 lb/45 Kg nto/MMH engines (20) Changed to reflect less total IMP
Auxiliary Propellant Baseline O_2, H_2 accumulator tanks (4) O_2, H_2 turbopumps (4) O_2, H_2 gas generators (2) O_2, H_2 heat exchangers (4) Lines and valves (2 sets)	APS — No change	APS — No change	APS — Indep. tanks due to less IMP Delete cryo conditioning
Sig and Pwr Dist Wiring Baseline	Wiring — Less complex	Wiring — Less complex	Wiring — Less complex

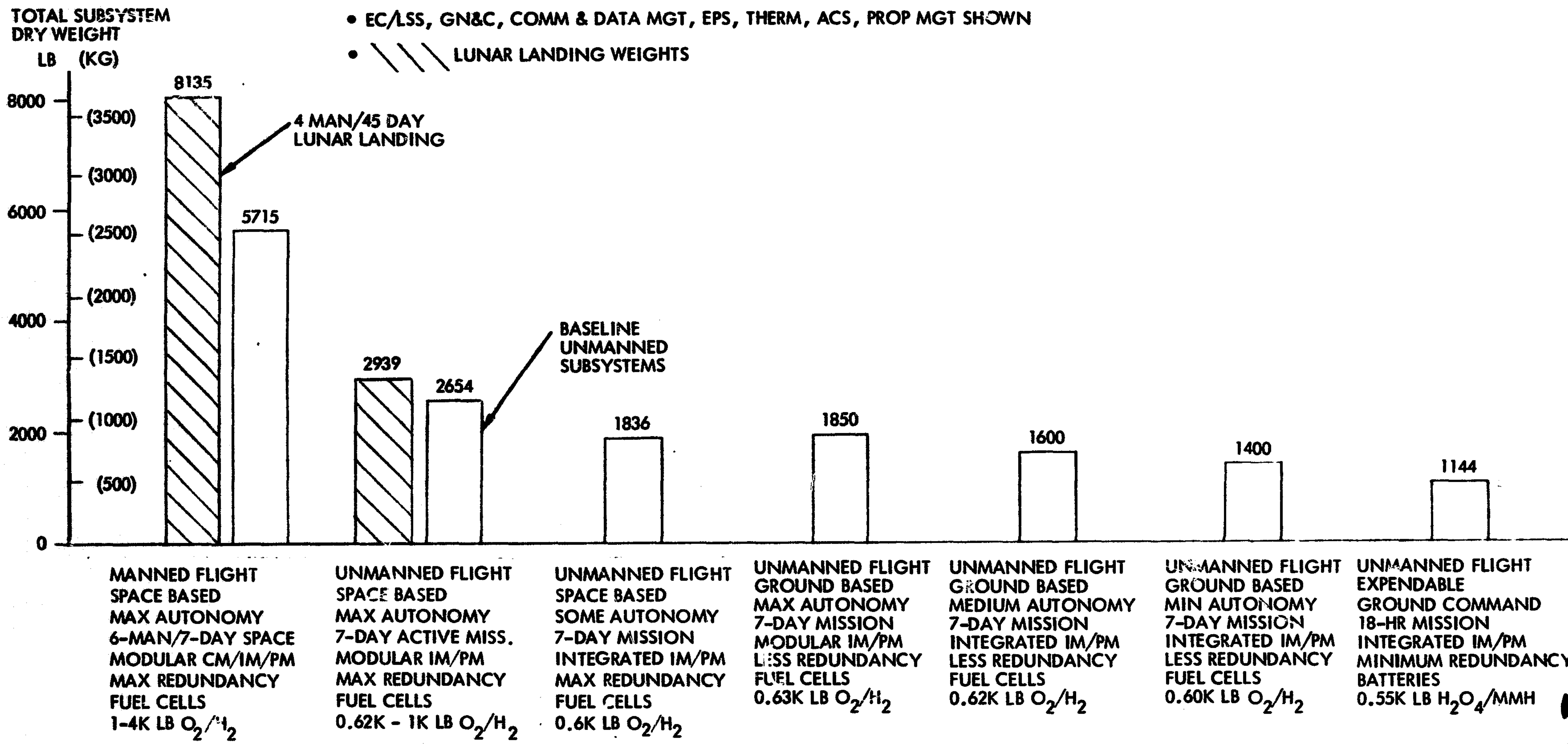


Figure 12-6. Subsystem Weight Vs Capability

12.3 IPP ELEMENT REQUIREMENTS IMPOSED BY TUG

Throughout the subsystems studies, as well as in other areas of the contract effort, requirements for other vehicles have been assumed to provide a basis for decision. An accounting of the more important of these requirements is made in this section.

The more obvious requirements for tug support by other vehicles are propellant and crew provision resupplies. The quantity of propellant to be transferred to tug tanks is a function of the mission. The method of orbital tanking is discussed in the appropriate design and operations sections. In most cases, the tug docks an aft docking system to the supplying vehicle so that shorter propellant transfer lines are used. Line connectors are mated after the completion of docking and checkout.

Crew provision resupplies are most easily transferred from the supplying vehicle through a CM pressurized hatch. A list of the resupply items is presented in Table 12-8. All of the data are directly traceable to requirements presented in the ECLSS section of this report. Included in the list are expendable, consumables, and an estimated 5 percent of the total onboard inventory of spare parts. The method of transferring these items depends upon the conditions in the supplying vehicle: a container holding the entire inventory could be attached to the CM port and unloaded as each item is stowed by the CM crew.

A large number of other requirement areas are noted in Table 12-9. All of the interface requirements should be defined in more detail by cooperative teams with vested interests in each of IPP elements.

12.4 TUG QUIESCENT STORAGE

If the tug is to be reusable and space-based, provisions for storage between missions must be made. The requirements for quiescence are 180 days in space or 30 days on the lunar surface. The space storage period may be spent docked to another vehicle or free-flying. In any case, reactivation should be accomplished within 2 hours.

The tug in free-flying quiescence condition needs attitude stabilization from the ACS to satisfy various types of requirements. Whether the stabilization is necessary throughout the storage period or only during the reactivation period depends upon these requirements. If propellant boiloff is to be minimized by pointing a minimum, highly insulated portion of the tug surface area at the sun, this can only be provided by continuous attitude stabilization. Other requirements also may lead to continuous attitude stabilization during free-flying quiescence, but they are not apparent at this time.

Table 12-8. Crew Module Resupply Summary

Description	Source/Type	Space Missions												Lunar Landing Missions							
		2 Man/7 Days				4 Man/7 Days				6 Man/7 Days				4 Man/31 Days				4 Man/45 Days			
		Weight		Volume		Weight		Volume		Weight		Volume		Weight		Volume		Weight		Volume	
		Lb	Kg	Ft^3	M^3	Lb	Kg	Ft^3	M^3	Lb	Kg	Ft^3	M^3	Lb	Kg	Ft^3	M^3	Lb	Kg	Ft^3	M^3
Expendables and Consumables																					
Housekeeping: cleaners, trash bags, char filters	EOSS Extrap	6.6	3.0	0.30	0.008	13.1	5.9	0.60	0.017	19.7	8.9	0.90	0.025	55.0	24.9	2.50	0.071	82.0	37.2	3.73	0.106
Food mgt: dehyd/froz food, utensils, soap	Skylab	61.5	27.9	2.79	0.079	123.0	55.8	5.57	0.158	184.0	83.5	8.38	0.237	450.0	204.1	20.40	0.578	664.0	301.2	30.20	0.855
Waste mgt: chemical and bacterial filters	Skylab	15.0	6.8	1.65	0.047	27.5	12.5	3.02	0.086	40.0	18.1	4.40	0.125	29.0	13.1	3.20	0.091	30.0	13.6	3.30	0.093
Temp and humid cont: filters	Apollo/Skylab	5.9	2.7	0.14	0.004	11.3	5.1	0.27	0.008	16.7	7.6	0.40	0.011	11.3	5.1	0.27	0.008	11.3	5.1	0.27	0.008
Atmos purif: particle and mole filters, char	EOSS Extrap	5.6	2.5	0.17	0.005	6.1	2.8	0.18	0.005	6.7	3.0	0.20	0.006	10.0	4.5	0.30	0.008	12.2	5.5	0.37	0.010
Atmos circ: LiOH	Apollo/Skylab	63.0	28.6	2.17	0.061	126.0	57.2	4.35	0.123	189.0	85.7	6.50	0.184	456.0	206.8	15.70	0.445	662.0	300.3	22.80	0.646
Crew support: clothes, bedding, towels, soap, med	EOSS Extrap	12.8	5.8	0.58	0.016	25.5	11.6	1.14	0.032	38.3	17.4	1.74	0.049	108.0	49.0	4.90	0.139	159.0	72.1	7.20	0.204
Nitrogen: cryo baseline	Estimate	3.4	1.5	--		3.4	1.5	--		3.4	1.5	--		11.0	5.0	--		15.4	7.0	--	
Total CM ECLSS expend consumables	–	173.8	78.8	7.80	0.220	335.9	152.4	15.13	0.429	497.8	225.7	22.52	0.637	1130.3	512.5	47.27	1.340	1635.9	742.0	67.87	1.922
Spare Equipment: stowable parts, controls, filters, hoses and bags	Apollo/Skylab/EOSS	5.0	2.3	0.31	0.009	7.6	3.4	0.51	0.014	10.2	4.6	0.70	0.020	9.1	4.1	0.57	0.016	9.1	4.1	0.57	0.016
Total CM ECLSS resupply	–	178.8	81.1	8.11	0.229	343.5	155.8	15.64	0.443	508.0	230.3	23.22	0.657	1139.4	516.6	47.84	1.356	1645.0	746.1	68.44	1.938

Table 12-9. Tug-Imposed Requirement Areas for IPP Elements

Tug Functional Area	Near-Earth Missions			Cislunar		Lunar		Tug	Payloads	Relay Satellites	Ground Stations
	Interfacing Vehicle										
	EOS	EOSS	OPD	RNS	CIS	OLS	LSB				
Propellant transfer	PR	R	R			R		PR	P		
Crew provision resupply	R	PR					P	PR			
Docking, checkout and separation	PR	P	P	PR	PR	PR	P	PR	P		
Rendezvous	PR	PR	PR	PR	PR	PR	PR	PR	PR	P	
Orbital assembly	R	PR	PR	PR	PR	PR		PR	P	P	
Transportation	R	P	P	P	P	P	P	PR	P	P	
Data link	PR	PR	PR	PR	PR	PR	PR	PR	PR	PR	PR
Command link	R	R	P	P	P	R	R	PR	P	PR	R
Crew transfer	PR	PR				PR	PR	PR			
Quiescent basing	R	R	R	R	R	R	R	PR	P	P	
Staging	R			R	R			PR	P		
Retrieval				P	P			PR	P	P	
Rescue	R	R		R	R	R		PR			
Maintenance	R	PR	PR	PR	PR	PR		PR	P	P	

Note R = Tug receiving or passive
P = Tug providing or active

Prolonged disturbance torques arising from external environment (solar pressure, gravity gradient, magnetic fields, etc.) typically are periodic and result in very low vehicle velocities. Internally produced torques (rotating machinery and outgassing) produce secular torques, which, over prolonged periods, cause large accumulated angular velocities. Neither type of torque will preclude stabilization during the reactivation period if communications contact can be established.

It is important to design the tug so that the stabilization system may be turned off during free-flying quiescence, because a rather large propellant consumption and an appreciable fraction of equipment life is involved. Unless a separate, low-grade stabilization system is employed for quiescence, nearly all other subsystems must be operating in support. On the other hand, with all subsystems turned off, reactivation could be accomplished by remote operation of circuit breakers in a preplanned sequence. Initial onboard power is supplied by batteries.

If the tug docks with another vehicle for quiescent storage, then the problem is greatly simplified. Here, thermal control is the major concern. The other vehicle might supply sufficient power, on the order of 60 watts, to permit operation of the coolant circulation pumps so tnat heat may be transported to the tug radiators. No advantage is seen in transporting heat to a docking port heat exchanger to be dumped by the other vehicle.

The computer may be deenergized during quiescence without loss of data, if the memory elements are nonvolatile. Both the operating and mass memories could be volatile, if they are resupplied with data and program records from a tape memory during reactivation.

To summarize, quiescent storage in space may be spent either free-flying or docked to another vehicle without great penalty. Quiescence requirements should be established to minimize expenditures of propellant and equipment operating life. Quiescent storage of the tug on the lunar surface should only be attempted if an external power source is provided to operate the coolant circulation and heat pump systems.

13.0 RECOMMENDED FUTURE SUBSYSTEMS EFFORT

The preceding sections indicate that the rapid advancement in most areas makes assessment of the current state of technology difficult. If compromises are made to take advantage of equipment commonality with EOS, these compromises should be noted and reassessed in the event of EOS equipment changes.

The prospects of large weight savings because of near-term technology advancement are very significant. Not only are software and sensor suppliers predicting weight and power reductions, but implementation of the changes will relax EPS and ATC requirements. As an estimate of the change in magnitude that could be expected if key predictions are realized, comparisons were made with data in Figure 12-6. New subsystem weights were totaled assuming the use of dual Micron IMU components, ground radio beacon tracking in place of an onboard horizon/earth edge tracker, star trackers and comm-link ranging in place of laser radar for rendezvous and docking, K-band communications in place of S-band, and the use of beam-lead software technology. No specific time period was associated with these advancements. The results imply that the unmanned baseline subsystem weight could be reduced approximately 800 pounds (362 kilograms). Therefore, these advancements are well worth pursuing.

Detailed future study descriptions for subsystems are presented in Volume 6 of this report and are reiterated in Volume 4.

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APPENDIX A

ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEM TRADE STUDY

APPENDIXES

These appendixes contain the reports from three tradeoff studies conducted under a separate, company-sponsored effort. A fourth report describes the Honeywell guidance, navigation and control study that was conducted for the tug. In each of these reports, tug requirements were used exclusively. However, since the contributors were using company funds, they were not required to include the international system of units. Thus, in all of the appendix reports, data are presented in English units only.

APPENDIX A

ENVIRONMENTAL CONTROL AND LIFE SUPPORT SUBSYSTEM TRADE STUDY

INTRODUCTION AND SUMMARY

This report presents the technical results of the environmental control and life support subsystem (EC/LSS) study performed in support of the Prephase A study of a reusable space tug. This preliminary design report features various studies that include development of requirements and constraints, consideration of operating modes, and tradeoff and comparison studies leading to selection of optimum subsystems and processes. The EC/LSS was evaluated over a range of mission objectives to provide support for a two- or six-man seven-day mission capability, with extended system growth for a four-man 45-day mission, including a 28-day stay on the lunar surface. Tradeoff and comparison studies were conducted in areas where multiple solutions were available. Tradeoff studies were performed to evaluate processes, such as regenerative (closed) types and nonregenerative (open) types of EC/LSS. Processes were evaluated and selected on a basis of weight, power, and volume. For establishment of EC/LSS characteristics of weight power, and volume, data developed for space station studies were adapted to tug crew sizes and mission duration requirements.

The objective of this report is to define the EC/LSS required to support a design configuration effort for earth orbit, lunar, and planetary missions, adopted to a reusable space tug program. The definition is at the assembly level, although definition to the subassembly level has been included in several instances to show more clearly how the significant EC/LSS functions are performed.

Study Objectives

Specific objectives of the study program reported herein are:

1. To identify and define candidate EC/LSS assembly concepts for potential application to the space tug program and make assembly level concept selection for each application.

2. To perform tradeoff analysis at the assembly level in order to select the preferred assembly concept approach. Estimate itemized weight, power, and volume for the EC/LSS.

3. To define the general characteristics of these selected approaches for space tug applications.

4. To define subsystem requirements for the EC/LSS commensurate with the various mission classes.

The specific assemblies and functions within the EC/LSS that were investigated under this study program are noted in Figure A-1. This functional matrix shows the specific assemblies and functions, which can be organized into three levels:

1. Specific assemblies or major EC/LSS areas such as atmospheric storage, active thermal control, etc.

2. Selected systems and functions such as cabin thermal loop, pressure control, or storage, etc.

3. System candidates affected by tradeoff analysis, such as the selection of the best CO_2 removal system using LiOh, molecular sieve, etc.

The major candidates affected by tradeoff analysis were in the areas of CO_2 removal, CO_2 reduction for water recovery, water electrolysis for O_2 recovery, trace contaminant control, and water reclamation. Candidate selections were made on a basis of weight, power, and volume. However, additional tasks for the evaluation of the EC/LSS will be conducted on the basis of reliability, safety, performance, availability, interfaces, and system flexibility. These tasks are not included as part of this study phase.

EC/LSS Summary

The major EC/LSS requirements and drivers that have a direct effect on vehicle volume, size, and weight and electrical power penalties for the selection of an EC/LSS were crew size and metabolic load, on-board consumable storage (a function of crew and mission duration), cabin volume, which affects cabin leakage and air circulation, and radiator heat rejection capabilities.

Table A-1 summarizes weight, power, and volume for the selected EC/LSS assemblies and subassemblies installed in the vehicle. Table A-2 presents the weight of EC/LSS consumables for the three missions considered in the tradeoff evaluation.

Results from tradeoff analyses, which minimized weight, power, and volume, show that the nonregenerative (open) type of system is best adapted to fulfill ll mission objectives and EC/LSS requirements of the tug program.

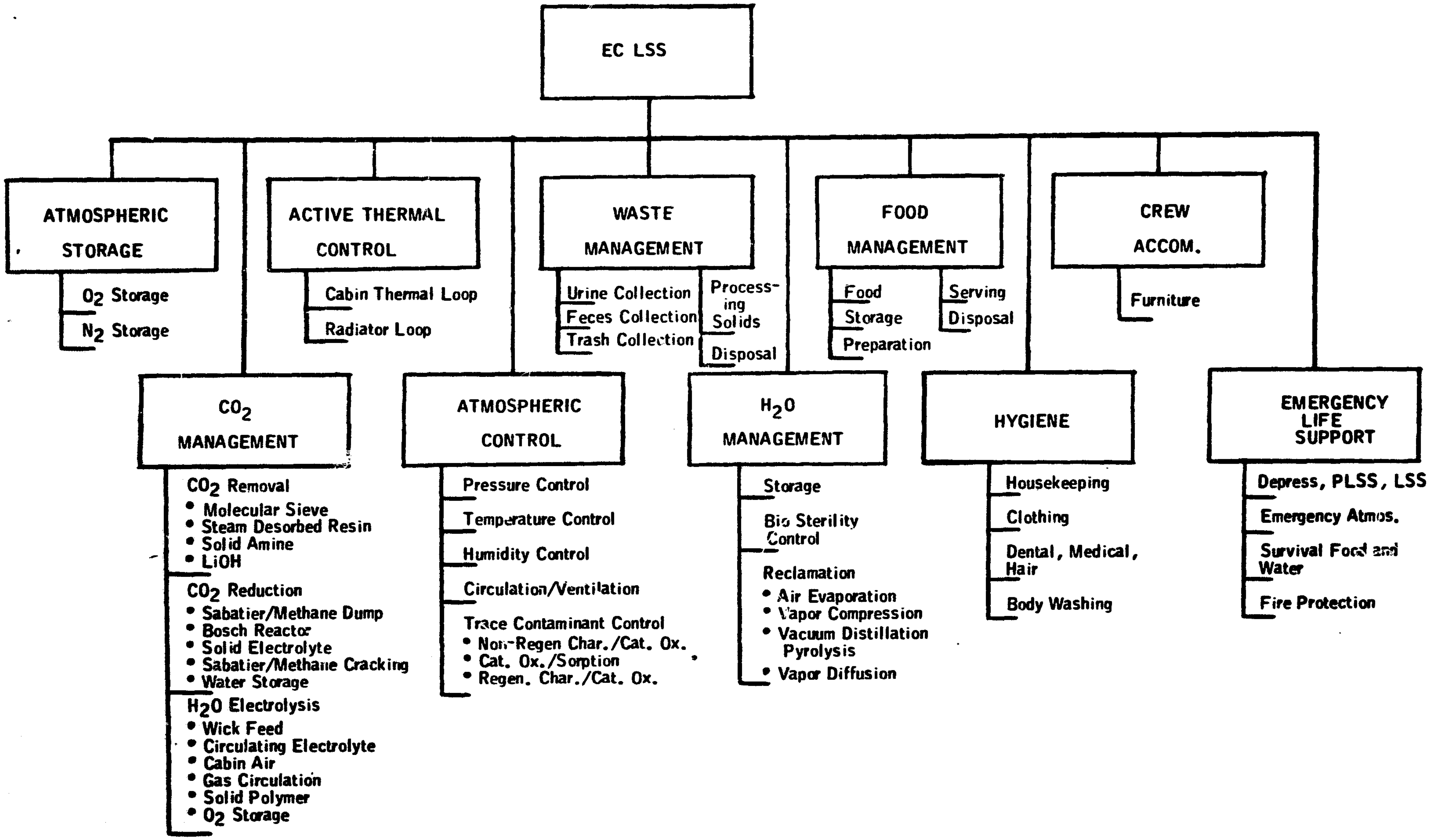


Figure A-1. ECLSS Candidate Assembly Concepts

Table A-1. EC/LSS System Selection

RECOMMENDED SYS. DESCRIPTION		FIXED EQUIPMENT		
CONCEPT	METHOD	WEIGHT LB (KG)	POWER WATTS	VOLUME CU. FT (M^3)
CONTAMINENT CONTROL	CAT. OX./SORPTION	36 (38.9)	145	4.52 (.128)
CO_2 REMOVAL	LiOH	0 (0)	--	0 (0)
CIRCULATION	FANS	121 (54.8)	150	18.87 (.534)
TEMP. & HUM. CONTROL	FANS, HX. & CONDEN HX.	310 (140)	250	35.55 (1.01)
PRESSURE CONTROL	5 PSIA, TWO GAS CONT.	41 (18.6)	30	2.14 (.061)
CABIN THERMAL LOOP	PUMPS & HX.	60 (27.2)	300	6.39 (.181)
WASTE MANAGEMENT	WASTE STORAGE	134 (60.6)	--	10.54 (.294)
EMER. LIFE SUPPORT	HIGH PRESS. GAS	81 (36.6)	--	1.80 (.051)
WATER & FOOD MGMT.	TYPICAL OF APOLLO	48 (21.7)	50	4.8 (.136)
CREW SUPPORT	TYPICAL OF APOLLO	154 (70)	--	7.0 (.198)
HOUSEKEEPING/ ATM. COND.	FILTERS, TRASH BAGS, ETC.	79 (35.8)	--	3.6 (.102)
EVA LIFE SUPPORT	PLSS & OPS (4 UNITS)	778 (352)	--	31.12 (.882)
TOTAL FIXED EQUIPMENT		1891 (856)	925	126.3 (3.57)

Table A-2. EC/LSS System Selection

CONSUMABLES — WEIGHT ~ LBS

CONCEPT	METHOD	2 MEN-7 DAYS (KG)	6 MEN-7 DAYS (KG)	4 MEN-45 DAYS (KG)
FOOD	FREEZE DRIED & FROZEN	54 (24.4)	161 (73)	665 (300)
LiOH	COMPOUND & CANISTER	69 (31.2)	186 (84)	809 (364)
EMERG. OXYGEN	RE PRESS & PLSS RESUPPLY	37 (16.7)	37 (16.7)	259 (117)
ATMOS. STORAGE				
CM CHARGE	O_2 & N_2 at 14.7 PSIA	95 (43)	95 (43)	95 (43)
OXYGEN	METABOLIC, LEAKAGE, ETC.	28 (12.7)	80 (36.1)	324 (146)
NITROGEN	HIGH PRESS.	3 (1.36)	3 (1.36)	17 (7.7)
WATER STORAGE	POTABLE	*	*	*
TOTAL CONSUMABLES		286 (129)	562 (254)	2169 (980)
TOTAL EC/LSS		2177 (985)	2453 (1110)	4060 (1835)

*WATER FROM FUEL CELLS

The regenerative (closed) system, such as molecular sieve or steam desorbed resin for the removal of CO_2, Sabatier or Bosch reactor for water recovery from CO_2 reduction, wick feed or gas circulation for O_2 recovery from water electrolysis are competitive for the longer mission stay times and larger crew sizes. The fixed equipment and spare weight, plus the electrical power penalty of these systems, are prohibitive for the tug program.

The fixed-equipment weight values in the tables include system spare weight. Expendable items such as chemicals normally programmed for replacement were included as a consumable. The total consumables did not include the weight of stored water. Water generation from fuel cell operation was sufficient to meet the crew drinking and washing requirements. CM configurations identifying the various location of equipment for the EC/LSS subsystem are shown in Figures A-2, A-3, and A-4.

PRIMARY DESIGN CRITERIA

The primary design criteria influencing the design of the EC/LSS are summarized in this section for reference. The criteria are given in the following categories: (1) Summary of EC/LSS Requirements (Table A-3), (2) Space Tug Mission Definition (Table A-4), (3) Crew Requirements (Table A-5) and (4) EC/LSS Subsystem Design Requirements (Table A-6). The criteria are listed in a form to provide a quick review of the EC/LSS design basis.

Requirements and Constraints

For the safety and comfort of the crewmen, the EC/LSS provides a pressurized oxygen-containing atmosphere, removes metabolic waste products, and meets food and personal hygiene requirements.

During tug manned operation, the cabin oxygen partial pressure and total pressure are maintained at levels suitable for sustained occupancy. A condensing heat exchanger and a lithium-hydroxide canister remove water vapor and carbon dioxide, respectively. Trace contaminants are controlled by leakage and adsorption on activated charcoal. Extra-tug activity uses portable life support system (PLSS) units, which are recharged by the tug subsystems, for life support.

The EC/LSS has significant interfaces with the electrical power, reactant storage, and structural subsystems. Electrical energy is required for operation of compressors, fans, pumps, and control and for recharging the PLSS batteries. An atmosphere supply is required continuously during manned operation to make up metabolic oxygen consumption and leakage. Water for drinking, for reconstitution, and PLSS cooling during EVA is obtained from storage tanks and from fuel-cell water generation.

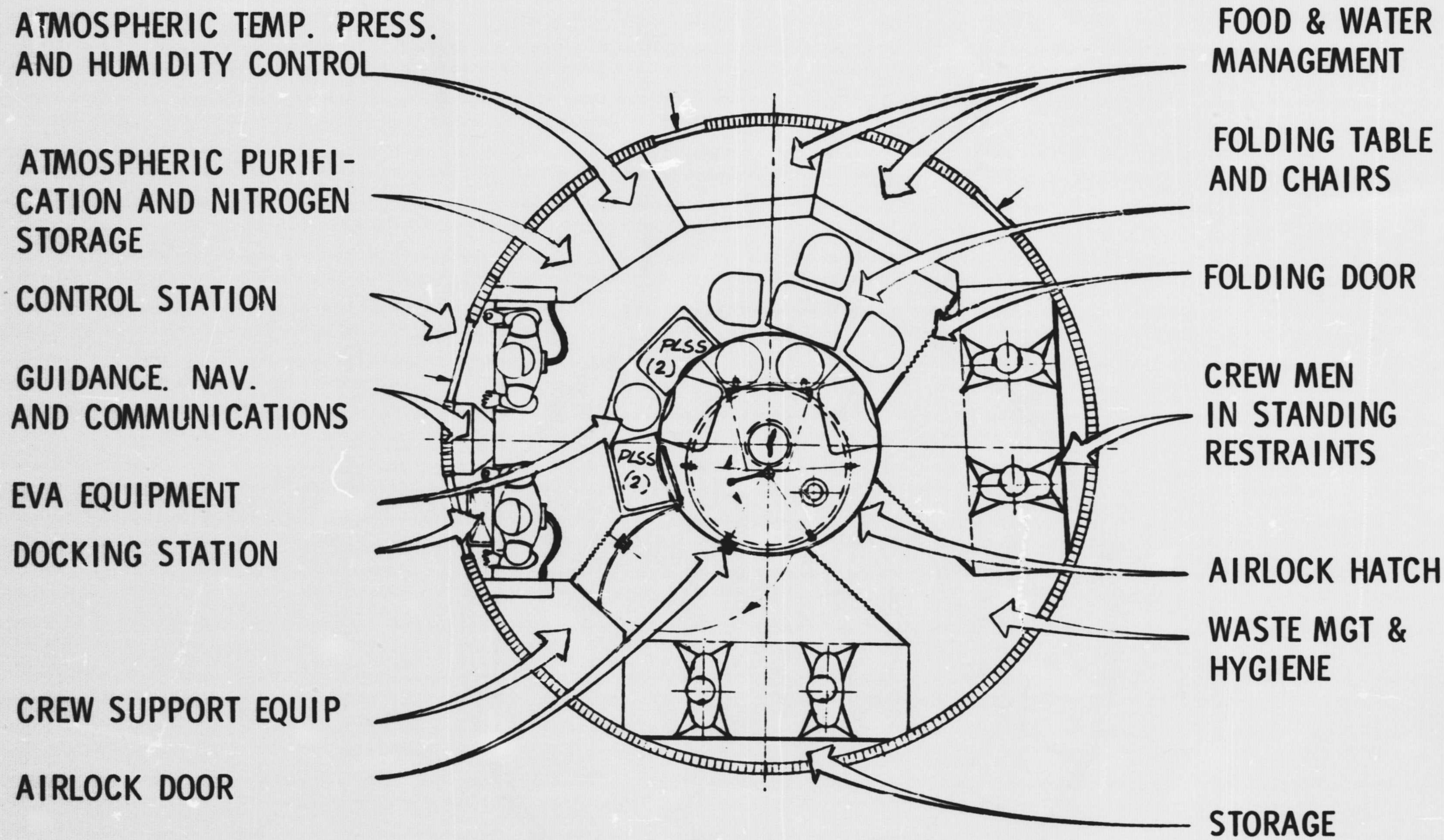


Figure A-2. Six-Man Crew Transport Seven-Day Configuration

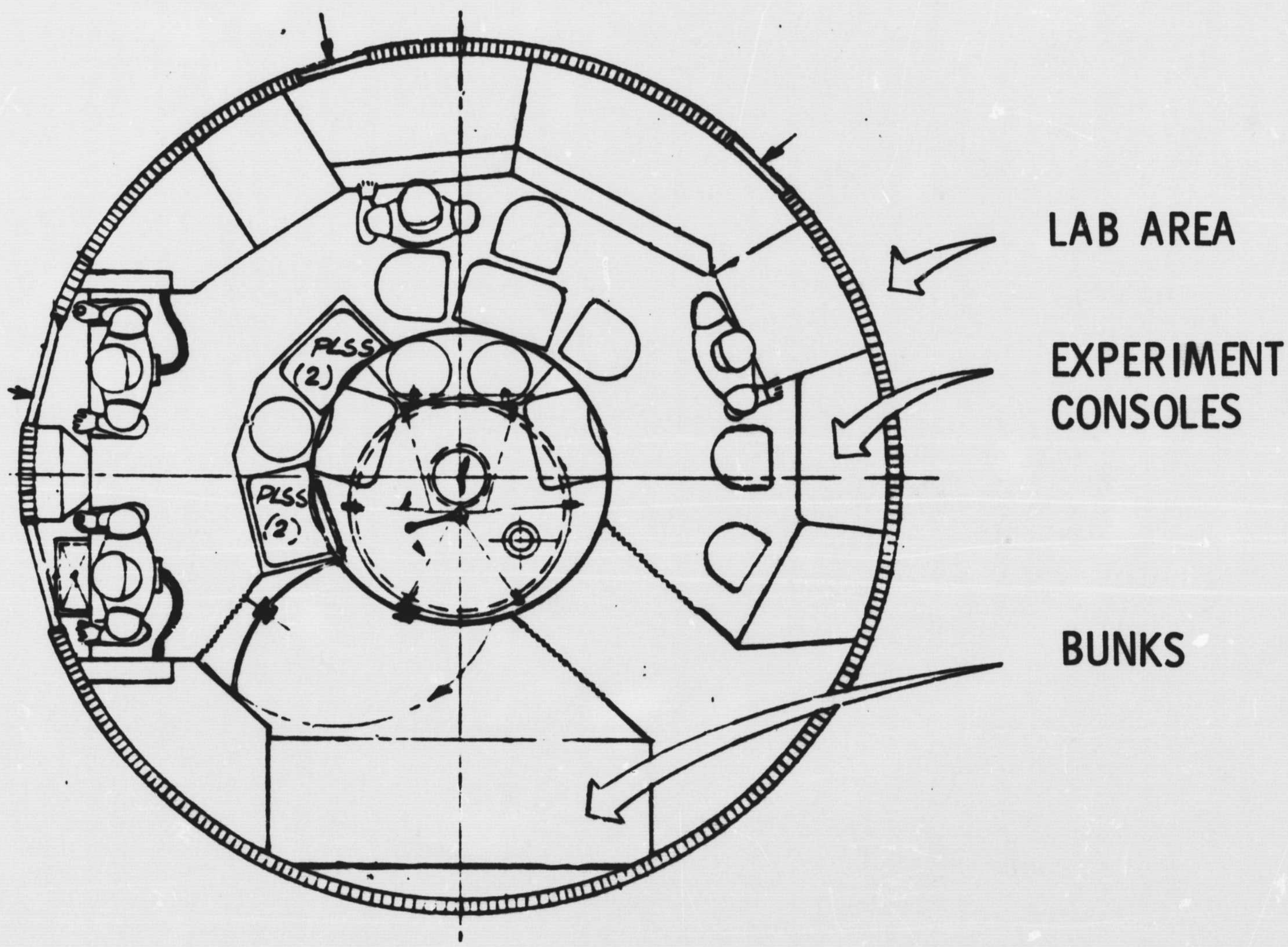


Figure A-3. Four-Man Lunar Lander - 28-Day-Stay Configuration

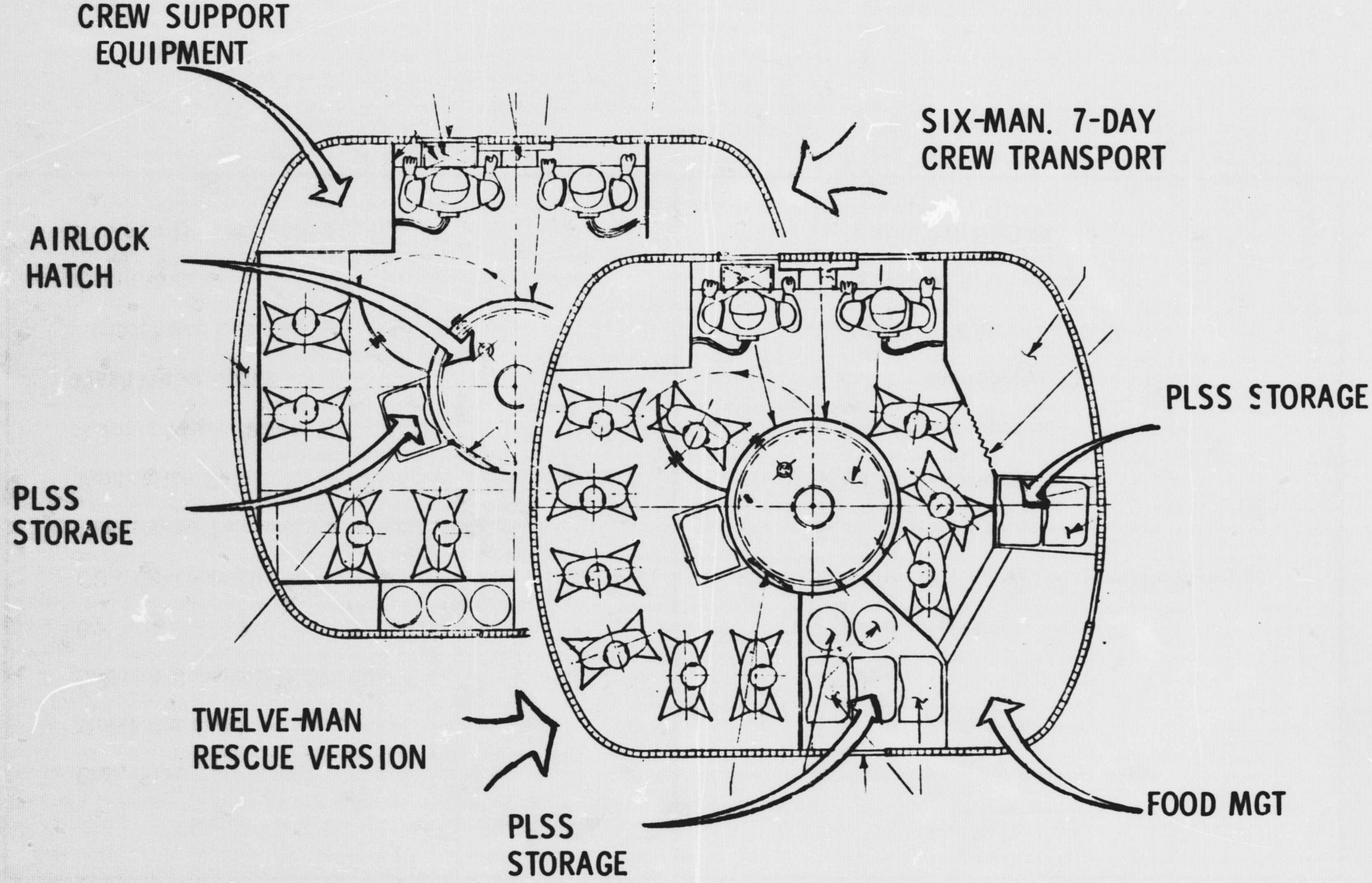


Figure A-4. Horizontal Cylinder Crew Module Configuration

Table A-3. Summary of ECLSS Requirements

Item	Characteristics
Crew Size	6 men normal, 12 men maximum
Metabolic Load	11,900 Btu/day-man
Onboard Consumable Storage	45 days
O_2	1.84 lb/man-day
CO_2 Concentration	5.0 mm Hg nominal, 7.6 mm Hg maximum
Atmosphere Temperature Selectability	65 to 75°F, nominal
Ventilation Rate	40 ft/min, nominal
Potable Water Usage	6 lb/man-day (approx.)
Wash Water Usage	4 lb/man-day (approx.)
Atmospheric Leakage	0.7 to 1.0 lb/day (N_2/O_2)
Cabin Volume	1250 ft^3 at 5.0 psia
Radiator Heat Rejection	5,000 to 20,000 Btu/hr

Table A-4. Space Tug Mission Definition

Item	Characteristics
Crew Size	Nominal: 6 men; Maximum: 12 men
Mission Duration	Nominal: 7 days; Maximum: 45 days
Resupply Period	Nominal: 7 days; Maximum: 45 days
Gravity	0 to 1 g (includes artificially induced gravity)
Vehicle Environment	Space vacuum to 14.7 psia
Maintenance	Inflight repair and maintenance
Power Supply	3.5 kw fuel cells (1.0 lb/watt penalty)
Emergency Reserve on Expendables	One day of essential requirements
Launch Expendables/Spares	45-day supply
Vehicle Configuration	See figures 1-2 through 1-4
Vehicle Total Volume	1250 to 2660 ft^3
Vehicle Leakage Rate	.7 to 1.0 lb/day O_2/N_2
Heat Rejection	Space radiators

Table A-5. Crew Requirements

Item	Characteristics	
Metabolic heat (total)	Nominal:	11,900 Btu/man-day
	Range:	10,300 - 13,600 Btu/man-day
	Design maximum:	650 Btu/man-hour
Latent load	Nominal:	3400 Btu/man-day
	Range:	2320 - 8500 Btu/man-day
	Design maximum:	360 Btu/man-day
Sensible load	Nominal:	8500 Btu/man-day
	Range:	4920 - 9680 Btu/man-day
Oxygen consumption	Nominal:	1.84 lb/man-day
	Range:	1.67 - 2.45 lb/man-day
CO_2 production	Nominal:	2.25 lb/man-day
	Range:	1.98 - 3.0 lb/man-day
Food consumption (dry)	Nominal supply:	1.68 lb/man-day
	Nominal intake:	1.50 lb/man-day
Dried foods	1.04 lb/man-day	
Wet foods (dry part)	0.64 lb/man-day	
Frozen		
Canned		
Fresh		
Food not ingested	0.18 lb/man-day	
Water in wet foods	0.96 lb/man-day	
Package-dried foods	0.73 lb/man-day	maximum
Package-wet foods	0.45 lb/man-day	maximum
Latent water loss	Nominal:	3.40 lb/man-day
	Range:	2.78 - 8.5 lb/man-day
Urine production		
Water	Nominal:	3.45 lb/man-day
	Range:	2.54 - 4.48 lb/man-day
Solids		0.13 lb/man-day
Feces production		
Water	0.25 lb/man-day	
Solids	0.13 lb/man-day	
Metabolic balance (lb/man-day)		
Input	6.32 water	
	1.84 O_2	
	1.50 dry food	
Output	3.40 latent H_2O	
	3.45 urine H_2O	
	0.25 fecal H_2O	
	2.25 CO_2	
	0.13 urine solids	
	0.13 fecal solids	
	0.05 hair, skin, etc.	

Table A-6. ECLSS Subsystem Design Requirements

Item	Characteristics
Total Pressure	5.0 psia nominal, 3.5 psia design minimum
O_2 Partial Pressure	3.5 psia nominal, 3.7 psia maximum
Diluent	Nitrogen
CO_2 Partial Pressure	Nominal: 7.6 mm Hg maximum Emergency maximum: 15.0 mm Hg for 2 hr
Cabin Trace Contaminants	See Table 2-5
Atmosphere Temperature	Control range between 65 and 75 F; control tolerance ±3 F of selected temperature
Cabin Ventilation Rate	40 ft/min nominal; 15 ft/min minimum; 100 ft/min maximum
Cabin Humidity	Absolute humidity range: 8 mm Hg to 12 mm Hg ppH_2O; relative humidity range: 30 to 50% approx; minimum dewpoint: 57 F
Water Requirements Model	
Crew Water Consumption	6.54 lb/man-day nominal supply; 6.32 lb/man-day nominal intake; 5.89-12.0 lb/man-day range
Drinking & added to dried food Water stored in other food Residual not ingested	5.17 lb/man-day .96 lb/man-day 0.22 lb/man-day
Wash Water	
Partial body hygiene (hand and face, hair groom)	4.0 lb/man-day
Other Water Requirements	
Urine Flush Vehicle Leakage	3.5 lb/man-day 0.3 lb/day
Water Purity	See Table 2-6
Special Life Support	
IVA EVA Expendables - Oxygen - Water	2 per mission with 2 men (normal) 2 per day with 4 men(28 day max.) 1.6 lb/4 hr. sortie 5.6 lb/4 hr. sortie

Manned Operation

An atmosphere will be supplied that provides a minimum oxygen partial pressure of 3.5 psia. The cabin carbon-dioxide partial pressure will not exceed 7.6 mm Hg. The normal operating level will be 5.0 mm Hg.

The cabin will be controlled within the range 50 ±20 percent relative humidity by condensation and removal of water. If maintenance of a water balance is required, the condensed water may be collected, processed, stored, and used by other subsystems.

The above EC/LSS functions will be supported by a tug atmosphere ventilation flow rate. The ventilation flow will be adequate to remove sensible heat, water vapor, carbon dioxide, and trace contaminants and to provide a substantially uniform atmosphere temperature and composition throughout the cabin.

Airborne trace contaminants will be controlled by tug leakage, air-lock losses, and adsorption in the activated charcoal contained in the lithium-hydroxide canisters. If needed, a catalytic burner will be added for the removal of low-molecular-weight trace contaminants.

A waste management system will be included for processing of biologicals and packaging waste materials. All processed waste will be stored aboard the CM, except for urine.

The water management system will store water carried from earth for an initial charge. All other water used will be generated by the electrical power system. The water balance may include that reclaimed from humidity condensate and excess water from fuel cell generation for thermal control system cooling.

Sufficient quantities of food and personal hygiene provisions will be included for the mission duration. The food supply will provide a caloric food value of 3280 calories per man-day for life support. The personal hygiene provisions will be analyzed to include requirements for clothing, shaving, hair/nail clipping, and tooth care.

Pertinent Data

Table A-7 lists the pertinent data on trace contaminants utilized; the aerospace drinking water standards are defined in Table A-8.

Table A-7. Maximum Concentration and Production Rate of Trace Contaminants

Contaminant	Production Rates			Space Maximum Allowable Concentration (mg/m^3)
	Non-Biological (gm/day)	Biological (gm/day)	Total (gm/day)	
Acetone	10.20	0.0059	10.20	240
Acetaldehyde	2.50	0.0023	2.50	36
Acetic Acid	0.25		0.25	2.5
Acetylene	2.50		2.50	180
Acetonitrile	0.25		0.25	7
Acrolein	0.25		0.25	0.25
Allyl Alcohol	0.25		0.25	0.5
Ammonia	2.50	12.	14.50	3.5
Amyl Acetate	0.25		0.25	53
Amyl Alcohol	0.25		0.25	36
Benzene	2.50		2.50	8
n-Butane	2.50		2.50	180
iso-Butane	0.25		0.25	180
Butene-1	2.50		2.50	180
cis-Butene-2	0.25		0.25	180
trans-Butene-2	2.50		2.50	180
1, 3 Butadiene	2.50		2.50	220
iso-Butylene	0.25		0.25	180
n-Butyl Alcohol	2.50	0.036	2.54	30
iso Butyl Alcohol	0.25		0.25	30
sec-Butyl Alcohol	0.25		0.25	30
tert-Butyl Alcohol	0.25		0.25	30
Butyl Acetate	0.25		0.25	71
Butraldehydes	0.25		0.25	70
Butyric Acid	0.25		0.25	14
Carbon Disulfide	0.25		0.25	6
Carbon Monoxide	2.50	0.4	2.9	29
Carbon Tetrachloride	0.25		0.25	6.5
Carbonyl Sulfide	0.25		0.25	25
Chlorine	0.25		0.25	1.5
Chloroacetone	0.25		0.25	100
Chlorobenzene	0.25		0.25	35
Chlorofluoromethane	0.25		0.25	24
Chloroform	2.50		2.50	24
Chloropropane	0.25		0.25	84
Caprylic Acid		X	0.25	155

Table A-7. Maximum Concentration and Production Rate of Trace Contaminants (Cont)

Contaminant	Production Rates			Space Maximum Allowable Concentration (mg/m^3)
	Non-Biological (gm/day)	Biological (gm/day)	Total (gm/day)	
Cumene	0.25		0.25	25
Cyclohexane	2.50		2.50	100
Cyclohexene	0.25		0.25	100
Cyclohexanol	0.25		0.25	20
Cyclopentane	0.25		0.25	100
Cyclopropane	0.25		0.25	100
Cyanamide	0.25		0.25	45
Decalin	0.25		0.25	5.0
1, 1 Dymethyl cyclohexane	0.25		0.25	120
trans 1, 2, dimethyl Cyclohexane	0.25		0.25	120
2, 2 Dimethyl butane	0.25		0.25	93
Dimethyl Sulfide	0.25		0.25	15
1, 1 Dichloroethane	2.50		2.50	40
Di iso Butyl Ketone	0.25		0.25	29
1, 4 Dioxane	2.50		2.50	36
Dimethyl Furan	0.25		0.25	3.0
Dimethyl Hydrazine	0.25		0.25	0.1
Ethane	2.50		2.50	180
Ethyl Alcohol	2.50	0.12	2.62	190
Ethyl Acetate	2.50		2.50	140
Ethyl Acetylene	0.25		0.25	180
Ethyl Benzene	0.25		0.25	44
Ethylene Dichloride	0.25		0.25	40
Ethyl Ether	2.50		2.50	120
Ethyl Butyl Ether	0.25		0.25	200
Ethyl Formate	2.50		2.50	30
Ethylene	2.50		2.50	180
Ethylene Glycol	0.25		0.25	114
trans 1, Methyl 3 Ethyl Cyclohexane	0.25		0.25	117

Notes: Decalin = decahydronaphthalene
1, 4 Dioxane = p-Dioxane
Ethylene dichloride = ethylene chloride = 1, 2 dichloroethane

Table A-7. Maximum Concentration and Production Rate of Trace Contaminants (Cont)

Contaminant	Production Rates: Non-Biological (gm/day)	Biological (gm/day)	Total (gm/day)	Space Maximum Allowable Concentration (mg/m^3)
Ethyl Sulfide	0.25		0.25	97
Ethyl Mercaptan	0.25	X	0.25	2.5
Freon 11	2.50		2.50	560
Freon 12	2.50		2.50	500
Freon 21	0.25		0.25	420
Freon 22	0.25		0.25	350
Freon 23	0.25		0.25	12
Freon 113	0.25		0.25	700
Freon 114	2.50		2.50	700
Freon 114 unsym	0.25		0.25	700
Freon 125	0.25		0.25	25
Formaldehyde	0.25		0.25	0.6
Furan	0.25		0.25	3
Furfural	0.25		0.25	2
Hydrogen	2.50	0.6	3.10	215
Hydrogen Chloride	0.25		0.25	0.15
Hydrogen Fluoride	0.25		0.25	0.08
Hydrogen Sulfide		0.0007	0.0009	1.5
Heptane	0.25		0.25	200
Hexene-1	0.25		0.25	180
n-Hexane	2.50		2.50	180
Hexamethylcyclotrisihexane	0.25		0.25	240
Indole	0.25	1.2	1.45	126
Isoprene	0.25		0.25	140
Methylene Chloride	2.50		2.50	21
Methyl Acetate	2.50		2.50	61

Notes: Freon 11 = Trichlorofluoromethane
Freon 12 = Dichlorodifluoromethane
Freon 21 = Dichlorofluoromethane
Freon 22 = Chlorodifluoromethane
Freon 23 = Fluoroform = Trifluoromethane
Freon 113 = Trichlorotrifluoroethane
Freon 114 = Dichlorotetrafluoroethane
Freon 125 = Pentafluoroethane

Table A-7. Maximum Concentration and Production Rate of Trace Contaminants (Cont)

Contaminant	Production Rates: Non-Biological (gm/day)	Production Rates: Biological (gm/day)	Production Rates: Total (gm/day)	Space Maximum Allowable Concentration (mg/m^3)
Methyl Butyrate	0.25		0.25	30
Methyl Chloride	0.25		0.25	21
2-Methyl-1 Butene	0.25		0.25	1430
Methyl Chloroform	2.50		2.50	190
Methyl Furane	0.25		0.25	3
Methyl Ethyl Ketone	2.50		2.50	59
Methyl Isobutyl Ketone	0.25		0.25	41
Methyl Isopropyl Ketone	2.50		2.50	70
Methyl Cyclohexane	0.25		0.25	200
Methyl Acetylene	0.25		0.25	165
Methyl Alcohol	2.50	0.12	2.62	26
3-Methyl Pentane	0.25		0.25	295
Methyl Methacrylate	0.25		0.25	41
Methane	29.5	7.2	36.7	1720
Mesitylene	0.25		0.25	2.5
mono Methyl Hydrazine	0.25		0.25	0.035
Methyl Mercaptan			0.25	2
Naphthalene	0.25		0.25	5.0
Nitric Oxide	0.25		0.25	32
Nitrogen Tetroxide	0.25		0.25	1.8
Nitrogen Dioxide	0.25		0.25	0.9
Nitrous Oxide	0.25		0.25	47
Octane	0.25		0.25	235
Propylene	2.50		2.50	180
iso-Pentane	2.50		2.50	295
n-Pentane	2.50		2.50	295

Notes: methyl ethyl ketone = 2-buranone
methyl isopropyl betane = 3-methyl 2 butanone = 2-methyl butanone-3
methyl acetylene = propine = propynl
mesitylene = 1, 3, 5 trimethyl benzene
Propylene = propene

Table A-7. Maximum Concentration and Production Rate of Trace Contaminants (Cont)

Contaminant	Production Rates Non-Biological (gm/day)	Biological (gm/day)	Total (gm/day)	Space Maximum Allowable Concentration (mg/m^3)
Pentene-1	0.25		0.25	180
Pentene-2	0.25		0.25	180
Propane	2.50		2.50	180
n-Propyl Acetate	0.25		0.25	84
n-Propyl Alcohol	2.50		2.50	75
iso-Propyl Alcohol	2.50		2.50	98
n-Propyl Benzene	0.25		0.25	44
iso-Propyl Chloride	0.25		0.25	260
iso-Propyl Ether	0.25		0.25	120
Proprionaldehyde	0.25		0.25	30
Propionic Acid	0.25		0.25	15
Propyl Mercaptan			0.25	82
Propylene Aldehyde	0.25		0.25	10
Pyruvic Acid		4.53	4.53	0.9
Phenol	0.25	4.53	4.78	1.9
Skatol			0.25	141
Sulfur Dioxide	0.25		0.25	0.8
Styrene	0.25		0.25	42
Tetrachloroethylene	0.25		0.25	67
Tetrafluoroethylene	0.25		0.25	205
Tetrahydrofurane	0.25		0.25	59
Toluene	2.50		2.50	75
Trichloroethylene	2.50		2.50	52
1, 2, 4 Trimethyl Benzene	0.25		0.25	49
1, 1, 3 Thrimethyl cyclohexane	0.25		0.25	140
Valeraldehyde			0.25	70
Valeric Acid			0.25	110
Vinyl Chloride	2.50		2.50	130
Vinyl Methyl Ether	0.25		0.25	60
Vinyldene Chloride	0.25		0.25	20
O-Xylene	2.50		2.50	44
m-Xylene	2.50		2.50	44
p-Xylene	2.50		2.50	44

Notes: Propylene aldehyde = crotonaldehyde
Tetrachlaroethylene = Perchloraethylene
Vinyl methyl ether = methoxy ethane

Table A-8. Aerospace Potable Water Specification

Chemical Requirements	Milligram/liter or ppm	Source
Total solids	1000.0	Space Science Board
Cadmium	.05	Space Science Board
Chromium, hexavalent	0.05	Space Science Board
Copper	3.0	Space Science Board
Lead	0.2	Space Science Board
Silver	0.5	Space Science Board
Iron	1.0	Air Force Potable Water Standard for 1967
Manganese	0.1	Air Force Potable Water Standard for 1967
Zinc	15.0	Air Force Potable Water Standard for 1967
Mercury	0.005	NASA PF-SPEC-1
Nickel	1.0	NR
Chemical oxygen demand	0.5	NR
Selenium	0.05	U.S. Dept. of Public Health
Other Standards	**Units**	**Source**
Color	15.0	Air Force Potable Water Standard for 1967
Turbidity	25.0	Air Force Potable Water Standard for 1967
Taste and odor (odor No.)	3.0	Air Force Potable Water Standard for 1967
pH	6.0-8.0	NASA PF-SPEC-1
Microorganisms	Essentially no coliforms	U.S. Dept. of Public Health
Particulates	Level 3	NR SPEC MA0610-017

Space Division
North American Rockwell

SYSTEM DESCRIPTION

The EC/LSS subsystems described herein are those selected on a weight, power, and volume penalty basis. Other evaluation criteria, such as reliability, safety, performance, availability, interfaces, and system flexibility, have not been introduced and may alter the above selections somewhat.

Atmospheric Storage

The atmospheric storage assembly, which is composed of an oxygen storage subassembly and a nitrogen storage subassembly, contains subcritical nitrogen tankage for leakage nitrogen makeup, and high-pressure oxygen for PLSS refill, CM repressurization, and other emergency use. The main oxygen supply for the CM and crew requirement will be stored in the main propulsion tanks. With proper filtering, the oxygen can be delivered to the CM in the gaseous state for normal metabolic consumption and CM leakage. There will be heaters and flow devices to control the rate of oxygen usage. Oxygen storage proved to be less of a vehicle penalty than generating oxygen by the water electrolysis method.

CO_2 Management

The CO_2 management assembly evaluated for this program consisted of subassemblies for CO_2 removal, water recovery from CO_2 reduction, and O_2 recovery from water electrolysis. All of these regenerable types were traded off against a nonregenerable system. Selection was made on a basis of weight, power, and volume penalty.

Several processes can provide carbon dioxide control. The two active processes for CO_2 control investigated in this program were nonregenerable absorption by lithium hydroxide and regenerable types such as molecular sieve, steam desorbed resin, and solid amine. The first process is irreversible and characterized by the use of expendables. The other processes, such as the molecular sieve, is a cyclic process in which one bed is regenerated while the alternate is being used. Both of these active control techniques may be supplemented to a small degree by normal cabin leakage. Systems based on absorption by nonregenerable sorbents are generally simple. The cyclic processes used in regenerable systems are, by comparison, more complex, but may offer weight savings for long-duration missions.

Tradeoff analyses based upon weight, power, and volume penalties show that for the seven-day tug mission, the nonregenerable absorption process using lithium hydroxide is by far the optimum approach. For the

45-day mission (28-day stay on the lunar surface), the regenerable process requires less volume, and the weight and power penalty combined are comparable.

Tradeoff studies were also conducted to determine the practicality of generating water from CO_2 reduction and the manufacturing of oxygen by water electrolysis. Results from the tradeoff analysis show that water and oxygen storage is by far more practical.

Therefore, for tug application, the CO_2 management assembly will consist of a nonregenerable (open) type of subsystem using lithium hydroxide for CO_2 removal. No provisions will be made for water recovery from CO_2 reduction or O_2 recovery from water electrolysis.

Atmosphere Control

The atmospheric control assembly consists of subassemblies for trace contaminant, humidity, temperature and pressure control, and circulation.

The trace contaminant control subassembly removes gaseous trace contaminants, aerosols, particulates, and bacteria from the cabin atmosphere. It consists of a debris trap, filters, ammonia sorbent bed, acid gas sorbent bed, catalytic oxidizer, and regenerable charcoal beds.

Atmospheric priessure control and composition is maintained by the pressure control subassembly. It consists of a total pressure sensor and partial pressure sensors, regulators, etc. to provide the proper scheduling of N_2 and O_2 to maintain the respective partial pressure levels.

The humidity control subassembly and the thermal control subassembly control the cabin temperature and the humidity level in the vehicle. Their functions are provided by packages consisting of fans, sensible heat exchangers, condensing heat exchangers, a water separator, check valves, temperature control valves, and fan bypass valves.

The circulation subassembly consists of additional fans to provide the necessary ventilation flow. Redundancy in all subassemblies has been considered for maintaining crew requirements during the 12-man rescue mission.

Active Thermal Control Assembly

The active thermal control assembly consists of the radiator loop and the cabin thermal loop assemblies.

The radiator loop is a refrigerant transport circuit that removes heat from the cabin thermal loop in the liquid transport circuit interface heat exchangers and rejects the heat to space via the vehicle radiator. The cabin thermal loop is a liquid circuit that removes heat from all the equipment requiring heat rejection and transfers this heat to the radiator loop in the interface heat exchangers.

Water Management

Water reclamation systems were evaluated and compared with the water storage-type of water supply. The regenerative systems evaluated were air evaporation, vapor compression, vacuum distillation pyrolysis, and vapor diffusion. Results indicate that the air evaporation has the least penalty based upon weight, power, and volume even though the expendable weight is quite large. This system does represent a very high penalty when compared with a water-storage-type system.

For tug applications, the water-storage system is practical because of its minimum water storage requirement at liftoff supplemented by fuel cell water during the mission. This approach results in a very low weight penalty to the vehicle.

The water storage subassembly consists of separate storage tanks for wash and potable water. Water bacteria control also incorporated.

Waste Management

The waste management assembly consists of subassemblies for collection of urine, collection and processing of solid wastes (feces and trash), and disposal of waste.

The Apollo approach has direct application to the tug program because of the similar operational systems. Urine will be vented overboard, and provisions will be made to prevent bacteria growth in the urine system. There are several potential approaches to the processing of the fecal waste. Chemical treatment as in Apollo, or vacuum drying, results in the minimum weight penalty to the vehicle. Feces could be collected and vacuum-dried in the fecal collection and processing subassembly. All trash could also be vacuum-dried and compacted for storage in the trash management subassembly.

Hygiene

The provisions for personal hygiene include clothing and medical and dental support, as well as general housekeeping functions. Clothing for the crew will be based on use of expendable clothing inserts. No cleaning or laundering of clothing is planned for this mission duration.

The category of cleaning supplies include wet packs for body cleaning. The water balance includes water for hygiene pruposes. Body shower provisions are not planned for the tug program.

Food Management

The food management assembly consists of subassemblies for food and provisions for its storage, preparation, serving, and disposal.

Crew Accommodations

The crew accommodations assembly consists of all the exercise and entertainment, as well as furniture for the crew.

Emergency and Auxiliary Life Support

This assembly consists of the PLSS, IVA/EVA-suited operation support equipment, and emergency oxygen subassemblies, and has provisions for fire fighting, repressure capabilities, and survival food and water. IVA-suited operation is provided by the emergency oxygen supply. EVA operation will be accomplished with an advanced PLSS with a closed loop or regenerative cycle concept.

SYSTEM TRADEOFF EVALUATION

The functional matrix chart (Figure A-5) shows the specific assemblies and associated functions within the EC/LSS investigated under this study program. The major candidates affected by tradeoff analysis were in the areas of CO_2 removal, CO_2 reduction for water recovery, water electrolysis for O_2 recovery, trace contaminant control, and water reclaimation. The remainder of the functional assemblies, such as atmospheric storage, active thermal control, waste management, water management, food management, atmospheric control, hygiene, crew accommodations, and emergency life support, were not included since these areas are not easily isolated as assemblies. Further, they are not easily evaluated at the assembly level since their selection rests heavily on subsystem integration requirements and design. For tug applications, these systems were scaled down directly from space station data based upon crew size.

Tradeoff studies were performed to evaluate processes, such as regenerative (closed) types and nonregenerative (open) types of EC/LSS. Processes were evaluated and selected on a weight, power, and volume basis only. Tradeoff data were based on the latest space station information

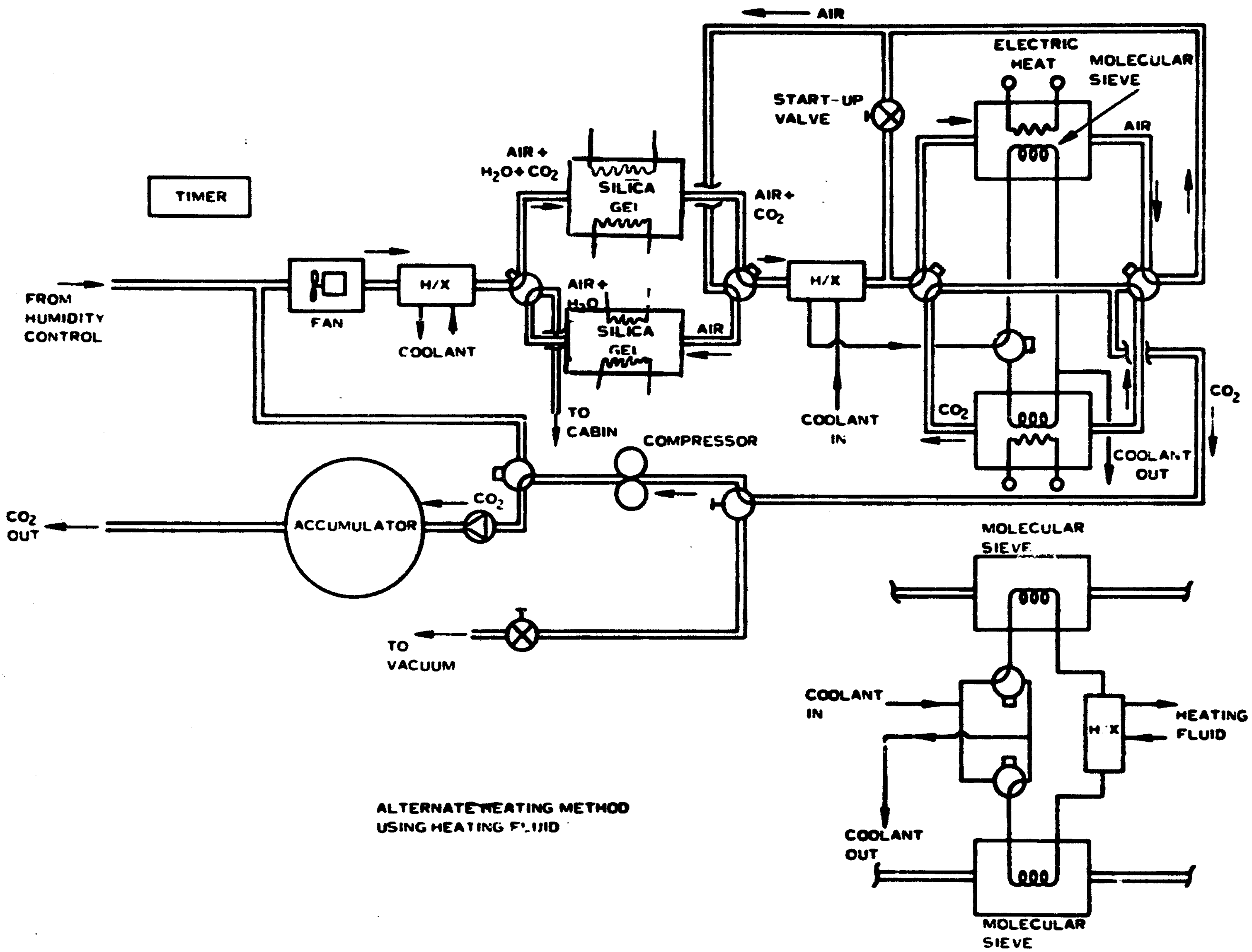


Figure A-5. Molecular Sieve Concept

available. These data were prepared from information originally supplied by Hamilton Standard (Reference A-1 and A-2) and converted into parametric curves. These parametric data were generated to help evaluate system penalties based upon weight, power, and volume. For space station ECLSS assembly data of weight, power, and volume to be adapted for tug applications, the space station trade study data utilizing process rates were converted into crew size and mission duration and plotted in parametric curve form. (See final paragraph of this appendix.) The parametric data generated herein for determining assembly weight, power, and volume as a function of crew size and mission duration are considered relative and are intended to be used as tools only for the valuation and selection of system or assembly processes. Tradeoff evaluation was conducted over a range of tug mission objectives to provide support for a two- or six-man seven-day mission capability, with extended system growth for a four-man 45-day mission, including a 28-day stay on the lunar surface.

Tradeoff data were based upon the following ground rules and assumptions:

1. System fixed weight and electrical power penalties were assumed to be directly proportional to crew size.

2. System spares weight was assumed to be directly proportional to mission time.

3. System expendable weight was assumed to be a function of the number of crew and mission duration.

4. The following process rates were used to convert space station trade study data into parametric data for tug crew size and mission duration:

   a. CO_2 removal — 2.25 lb CO_2/man day

   b. CO_2 reduction — 2.25 lb CO_2/man day

   c. H_2O reclamation — 0.618 lb H_2O/hr per man

   d. H_2O electrolysis — 0.193 lb H_2O/hr per man

   e. Contaminant control — Mission size (No. of men)

5. Tug volume penalties were obtained from a computed density factor based upon space station data of weight and volume. A separate density factor was used for each assembly concept.

