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Comparative Study of Heat Rejection Systems for Portable Life Support Equipment

FINAL REPORT

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COMPARATIVE STUDY OF HEAT REJECTION SYSTEMS FOR PORTABLE LIFE SUPPORT EQUIPMENT

FINAL REPORT

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FOREWORD

This document is submitted in accordance with the requirements of National Aeronautics and Space Administration Contract NAS9-8184. Technical assistance by members of the NASA Manned Spacecraft Center, Houston, Texas throughout the program and during preparation of this report is gratefully acknowledged.

ABSTRACT

A comparative study was undertaken to identify promising heat rejection systems which are smaller, require less expendables and/or are more reliable and less sensitive to contamination than currently available hardware. The study began with an investigation of, and data collection for, promising concepts and progressed into definition of desirable and undesirable system or method characteristics. At this point the least desirable concepts were discarded. Next, the sizing and arrangement of components for each potential candidate system was established, and a parametric analysis of each candidate system was performed. From the study, it was determined that currently available heat rejection system hardware with some improvement should be utilized in the next generation of portable life support systems (PLSS) for missions requiring less than forty hours of either earth orbital or lunar surface EVA per man. For missions of greater than forty hours of EVA per man the development of a regenerative heat storage sink is recommended with the option of a hybrid heat storage/heat rejection sink for some applications. For missions requiring greater than fifty hours per man of lunar surface EVA, serious consideration of the employment of a lunar radiator cart is recommended. Successors to the recommended next generation of PLSS heat rejection system were also considered. A potential candidate in this category is the suit integral diffusion-vaporization system for missions requiring less than forty hours of EVA per man.

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I. INTRODUCTION

Better Portable Life Support System (PLSS) heat rejection systems are essential if man's total potential for performing extravehicular tasks both on the lunar surface and in earth orbit is to be realized. Systems developed to date were designed for a relatively short operational life. This is reflected in the design of the heat sinks for both the Gemini Extravehicular Life Support System (ELSS), and Apollo PLSS. Both rely on the consumption of an expendable -- the evaporation of water -- to provide cooling, and both use contamination sensitive capillary flow devices. In addition, the Gemini ELSS could not be recharged in space and recharging of the Apollo PLSS is complicated by the requirement for deaeration of recharge water. This study was undertaken to define mission optimum heat rejection methods for the next generation of portable life support systems for both earth orbital and lunar surface extravehicular activity.

II. APPROACH

The sequence of tasks followed in this study was constructed with the intent of providing a broad review of current technology, identification of new concepts and finally formulation of recommendations for future hardware development. The logic flow of these tasks is shown in Figure 1. The heat loads used in the study are shown in Figure 2.

The study began with an investigation of, and data collection for, promising concepts. During conduct of this task a comprehensive literature search was performed from which concepts were identified and documented. Results of this effort were then reviewed by a Martin Marietta concept team composed of recognized authorities in the fields of heat transfer, zero gravity management of fluids, and life sciences. The action items and recommendations from this review were then acted upon and included into the study. In responding to concept team action items additional experts in the areas of physical metallurgy, space medicine and controls technology were consulted.

Desirable and undesirable characteristics associated with each concept were defined, and a preliminary screening process was initiated to select candidate systems for further consideration. The screening conclusions were subjected to reviews, first internally by the Martin Marietta Technical Approval Group, and subsequently by cognizant NASA personnel.

Next, the concurred upon candidate system approaches were analyzed to determine system requirements, including storage, power, sensors and controls, flow rates and coating insulation requirements. A detailed schematic was then prepared and components were sized and arranged to provide the required PLSS heat rejection requirements. Based on the sizing analyses weight, dimensional and power requirements were established.

A parametric analysis was then performed and the results compiled in graphs and tables. Based on conclusions drawn from the parametric analysis, recommendations were then made concerning improvements to developed heat rejection systems hardware and future new hardware development effort.

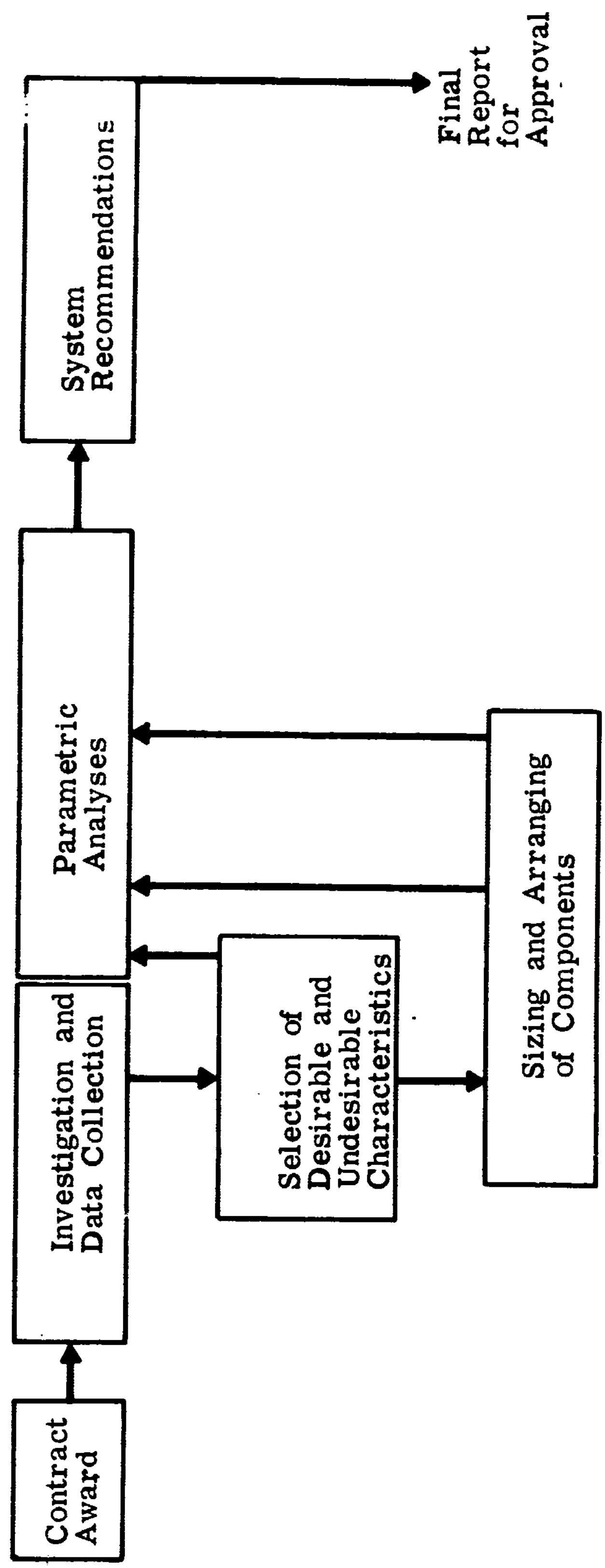


Figure 1 Study Approach Logic

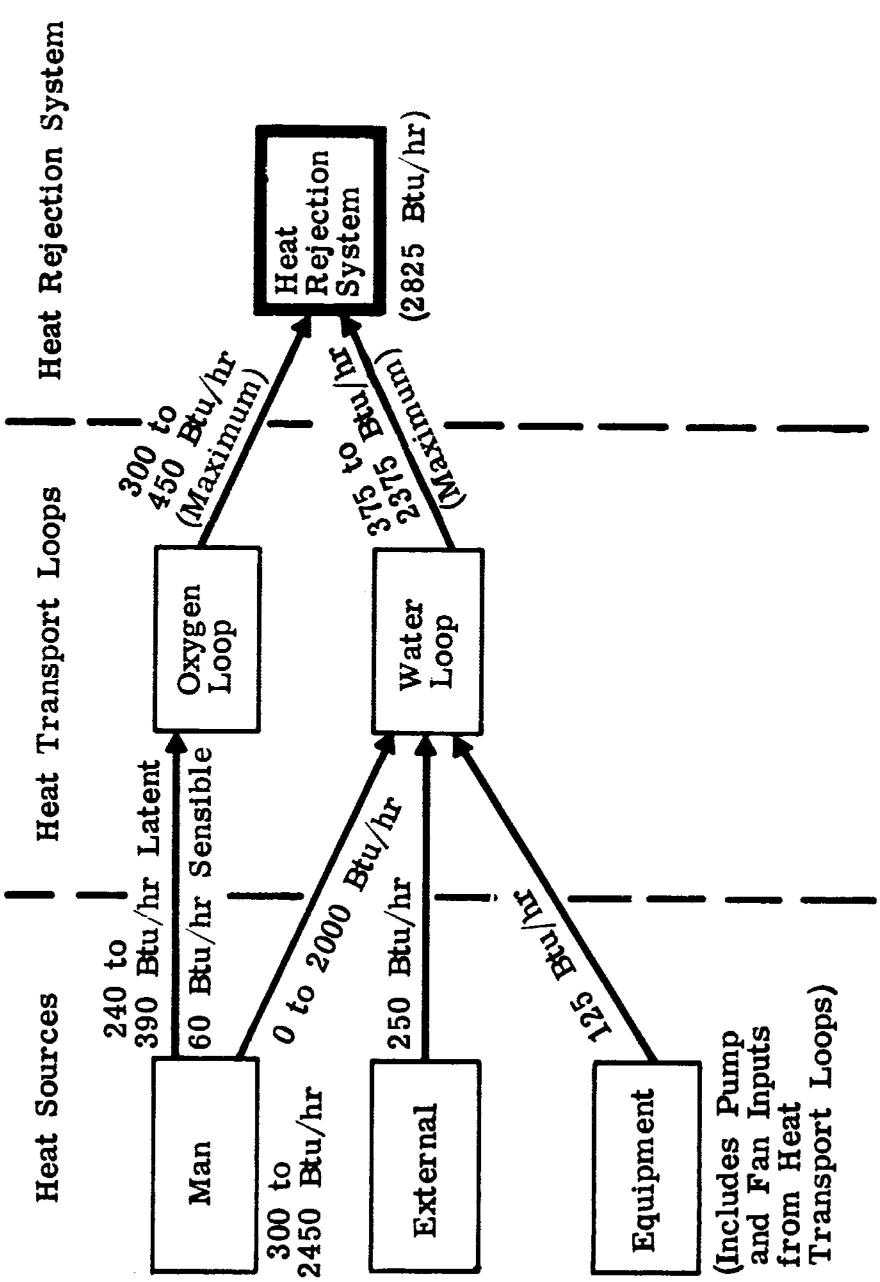


Figure 2 Heat Rejection Requirements fcr Portable Life Support Equipment

III. DISCUSSION

The objective of this program is the identification of promising heat rejection systems which require less volume on the man, a lower total launch weight and/or are more reliable and less sensitive to contamination than currently available hardware. The discussion begins with a description of each concept considered. Next, candidate system selection criteria is summarized and selected systems are listed. The discussion is concluded with a review of the parametric analysis.