CO_2 Management

Tradeoff studies were conducted in order to select the optimum CO_2 management system that would meet the tug crew and mission requirements. The areas of CO_2 management investigated were CO_2 removal, CO_2 reduction for water recovery, and H_2O electrolysis for O_2 recovery. Regenerative (closed) systems were evaluated and compared with a nonregenerative (open) type system on a weight, power, and volume penalty basis, and a selection was made.

CO_2 Removal

Carbon dioxide is generated continuously as a metabolic product in the tug during manned operations. The carbon dioxide must be removed from the tug atmosphere in order for a normal partial pressure of 5.0 mm Hg and a maximum operating value of 7.6 mm Hg to be maintained. Several methods are considered for this purpose. Lithium hydroxide, for instance, reacts with carbon dioxide, and this reaction can be used to remove CO_2 from the atmosphere. The reaction is, for practical purposes, irreversible. Thus a system utilizing lithium hydroxide to remove CO_2 is termed a nonregenerative system.

Lithium hydroxide (LiOH) is a granular solid that readily absorbs carbon dioxide in air in the presence of water vapor. The chemical reaction that takes place is

$$2\,LiOH + CO_2 \xrightarrow{H_2O} Li_2\,CO_3 + H_2O + \text{Heat}$$

The LiOh combines with CO_2 in the presence of some water vapor to become $Li_2\,CO_3$, and H_2O and heat are liberated in the process. LiOH may be packed in a replaceable cartridge or canister and located in a loop in the entrance portion of the environmental control system. Filters must be provided to keep LiOH dust from being circulated into the cabin. For this study one pound of LiOH was assumed to absorb 0.925 pounds of CO_2. Hardware weight is assumed to be 60 percent of that of the sorber weight.

Substituting a material for LiOH that can be reactivated and reused many times is the basis for the design of a regenerative CO_2 removal system. For tug applications three types of regenerative CO_2 removal systems were selected for tradeoff purposes: (1) molecular sieve, (2) steam desorbed resin, and (3) solid amine. Schematics of these concepts are presented in Figures A-5, A-6, and A-7. Description and operations are not included herein but can be obtained from References A-1 and A-2.

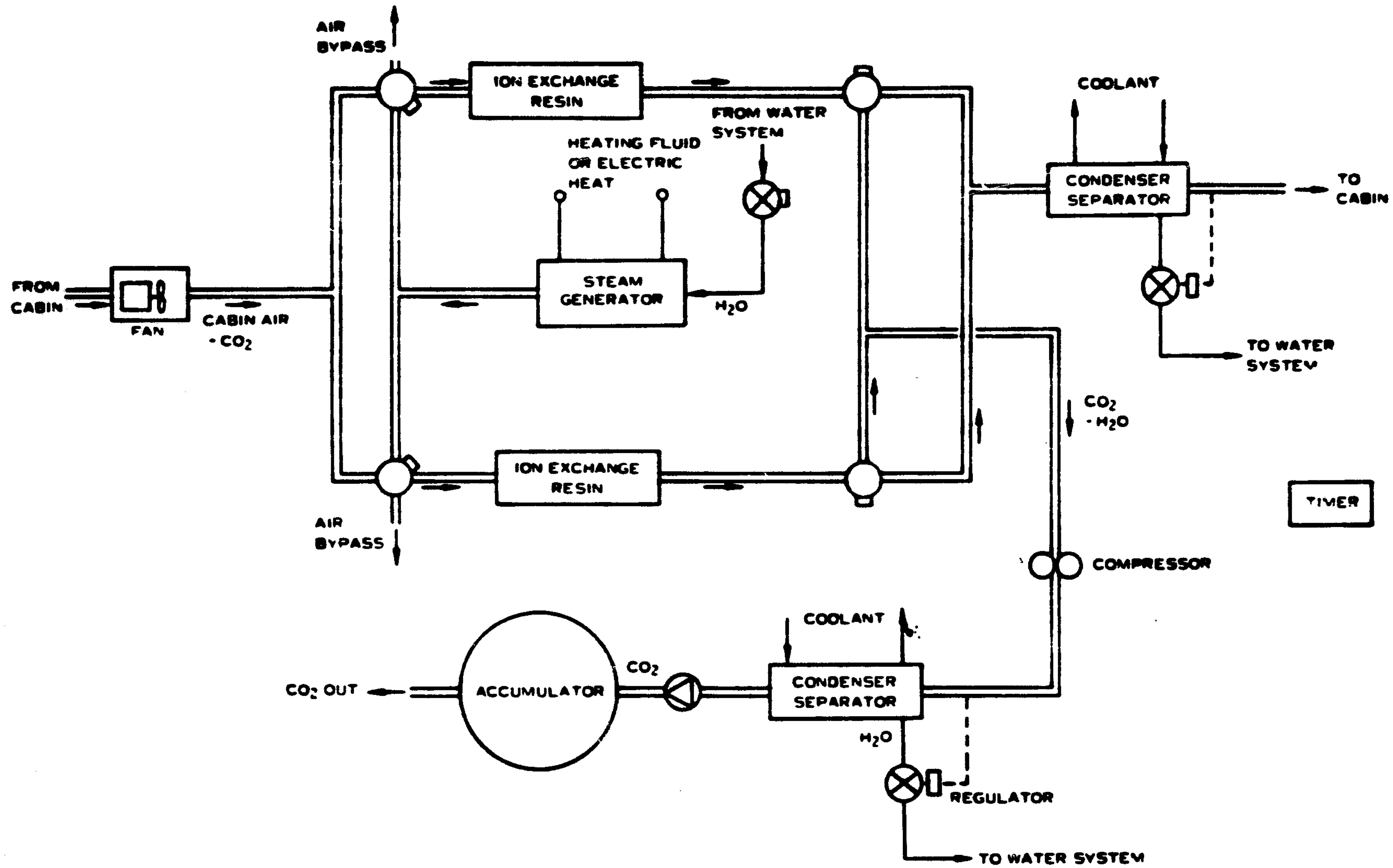


Figure A-6. Steam-Desorbed Resin Concept

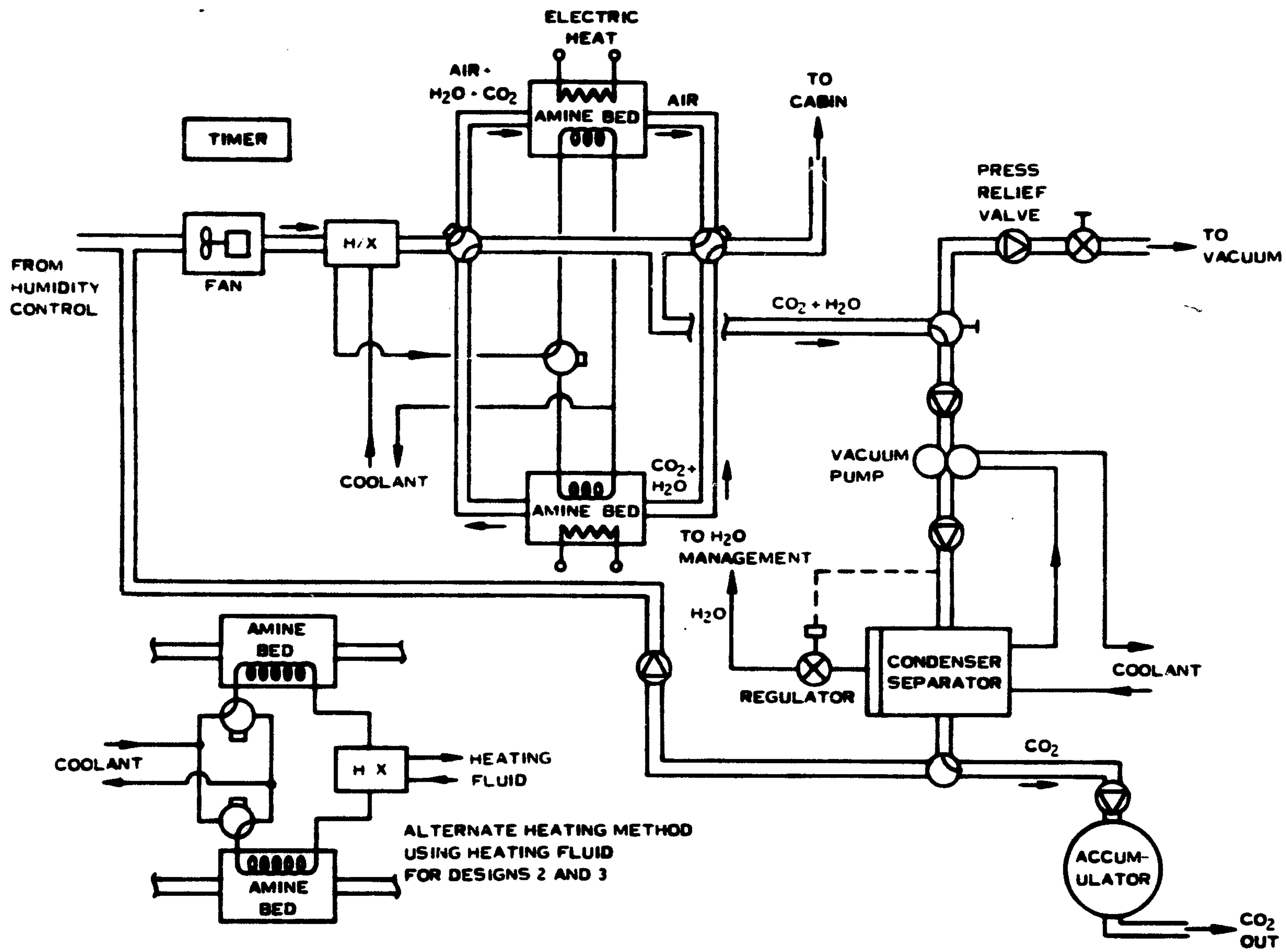


Figure A-7. Solid Amine Concept

Tradeoff curves were constructed show the fixed system weight, electrical power, and the spares weight for the three regenerative system. Bases for the curves were obtained from parametric data in Section 14. 1. 7. System fixed weight and electrical power were plotted as a function of crew size (Figure A-8). System spare weight was plotted against mission duration, (Figure A-9).

On a weight penalty basis (fixed and spare weight) the steam-desorbed resin concept has the minimum weight penalty. Considering the A-10 electrical power penalty, the molecular sieve requires the least power. Figure A-10 shows the total launch weight penalty (fixed plus spare weight) for the nonregenerative and regenerative systems. For a two-man tug concept, the LiOH sorption system shows a minimum weight up to 26 days of mission time, at which point the steam-desorbed resin regenerative system shows the least weight peanlty. For a six-man tug, the corss-over point between a nonregenerative and regenerative concept is about 14 days. Table A-9 summarizes the weight, power, and volume of all the candidates for the three missions: two men-seven days, six men-seven days and four men-45 days.

Review of the Table A-9 shows that the selected system candidate for the removal of CO_2 is the nonregenerative LiOH system. This system shows a very low weight penalty for the seven-day missions and is comparable with the steam-disorbed resin for the longer mission of 45 days when electrical power penalty is considered. A total weight penalty comparison based upon an electrical power penalty of one lb/watt for the longer mission favors the nonregenerative system. Results show a weight penalty of 809 pounds for LiOH as compared to 832 pounds for the steam desorbed resin concept.

CO_2 Reduction

Four regenerative concepts were evaluated for the reduction of CO_2 to form water and by-products. These concepts are schematically shown in Figure A-11, Sabatier - methane dump; Figure A-12, Bosch reactor; Figure A-13, solid electrolyte; and Figure A-14, Sabatier-methane cracking. Tradeoff study curves were constructed to show the system fixed weight and power penalties versus crew size (Figure A-15), system spare weight as a function of mission time (Figure A-16), and expendable weight as a function of man-days (Figure A-17). Review of the four concepts show that the Sabatier-methane dump has the lowest weight penalty for system fixed weight, spare weight and is also very low on electrical power demand. Figure A-17 indicates that the expendable weight is the highest by a large margin for the Sabatier-methane dump concept, whereas the Bosch reactor requires the minimum quantity of expendables. However, when comparing these concepts on a total system weight basis (Figure A-18), it can be shown that the Sabatier-methane dump has the minimum weight penalty for the tug mission

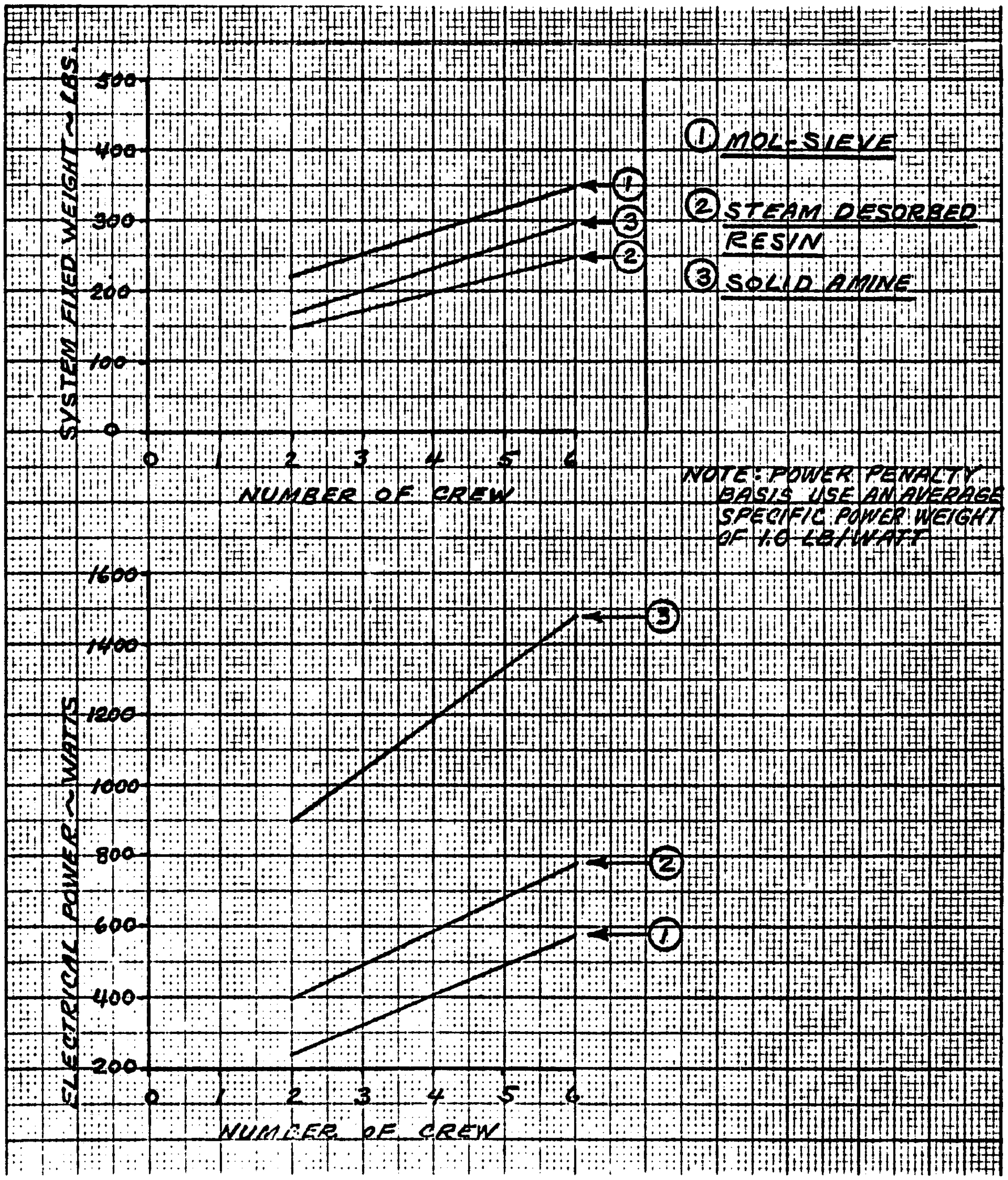


Figure A-8. CO_2 Removal System Weight Trade Study System Fixed Weight and Electrical Power Penalty

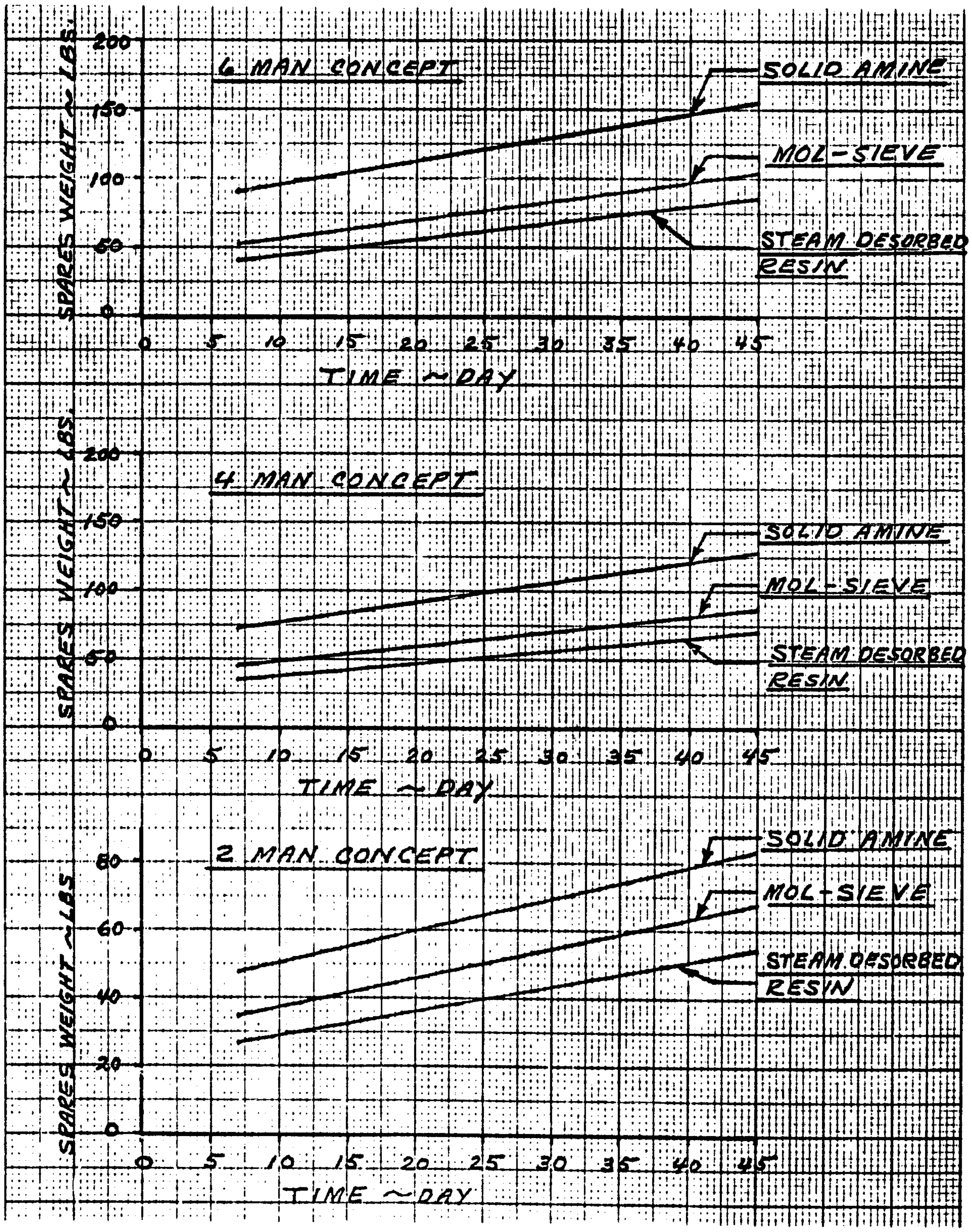


Figure A-9. CO_2 Removal Spares Weight Trade Study

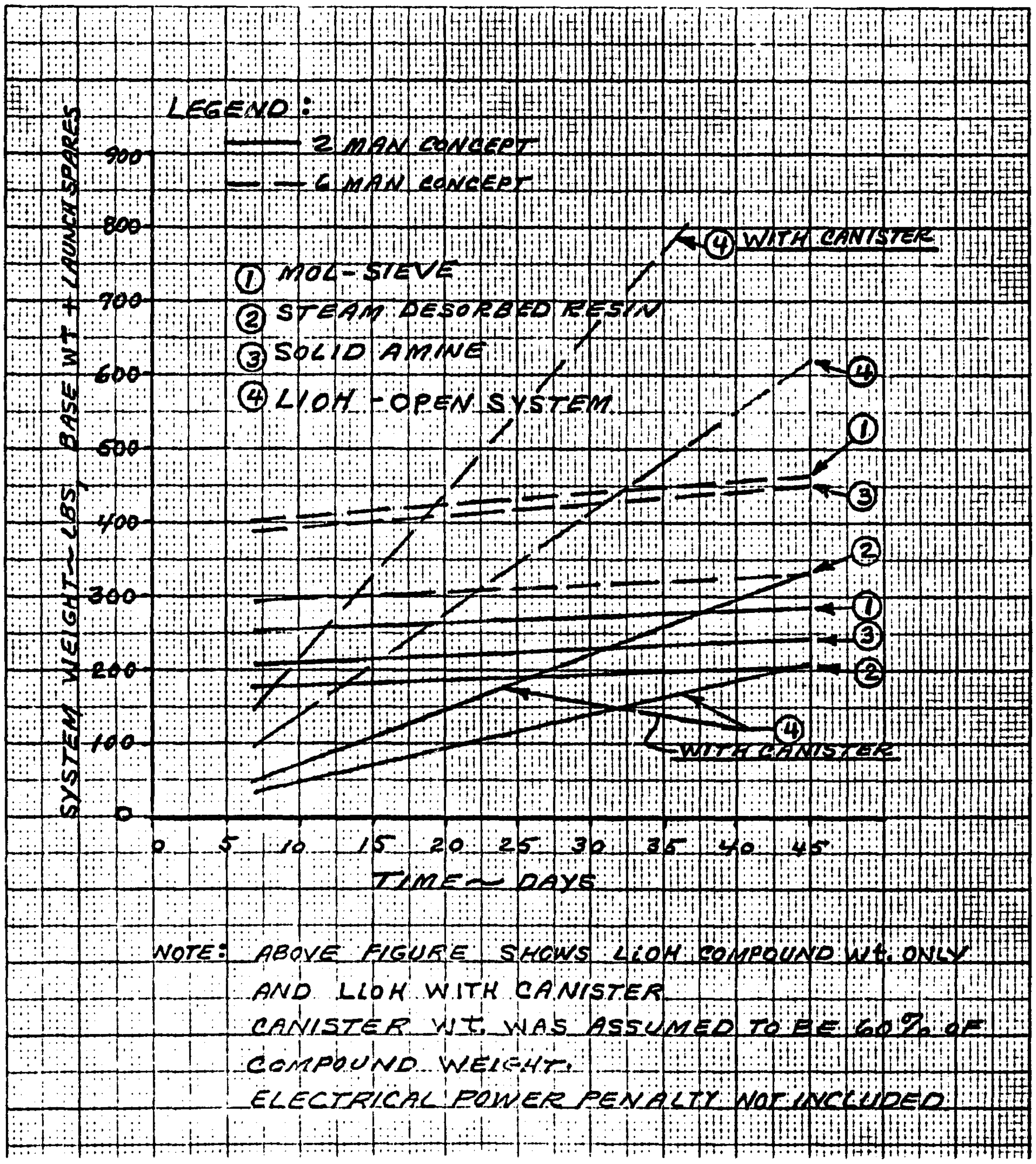


Figure A-10. CO_2 Removal System Weight Trade Study Total Launch Weight System Comparison

Table A-9. CO_2 Removal

MISSION	CANDIDATES	WEIGHT PENALTY ~ LBS				ELEC. PWR.		VOLUME ~ FT^3		
		Fixed	Spares	Expend	Total	Watts	Cycle	Fixed	Spares	Total
2 Men	(1) Mole Sieve	220	35	--	225	240	Cont.	7.6	1.2	8.80
7 Days	(2) Steam Desorbed Resin	150	27	--	177	400	Cont.	5.2	0.93	6.13
	(3) Solid Amine	150	48	--	198	900	Cont.	5.2	1.65	6.85
*	(4) LiOH	56	13	--	69	--	--	2.0	0.46	2.46
6 Men	(1) Mole Sieve	340	55	--	395	580	Cont.	11.70	1.90	13.60
7 Days	(2) Steam Desorbed Resin	235	42	--	277	780	Cont.	8.10	1.45	9.55
	(3) Solid Amine	285	92	--	377	1480	Cont.	9.80	3.16	12.96
*	(4) LiOH	152	34	--	186	--	Cont.	5.43	1.21	6.64
4 Men	(1) Mole Sieve	280	87	--	367	400	Cont.	9.65	3.0	12.65
45 Days *	(2) Steam Desorbed Resin	200	72	--	272	560	Cont.	6.90	2.48	9.38
	(3) Solid Amine	230	129	--	359	1180	Cont.	7.91	4.44	12.35
*	(4) LiOH	664	145	--	809	--	--	23.60	5.19	28.79

NOTE: Expendables include such items as filters, catalysts or chemicals normally programmed for replacement.

* Selected system candidate based on minimum weight and power penalty.

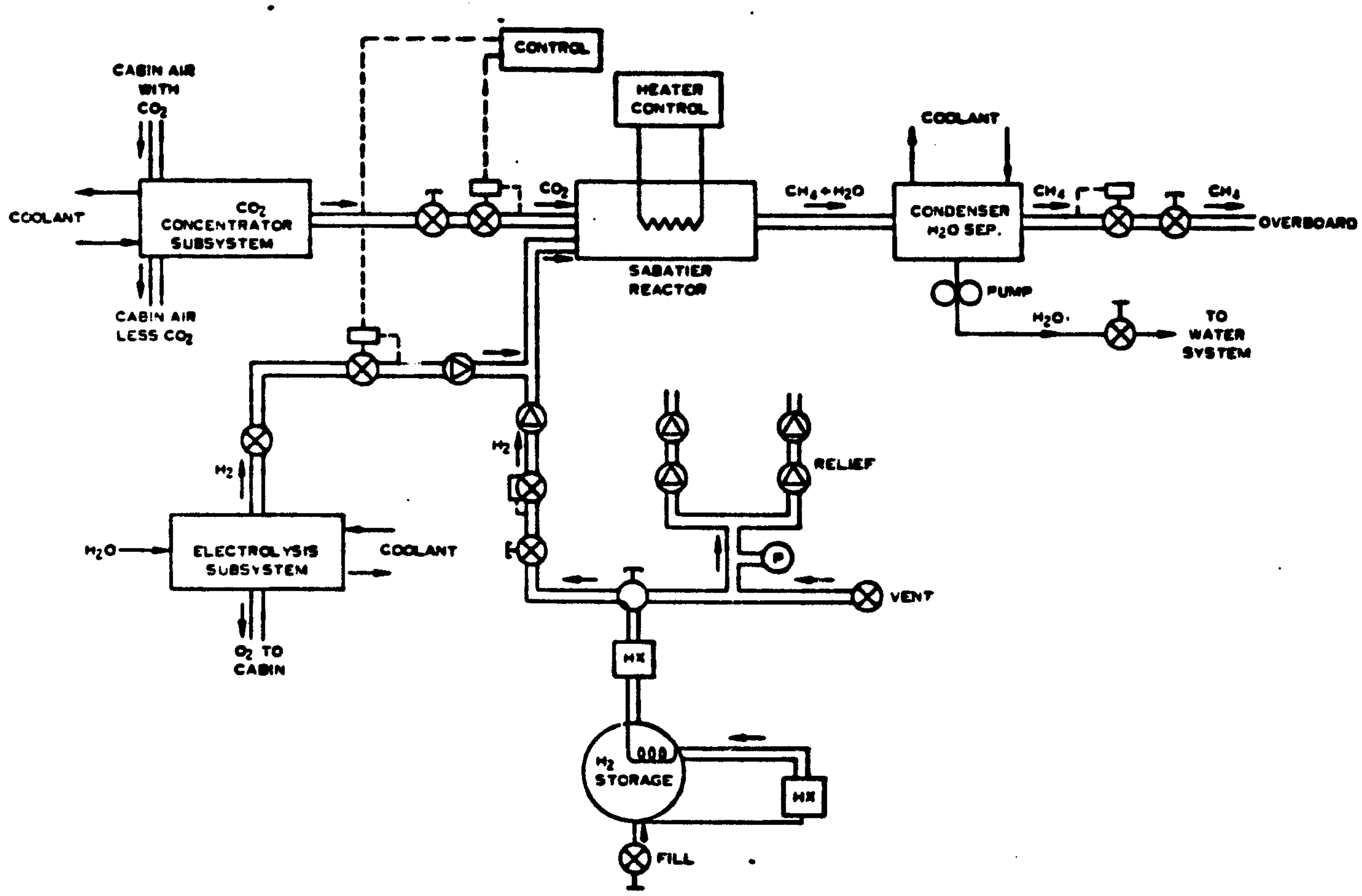


Figure A-11. Sabatier CO_2 Reduction With Methane Dump Concept

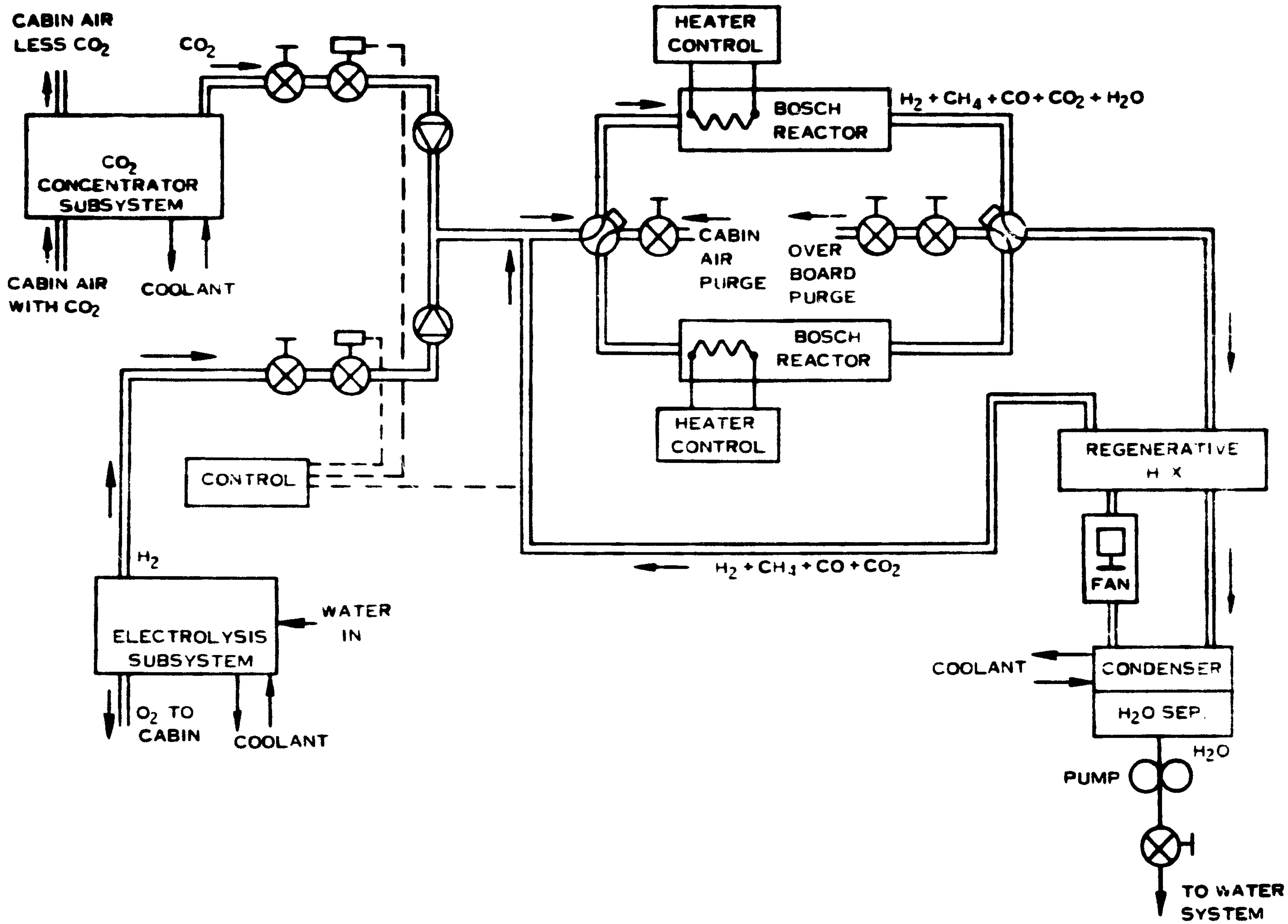


Figure A-12. Bosch CO_2 Reduction Concept

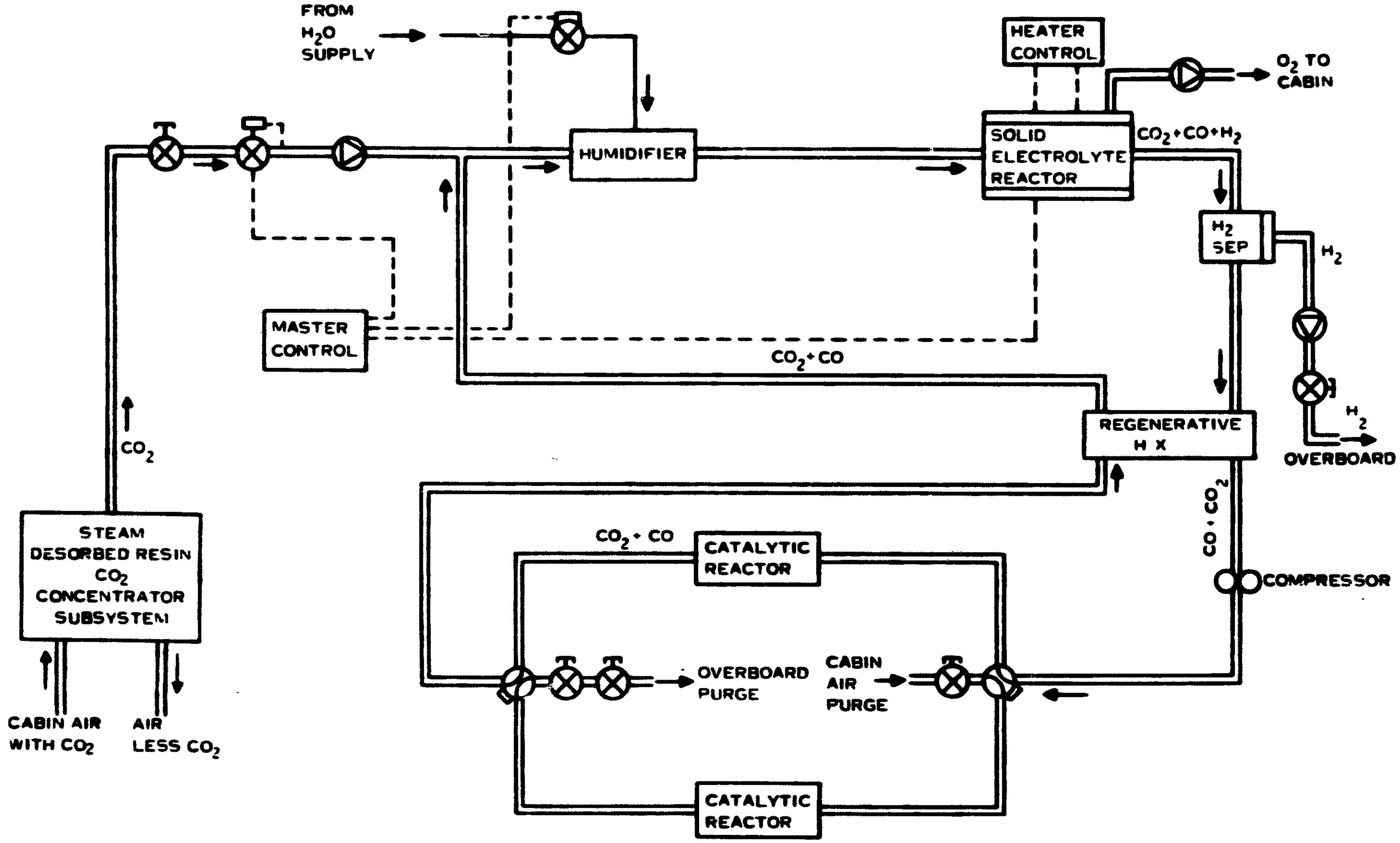


Figure A-13. Solid Electrolyte Concept

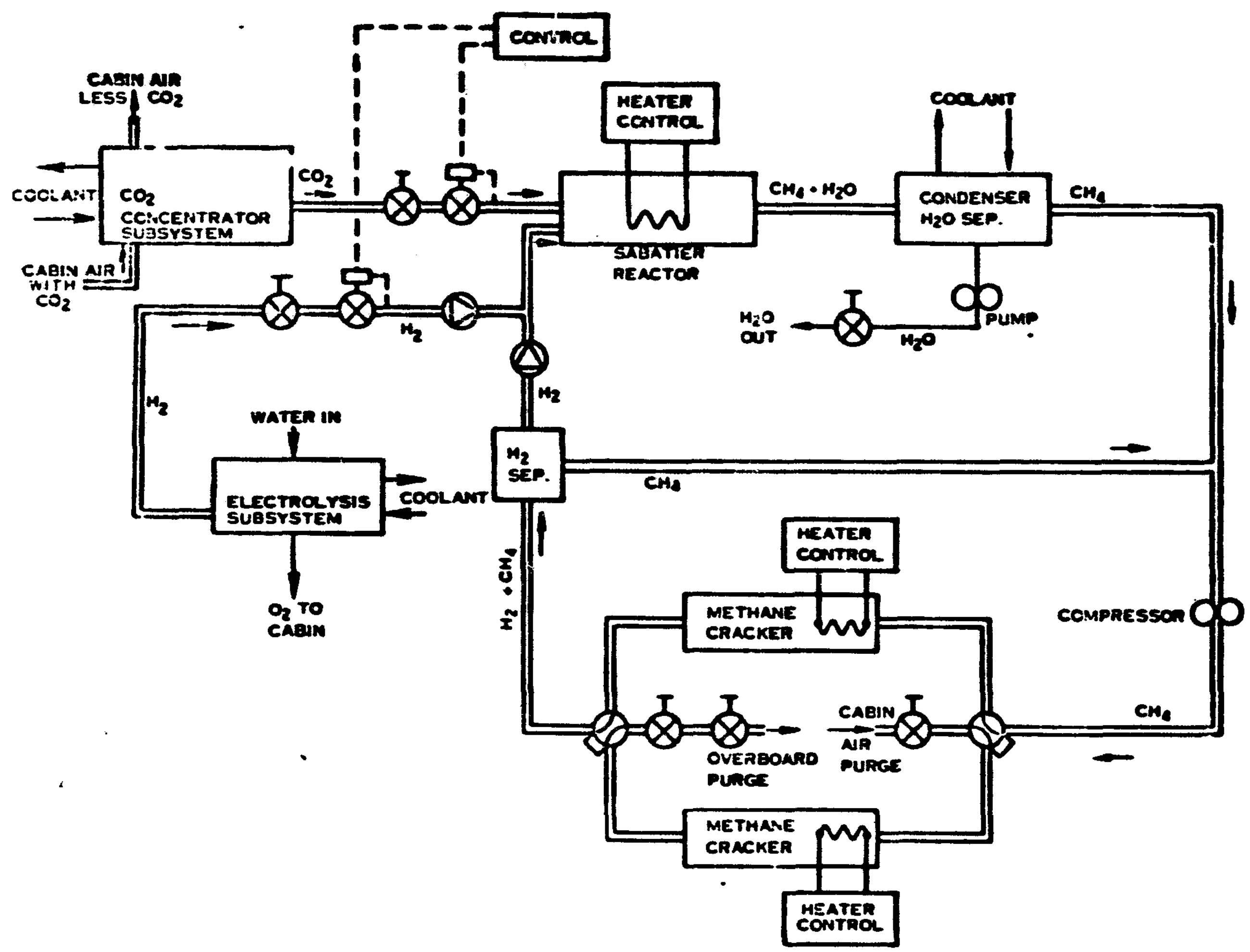


Figure A-14. Sabatier CO_2 Reduction With Methane Cracking Concept

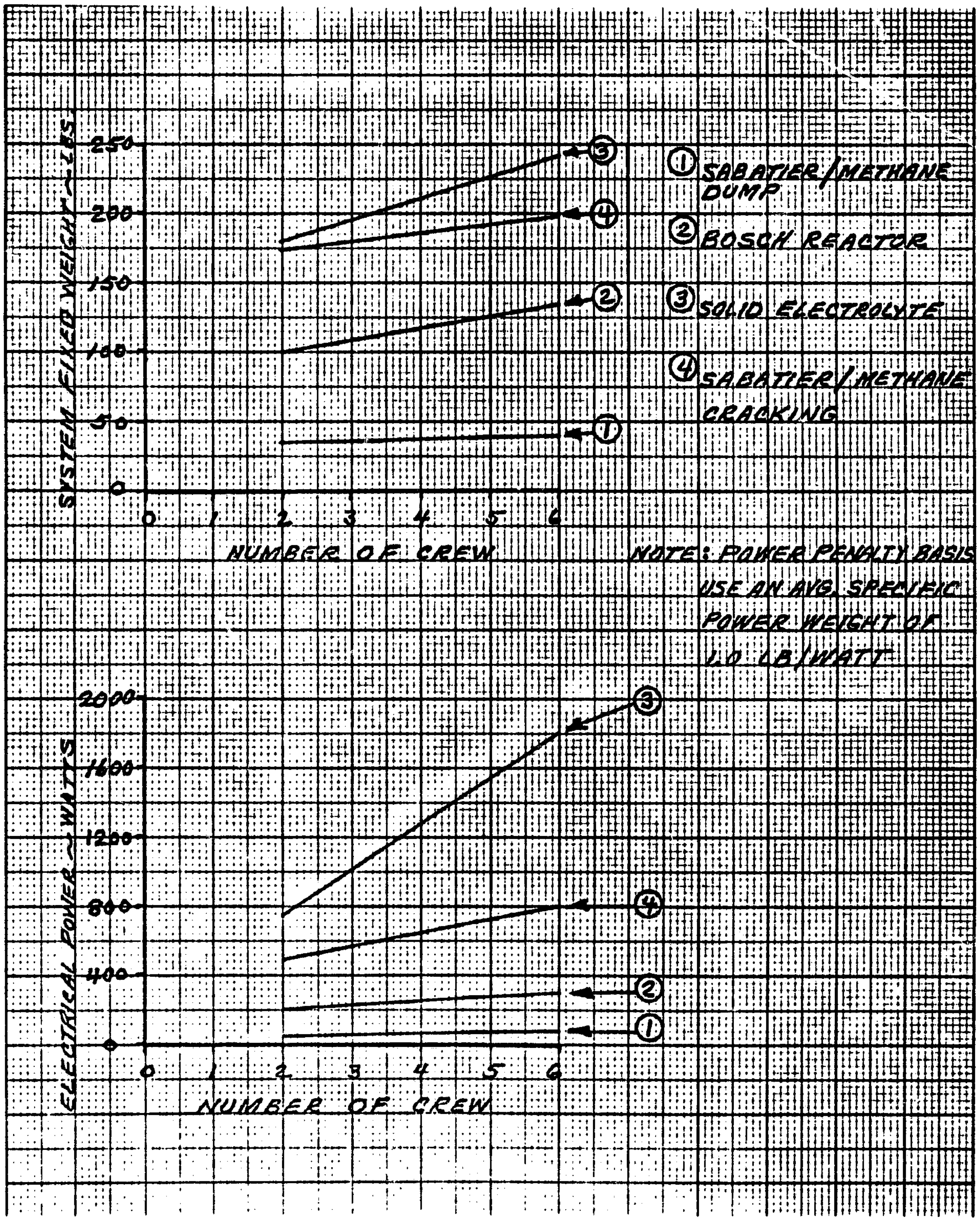


Figure A-15. CO_2 Reduction System Weight Trade Study System Fixed Weight and Electrical Power Penalty

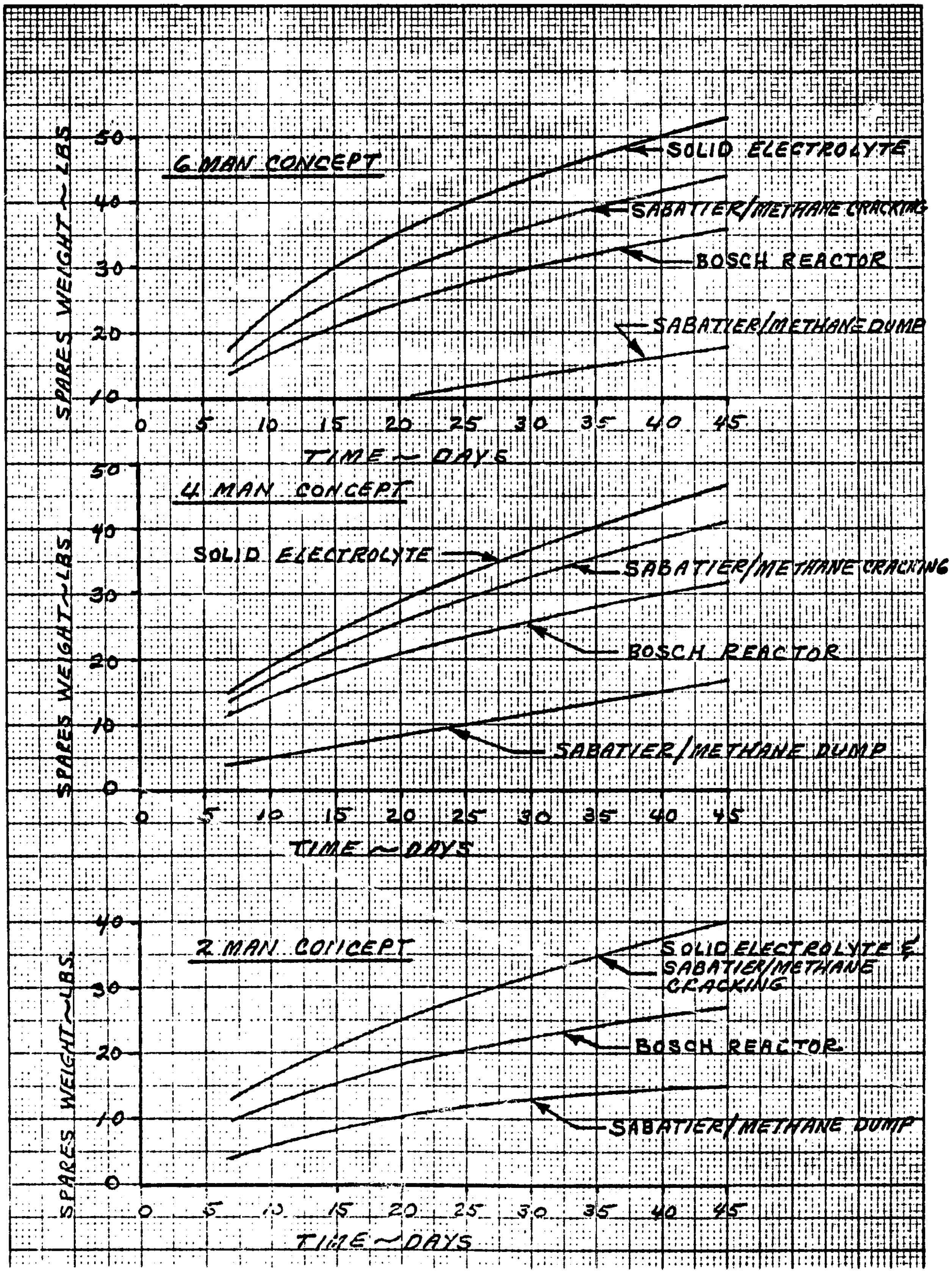
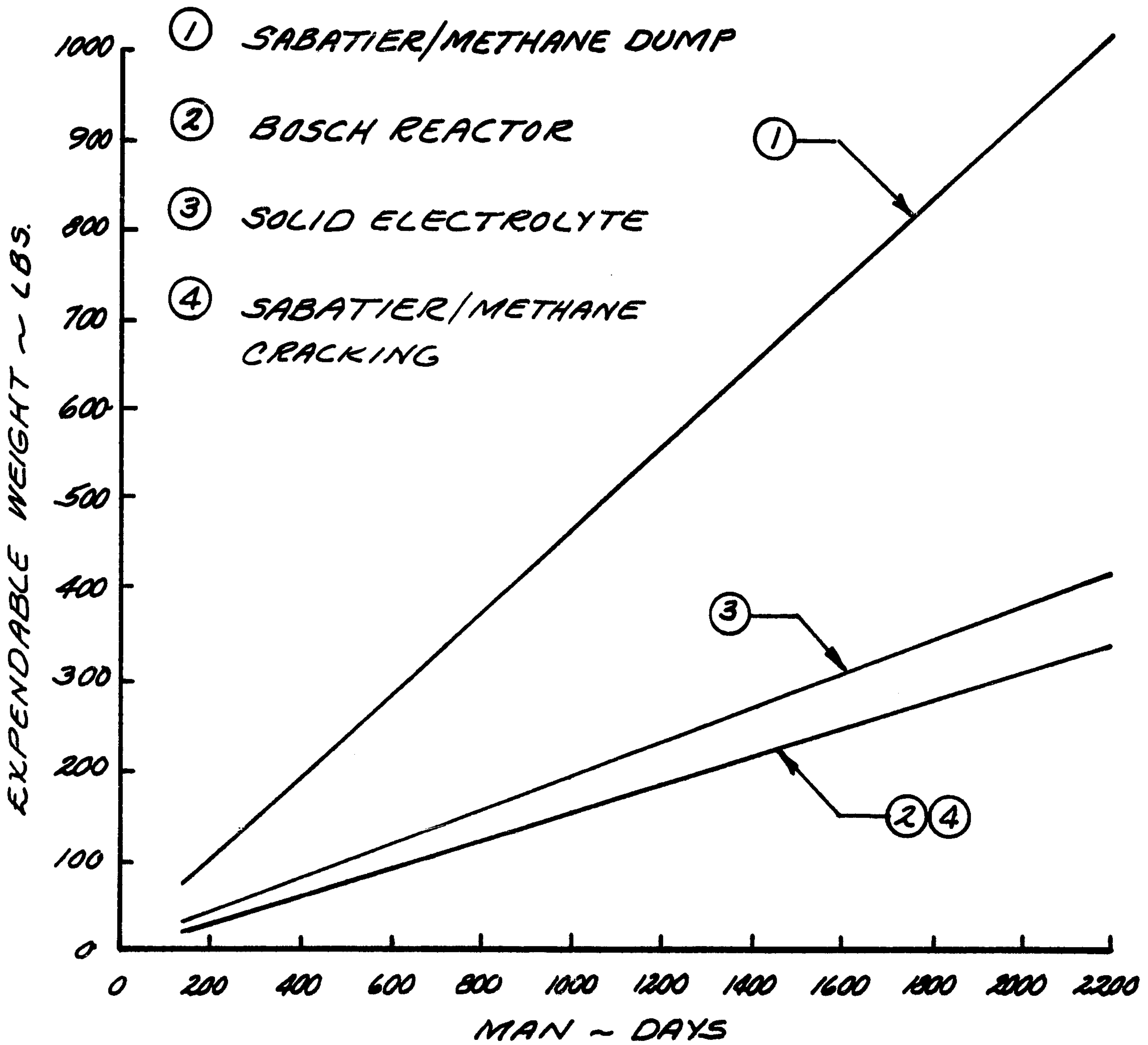


Figure A-16. CO_2 Reduction Spares Weight Trade Study



NOTE: EXPENDABLES INCLUDE SUCH ITEMS AS FILTERS, CATALYSTS OR CHEMICALS NORMALLY PROGRAMMED FOR REPLACEMENT.

Figure A-17. Expendable System Weight Trade Study

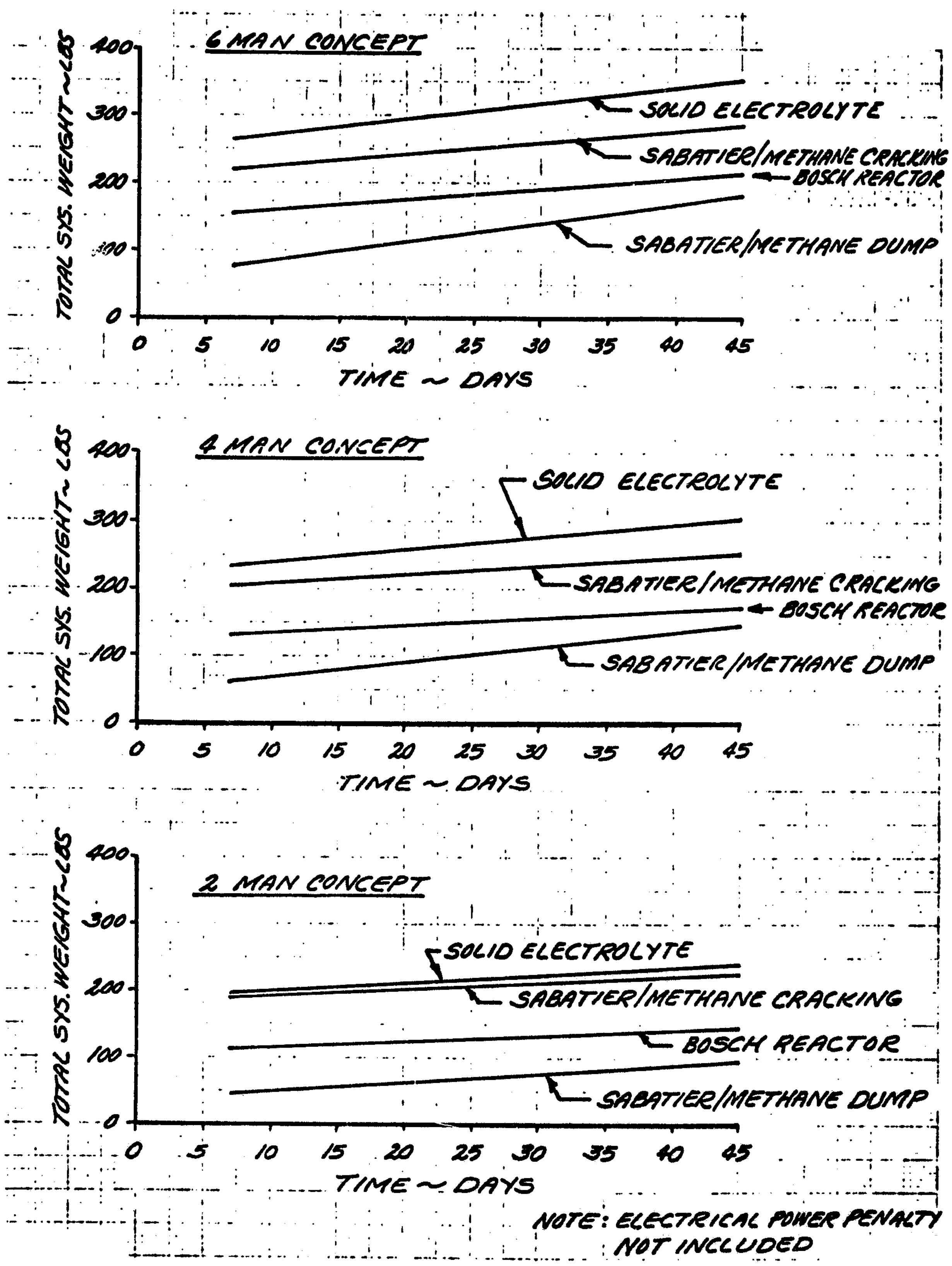


Figure A-18. CO_2 Reduction Total System Weight Trade Study requirement. For longer mission durations the slope of the curves indicate that the Bosch reactor concept would become optimum. The solid electrolyte concept is the least desirable due to its large system weight penalty plus its high electrical power demand.

Table A-10 summarizes the weight, power, and volume of the candidates selected for water recovery by CO_2 reduction. Results indicate that the Sabatier-methane dump concept is the optimum on a weight, power, and volume basis. Even when comparing this regenerative concept with a stored water system (based upon 9 pounds of water per man day) the regenerative concept shows the minimum weight penalty.

For tug applications, however, electrical power will be generated by fuel cell operation. Since water is a by-product and the water generation rate exceeds the crew consumption rate, there is no need for an additional water generation source.

H_2O Electrolysis

Methods of generating oxygen by regenerative means were investigated for the tug program. Five concepts of water electrolysis were reviewed and are schematically shown (Figures A-19 through A-23): (1) wick feed, (2) circulating electrolyte, (3) cabin air, (4) gas circulation, and (5) solid polymer.

Similar weight tradeoff curves showing system fixed weight, spare weight, electrical power penalty and total system weight penalty were plotted (Figures A-24, and A-26).

Table A-11 summarizes the candidate weight, power, and volume penalties for the three tug mission requirements. Results indicate that the gas circulation and solid polymer concepts are comparable and represent the lowest system weight penalty to the vehicle. However, the overall weight penalty based on the electrical power demand results in a penalty of 2200 to 3200 pounds. Generating oxygen by electrolysis for the tug program is costly when compared with a stored oxygen system. Based upon the normal and emergency oxygen requirements, the total weight of oxygen required for the most critical mission will not exceed 700 pounds. Therefore, for short mission requirements similar to tug, a non-regenerative (open) type of stored system is preferable.

Contaminant Control

Contaminants are those chemical compounds found in trace amounts in air or water which might be harmful to man if present in high concentrations. They are likely to build up during long-term manned space missions in which

Table A-10. CO_2 Reduction

MISSION	CANDIDATES	WEIGHT PENALTY ~ LBS.				ELEC. PWR.		VOLUME ~ FT^3		
		Fixed	Spares	Expend	Total	Watts	Cycle	Fixed	Spares	Total
2 Men 7 Days	*① Sabatier/Methane Dump	35	4	7	46	50	Cont.	3.1	0.53	3.63
	② Bosch Reactor	100	10	5	115	200	Cont.	8.9	0.66	9.56
	③ Solid Electrolyte	180	13	6	199	750	Cont.	16.0	0.84	16.84
	④ Sabatier/Methane Cracking	175	12	5	192	500	Cont.	15.5	0.76	16.26
6 Men 7 Days	*① Sabatier/Methane Dump	40	5	35	80	75	Cont.	3.54	1.78	5.32
	② Bosch Reactor	135	14	9	158	300	Cont.	11.95	1.02	12.97
	③ Solid Electrolyte	242	17	10	269	1800	Cont.	21.40	1.20	22.60
	④ Sabatier/Methane Cracking	200	14	9	223	800	Cont.	17.70	1.02	18.72
4 Men 45 Days	*① Sabatier/Methane Dump	38	17	90	145	60	Cont.	3.35	4.75	8.10
	② Bosch Reactor	118	32	28	178	240	Cont.	10.41	2.66	13.07
	③ Solid Electrolyte	215	47	40	302	1300	Cont.	19.10	3.86	22.96
	④ Sabatier/Methane Cracking	186	41	28	255	650	Cont.	16.50	3.06	19.56

NOTE: Expendables include such items as filters, catalysts or chemicals normally programmed for replacement.

* Selected system candidate based on minimum weight and power penalty.

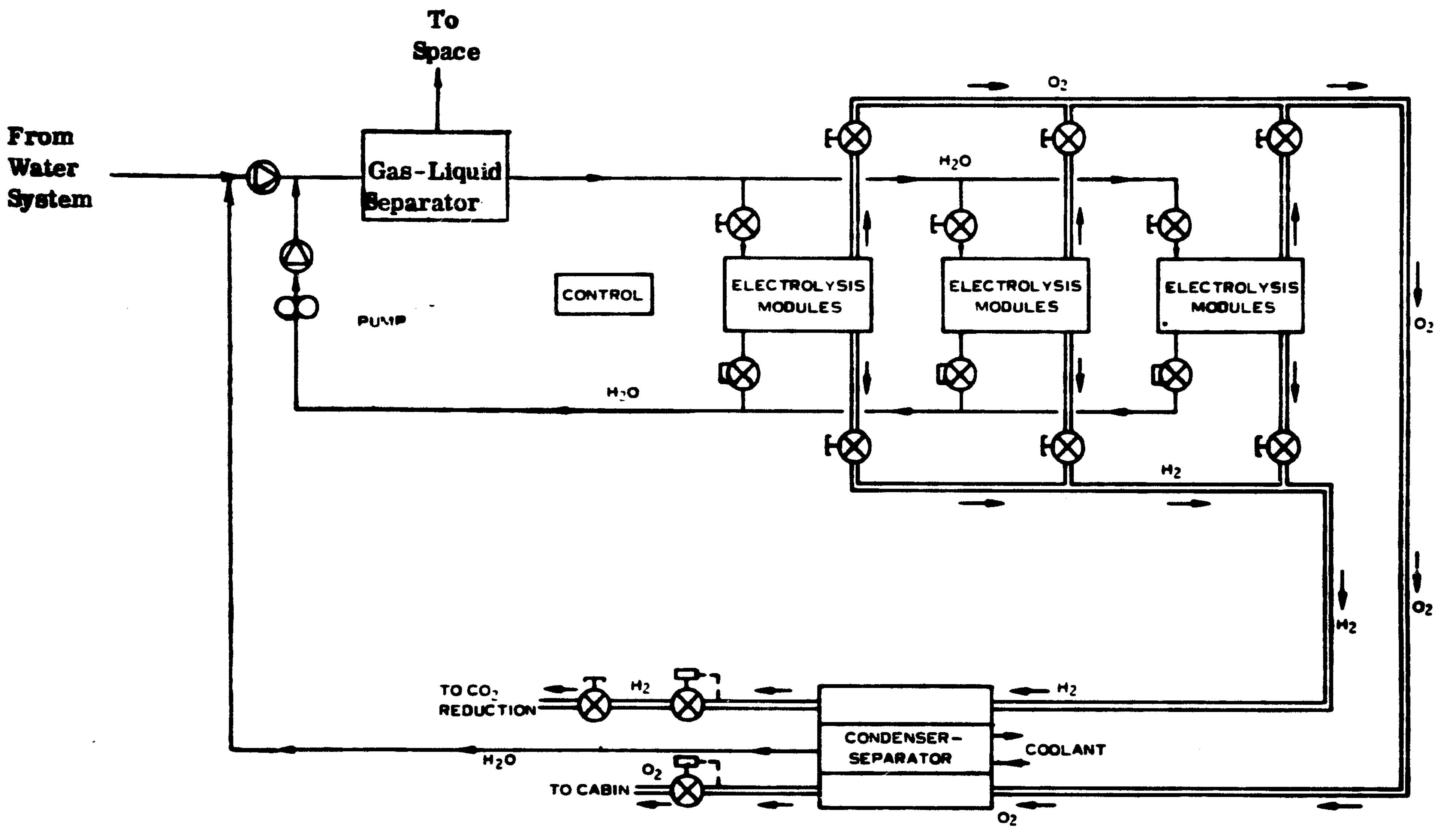


Figure A-19. Wick Feed Concept

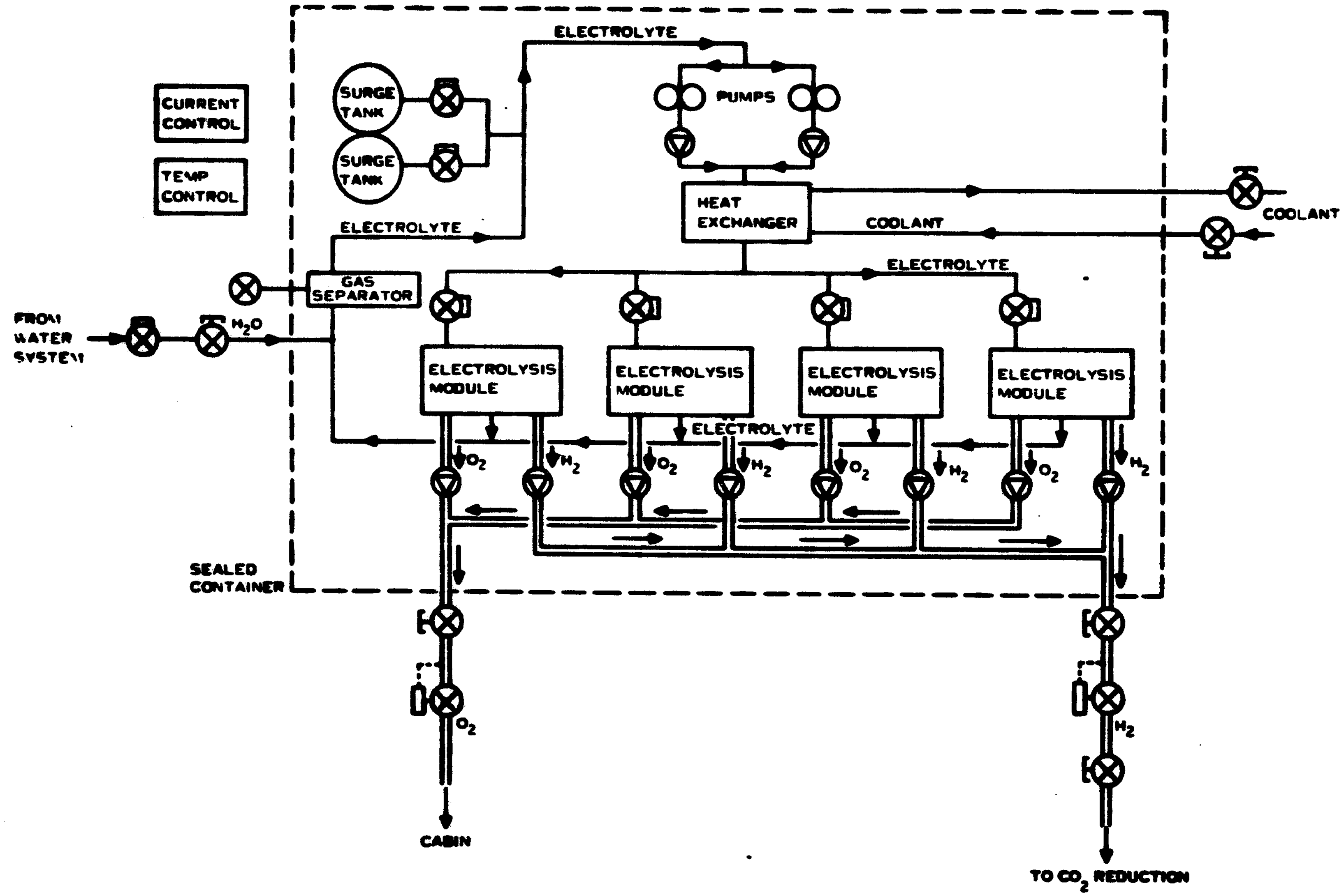


Figure A-20. Circulating Electrolyte Concept

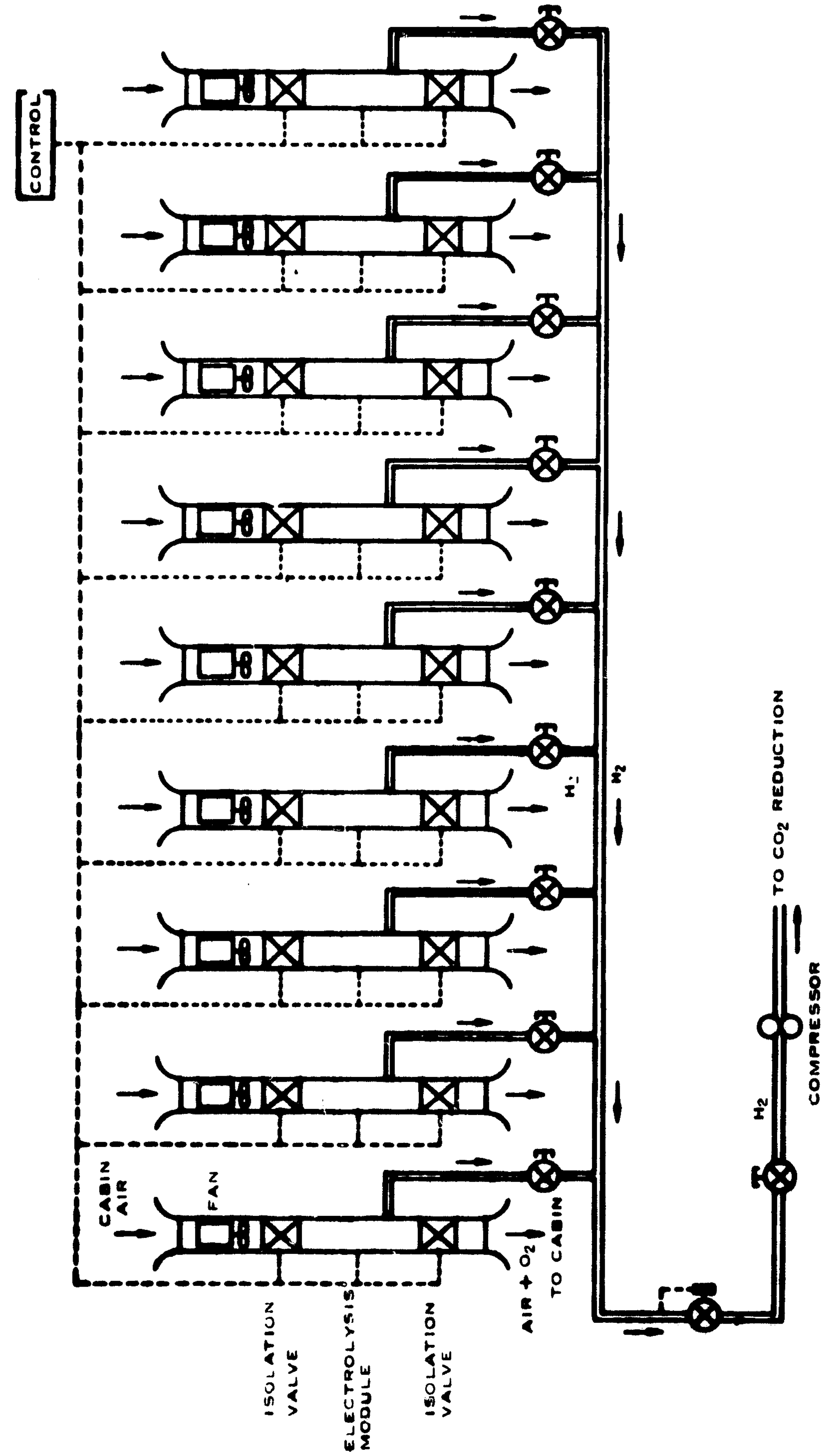


Figure A-21. Cabin Air Concept

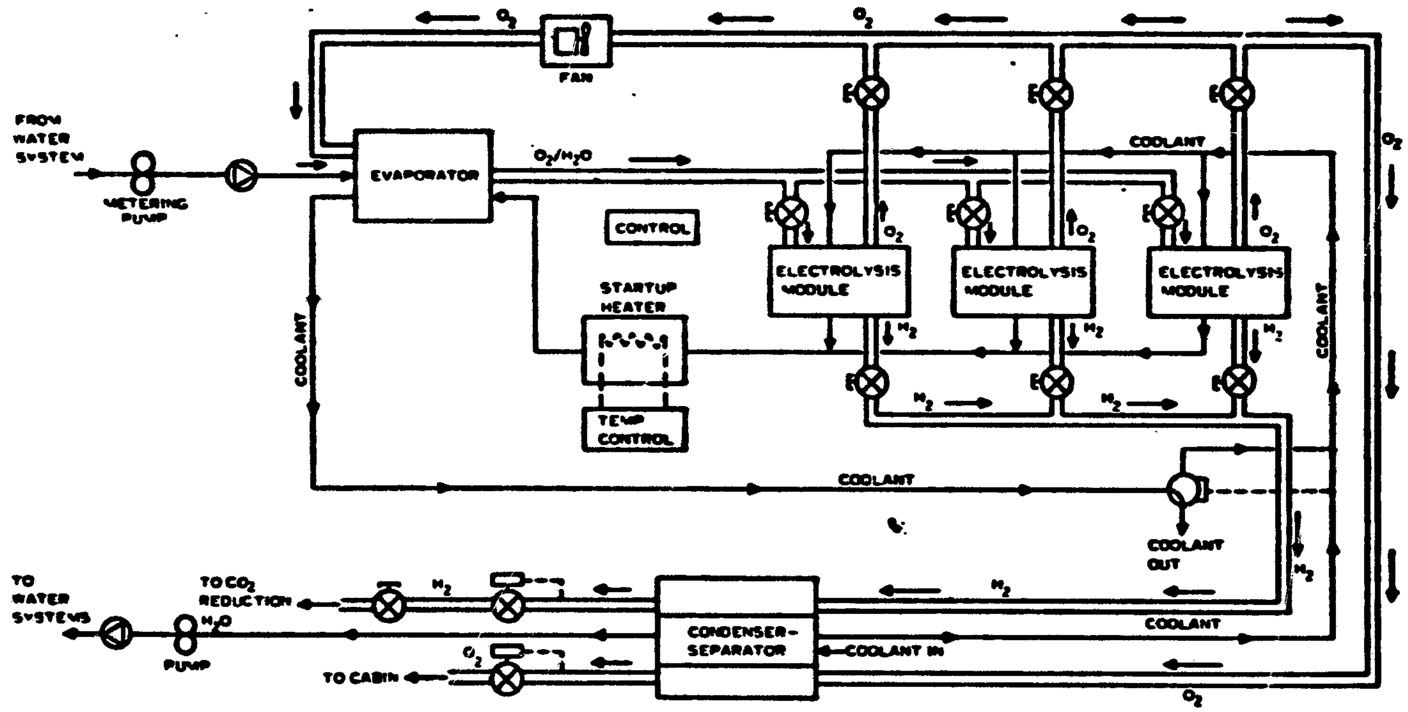


Figure A-22. Gas Circulating Concept

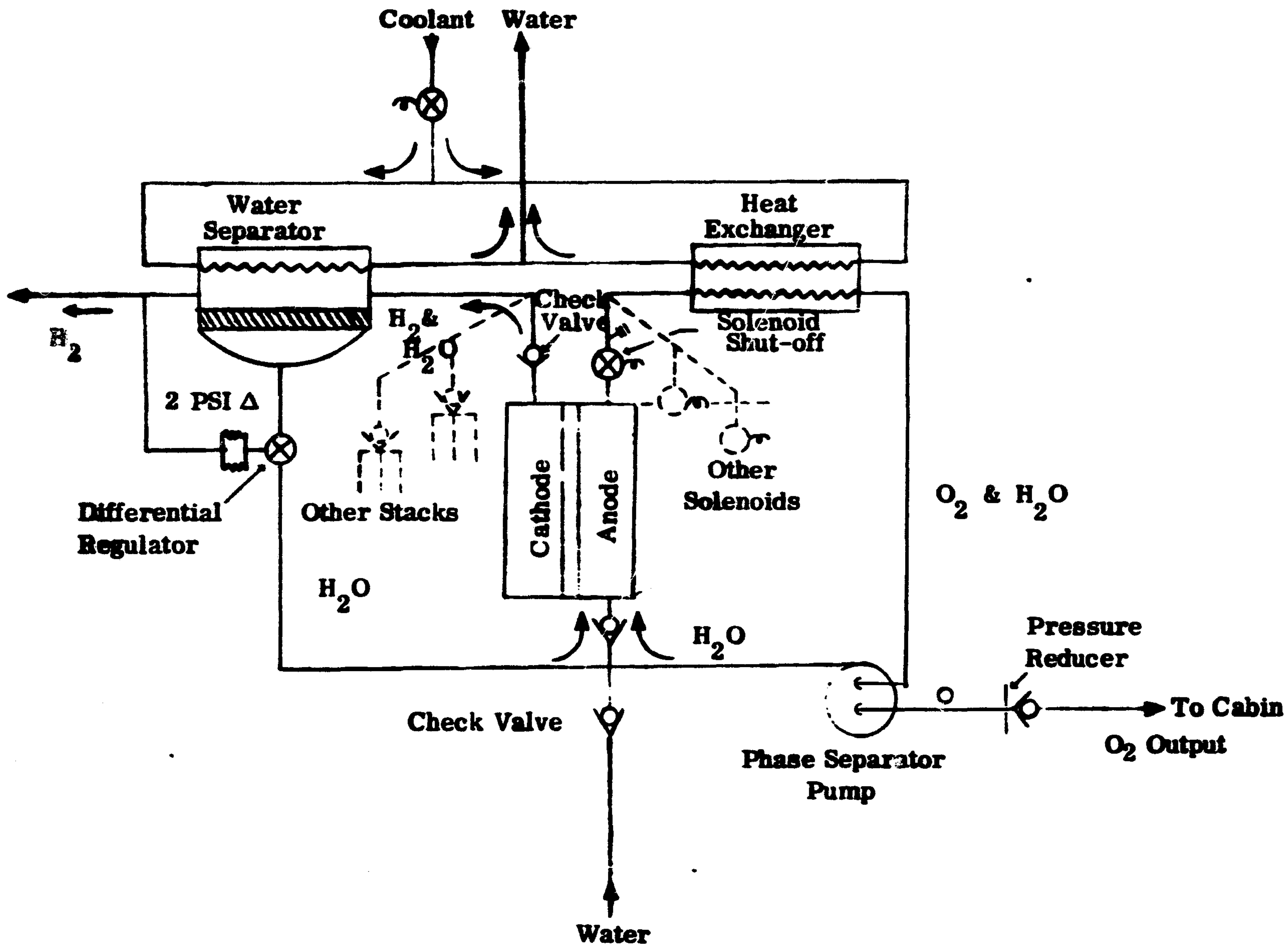


Figure A-23. Water Electrolysis Solid Polymer Electrolyte

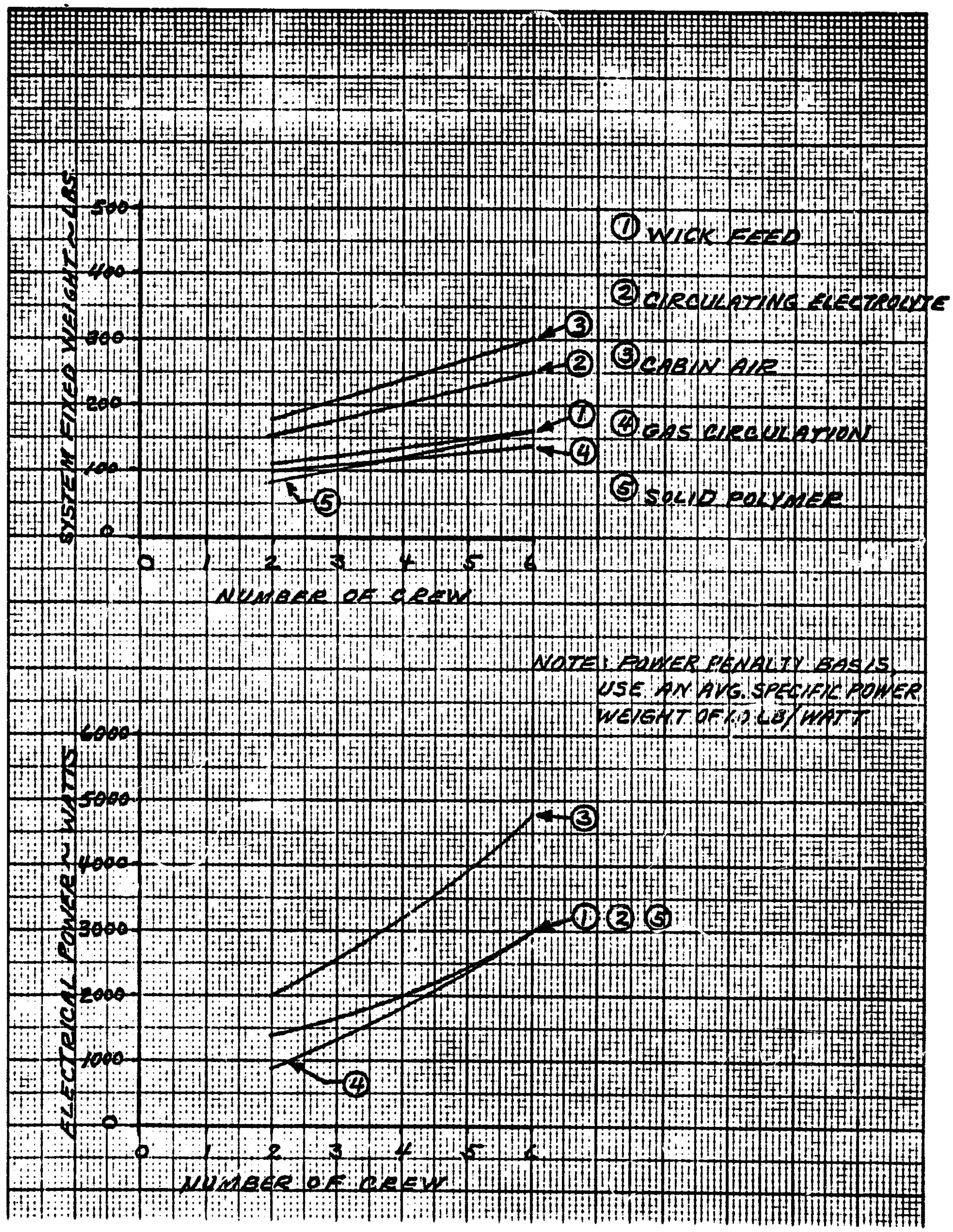


Figure A-24. Water Electrolysis System Weight Trade Study System Fixed Weight and Electrical Power Penalty

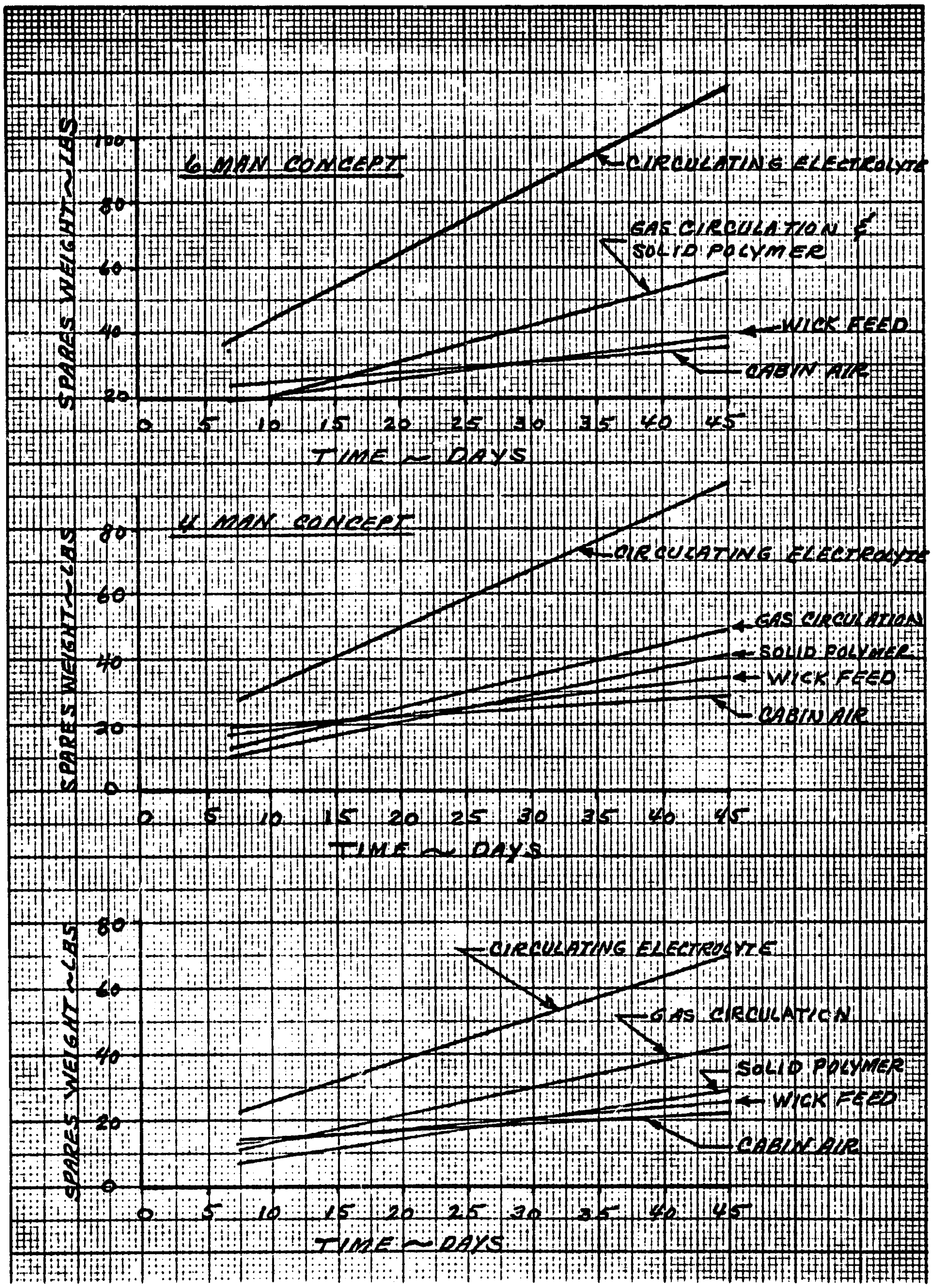


Figure A-25. Water Electrolysis Spares Weight Trade Study

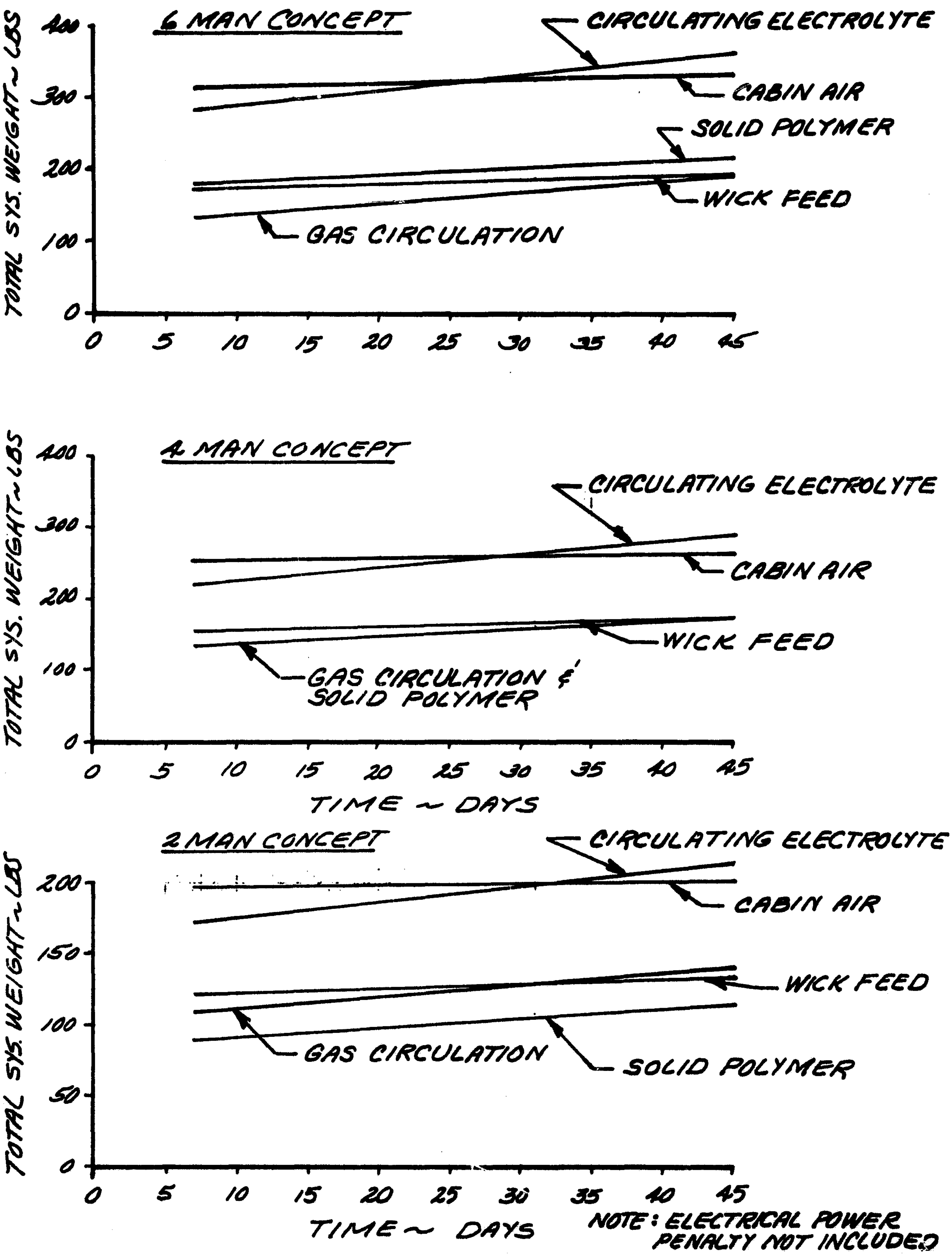


Figure A-26. Water Electrolysis Total System Weight Trade Study

Table A-11. H_2O Electrolysis

MISSION	CANDIDATES	WEIGHT PENALTY ~ LBS.				ELEC. PWR.		VOLUME ~ FT^3		
		Fixed	Spares	Expend	Total	Watts	Cycle	Fixed	Spares	Total
2 Men 7 Days	① Wick Feed	110	13	--	123	1400	Cont.	4.08	0.48	4.56
	② Circulating Electrolyte	150	21	--	171	1400	Cont.	5.58	0.78	6.36
	③ Cabin Air	180	14	--	194	2000	Cont.	6.70	0.52	7.22
*	④ Gas Circulation	100	11	--	111	900	Cont.	3.72	0.41	4.13
*	⑤ Solid Polymer	85	7	--	92	1400	Cont.	3.16	0.26	3.42
6 Men 7 Days	① Wick Feed	160	19	--	179	3000	Cont.	5.94	0.72	6.66
	② Circulating Electrolyte	250	35	--	285	3000	Cont.	9.30	1.32	10.62
	③ Cabin Air	300	24	--	324	4800	Cont.	11.15	0.91	12.06
*	④ Gas Circulation	140	15	--	155	3000	Cont.	5.20	0.56	5.76
*	⑤ Solid Polymer	160	13	--	173	3000	Cont.	5.94	0.49	6.43
4 Men 45 Days	① Wick Feed	140	34	--	174	2000	Cont.	5.21	1.28	6.49
	② Circulating Electrolyte	200	92	--	292	2000	Cont.	7.44	3.47	10.91
	③ Cabin Air	240	29	--	269	3200	Cont.	8.92	1.09	10.01
*	④ Gas Circulation	120	48	--	168	1800	Cont.	4.46	1.81	6.27
*	⑤ Solid Polymer	120	41	--	161	2000	Cont.	4.46	1.55	6.01

NOTE: Expendables include such items as filters, catalysts or chemicals normally programmed for replacement.

* Selected system candidate based on minimum weight and power penalty.

the atmosphere is being regenerated. There are various sources of these contaminants: the metabolic processes of the crew that result in saliva, urine, feces, flatus, and expired air; gassing products from food and supplies stored and used aboard the spacecraft; and gassing products resulting from the operation of the various systems within the spacecraft. Other sources of contaminants are the materials from which the spacecraft is made and any reaction products of individual gassing components. Table A-7 presents a complete list of the possible trace contaminants that may be present in the tug command module. Three types of contaminant control devices will be used on Tug: (1) fiberglass particulate filters to remove the particulate matter and aerosols from the cabin air; (2) activated charcoal filters located downstream of the particulate filters to remove the high molecular weight organics; and (3) a regenerative type of contaminant control which oxidizes many of the potential contaminants into water and carbon dioxide.

For tug applications, three methods of contaminant control by regenerative means were investigated: (1) non-regenerable charcoal/catalytic oxidation (Figure A-27); (2) catalytic oxidation/sorption (Figure A-28); and (3) regenerable charcoal/catalytic oxidation (Figure A-29).

Tradeoff curves show the weight and power penalties for the three candidates: Figure A-30 shows the system fixed weight and electrical power penalties as a function of crew size; Figure A-31 shows the system spares weight versus mission duration; and Figure A-32 shows the system expendable weight versus man days. The total system weight tradeoff is presented in Figure A-33.

Table A-12 is a summary of the tradeoff study and indicates that the catalytic oxidation sorption concept is the best choice from a weight penalty basis and is comparable with the non-regenerable charcoal/catalytic oxidation concept in electrical power demand.

Expandable weight is also much lower for the catalytic oxidation/ sorption concept.

Water Reclamation

An area of major consideration on any manned space mission is water management. The water management system is a very important aspect of manned space travel; selecting a proper technique could result in significant weight savings for a medium- or long-term mission.

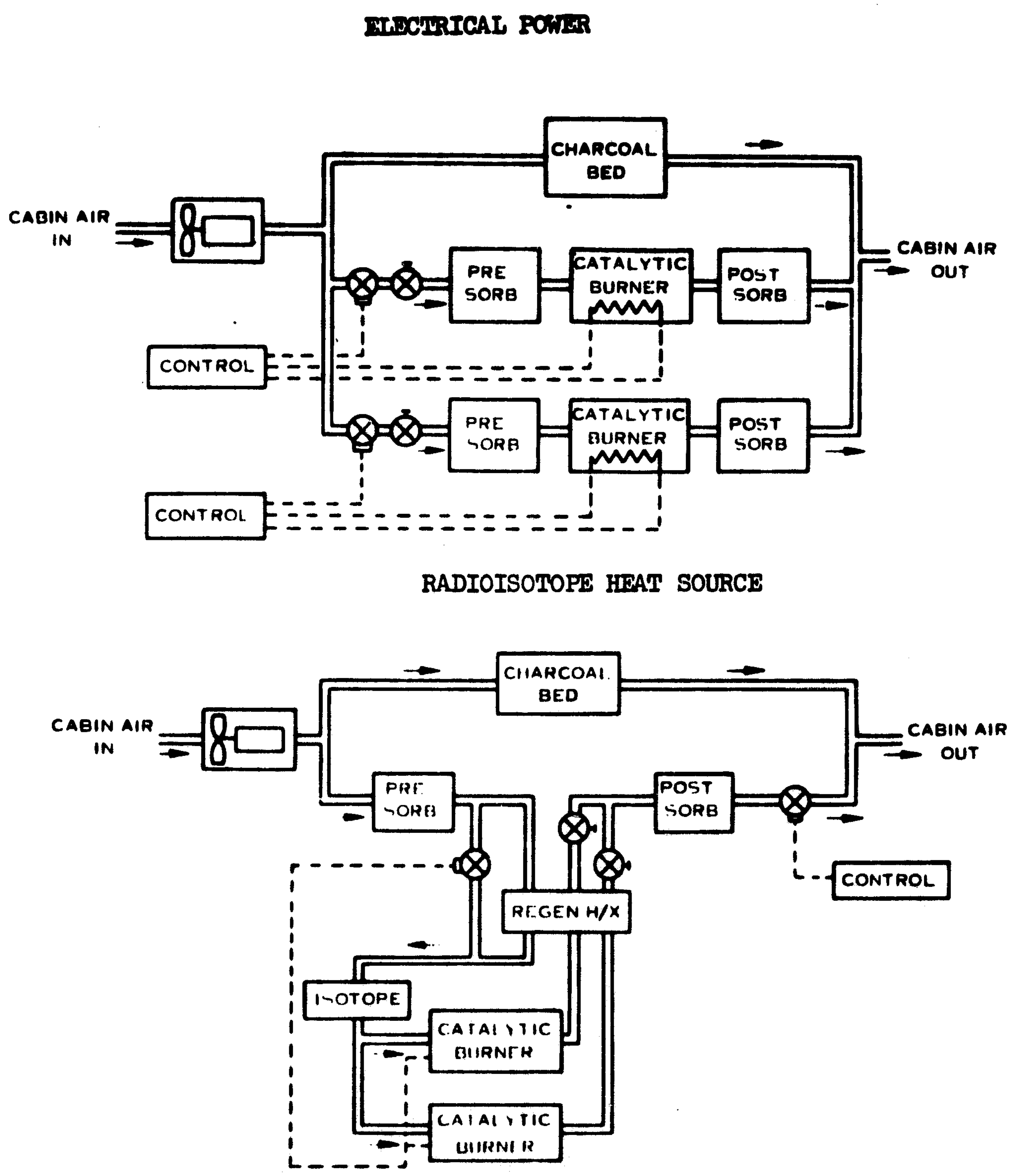
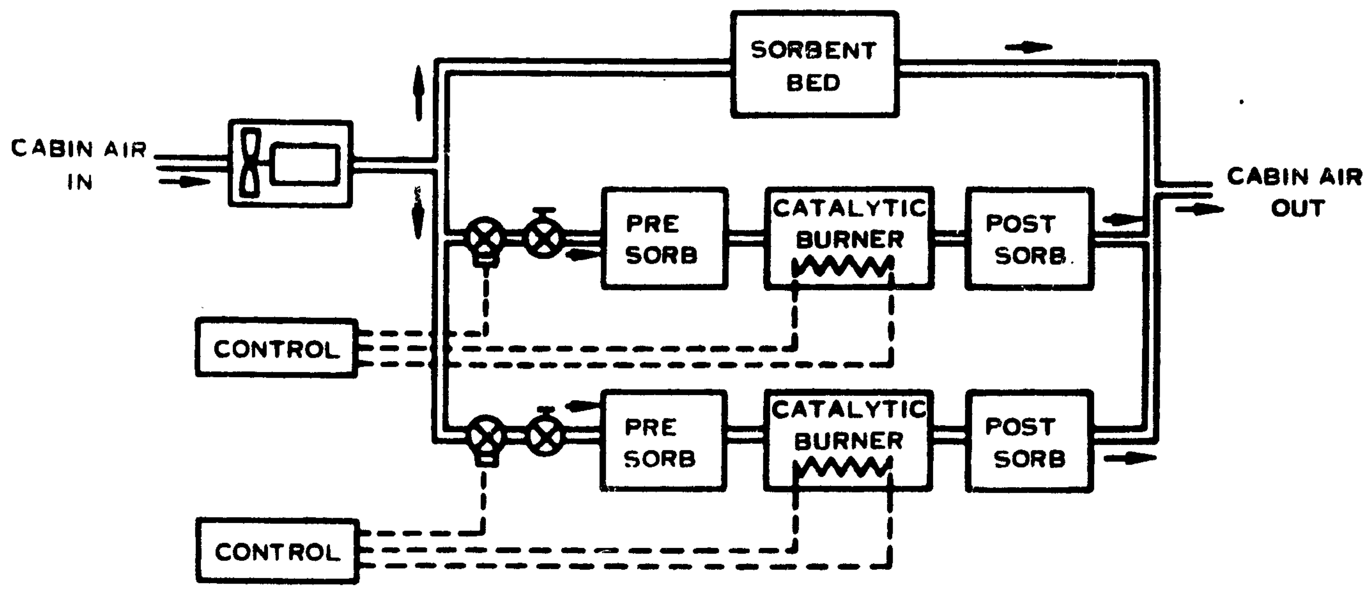


Figure A-27. Nonregenerable Charcoal/Catalytic Oxidation

ELECTRICAL POWER



RADIOISOTOPE HEAT SOURCE

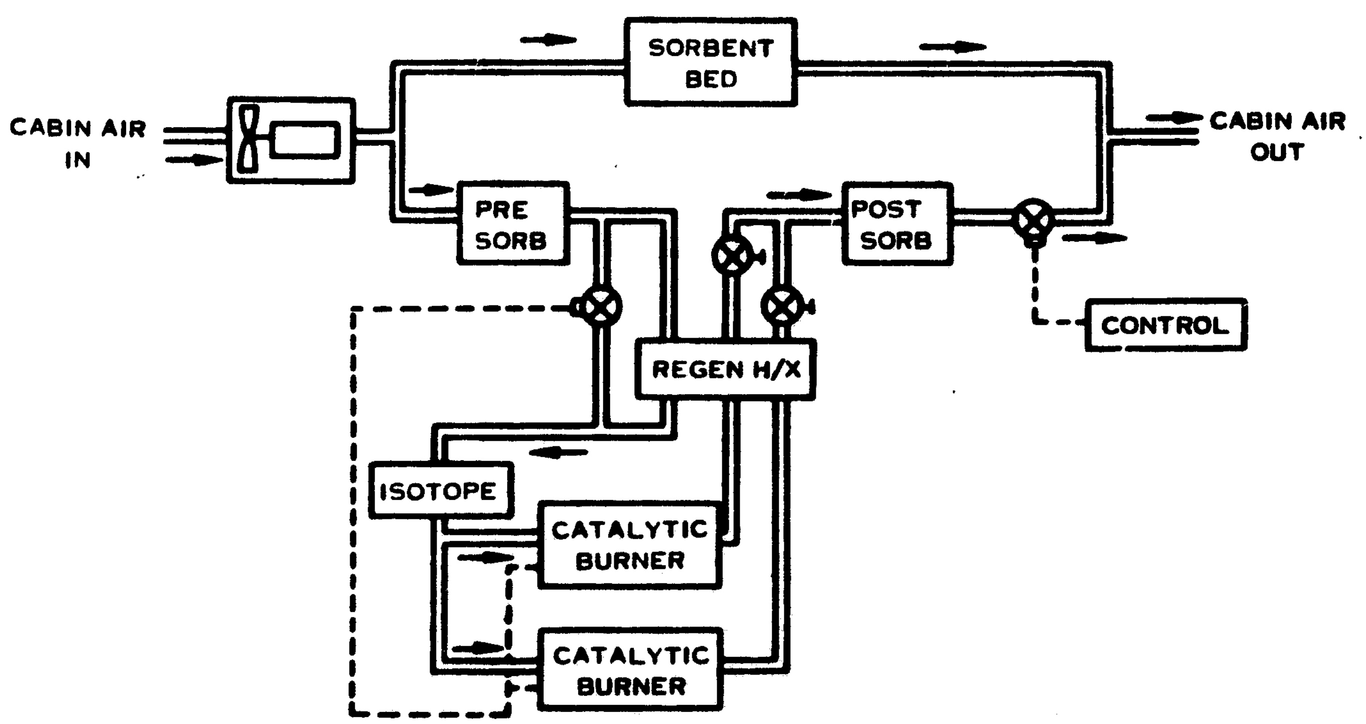


Figure A-28. Catalytic Oxidation/Sorption

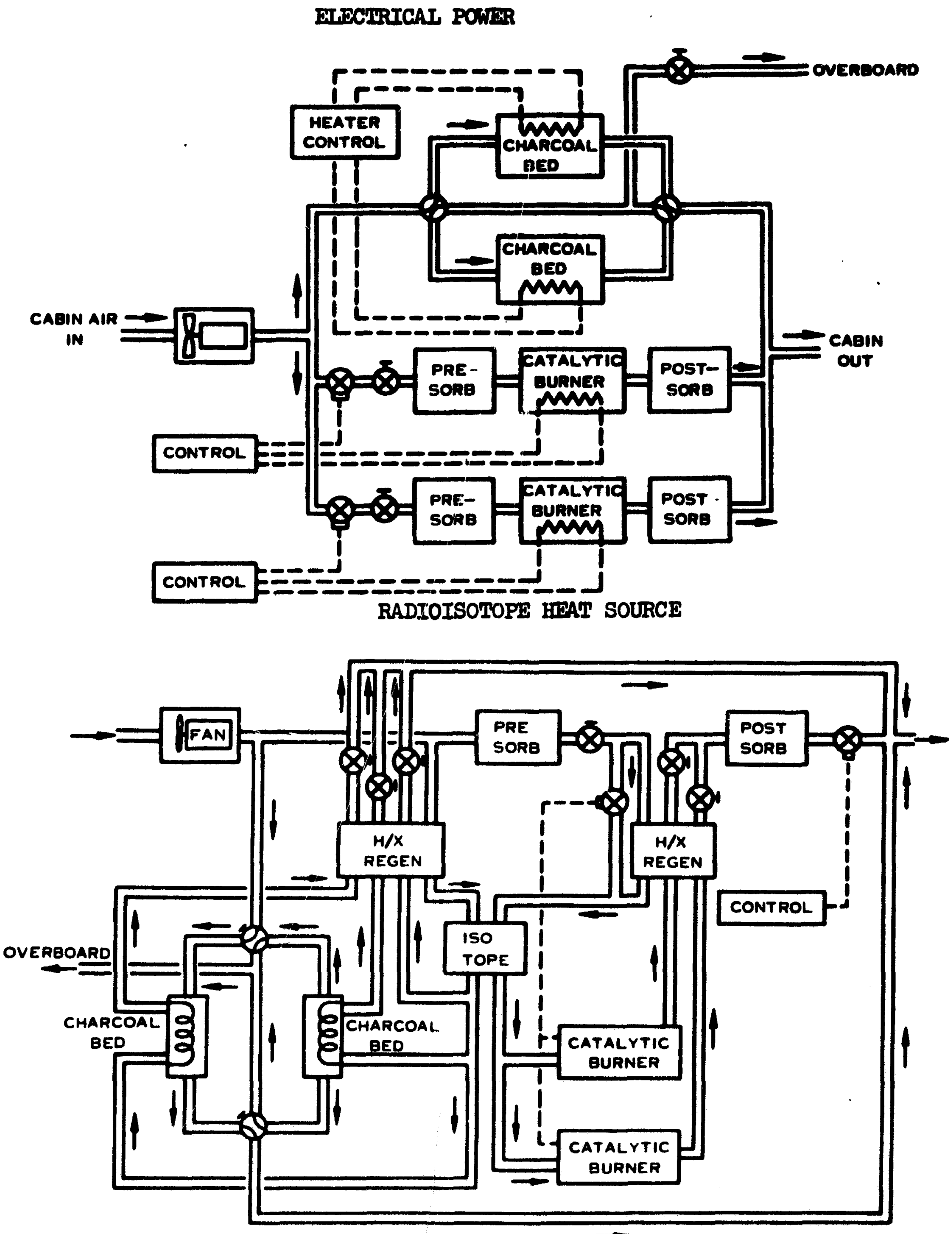


Figure A-29. Regenerable Charcoal/Catalytic Oxidation

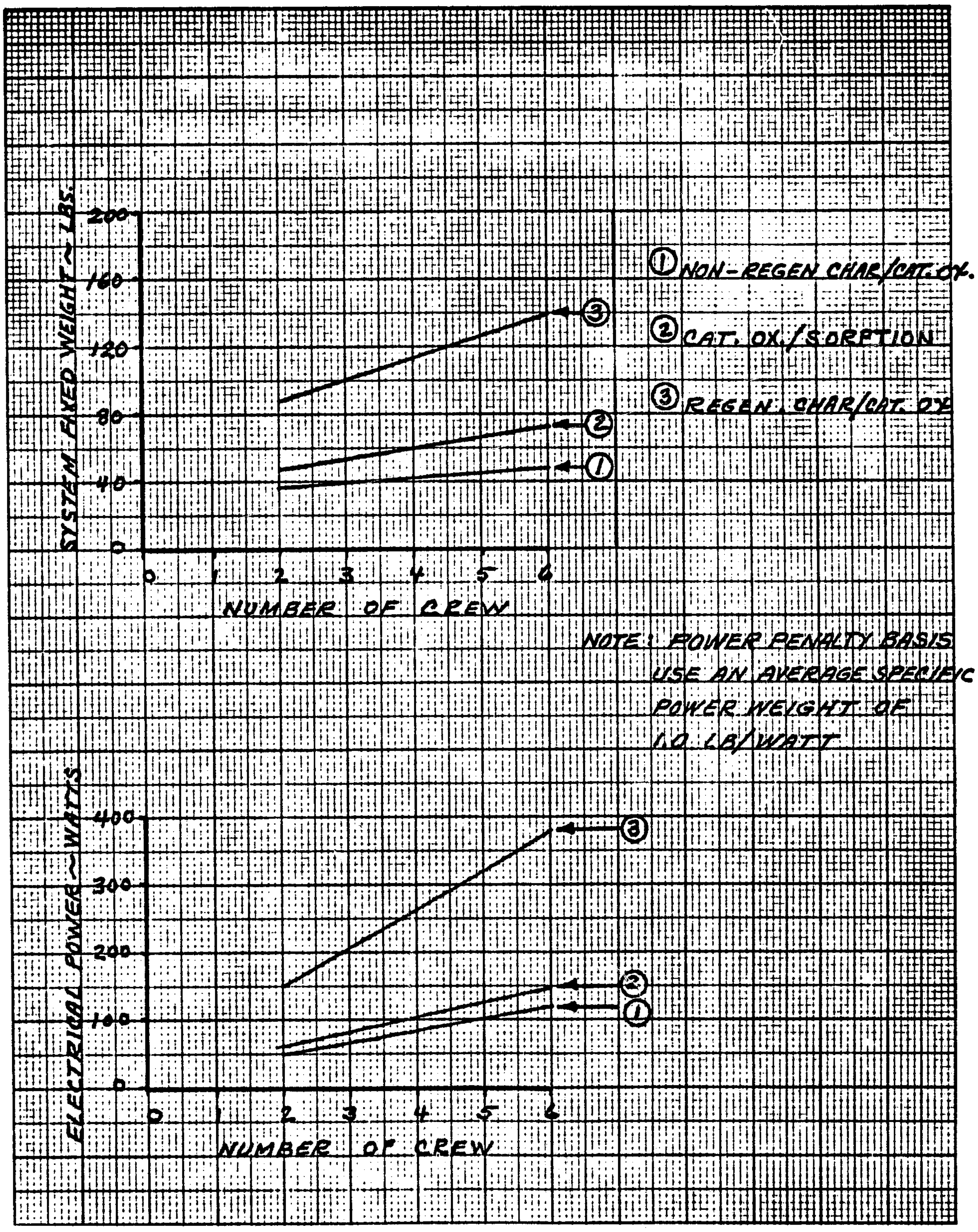


Figure A-30. Contaminant Control System Weight Trade Study System Fixed Weight and Electrical Power Penalty

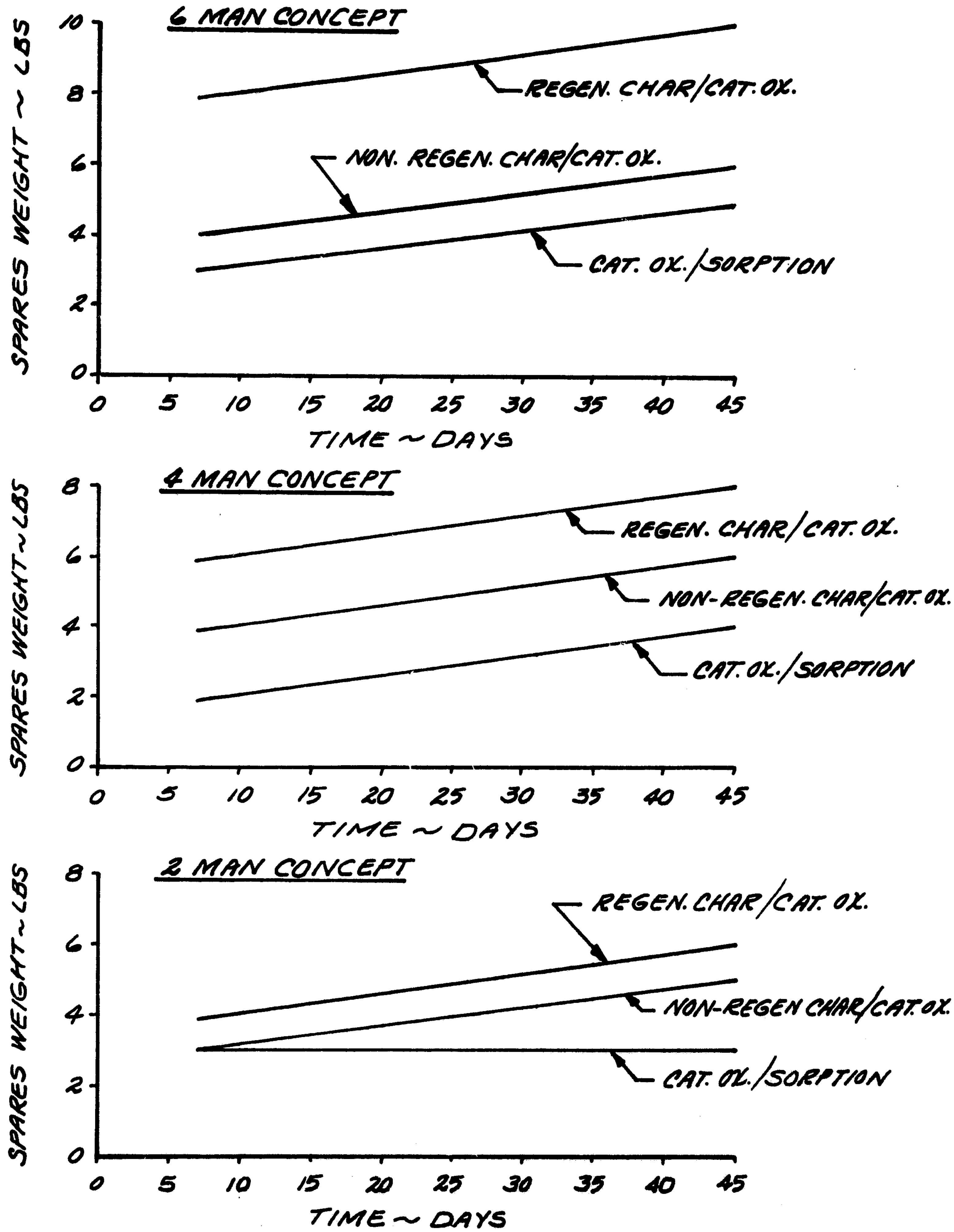
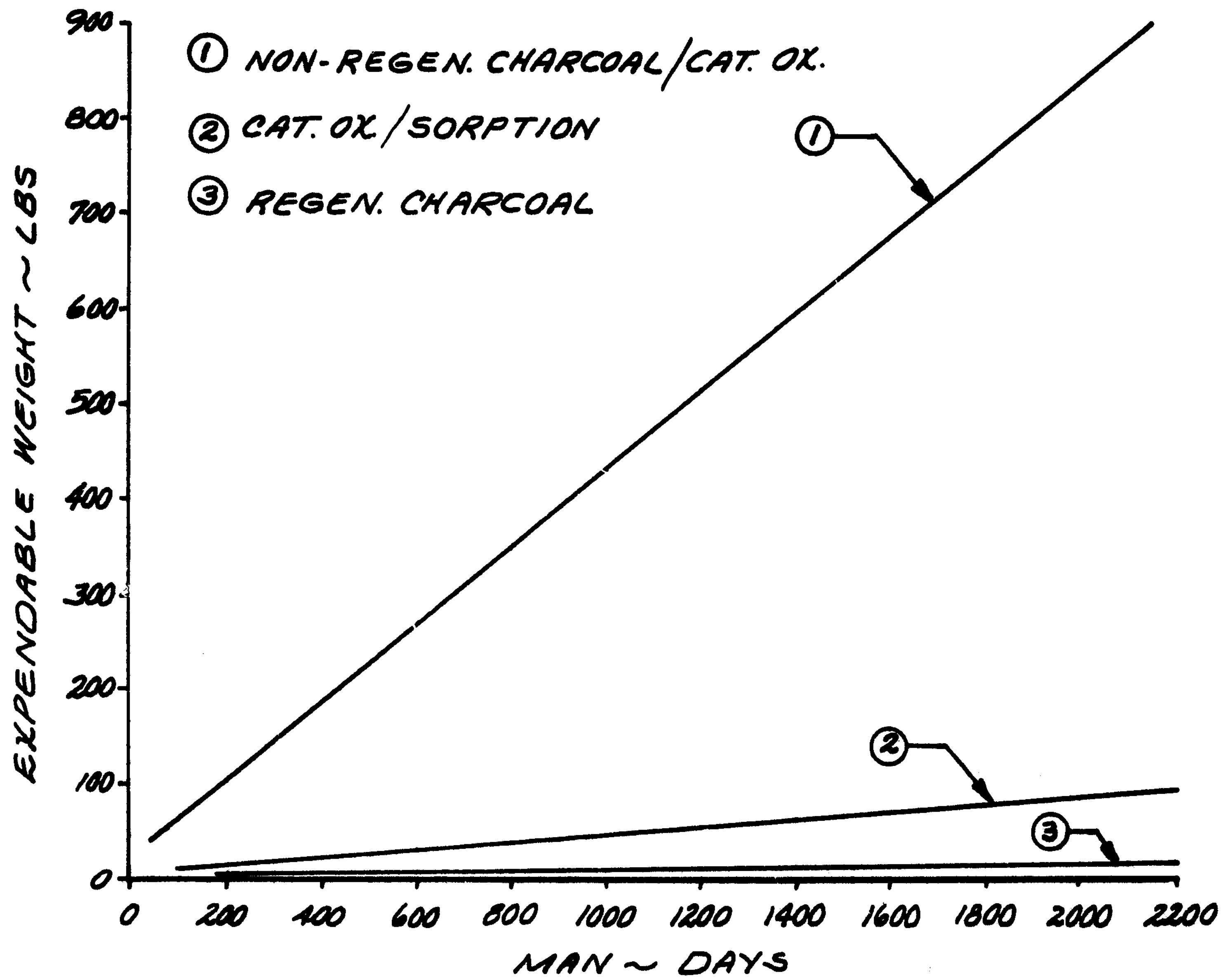


Figure A-31. Contaminant Control Spares Weight Trade Study



NOTE: EXPENDABLES INCLUDE SUCH ITEMS AS FILTERS, CATALYSTS OR CHEMICALS NORMALLY PROGAMMED FOR REPLACEMENT.

Figure A-32. Contaminant Control Expendable System Weight Trade Study

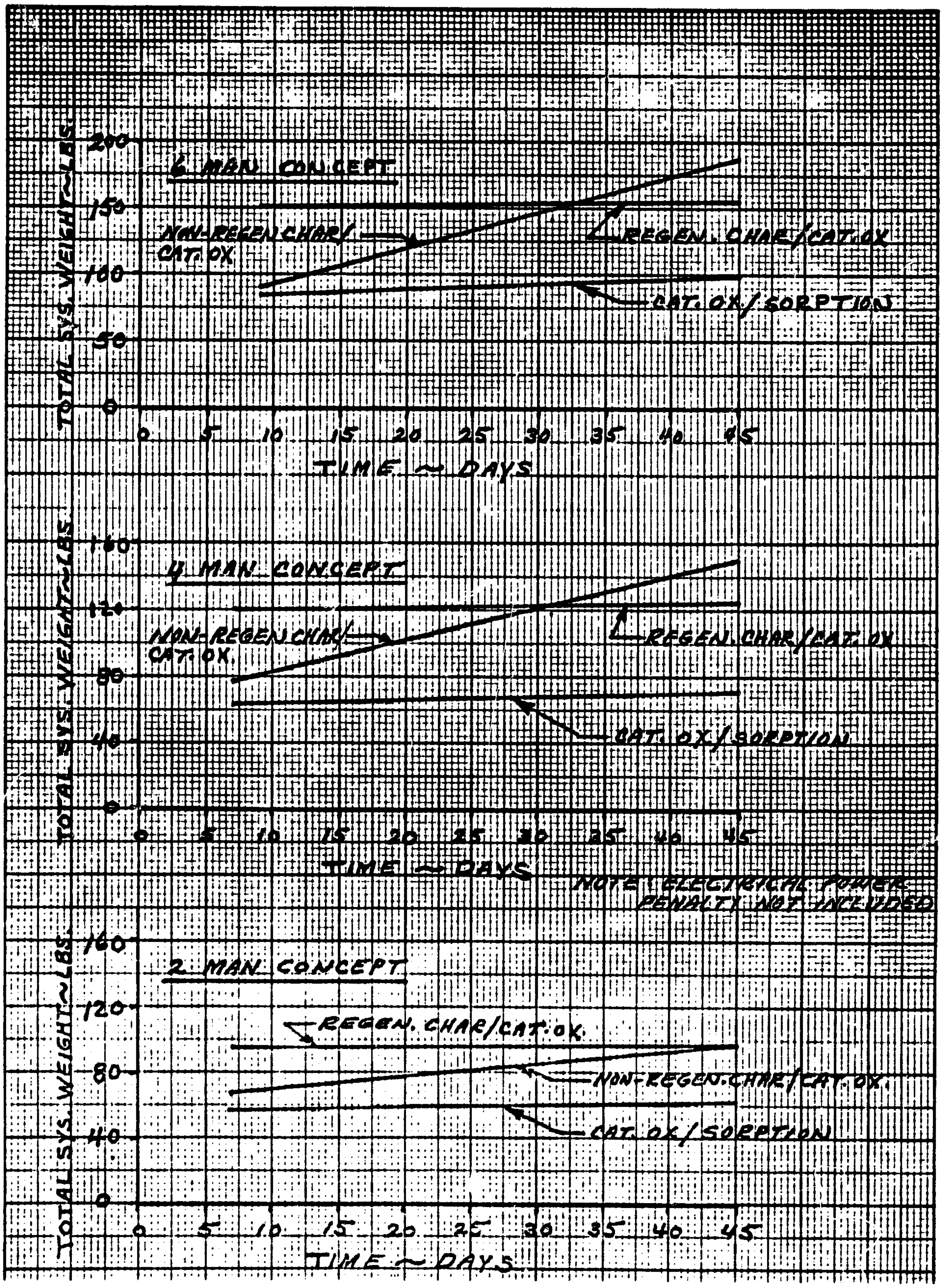


Figure A-33. Contaminant Control Total System Weight Trade Study

Table A-12. Contaminant Control

MISSION	CANDIDATES	WEIGHT PENALTY ~ LBS.				ELEC. PWR.		VOLUME ~ FT^3		
		Fixed	Spares	Expend	Total	Watts	Cycle	Fixed	Spares	Total
2 Men	① Non-Regen-Char/Cat. Ox.	36	3	25	64	50	Cont.	2.10	0.84	2.94
7 Days *	② Cat. Ox./Sorption	48	3	6	57	60	Cont.	2.70	0.27	2.97
	③ Regen-Char./Cat. Ox.	88	5	1	94	150	Cont.	4.95	0.18	5.13
6 Men	① Non-Regen-Char/Cat. Ox.	48	4	40	92	120	Cont.	2.70	1.32	4.02
7 Days *	② Cat. Ox./Sorption	74	4	8	86	145	Cont.	4.16	0.36	4.52
	③ Regen-Char/Cat. Ox.	140	8	3	151	380	Cont.	7.85	0.33	8.18
4 Men	① Non-Regen-Char/Cat. Ox.	44	6	100	150	80	Cont.	2.48	3.19	5.67
45 Days *	② Cat Ox./Sorption	52	4	15	71	100	Cont.	2.92	0.57	3.49
	③ Regen-Char/Cat. Ox.	112	8	4	124	260	Cont.	6.30	0.36	6.66

NOTE: Expendables include such items as filters, catalysts or chemicals normally programmed for replacement.

* Selected system candidate based on minimum weight and power penalty.

Table A-13. Water Reclamation

MISSION	CANDIDATES	WEIGHT PENALTY ~ LBS.				ELEC. PWR.		VOLUME ~ FT^3		
		Fixed	Spares	Expend	Total	Watts	Cycle	Fixed	Spares	Total
2 Men 7 Days	* ① Air Evaporation	86	36	14	136	350	Cont.	8.60	5.72	14.32
	② Vapor Compression	255	97	4	356	200	Cont.	25.50	11.60	37.10
	③ Vacuum Distillation Pyro.	255	106	1	362	800	Cont.	25.50	12.30	37.80
	④ Vapor Diffusion	322	22	2	344	500	Cont.	32.20	2.78	34.98
6 Men 7 Days	* ① Air Evaporation	120	54	17	191	900	Cont.	12.0	8.25	20.25
	② Vapor Compression	275	105	4	384	300	Cont.	27.5	12.50	40.00
	③ Vacuum Distillation Pyro.	275	114	1	390	1400	Cont.	27.5	13.10	40.60
	④ Vapor Diffusion	352	25	3	380	1200	Cont.	35.2	3.23	38.43
4 Men 45 Days	* ① Air Evaporation	103	80	25	208	600	Cont.	10.3	12.10	22.40
	② Vapor Compression	265	167	8	440	240	Cont.	26.5	20.10	46.60
	③ Vacuum Distillation Pyro.	265	191	2	458	1150	Cont.	26.5	22.40	48.90
	④ Vapor Diffusion	338	47	4	389	800	Cont.	33.8	5.88	39.68

NOTE: Expendables include such items as filters, catalysts or chemicals normally programmed for replacement.

* Selected system candidate based on minimum weight and power penalty.

For a 45-day mission, a review of several methods of water reclamation seemed practical, since the time span in long enough so that a significant weight saving could be made.

Four concepts of water reclamation systems were reviewed and are schematically shown: air evaporation (Figure A-34), vapor compression (Figure A-35); vacuum distillation pyrolysis (Figure A-36); and vapor diffusion (Figure A-37).

Tradeoff study curves were constructed to show the system fixed weight, spare weight, expendable weight, electrical power penalty and the total system weight penalty for each concept (Figures A-38 through A-41).

Table A-13 is a summary of the tradeoff study and shows that the air evaporation concept for reclaiming water has the lowest weight penalty, and the vapor diffusion is the next best. However, when considering the overall weight and electrical power penalties, the vapor compression concept is the best selection.

For tug applications a regenerative system presents the lower weight penalty when compared with a stored water system. However, since fuel cells will be used for the tug, and water generation is a by-product of these cells, there is no practical need for a regenerative system.

Table A-14 shows the water consumption and generation requirements.