In an attempt to perform a comprehensive study, review of heat rejection approaches was not limited to prior aerospace applications. From the review, a large number of concepts evolved. Each concept is summarized in this discussion without regard as to whether it was later selected for a candidate system. The purpose of including each concept is to provide subsequent investigators with maximum visibility concerning the information and considerations which formed the basis of this study. Screening of concepts was based on whether an advantage consistent with the objective of this study was apparent. The concepts were synthesized into systems, and candidate systems were selected from these for further consideration. The parametric analysis was also constructed with the intent of providing maximum visibility; therefore, all assumptions are discussed and all values used in constructing the curves are tabulated.

A. CONCEPT DESCRIPTIONS

The following methods and concepts which are applicable to portable life support systems (PLSS) were considered during this study. Heat sinks are considered under two classifications, 1) heat rejection sinks, and 2) heat storage sinks. The discussion also includes heat transport from the LCG to the heat sinks by liquid loops and gas loops. In addition, heat transfer from the astronaut's skin by heat pipes, helium gas conduction through the suit, capillary-pumped liquid-vapor loop, and several types of refrigeration are considered.

1. Heat Rejection Sinks

a. Wick fed evaporator

A coolant is circulated by a pump through the astronaut's liquid cooled garment (LCG) where the coolant absorbs the astronaut's metabolic heat. The coolant is passed through an external plate and fin evaporator-heat exchanger (E-HX) where this heat is transferred to an evaporant, water. The E-HX is vented to space vacuum through a back-pressure regulating valve which maintains a constant pressure within the E-HX so that the evaporant boils within a narrow temperature band above freezing. The water which is fed into the E-HX travels by capillary action along the wicks and wets the adjacent heat transfer surfaces where the heat from the coolant causes the water to evaporate. The resulting vapor flows into a header and through the back-pressure regulation valve through which it escapes to space. The water which is converted into vapor is replaced by feeding additional water from a pressurized reservoir into the wicks of the E-HX at the rate required to keep the wicks wetted. A schematic diagram of a wick fed evaporator heat sink incorporated into a heat rejection system coolant loop is shown in Figure 3. Water is selected as the evaporant because of its high heat of vaporization, 67,000 Btu/cu ft, 1073 Btu/lb. Water has the additional obvious advantages of being nontoxic, nonirritant and noncombustible. It is also inexpensive and readily available. In addition, the boiling temperature of the water in the evaporator can be easily controlled by the pressure regulating valve which vents the expended water vapor to space. Space qualified hardware has been developed and used in space. Disadvantages of the wick fed evaporator include the expenditure of twelve pounds of water for each four-hour extra vehicular activity (EVA) mission. The wicks are sensitive to contamination and tend to degrade in performance with time. Improvement of wick performance appears feasible through development of a monomolecular surface treatment for the wicks and/or control of water pH and quality with additives. (See Recommendation No. 1, page V-1 of this report.)

Another disadvantage which may not be readily apparent is aeration of the evaporant supply caused by gas permeation of the bladder in the storage tank. Often, the evaporant supply is tied into the circulating H_2O loop for filling, accumulator effect and possibly other considerations. The result is deaeration at the pump and pumping rate decrease resulting from pump cavitation.

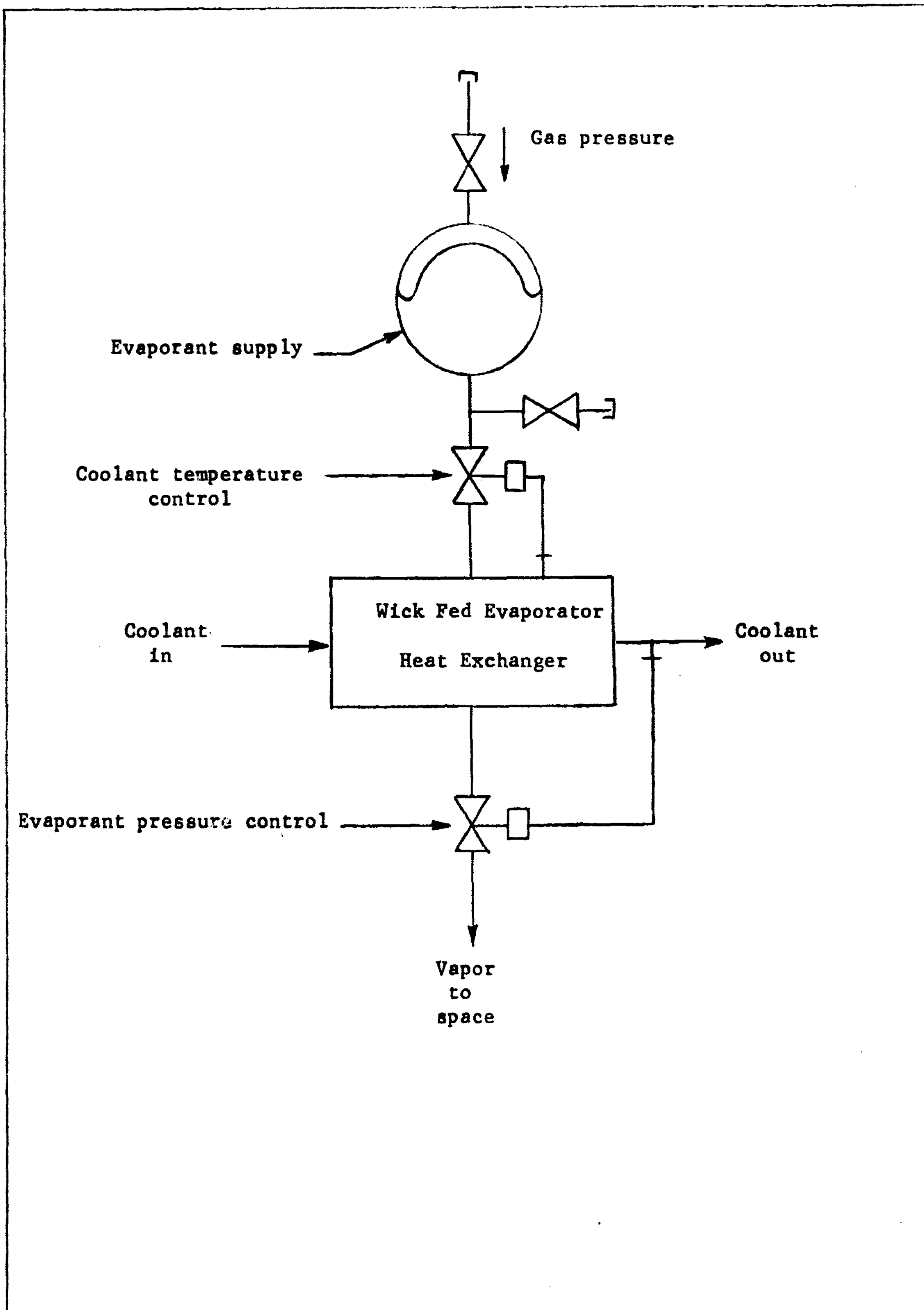


Figure 3 Wick Fed Evaporator

b. Forced vortex boiling evaporator

The coolant which is pump-circulated through the astronaut's LCG is passed through a shell and tube evaporator heat exchanger (E-HX) where the LCG heat is given up to the evaporant, water. The uniform evaporant wetting of the internal E-HX surfaces is accomplished by the insertion of a spirally-twisted metal ribbon in each tube. The ribbon is tightly fitted to the tube wall for good thermal conductivity. The evaporant is introduced into the inlet end of each tube and the spiral flow path which must be followed causes the liquid to be thrown against the inside walls of the tube by centrifugal force. The heat from the coolant provides the energy for vaporization of the evaporant. The vapor produced flows from the tubes into a header and from there through a back-pressure regulation valve to space. The valve is adjusted to control evaporant boiling within a narrow temperature band above freezing. Evaporant water is fed from a pressurized reservoir into the E-HX as required. A schematic diagram of a forced vortex boiling evaporator is shown in Figure 4 and a typical core design is shown in Figure 5.

The major advantage of the forced vortex evaporator is that centrifugal force slings the evaporant against the evaporator tube walls thereby eliminating capillary devices as the means of wetting the heat transfer surfaces. The advantages listed under wick fed evaporator for the use of water as the evaporant also apply to this evaporator.