For normal metabolic and washing needs there is ample water available for the 6-man or 4-man CM configuration. For EVA operations using PLSS on the lunar surface the water requirement for a 4 hour sorties is 5.6 pounds. Four men would increase this water usage requirement to 22.4 pounds. The

Table A-14. Water Consumption and Generation Requirements

Requirements	Water Consumption	Water Generation by Fuel Cells*
Metabolic	6.13 lb/man day	
Washing	4.0 lb/man day	
Total	10.13 lb/man day	70 lb/day
6 men	60.78 lb/day	70 lb/day
4 men	40.52 lb/day	70 lb/day
*Based on 3.5 kw at 20 lb H_2O/day/kw		

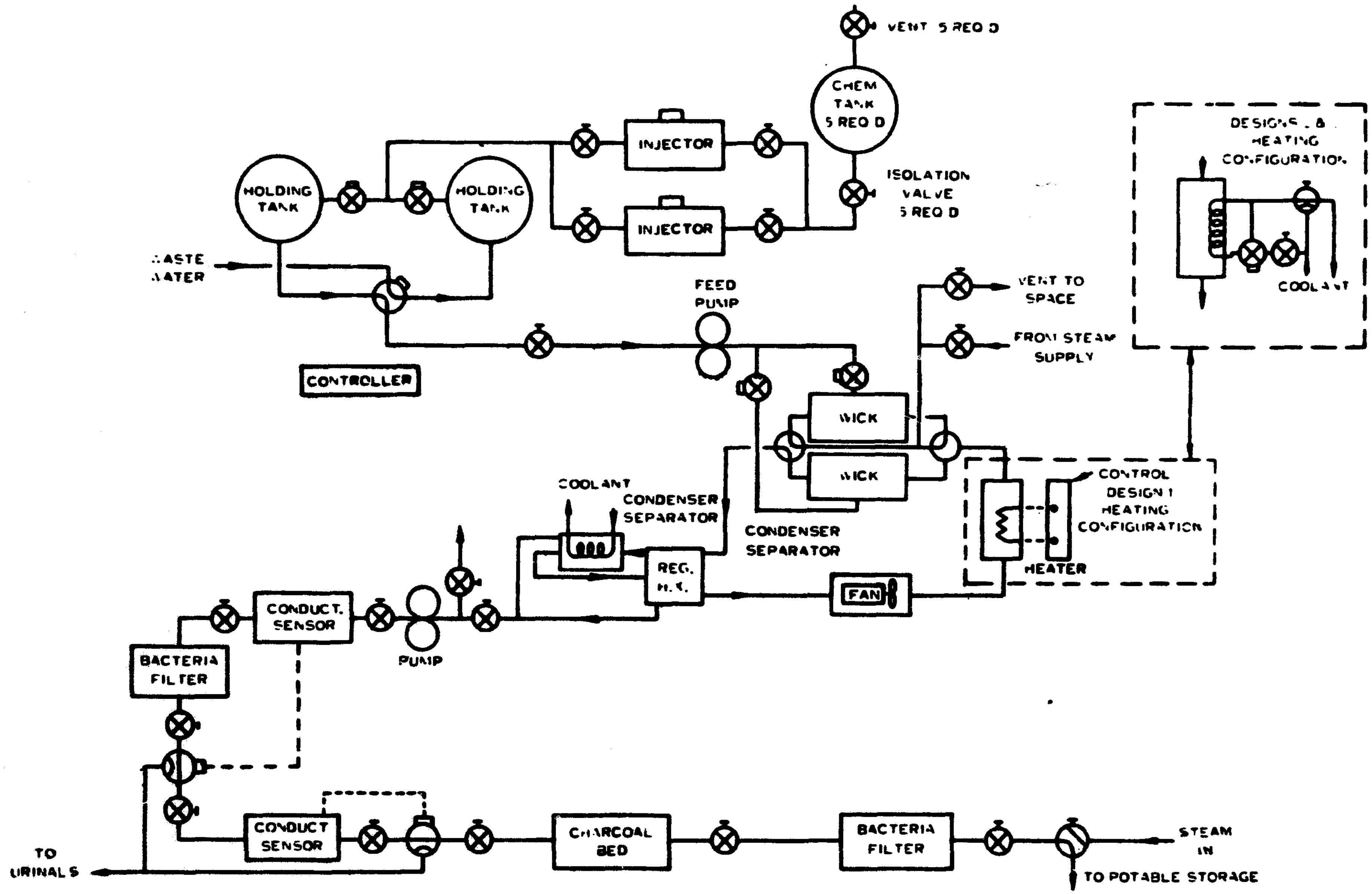


Figure A-34. Closed Cycle Air Evaporation Concept

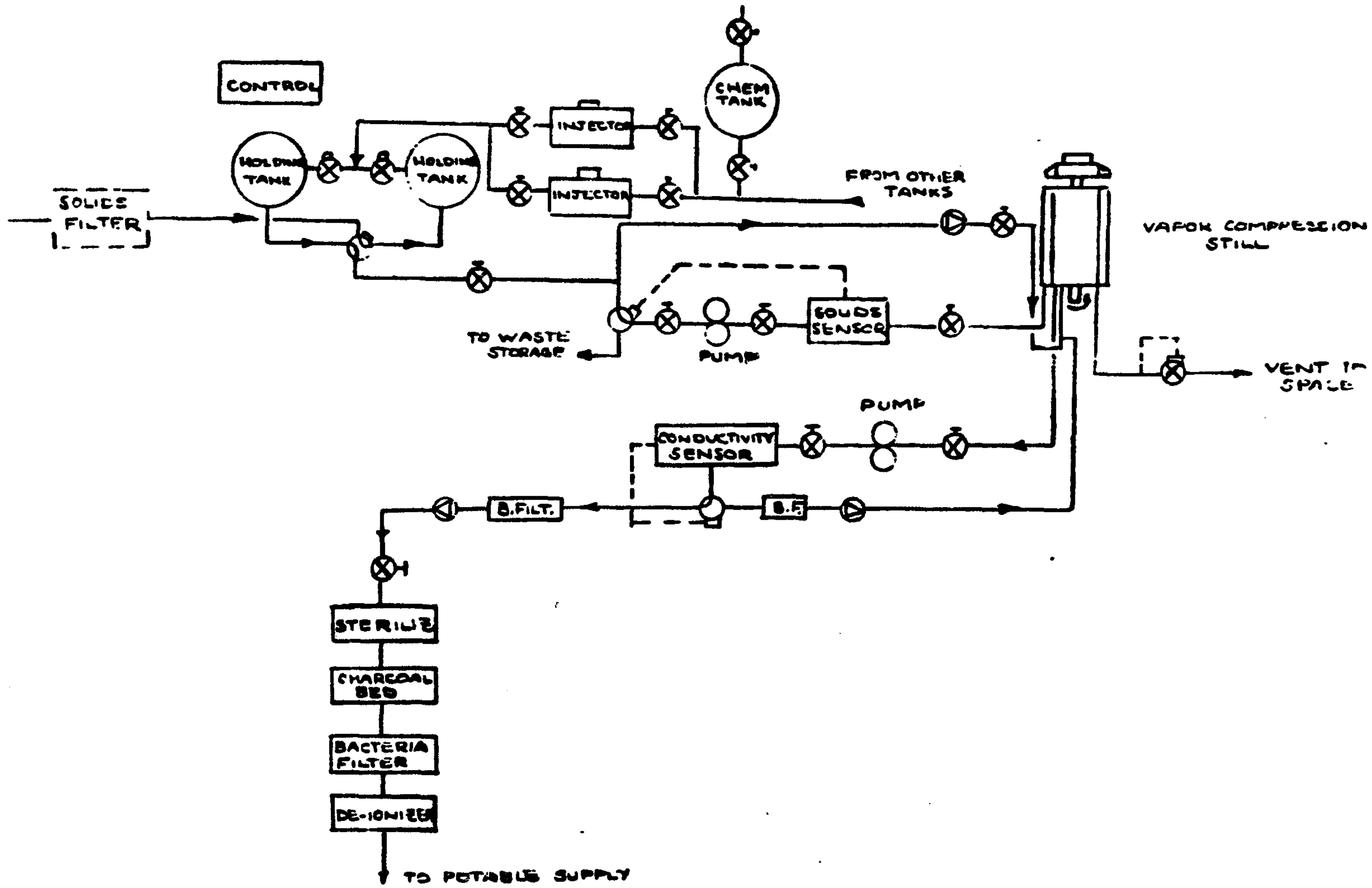


Figure A-35. Vapor Compression

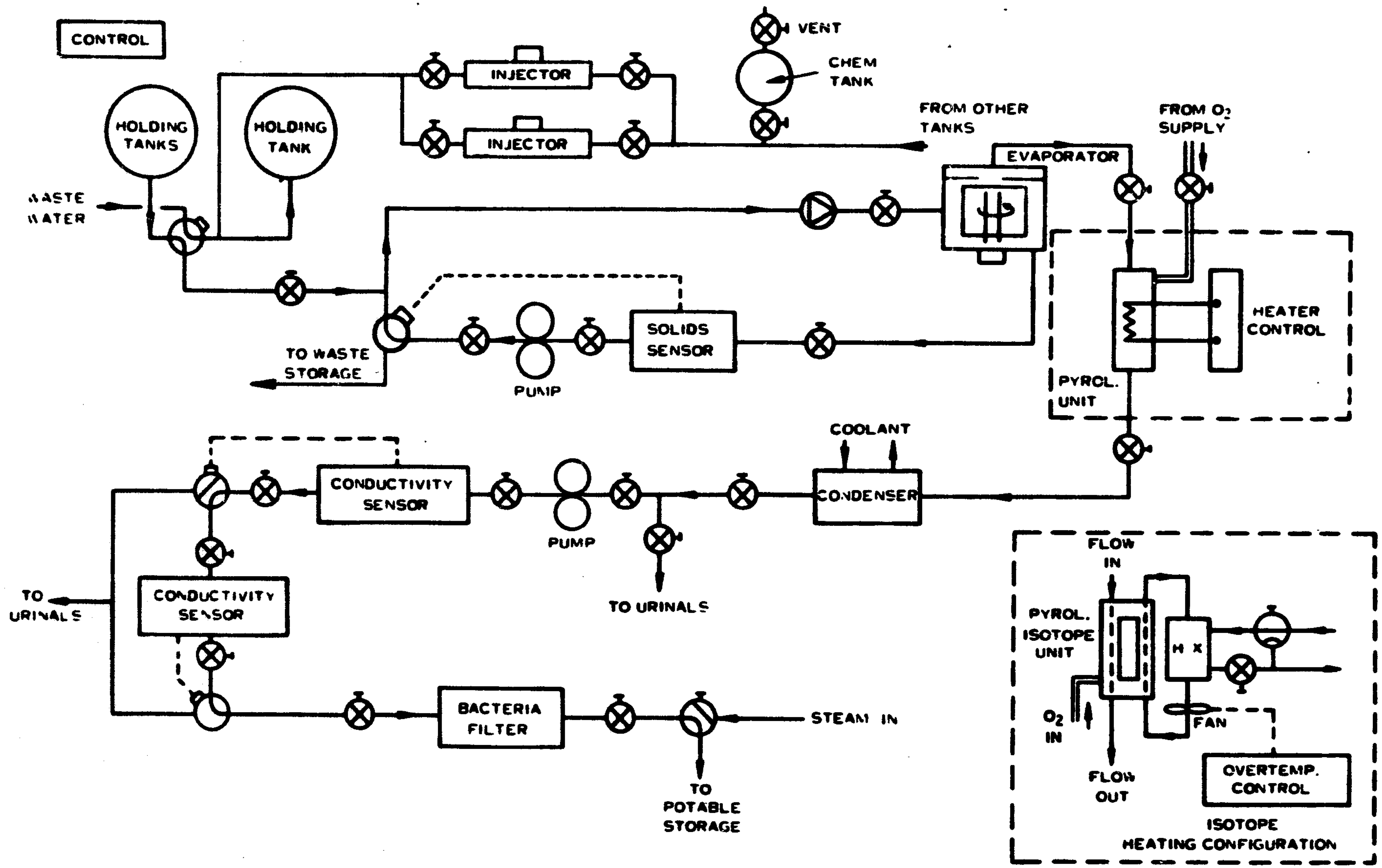


Figure A-36. Vacuum Distillation/Pyrolysis Concept

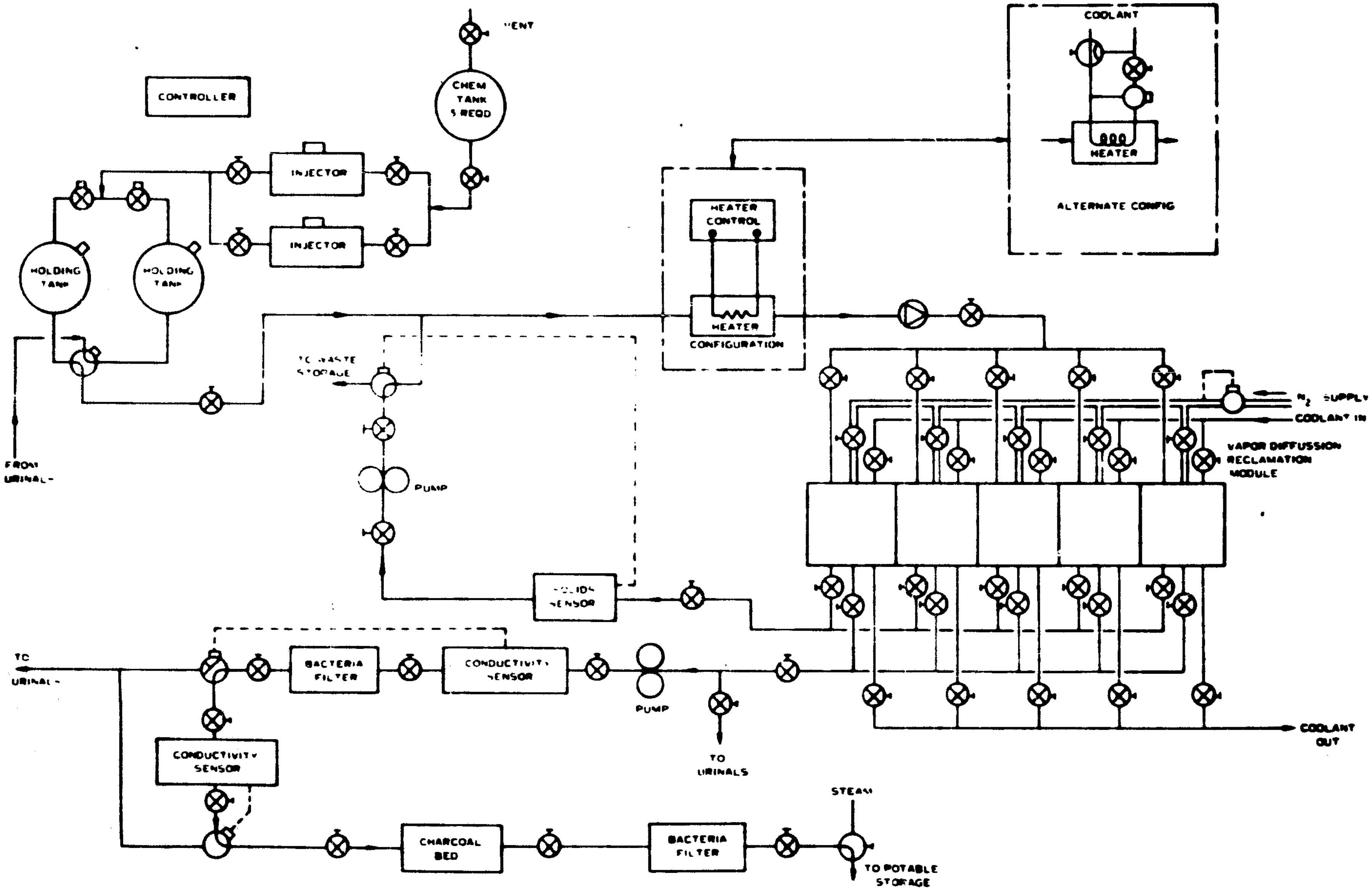


Figure A-37. Vapor Diffusion Concept

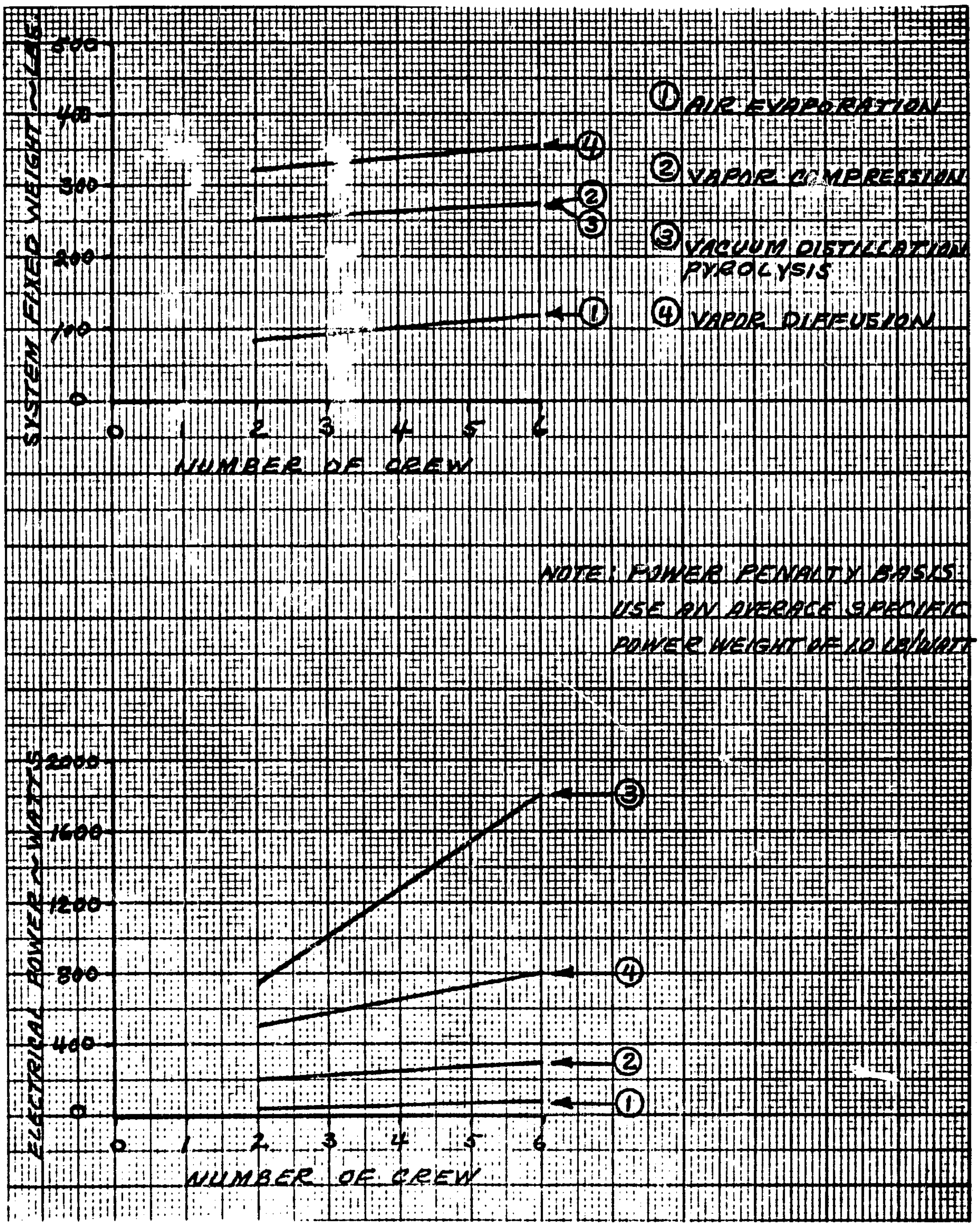


Figure A-38. Water Reclamation System Weight Trade Study System Fixed Weight and Electrical Power Penalty

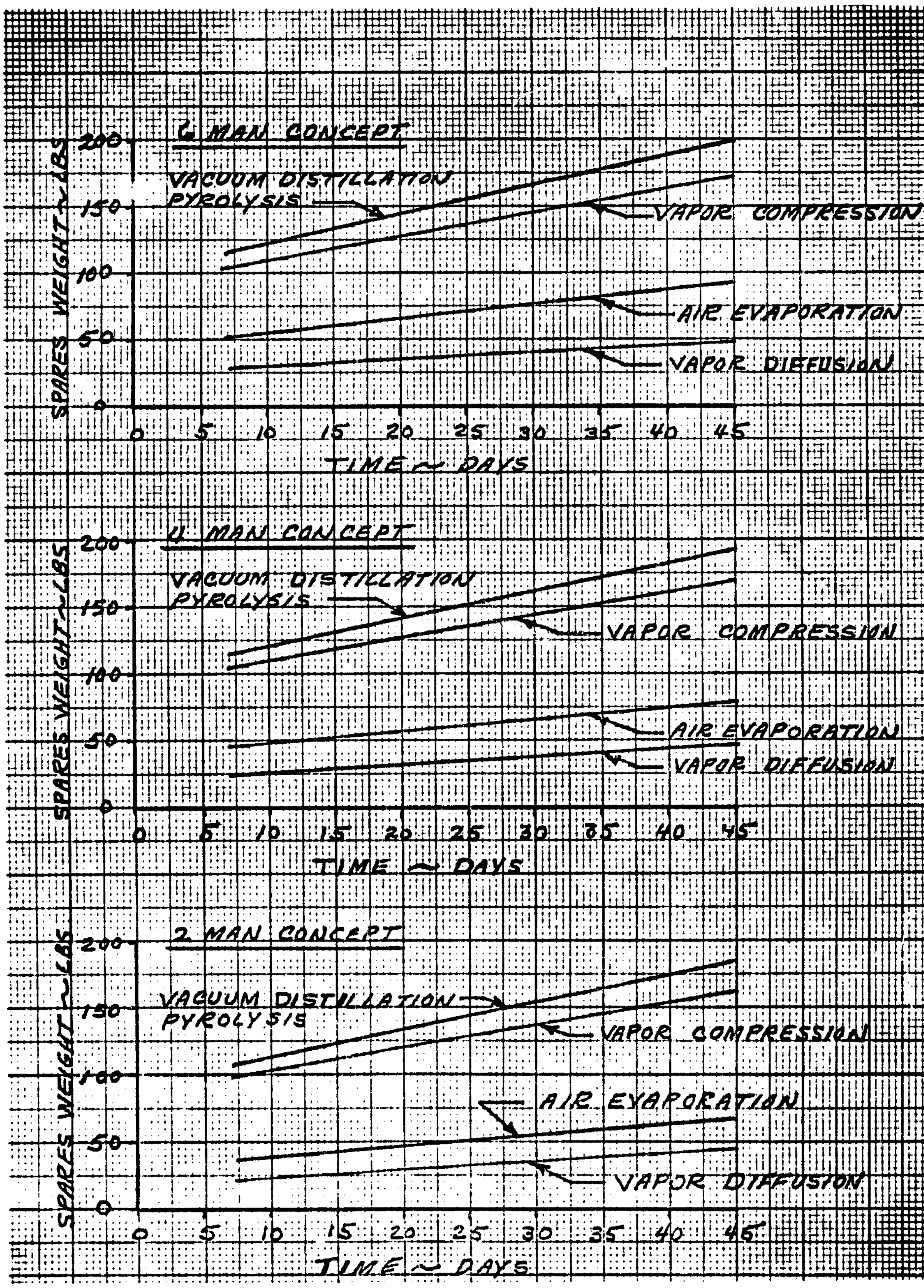
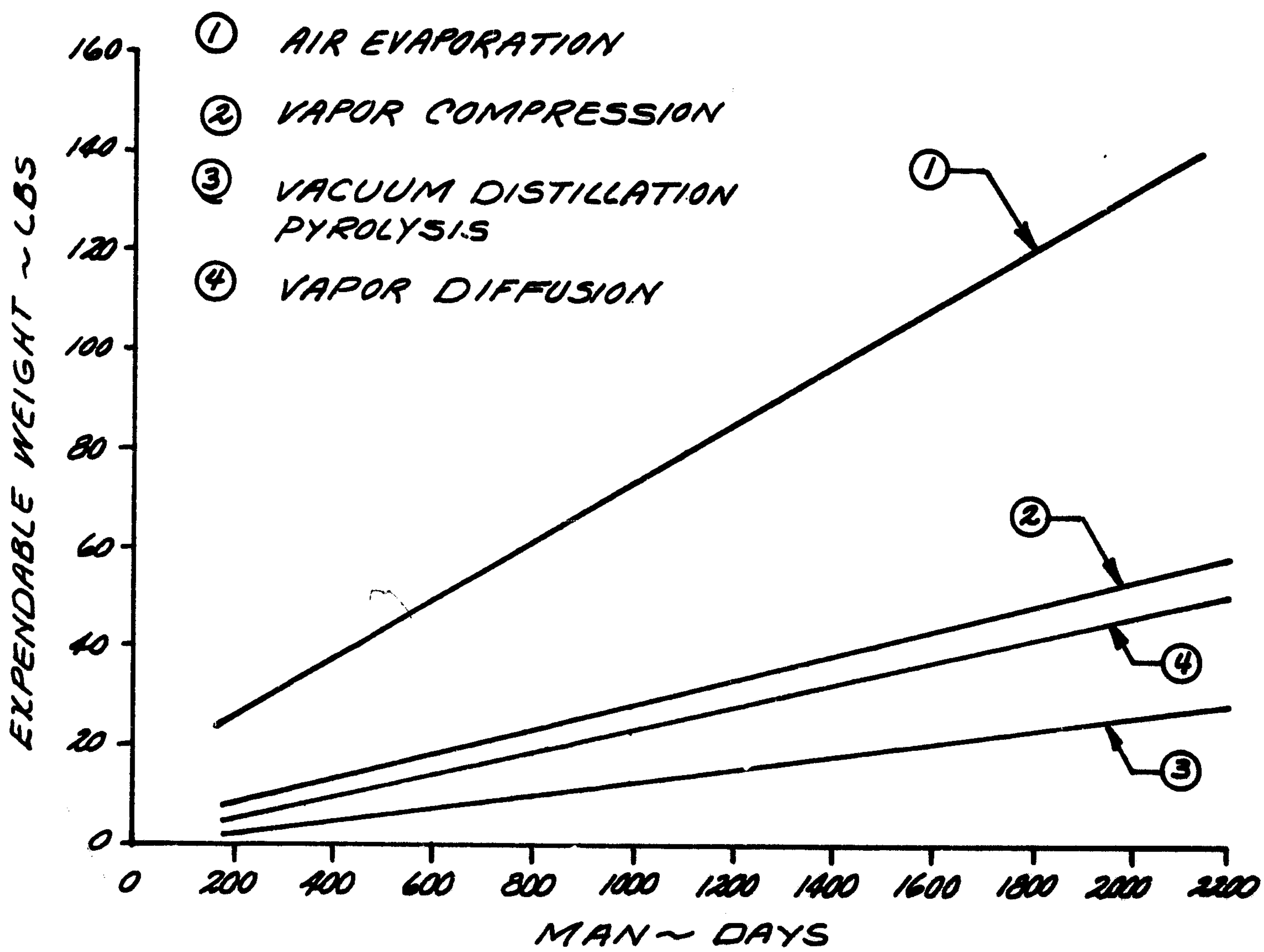


Figure A-39. Water Reclamation Spares Weight Trade Study



NOTE: EXPENDABLES INCLUDE SUCH ITEMS AS FILTERS, CATALYSTS OR CHEMICALS NORMALLY PROGRAMMED FOR REPLACEMENT.

Figure A-40. Expendable System Weight Trade Study

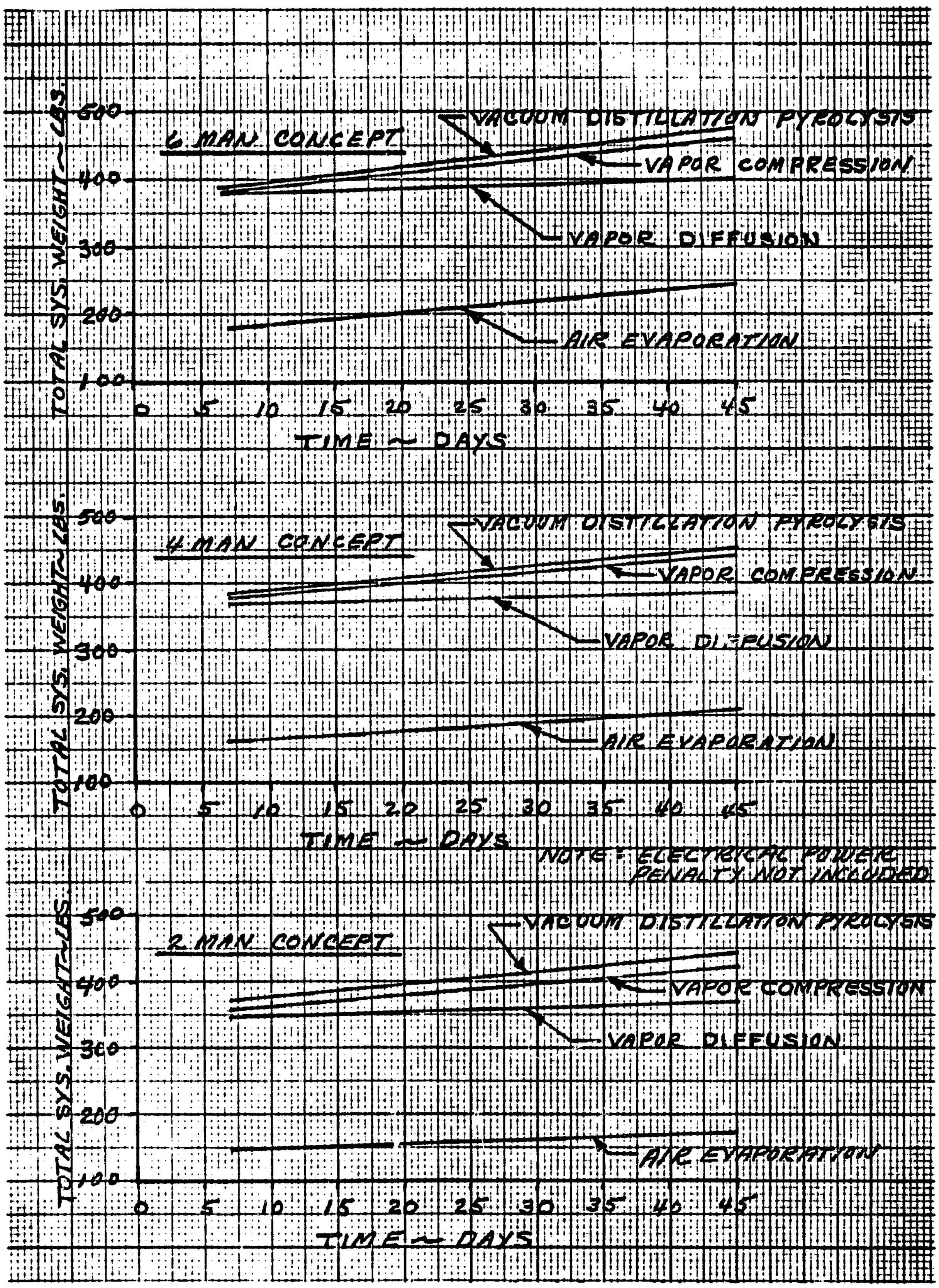


Figure A-41. Water Reclamation Total System Weight Trade Study resultant water usage rate with PLSS operation is 66.92 lbs/day with 70 lb/day being generated. The present water balance indicates that at peak electrical load output the fuel cells can generate sufficient water to meet the crew water consumption requirements. However, if longer EVA's are planned (8 hrs/sortie), or conditions where electrical power demand is reduced, water generation by fuel cell operation would not be sufficient and a regenerative cycle would be practical to supplement the delta water consumption requirements.

Secondary Subsystems Evaluation

Tradeoff and comparison studies were performed in those areas where multiple solutions were available. The tradeoffs and recommendations presented in the preceding sections were made for the selection of an optimum assembly process to be used for tug applications. However, there are other Environmental Control Systems areas where multiple solutions are not available, but in those cases selection as part of the ECLSS rest heavily with subsystem integration requirements and design. Subsystems or assemblies that fall within this category are listed below.

1. Atmospheric control
   - Circulation/ventilation
   - Temperature and humidity control
   - Pressure control
2. Active thermal control
   - Cabin thermal loop
   - Radiator loop
3. Water management
   - Wash and condensate recovery
   - Water storage and control
4. Waste management

5. Emergency life support

   Emergency atmosphere and storage

   EVA life support system

6. Food management

7. Crew support and accommodations

8. Hygiene

   Housekeeping and atmosphere conditioning

For tug applications the weight of these subsystems were scaled down directly from space station data references 14. 1-1 and 14. 1-2. Fixed system weight for the above subassemblies were assumed to be a function of crew size and are presented in Figures A-42 and A-43. Tables A-15 through A-20 show subsystem weight, power and volume penalty for the following tug missions: 2 men for 7 days; 6 men for 7 days; and 4 men for 45 days. Wash and condensate recovery plus water storage and control were considered as a weight penalty assessed only to a closed loop system. Tables for the open system did not include these weight values.

Volume data were obtained from computed density factors based upon space station data on subassembly weight and volume. A list of the density values used to obtain volume sizes for tug are presented in Table A-21. Electrical power data were obtained directly from space station subsystem evaluation studies (Reference A-1 and A-2).

Consumables

Consumables for tug applications were evaluated by the use of a RAX Computer Program (Reference A-3) designed for space station studies. The program is flexible and readily adaptable for tug crew size and mission requirements.

The program provides the following consumable weight information.

Food management:

Dehydrated food	Other food package
Dehydrated food package	Water in food
Other food	Food package reserve

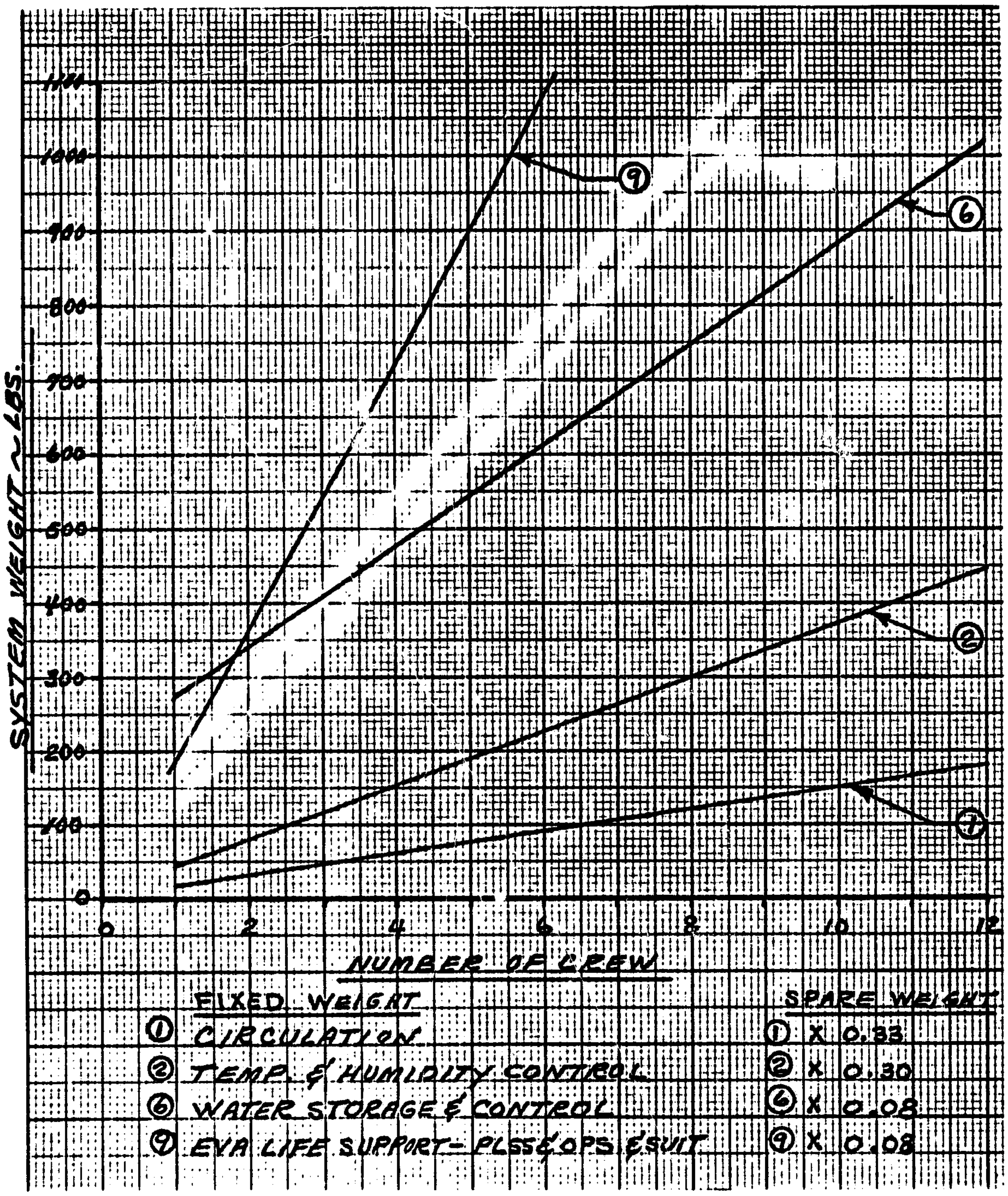


Figure A-42. ECLSS Secondary Subsystem Weight
(Fixed Weight per Subsystem)

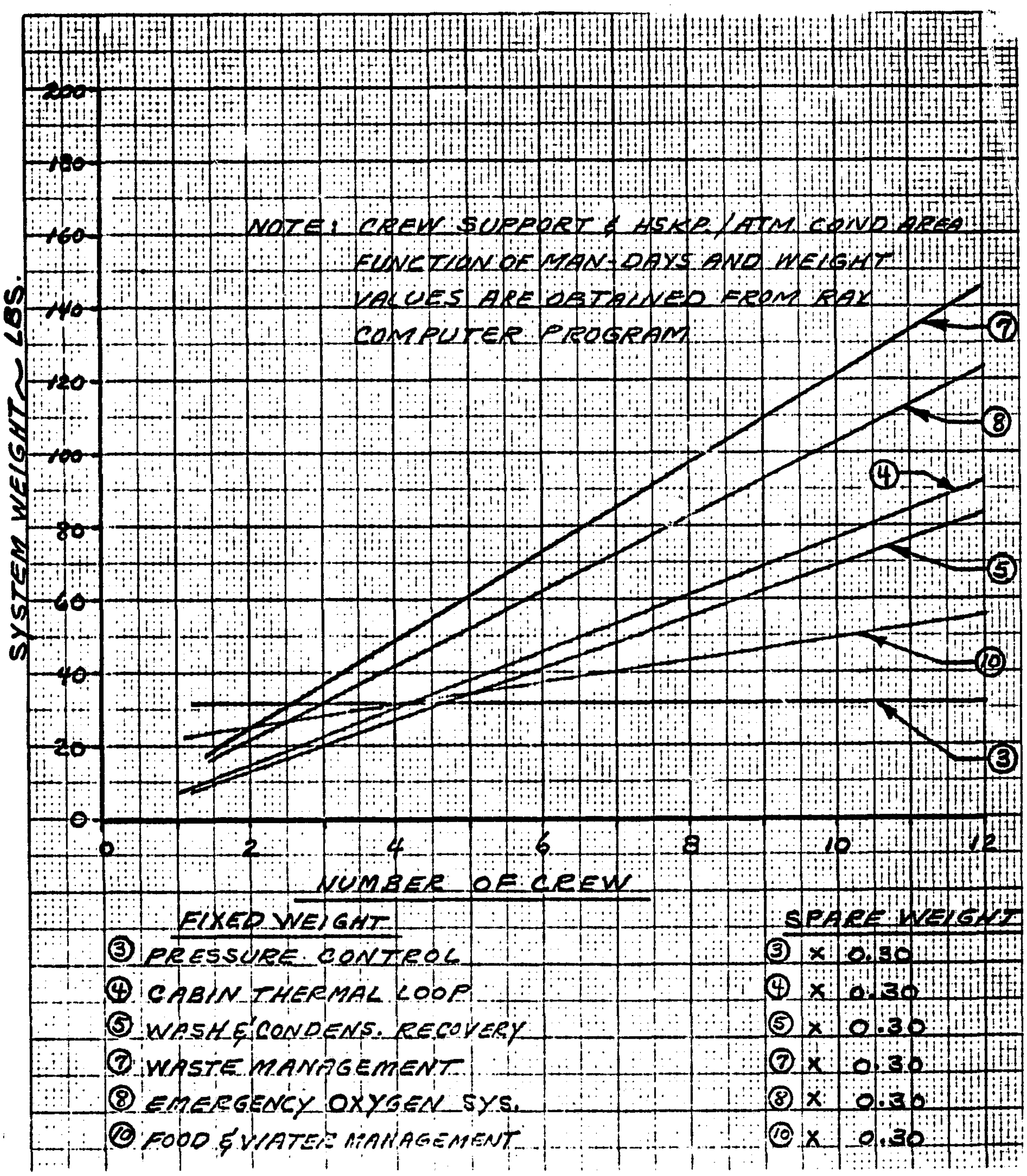


Figure A-43. ECLSS Secondary Subsystem Weight Study
(Fixed Weight per Subsystem)

Table A-15. ECLSS Secondary Systems (Closed System)
Mission ~2 Men, 7 Days

SUBSYSTEMS REQUIRED	WEIGHT PENALTY ~ LBS.				ELEC. POWER		VOLUME ~ FT^3		
	FIXED	SPARES	EXPEND.	TOTAL	WATTS	CYCLE	FIXED	SPARES	TOTAL
① Circulation	31	10	--	41	50	Cont.	4.62	1.77	6.39
② Temp. & Hum. Control	80	24	6	110	250	Cont.	11.90	5.31	17.21
③ Pressure Control	30	10	--	40	10	Cont.	1.52	0.57	2.09
④ Cabin Thermal Loop	15	4	--	19	200	Cont.	1.61	0.41	2.02
⑤ Wash & Condens. Recovery	14	4	7	25	150	Cont.	4.68	1.83	6.51
⑥ Water Storage & Control	340	28	--	368	130	Cont.	62.00	2.48	64.48
⑦ Waste Management	24	8	15	47	190	Int.	1.26	2.52	3.78
Emer. & Aux. Life Support Provisions									
⑧ 1. Emerg. Oxygen Sys.	21	7	--	28	--	--	0.46	0.16	0.62
⑨ 2. EVA Life Support Sys.	2 units 360	29	--	389	--	--	14.40	1.16	15.56
⑩ Food Mgmt - Apollo Type	25	8	--	33	50	Int.	2.50	.80	3.30
⑪ Crew Support	--	--	13	13	--	--	--	.59	0.59
⑫ Hskp/Atm. Cond.	5	2	--	7	--	--	0.23	0.09	0.32
TOTAL	945	134	41	1120	1030		105.18	17.69	122.87

NOTE: *WATER FROM FUEL CELLS

EXPENDABLES INCLUDE SUCH ITEMS AS FILTERS, CATALYSTS OR CHEMICALS NORMALLY PROGRAMMED FOR REPLACEMENT.

EVA LIFE SUPPORT SYS. INCLUDES A PLSS & OPS.
IF A RESISTANCE OVEN IS USED ADD 75 LBS & 80 WATTS AVG. POWER

Table A-16. ECLSS Secondary Systems (Closed System)
Mission ~6 Men, 7 Days

SUBSYSTEMS REQUIRED	WEIGHT PENALTY ~ LBS.				ELEC. POWER		VOLUME ~ FT^3		
	FIXED	SPARES	EXPEND.	TOTAL	WATTS	CYCLE	FIXED	SPARES	TOTAL
① Circulation	91	30	--	121	150	Cont.	13.55	5.32	18.87
② Temp. & Hum. Control	225	68	17	310	250	Cont.	33.50	2.05	35.55
③ Pressure Control	31	10	--	41	30	Cont.	1.57	0.57	2.14
④ Cabin Thermal Loop	46	14	--	60	300	Cont.	4.96	1.43	6.39
⑤ Wash & Condens. Recovery	42	14	25	81	150	Cont.	14.0	6.50	20.50
⑥ Water Storage & Control	610	50	--	660	130	Cont.	710.20	4.42	114.62
⑦ Waste Management	73	21	40	134	190	Int.	3.84	6.70	10.54
Emer. & Aux. Life Support Provisions									
⑧ 1. Emer. Oxygen Sys.	62	19	--	81	--	--	1.38	0.42	1.80
⑨ 2. EVA Life Support Sys.	2 units 360	29	--	389	--	--	14.40	1.16	15.56
⑩ Food Mgmt - Apollo Type	37	11	--	48	50	Int.	3.7	1.10	4.8
⑪ Crew Support	--	--	38	38	--	--	--	--	1.72
⑫ Hskp/Atm. Cond.	18	2	--	20	--	--	0.82	0.09	0.91
TOTAL	1595	268	120	1983	1250		201.92	31.48	233.40

NOTE: *WATER FROM FUEL CELLS

EXPENDABLES INCLUDE SUCH ITEMS AS FILTERS, CATALYSTS OR CHEMICALS NORMALLY PROGRAMMED FOR REPLACEMENT.

EVA LIFE SUPPORT SYS. INCLUDES A PLSS & OPS.
IF A RESISTANCE OVEN IS USED ADD 75 LBS & 80 WATTS AVG. POWER

Table A-17. ECLSS Secondary Systems (Closed System)
Mission ~4 Men, 45 Days
3 Day Flt., 28 Days Surface, 14 Day Surface Contingency

SUBSYSTEMS REQUIRED	WEIGHT PENALTY ~ LBS.				ELEC. POWER		VOLUME ~ FT^3		
	FIXED	SPARES	EXPEND.	TOTAL	WATTS	CYCLE	FIXED	SPARES	TOTAL
① Circulation	60	20	--	80	100	Cont.	8.95	3.54	12.49
② Temp. & Hum. Control	152	45	8	205	250	Cont.	22.50	1.08	23.58
③ Pressure Control	30	10	--	40	20	Cont.	1.52	0.57	2.09
④ Cabin Thermal Loop	30	10	--	40	250	Cont.	3.22	1.03	4.25
⑤ Wash & Condens. Recovery	28	9	14	51	150	Cont.	9.35	3.82	13.17
⑥ Water Storage & Control	475	38	--	513	130	Cont.	86.30	3.37	89.67
⑦ Waste Management	49	15	30	94	190*	Int.	2.59	4.95	7.54
Emer. & Aux. Life Support Provisions									
⑧ 1. Emerg. Oxygen Sys.	42	12	--	54	--	--	0.94	0.38	1.32
⑨ 2. EVA Life Support Sys.	4 units 720	58	--	778	--	--	28.80	2.32	31.12
⑩ Food Mgmt - Apollo Type	31	10	--	41	50	Int.	3.1	1.0	4.1
⑪ Crew Support	--	--	154	154	--	--	--	7.0	7.0
⑫ Hskp/Atm. Cond.	63	16	--	79	--	--	2.9	0.7	3.6
TOTAL	1680	243	206	2129	1140		170.17	29.76	199.93

NOTE: *WATER FROM FUEL CELLS

EXPENDABLES INCLUDE SUCH ITEMS AS FILTERS, CATALYSTS OR CHEMICALS NORMALLY PROGRAMMED FOR REPLACEMENT.

EVA LIFE SUPPORT SYS. INCLUDES A PLSS & OPS.
IF A RESISTANCE OVEN IS USED ADD 75 LBS & 80 WATTS AVG. POWER

Table A-18. ECLSS Secondary Systems (Open System)
Mission ~2 Men, 7 Days

SUBSYSTEMS REQUIRED	WEIGHT PENALTY ~ LBS.				ELEC. POWER		VOLUME ~ FT^3		
	FIXED	SPARES	EXPEND.	TOTAL	WATTS	CYCLE	FIXED	SPARES	TOTAL
① Circulation	31	10	--	41	50	Cont.	4.62	1.77	6.39
② Temp. & Hum. Control	80	24	6	110	250	Cont.	11.90	5.31	17.21
③ Pressure Control	30	10	--	40	10	Cont.	1.52	0.57	2.09
④ Cabin Thermal Loop	15	4	--	19	200	Cont.	1.61	0.41	2.02
Wash & Condens. Recovery	--	--	--	--	--	--	--	--	--
*Water Storage & Control	--	--	--	--	--	--	--	--	--
⑦ Waste Management	24	8	15	47	--	--	1.26	2.52	3.78
Emerg. & Aux. Life Support Provisions									
⑧ 1. Emer. Oxygen Sys.	21	7	--	28	--	--	0.46	0.16	0.62
⑨ 2. EVA Life Support Sys.	2 units 360	29	--	389	--	--	14.40	1.16	15.56
⑩ Food & Water Mgmt	25	8	--	33	50	Int.	2.50	.80	3.30
⑪ Crew Support	--	--	13	13	--	--	--	.59	0.59
⑫ Hskp./Atm. Cond.	5	2	--	7	--	--	0.23	0.09	0.32
TOTAL	591	102	34	727	560		38.50	13.38	51.88

NOTE: *WATER FROM FUEL CELLS

EXPENDABLES INCLUDE SUCH ITEMS AS FILTERS, CATALYSTS OR CHEMICALS NORMALLY PROGRAMMED FOR REPLACEMENT.

EVA LIFE SUPPORT SYS. INCLUDES A PLSS & OPS.
IF A RESISTANCE OVEN IS USED ADD 75 LBS & 80 WATTS AVG. POWER

Space Division
North American Rockwell

Table A-19. ECLSS Secondary Systems (Open System)
Mission ~6 Men, 7 Days

SUBSYSTEMS REQUIRED	WEIGHT PENALTY ~LBS.				ELEC. POWER		VOLUME ~FT³		
	FIXED	SPARES	EXPEND.	TOTAL	WATTS	CYCLE	FIXED	SPARES	TOTAL
① Circulation	91	30	--	121	150	Cont.	13.55	5.32	18.87
② Temp. & Hum. Control	225	68	17	310	250	Cont.	33.50	2.05	35.55
③ Pressure Control	31	10	--	41	30	Cont.	1.57	0.57	2.14
④ Cabin Thermal Loop	46	14	--	60	300	Cont.	4.96	1.43	6.39
Wash & Condens. Recovery	--	--	--	--	--	--	--	--	--
*Water Storage & Control	--	--	--	--	--	--	--	--	--
⑦ Waste Management	73	21	40	134	--	--	3.84	6.70	10.54
Emer. & Aux. Life Support Provisions									
⑧ 1. Emer. Oxygen Sys.	62	19	--	81	--	--	1.38	0.42	1.80
⑨ 2. EVA Life Support Sys.	2 units 360	29	--	389	--	--	14.40	1.16	15.56
⑩ Food & Water Mgmt	37	11	--	48	50	Int.	3.7	1.1	4.8
⑪ Crew Support	--	--	38	38	--	--		1.72	1.72
⑫ Hskp./Atm. Cond.	18	2	--	20	--	--	0.82	0.09	0.91
TOTAL	943	204	95	1242	780		77.72	20.56	98.28

NOTE: *WATER FROM FUEL CELLS

EXPENDABLES INCLUDE SUCH ITEMS AS FILTERS, CATALYSTS OR CHEMICALS NORMALLY PROGRAMMED FOR REPLACEMENT.

EVA LIFE SUPPORT SYS. INCLUDES A PLSS & OPS.
IF A RESISTANCE OVEN IS USED ADD 75 LBS & 80 WATTS AVG. POWER

Table A-20. ECLSS Secondary Systems (Open System)
Mission ~4 Men, 45 Days
3 Day Flt., 28 Day Surface, 14 Days Surface Contingency

SUBSYSTEMS REQUIRED	WEIGHT PENALTY ~ LBS.				ELEC. POWER		VOLUME ~ FT^3		
	FIXED	SPARES	EXPEND.	TOTAL	WATTS	CYCLE	FIXED	SPARES	TOTAL
① Circulation	60	20	--	80	100	Cont.	8.95	3.54	12.49
② Temp. & Hum. Control	152	45	8	205	250	Cont.	22.50	1.08	23.58
③ Pressure Control	30	10	--	40	20	Cont.	1.52	0.57	2.09
④ Cabin Thermal Loop	30	10	--	40	250	Cont.	3.22	1.03	4.25
Wash & Condens. Recovery	--	--	--	--	--	--	--	--	--
*Water Storage & Control	--	--	--	--	--	--	--	--	--
⑦ Waste Management	49	15	30	94	--	--	2.59	4.95	7.54
Emer. & Aux. Life Support Provisions									
⑧ 1. Emer. Oxygen Sys.	42	12	--	54	--	--	0.94	0.38	1.32
⑨ 2. EVA Life Support Sys.	4 units 720	58	--	778	--	--	28.80	2.32	31.12
⑩ Food & Water Mgmt	31	10	--	41	50	Int.	3.1	1.0	4.1
⑪ Crew Support	--	--	154	154	--	--	--	7.0	7.0
⑫ Hskp/Atm. Cond.	63	16	--	79	--	--	2.9	0.7	3.6
TOTAL	1177	196	192	1565	670		74.52	22.57	97.09

NOTE: *WATER FROM FUEL CELLS

EXPENDABLES INCLUDE SUCH ITEMS AS FILTERS, CATALYSTS OR CHEMICALS NORMALLY PROGRAMMED FOR REPLACEMENT.

EVA LIFE SUPPORT SYS. INCLUDES A PLSS & OPS.
IF A RESISTANCE OVEN IS USED ADD 75 LBS & 80 WATTS AVG. POWER

Table A-21. Density Values for Computing Subsystem Volumes

ECLSS CONCEPT	FIXED SYS-DENSITY LB/FT^3	SPARES-DENSITY LB/FT^3
CO_2 Removal	29	29
CO_2 Reduction	11.3	22.5
Water Electrolysis	26.9	26.5
Contaminant Control	17.8	33.2
Water Reclamation	10	8.7
Circulation	6.7	5.65
Temp. & Humidity Control	6.75	41.5
Pressure Control	19.7	17.5
Cabin Thermal Loop	9.3	9.7
Wash & Condensation Reclamation	3	6
Water Storage & Control	5.5	11.3
Waste Management	19	9.1
Emergency Life Support	45	--
EVA Life Support		
PLSS	26 } 25 avg.	--
OPS	23 } 25 avg.	--
Food Management	10	--
Food Consumables & Hygiene	22	--

NOTE: To determine the volume of a subsystem fixed hardware and spares, divide the fixed weight or spare weight value by the subsystem density listed in the above table.

Food management (Continued)

Utensils, soap, etc.

Crew support:

Personal hygiene

Clothing, towels, soap

Medicines

Toilet paper, covers, etc.

Housekeeping/atmosphere cond.

Cleansers, trash bags, etc.

Charcoal - odor

Filters - air

Atmospheric storage

Oxygen

Metabolic and leakage

Tank weight

Reserve O_2

Reserve tank weight

Nitrogen

Leakage

Tank weight

Reserve

Tank weight

Water storage

Metabolic and wash

Tank

Reserve

Tank

Tables A-22 and A-23 show the consumable breakdown for the three tug missions.

Food will consist of both freeze dried and frozen types and the amount of each will depend upon the length of mission. For longer missions a greater percentage of the frozen food will be stowed. To prepare the frozen food a resistance oven will be provided. This will add 75 pounds of weight and require an added 80 watts of average power. For the 7-day mission it is planned that only freeze-dried food be used.

LiOH was placed in the consumable category for the weight evaluation and the amount required for this study was based upon one pound of LiOH compound to absorb 0.925 pound of CO_2. Canister weight was assumed to be 60 percent of the compound weight.

Table A-22. ECLSS Final Weight Summary (Closed System)

SELECTED SYSTEM		TOTAL SYSTEM WEIGHT LBS			ELECT. POWER WATTS			VOLUME FT^3		
CONCEPT	METHOD	2 Men 7 Days	6 Men 7 Days	4 Men 45 Days	2 Men 7 Days	6 Men 7 Days	4 Men 45 Days	2 Men 7 Days	6 Men 7 Days	4 Men 45 Days
CO_2 Removal	Steam Desorbed Resin	177	277	272	400	780	560	6.13	9.55	9.38
CO_2 Reduction	Sabatier/Methane Dump	46	80	145	50	75	60	3.63	5.32	8.10
H_2O Electrolysis	Gas Circulation		155			3000			5.76	
	Solid Polymer	92		161	1400		2000	3.42		6.01
Contaminent Contr.	Cat. Ox./Sorption	57	86	71	60	145	100	2.97	4.52	3.49
H_2O Reclamation	Air Evaporation	135	191	208	350	900	600	14.32	20.25	22.40
Circulation	Fans	41	121	80	50	150	100	6.39	18.87	12.49
Temp. & Hum. Control	Fans, Hx. & Conden. Hx.	110	310	205	250	250	250	17.21	35.55	23.58
Pressure Control	5 to 7 PSIA Normal	40	41	40	10	30	20	2.09	2.14	2.09
Cabin Thermal Loop	Pumps & Hx.	19	60	40	200	300	250	2.02	6.39	4.25
Wash & Condens. Rec.	Reverse Osmosis	25	81	51	150	150	150	6.51	20.50	13.17
H_2O Storage & Cont.	Potable Tanks & Pumps	368	660	513	130	130	130	64.48	14.62	89.67
Waste Management	Dry John & Waste Storage	47	134	94	190	190	190	3.78	10.54	7.54
Emer. Life Support	High Press. Gas Storage	28	81	54	--	--	--	0.62	1.80	1.32
EVA Life Support	PLSS & Ops.	389	389	778	--	--	--	15.56	15.56	31.12
Food Mgmt.	Reconstituted with H_2O	33	48	41	50	50	50	3.3	4.8	4.1
Crew Support	Personal Hygiene	13	38	154	--	--	--	0.59	1.72	7.0
Hskp/Atm. Cond.	Filters, Trash Bags, etc.	7	20	79	--	--	--	0.32	0.91	3.6
SUBTOTAL		1627	2771	2986	3290	6150	4460	153.3	278.8	249.3
Consumables										
Food	Freeze Dried	54	161	665	--	--	--	2.46	7.33	30.20
Emer. Oxygen	Repress & PLSS Resupply, Leakage	57.0	65.0	374.0	--	--	--	--	--	--
Atmosp. Storage										
CM Charge	O_2 Plus N_2 @ 14.7 PSIA	95	95	95	--	--	--	--	--	--
Oxygen	CO_2 Reduction	--	--	--	--	--	--	--	--	--
Nitrogen	Tank	3	3	17	--	--	--	--	--	--
Water Storage	H_2O Electrolysis	--	--	--	--	--	--	--	--	--
SUBTOTAL		209.0	324.0	1151.0	--	--	--	2.46	7.33	30.20
TOTAL		1836	3095	4137	3290	6150	4460	155.7	286.1	279.5

Table A-23. ECLSS Final Weight Summary (Open System)

SELECTED SYSTEM		TOTAL SYSTEM WEIGHT LBS			ELECT. POWER WATTS			VOLUME FT^3		
CONCEPT	METHOD	2 Men 7 Days	6 Men 7 Days	4 Men 45 Days	2 Men 7 Days	6 Men 7 Days	4 Men 45 Days	2 Men 7 Days	6 Men 7 Days	4 Men 45 Days
Contaminent Control	Cat. Ox./Sorption	57	86	71	60	145	100	2.97	4.52	3.49
CO_2 Removal	LiOH	69	186	809	--	--	--	2.46	6.64	28.79
Circulation	Fans	41	121	80	50	150	100	6.39	18.87	12.49
Temp. & Hum. Cont.	Fans, Hx. & Conden. Hx.	110	310	205	250	250	250	17.21	35.55	23.58
Pressure Control	5 to 7 PSIA Normal	40	41	40	10	30	20	2.09	2.14	2.09
Cabin Thermal Loop	Pumps & Hx.	19	60	40	200	300	250	2.02	6.39	4.25
Waste Mgmt.	Waste Storage	47	134	94	--	--	--	3.78	10.54	7.54
Emer. Life Support	High Press. Gas Storage	28	81	54	--	--	--	0.62	1.80	1.32
EVA Life Support	PLSS & Ops.	389	389	778	--	--	--	15.56	15.56	31.12
Water & Food Mgmt.	Typical of Apollo	33	48	41	50	50	50	3.3	4.8	4.1
Crew Support	Personal Hygiene	13	38	154	--	--	--	0.59	1.72	7.0
Hskp/Atm. Cond.	Filters, Trash Bags, etc.	7	20	79	--	--	--	0.32	0.91	3.6
SUBTOTAL		853	1504	2445	620	925	770	56.3	109.5	129.3
Consumables										
Food	Freeze Dried	54	161	665	--	--	--	2.46	7.33	30.20
Emerg. Oxygen	Repress & PLSS Resupply	37	37	259	--	--	--	--	--	--
Atmos. Storage										
CM Charge	O_2 Plus N_2 @ 14.7 PSIA	95	95	95	--	--	--	--	--	--
Oxygen	Metabolic, Leakage, etc.	28	80	324	--	--	--	--	--	--
Nitrogen	High Press Tank	3	3	17	--	--	--	--	---	--
Water Storage	Potable Tanks	*	*	*	--	--	--	--	--	--
SUBTOTAL		217	376	1360	--	--	--	2.46	7.33	30.20
TOTAL		1070	1880	3805	620	925	770	58.8	116.8	159.5

*WATER FROM FUEL CELLS

Water storage was computed by the RAX program but this item was not included in the weight summary tables for consumables. The amount of water required versus the water generated by the fuel cells was itemized in Table A-12. Since the water generated is greater than the water being consumed by the crew there would be no water weight penalty to the ECLSS and therefore no values were placed on the tables.

Oxygen storage is based upon the crew metabolic and CM leakage rates only. The O_2 tank weight and reserve O_2 and tank weight printed out in the RAX program is not applicable to tug applications since these are directly chargeable to the main propulsion system.

The process rates used in the RAX program to evaluate consumables are listed below.

Food Weight

Dehydrated food	1.04 lb/man day
Dehydrated food package	0.73 lb/man day
Other food, (frozen dry)	0.64 lb/man day
Water stored in other food (ice)	0.96 lb/man day
Other food package (frozen food)	0.45 lb/man day
Utensils, soap, etc.	0.0125 lb/man day
Total	3.8325 lb/man day

Water Weight

Water for drinking in food reconstitution	5.17 lb/man day
Hand washing (used for reserve)	4.0 lb/man day

Water tank 5% of total water weight

Oxygen

Metabolic Osygen Consumption	1.84 lb/man day
Leakage	0.362 lb/day
Tank weight	5% of total O_2 weight

Nitrogen

Leakage	0.316 lb/day
Tank weight	5% of total N_2 weight

Crew support

Personal hygiene	0.50 lb/man day
Clothing, towels, soap	0.13 lb/man day
Medicines	0.035 lb/man day
Toilet paper, covers, etc.	0.25 lb/man day
Total	0.915 lb/man day

Housekeeping/atmosphere cond.

Cleansers, trash bags, etc.	0.20 lb/man day
Charcoal-odor	0.20 lb/man day
Filter-air	0.07 lb/man day
Total	0.47 lb/man day

LiOH elements

LiOH compound	2.3 lb/man day
Canisters	60% of compound weight
Spares	22% of compound and canister weight

Emergency Oxygen		2 men	6 men	4 men
		1 day	1 day	28 day
CM repress.	1250 ft^3 at 5 psia, 70 F			
PLSS recharge	1 lb/charge = 4-hr supply	2 lb	2 lb	224 lb

28 day on lunar surface
8 hr supply 2 lb
4 units

} 224 lb

System Tradeoff Summary

Results of the tradeoff analysis show that the nonregenerative (open) type of system is best adapted to fulfill all EC/LSS requirements and mission objectives of the tug program. The regenerative (closed) systems, such as molecular-sieve or steam-desorbed resin for the removal of CO_2, Sabatier or Bosch reactor for water recovery from CO_2 reduction, wick feed or gas circulation for O_2 recovery from water electrolysis, become competitive for the longer mission stay times and larger crew sizes. A regenerative system nearly becomes competitive in the removal of CO_2 for the four-man, 45-day mission. Total weight, including power penalties for the competitive candidates, was 809 pounds for LiOH and 832 pounds for steam-desorbed resin. A two-bed molecular sieve used on Skylab is being evaluated, and preliminary data point to a lighter weight system, which may be competitive to LiOH for a four-man, 45-day mission. Table A-24 and A-25 were compiled to compare between a regenerative (closed) loop EC/LSS system with a nonregenerative (open) loop EC/LSS system. These tables summarize weight, power, and volumes for all three tug missions. Table A-24 shows the results of the best regenerative concepts, which were selected from tradeoff studies. Table A-25 shows results of an open-loop EC/LSS system. The distinction between a closed- and open-loop EC/LSS system used for comparative analysis is identified by the methods selected. A closed-loop system is identified by use of regenerative subsystem. The open-loop system is identified by the use of nonregenerative types of subsystem.

Review of Tables A-24 and A-25 clearly indicates that the overall system weight, power, and volume for the closed-loop system is considerably higher than for the open-loop system. However, when the consumable weight penalties are considered the two systems become competitive for the longer mission durations.

Table A-24. EC/LSS Final Weight Summary (Closed System)

SELECTED SYSTEM		TOTAL SYSTEM WEIGHT LBS			ELECT. POWER WATTS			VOLUME FT^3		
CONCEPT	METHOD	2 Men 7 Days	6 Men 7 Days	4 Men 45 Days	2 Men 7 Days	6 Men 7 Days	4 Men 45 Days	2 Men 7 Days	6 Men 7 Days	4 Men 45 Days
CO_2 Removal	Steam Desorbed Resin	177	277	272	400	780	560	6.13	9.55	9.38
CO_2 Reduction	Sabatier/Methane Dump	46	80	145	50	75	60	3.63	5.32	8.10
H_2O Electrolysis	Gas Circulation		155			3000			5.76	
	Solid Polymer	92		161	1400		2000	3.42		6.01
Contaminent Contr.	Cat. Ox./Sorption	57	86	71	60	145	100	2.97	4.52	3.49
H_2O Reclamation	Air Evaporation	135	191	208	350	900	600	14.32	20.25	22.40
Circulation	Fans	41	121	80	50	150	100	6.39	18.87	12.49
Temp. & Hum. Control	Fans, Hx. & Conden. Hx.	110	310	205	250	250	250	17.21	35.55	23.58
Pressure Control	5 to 7 PSIA Normal	40	41	40	10	30	20	2.09	2.14	2.09
Cabin Thermal Loop	Pumps & Hx.	19	60	40	200	300	250	2.02	6.39	4.25
Wash & Condens. Rec.	Reverse Osmosis	25	81	51	150	150	150	6.51	20.50	13.17
H_2O Storage & Cont.	Potable Tanks & Pumps	368	660	513	130	130	130	64.48	114.62	89.67
Waste Management	Dry John & Waste Storage	47	134	94	190	190	190	3.78	10.54	7.54
Emer. Life Support	High Press. Gas Storage	28	81	54	--	--	--	0.62	1.80	1.32
EVA Life Support	PLSS & Ops.	389	389	778	--	--	--	15.56	15.56	31.12
Food Mgmt.	Reconstituted with H_2O	33	48	41	50	50	50	3.3	4.8	4.1
Crew Support	Personal Hygiene	13	38	154	--	--	--	0.59	1.72	7.0
Hskp/Atm. Cond.	Filters, Trash Bags, etc.	7	20	79	--	--	--	0.32	0.91	3.6
SUBTOTAL		1627	2771	2986	3290	6150	4460	153.3	278.8	249.3
Consumables										
Food	Freeze Dried	54	161	665	--	--	--	2.46	7.33	30.20
Emer. Oxygen	Repress & PLSS Resupply, Leakage	57.0	65.0	374.0	--	--	--	--	--	--
Atmosp. Storage										
CM Charge	O_2 Plus N_2 @ 14.7 PSIA	95	95	95	--	--	--	--	--	--
Oxygen	CO_2 Reduction	--	--	--	--	--	--	--	--	--
Nitrogen	Tank	3	3	17	--	--	--	--	--	--
Water Storage	H_2O Electrolysis	--	--	--	--	--	--	--	--	--
SUBTOTAL		209.0	324.0	1151.0	--	--	--	2.46	7.33	30.20
TOTAL		1836	3095	4137	3290	6150	4460	155.7	286.1	279.5

Table A-25. EC/LSS Final Weight Summary (Open System)

SELECTED SYSTEM		TOTAL SYSTEM WEIGHT LBS			ELECT. POWER WATTS			VOLUME FT^3		
CONCEPT	METHOD	2 Men 7 Days	6 Men 7 Days	4 Men 45 Days	2 Men 7 Days	6 Men 7 Days	4 Men 45 Days	2 Men 7 Days	6 Men 7 Days	4 Men 45 Days
Contaminent Control	Cat. Ox./Sorption	57	86	71	60	145	100	2.97	4.52	3.49
CO_2 Removal	LiOH	69	186	809	--	--	--	2.46	6.64	28.79
Circulation	Fans	41	121	80	50	150	100	6.39	18.87	12.49
Temp. & Hum. Cont.	Fans, Hx. & Conden. Hx.	110	310	205	250	250	250	17.21	35.55	23.58
Pressure Control	5 to 7 PSIA Normal	40	41	40	10	30	20	2.09	2.14	2.09
Cabin Thermal Loop	Pumps & Hx.	19	60	40	200	300	250	2.02	6.39	4.25
Waste Mgmt.	Waste Storage	47	134	94	--	--	--	3.78	10.54	7.54
Emer. Life Support	High Press. Gas Storage	28	81	54	--	--	--	0.62	1.80	1.32
EVA Life Support	PLSS & Ops.	389	389	778	--	--	--	15.56	15.56	31.12
Water & Food Mgmt.	Typical of Apollo	33	48	41	50	50	50	3.3	4.8	4.1
Crew Support	Personal Hygiene	13	38	154	--	--	--	0.59	1.72	7.0
Hskp/Atm. Cond.	Filters, Trash Bags, etc.	7	20	79	--	--	--	0.32	0.91	3.6
SUBTOTAL		853	1504	2445	620	925	770	56.3	109.5	129.3
Consumables										
Food	Freeze Dried	54	161	665	--	--	--	2.46	7.33	30.20
Emerg. Oxygen	Repress & PLSS Resupply	37	37	259	--	--	--	--	--	--
Atmos. Storage										
CM Charge	O_2 Plus N_2 @ 14.7 PSIA	95	95	95	--	--	--	--	--	--
Oxygen	Metabolic, Leakage, etc.	28	80	324	--	--	--	--	--	--
Nitrogen	High Press Tank	3	3	17	--	--	--	--	---	--
Water Storage	Potable Tanks	*	*	*	--	--	--	--	--	--
SUBTOTAL		217	376	1360	--	--	--	2.46	7.33	30.20
TOTAL		1070	1880	3805	620	925	770	58.8	116.8	159.5

*WATER FROM FUEL CELLS

Figure A-44 substantiates this. Weight tradeoff curves plotted against mission duration (time) show the weight penalty trends for an open versus closed system. This evaluation did not include the weight penalty imposed on the system by electrical power requirements. Fixed weight and consumable weight were the bases for this presentation. Fuel cells were assumed to be the source of water supply for the open system. The weight of the EVA equipment was not included. Results show that selection of an EC/LSS by weight depends upon crew size and mission duration. For tug mission objectives, two or six men for seven days or four men for 45 days, a nonregenerative (open) system seems to be optimum and will be selected. The points of intersection of the open versus closed system plots identifies a boundary where an open system becomes less competitive. For larger crews on longer missions, a closed system presents the minimum weight penalty.

If electrical power were included in the overall weight penalty, the intersection points would move to the right. For tug mission objectives the electrical power requirement was approximately six times higher for a closed system than for an open system.

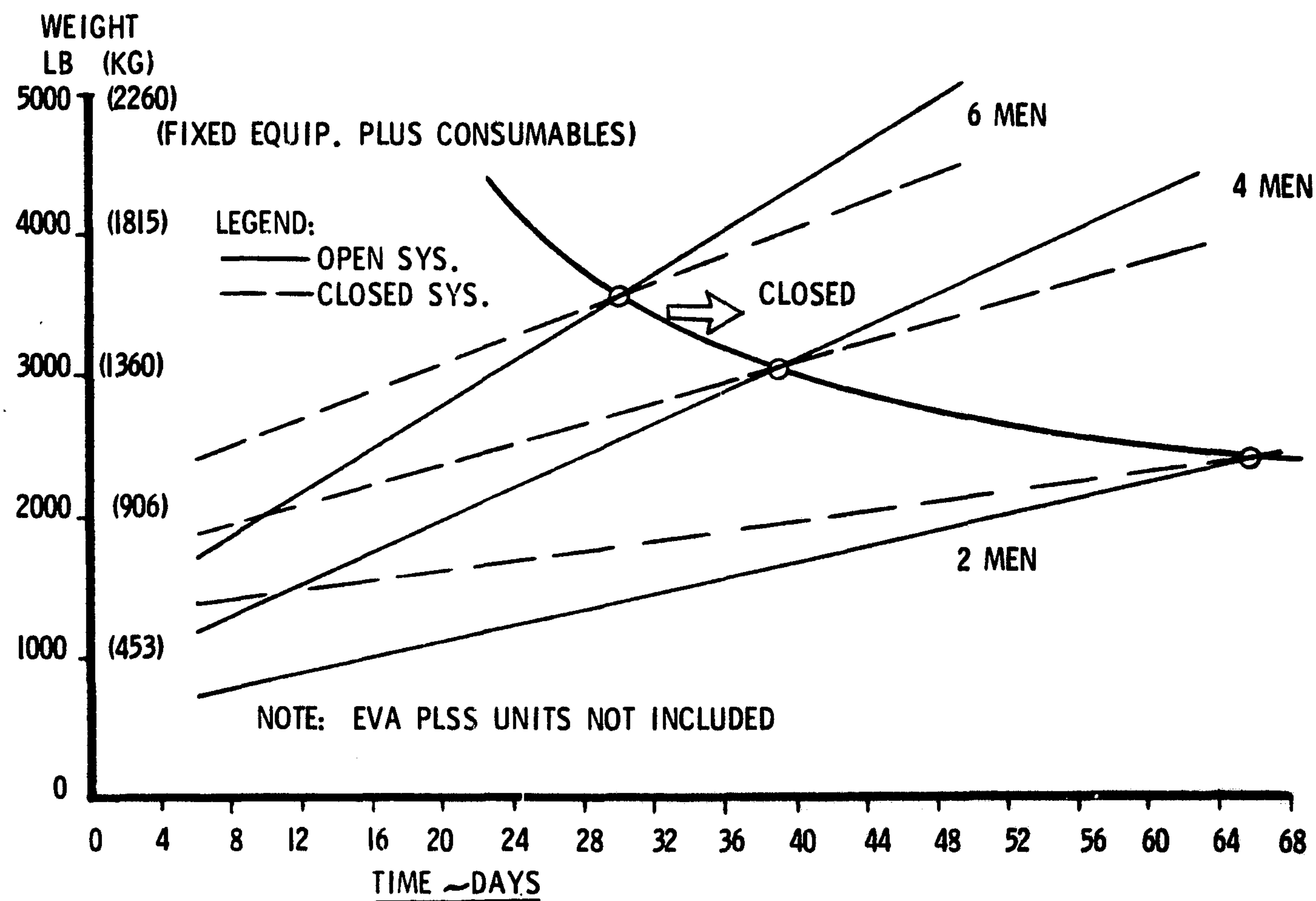


Figure A-44. ECLSS Weight Tradeoff Open Versus Closed System

Crew Accommodations

Habitability will be important in the success of long manned space missions. For the space tug to be habitable, the design must be such that crewmen's physiological and psychological needs are satisfied. In addition, to conduct work efficiently, a crewman requires functional and well organized work spaces.

Numerous space flight simulation studies, operational situations, and hypothetical long-duration space missions have been analyzed to obtain guidelines for estimating man's living space requirements. Celentano efforts on crew living space requirements for long durations have been recognized and were used in this tudy. The Celentano range used for tug is identified as the values of cubic feet per man between optimal and performance limits. For a mission duration of 45 days, the range varies from 420 to 190 ft^3/man. For seven days, an extrapolation was used, and the values range from 120 to 50 ft^3/man.

A tradeoff evaluation was conducted between an EC/LSS open and closed loop system to determine the minimum volume penalty to the vehicle. Figure A-45 shows evaluation results. They indicate that the open-loop system imposed the least volume penalty to the vehicle for tug applications. The closed-loop system required about twice as much volume as did the open loop system. However, the slope of the curves indicate that for longer missions the closed-loop system would have the minimum weight penalty.

In determining the volume allocation of the crew, the following non-EC/LSS systems were used, along with the EC/LSS volume penalty.

System	Volume (ft^3)
Guidance, navigation, & control	6.25
Communications, data management & displays	24.38
Power distribution & conversion	3.0
Total CM astrionics	33.63

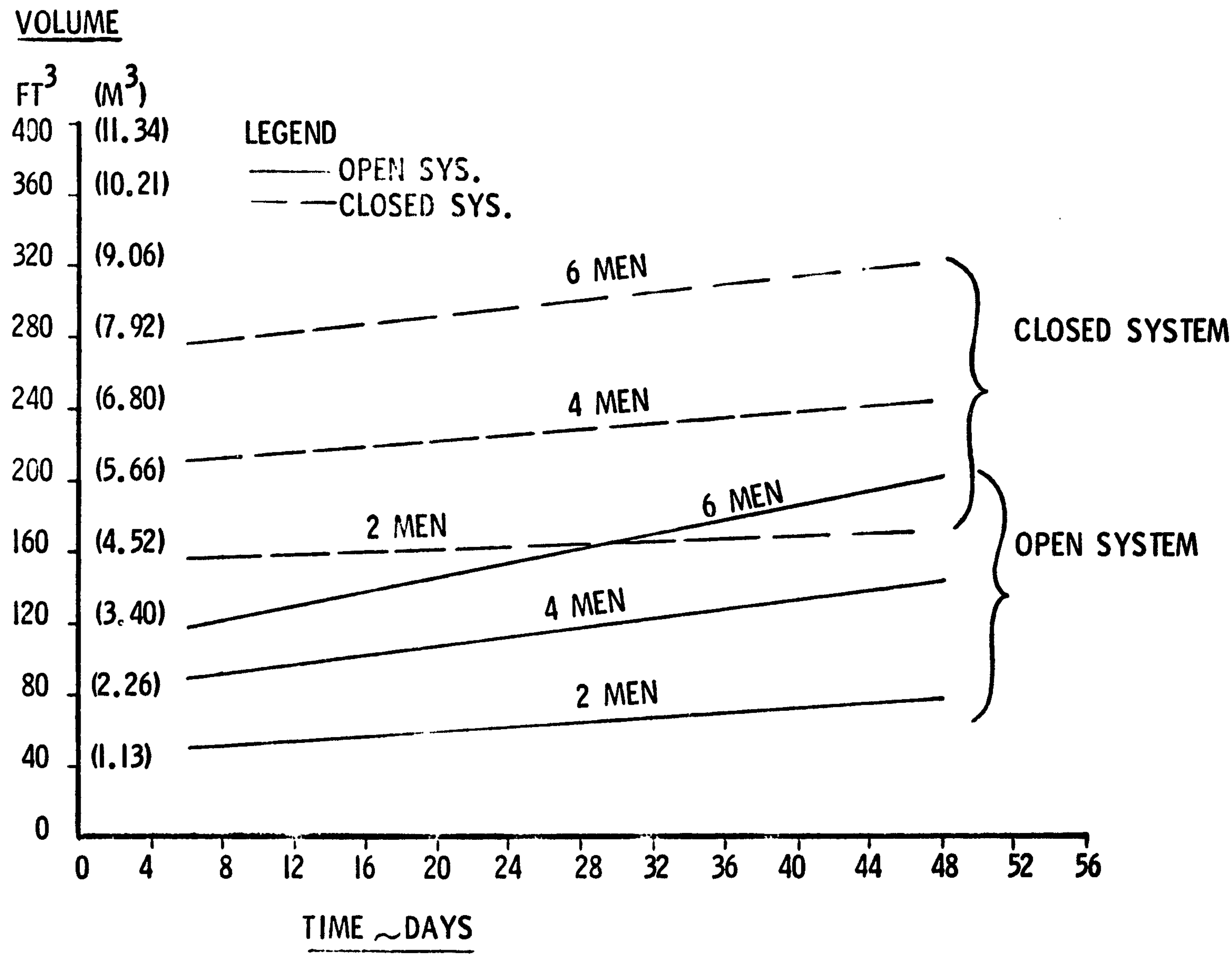


Figure A-45. ECLSS Volume Tradeoff— Open Versus Closed System (Fixed Equipment Plus Consumables)

Crew Furnishings:		Vol (ft^3)	No. of Units	Total Vol (ft^3)	Weight (lb)
1.	Sleeping restraints/bunks	1.35	6	8.1	--
2.	Seating restraints/chairs	3.35	6	20.2	--
3.	Galley	15.0	1	15.0	--
4.	Medical treatment area	12.4	1	12.4	--
5.	Knee space (for work surfaces)	6.6	3	19.8	--
6.	Air lock	42.5	1	42.5	--
	Total crew accommodations			118.0	

Figure A-46 is a bar chart of the volume allocation for the responsible subsystems based upon requirements for a four-man, 45-day mission. The EC/LSS assessment includes volumes for fixed and spares equipment plus expendables and consumables such as food, personel hygiene, medicines, clothing, towels, soap, etc. Crew furniture and command module astrionics volumes are as listed previously.

Figure A-46 also shows a family of curves for the crew available living space as a functions of CM diameter. The Celentano ranges are also identified. The purpose of this evaluation is to determine the optimum CM diameter that will provide an adequate crew living and working space in tug and be within the crew optimal and performance limits.

For a tug mission of four men for 45 days, the curves indicate that CM diameters of 15 feet or two modules of 12 feet will satisfy the crew requirements. The 22-foot CM diameter is too large. For the seven-day mission with six men, the 15-foot CM diameter seems to be best. The other concepts are too large. Therefore, for mission objectives for tug and to satisfy crew requirements, a 15-foot CM is required.

The weight evaluation of the crew furnishing is not included in this report since the weight breakdown was not available. This evaluation will be covered during the Phase A study of the Reusable Space Tug Program.

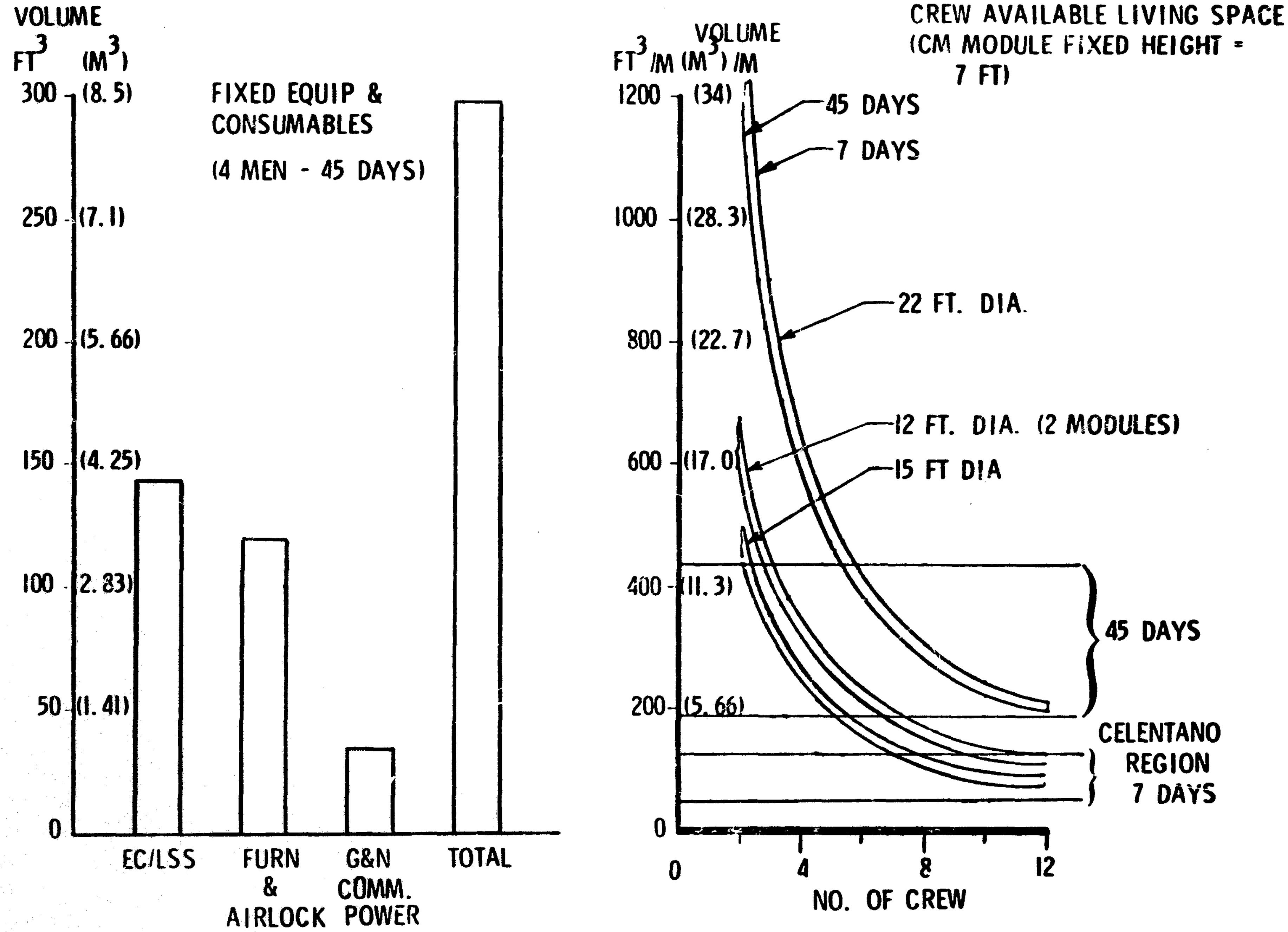


Figure A-46. Volume Relocation

Conclusions and Recommendations

Specific assemblies and associated functions within the EC/LSS were investigated, and trade studies were conducted to determine the best EC/LSS selection based upon evaluation factors of weight, power, and volume scaled down from space station studies. The EC/LSS was evaluated over a range of tug mission objectives to provide support for a two- or six-man seven-day mission capability, with extended system growth for a four-man, 45-day mission,including a 28-day stay on the lunar surface. Results of the tradeoff studies indicate that a nonregenerative (open) system presents the least overall weight, power, and volume penalty to the vehicle. This system selection has the flexibility of a single space tug design, which can effectively accomplish the broad spectrum of the proposed and the potential tug missions defined previously. Based upon the ground rules and assumptions established for this study, the EC/LSS selection that fulfills tug mission objectives is summarized in Tables A-1 and A-2.

Regenerative (closed) loop systems evaluated during this study phase show a high weight and power penalty for tug applications. However, they become competitive for the longer mission stay times and larger crew size.

System selections based upon a weight, power, and volume evaluation must be considered limited. Other factors may have a higher rating when the final design is established. Additional rational factors for the selection of an EC/LSS should be reviewed and their merits evaluated with the result of weight, power and volume before they are committed to a baseline configuration.

Other factors to be considered are start-of-art status, availability, development, reliability, safety, performance, cost, interfaces, and system flexibility.

The following recommendations should be considered during the Phase A Study period for the Reusable Space Tug.

1. Reassess the predicted state of art during the design phase.

2. Establish on a subsystem level the functiona design, including schematics, circuit analysis and controls, component, and hardware identification.

3. Evaluate the advantages and weight savings of collecting CO_2 aboard the vehicle and then processing this gas after returning to the space station.

4. This study was based upon evaluations factors of weight, power, and volume in the selection of the EC/LSS, but the relative importance of other rational factors in the overall selection should be considered. Weighted values should be given to the evaluating factors, and their order of priority should be established.

EC/LSS Parametric Data for Evaluating Regenerative Systems

Presented in this section are parametric data to be used for the selection of regenerative systems related to the EC/LSS. The major system candidates that can be evaluated by the use of these parametric curves are in the area of CO_2 removal, CO_2 reduction for water recovery, water electrolysis for O_2 recovery, trace contaminant control and water reclamation. The curves are shown in Figures A-47 through A-60.

The assumptions and rational that underly development of the accompanying parametric data are noted here:

1. Space station parametric data (reference A-2), which were based on process rates, were converted into crew size and mission duration.

2. System fixed weight and electrical power penalties were assumed to be directly proportioned to crew size.

3. System spare weight was assumed to be directly proportional to mission time and presented as a percentage of the system fixed weight.

4. System expendable weight was assumed to be a function of man-days.

5. Volume penalties were obtained from computed density factors based upon space station subassemblies weight and volume. These density factors are listed in Table A-21 of the report.

The parametric data generated herein are considered relative, based upon the authenticity of their source. They are intended to be used as tools only for the evaluation and the selection of regenerative processes. These data can be used over a wide range of crew size and mission durations such as 0 to 12 crew members and mission durations up to 180 days.

Specific component or hardware weight and volume cannot be obtained from this report because the data are only for evaluation and tradeoff studies at the subsystem or assembly level.

Curves for the assessment of weight of secondary subsystems are scaled linearly with crew size. The subsystems included in these curves are circulation, temperature and humidity control, pressure control, cabin thermal loop, wash and condensate recovery, water storage and control, waste management, emergency oxygen system, EVA life support, and food and water management.

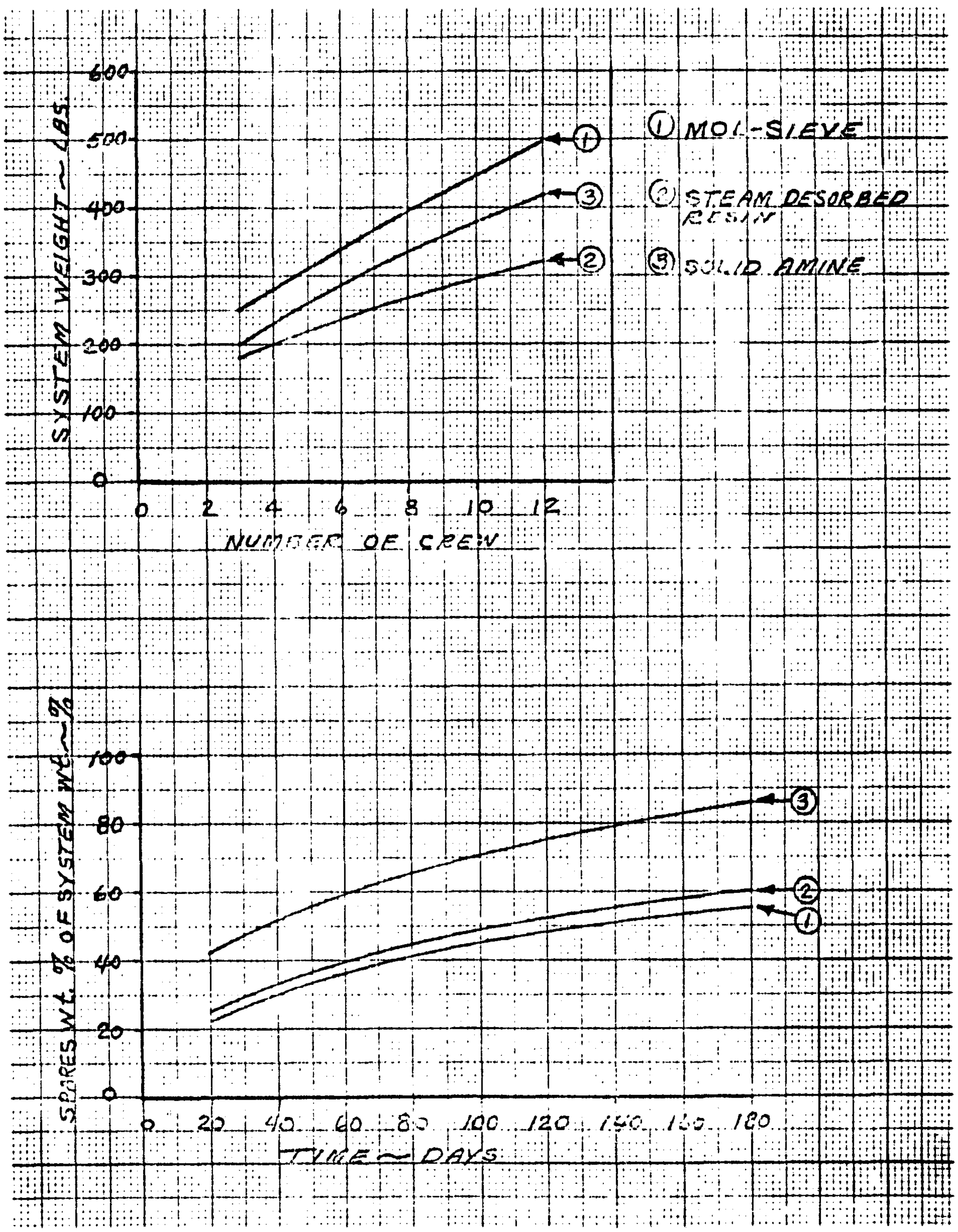


Figure A-47. CO_2 Removal - System Weight Trade Study Closed Loop System

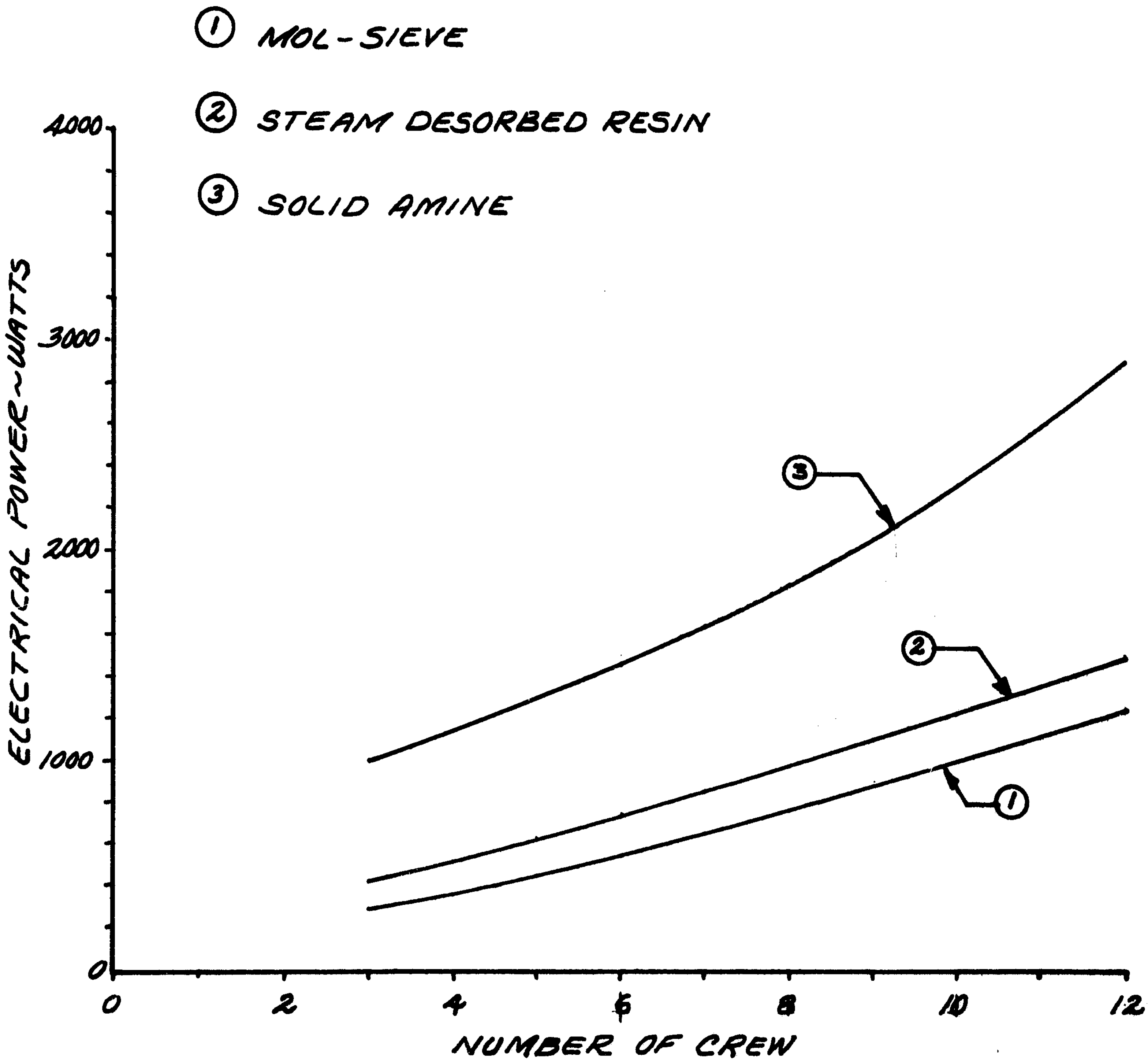


Figure A-48. CO_2 Removal—Electrical Power Penalty

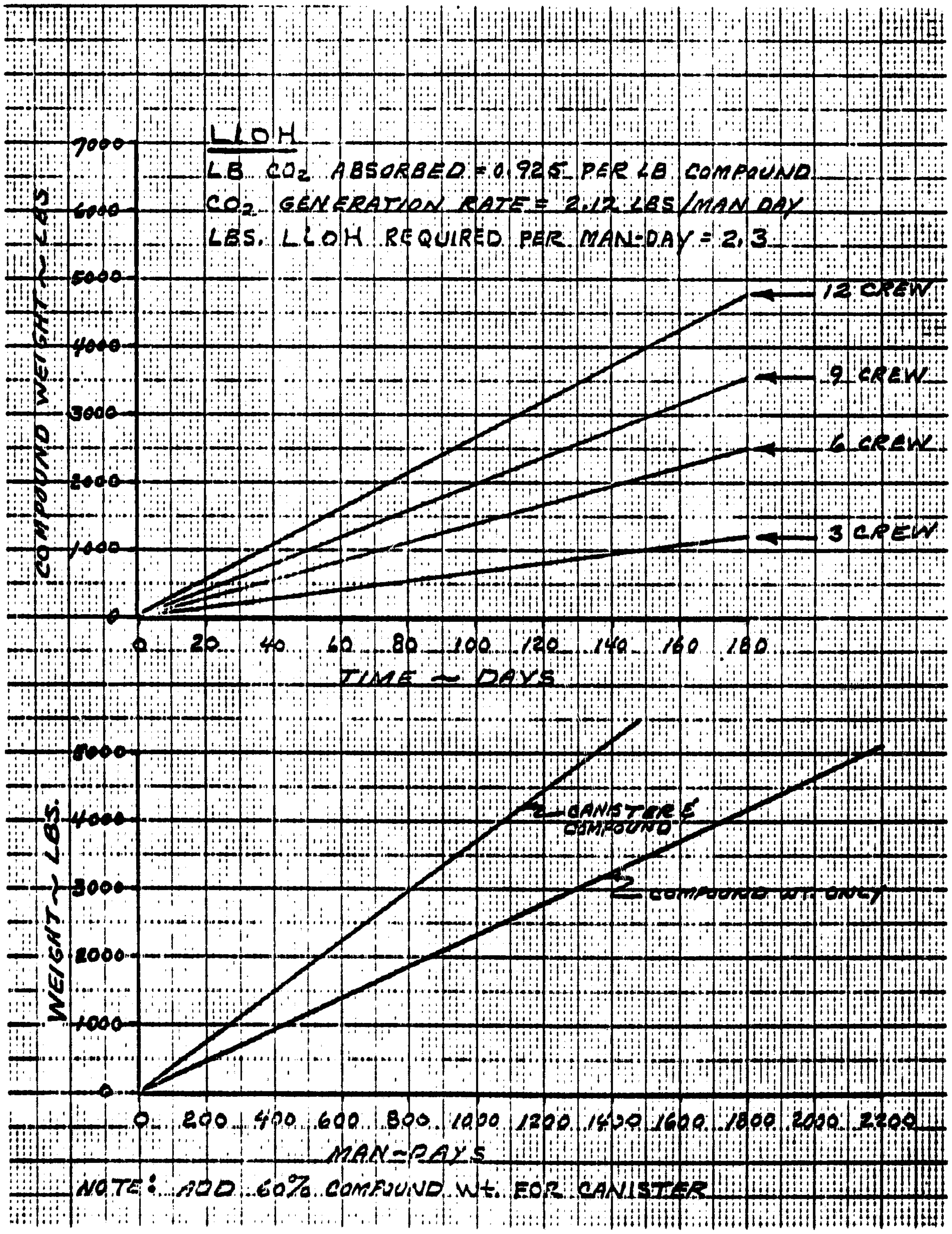


Figure A-49. CO_2 Removal - System Weight Trade Study Open Loop System

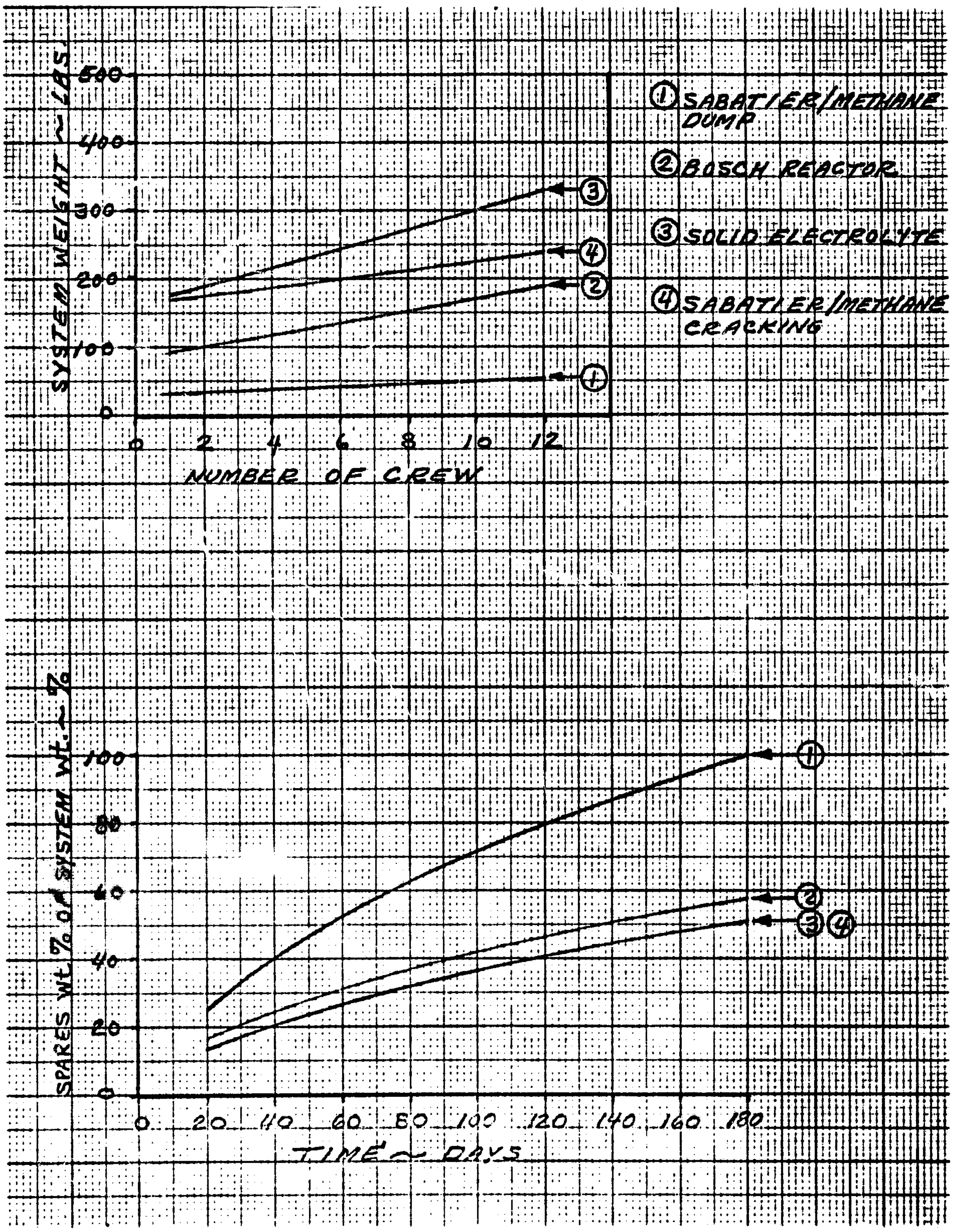


Figure A-50. CO_2 Removal - System Weight Trade Study Closed Loop System

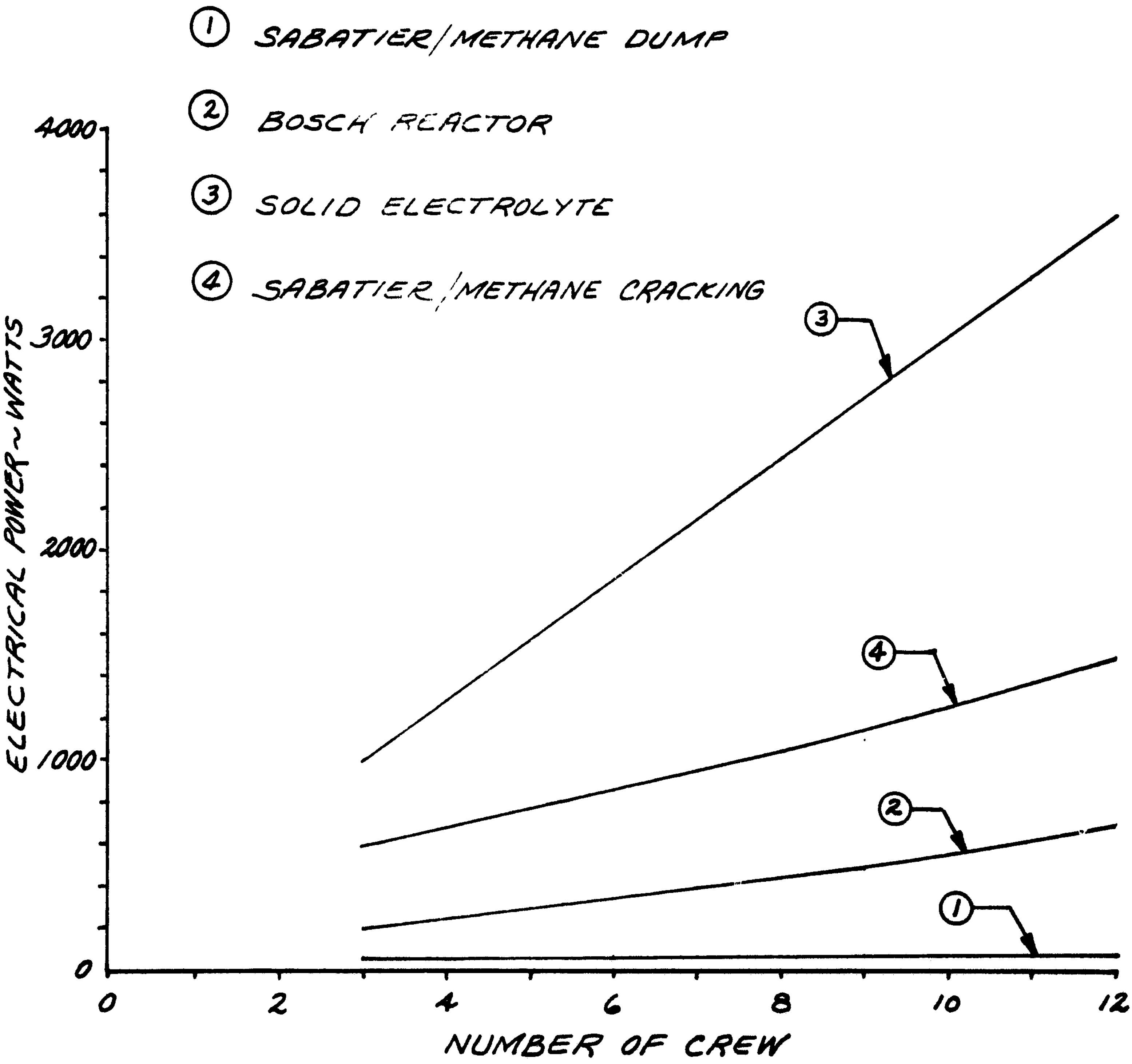


Figure A-51. CO_2 Reduction—Electrical Power Penalty

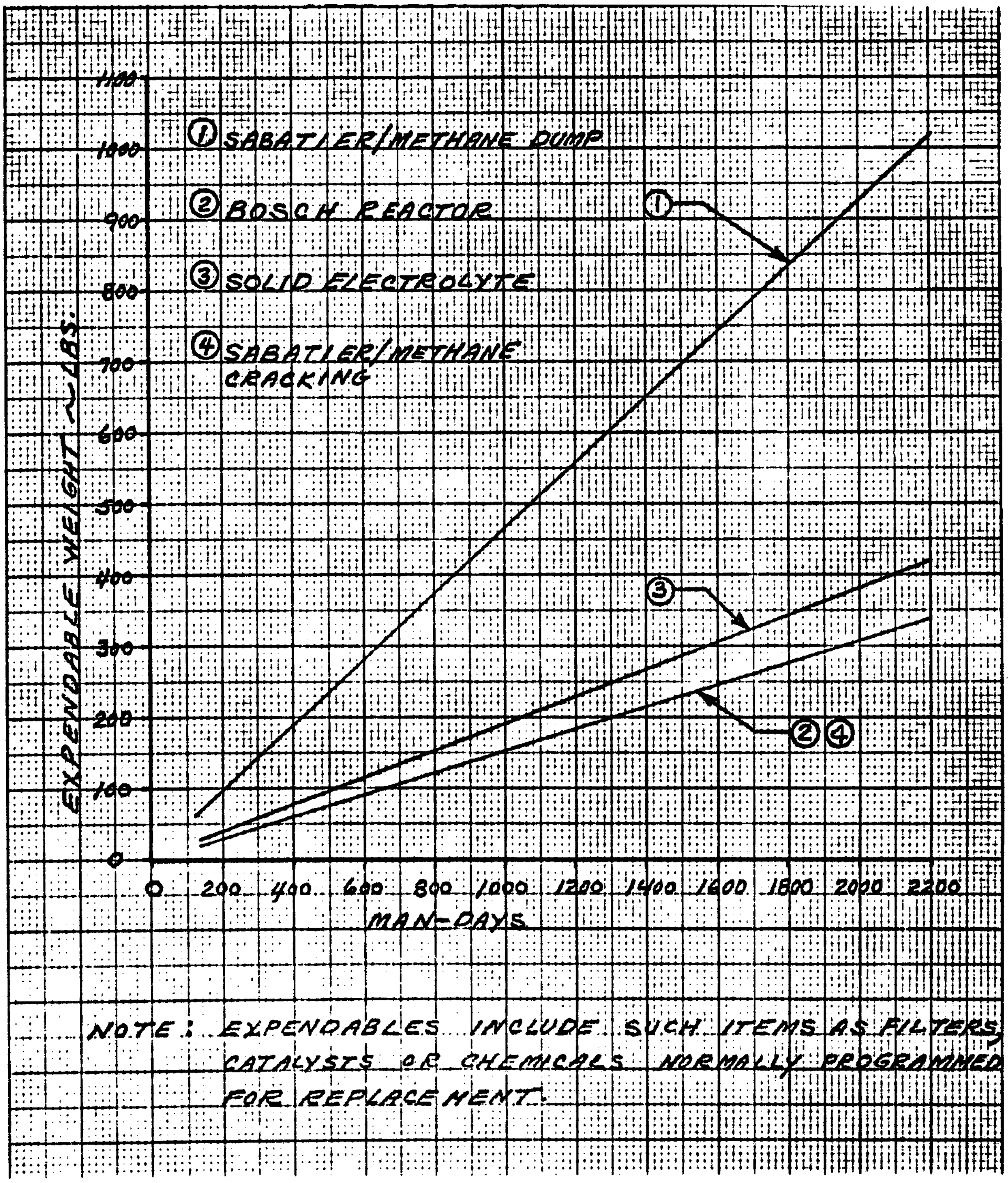


Figure A-52. CO_2 Reduction - System Weight Trade Study Closed Loop System

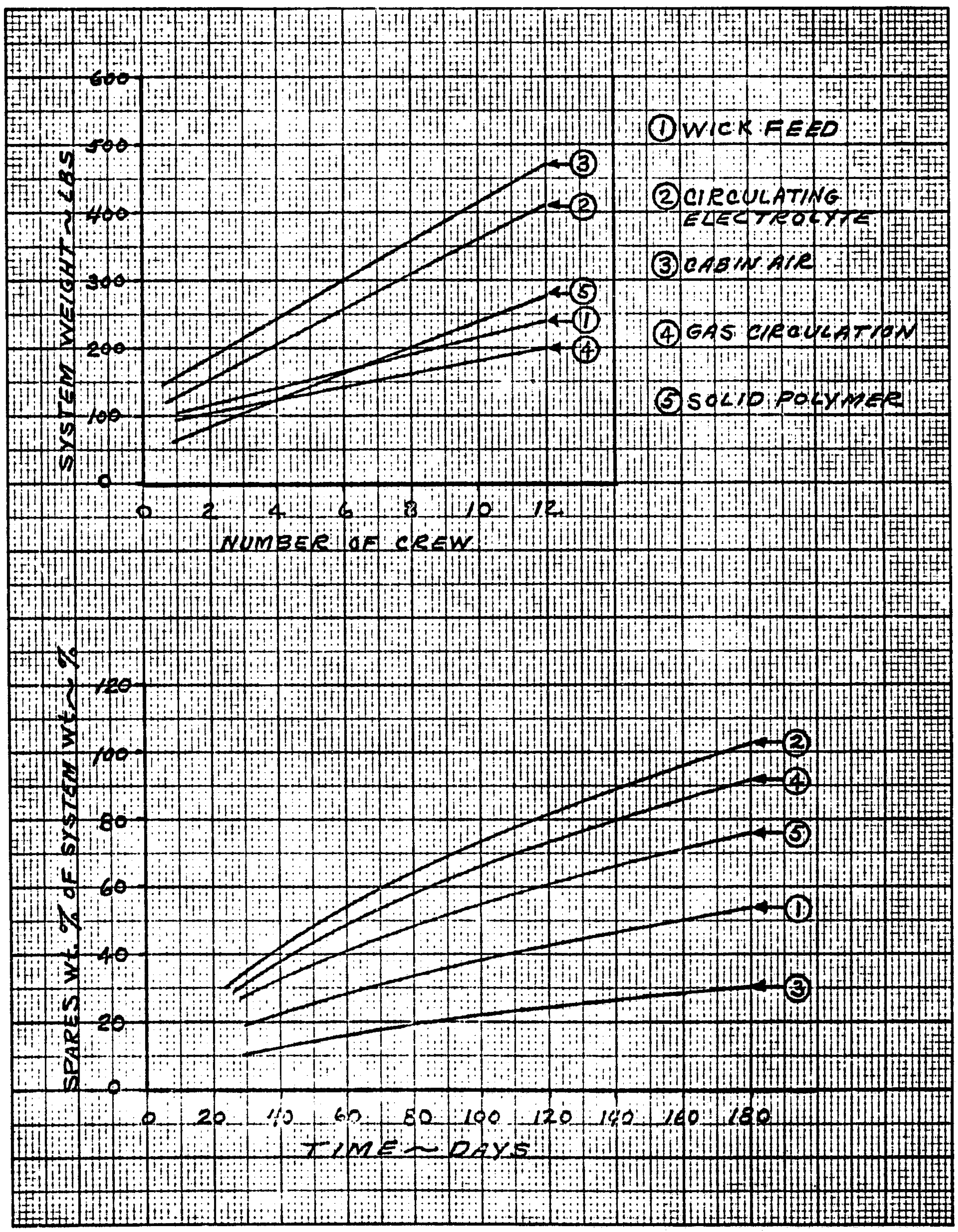


Figure A-53. Water Electrolysis System - Weight Trade Study Closed Loop System

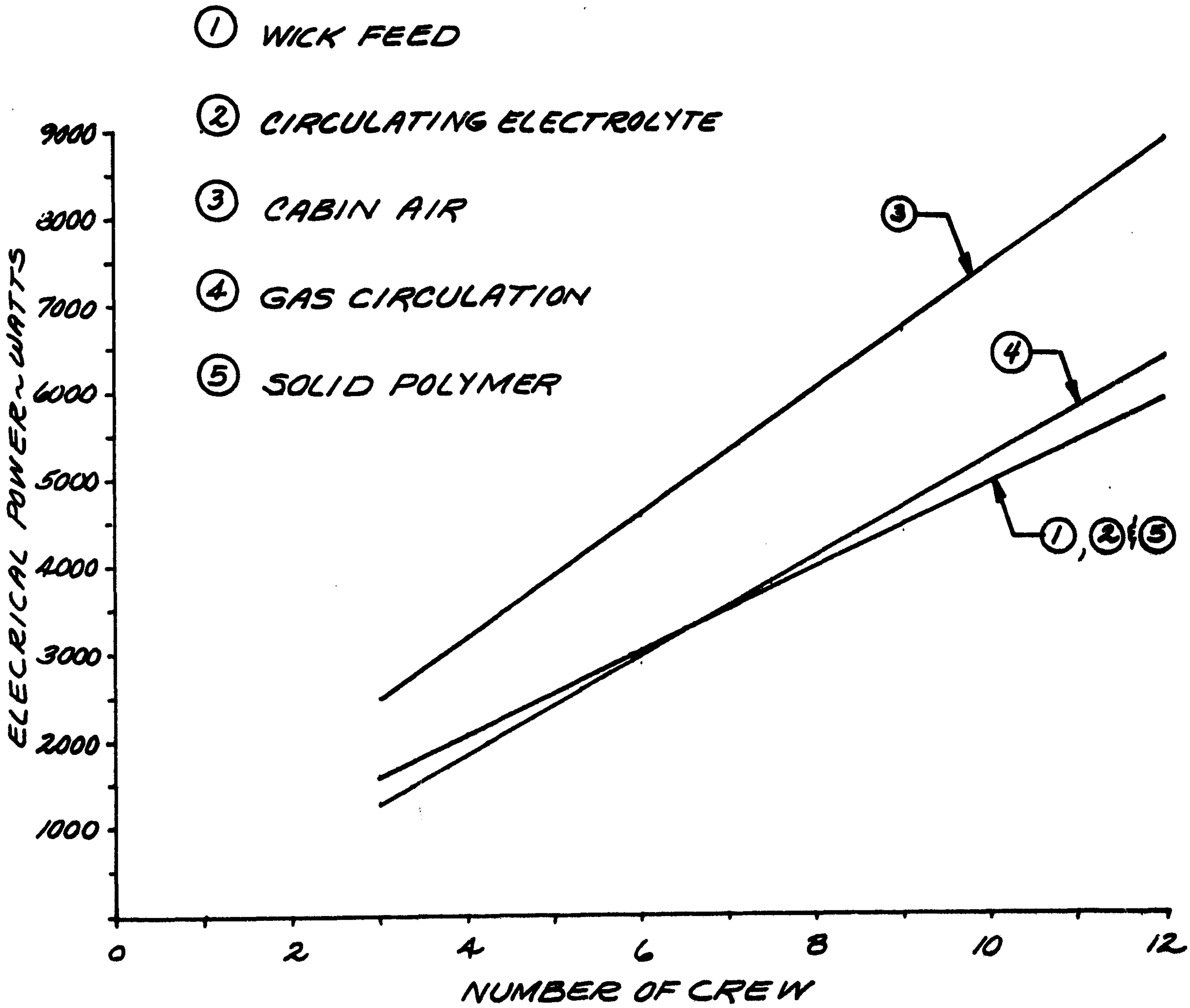


Figure A-54. Water Electrolysis—Electrical Power Penalty

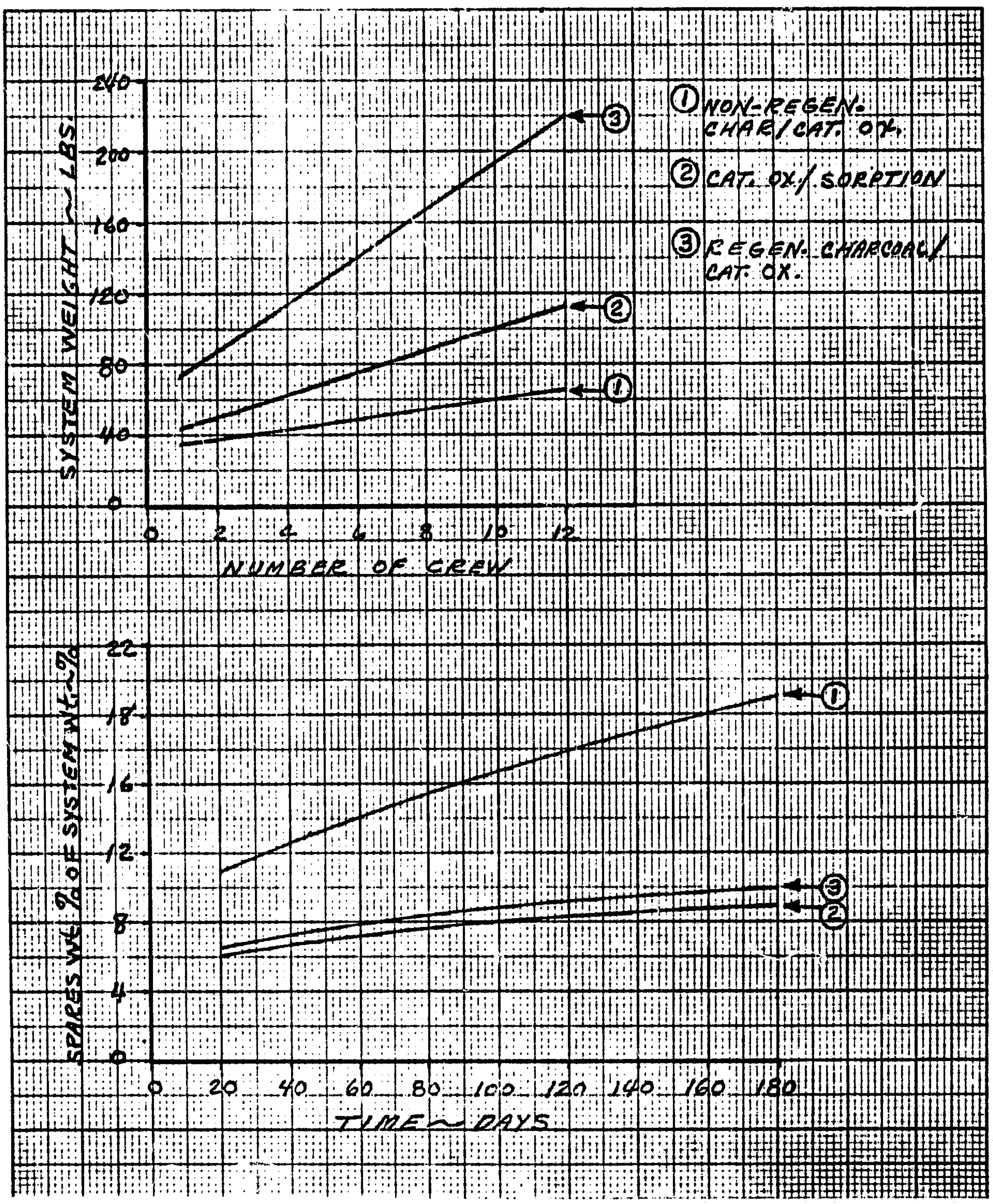


Figure A-55. Contaminant Control - System Weight Trade Study Closed Loop System

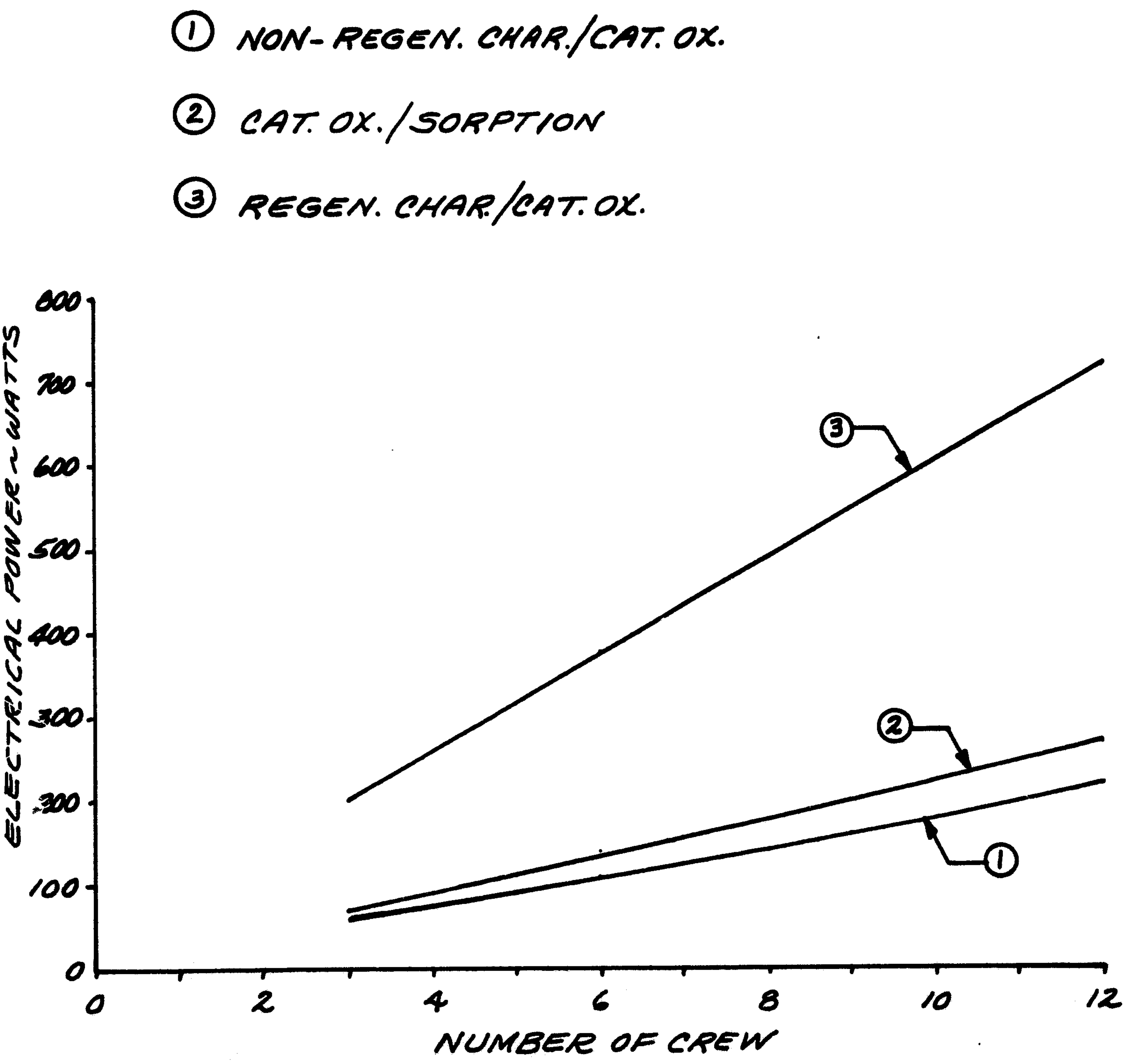


Figure A-56. Contaminant Control - Electrical Power Penalty

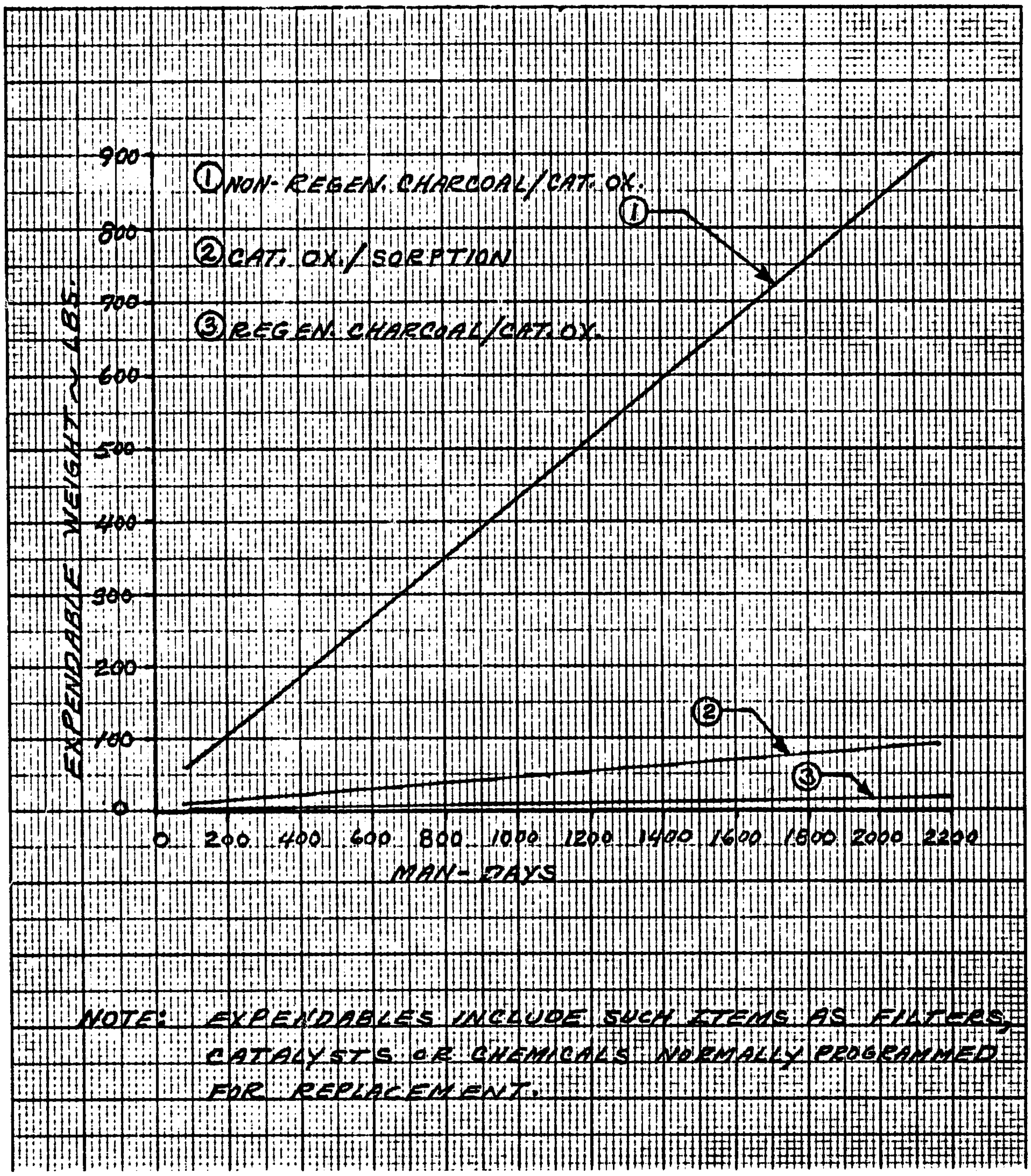


Figure A-57. Contaminant Control - System Weight Trade Study Closed Loop System

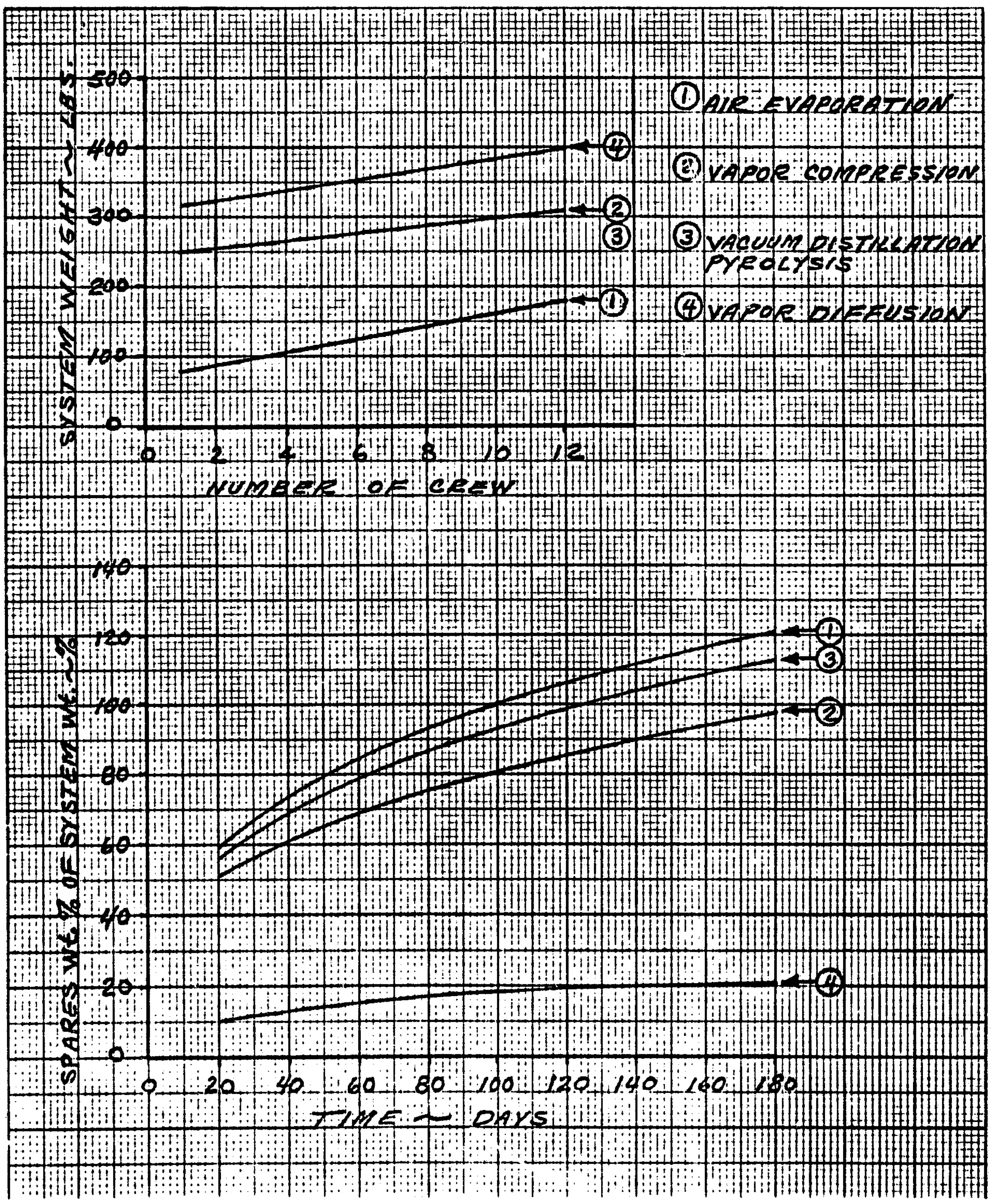


Figure A-58. Water Reclamation - System Weight Trade Study Closed Loop System

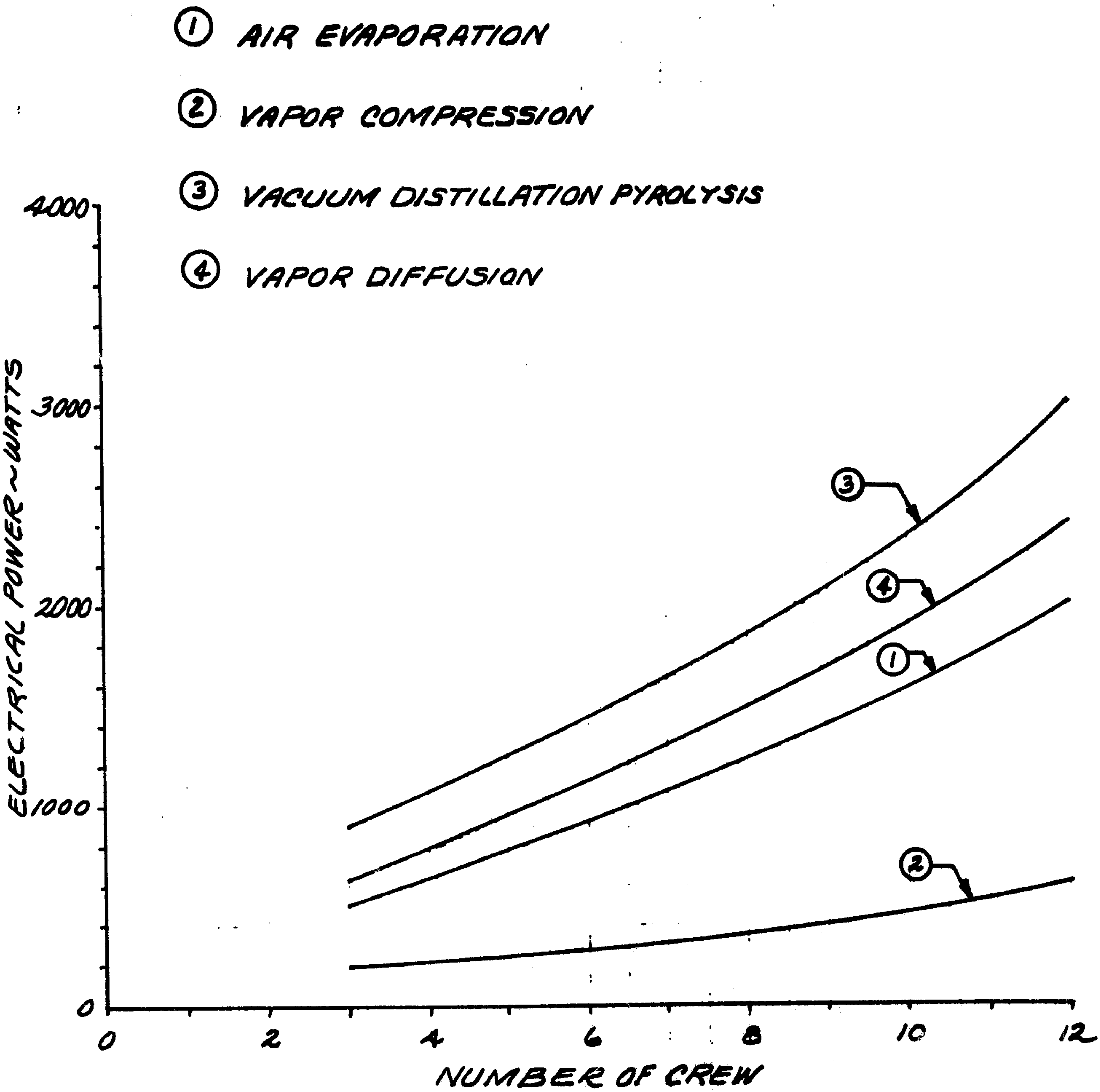


Figure A-59. Water Reclamation—Electrical Power Penalty

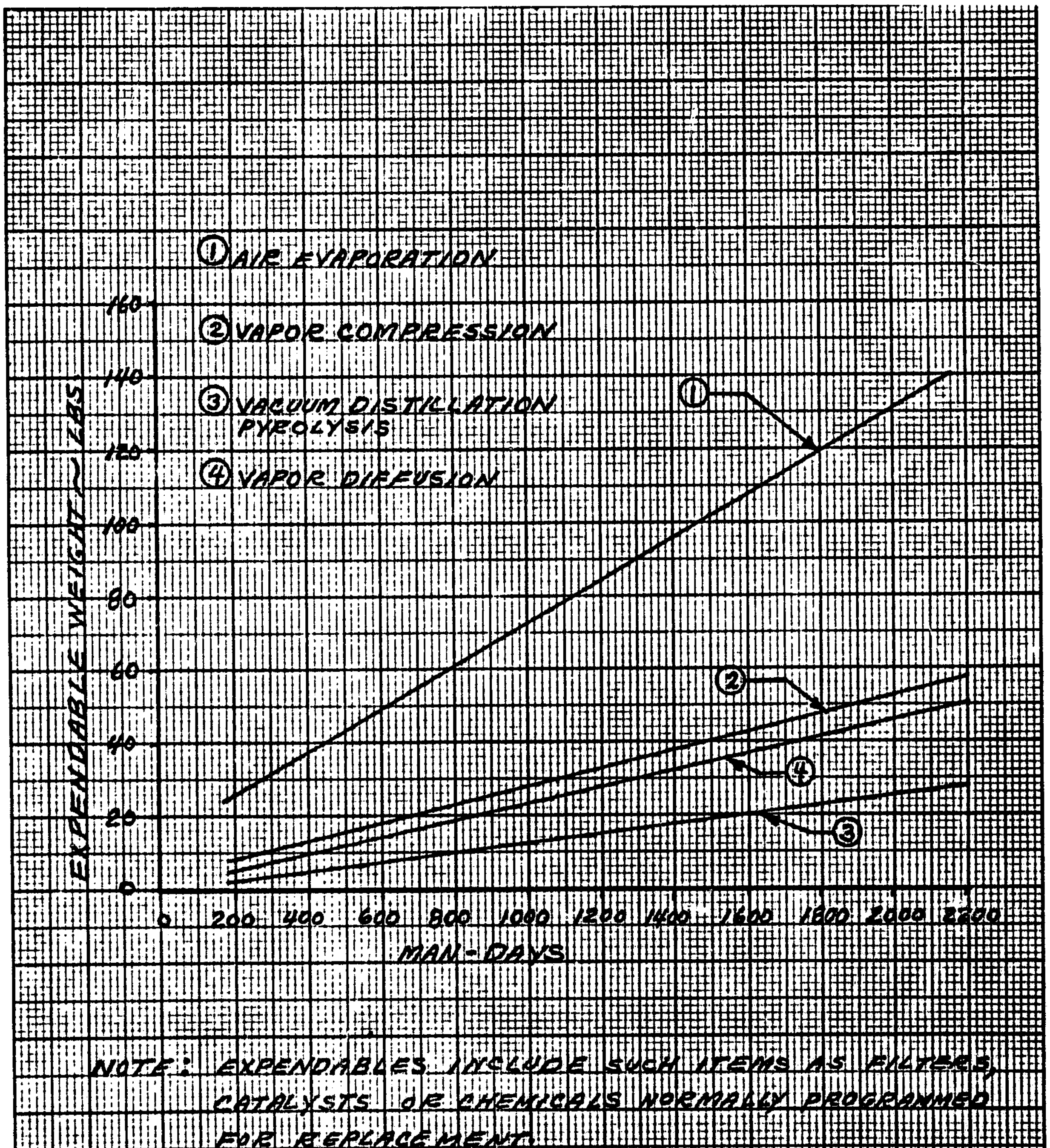


Figure A-60. Water Reclamation - System Weight Trade Study Closed Loop System

APPENDIX B

ACTIVE THERMAL CONTROL
TRADE STUDY

APPENDIX B

ACTIVE THERMAL CONTROL TRADE STUDY

Active thermal control is concerned with the transport of waste heat from sources inside the spacecraft (electronic equipment coldplates, fuel cells, crew members, lights, and other equipment) to a suitable heat sink. The latter is generally deep space, although the latent and sensible heats of expendable fluids may also be used, especially during missions of relatively short duration. Specific items of thermal control equipment applicable to this study and discussed in detail are (1) space radiators, (2) heat pumps, (3) water boilers and sublimators, and (4) GH_2 heat exchangers. A preliminary investigation of the feasibility of using an absorption refrigeration cycle is also reported.