The disadvantages are that an expenditure of approximately twelve pounds of water are required for a four-hour EVA mission, a distributor is required to equalize the flow of water between the evaporator tubes, and flight qualified hardware has not been developed. However, a conceptual prototype has been built and performance has been demonstrated. Aeration of the evaporant supply is also a disadvantage applicable to this approach (see discussion under "Wick Fed Evaporator", page III-2).

c. Porous plate sublimator

The coolant which is pump-circulated through the astronaut's LCG is passed through a porous plate sublimator heat exchanger (S-HE) where the LCG heat is given up to

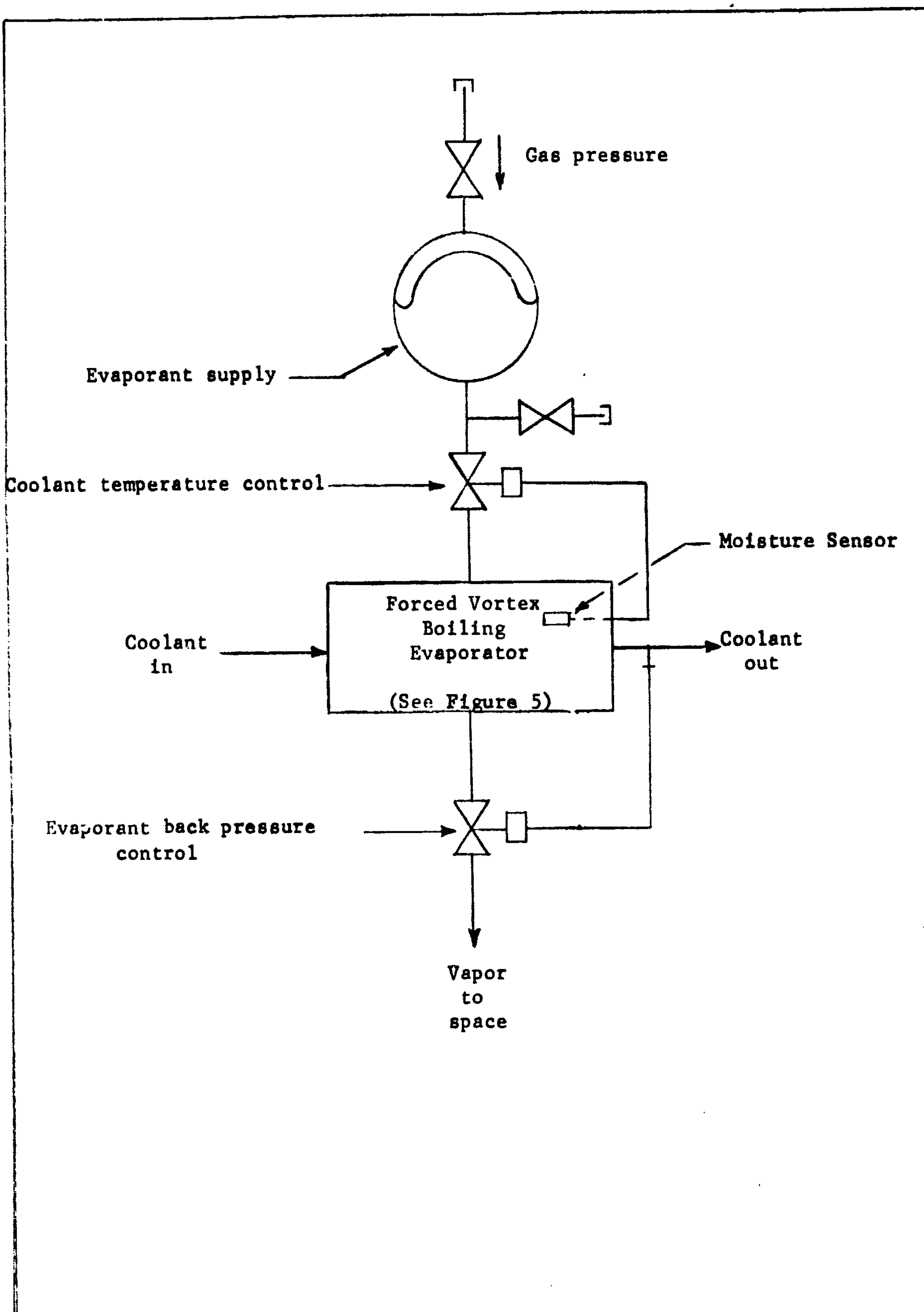


Figure 4 Forced Vortex Boiling Evaporator

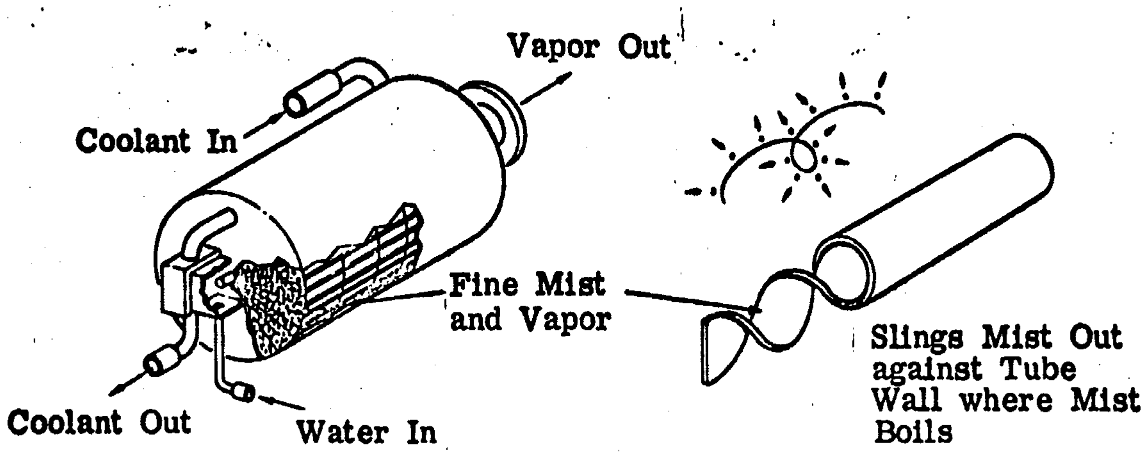


Figure 5 Detail of Forced Vortex Boiling Ev.porator

the sublimant, ice. The ice is formed on the external side of the porous plate by the water vapor escaping through the porous media under low load conditions. After the ice is frozen on the outer side of the plate, the system can be operated at rated heat flows. The sublimator is fed with water from a pressurized reservoir to replace the water that is sublimed to space. The coolant passes through passages within the sublimator and gives up heat to the feedwater within the sublimator. This heat is transferred by conduction into the ice where it elevates the temperature of the ice and thereby increases the rate of sublimation until equilibrium occurs. A schematic diagram of a sublimator heat sink is shown in Figure 6.

The advantages of the sublimation method include: a) a high heat absorption rate (the net change of the feed water is from liquid to steam with the absorption of 1073 Btu/lb); b) water and ice are nontoxic nonirritant, noncombustible, available and inexpensive; c) once in operation, the sublimator tends to be self regulating; and d) hardware employing this method has been space qualified.

The disadvantages of the sublimation method are: a) the sublimant, water vapor, is expended to space (the quantity required is 333 cu in, 12 lbs, for a four-hour mission rejecting 2825 Btu/hr); b) ice build-up for restart must be done under low load; c) porous plates are capillary devices which are sensitive to contamination and usually suffer some degree of degradation in performance with time; and d) any stoppage of coolant flow may result in sublimator freezeup.

Another less apparent disadvantage is aeration of the evaporant supply (see discussion under "Wick Fed Evaporator", page III-2).

Improvements in porous plate performance appear feasible through development of a monomolecular surface treatment for the porous plate and/or control of water pH and quality with additives (see Recommendation No. 1, page V-1 of this report). Also, an effort is now underway to arrive at a sublimator without microporous media. Development testing has been encouraging to date.