RADIATORS

Radiators generally perform the function of rejecting heat to the deep space heat sink by means of radiation heat transfer. The performance of this function is affected by the temperature level at which the radiator is required to operate and by the external thermal environment to which the radiator surface is exposed. The operating temperature level for the radiator is determined by the requirements for specific heat transport fluid temperatures necessary for thermal control and by the effects on the surface of the external thermal environment. These effects can be controlled to some extent by selecting radiator surface thermal control coatings that display desirable ratios of α_s/ϵ (solar absorptivity/infrared emissivity). How the variation in the α_s/ϵ ratio affects radiator heat rejection is illustrated in Figure B-1 for two values of α_s and two different thermal environments. The lower of the two absorptivities (0.18) represents the characteristics of the thermal control coating used on the surface of the Apollo ECS radiator panels at the time of application, while an absorptivity of 0.3 represents a degraded value for the same zinc-oxide potassium-silicate paint. The degradation in the value of α_s is believed to be caused primarily by proton bombardment, and must be taken into account in the design of radiator systems that operate over long periods in environments that are likely to produce such degradation.

Figure B-1 indicates that, for a solar oriented radiator, a low value of α_s is desirable in order to maximize heat rejection from a given area. It may also be seen that radiator surface temperatures in excess of 135 F are

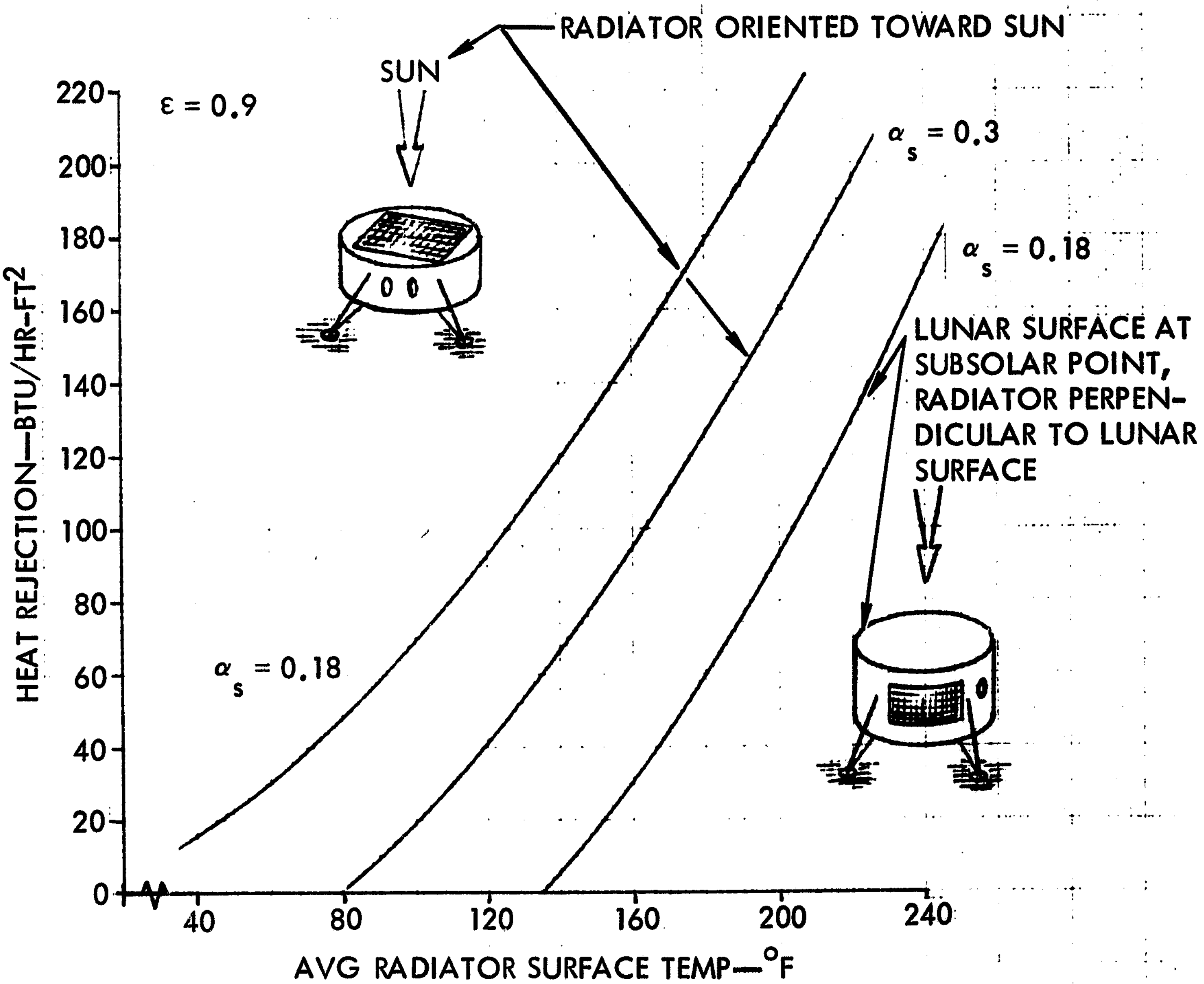


Figure B-1. Radiator Heat Rejection Capability required in order to reject heat from a vertical radiator on the lunar surface, even if the radiator is not subject to direct solar irradiation (at the subsolar point).

The use of a vertical radiator on the lunar surface is complicated by the fact that the radiator "sees" the lunar surface equally well as it "sees" space. Figure B-2 presents heat rejection capability versus surface temperature for various sun zenith angles for a white surface finish (magnesium oxide). It can be seen that maximum heat rejection capability during the lunar day occurs when the sun is on the horizon, and minimum heat rejection capability occurs when the sun is directly overhead. The importance of proper surface finish becomes evident when Figure B-3 is investigated. This graph presents heat rejection capability as before, except the radiator surface is assumed to have a finish with properties similar to black paint (solar absorptivity = 0.9, infrared emissivity = 0.9). In this case, maximum heat rejection capability is considerably lower than before and occurs when the sun is directly overhead. Minimum heat rejection capability occurs when the sun is close to the horizon. In both cases, however, the radiator surface temperatures necessary for heat rejection are well above the usual temperature levels required for spacecraft thermal control.

The radiator performance shown in Figures B-2 and B-3 leads to the conclusion that the effect of thermal radiation from the moon's surface poses a more serious problem than direct radiation from the sun. It follows, then, that a radiator designed to operate on the lunar surface should be oriented parallel to the lunar surface and insulated as much as possible from this surface. The performance of a radiator surface oriented and insulated in this manner is shown in Figure B-4 as a function of surface temperature and for various sun zenith angles. The surface properties (α_s = 0.18, ϵ = 0.95) applicable to the performance characteristics shown are almost identical to those for the zinc-oxide potassium-silicate paint mentioned above; and they are considered to be usable in preliminary trade-off studies without adjustment.

Figure B-5 illustrates the effect of radiator surface temperature on radiator area relative to the area required at 40 F. For example, a radiator operating at an average surface temperature of 80 F will require only 33 percent of the area required for a 40 F radiator when both are rejecting the same heat load. This area-temperature relationship is shown in Figure B-5 for the same radiator paint mentioned earlier (α_s = 0.18, ϵ = 0.9) and for a solar oriented radiator panel. It is evident from this graph that radiator operation at higher temperature levels is desirable from the standpoint of reducing radiator surface area requirements.

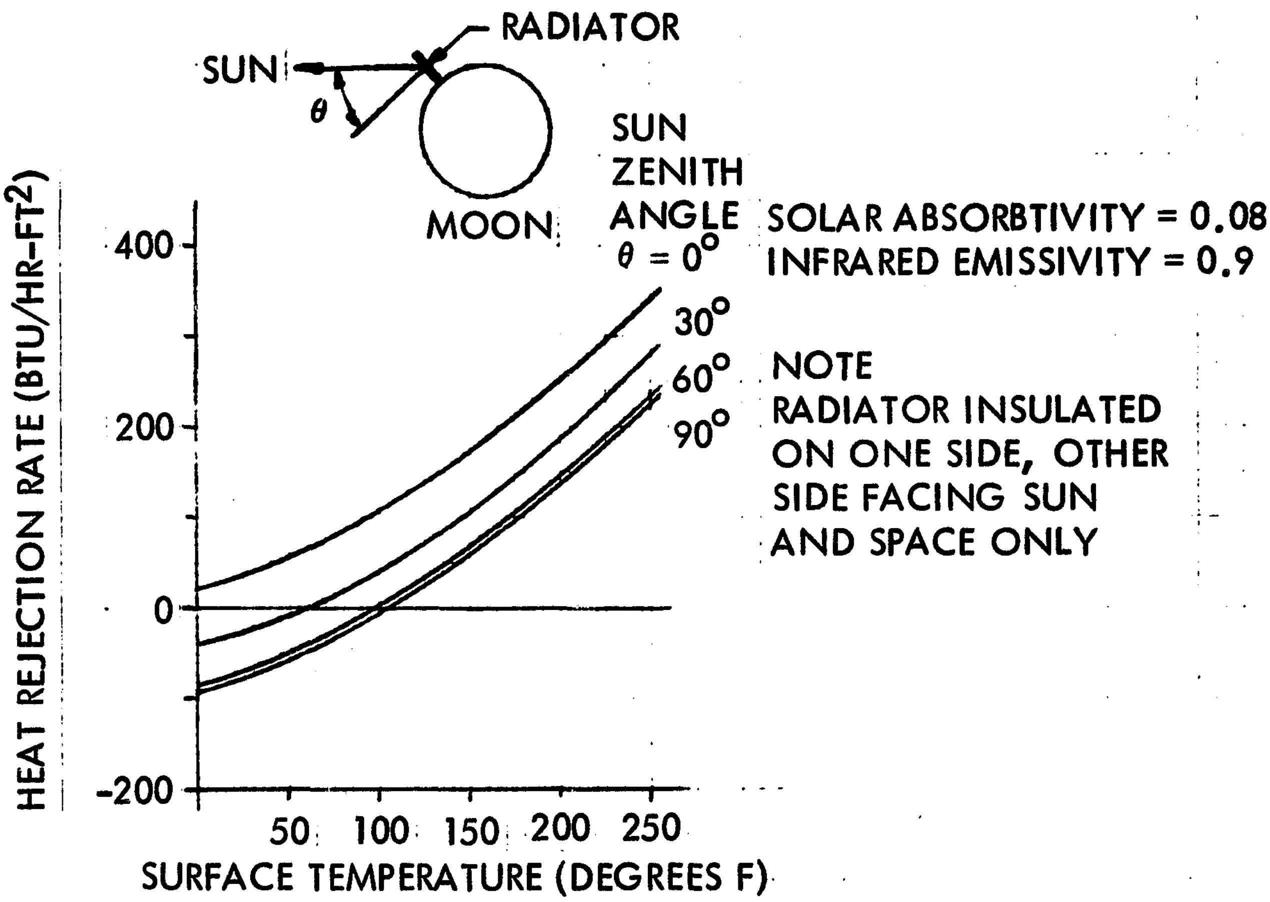


Figure B-2. Vertical Radiator With Magnesium Oxide Coating; Heat Rejection Rate Versus Surfaces Temperature for Various Sun Zenith Angles

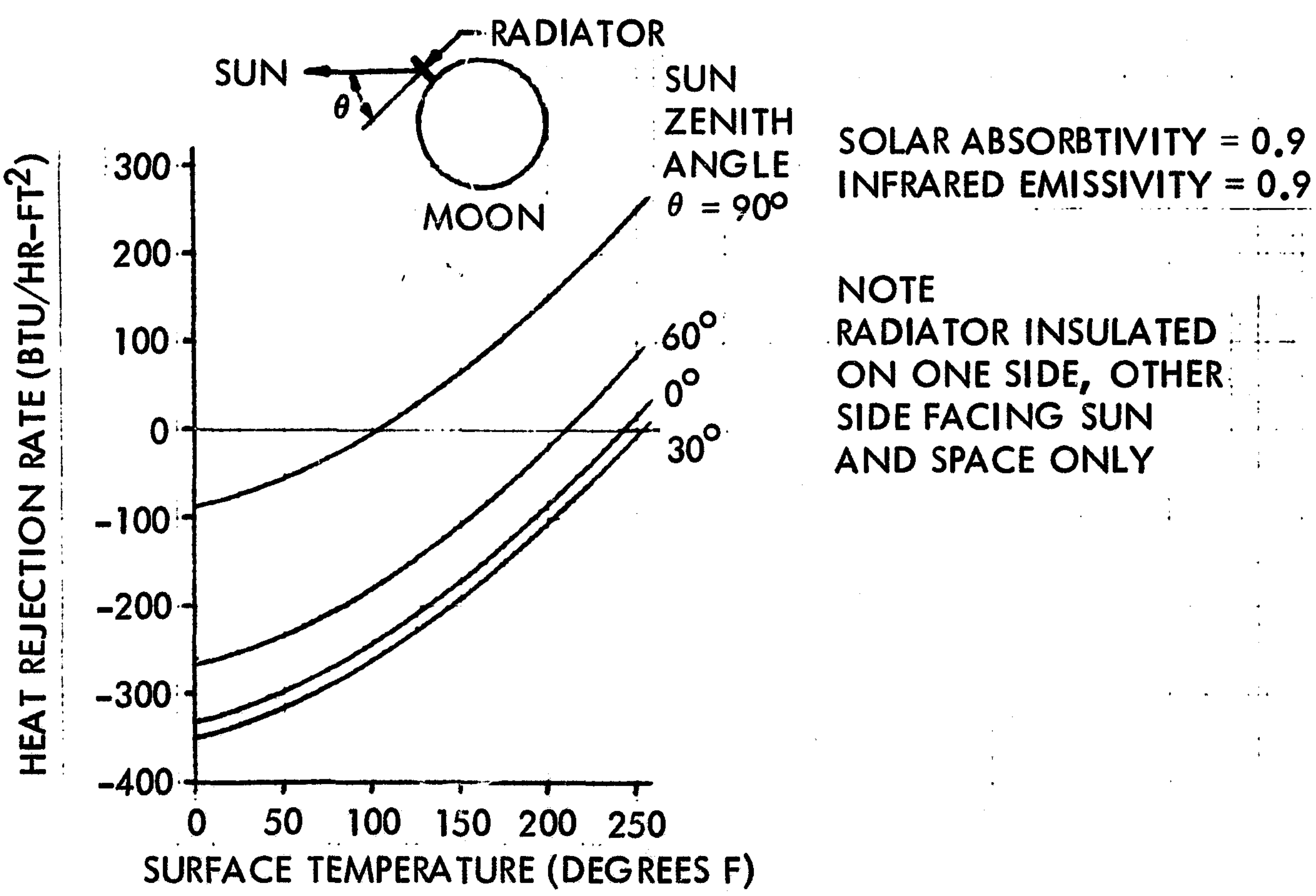


Figure B-3. Vertical Radiator With Surface Absorbtivity and Emissivity = 0.9; Heat Rejection Rate Versus Surface Temperature for Various Sun Zenith Angles

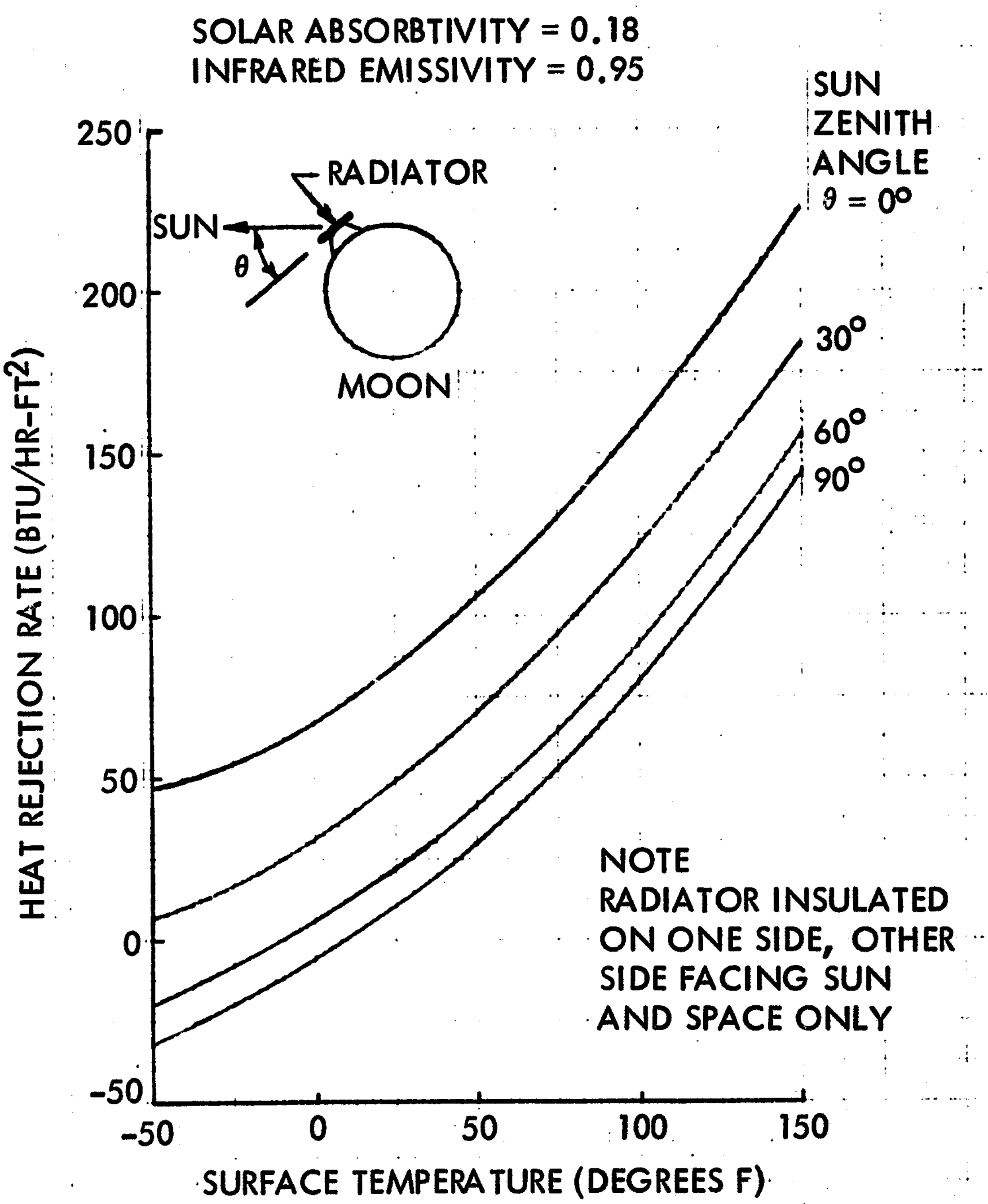


Figure B-4. Flat Radiator Painted White; Heat Rejection Rate Versus Surface Temperature for Various Sun Zenith Angles

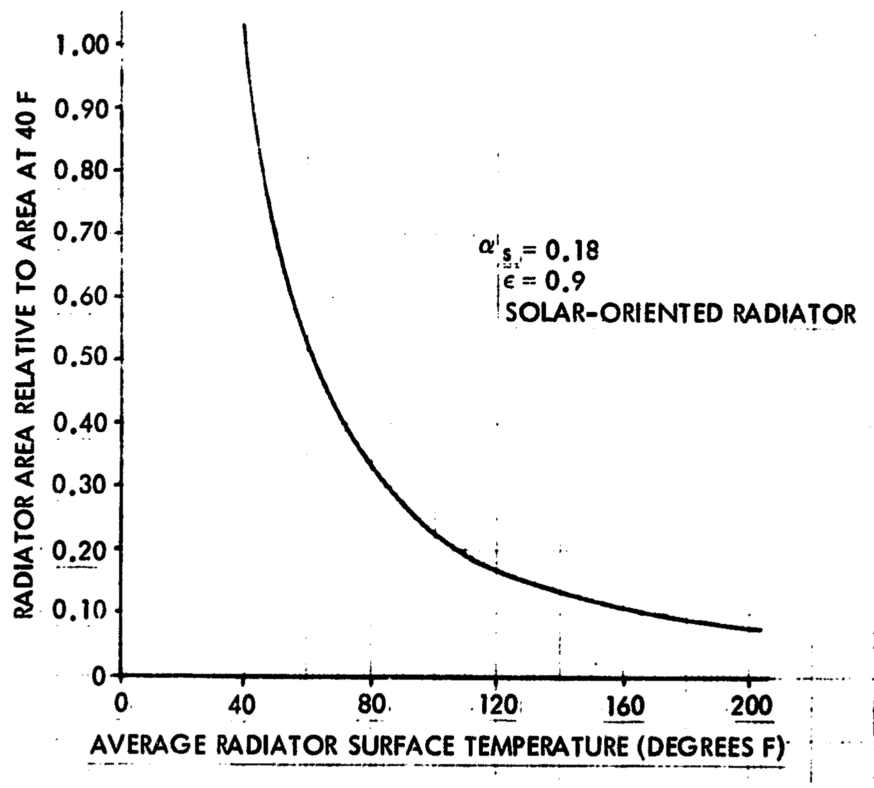


Figure B-5. Relative Radiator Area Requirements

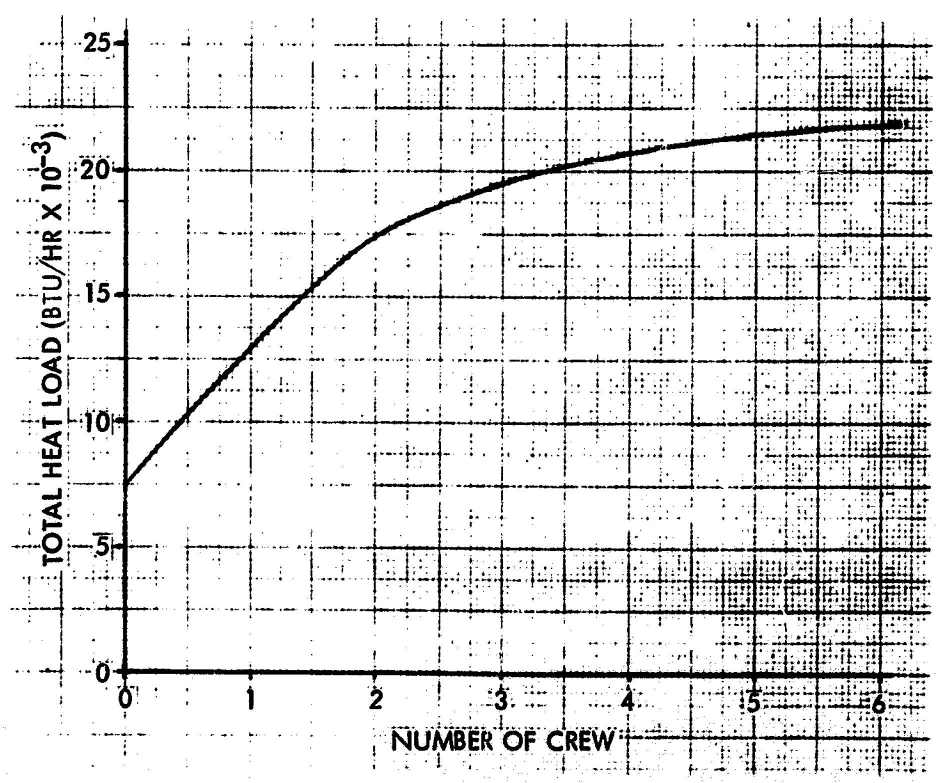


Figure B-6. Total Heat Load Versus Crew Size

In order to use radiator performance data in the calculation of required radiator surface area, it is necessary to establish the range of heat loads that will be considered in the various trade-off studies. This heat load range is shown in Figure B-6 as a function of crew size for the tug, and was calculated under the following assumptions:

Crew metabolic heat load = 500 Btu/hr per man

Electrical heat load = 0.657 kw/kw generated

Unmanned mission load = 1.33 kw

Two-man mission load = 2.91 kw

Four-man and six-man mission load = 3.33 kw

For a two-man crew, for example, these assumptions result in a total heat load of

$$(2 \times 500 \text{ Btu/hr}) + (1.657 \times 2.91 \text{ kw} \times 3412 \text{ Btu/kw-hr}) = 17{,}400 \text{ Btu/hr}$$

Note that the 1.657 represents the inefficiency of power generation plus the requirement to dissipate the total usable power.

Radiator area requirements for solar orientation and covering the heat load range of Figure B-6 are shown in Figures B-7, B-8, and B-9 with radiator surface temperature as a parameter. These area requirements were calculated for both virgin and degraded thermal coating properties (α_s = 0.18 - 0.3); and for convenient identification of specific mission requirements, vertical lines representing various crew sizes have been added to the graphs. Examination of these graphs, specifically Figure B-9, reveals that the effect of α_s degradation on required radiator area becomes less pronounced as radiator operating temperature is increased. It should also be noted that the radiator weight penalty shown in Figure B-9, calculated at the rate of 0.2 pound per square foot, represents only the difference in weight between a spacecraft outer skin panel incorporating a fluid radiator and a panel without such radiator provisions.

Estimates of weight and power requirements for a coolant circulating system associated with a space radiator (coolant pumps, plumbing, cold-plates, heat exchangers, valves, and control elements) are shown in Figure B-10 as a function of heat load. These estimates are based on the Apollo system, which weighs 185 pounds (exclusive of radiators) and handles an average heat rejection load of 5000 Btu/hr in earth orbit. The electrical power requirements of this system are approximately 60 watts for operation of a coolant circulating pump and a control valve system.

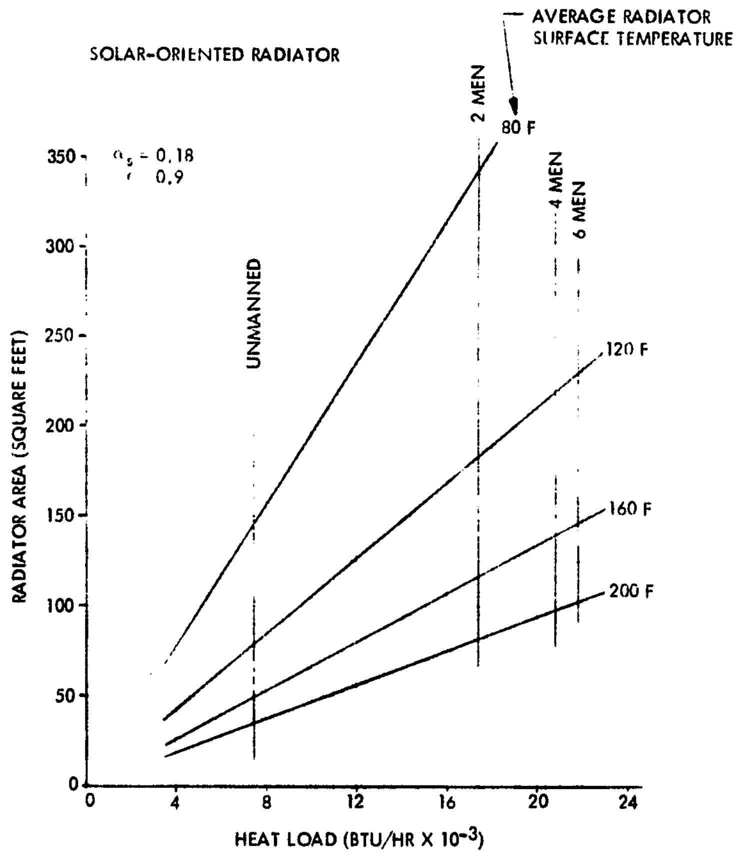


Figure B-7. Radiator Area Versus Heat Load—Solar-Oriented Radiator (Average Surface Radiator Temperature 80F)

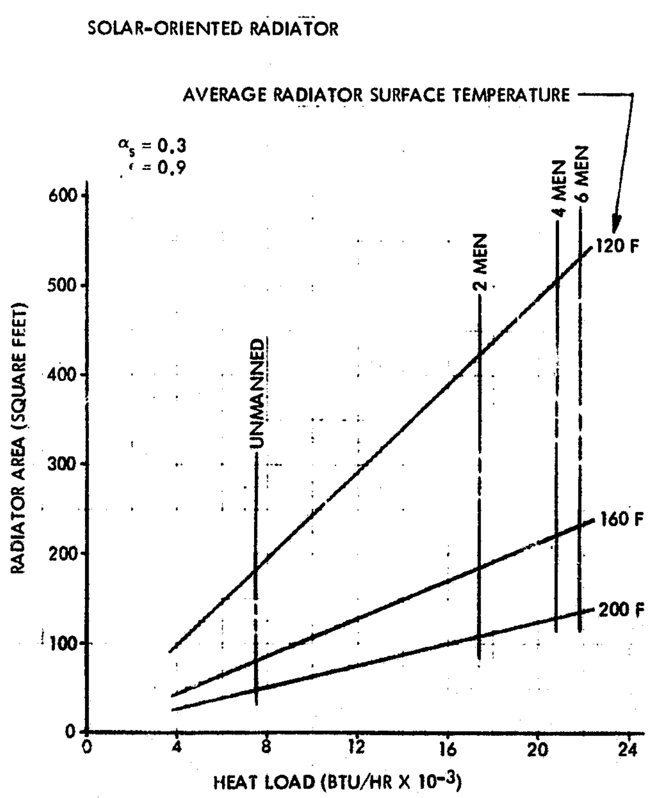


Figure B-8. Radiator Area Versus Heat Load—Solar-Oriented Radiator (Average Radiator Surface Temperature 120 F).

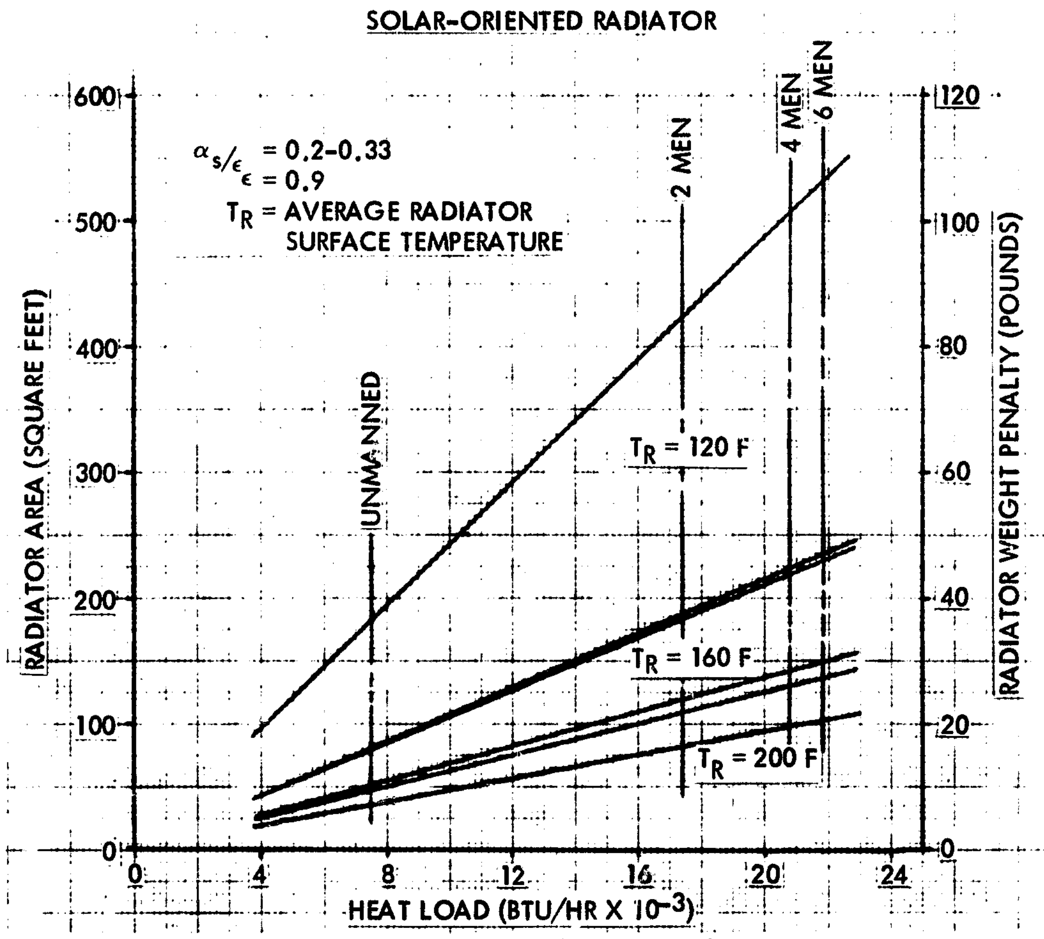


Figure B-9. Radiator Area and Weight Penalty Versus Heat Load—Solar-Oriented Radiator

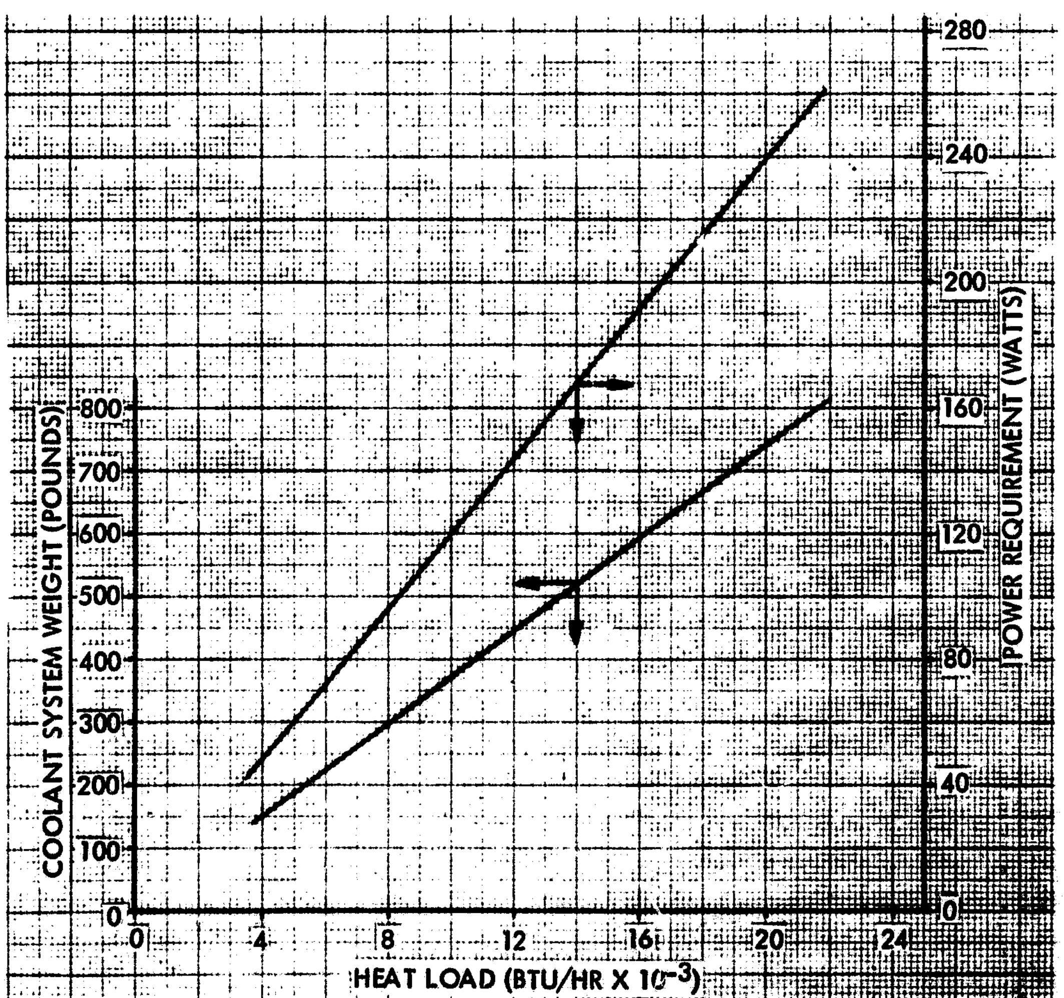


Figure B-10. Estimate of Coolant System Weight and Power Requirement—Radiator Weight Not Included

Earlier in this discussion on radiator system characteristics, it was mentioned that radiator surface thermal control coatings are subject to degradation in the value of α_s. The magnitude of this degradation as a function of time is shown in Figure B-11 (Reference 2-1) for ARF-2, which is the same zinc-oxide potassium-silicate paint discussed previously. This curve indicates that α_s will degrade to a value of 0.3 (α/α_o = 1.65) after an exposure of approximately 3500 sun hours. It should be noted, however, that radiator surfaces are not subject to proton bombardment in near earth orbits and that the exterior paint formulation used on the Apollo radiators resists degradation caused by ultraviolet radiation.

To overcome the problem of α_s degradation, coatings other than paints are being developed. One class of these is the so-called second surface mirror type coatings, such as aluminized Teflon foil, which exhibit very low values of α_s and are not believed to be subject to degradation. However, these coatings present problems in application and ground handling in that extreme care must be taken to prevent scratching their highly polished surfaces; and their successful application to a long-duration space radiator system remains to be demonstrated.

HEAT PUMPS

In the context of this discussion of active thermal control systems, a heat pump is essentially a device for transferring a heat load from the temperature level of the source to a higher temperature level in the rejection system (radiator). A heat pump, or refrigerator, may be required when the temperature of a process or piece of equipment being controlled must be maintained below a level obtainable by means of passive or semi-active control methods. Refrigeration may also be employed to raise the operating temperature level of the radiator system and thereby reduce radiator surface area requirements. Such elevation in radiator operating temperature may be advantageous if the resulting reduction in radiator weight penalty is greater than the power and weight penalties imposed by the refrigeration system or if the area available for radiators is limited. A typical example of the reduction in radiator area requirements that may be achieved is illustrated in Figure B-12 for an assumed heat load of 5000 Btu/hr and a heat transport fluid temperature of 75 F. Without the use of a heat pump, the required radiator area is shown to be 142 square feet. This requirement is reduced to about 37 square feet if the average radiator surface temperature is raised to 200 F by means of a refrigeration system. This elevation of the radiator temperature is seen to be achieved at a cost of 790 watts, representing a weight penalty that may or may not be tolerable. But before proceeding with further details regarding heat pump weight penalty, it is appropriate to describe the operation of heat pump systems applicable to this study. These are the basic and the cascaded vapor compression refrigeration systems.

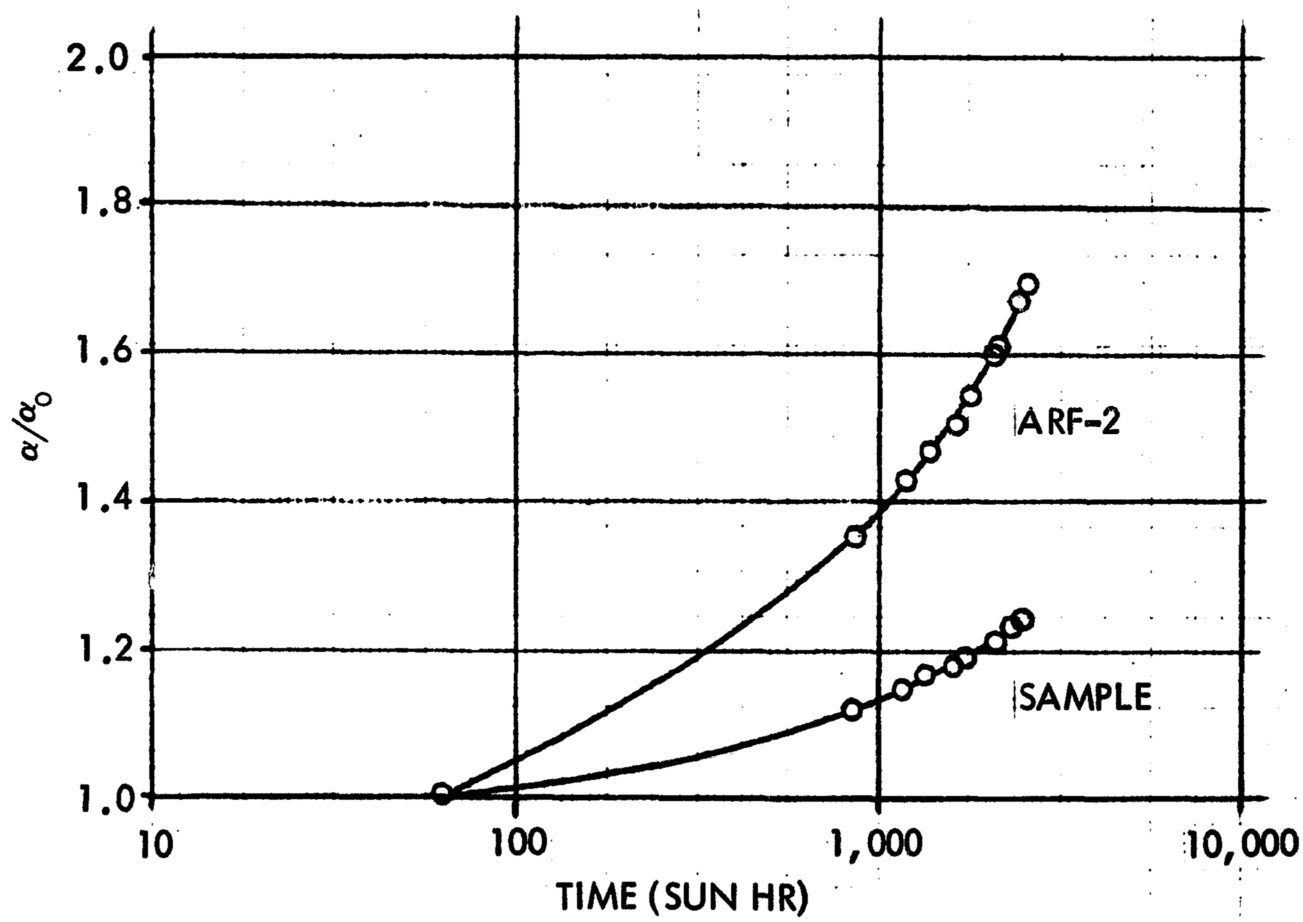


Figure B-11. Degradation of ARF-2 Sample Versus Integrated Exposure Time Mariner IV Absorptivity Standard

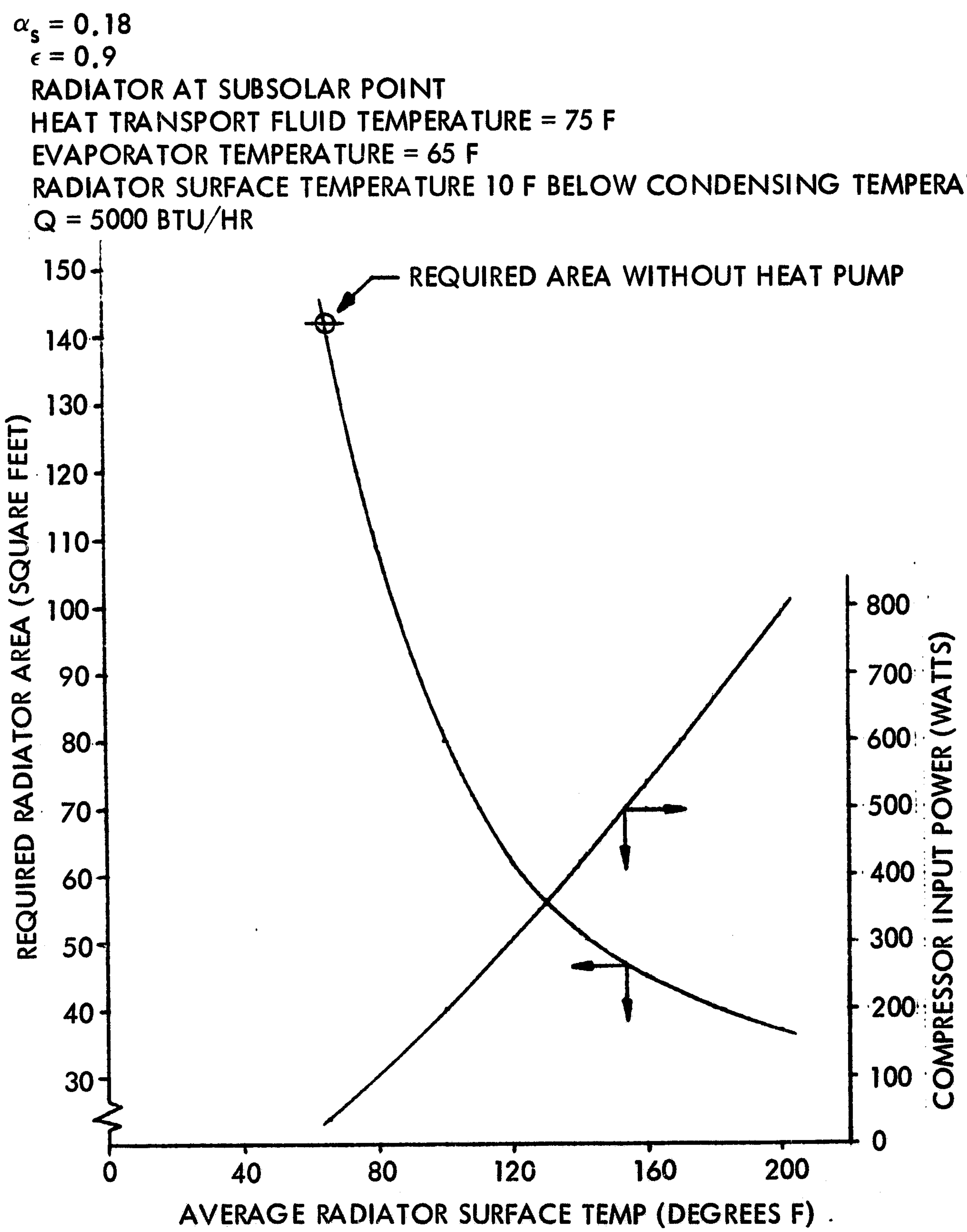


Figure B-12. Radiator Area and Compressor Power Requirements

Basic Vapor Compression Refrigeration System

The basic vapor compression refrigeration system utilizes the latent heat of vaporization of a working substance (refrigerant) to provide a low temperature heat sink for a refrigeration or cooling load. Referring to the schematic representation of the system and the corresponding pressure-enthalpy diagram for the refrigerant (Figure B-13), the functioning of the system may be understood by following the state points of the refrigerant around the underlying thermodynamic cycle.

At state point 1, a mixture of liquid and vapor enters the evaporator which is a heat exchanger designed to transfer the refrigeration or cooling load (Q_a) to the refrigerant. Absorption of this load provides the heat to transform all of the refrigerant into vapor, which leaves the evaporator with a few degrees of superheat at state point 2. The pressure of the vapor leaving the evaporator is raised in the compressor to permit transfer of the cooling load plus the compressor input power at a higher temperature level in the condenser. This compression takes place along the path from state point 2 to state point 3. The high-pressure/high-temperature vapor leaving the compressor enters the condenser, where it is cooled by rejecting the total of cooling and compressor heat loads (Q_a) to the heat sink. The latter may be a circulating gas or liquid, or it may be deep space itself. In the event that heat rejection is to take place directly to deep space, the condenser becomes a space radiator with integral fluid tubes. The refrigerant leaves the condenser as saturated liquid (state point 4) and returns to the evaporator via the expansion valve. This valve throttles the refrigerant from the condensing pressure level to that maintained in the evaporator. The valve also acts as a metering device that insures that the refrigerant flow rate corresponds to the cooling load.

An actual vapor compression refrigeration system, in which the compressor is driven by a constant speed electric motor, is normally equipped with additional control devices such as hot gas bypass and liquid quench valves. These are designed to prevent the evaporator pressure from falling below a lower limit under low refrigeration load conditions. However, they have been omitted from the schematic for sake of simplicity.

The design of a vapor compression refrigeration system for operation in a zero-g environment required special attention to those processes that normally rely on gravity for efficient operation. In the evaporator, for example, bubbles of refrigerant vapor are removed from heat transfer surfaces as soon as they are found, and their place is taken by liquid refrigerant. This removal of vapor bubbles is dependent on body forces that are not available in a zero-gravity environment. To overcome this deficiency, evaporators employing capillary material to produce continuous wetting of heat transfer surfaces have been designed. An alternative method

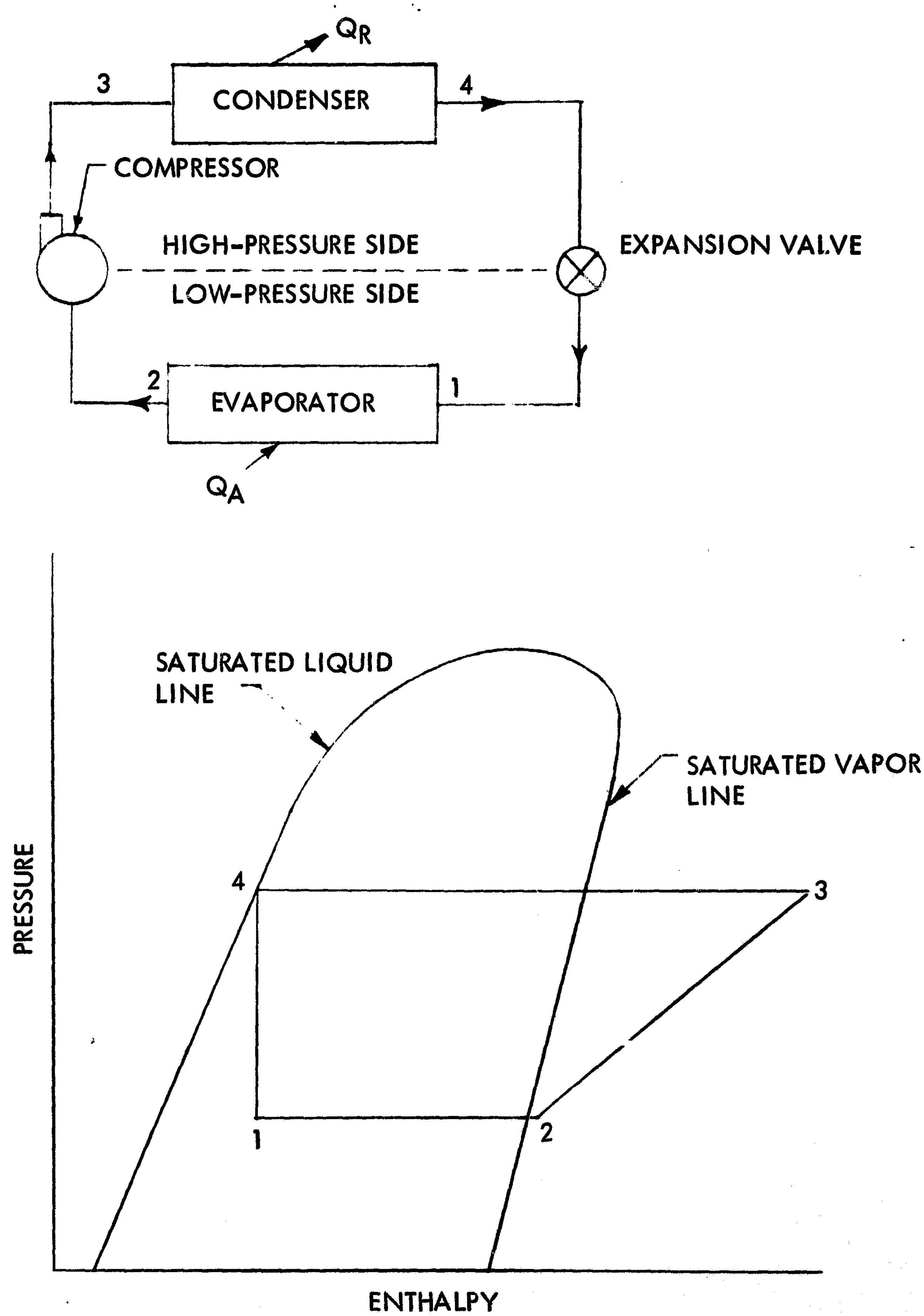


Figure B-13. Basic Vapor Compression Refrigeration System is to induce centrifugal fluid forces by inserting twisted ribbons in evaporator tubes. The induced forces will tend to hold the refrigerant liquid in contact with the tube walls.

Similar zero-g considerations apply to the design of condensers. One applicable design method consists of using condenser tubes that are tapered towards the outlet, thereby maintaining relatively constant fluid velocities as vapor condenses into liquid. Inertial forces induced by a spiral counterflow design may also be employed.

Cascaded Vapor Compression Refrigeration System

A cascaded vapor compression refrigeration system consists essentially of two or more separate but interconnected vapor compression systems operating at different temperature levels. A cascaded system may be employed when the difference in temperature level between evaporator and condenser is too large to be handled by a single vapor compression system.

An example of a cascaded vapor compression refrigeration system is shown schematically in Figure B-14 along with a temperature-entropy diagram to illustrate operation of the system. The lower temperature portion of the system, represented by the solid line cycle 1-2-3-4 on the temperature-entropy diagram, absorbs a refrigeration or cooling load (Q_a) in the same manner as in the basic vapor compression refrigeration system. However, the condenser of this lower temperature portion is replaced by a heat exchanger that also serves as the evaporator for the upper temperature portion of the system. This portion of the system is represented by the dashed line cycle 5-6-7-8 on the temperature-entropy diagram. In the heat exchanger, the refrigeration or cooling load plus the compressor input power load of the lower temperature portion of the system are transferred to the refrigerant circulating through the upper temperature portion. In the process, the latter is being evaporated, while the lower temperature refrigerant is being condensed. The pressure of the vapor leaving the heat exchanger at state point 6 is raised in the compressor of the upper temperature portion of the system to permit rejection of the load absorbed in the heat exchanger plus the input power load of the second compressor in the condenser.

Control system components that permit operation of the system under variable cooling load conditions, and that provide the proper matching of capacities between the lower and upper temperature portions, have been omitted from the schematic for the sake of clarity.

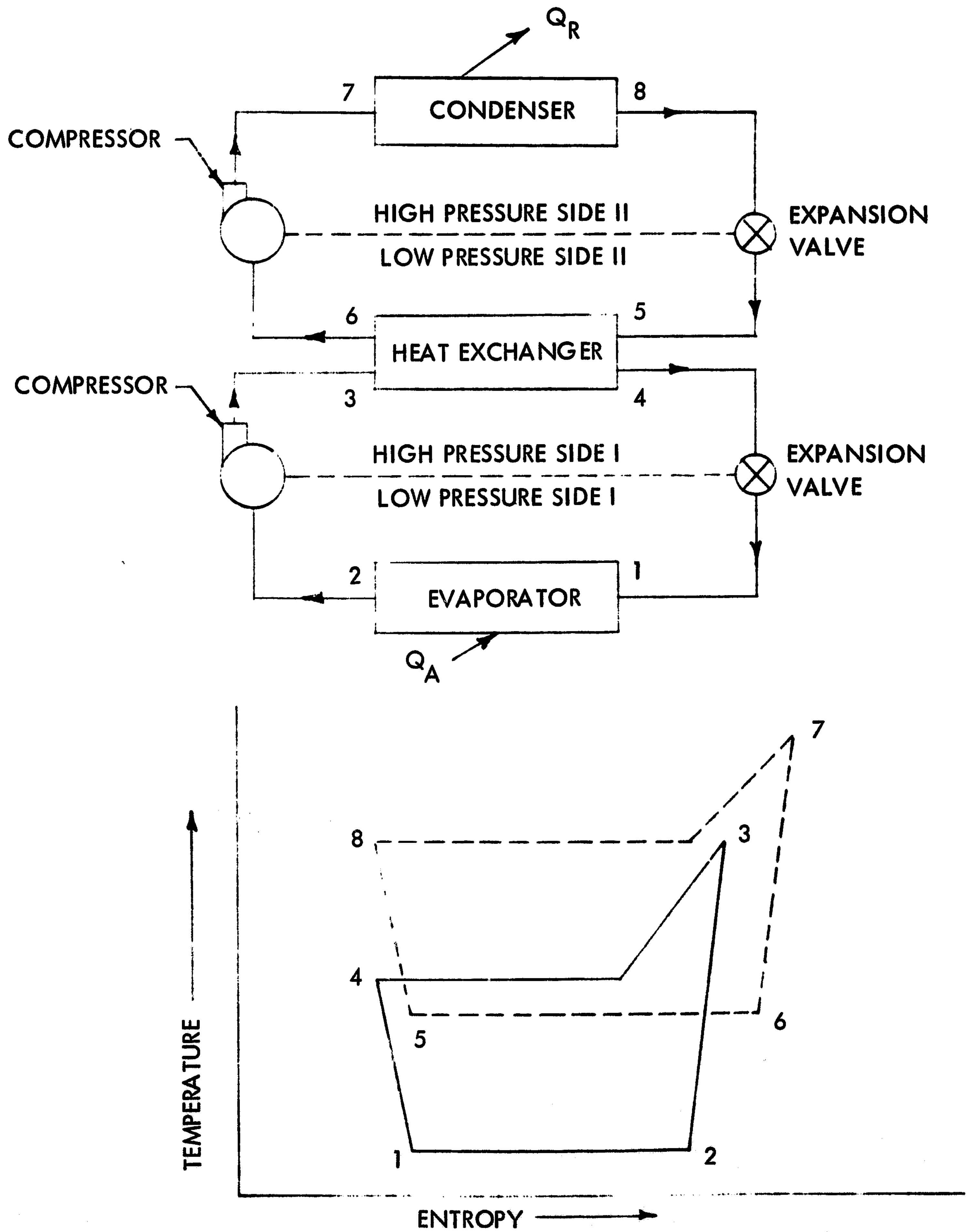


Figure B-14. Cascaded Vapor Compression Refrigeration System

Compressor input power requirements as a function of condensing temperature and with evaporator temperature as a parameter are shown in Figures B-15 and B-16 for the basic and the cascaded vapor compression systems, respectively. These graphs are based on the thermodynamic properties of Refrigerant-11 and Refrigerant-12 and on an assumed compressor efficiency of 70 percent. Refrigerant vapor was assumed to leave the evaporator with 5 F of superheat, and the condition of the fluid at the exit of the condenser was assumed to be saturated liquid. In the cascaded system, condensation of the lower temperature fluid (R-12) was assumed to take place at 100 F, while evaporation of the higher temperature fluid (R-11) was assumed to be at 90 F. Thus, a 10 F temperature differential was provided in the heat exchanger connecting the two portions of the system. All data shown are for a heat load of 1000 Btu/hr; and simple proportioning may be used in the application of the data to other heat loads.

The results of using the heat pump data of Figures B-15 and B-16 and appropriate radiator performance data (Figure B-1) are shown in Figure B-17 for the applicable heat load range. This graph shows required condensing radiator area as a function of heat load, with evaporator and radiator surface temperatures as parameters. Corresponding estimates of heat pump system weight penalty are illustrated in Figures B-18, B-19, and B-20. The power penalty in these graphs is based on an assumed penalty factor of 1.0 pound per watt, and radiator weight penalties were calculated at the same rate of 0.2 pound per square foot used previously. Additional system weights (fluid and equipment other than radiators) are based on a recent study (Reference 2-2) which estimated this weight to be 40 pounds for a vapor compression system handling a load of 1 kilowatt (3412 Btu/hr).

WATER BOILERS AND SUBLIMATORS

In lieu of rejecting waste heat to the deep space heat sink, the latent heat of vaporization of water may be utilized as a heat sink if excess quantities of water are available. Figure B-21 indicates the weight penalties involved assuming the entire waste heat load is absorbed by boiling water. Weights were established on the basis of 1000 Btu heat sink capacity per pound of water and a 5 percent allowance for tankage, plumbing, etc. In connection with the use of water, it is worth noting that an evaporator (boiler) has been developed for the Apollo ECS for operation in a zero-g environment. This evaporator has a maximum heat load capacity of about 7500 Btu/hr and weighs approximately 15 pounds. Another device for utilizing boiling water as a heat sink is a sublimator, developed for use on the lunar module. Although somewhat heavier than an evaporator of the same capacity, the sublimator offers the advantage of fewer operating and control problems over a varying load range.