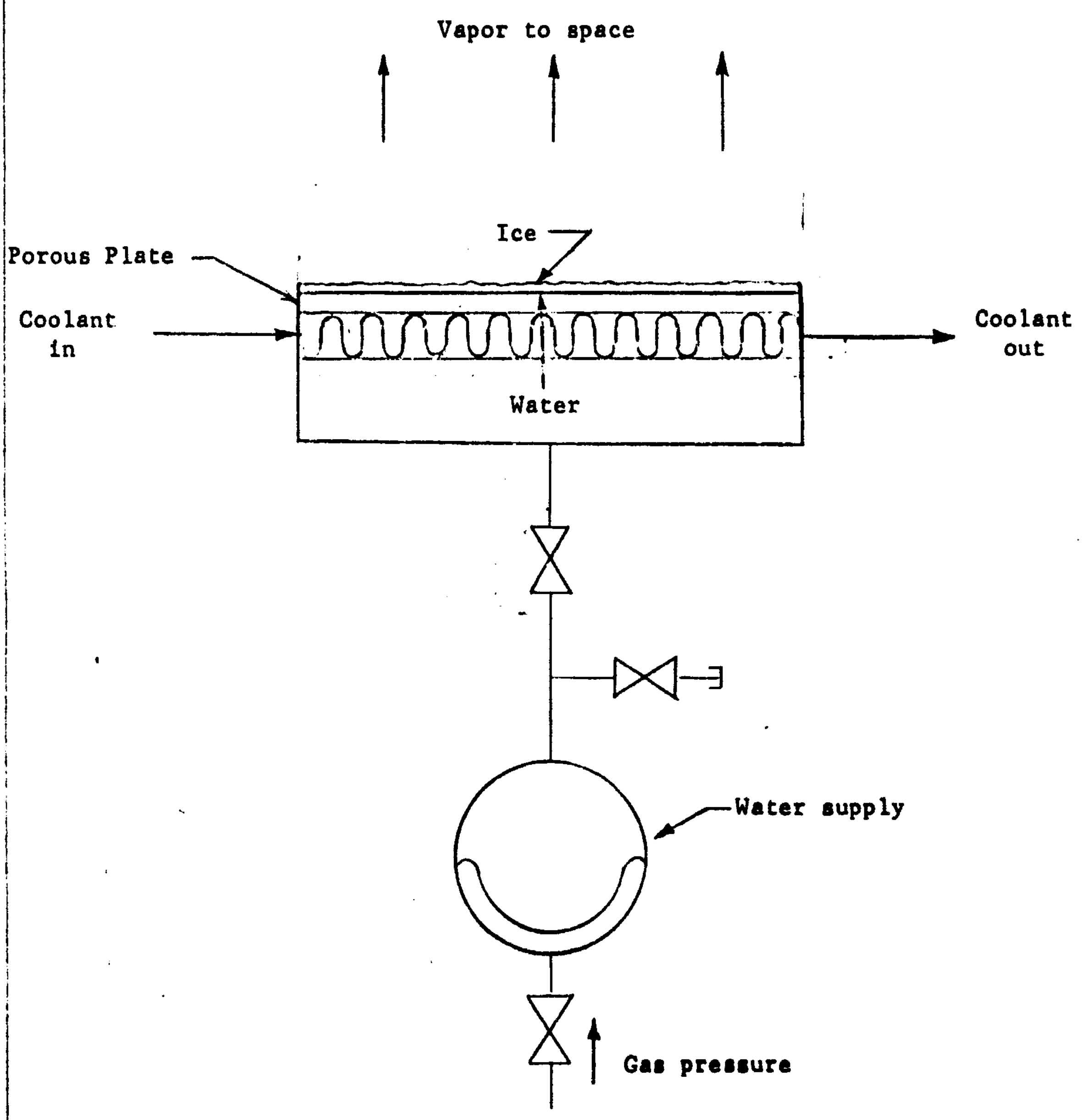


Figure 6 Porous Plate Sublimator

d. Gelled-Water Fed Sublimator

The degradation of porous plates with time in the aforementioned sublimator may be a problem when many usages for a single mission are required. The porous plate and water can be eliminated and replaced by employing a fine-mesh screen and gelled water. The gelled water can be produced by adding about one-half percent by weight of a gelling agent, and the gelled water is contained in a pressurized reservoir and fed directly into the heat exchanger core. The gel fills the core and is retained at the surface exposed to the vacuum by the surface tension of the gel interacting with the screen. The mesh of the screen and the yield stress of the gel must be matched to meet all mission-use conditions. The vacuum on the exposed face of the gel retaining screen causes a layer of the gelled water to freeze but the gel within the core will not freeze because of the heat being delivered to the core by the circulating coolant. The frozen gel-ice layer will vary in thickness with variations in heat load and the subliming gel-ice is continuously replaced by the gelled water which maintains a pressurized interface. The gelling agent will float off the subliming surface as a fine powder with the escaping water vapor. The possible chemical breakdown of the gelling agent as the result of ultraviolet radiation could result in the reduction or reducing of viscosity of the gel. This can be overcome by pH control, use of additives or a solar shield. The heat absorbing capacity of the gelled ice is near that of pure ice with the difference being the reduction in capacity resulting from the one-half percent gelling agent. A schematic diagram of a gelled ice sublimator is shown in Figure 7. The gelled ice sublimation heat sink eliminates capillary devices with their sensitivity to contamination.

The disadvantages of gelled water in a sublimator are: a) the water vapor formed is expelled to space, therefore approximately 333 cu in, 12 lb is required for a four-hour mission having an 2825 Btu/hr heat rejection rate; and b) concept verification and hardware development is required for this heat rejection method. Aeration of the evaporant supply is also a disadvantage of this concept (see discussion under "Wick Fed Evaporator", page III-2).

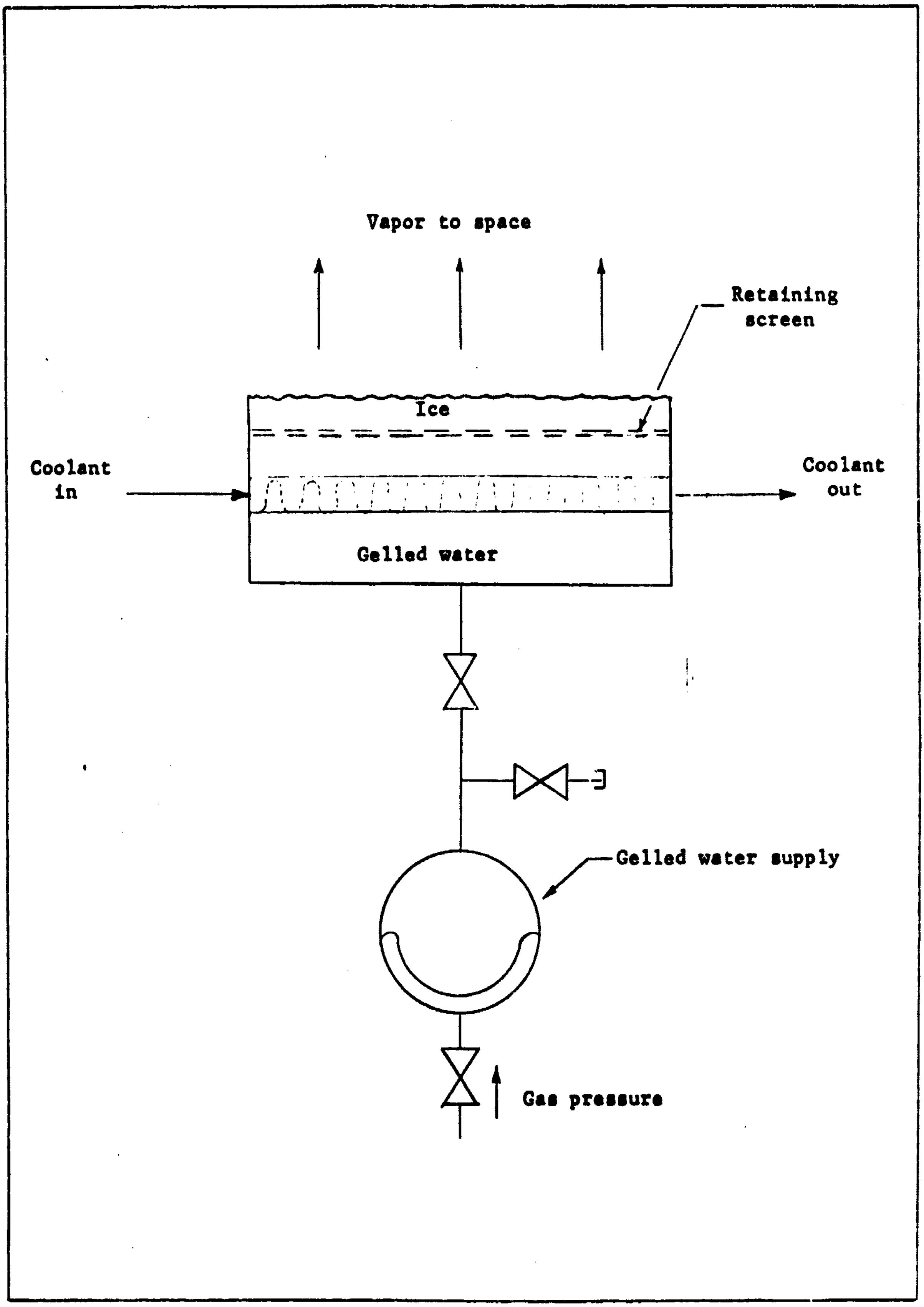


Figure 7 Gelled Water Fed Sublimator

e. Integral diffusion-vaporization

The integral diffusion-vaporization system consists of several cooling patches which cover non-articulated portions of the body, associated plumbing and controls, and a supporting garment. Next to the skin is a layer of absorbent mesh. The absorbent mesh provides a comfortable substrate for the cooling patches and also provides a wick to absorb moisture from the skin. This moisture is wicked to areas between the cooling patches where it is evaporated.

The cooling patch which acts as a low-pressure boiler is made up of five basic layers: an outer and inner impermeable membrane, a wicking layer, a boiler void area, and reflective foil layers. The wicking layer is a reservoir which maintains "wet" contact with the membrane on the skin side of the patch. This wicking layer is attached to the boiler void layer, which provides a path for the vaporized water to follow in leaving the patch. The foil layer acts as a radiation shield protecting the cooling patch from external radiant heat.

The absorbent mesh and cooling patches are held in place by a nonstretchable fabric cover. The patches are connected to form a complete garment by a stretch type material at the articulated joint areas. Water supply lines are routed inside of the vacuum lines to each cooling patch. A water reservoir in the form of a belt is attached to the main torso cooling patch. The reservoir supplies water to each cooling patch through water supply lines and each patch boiler is vented through the vacuum line to an orifice-limiting control valve. Selective pressure control and temperature control can be obtained by isolating sections of the suit into separate compartments, each with its own regulating valve. The water evaporated from the wetted wick is replaced from a pressurized reservoir. A cross-section of an integral diffusion-vaporization heat sink cooling method integral with a space suit is shown in Figure 8.

Major advantages are that the heat sink can be integrated into the space suit; and water, with its high heat of evaporation, its non-toxic, nonirritant, noncombustible characteristics, its ready availability, and low cost can be utilized as the evaporant. Good heat rejection efficiency has been demonstrated by an experimental test section

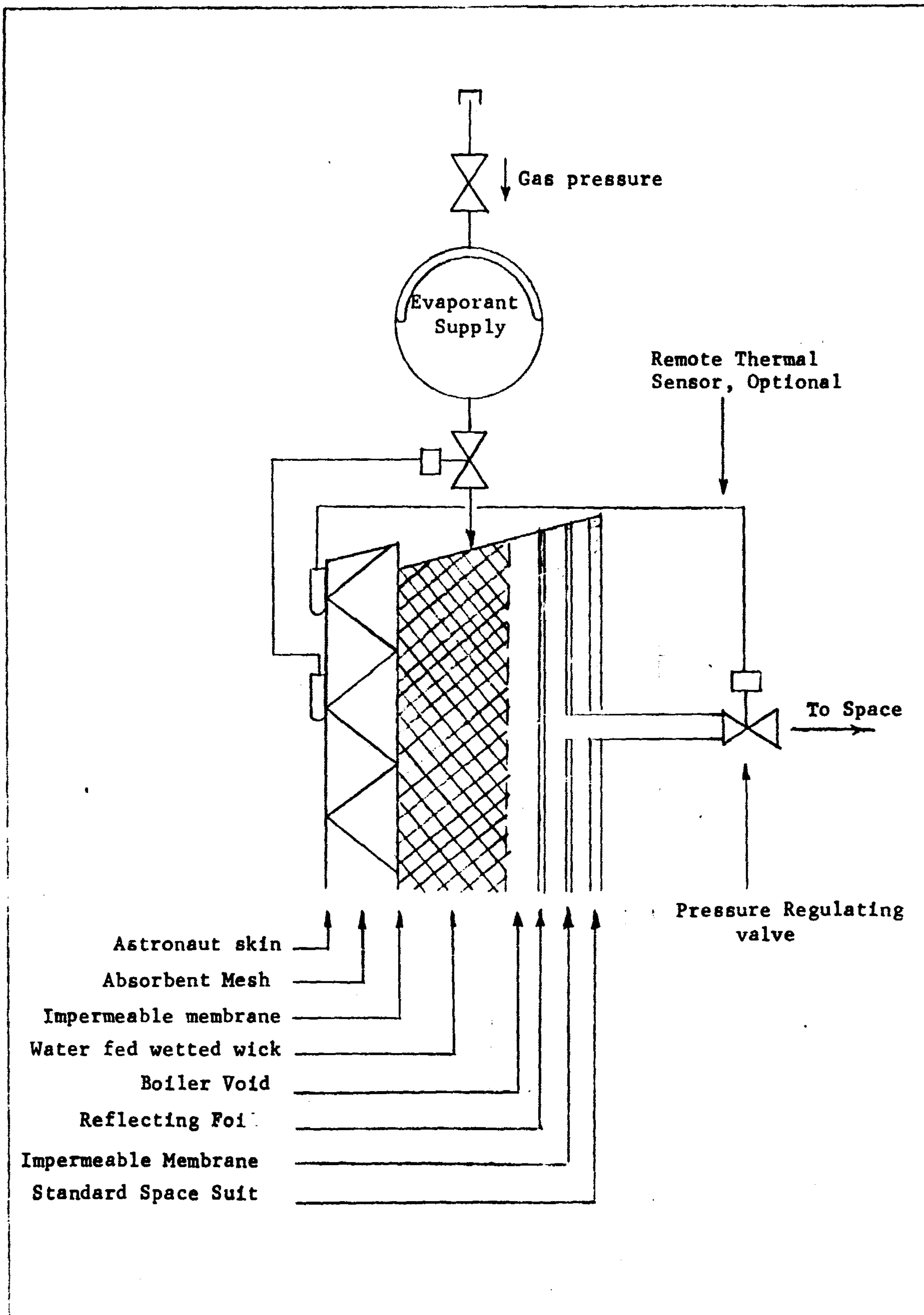


Figure 8 Cross-Section of Diffusion-Vaporization Heat Sink Integral with Space Suit

which rejected 182 Btu/hr, sq. ft. (in one-g) or the equivalent of 2910 Btu/hr for 16 sq. ft. surface (assumes coverage of approximately 80% of the body with cooling patches).

The disadvantages of the diffusion-evaporation heat sink are: a) the evaporated water vapor is expended to space (333 cu. in., 12 lbs.) is required for a four-hour mission with a heat rejection rate of 2825 Btu/hr; b) the suit-integrated wick, vapor passages and water distribution system add to the suit thickness and may constrain astronaut mobility; c) the heat rejection rate of the suit and uniform heat rejection throughout the suit can degrade with time as a result of contamination of the suit wicking; and d) under low load the astronaut's skin temperature may be reduced to an uncomfortable level.

This approach is being used in the Evaporative Cooled Garment System (ECGS) which is currently being developed by McDonnell-Douglas for the NASA-Manned Spacecraft Center.

f. Integral transpiration-vaporization

This system requires a membrane that is highly permeable to water vapor but impermeable to atmospheric gases. The water vapor that is produced in the process of cooling the astronaut passes through the permeable membrane and escapes to space vacuum.

Oxygen gas or an oxygen-nitrogen gas mixture is circulated through the thermally conductive spacer which contacts the skin. An impermeable membrane suit encloses the spacer and retains the oxygen or oxygen-nitrogen gas atmosphere. A water-wetted wick suit surrounds the outside of the impermeable membrane and the heat conducted from the skin evaporates the water. The water vapor produced flows outward through porous insulation and escapes to the vacuum of space by transpiring through microscopic openings in the outer membrane enclosing the suit. The outer membrane area must be selected to transpire water vapor at the rate of three pounds per hour. Since this is a constant rate, heat rejection control must be obtained by controlling the rate at which water is fed into the wick suit. The rate of heat rejection varies over the body, therefore, for temperature control and body comfort, the rate of water feed must be varied on a selective basis by body areas. Some of the water vapor produced in high heat transfer areas must diffuse laterally through the porous insulation to lower transfer areas, because the vapor transpires uniformly through the vapor-permeable outer membrane. An alternate approach is to sectionalize the evaporating-transpiring areas and select vapor-permeable membranes with transpiration rates proportional to the vapor heat transfer and vapor generation rates. A cross-section of a transpiration space suit for astronaut heat rejection is shown in Figure 9.

The transpiration space suit utilizes water with its broad compatibility and its high heat of evaporation. Water vapor permeable membranes have been fabricated and are undergoing performance tests to verify their application for space suits.

Disadvantages of transpiration space suits include the requirement for an expendable water supply, 333 cu. in., 12 lbs., for a four-hour mission with an average 2825 Btu/hr heat rejection rate, performance degradation with time due to physical changes in the water-vapor-permeable membrane, clogging of microscopic pores in the membrane occurring as the result of the suit outer surface making contact with contaminated surfaces or of poor feed water quality, variation in cooling rate as a function of capillary pumping or feed water flow rate, and low load conditions causing ice formation within the suit with possible retardation of astronaut mobility.

The principal difference between vapor-diffusion and transpiration is that vapor-diffusion evaporation is actively controlled by the pressure control valve whereas transpiration relies on diffusion directly to the outer surface of the suit where evaporation or sublimation occurs in response to the internal heat load.

g. Integrated magnetohydrodynamic transpiration

This system utilizes a magnetic field to improve evaporation and transpiration of the evaporant. An electric potential is established between the evaporating area and the transpiring area, which in turn produces a magnetic field between these two areas. The magnetic-electric movement of a gas, like water vapor, because of its polar molecular form, tends to improve both the evaporation heat flux and the transpiration of the water vapor. This principle has been investigated as a means of controlling excessive aerodynamic heating at supersonic velocities

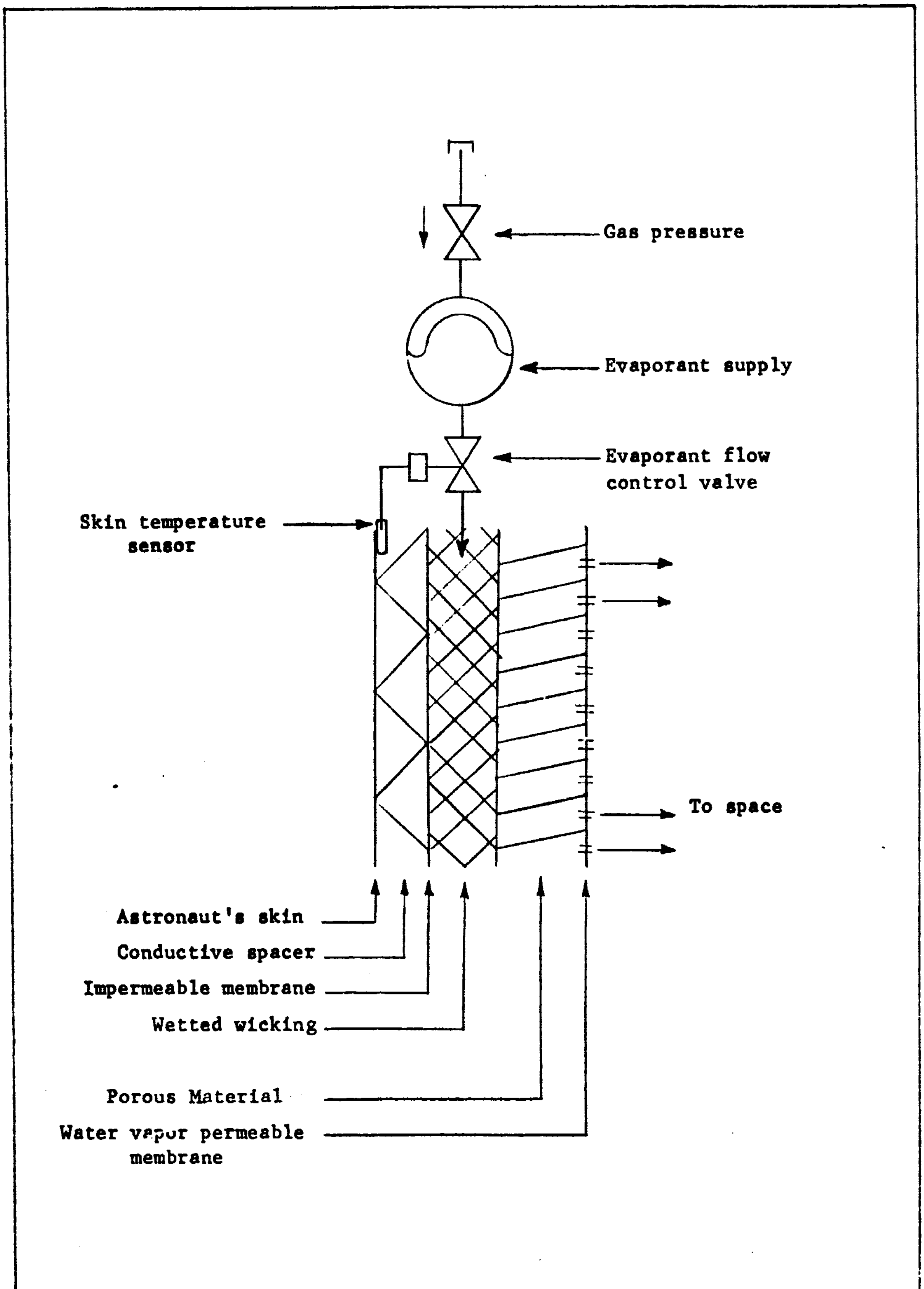


Figure 9 Cross-section of Integral Transpiration-Vaporization Heat Rejection Sink

for missile and rocket nosecones and there are indications that surface cooling by evaporation and transpiration may be improved at the temperatures and pressures occurring in heat rejection systems for portable life support equipment. A cross-section of a space suit with the magnetohydrodynamic transpiration concept applied is shown in Figure 10.