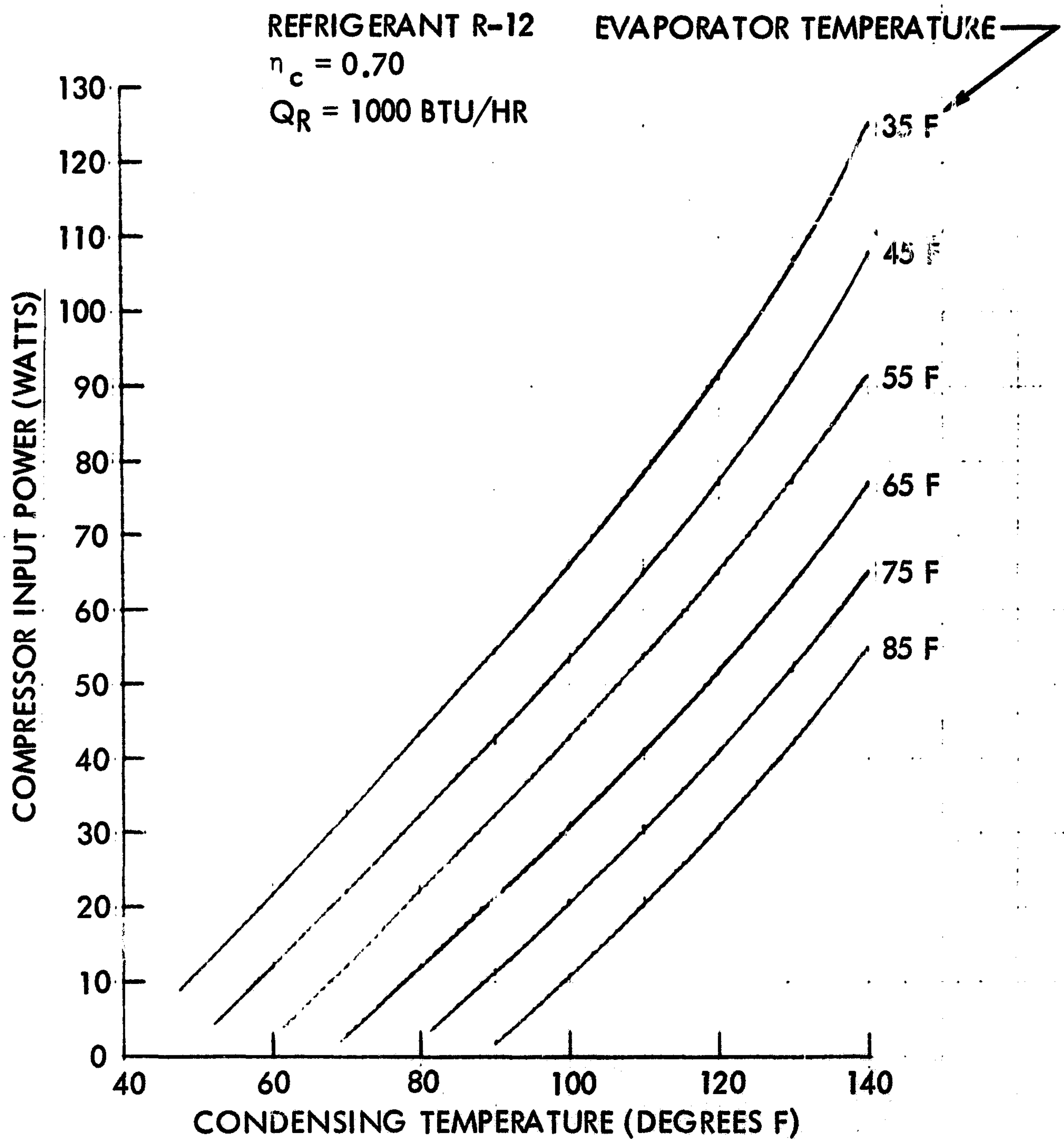


Figure B-15. Heat Pump Compressor Input Power Versus Condensing Temperature Basic Vapor Compression Refrigeration System

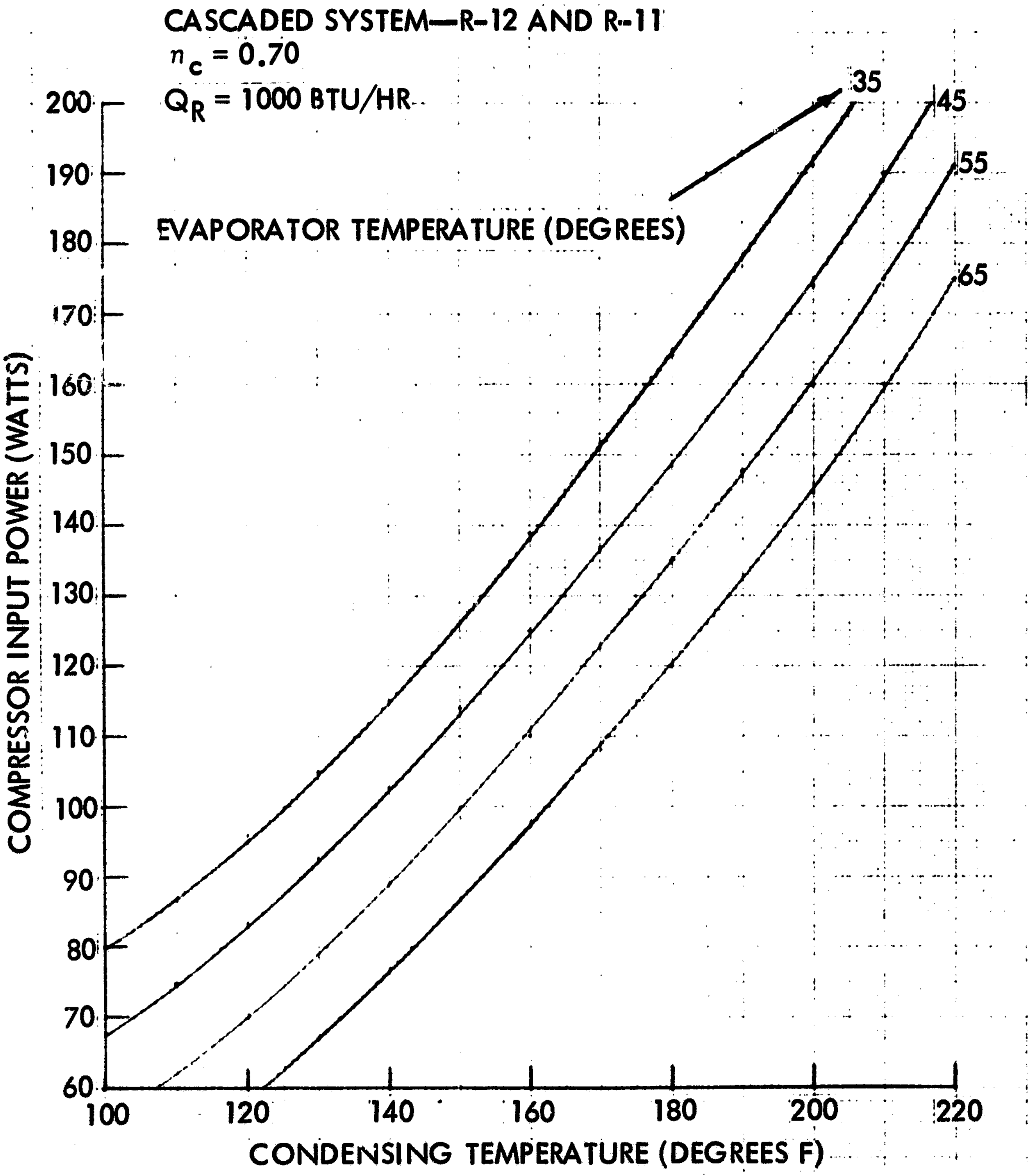


Figure B-16. Heat Pump Two-Stage Compressor Input Power Versus Condensing Temperature

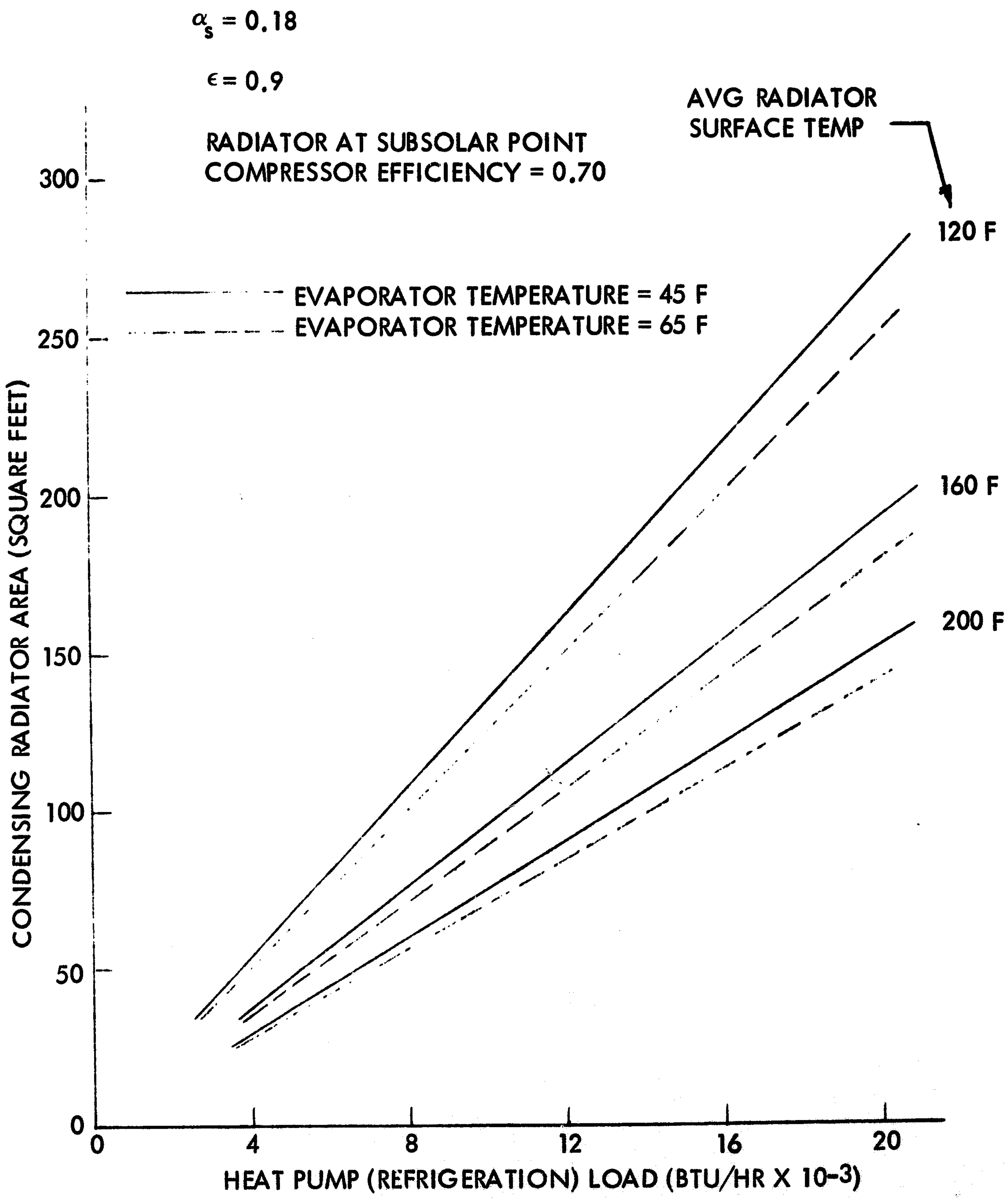


Figure B-17. Condensing Radiator Area Versus Heat Load

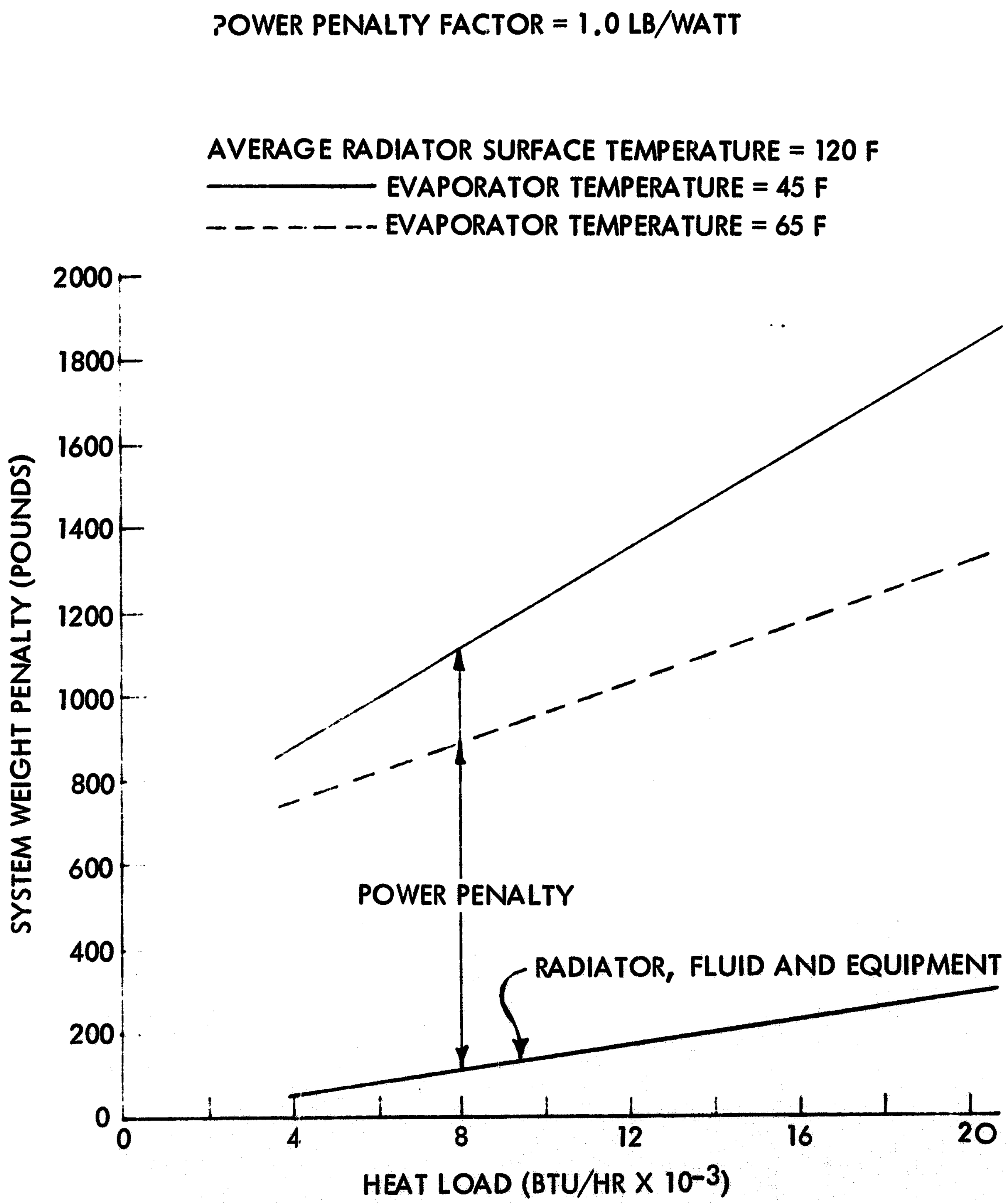


Figure B-18. Estimate of Heat Pump System Weight Penalty—Average Radiator Temperature 120 F

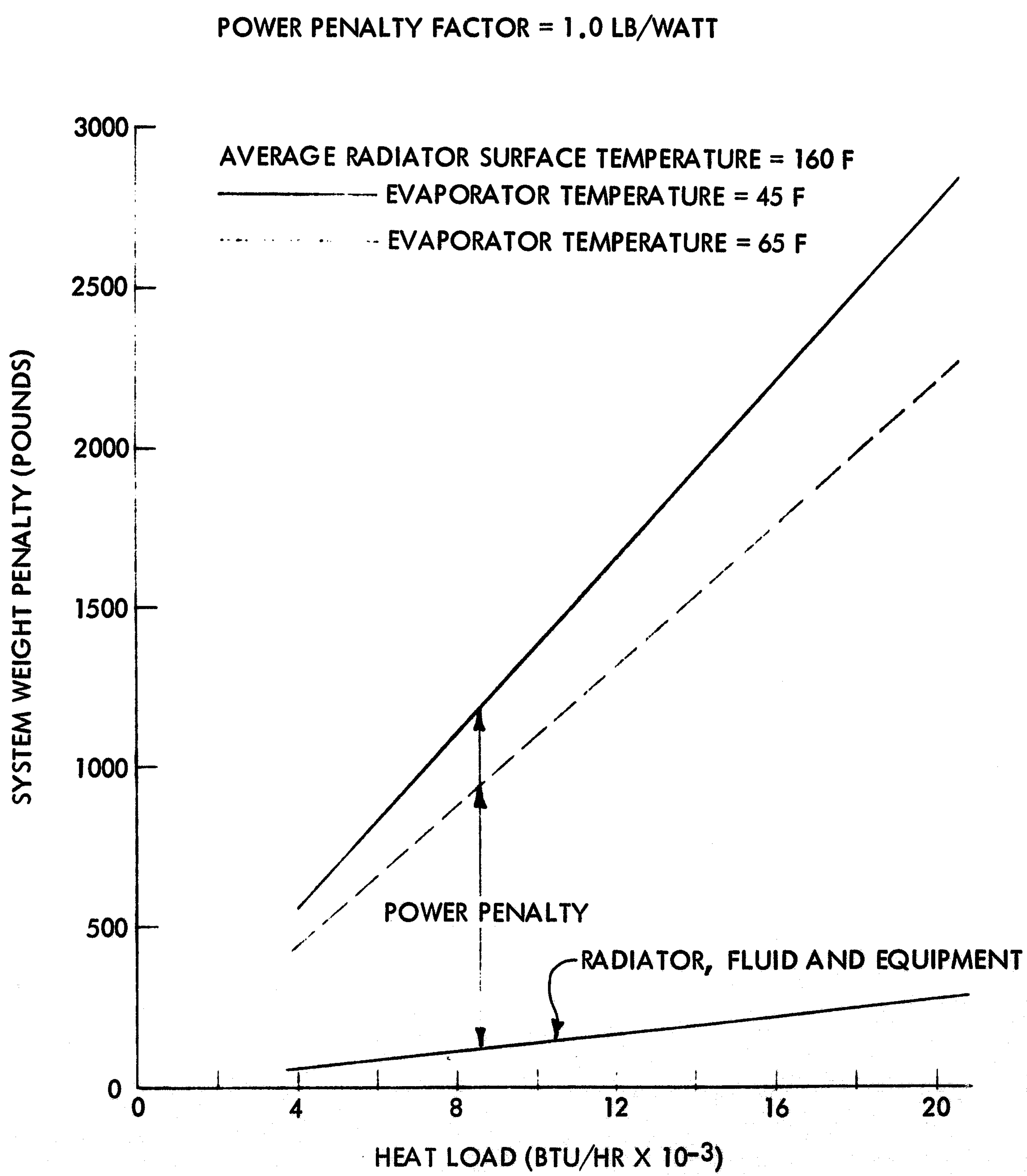


Figure B-19. Estimate of Heat Pump System Weight Penalty—Average Radiator Temperature 160 F

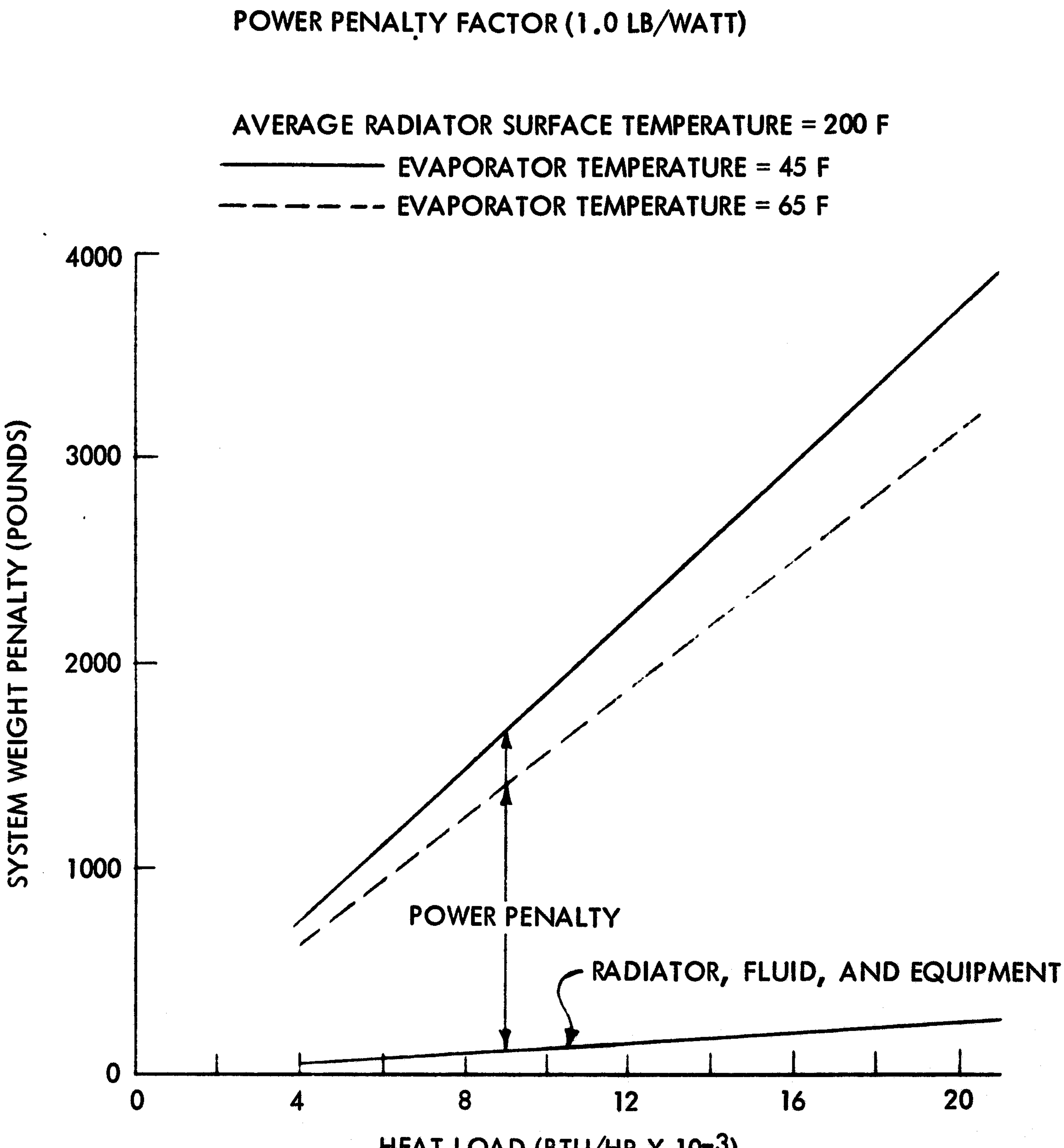


Figure B-20. Estimate of Heat Pump System Weight Penalty—Average Radiator Temperature 200 F

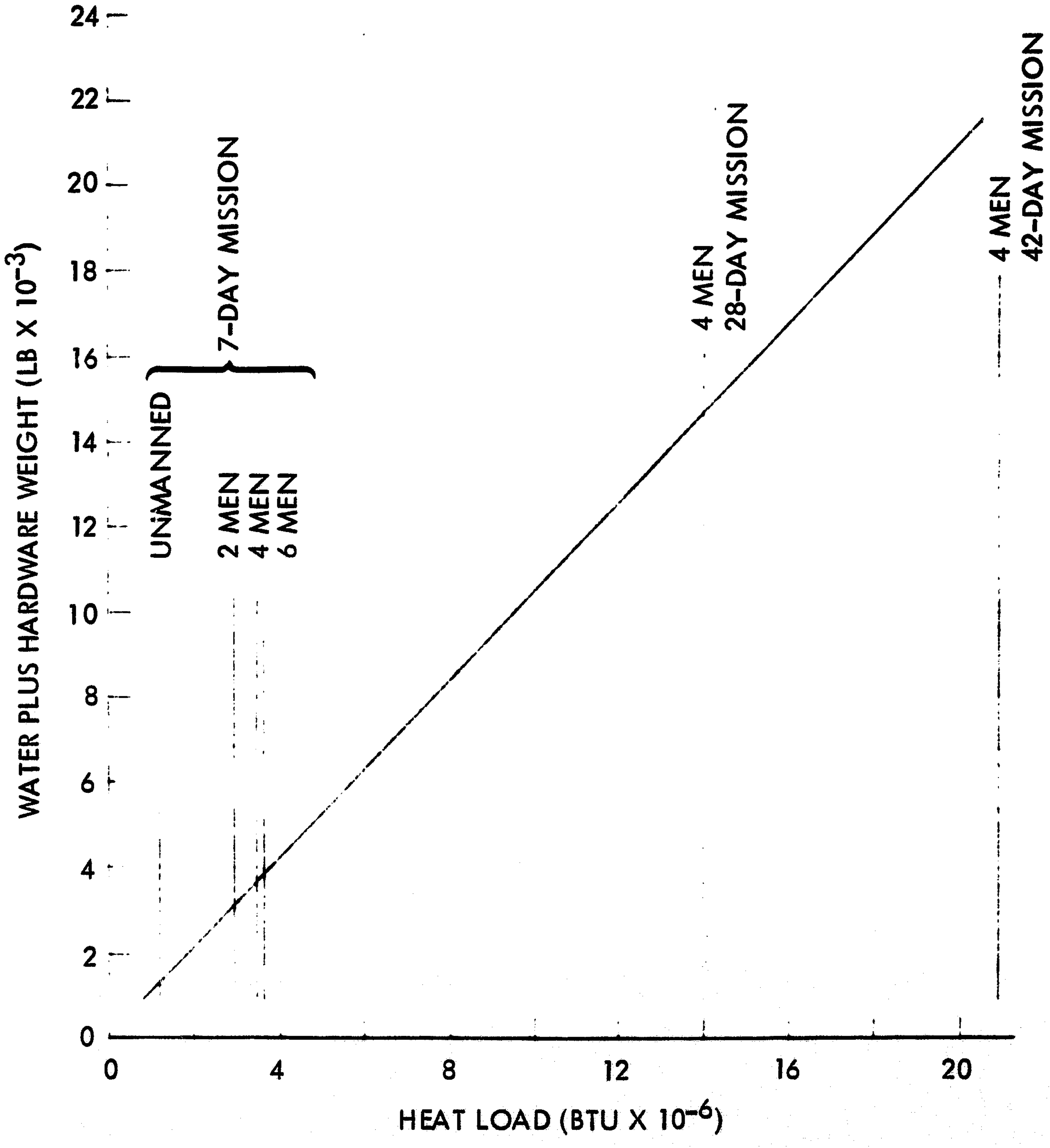


Figure B-21. Water System Weight Versus Heat Load

GH_2 HEAT EXCHANGERS

If appreciable quantities of hydrogen vent gas are available during extended lunar surface operations, serious consideration can be given to the use of this gas as a heat sink. Due to its high specific heat (more than 3.0 Btu/lb-F), each pound of hydrogen gas can absorb in excess of 1000 Btu. This figure is based on an assumed vent gas temperature of -400 F and heating of the gas to 0 F before it is permitted to escape to space. Heating of the gas can be achieved in a suitable heat exchanger in which heat is transferred from a circulating fluid to the gas. Special precautions must be taken in the design of such a heat exchanger system to prevent freezing of the heat transport fluid.

ABSORPTION REFRIGERATION CYCLE STUDY FOR REUSABLE SPACE TUG

The absorption refrigeration cycle considered in this study consists of an aqua-ammonia-water system. The ammonia acts as a refrigerant and the water acts as an absorber. Variation of radiator surface area, total flowrate, and power requirements as a function of generator pressure and refrigeration heat load is presented. The absorption refrigeration cycle is compared with the vapor compression cycle.

Study results reveal that the vapor compression cycle would require approximately 35 times more power and 38 percent less total radiator surface area as compared with the absorption refrigeration cycle in order to accommodate the same refrigeration heat load. No attempt was made in this study to optimize design of the absorption refrigeration cycle because considerable time and effort would be required.

Basic Absorption Refrigeration Cycle

The basic components of the absorption refrigeration cycle considered in this analysis are an absorber, pump, heat exchanger generator, condenser, and evaporator. Figure B-22 shows a flow diagram of the basic system and Figure B-23 shows the temperature concentration diagram. The absorption refrigeration cycle requires two working fluids, the refrigerant and the absorbent, the refrigerant circulator from the absorber, to the generator, to the condenser, to the evaporator, and back to the absorber. The absorbent ideally circulates from the absorber to the generator and back to the absorber. The basic function of the refrigerant is to remove the refrigeration heat load from the evaporator and reject it in space through the condenser radiator. The main function of the absorbent is to absorb the refrigerant vapor in the absorber, compress it to its liquid state at the absorber pressure and temperature, and finally return it to the generator

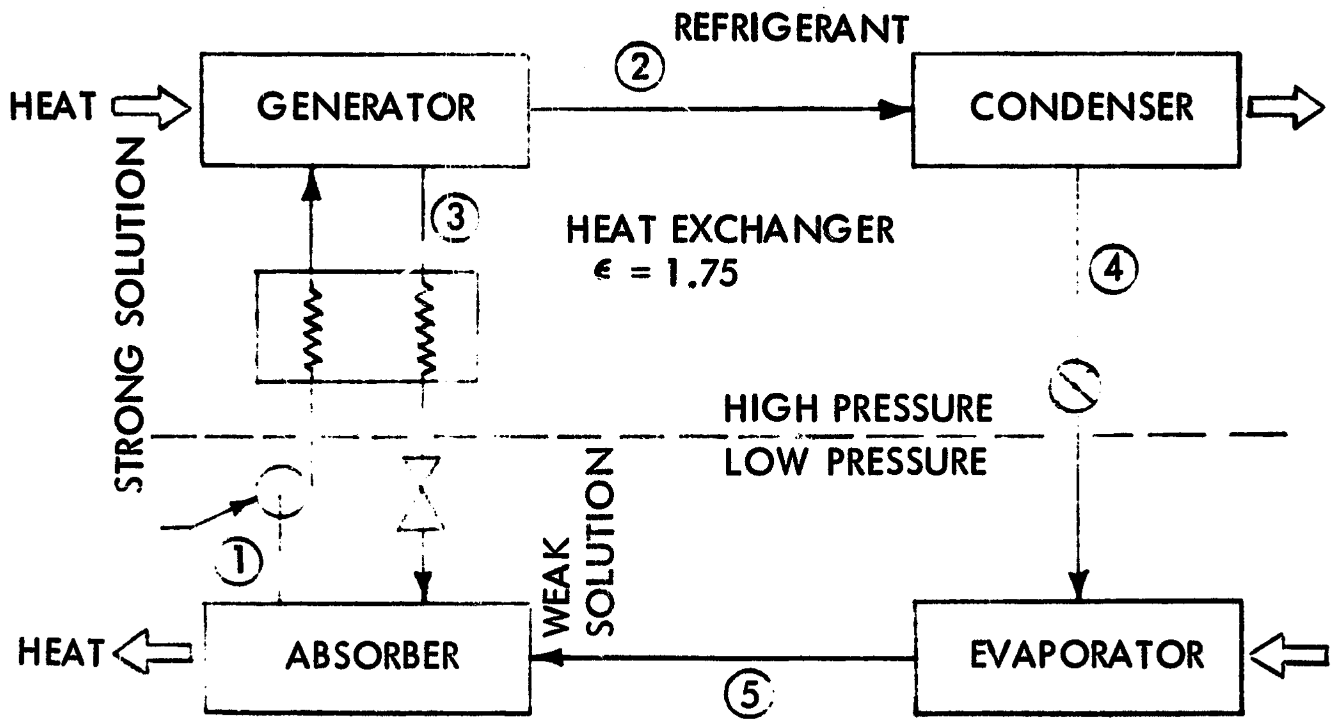


Figure B-22. Basic Absorption Cycle

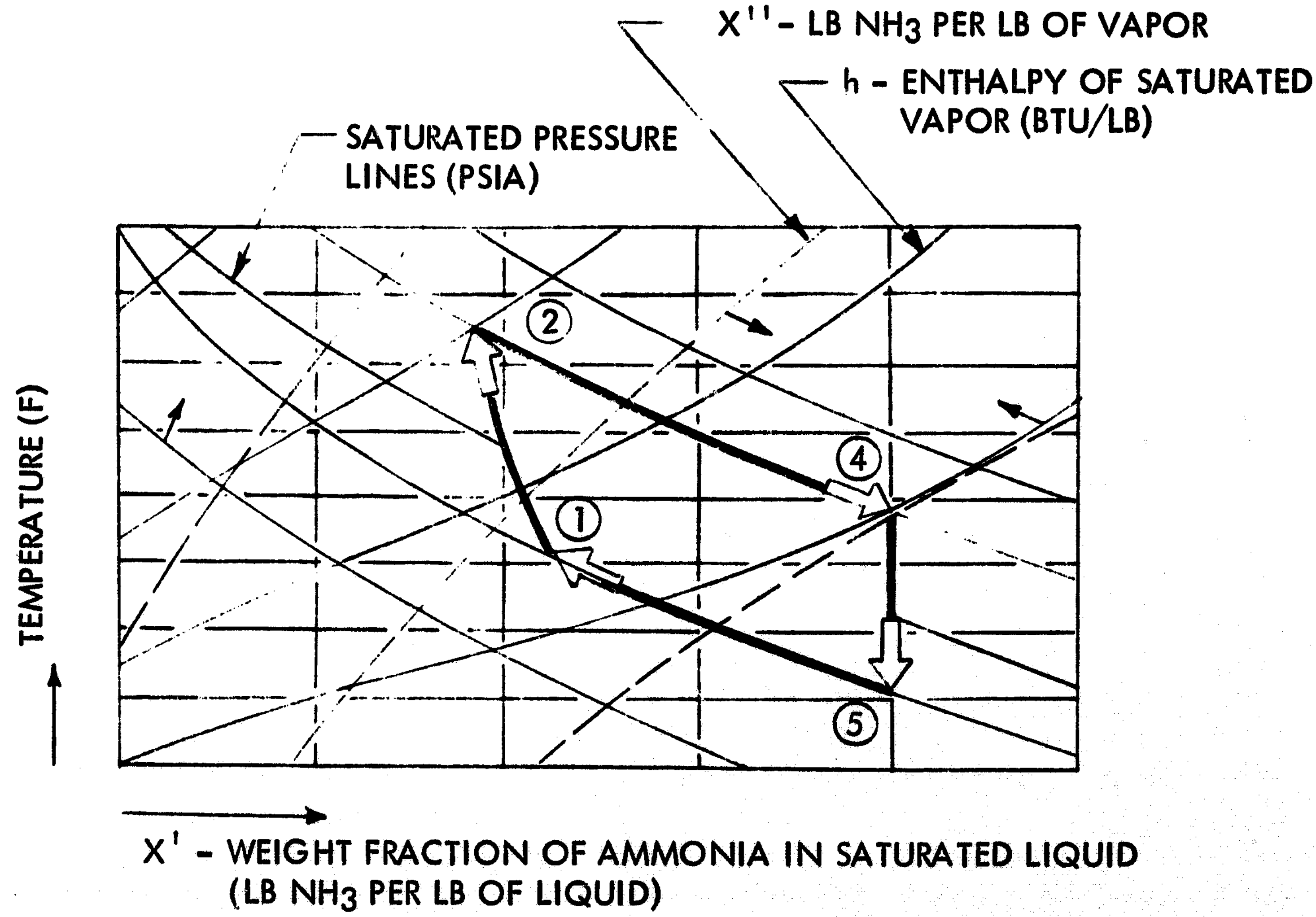


Figure B-23. Typical Temperature Concentration Diagram with the aid of the pump. Since the generator is at higher pressure than the absorber, a pump is required to make the transfer. The pump is not a compressor because the absorber has already provided the compression; it is merely a circulating device which provides enough power to overcome pressure head and friction losses through the system. The amount of power required to drive the pump is considerably smaller in comparison to that required to drive a vapor compressor for the same refrigeration.

To utilize an absorption refrigeration cycle as a device to provide air conditioning on the reusable space tug, the absorber and condenser would also function as space radiators and the generator would function as a solar absorber. It is obvious from Figure B-22 that the total radiator surface area would be larger in comparison to that required for a vapor compressor cycle where only one condenser radiator is required.

The purpose of this study is to evaluate the total radiator surface area and the power requirements in order to compare this system with the vapor compression refrigeration cycle. This study is based on the ammonia-water absorption refrigeration cycle. Its highly critical pressure and temperature and high latent heat of vaporization make its use attractive in the absorption refrigeration cycle. This study considers the effects of generator pressures, condenser temperature, and absorber temperature on the radiator's surface area and pump power requirements.

In order to compare the absorption cycle with the compression refrigeration cycle, the total radiator surface area and power requirements of both systems are presented as a function of the refrigeration heat loads.

Results and Discussion

Figure B-24 presents the radiator surface area as a function of the generator pressure with solar absorptivity to emissivity ratio (α_s/ϵ) as a parameter. It is obvious from Figure B-24 that the generator should operate at high pressures if it is to keep the radiator surface area small. Also, the importance in maintaining low solar absorptivity for the condenser and absorber radiators is plainly demonstrated in Figure B-24.

Figure B-25 shows that the total flowrate (strong solution), and thus the power requirements, are increasing with increasing generator pressure. Consequently, the designer must sacrifice power requirements for lower radiator surface area or vice versa.

Figure B-26 shows the variation of generator, absorber, and condenser temperature with increasing generator pressure for constant evaporator pressure, temperature, and constant X'' (lb - NH_3/lb vapor). Both condenser and absorber temperatures are assumed equal in this evaluation.

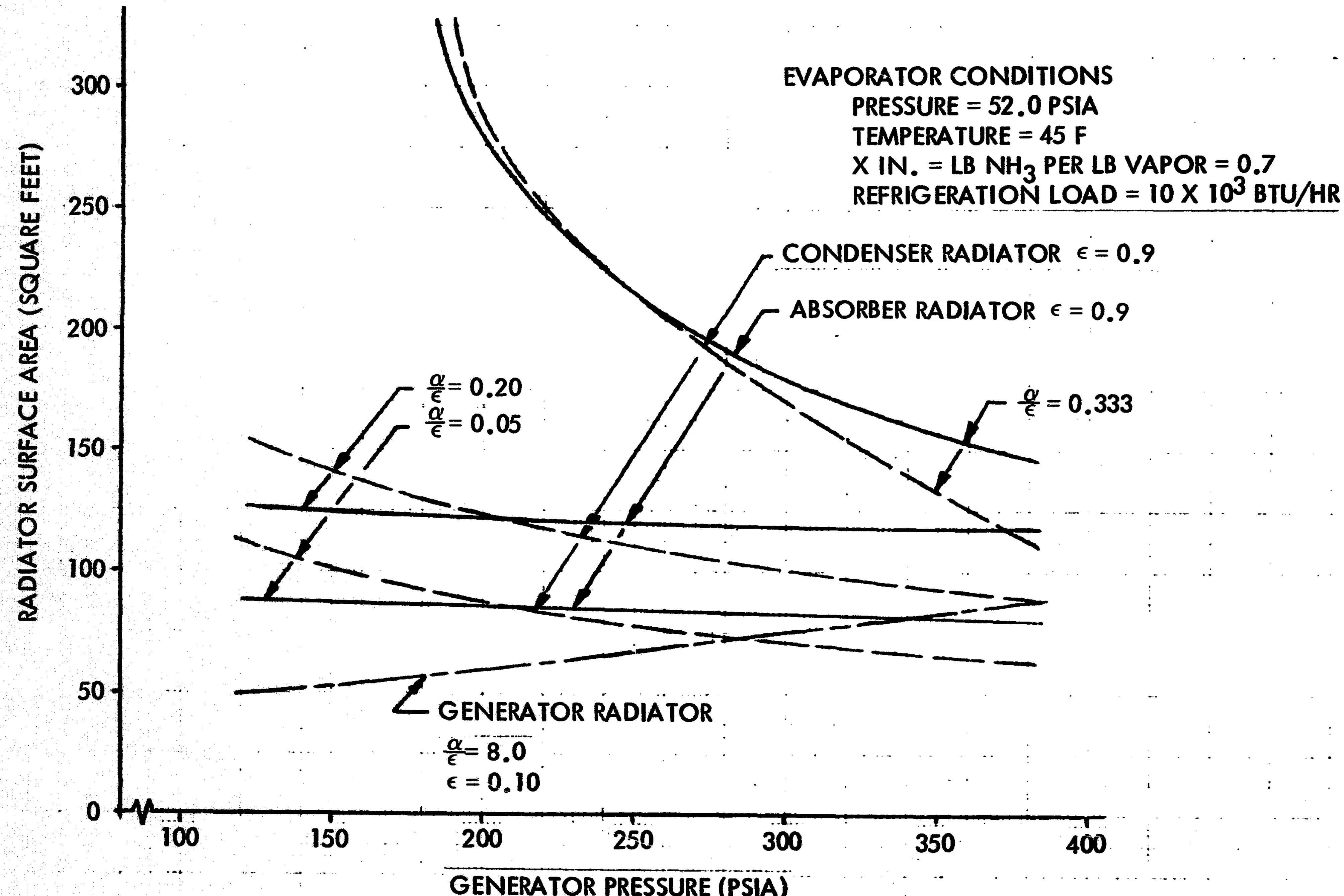


Figure B-24. Radiator Surface Area Versus Generator Pressure

Space Division
North American Rockwell

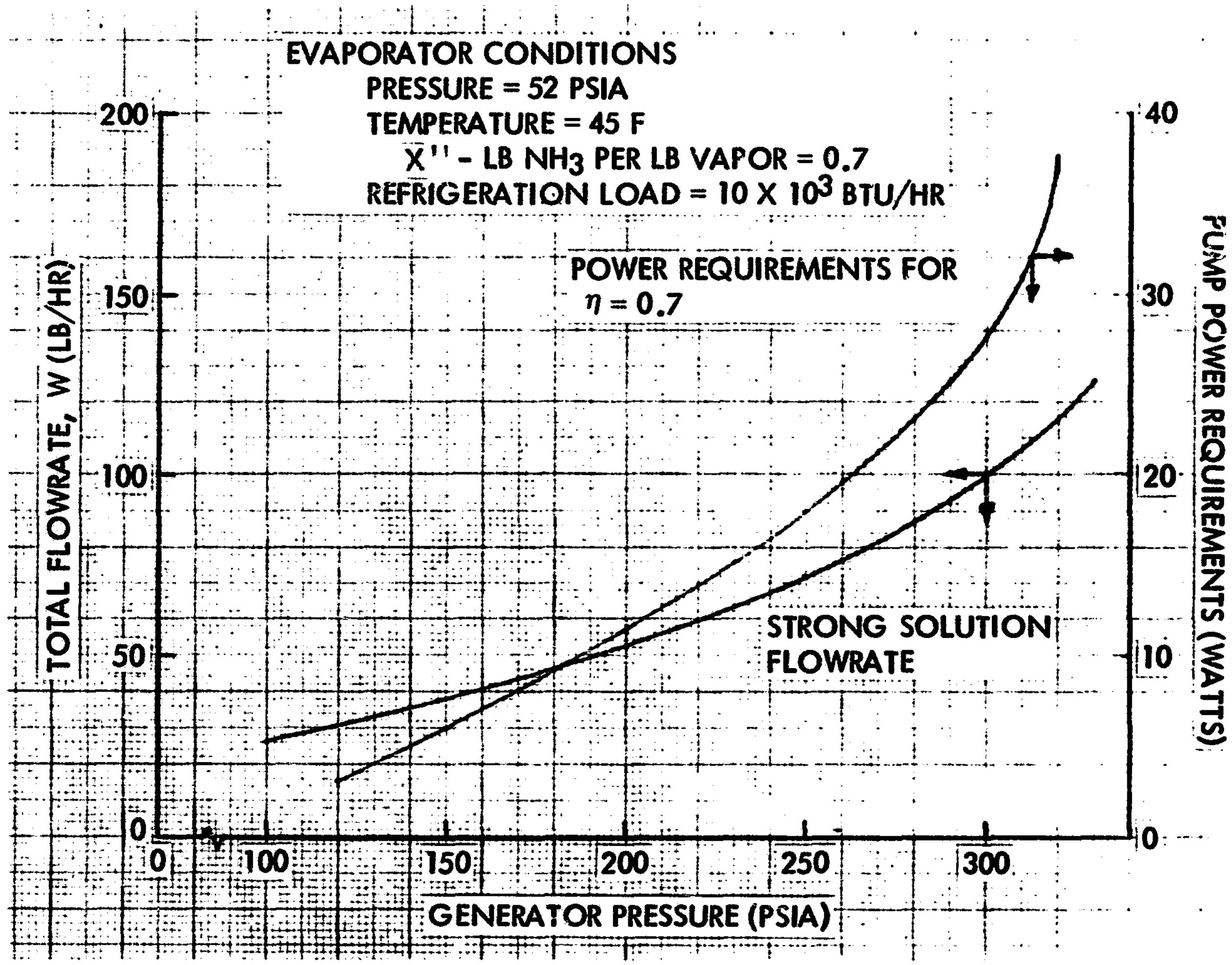


Figure B-25. Total Flow Rate and Power Requirements Versus Generator Pressure

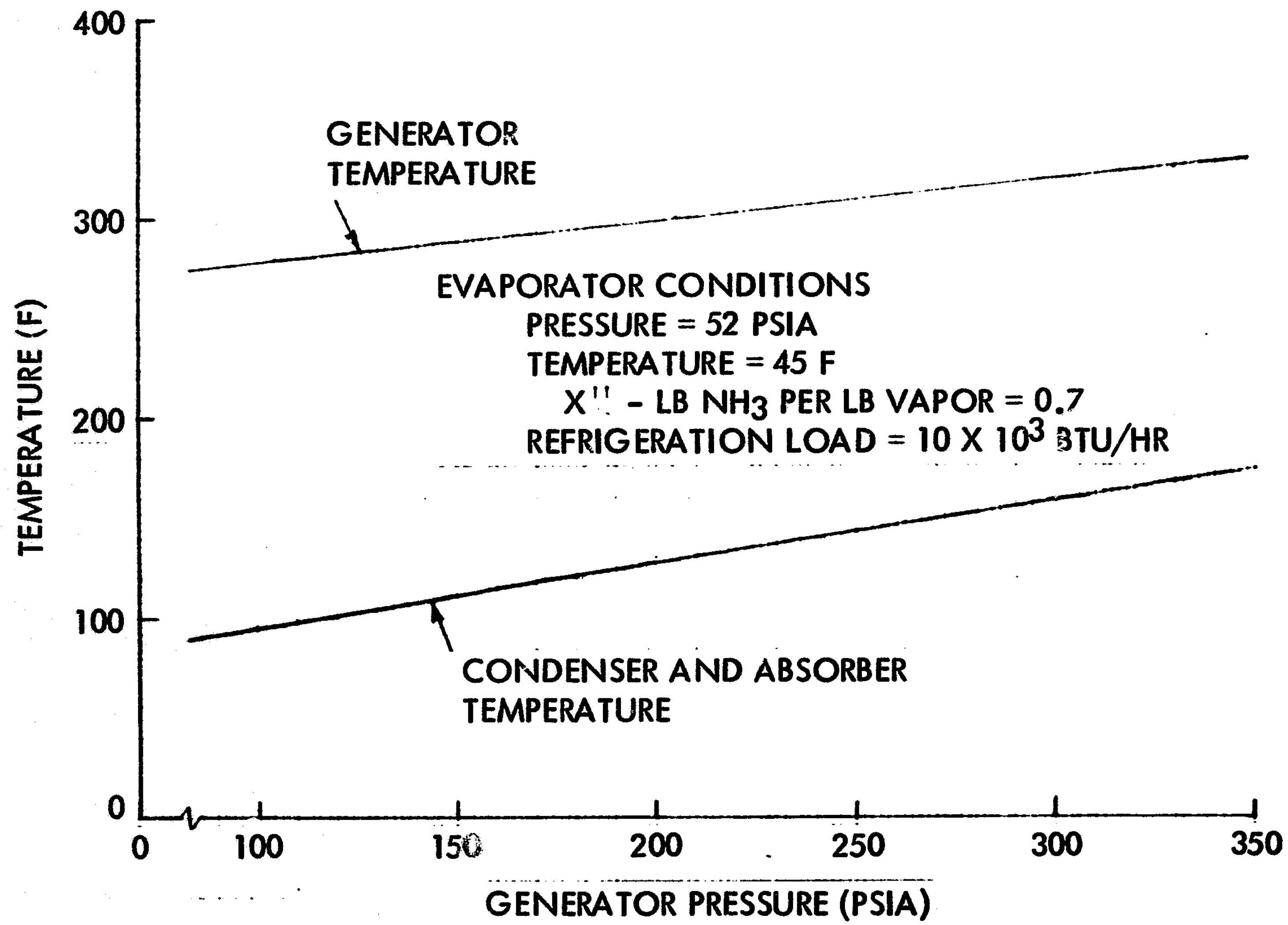


Figure B-26. Temperature Versus Generator Pressure

Figure B-26 shows that the temperatures of both condenser and absorber are increasing faster (dT/dP) than the generator temperatures with increasing generator pressures. Since the major portion of the radiator's surface area depends on the absorber and condenser surface temperature, it would be highly desirable to operate the absorption system at higher generator temperatures. The maximum operating generator pressure and temperature would be of course established by the critical temperatures and pressures of the refrigerant.

Figure B-27 presents the total radiator surface area and pump power requirements as a function of the refrigeration heat load. Superimposed in this figure are the data obtained for the vapor compression refrigeration cycle. Figure B-27 indicates that for the same refrigeration heat load (10,000 Btu/hr), the vapor compression cycle power requirement is 35 times higher than the power requirement for an absorption refrigeration cycle. The total radiator surface area, however, for the vapor compression cycle, is 38 percent of the total area of the absorption refrigeration cycle. Consequently, the choice of the cycle refrigeration system would depend on whether the power requirements or the radiator surface area is the governing designing factor for the reusable space tug.

An example of a trade between the two systems involving radiator area weight penalty and a power weight penalty may be made as follows for a refrigeration load of 10,000 Btu/hr:

	Radiator Area			Power		
		Penalty (Lb)				
Refrigeration Cycle	Area Ft^2	<150 ft^2 (0.2 lb/ft^2)	>150 ft^2 (2 lb/ft^2)	Watts	Penalty 1 lb/watt (lb)	Total Weight Penalty (lb)
Absorption	300	30	300	35	35	365
Vapor Compression	110	22	-	1,060	1,060	1,082

This comparison shows the absorption cycle to have only one third of the weight penalty of the vapor compression cycle. The weight penalty for the radiator was assumed to be only 0.2 pound per square foot for a 150 square foot area on the top of the tug and 2.0 pounds per square foot for a deployable radiator greater than the first 150 square feet.

CONDITIONS:

1. AQUA-AMMONIA/WATER ABSORPTION SYSTEM
2. MOTOR EFFICIENCY $\eta = 0.7$
3. EVAPORATOR TEMPERATURE = 45 F
4.

RADIATOR	α/ϵ	ϵ	PRESSURE (PSIA)	TEMPERATURE (F)
CONDENSER	0.2	0.9	300	150
ABSORBER	0.2	0.9	72	150
GENERATOR	8.0	0.1	300	300

5. RADIATOR SURFACE AREA IS BASED ON THE OUTLET TEMPERATURE
6. HEAT EXCHANGER EFFECTIVITY = 0.75

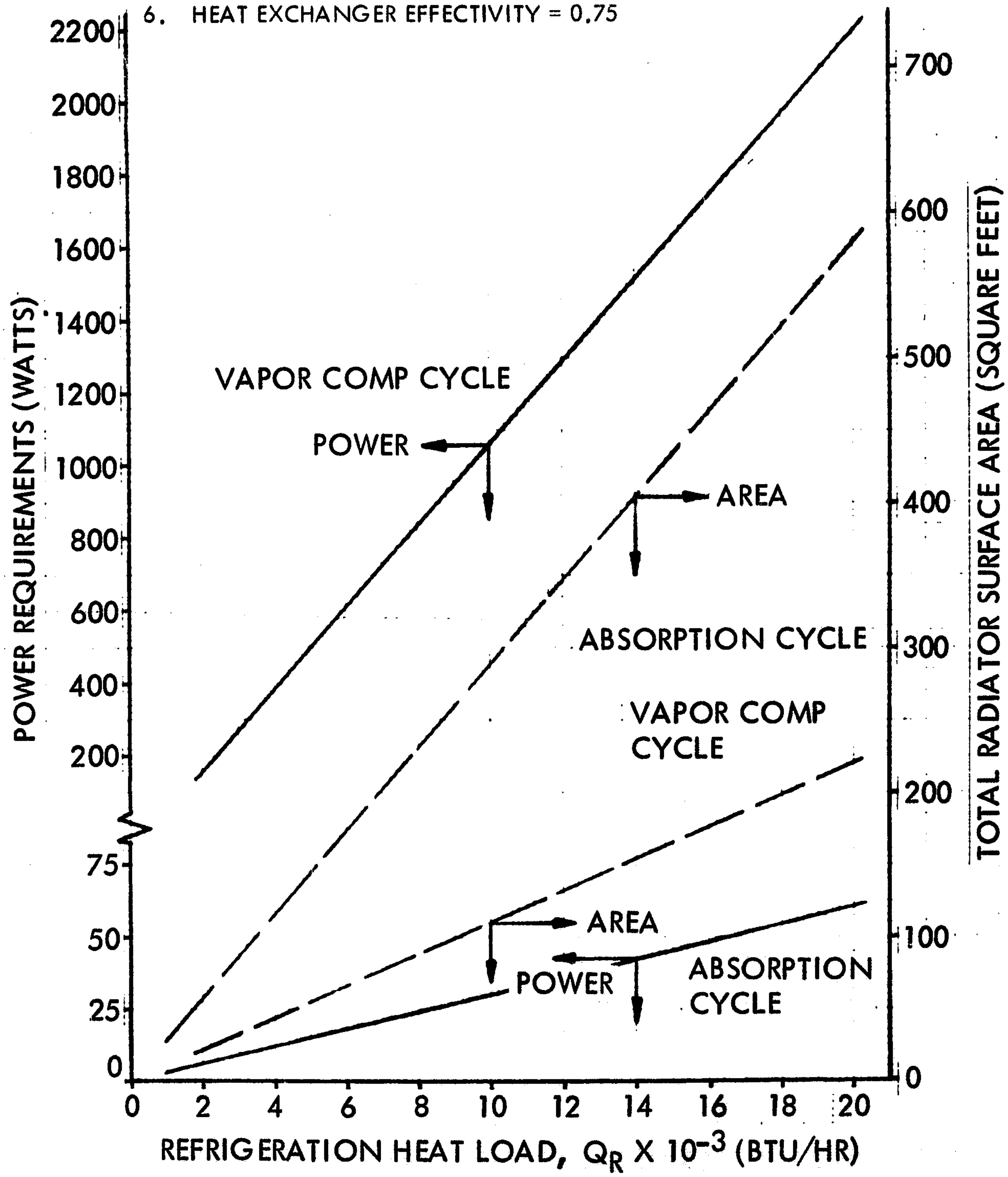


Figure B-27. Comparison of Power Requirements and Total Radiator Surface Area Versus Refrigeration Heat Load Between Vapor Compressor

The fact that the inlet temperature for both the condenser and the absorber would be higher would tend to reduce the required radiator area. However, this fact was not considered because other factors in the detailed design were also ignored which would increase the radiator area. With a tube and fin radiator the temperature distribution on the fin will be a function of the fin length, the fin thickness, the absolute temperature of the fin base at the tube wall, and the environmental heat incident on the fin surface. These factors comprise the fin effectiveness and tend to increase the radiator area required. These compensating details were ignored at this time so that a comparable comparison could be made between the two refrigeration cycles on a common basis.

The effects of increasing the absorber outlet temperature on the radiator surface area and power requirements are presented in Figures B-28 and B-29. Note in Figure B-28 that the condenser surface area remains constant with increasing absorber temperature. The condenser surface area is only a function of the condenser temperature and the refrigeration load. Since those two parameters remain constant, the condenser area requirements will remain fixed. The generator surface area, however, increases with increasing absorber temperature although the generator temperature remains constant. The increase of generator surface area is due to the increase of strong solution, and thus condenser heat flux requirements with increasing absorber temperature.

Power requirements are increasing much faster with increasing absorber temperature as compared to the increase of power requirements with increasing generator pressure. Figure B-30 presents the effects of condenser temperature on the radiator surface area. The condenser surface area remains constant since all rejection heat load, condenser temperature, and refrigeration flow rate remain fixed. The generator radiator surface area, however, is increasing with increasing condenser outlet temperature. This increase is attributed to the increase of the generator temperature which, in turn, increases the radiation heat losses.

Figure B-31 presents the influence of condensing temperature on the total flow rate and pump power requirements. Inspection of Figure B-31 reveals that power requirements drop with increasing condenser temperature for constant generator pressure and evaporator temperature. A comparison between Figures B-29 through B-31 indicates that both radiator surface area and power requirements can be reduced by maintaining constant absorber temperature and generator pressure, while increasing the radiator temperature. This, of course, necessitates a reduction in evaporator and absorber pressures.

COMPONENTS	PRESSURE (PSIA)	TEMPERATURE (F)	X'' $\frac{\text{LB } NH_3}{\text{LB MIX.}}$
GENERATOR	300	265	0.90
CONDENSER	300	130	0.90
EVAPORATOR	72	45	0.90
ABSORBER	72		0.90

REFRIGERATION = 10 X 10^3 BTU/HR

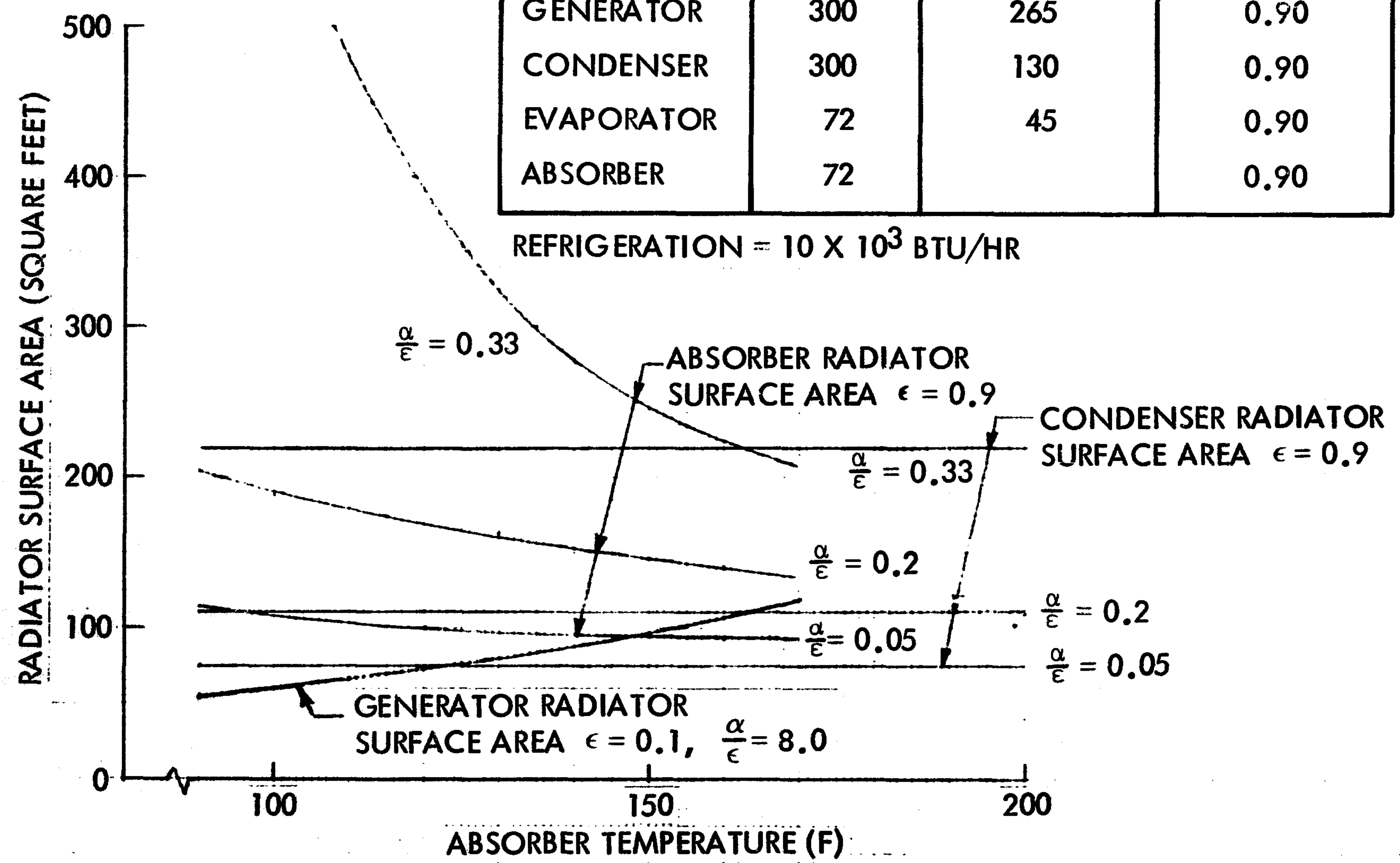
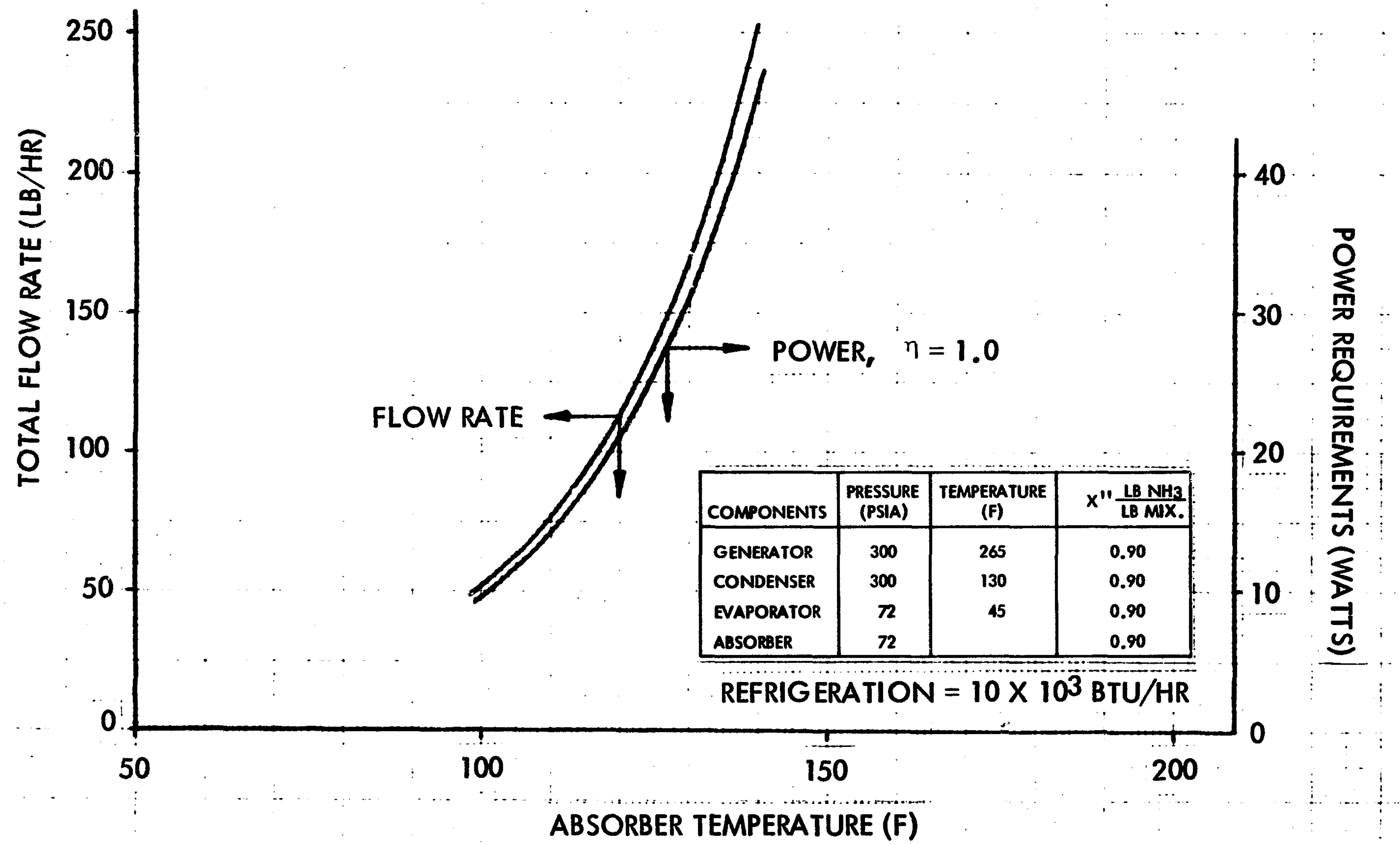


Figure B-28. Radiator Surface Area Versus Absorber Temperature



COMPONENTS	PRESSURE (PSIA)	TEMPERATURE (F)	X'' LB NH3 / LB MIX.
GENERATOR	300	265	0.90
CONDENSER	300	130	0.90
EVAPORATOR	72	45	0.90
ABSORBER	72		0.90

Figure B-29. Total Flowrate and Power Requirements Versus Absorber Temperature

COMPONENTS	PRESSURE (PSIA)	TEMPERATURE (F)	X'' LB NH3 / LB MIX.
GENERATOR	300	265-322	0.7-0.9
CONDENSER	300		0.7-0.9
ABSORBER	70-52	130	0.7-0.9
EVAPORATOR	70-52	45	0.7-0.9

$Q_R = 10 \times 10^3$ BTU/HR

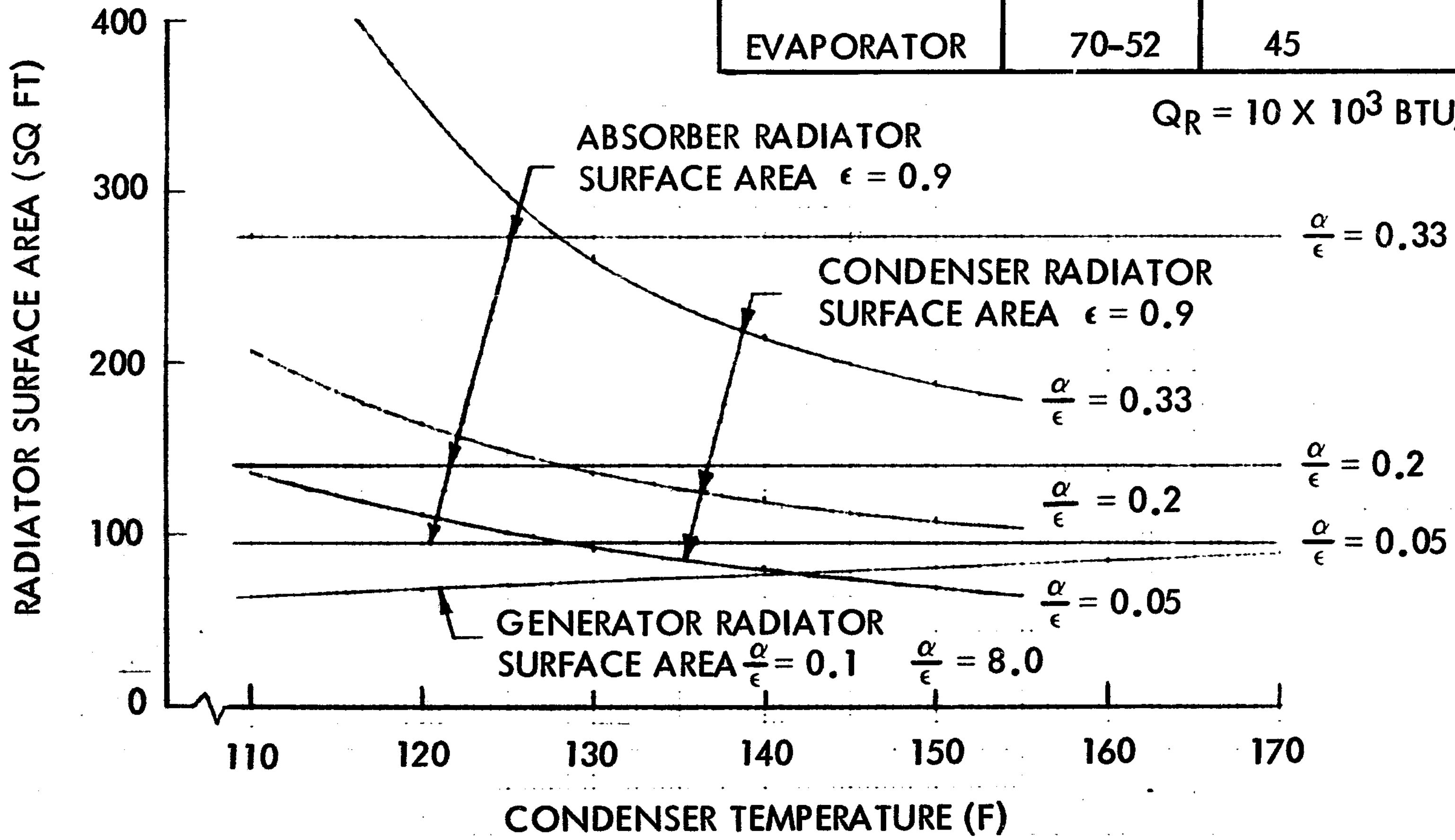


Figure B-30. Radiator Surface Area Versus Condenser Temperature

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COMPONENTS	PRESSURE (PSIA)	TEMPERATURE (F)	X'' $\frac{\text{LB NH}_3}{\text{LB MIX.}}$
GENERATOR	300	265-322	0.7-0.9
CONDENSER	300		0.7-0.9
ABSORBER	70-52	130	0.7-0.9
EVAPORATOR	70-52	45	0.7-0.9

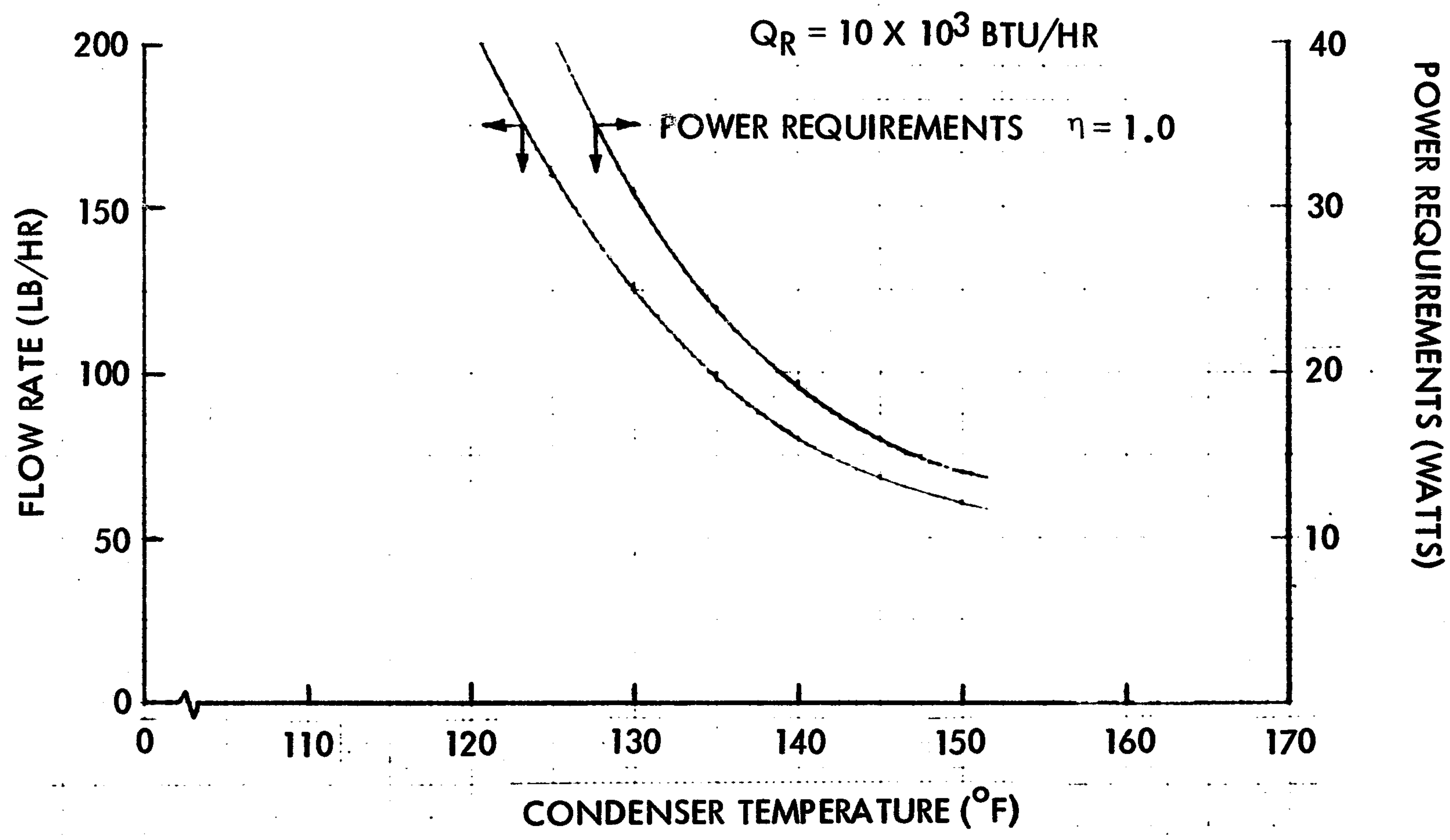

Figure B-31. Flowrate and Power Requirements Versus Condenser Temperature

APPENDIX C

GN&C SUBSYSTEM RECOMMENDATIONS

SUBMITTED BY HONEYWELL
AEROSPACE DIVISION

NOV 1 1970

AEROSPACE DIVISION
MINNEAPOLIS OPERATIONS

C.E.L. NO. NR-TUG-70-02
DATE 2 November 1970

CUSTOMER ENGINEERING LETTER

ADM-409, REV. 2/67

PAGE 1 OF 2

North American Rockwell Corporation
Space Division
12214 South Lakewood Blvd.
Downey, California

Attention: Mr. G. Hanley
Engineering Manager
Space Tug Program

Reference: 1. Letter to C. E. Jennings dated September 17, 1970, Space Tug GN&C Effort (with attached statement of work)

2. CEL No. NR-TUG-70-01 to G. E. Hanley dated October 8, 1970, Preliminary Equipment List for the Space Tug GN&C.

Subject: FINAL REPORT - SPACE TUG GN&C

Dear Sir:

The attached report summarizes Honeywell's effort in supporting NR to define a Guidance, Navigation and Control (GN&C) system to fulfill the requirements of a Space Tug mission as outlined in the Reference 1 Statement of Work. This report in conjunction with the Reference 2 Preliminary Report forms the total system description with appropriate assumptions and guidelines.