Advantages of this application to space suit heat rejection are the potential increase in heat flux between the warm surface and the evaporant under high heat flow rates, the greater ease in transpiring water vapor through the water-vapor-permeable, non-liquid permeable membrane, and the use of water as the evaporant with its broad compatibility and its high heat of evaporation.

Disadvantages of this application to space suits are that the evaporant is expended to space, electrical apparatus is required to produce the magnetic field, the evaporating surface and the transpiring surface must both be electrically conductive, additives may be required in the evaporant to increase its conductivity, and performance may degrade with time due to contamination of the evaporating surface and clogging of the pores in the transpiring surface. Precautions would be necessary to prevent the electromagnetic field from interferring with spacecraft systems and scientific experiments.

While this application of magnetohydrodynamics may be feasible for space suit heat rejection, it would appear to be more practical for larger heat rejection requirements such as space power plants.

h. Cryogenic oxygen or hydrogen evaporation

A primary coolant such as water is pumped through the LCG to absorb the astronaut load. The heat absorbed by the primary coolant is transferred to a secondary coolant in a liquid heat exchanger. The secondary coolant is pumped through a heat exchanger which transfers the astronaut heat to a cryogenic holding tank containing liquid hydrogen or liquid oxygen. The heat exchanger is designed to be low-temperature-limiting to prevent freezing of the coolant under all operating conditions. The cryogenic tank is designed to distribute the incoming LCG heat uniformly throughout the tank and to separate the boiled-off gas from the liquid so that only gas will flow from the tank. Exiting cryogenic fluid is routed

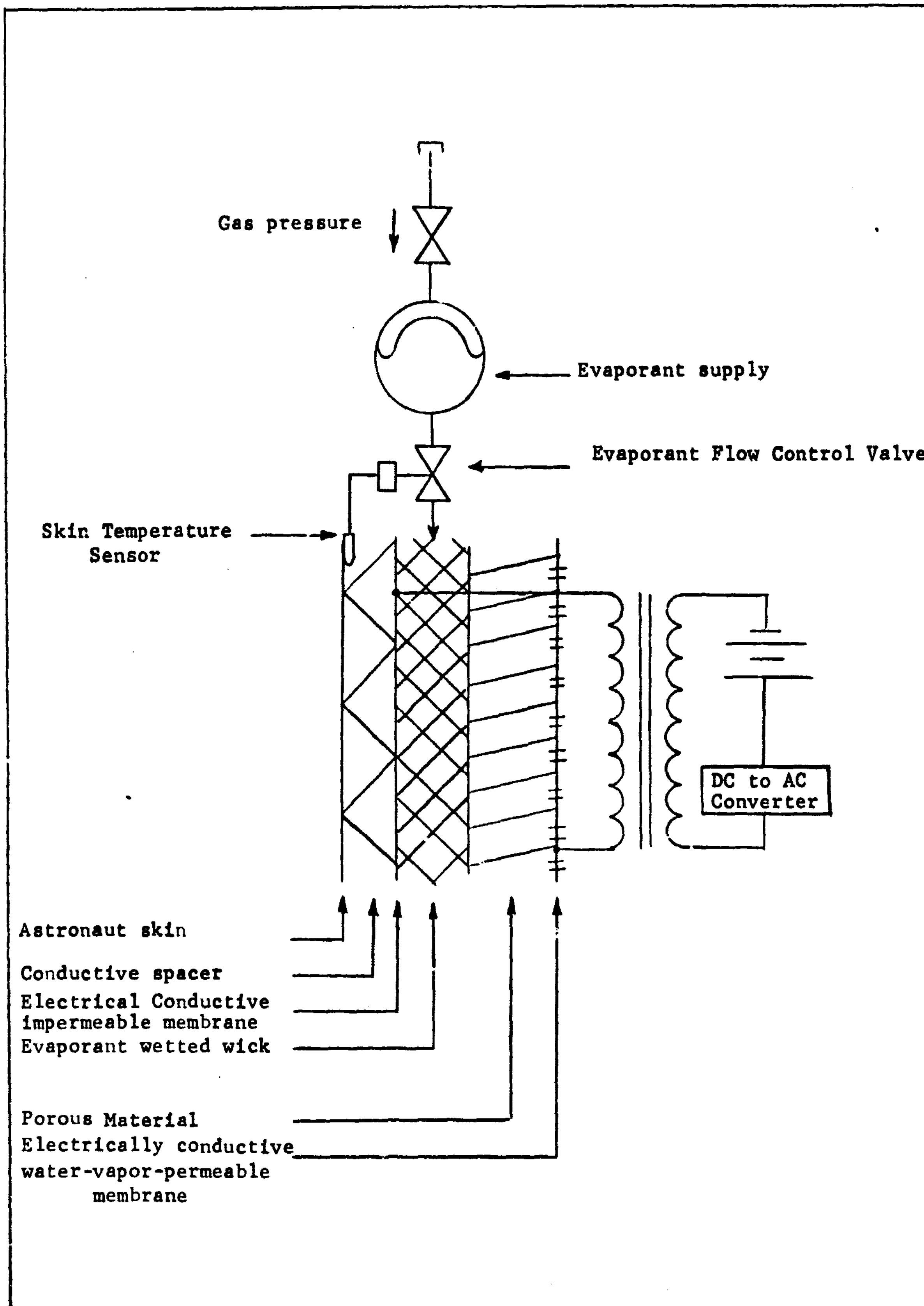


Figure 10 Cross Section of Integral Magnetohydrodynamic Heat Sink Transpiration

through an external gas-liquid heat exchanger where it is vented as a superheated gas. The heat of vaporization of cryogenic oxygen is 12,200 Btu/cu ft, 171 Btu/lb. The heat of vaporization of cryogenic hydrogen is 8380 Btu/cu ft, 194 Btu/lb. A schematic diagram of a cryogenic evaporative heat sink is shown in Figure 11.

The advantages of a cryogenic oxygen evaporative heat sink are that the expended gas is nontoxic and could possibly be recovered and utilized for life support. The advantage of cryogenic hydrogen is that it is nontoxic.

The disadvantages of the cryogenic oxygen heat sink include: a) Liquid oxygen has a low volumetric heat absorbing capacity (0.93 cu ft, 66 lbs, are required for a four-hour EVA); b) Oxygen gas is expended to space; c) Oxygen-clean system and handling equipment is required; d) Low liquid oxygen boiling temperature requires a secondary coolant loop and a suitable secondary coolant; e) There is cryogenic boil-off loss during storage; f) An additional heat exchanger is required to utilize the sensible heat absorbing capacity of the vented oxygen gas; g) Many materials are flammable or explosive in contact with oxygen; and h) Special safety procedures must be followed by handling personnel.

The disadvantages of cryogenic hvdrogen heat sink are similar except that: the low volumetric heat absorbing capacity of hydrogen; means that 13.1 cu ft, 58 lbs, are required for a four-hour EVA; and hydrogen is flammable or explosive when mixed with oxygen.

i. Irreversible endothermic chemical reactions

Ths decomposition of ammonium bicarbonate to form ammonia, water and carbon-dioxide takes place in stepwise fashion over a range of 95 to 140°F at one atmosphere and with a predicted range of 85 to 130°F if the reaction is vented to a vacuum. Even though the potential quantity of heat which can be absorbed by this reaction is 90,000 Btu per cu ft compared to 67,000 Btu per cu ft with water, the heat sink temperature level is too high to absorb the available heat unless a heat pump is employed. Consequently this system appears unattractive for the required application.

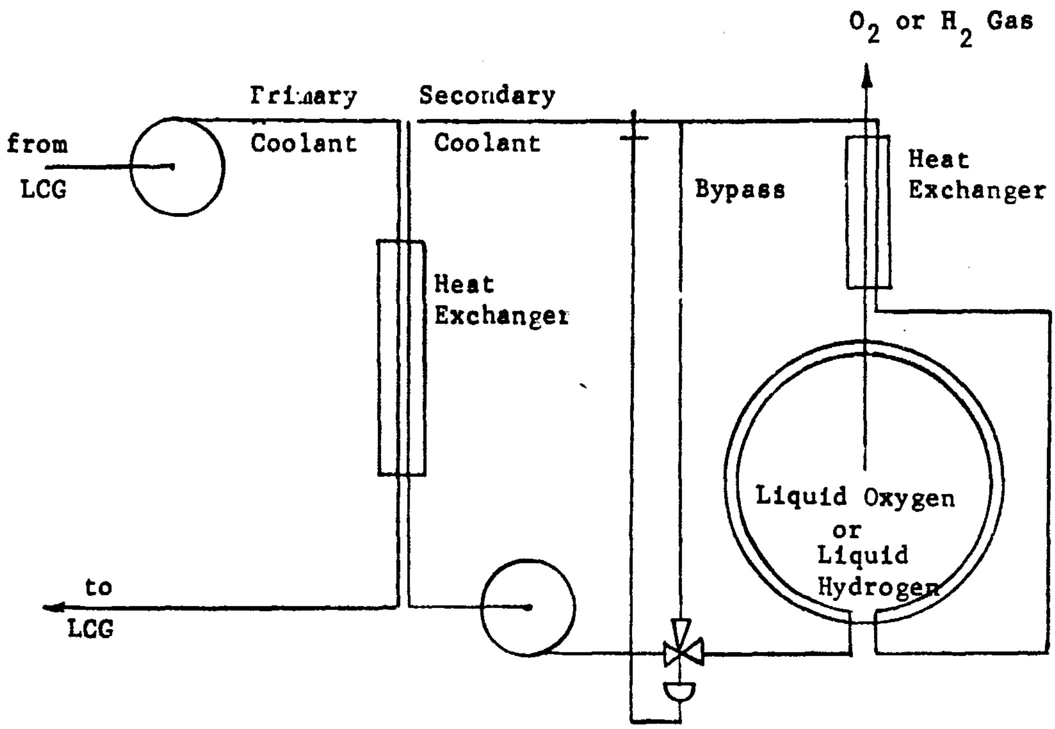


Figure 11 Cryogenic Oxygen or Hydrogen Evaporative Heat Sink

Other disadvantages in addition to the temperature level at which the heat can be absorbed are: a) a by-product of the chemical reaction, ammonia, is toxic; b) all by-products of the chemical reaction are expendable and must be disposed of; and c) the volume, mass, power requirements, and controls for a heat pump add to the disadvantages of this method of heat rejection.

j. Radiators

This heat rejection method is effective for a radiating surface that faces deep space. A radiator sized for the design of a heat rejection load of 2825 Btu per hour is too large to mount on a man. An alternate method of providing a remote radiator which is attached and provides cooling to the astronaut by means of an umbilical was considered. The difficulty of maneuvering a large radiator and keeping it pointed to deep space makes this approach unattractive for earth orbital EVA operations. However, the possibility of mounting such a radiator on wheels for lunar operations appears to be feasible. The radiator system would be connected to the astronaut by means of an umbilical. The resulting cart could then be pulled by the astronaut while changing work stations; however, he would be fairly free to work at one location within the limiting radius of the umbilical. While this approach violates the study design constraint that the heat rejection system be mounted on the man, it is felt that the prospect of a system requiring no expendables makes such an approach worthy of further consideration. As a result a study of various radiator configurations was conducted and is summarized in the following paragraphs. The configurations considered are:

1. Fins
2. Hemisphere
3. Conical spires protruding from a hemisphere
4. Rotating belt or cylinder
5. Rotating disk
6. Flat sheets or plates

The evaluation process consisted of two steps: 1) a qualitative analysis to inspect the potential efficiency of the configuration, and 2) a quantitative analysis which yielded actual sizes and configurational details necessary for the transfer of 2825 Btu/hr.

The basic heat transfer relationship used was:

$$\dot{Q} = \epsilon \sigma A F (T_r^4 - T_s^4)$$

where: $\dot{Q}$ = heat transfer rate, Btu/hr

ϵ = emissivity, assumed .95

σ = Stefan-Boltzmann constant, $.1714 \times 10^{-8}$ Btu/hrR^4 ft^2

A = Area of radiating surface, sq ft

F = View factor

T_r = Temperature of radiating surface, 515°R assumed

T_s = Temperature of heat sink, 0°R assumed for space

In the analysis of lunar radiators the radiant heat input from the sun, 440 Btu/hr sq ft, and the radiation from the sunlit lunar surface, which may reach temperatures above 220°F, must be considered. Shading from these two thermal inputs must be provided if the radiator operates during the lunar day. Surfaces used for shades must have high reflectivity values whereas surfaces used as radiators must have high emissivities. Surface coatings which effectively combine these qualities are available.

The results of this investigation are summarized below:

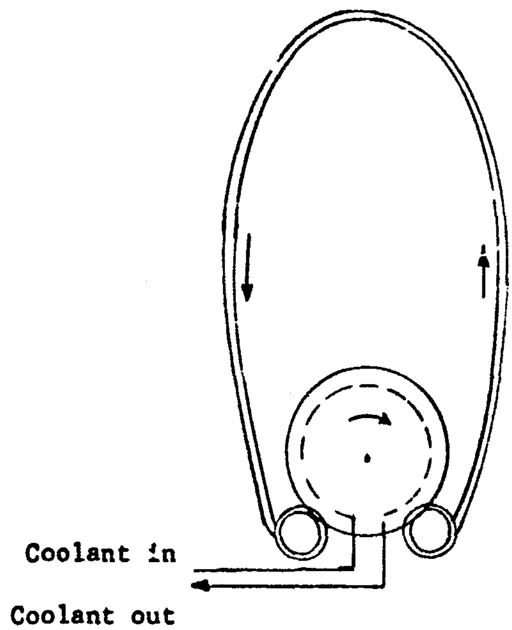
1. Fins - For the pure radiation heat transfer under consideration there is an extreme amount of visual interaction between the fins which leads to a low view factor and poor performance. Fin performance varied with configuration but no configuration had sufficient heat rejection characteristics.

2. Hemisphere - The hemispherical radiating surface has a view factor of 1.0 for all practical purposes. A hemisphere with a diameter of 3.8 feet is needed to radiate 2825 Btu/hr. This configuration needs shielding from direct solar radiation in order to be effective, however, the shielding will reduce the view factor and further increase the large diameter of the hemisphere.

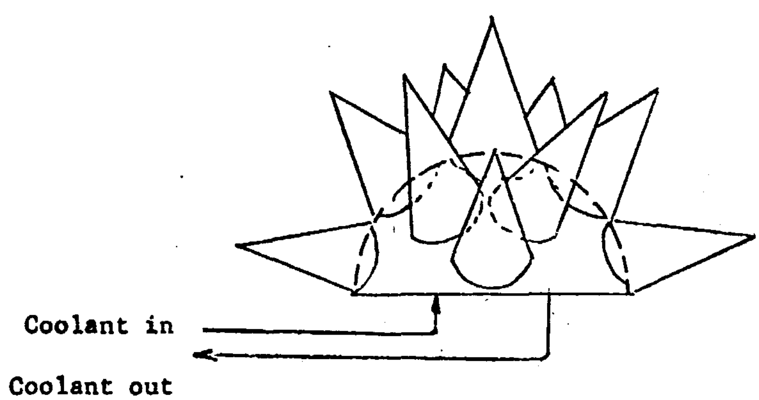
3. Conical Protrusions (see Figure 12) - Several configurations with respect to cone length, taper and number were analyzed when protruding radially from a 2 ft. diameter hemisphere. It was found that the cones had to be quite large to provide the necessary surface area. The best configuration analyzed was one with 10 cones each 3 ft. long with 15 degree apex. Considering interaction with the lunar surface and shading from sunlight, the heat transfer rate was only 2000 Btu/hr. This configuration was abandoned because of its large size and inadequate heat transfer rate.

4. Rotating Belt or Cylinder (see Figure 12) - This device involved heat transfer to a thin metallic belt by conduction, convection or radiation and then radiation to space when the belt has moved beyond the initial heat transfer zone. Evaluating the three initial modes; conduction between two solid bodies in space is inefficient because of the lack of any gas in the pores of the two surfaces, convection would require an elaborate sealing system to prevent leakage of the fluid, and the initial radiation process would be too inefficient. Because of inherent disadvantages, this method was not considered further.

5. Rotating Disk (see Figure 13) - The same comments concerning the moving belt radiator apply here except that the heat exchanger compactness of the rotating disk design permits the initial radiation process to be more efficient. For this reason the system was given more analysis in hopes that it would be compact enough to justify usage of the motor needed to ratate the disks. In order to get adequate heat transfer, four disks, each 3 feet in diameter and .030 inch thick and rotating at a speed of 15 revolutions per hour, were needed to radiate a maximum heat load of 2825 Btu/hr. This configuration has overall

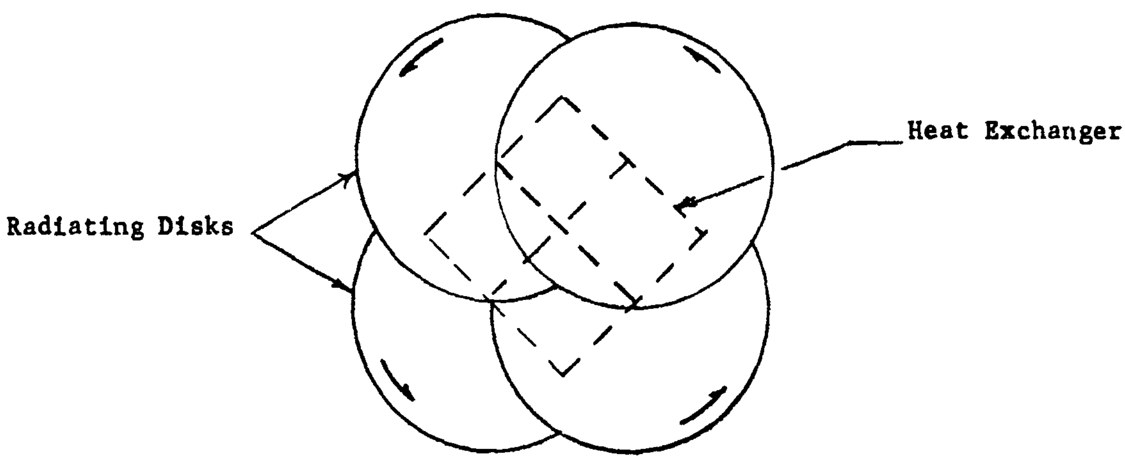


Traveling Belt Radiator

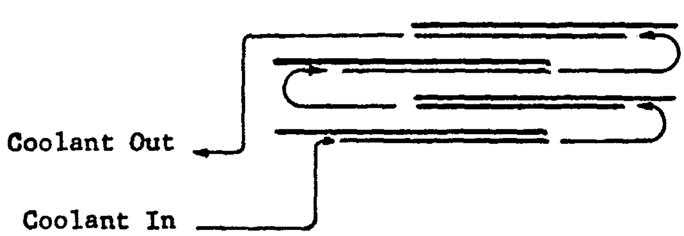


Conical Protrusion Radiator

Figure 12 Types of Space Radiators



Plan View



Elevation View

Figure 13 Rotating Disk Radiator

dimensions of 4.5' x 4.5' x .5' which was considered too large even before the motor and its power supply were added.

6. Sheets or Plates - The area necessary to transfer the required amount of heat from a vertical plate with a view factor of F = 0.45 is 55.0 sq ft. Because of the severe radiant heat load from the sun and hot lunar surface, both shading and a high emissivity, low absorptivity surface coating, $\epsilon = .94, \alpha = 0.09$, should be used. (This is supported by the fact that a flat horizontal surface 50 sq ft in area, which is the approximate radiating area needed, will absorb 2000 Btu/hr when unshaded and having a surface absorptivity of 0.09 and a view factor of 1.00. With shading the desired heat rejection rate of 2825 Btu/hr can be obtained.)