As stated in the preliminary report, the Space Tug GN&C definition takes advantage of work performed on the Space Shuttle Program. Eleven of the nineteen Space Tug equipments are identical to Space Shuttle hardware. Implicit in this approach is that the Tug performance requirements are similar to those of the Shuttle.

Included in the report are:

- system description
- equipment list (with size, weight and power)
- system cable weight
- detailed device descriptions with performance capabilities
- equipment usage per mission operation
- system performance estimates

The information provided satisfies the statement of work requirements plus additional information requests made by North American. Although the report

Space Division
North American Rockwell

fulfills the final data submittal requirement, additional information on the system will be supplied on request. Please contact the author, D. V. Wilson, on Extension 5832.

C. A. Moosbrugger
Project Engineer
GN&C Systems Integration

D. V. Wilson
Principal Development Engineer
Space Shuttle Navigation

J. W. Lundquist
Manager
Shuttle Program Administration

cc: G. McKee - NR
C. Jennings - NR

UNITERMS:

Space
Tug
GN&C
CEL
Final Report

ATTACHMENT

FINAL REPORT

SPACE TUG GN&C

1.0 INTRODUCTION

In support of the NR pre-phase A Space Tug Study, Honeywell has performed a Guidance, Navigation and Control (GN&C) Study. The purpose of the study is to define the total GN&C equipment to be used over the total Space Tug mission regime. This includes both manned and unmanned operations with emphasis placed on unmanned, automatic operation. The depth of the system definition presented here is felt to be consistent with that of a pre-phase A program.

2.0 SUMMARY

This report provides additional system description of the Space Tug GN&C to support the Reference 2 preliminary report.

A system description is presented for both a fail operational/fail operational/fail safe (FO/FO/FS) and a FO/FS redundancy requirement.

The size, weight and power of each LRU in the system is also given. Although this information was presented in Reference 2, it is included here because of refinements made to some of the devices. A summary of the results are shown be

	Redundancy Requirement	
	FO/FO/FS	FO/FS
Size	44,655 in^3	33,157 in^3
Weight	945.3 lb.	673.6 lb.

A discussion of cable weight is also included since this usually represents a significant fraction of the total system weight. The discussion covers 1) the data bus and associated line coupling units, 2) power cabling and 3) actuator cabling.

A description of each device is also included with performance capabilities given for the sensors. In some cases, the device descriptions are not known in detail.

The last section describes equipment usage during various mission phases or operations. This information is included to facilitate estimating power and for the development of power time lines. In addition, this section also supplies estimates of navigation performance capability based on the accuracies presented for the sensors.

3.0 Discussion

3.1 System Description

3.1.1 Block Diagram

The system proposed for the Space Tug is similar in concept to that currently used in the Space Shuttle program. In fact, many of the equipment types are identical to those in the Shuttle System and, thus, will enjoy the very significant benefit of being space qualified. This of course assumes that the Space Tug GN&C performance requirements are not significantly different from that of the Space Shuttle.

All devices which require interface with the central GN&C computer communicate to the computer via a common data bus. Figures 1 and 2 illustrate the concept for both FO/FO/FS and a FO/FS redundancy requirement. The GN&C equipment is shown to be distributed between the Intelligence Module (IM) and the Crew Module (CM), with an interface with the propulsion module. All required GN&C processing is accomplished in the IM. Communication of the computer with the individual Line Replaceable Units (LRU) is maintained under the control of the computef. During a particular mode of operation, each LRU connected to the data bus is sequentially interrogated by the computer. (The interrogation rates may be different depending on the mission phase.) If data is to be transmitted to a particular LRU, the data word, preceded by the device address, is serially clocked onto the data bus. The device after decoding the address accepts the data and proceeds to act upon it. Since each device address is unique, there is no possibility that other devices will act on the data. If data is required from a particular device, a short word containing the device address and a control word is clocked onto the data bus. The data from the device is then clocked onto the bus under control of the computer clock and intercepted by the computer. This time sharing approach using a common signal interface bus provides a very significant reduction in the amount of spacecraft cabling that would otherwise be required by the more conventional wiring approaches.

Redundancy of devices is illustrated in the figures by the multiplicity of blocks. The GN&C computer provides automatic failure detection and switch out of a malfunctioning circuit using voting techniques among the redundant circuits.

3.1.2 Size, Weight and Power

The size, weight and power of each device in the system is given in Tables I and II for both FO/FO/FS and FO/FS redundancy requirements. These tables are nearly identical to those in the Reference 2 preliminary report except for refinements in the following LRU's:

- IMU (FO/FS)
- TVC Gimbal Servo
- Horizon Sensor
- MFED Power Interface
- Radar Altimeter
- GN&C Power Conditioning and Distribution

The equipments numbered 2-9 in the table are currently not a part of the Shuttle GN&C baseline system. However, they may be included in the baseline when Shuttle sensor trades studies are completed.

Equipments numbered 1, 10-19 are part of the Shuttle baseline. The size, weight, power and LRU description given in this report represents the Oct. 1, 1970 baseline description for these equipments. As greater GN&C definition emerges, resulting from Shuttle studies, these parameters will be refined. It is suggested that the NR Space Tug Team consult with the NR Space Shuttle team, occasionally, to keep current with the GN&C baseline LRU's applicable to the Space Tug Program.

The tables do not indicate the additional weight due to spacecraft cabling. This is covered separately in the next section.

A discussion of the rationale for selecting the LRU quantities given in the tables is included in the Appendix.

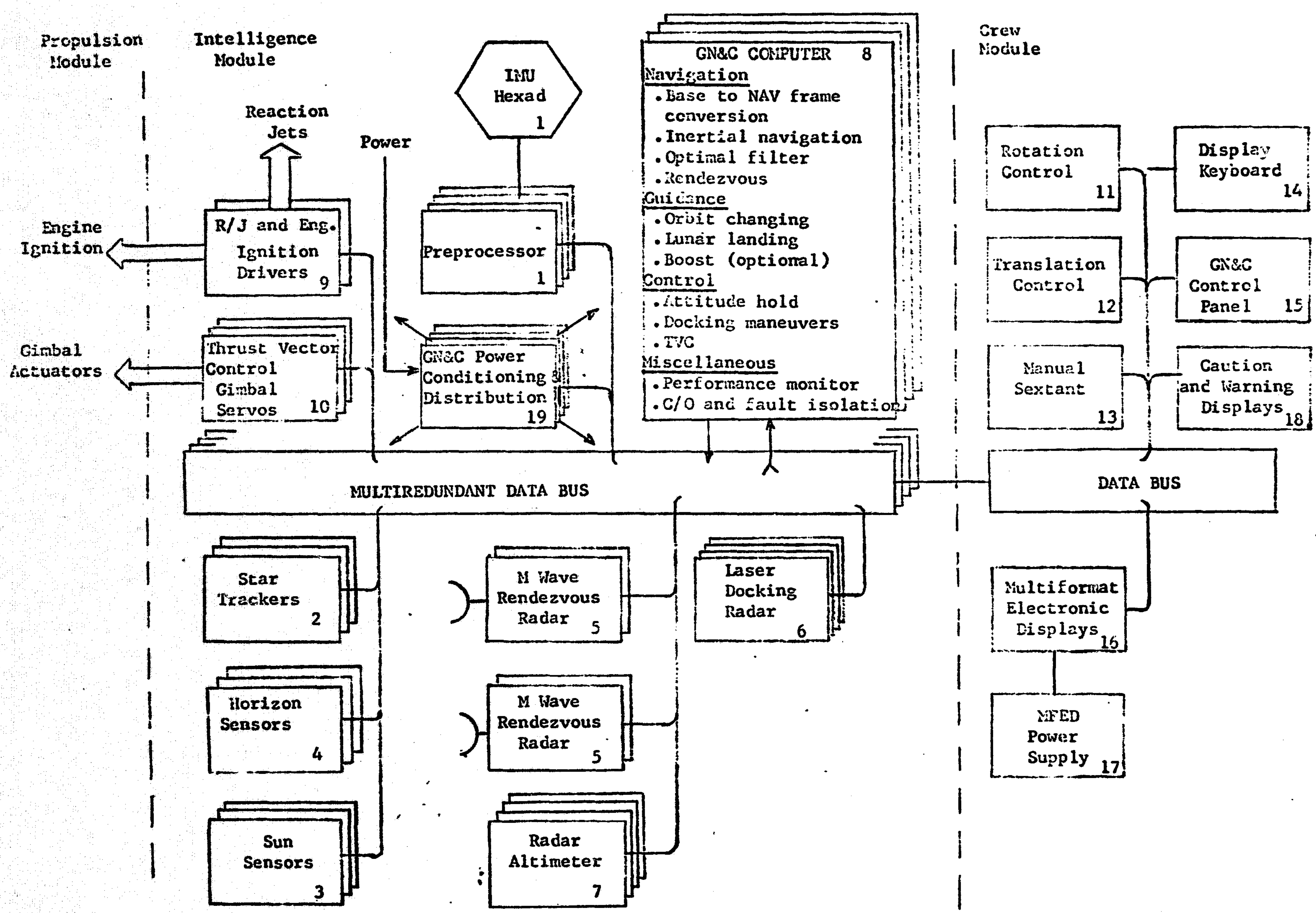

FIGURE 1 SPACE TUG GN&C (FO/FO/FS)

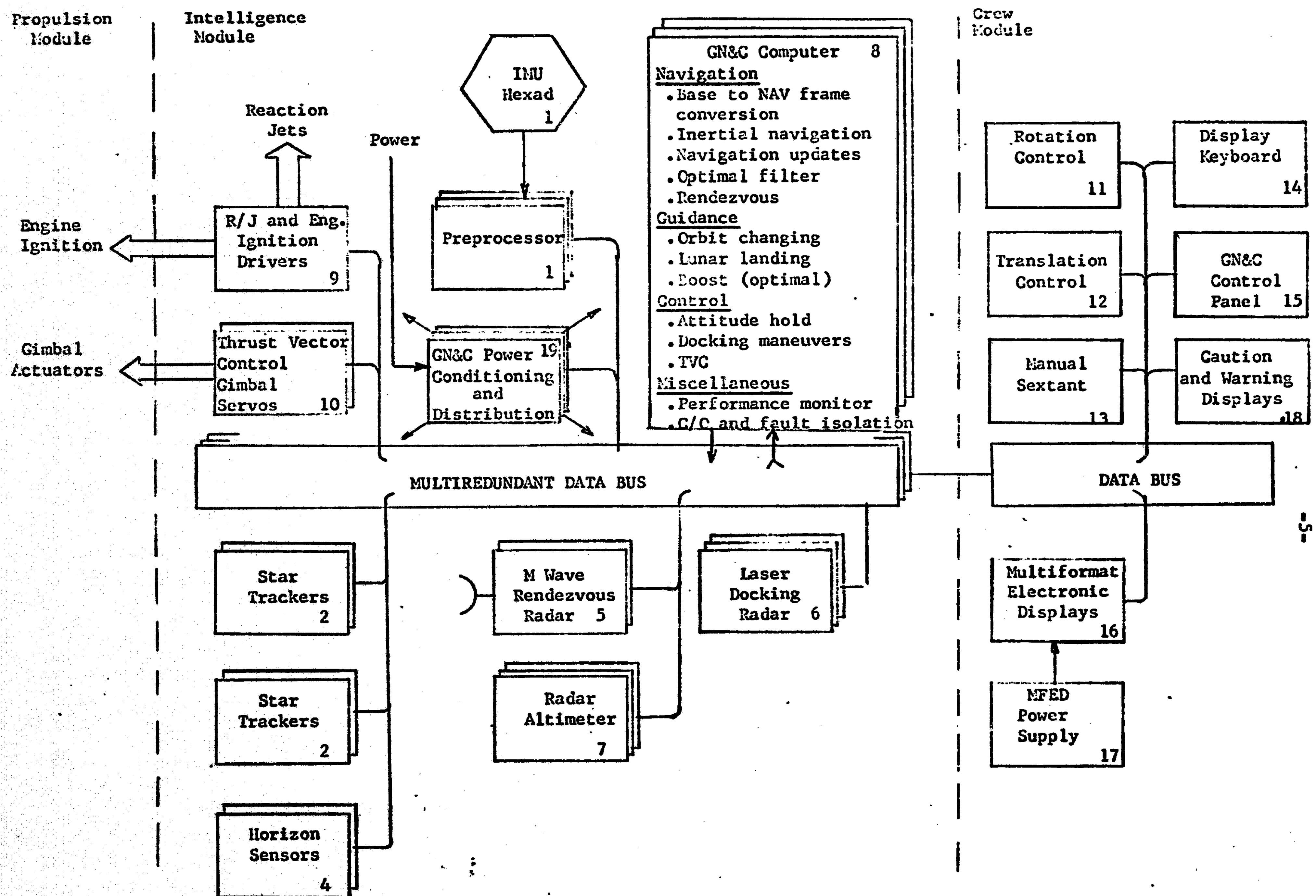

FIGURE 2 SPACE TUG GN&C (FO/FS)

TABLE 1 SPACE TUG GN&C EQUIPMENT

REDUNDANCY REQUIREMENT - FO/FO/FS

LRU	LOC	QTY	WEIGHT LB/DEVICE	VOLUME IN^3/DEVICE	POWER WATT/DEVICE	TOTAL WEIGHT	TOTAL VOLUME	TOTAL* POWER
1. IMU HEXAD	IM	1	111	3,570	243	111	3,570	243
2. STAR TRACKER	IM	3	7.5	264	10	22.5	792	30
3. SUN SENSOR	IM	3	3	18	2	9	54	6
4. HORIZON SENSOR	IM	2	45	767	38	90	1,534	76
5. μ WAVE REND. RADAR	IM	2	110	9,170	600	220	18,340	1,200
6. LASER DOCKING RADAR	IM	3	30	1,480	30	90	4,440	90
7. RADAR ALTIMETER	IM	4	10.2	2,025	45	40.8	8,100	180
8. GN&C COMPUTER	IM	4	33	1,200	45	132	4,800	180
9. R/J AND ENGINE IGN. DRIVER	IM	2	20	530	16 (quiescent)	40	1,060	32
10. TVC GIMBAL SERVO	IM	1	25	620	58	25	620	58
11. ROTATION CONTROL	CM	1	12.5	215	12	12.5	215	12
12. TRANSLATION CONTROL	CM	1	9	175	20	9	175	20
13. MANUAL SEXTANT	CM	1	38	2,214	14.6	38	2,214	14.6
14. DISPLAY KEYBOARD	CM	1	7.5	185	37	7.5	185	37
15. GN&C CONTROL PANEL	CM	1	5	81	25	5	81	25
16. MULTIFORMAT ELECT DISPLAYS	CM	1	54	1,950	52	54	1,950	52
17. MFED POWER/INTERFACE	CM	1	14	425	22	14	425	22
18. CAUTION & WARNING DISPLAYS	CM	1	15	300	75	15	300	75
19. GN&C PWR COND/DISTRIB	IM	1	10	200	1	10	200	1
						945.3	44,655	

*All devices are turned on

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North American Rockwell

TABLE II SPACE TUG GN&C EQUIPMENT

REDUNDANCY REQUIREMENT - FO/FS

	LRU	LOC	QTY	WEIGHT LB/DEVICE	VOLUME IN^3/DEVICE	POWER WATT/DEVICE	TOTAL WEIGHT	TOTAL VOLUME	TOTAL* POWER
1.	IMU	IM	1	92.5	2,970	203	92.5	2,970	203
2.	STAR TRACKER	IM	2	7.5	264	10	15	528	20
3.	SUN SENSOR	IM	2	3	18	2	6	36	4
4.	HORIZON SENSOR	IM	1	45	767	38	45	767	38
5.	μ WAVE REND. RADAR	IM	1	110	9,170	600	110	9,170	600
6.	LASER DOCKING RADAR	IM	2	30	1,480	30	60	2,960	60
7.	RADAR ALTIMETER	IM	3	10.2	2,025	45	30.6	6,075	135
8.	GN&C COMPUTER	IM	3	33	1,200	45	99	3,600	135
9.	R/J AND ENGINE IGN DRIVER	IM	1	20	530	16	20	530	16
10.	TVC GIMBAL SERVO	IM	1	16.5	413	38.8	33	826	77.6
11.	ROTATION CONTROL	CM	1	12.5	215	12	12.5	215	12
12.	TRANSLATION CONTROL	CM	1	9	175	20	9	175	20
13.	MANUAL SEXTANT	CM	1	38	2,214	14.6	38	2,214	14.6
14.	DISPLAY KEYBOARD	CM	1	7.5	185	37	7.5	185	37
15.	GN&C CONTROL PANEL	CM	1	5	81	25	5	81	25
16.	MULTIFORMAT ELECT DISPLAYS	CM	1	54	1,950	52	54	1,950	52
17.	MFED POWER/INTERFACE	CM	1	14	425	22	14	425	22
18.	CAUTION & WARNING DISPLAYS	CM	1	15	300	75	15	300	75
19.	GN&C PWR COND/DISTRIB	IM	1	7.5	150	0.8	7.5	150	1
							673.6	33,157	

*All devices are turned on

3.1.3 Device Cable Weight

It has been found in the past that the weight of cable harness interconnecting the various devices of a system is a very significant fraction of the total system weight. This is one of the major reasons that a data bus approach is used in the Space Shuttle and is proposed for the Space Tug. The data bus, however, replaces only the signal interconnections between devices. There still remains a large amount of conventional cabling necessary to provide power to the various devices as well as provide analog or discrete signals to actuators. In all cases, the estimated weight of a cable is given in pounds per 1000 feet of cable since it is not presently known how the devices will be distributed on the Space Tug. The weight given in this fashion allows NR to vary equipment locations and compute a new cabling weight for each configuration.

3.1.3.1 Data Bus - The IM data bus as shown in Figure 1 is quad redundant for the FO/FO/FS requirement and as shown in Figure 2 is triple redundant for the FO/FS requirement. The CM data bus is not redundant in both cases. The weights are as follows:

IM (FO/FO/FS) - 80 lbs/1000 ft.
IM (FO/FS) - 60 lbs/1000 ft.
CM - 20 lbs/1000 ft.

In addition, the data bus requires line coupling units (LCU) to interface the bus with each redundant device. The size, weight and power of each LCU is 2cu.in.,0.25 lb. and 0.1 watts respectivel A summary of LCU requirements is given below.

	Qty.	Size Cu. In.	Weight lbs	Power watts
IM (FO/FO/FS)	41	82	10.25	4.1
IM (FO/FS)	29	58	7.25	2.9
CM	7	14	1.75	0.7

3.1.3.2 Power Cabling - The size of the power cabling is a function of the wire size and number of wires to each redundant device from the GN&C Power Conditioning and Distribution Assembly.

Wire size is a function of the current carrying capacity required as well as the allowable voltage drop due to wire resistance. The current required can be determined by dividing the power required by each redundant device by the voltage level to the device (in most cases 28 VDC is assumed). Since line voltage drop is a function of distance which is not known, a conservative approach to sizing the wire is taken. It is assumed that the wire capacity must not exceed 300 circular mils per ampere.

The number of power wires to each device is a function of the level of redundancy within the device. In each case, two wires to each redundant circuit is assumed, a power line and a return. Table III summarizes the power cabling requirements for the system. In calculating the cable weight for a particular device, the distance between the device and the GN&C Power Conditioning and Distribution assembly must first be estimated.

3.1.3.3 Actuator Cabling - The R/J and Engine Ignition Driver Assemblies plus the Thrust Vector Control Gimbal Servo Assembly interface with the reaction jets and gimbal servo motors directly. Each R/J and Engine Ignition Driver Assembly provides 16 ON OFF signals, one to each reaction jet. These are bundled in groups of 4 (with each signal having a return) since the jets are clustered in groups of four. This assembly also provides engine firing commands to the propulsion engines. Assuming three redundant ignition systems per engine, each cable contains 6 wires (hi and return). The cable requirements are summarized in Table IV.

The TVC Gimbal Servo Assembly provides servo control signals to six actuators per engine (3 per gimbal). In addition, it accepts 6 gimbal position and 6 gimbal velocity transducer signals (3 per gimbal) and provides a voltage reference to the transducers. Table IV summarizes the cable requirements. It is seen that each cable contains 48 wires.

Figure 3 illustrates the cable interfaces between the driver devices and the actuators and may be used in estimating actuator cable weights.

-11-

TABLE III LRU POWER CABLING (FO/FO/FS)

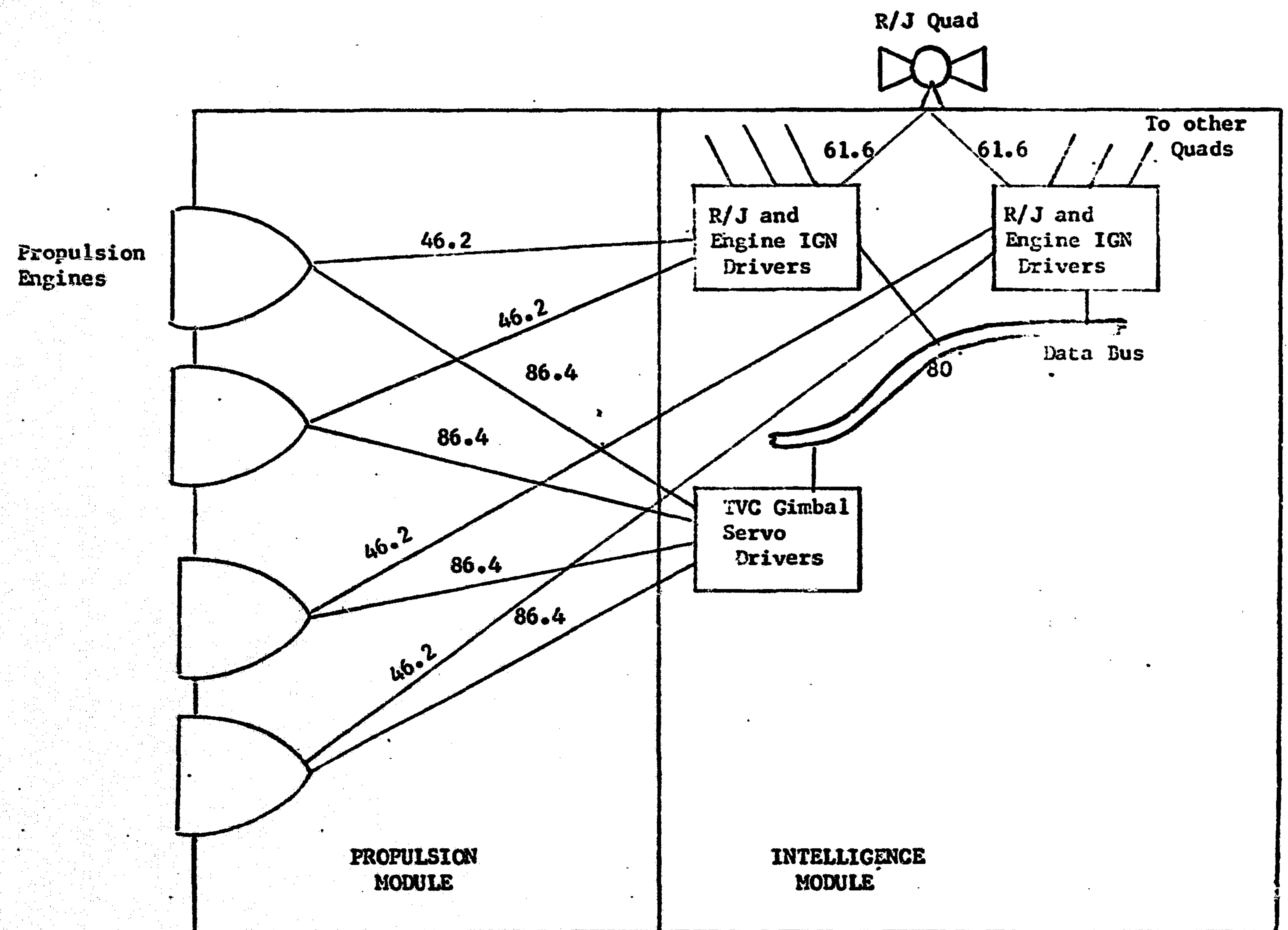
LRU	LOC	WIRE SIZE	NO. OF WIRES	WEIGHT* LBS/1000 FT	QTY. OF LRU'S
1. IMU HEXAD	IM	14	8	112	1
2. STAR TRACKER	IM	26	2	3.6	3
3. SUN SENSOR	IM	26	2	3.6	3
4. HORIZON SENSOR	IM	26	2	3.6	3
5. μ WAVE REND. RADAR	IM	14	4	56	2
6. LASER DOCKING RADAR	IM	24	2	5	3
7. RADAR ALTIMETER	IM	24	2	5	4
8. GN&C COMPUTER	IM	22	2	7	4
9. R/J AND ENGINE IGN DRIVER	IM	26	8	14.4	2
10. TVC GIMBAL SERVO	IM	26	16	28.8	1
11. ROTATION CONTROL	CM	26	2	3.6	1
12. TRANSLATION CONTROL	CM	26	2	3.6	1
13. MANUAL SEXTANT	CM	26	2	3.6	1
14. DISPLAY KEYBOARD	CM	24	2	5	1
15. GN&C CONTROL PANEL	CM	24	2	5	1
16. MULTIFORMAT ELECT DISPLAYS	CM	22	2	7	1
17. MFED POWER/INTERFACE	CM	26	2	3.6	1
18. CAUTION & WARNING DISPLAYS	CM	20	2	10.4	1
19. GN&C PWR COND/DISTRIB	IM	26	16	14.4	1

* Per LRU

TABLE IV R/J AND PROPULSION ENGINE CABLING (FO/FO/FS)

LRU	LOC	WIRE SIZE	NO.OF WIRE	WEIGHT * LBS/1000 FT.	NO. OF CABLES PER LRU	QUANTITY OF LRU'S
R/J AND ENGINE IGN DRIVER	IM					2
• R/J	IM	18	8	61.6	4	
• ENGINE IGN	PM	18	6	46.2	2	
TVC GIMBAL SERVO	IM					1
• ACTUATORS**	PM	26	12	21.6		
• TRANSDUCER INPUTS	PM	26	12	21.6		
• TRANSDUCER OUTPUTS	PM	26	24	43.2		
		TVC TOTAL		86.4	4	

* Per Cable
** Assume hydraulic actuators similar to Space Shuttle



Cable weight numbers in lbs/1000 ft.

FIGURE 3 REACTION JET AND PROPULSION ENGINE CABLING

3.2 Device Description

The following describes the functions of each LRU as well as the contents. In all cases each LRU contains at least one acquisition, control and test unit (ACT) to act as a buffer between the operational part of each LRU and the Line Coupling Units. The ACT unit is standard equipment used on the Shuttle program. It decodes the address field and function code of the data word transmitted on the bus and transfers data onto and off of the data bus.

The redundancy provided in the CM Display & Control equipment is generally more than is necessary for the Space Tug application. However, this allows the use of identical shuttle equipment which will have been space qualified and available for use in the Space Tug Program.

3.2.1 IMU Hexad - The IMU Hexad consists of a strapdown sensor package containing gyros and accelerometers and quad redundant preprocessors (FO/FO/FS). It provides an incremental measurement of vehicle velocity and attitude with respect to a skewed reference frame. The preprocessor transforms the skewed reference frame and, in addition, provides automatic failure detection and correction of failed sensors. A gross description of the contents of the IMU Hexad follows:

- 6 GG334 gas bearing gyros
- 6 GG177 accelerometers
- 12 Pulse rebalance electronic loops
- 4 Self contained power supplies
- 4 Preprocessors
- 12 Temperature control loops
- 4 ACT units

The above is reduced by one level of redundancy for FO/FS.

3.2.2 Star Tracker - The Star Tracker consists of a non-gimballed tracking head with an associated electronics assembly. It operates both in an acquisition and track mode. It provides two axis position of a star with respect to its bore sight to an accuracy of ±0.05° over a cone angle of 20 degrees. An automatic sun shutter provides protection of the tracker head during sun viewing periods. The contents of the Star Tracker include:

- 1 Image dissector tube
- 1 Sun shutter with sun sensor and control electronics
- 2 Deflection amplifiers (one for each axis)
- 1 AGC circuit
- 1 Acquisition/track logic circuit
- 1 Timing and sweep circuit
- 1 ACT unit

3.2.3 Sun Sensor - The Sun Sensor Assembly consists of a sensor head using silicon quad cells with a shadow mask plus supporting electronics. The sensor provides two axis of sun position information with respect to the boresight over a square FOV of ±15 degrees in each axis. The accuracy is better than 0.075° in each axis. The electronics removes the cosine law effect from the sensor output, provides automatic gain control over a dynamic range of 1000 and resolves the output of the four silicon cells into two attitude error signals. The contents of the Sun Sensor Assembly include:

- 1 Quad silicon cell sensor head
- 1 Signal processing circuit
- 1 ACT unit

3.2.4 Horizon Sensor - The horizon sensor, of the edge tracking variety, utilizes 4 heads to provide two axis attitude errors of the vehicle with respect to the lotal vertical. A common electronics package processes the individual sensor head signals to provide body attitude error signals. Since three heads are sufficient to provide both error signals, the fourth head provides redundancy. The electronics package is also dual redundant to maintain complete redundancy of the Horison Sensor subsystem. The system is capable of measuring attitude to about ±0.05 degrees over an attitude range of 80 to 25,000 nautical miles. The system contains the following elements:

- 4 Tracking-mirror subsystems
- 4 Optical-angle readouts
- 4 IR horizon-tracking subsystems
- 4 Sun rejection detectors
- 4 Heater circuits
- 2 Input power preregulators
- 2 Pitch/roll output circuitry
- 2 ACT units

3.2.5 Microwave Rendezvous Radar

The S-Band Solid State Rendezvous Radar is capable of tracking both cooperative and non-cooperative targets. The two axis, electro-mechanically steered antenna is comprised of a 16 element beam forming array (provides graceful degradation). Dual electronic assemblies are provided for each antenna assembly. For tracking cooperative targets, the system operates in a CW tone ranging mode to provide a range, range rate and angle accuracies of about ±0.1% or 50 feet, ±1 ft/sec. and ±0.2 degree, respectively, over a range of 500 ft. to 1500 nm. For tracking non-cooperative targets, an intermittent CW mode is used to provide range, range rate and angle accuracies of ±5% or 25 feet, ±5 ft/sec. and ±0.4 degrees, respectively, over a range of 500 ft. to 30 nm. The cooperative target requires a transponder. Elements of the rendezvous system are given below:

- 1 Sixteen element antenna array
- 1 Two axis mechanically gimballed assembly
- 2 Electronics assemblies
- 2 ACT units

3.2.6 Laser Docking Radar

The Laser Docking Radar uses a laser transmitter to provide a pulsed, narrow light beam which is then scanned by a beam steerer over a scan field of view of 30 degrees by 30 degrees. A receiver using an image disector tube (IDT) senses the return beam from the passive vehicle. The transmitter/receiver combination on the active docking vehicle then provides a measure of range, range rate and angle between the active and passive vehicles.

For a cooperative target, optical corner cube reflectors are mounted in a known geometrical pattern on the passive vehicle to return the laser beam. The non-cooperative target depends on skin reflection. The system provides a range accuracy of ±0.02% or 10 cm, range rate accuracy of ±1.0% or 0.5 m/sec. and angle accuracy of ±0.02 degree transverse axis and ±1.0 degree longitudinal axis over a range of 0-75 nautical miles.

The contents of the Laser Docking Radar are listed below:

- 1 Laser transmitter
- 1 Beam steerer
- 1 Scanning optical detector (IDT)
- 1 Receiver optics
- 1 Ranging circuit
- 1 Threshold circuit
- 1 Timing circuit
- 1 Target selection logic
- 1 Output processor
- 1 ACT unit

3.2.7 Radar Altimeter

The Radar Altimeter uses pulsed radar, leading edge tracking to provide an altitude and altitude rate from 0 to 50,000 feet altitude. The altitude accuracy is less than ±(2 feet + 0.1% of altitude); altitude rate is less than ±(3 ft/sec. + 3% of rate). The major components include:

- 1 Transmitter
- 1 Receiver
- 1 Tracker
- 1 Power supply
- 2 Horn antennas (transmit and receive)
- 1 ACT unit

3.2.8 GN&C Computer

It is assumed that a computer with a 32K memory and 24 bit word length is sufficient to perform all of the Space Tug GN&C functions as well as perform checkout and fault isolation of the entire system. The computer interfaces with the data bus to control the operation of all LRU's. It provides automatic mode sequencing, controls the data rate to and from the sensors and determines what LRU's are turned on and what level of redundancy must be provided for a particular mission operation. The computer is modularized so that additional memory modules may be added. The following modules comprise the GN&C computer:

- 1 Memory section
- 1 CPU
- 1 I/O section (includes bus control unit)
- 1 Power supply

(Although not shown as part of the system, a communications receiver must tie into the data bus to provide computer command signals during manual remote operation.)

3.2.9 R/J and Engine Ignition Control

The R/J and Engine Ignition Control Assembly provides control of the valve solenoids supplying O_2/H_2 to the 16 reation jet combustion chambers. Ignition circuits in the assembly ignite the propellant mixture in both the reaction jet system and the main propulsion system. The contents of each assembly include:

- 16 "ON-OFF" driver stages
- 16 Transient suppression circuits
- 22 Ignition circuits
- 2 ACT units

3.2.10 TVC Gimbal Servo

The Thrust Vector Control Gimbal Servo Assembly provides proportional control of the actuators which position the two axes of each propulsion engine in the Propulsion Module. For a FO/FO/FS requirement, 3 actuators per gimbal for each of the 4 engines are used. In addition, 3 gimbal position and 3 gimbal rate transducers on each gimbal provide redundant feedback to stabilize the system. The device contains:

- 24 Servo loop electronics channels
- 4 ACT units

3.2.11 Rotation Controller

Since there is only one pilot station in the Crew Module, one Rotation Controller is provided. The controller permits manual control of the Space Tug attitude in all three axes. The contents include:

- 1 Stick
- 12 Breakout switches (4 per axis)
- 6 Linear transducers (2 per axis)
- 2 ACT units

3.2.12 Translation Control

The Translation Control permits manual control of the Space Tug translational position along all three axes. The device includes:

- 1 T-handle
- 24 Translation switches (8 per axis)
- 4 Abort switches (manual abort during boost)
- 4 ACT units

3.2.13 Manual Sextant

The Manual Sextant can be used for both star sightings and land mark tracking to provide IMU alignments and near earth navigation as a backup to the star sensor, sun sensor and horizon sensor. Star sighting accuracies are of the order of 10 arc sec. while landmark tracking approaches 30 arc sec. The contents of this LRU is as follows:

- 1 Sextant
- Two DOF gimbal assembly
- 2 Dual redundant servo loops
- 1 DC power supply
- 1 Control panel
- 1 ACT unit

3.2.14 Display Keyboard

The Display Keyboard permits manual communication with the GN&C processor for manual commands, manual checkout and status, data entry or program changes, etc. It contains switches for introducing data and E/L displays for readout. The contents of the display include:

- 29 Dual switches
- 4 E/L displays
- 2 ACT units

3.2.15 GN&C Control Panel

The GN&C Control Panel provides manual selection of GN&C modes. The panel contains:

- 20 Illuminated switches
- 2 ACT units

3.2.16 Multiformat Electronic Displays

The multiformat display uses a cathode ray tube to provide a highly versatile display system for portraying either graphically or alpha/numerically GN&C parameters for the various modes of operation. The contents include:

- 1 CRT
- 2 Processors
- 2 ACT units

3.2.17 MFED Power/Interface

This LRU supplies power and provides auxilliary electronics to the Multiformat Electronic Displays. It contains:

- 1 Power supply
- 1 Auxilliary electronics circuit

This device is hardwired to the MFED. It was not made a part of the MFED because the total weight would exceed acceptable levels for a LRU.

3.2.18 Caution and Warning Display

The Caution and Warning Display provides annunciation of critical failures during manual or manned, automatic operation. The detailed contents have not yet been determined because of the strong dependence on failure mode and effects analysis which is yet to be performed.

3.2.19 GN&C Power Conditioning and Distribution

This LRU distributes the power from the main power source to all LRU's. It also conditions/filters the raw airframe DC and AC power. Its contents include:

- 4 Transient and EMI suppression filters
- 4 DC regulator circuits
- 4 ACT units

3.3 Mission Operation

This section discusses the usage of the GN&C equipment during the various mission phases. Estimates of navigation accuracy are also given.

3.3.1 Equipment Usage

A summary of the equipment necessary to perform various Space Tug GN&C operations are tabulated in Tables V and VI. Table V indicates equipment usage during pure automatic operation (whether manned or unmanned) and Table VI lists equipment usage when all operations are performed manually. Both tables assume that the PM is attached. During automatic IM operation only, Table V can be used with the exception that the TVC gimbal servo assembly is not used.

These tables in conjunction with Tables I and II can be used to derive GN&C power time lines for both the FO/FO/FS and FO/FS redundancy requirements once the mission time line is known. In computing power, it is assumed that automatic operation requires that all redundant elements of a piece of equipment being used be turned "on" to facilitate automatic failure detection and correction. For manual operation, it is assumed that during non-critical operations, only one element of each equipment being used is turned "on" since the crew can manually detect and correct failure. However, during manual time critical operations such as Delta-V corrections and lunar landing and ascent, all redundant elements should be turned "on".

In addition to establishing power time lines, the tables can be used to define the equipment necessary for reduced mission capability. For instance, deleting the lunar operations results in deleting the radar altimeter.

TABLE V EQUIPMENT USAGE DURING AUTOMATIC OPERATION

MISSION OPERATION / LRU	Boost Guidance	Paring Orbit Navigation	Orbit Changing	Synch. Orbit Navigation	IMU Alignment	Checkout	Rendezvous	Docking	Translunar Coast Navigation	ΔV corrections	Lunar Orbit	Lunar Landing	Lunar Ascent
1. IMU HEXAD	X	X	X	X	X	X	X	X	X	X	X	X	X
2. STAR TRACKER	O	X	O	X	X	X	O	O	X	X	X	O	O
3. SUN SENSOR	O	O	O	O	X	X	O	O	X	X	X	O	O
4. HORIZON SENSOR	O	X	O	O	O	X	O	O	O	O	O	O	O
5. μ WAVE REND. RADAR	O	O	O	O	O	X	X	O	O	O	O	O	O
6. LASER DOCKING RADAR	O	O	O	O	O	X	O	X	O	O	O	O	O
7. RADAR ALTIMETER	O	O	O	O	O	X	O	O	O	O	O	X	X
8. GN&C COMPUTER	X	X	X	X	X	X	X	X	X	X	X	X	X
9. R/J AND ENGINE IGN. DRIVER	O	X	X	X	X	X	X	X	X	X	X	X	X
10. TVC GIMBAL SERVO	O	O	X	O	O	X	X	O	O	X	O	X	X
19. GN&C PWR COND/DISTRIB	X	X	X	X	X	X	X	X	X	X	X	X	X

X = Power on

O = Power off

TABLE VI EQUIPMENT USAGE DURING MANUAL OPERATION

LRU \ MISSION OPERATION	Boost Guidance	Parking Orbit Navigation	Orbit Changing	Synch. Orbit Navigation	IMU Alignment	Checkout	Rendezvous	Docking	Translunar Coast Navigation	ΔV Corrections	Lunar Orbit	Lunar Landing	Lunar Ascent
1. IMU HEXAD	X	X	X	X	X	X	X	X	X	X	X	X	X
2. STAR TRACKER	O	O	O	O	O	X	O	O	O	O	O	O	O
3. SUN SENSOR	O	O	O	O	O	X	O	O	O	O	O	O	O
4. HORIZON SENSOR	O	O	O	O	O	X	O	O	O	O	O	O	O
5. μ WAVE REND. RADAR	O	O	O	O	O	X	O	O	O	O	O	O	O
6. LASER DOCKING RADAR	O	O	O	O	O	X	O	O	O	O	O	O	O
7. RADAR ALTIMETER	O	O	O	O	O	X	O	O	O	O	O	X	X
8. GN&C COMPUTER	X	X	X	X	X	X	X	X	X	X	X	X	X
9. R/J AND ENGINE IGN. DRIVER	O	X	X	X	X	X	X	X	X	X	X	X	X
10. TVC GIMBAL SERVO	O	O	X	O	O	X	X	O	O	X	O	X	X
11. ROTATION CONTROL	O	X	X	X	X	X	X	X	X	X	X	X	X
12. TRANSLATION CONTROL	O	O	X	O	O	X	X	X	O	X	O	X	X
13. MANUAL SEXTANT	O	X	O	X	X	X	X	X	X	O	X	O	O
14. DISPLAY KEYBOARD	O	X	X	X	X	X	X	X	X	X	X	X	X
15. GN&C CONTROL PANEL	O	X	X	X	X	X	X	X	X	X	X	X	X
16. MULTIFORMAT ELECT DISPLAYS	O	X	X	X	X	X	X	X	X	X	X	X	X
17. MFED POWER/INTERFACE	O	X	X	X	X	X	X	X	X	X	X	X	X
18. CAUTION & WARNING DISPLAYS	O	X	X	X	X	X	X	X	X	X	X	X	X
19. GN&C PWR COND/DISTRIB	X	X	X	X	X	X	X	X	X	X	X	X	X

X = Power On

O = Power off

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3.3.2 System Performance

The following is a brief discussion of the performance capability of the equipment described in this report. In lieu of having definite performance requirements from which to define equipment, it is assumed that the equipment defined for the Space Shuttle program is sufficient to meet Space Tug requirements. Engineering judgement is used on those equipment not having a Space Shuttle counter part. Individual sensor accuracies have been previously discussed under Section 3.2 . This discussion uses the sensor accuracies to estimate navigation update accuracy, attitude update accuracy and Delta-V pointing error for the various mission operations.

3.3.2.1 Near Earth and Synchronous Orbit Navigation

Navigation updates for earth orbits use the star tracker and horizon sensor to establish orbital position during automatic operation. Using a ±0.05 degree accuracy for both sensors, a navigation accuracy of 4.2 nm (1σ) is projected.

During manual operation, the sextant is used. It is estimated that star sightings can be made to an accuracy of 0.003° while landmark tracking can be performed to an accuracy approaching 0.008°. In this case, updates on the order of 0.2 nm (1σ) can be expected.

3.3.2.2 IMU Alignment

IMU alignment, performed at any time during the space tug mission, uses the star tracker and sun sensor to provide inertial attitude during automatic operation. Based on sensor accuracies of 0.05 degree and 0.075 degrees for the star tracker and sun sensor, respectively, an alignment error of less than 0.1 degree per axis is projected. This accuracy can be improved upon considerably by using the manual sextant in which case the error approaches .005 degree per axis. Since the sextant and IMU are separated, it is expected that this accuracy would be difficult to achieve due to relative structural bending or misalignment between the LM and CM. Optical monitors linking the two systems would be required in order to achieve 0.005 degree accuracies.

3.3.2.3 Translunar Coast Navigation

During Translunar Coast or other deep space operations, it is assumed that earth based tracking would be used (Deep Space Network). The present state of development for this system yields accuracies on the order ±50 feet (1σ)

3.3.2.4 ΔV Corrections

Based on Apollo experience, delta velocity vector misalignments will be in the order of 0.667 degree (1σ) for propulsion engine burns of 60 seconds or less. The misalignment error improves for longer burns; approaching 0.43 degrees for burn times between 60 and 250 seconds.

References

1. Letter to C. F. Jennings dated September 17, 1970, Space Tug GN&C Effort (with attached Statement of Work).

2. CEL NO. NR-TUG-70-01 to G. E. Hanley dated October 8, 1970, Preliminary Equipment List for the Space Tug GN&C.

APPENDIX

REDUNDANCY RATIONALE

In interpreting a FO/FO/FS requirement for the Space Tug, it is assumed that during manned missions the occurrence of two critical failures is sufficient to abort the mission (this leaves two operational paths). This is similar to the Space Shuttle interpretation. Applied to the Space Tug, this means that generally equipment in the IM will be quad redundant except for those equipment that can be backed up manually in the CM.

As an example, the crew module contains a sextant which can be used for reference alignment and docking therefore only three each of star trackers, sun sensors, and docking radar systems are needed instead of four.

The IMU, computer and power conditioning and distribution unit are used during all unmanned and manned operations, consequently, these are quad redundant.

The TVC gimbal servo is triple redundant on each motor. Since there are four motors, any two of which may fail, the level of redundancy actually provided by the TVC gimbal servo is FO/FO/FS/FS.

The two R/J and engine ignition drivers provide quad redundancy assuming that maneuvers can be accomplished using one jet instead of two.

Each horizon sensor is dual redundant, thus, two of them provide quad redundancy. Actually, the manual sextant can back up the horizon sensor thereby providing a fifth level of redundancy. However, to delete one horizon sensor results in triple redundancy which is unacceptable. Therefore, two are needed.

Honeywell assumed that two rendezvous radar antennas are sufficient (FO/FS) with dual electronics for each antenna. Like the horizon sensor, the manual sextant can be used as a backup thus providing an extra level of redundancy. Rather than provide dual electronics for one antenna and single electronics for the other (thus providing quad redundancy with the sextant), it is felt that equipment similarity would be highly desirable and the extra redundancy level provided by two dual electronic assemblies should be tolerated.

The radar altimeter does not have a manual backup, consequently, four are required.

All of the equipment in the CM are not required to be redundant. However, a certain degree of redundancy is inherent in these equipments since they are identical to similar hardware in the Space Shuttle which is required to be redundant.

A FO/FS redundancy requirement is similar to the above except one less redundancy level is provided.

APPENDIX D

MANIPULATION REQUIREMENTS FOR SPACE TUG OPERATIONS

MANIPULATION REQUIREMENTS FOR SPACE TUG OPERATIONS

INTRODUCTION

The requirements and capabilities for performing useful work in space have been subjected to much study, and the work has been reported in detail.

The work includes manipulative ability of astronauts, unmanned remote control manipulative vehicles, and manned manipulative vehicles.

The reports on satellite maintenance requirements, astronaut rescue requirements, and logistics support to space stations apply generally to tug operations.

Tug, as a manned maneuverable vehicle, can be provided with manipulation capability by several means

- EVA from tug as a tender
- RMU from tug as a controller
- Manipulation by tug crew and tug-mounted arms

The objective of this analysis is to identify manipulative tasks that need to be accomplished and to evaluate candidate systems and their impact on tug design and operations. In the Prephase A tug study, the evaluations are somewhat speculative.

CLASSIFICATION

In the system involving space tug, there are several means for providing manipulative and remote handling tasks. These may be classified as special purpose and general purpose. General purpose methods require a flexibility and dexterity that is substantially greater than for special purpose methods. In addition, general purpose (dexterous) manipulation can be accomplished directly by man EVA or indirectly by RMU manipulators or tug manipulators. Special purpose manipulation and remote handling are provided by the tug's capability to couple to cargo and maneuver it and dock it to EOSS. Also, special purpose manipulation is provided by loading devices, booms and materials handling equipment in space stations.

The purpose of drawing these distinctions is to avoid argument based on pure definitions. More important, however, we will cover the situation where general purpose manipulative capability is required as back-up to primary special purpose methods to improve system operational reliability.

Table D-1 shows this classification scheme and examples of tasks. The general purpose methods identify candidate systems that can provide dexterous manipulative capability to tug operations. The tug/RMU and tug/manipulator options are classed as teleoperator systems.

The requirement for these systems as back-ups, or complements, to the special-purpose methods will be considered in the analysis.

TUG FUNCTIONS AND MANIPULATION

Some of the principal functions of interest are shown in Table D-2.

SPECIAL DESIGN REQUIREMENT

Three basic manipulator system design requirements are frequently ignored at the design concept stage of development. The impact of the following special requirements on system analysis and concept development is substantial.

Hovering and Coupling Compliance

The ability of astronauts and manipulator operators to grab moving objects has been demonstrated. However, the complexity of the task is severely limited by the relative velocity of motions between the terminal device and the work object.

Hovering and Docking

Considering the hovering and maneuverability of a tug vehicle and a destabilized vehicle, or a tumbling astronaut, it becomes evident that the energy and flight-control-sensor demand can become extremely high. Furthermore, the grapple, snare, or net device required for object acquisition must accommodate greater and greater demands on relative-position precision. Energy limitations, vehicle control capability, manipulator dexterity, and acquisition-tool accommodations establish the limitations to feasible tasks for such station keeping and fly-by operations. The acquisition relative velocities of objects with various masses and C.G. misalignments have been studied by many concerns. The conclusion is soon reached that the maneuvering capability between spacecraft and work objects is obviously limited by energy demand, pilot instruments, and pilot capability.

Table D-1. Manipulative Capability - Classification and Examples

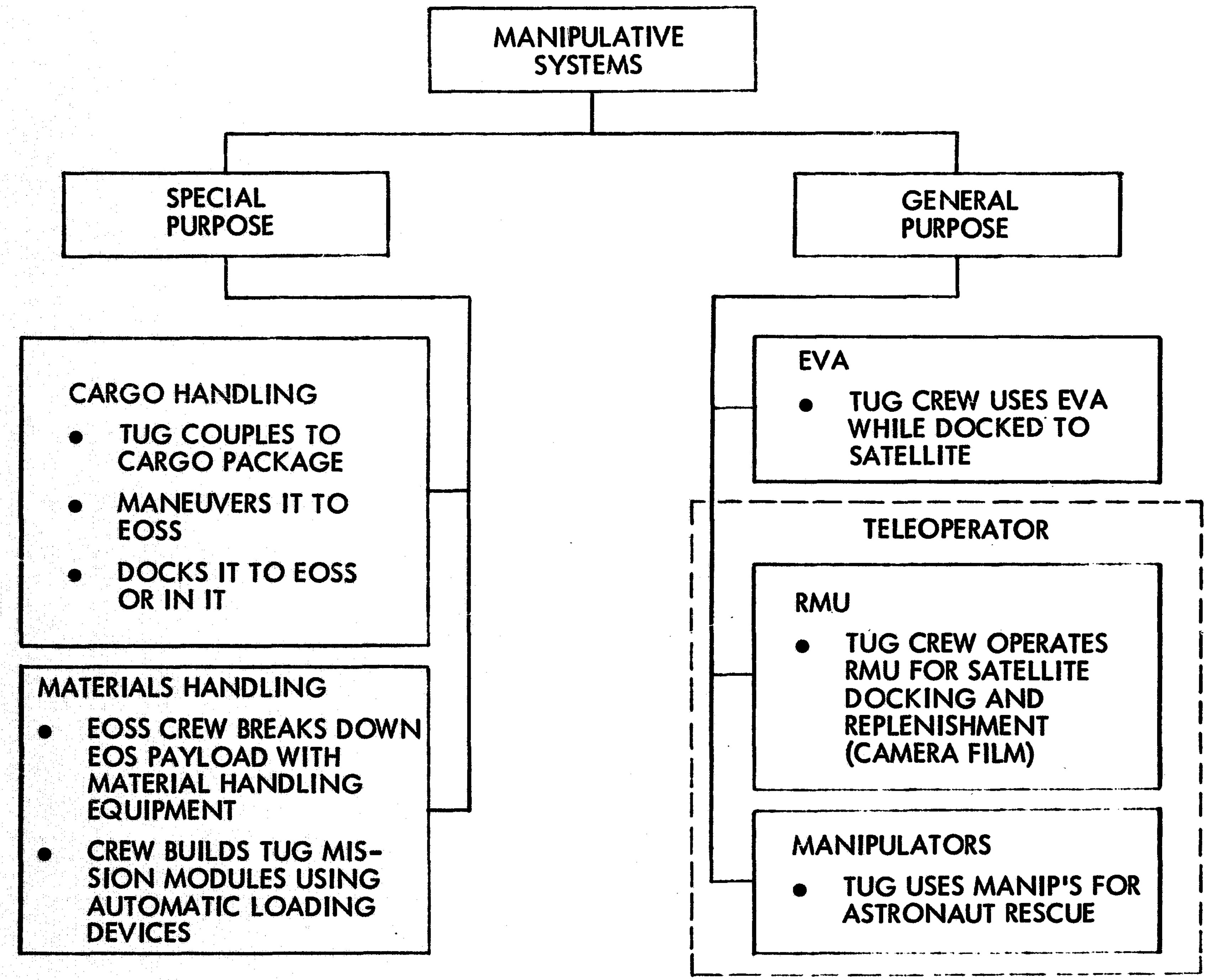

Table D-2. Functions Requiring Tug Manipulation

Tug Function	Manipulation Task
Observation, detection, and photo documentation	• Position cameras, radiation counters, lights, sunshades and mirror (fly-by or docked or grappled)
Transportation, positioning, and stabilization	• Acquire objects for transport
	• Provide reactionless release for deployment after docking
	• Operate spin/despin devices
Construction, rigging, and assembly	• Grapple structural elements for positioning for in-orbit assembly
	• Tether object to EOSS (clear of docking ports)
	• Deploy and attach lunar landing gear in orbit
Maintenance, experiment exchange, and replenishment	• Acquire satellite by grapple and exchange experiment package or solar paddle
	• Change film casette
Recovery	• Net/snare space junk for disposal
Retrieval	• Grapple inoperative satellite and hold in-transit to EOSS
Rescue	• Net/snare astronaut in tug rescue operations
	• Grapple space vehicle for EVA crew transfer into tug
Hazardous material handling and disposal	• Handle and dispose radioactive items—dangerous chemicals and explosives.

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This is combined with the same factors for the vehicle manipulator system and its operator in closing the gap between vehicle and work object for acquisition prior to coupling for dexterous manipulation or EVA tasks. The demand on the system is thus compounded to where the positioning precision, and the relative velocities in accomplishing capture (grappling or docking), must be conservative. Nonalignment of the two C.G.'s causes further complications.

It is, therefore, not likely that any degree of dexterous manipulation will be possible between a hovering spacecraft and a work object. Acquisition and docking, however, is an established capability.

Docked Manipulation

For dexterous manipulation tasks, the spacecraft must be coupled, or attached by grapples to the object being worked on. The compliance of this coupling directly affects the precision of manipulation that is possible.

Atomics International has found that the spring rate of the XYZ crane supporting a manipulator arm in hot cell work can be too low (too compliant). The dexterity of the manipulation can be substantially lowered by such excessive compliance. NR's Ocean Systems Operations found it necessary to dock a neutrally buoyant submersible to a rail to control compliance of the system sufficient to permit manipulation of valves and other tasks. It has been determined that grappling with one arm, while manipulating by another, will not provide the required fixity of the coupling. The neuter docking device for tug will provide a rigid attachment means. However, manipulator locations respective to the docking device are limited in two ways:

1. Docking is only possible with objects providing a compatible system.

2. Manipulative tasks can only be accomplished within the reach of the arms from this fixed point on the work object.

Therefore, this configuration is more of a special case than would be useful in evaluating general purpose manipulator systems for teleoperator applications

Visual Feedback

The principal (and often only) feedback loop for manipulation is human vision. For multiple points of view of the work task from within the spacecraft, both direct viewing and TV viewing are needed. However, multiple camera and stereo camera viewing have been used successfully.

In the task of capturing a non-cooperative object, the vehicle pilot and the manipulator operator will probably work in conjunction with each other. If viewports are provided, the field of view should overlap so that each operator can see the work at the same time. A practical consideration of viewport locations on tug may show the need for portable controls. The problem may be to find one or more viewports that bear on the target and then the pilot and operator would need their controls at those particular ports.

Lights and sunshades are required to control illumination levels. Sunshades may also be required to limit servo-temperature rise when operation is in direct sunlight. A manipulator arm is excellent for positioning cameras, lights, mirrors, and sunshades for manipulator tasks. A manipulator can remove such devices from their stored location, attach them to the work object, or hold and operate them.

The use of a manipulator arm as a camera positioner provides an interesting capability. It allows the tug to inspect itself in areas that may not be otherwise viewed except by EVA or another vehicle.

Mass Reactions

Some problems of mass reaction during docking were mentioned in discussion of the transfer from hovering to docking.

Another consideration relates to the use of reaction tools such as impact wrenches. This reduces the demand on the grappling system to minimize relative motion during manipulation.

Manipulators can also be used in conjunction with the tug's docking device to minimize disturbance to the work object during release of the object. In this case the two arms would maintain control of the object during release and separation of the drogue or the docking port. Thereafter the arms would release the grab points on the object without imparting forces to the object.

Summary

Three special system design requirements have been outlined in this section. They are summarized in Table D-3. These considerations are important in tug configuration evaluation and in candidate manipulator system evaluation.

Table D-3. Special System Design Requirements

1. Coupling Compliance
   - EVA and manipulator dexterity limited in accommodation of relative motion
   - Tug or RMU must lock rigidly to work object. EVA astronaut must tie to work object
2. Visual Feedback
   - Principal feedback loop. Direct viewing and/or multi-viewpoint TV required.
   - Illumination, sunshades, and mirrors needed.
3. Mass Reactions
   - High force/low reaction tools
   - Relative velocity: mass, limited for acquisition
   - Low-reaction acquisition and release required.

CANDIDATE MANIPULATOR SYSTEMS

As shown, it is apparent that tug can be provided with three manipulative options: tug-based EVA, tug-based-and-controlled RMU, and tug-mounted and controlled manipulator/grappling system.

Remote Maneuvering Unit

The GE space slave RMU is a general purpose teleoperator with arms, tools, propulsion, TV, lights, and communication links to control stations on earth, on EOSS or on tug. When control is from tug, the communication time-delay problem becomes small compared to the substantial problem of earth-based control of the RMU for manipulation on a planetary mission.

GE studies have established the requirements for manipulation and demonstrated feasibility by analysis and simulation. It can be argued that tug is the logical control base for an RMU. This option also meets the modular and kit concepts of the tug system since the RMU can be left at the EOSS, or even controlled by EOSS for local jobs, when not in use on tug.

With this configuration, tug could dock to a work object by either the docking port or manipulator arms. It could then deploy the RMU for exterior inspection and work while the tug crew entered the satellite and performed interior maintenance task.

The RMU might be particularly suitable for spin and despin operations.

Astronaut EVA

The tug can obviously use astronaut EVA methods to acquire manipulative capability. In this case an air lock would be required. Since the astronaut would probably leave the air lock through the docking port with safety line and life support hoses, it can be argued that tug should be equipped with manipulator arms that would be used to capture the work object and lock on prior to EVA operations.

The astronaut could rig tethering struts or cables so as to free the arms. One arm with a basket could be used as a cherry picker to position the astronaut on the work.

This configuration, with two general purpose arms, would be similar to the mini-tug concept. The relatively simple two-arm configuration does not meet teleoperator capabilities except in conjunction with EVA.

Since astronaut EVA is presumed to be available, the impact on tug would be the air lock, two general purpose arms, and the required illumination and TV systems.

Tug Manipulator Configuration

As a manned teleoperator, the tug would require the same arms, tools, lights, cameras, and controls as the RMU except for the communication link.

The tug could carry arms and grapples with greater reach and force capability. On the other hand, the bulk and mass of tug compared to RMU would bar its access to certain jobs where smaller and more agile RMU/EVA options would be able to work.

Table D-4 summarizes the manipulator mode options.

Table D-4. Tug Manipulative Options

- Tug Manipulator

  (Tug with arms, grapples, lights, and TV)

  Grapples to work object and performs tasks with arms and tools

- Tug EVA

  (Tug with arms, air lock, lights, and TV)

  Grapples work object during EVA.

- Tug RMU

  (Tug with command controls, lights, and TV)

  Transports RMU to job and controls

  RMU docking and manipulation—monitors

  RMU TV and sensors.

SUMMARY AND CONCLUSIONS

Present tug system concepts have concentrated on a system of docking-port devices that enable the tug to couple to various compatible packages and objects to maneuver them, assemble them to other structures and packages, and transfer personnel and cargo from one unit to another.

More dexterous manipulative capability than that is required to work on noncompatible objects. Teleoperator studies and space maintenance studies (by NR and others) have shown the requirement to perform such dexterous tasks.

Dexterous manipulation capability is also required for back-up and assistance to special purpose materials handling and EVA methods.

Dexterous manipulative capability can be provided to tug operations by three methods:

1. Tug-controlled-and-based RMU (teleoperator)

2. Tug-based EVA with air lock and two manipulator systems

3. Tug-mounted-and-controlled manipulator - tool and grapple systems (with teleoperator capability)

The tug-EVA dual arm appears to merit detailed consideration. The air lock might be mounted in a separate neuter docking cone that also carries the dual manipulator arms.

As one of a few manned spacecraft on the spot, tug should be capable of coping with targets of opportunity.