   A vertical plate with the longitudinal edge pointing toward the sun so that both side surfaces can radiate to deep space appears to be a practical lunar application. The vertical plate would contain passages through which the coolant carying the heat from the LCG would be pumped. The proposed configuration is shown in Figure 14. Surfaces 1 and 2 partially shade surface 3 from the hot moon surface during the lunar day. Surfaces 4 and 5 are designed to shade surface 3 from the sun when the plane of surface 3 passes through the sun. In order to utilize these shading surfaces a pivot device would be built into the lunar cart as shown in Figure 15 to allow vertical axis rotation of surface 3. This movement will allow proper alignment of the shading surfaces 4 and 5 for any sun/cart orientation. It should be noted that the sun's rays may be allowed to impinge on either end of the radiating plane. Therefore, the pivot angle need only be 90 degrees. A friction device may be used to secure surface 3 in any rotationally adjusted position. The coolant passages in surface 3 must be connected to the balance of the thermal loop by flexible hoses.

   Figure 15 shows the dimensions of the Lunar ECS Cart. The included shading angle from the bottom of surface 3 to the outer edges of surfaces 4 and 5 is 11 degrees. This provides some tolerance in adjustment of surface 3 as the included sun angle

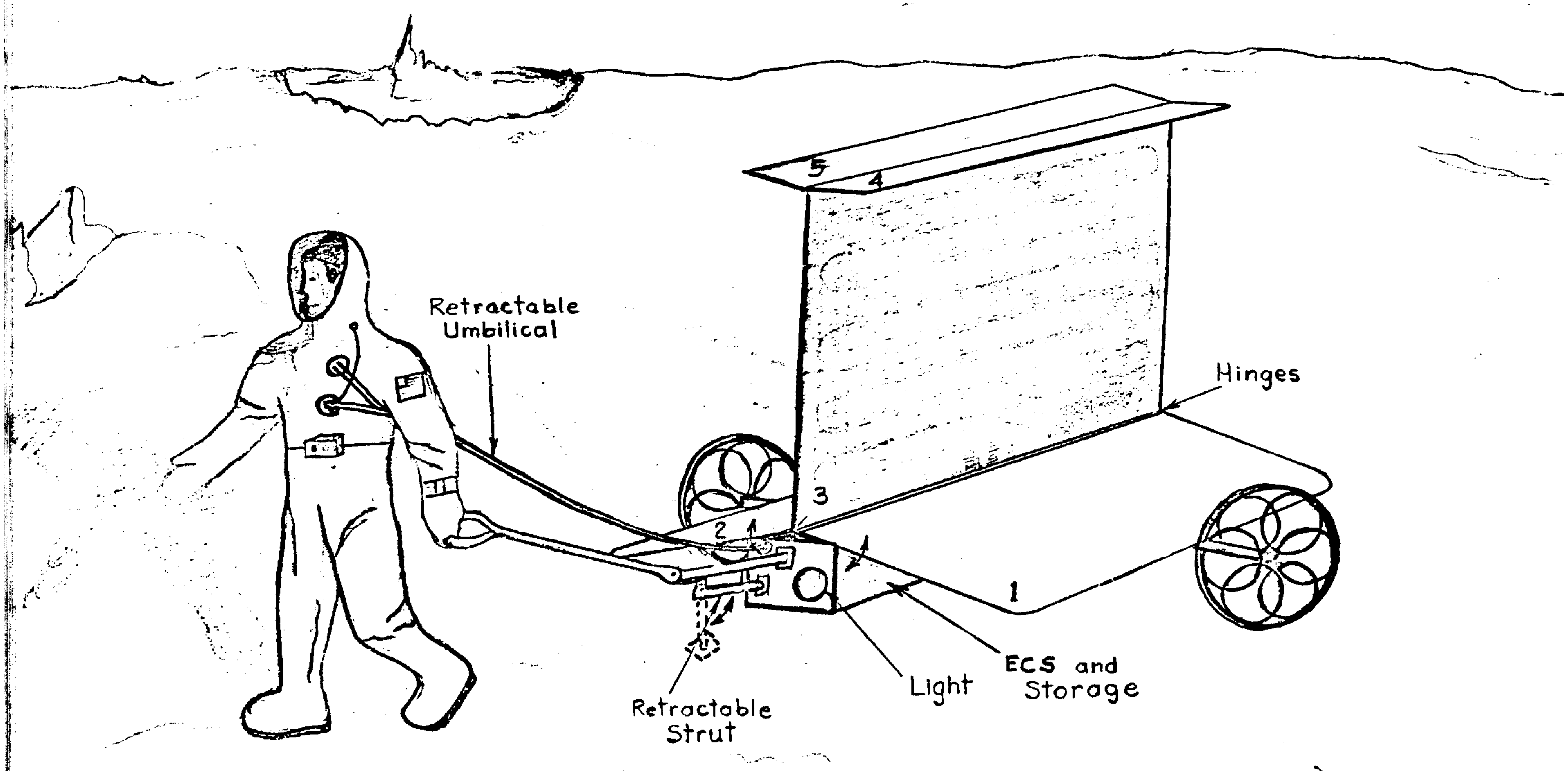


Figure 14

Lunar ECS Cart

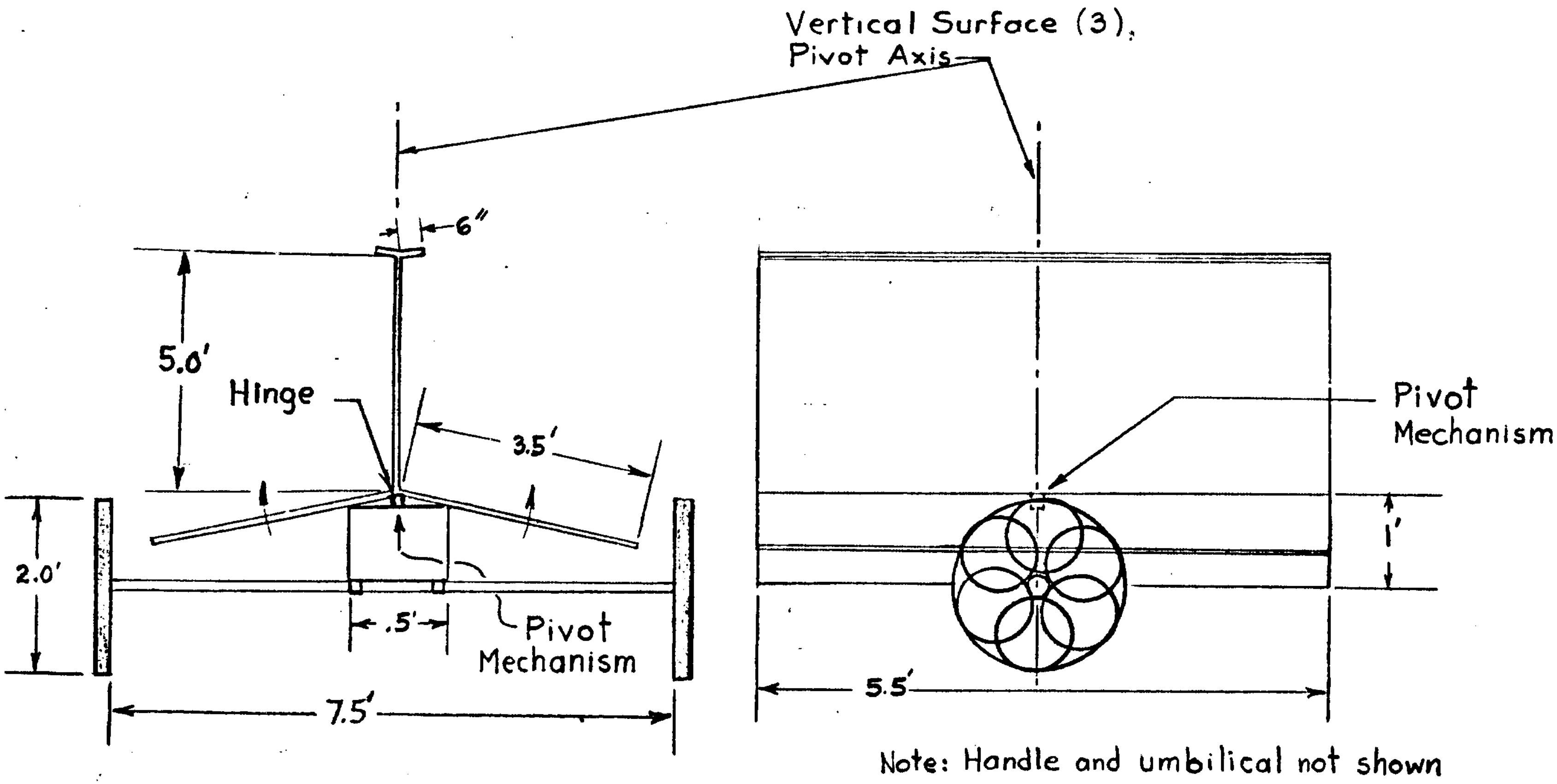


Figure 15

Lunar ECS Cart, Major Dimensions

is approximately 30 minutes. Also, the astronaut, if he travels in a straight path, will not have to stop and readjust as often to compensate for sun azimuth angle change.

The astronaut can maneuver in the vicinity of the cart on a 20-foot multiple-function umbilical which provides coolant supply and return, oxygen supply, and telemetry and communications. The cart in addition to providing a convenient carrier for the ECS is also available for carrying other tools and equipment.

Because of the low-temperature lunar night a secondary coolant loop with a low freezing point below -243°F and a critical temperature above 214°F, such as Freon 12 or R-500 would be required in the radiator.

Another feature of this design is that surfaces 1, 2, 4 and 5 can fold against surface 3; surface 3 can fold longitudinally; and the wheels may retract or fold in to form a compact package for stowage. The outside dimensions of such a package are estimated at 5.5' x 2.5' x 1'.

The principal advantage of a radiator is that no expendables are required for any number of EVA hours.

The disadvantages are the size and mass required to radiate the rejected heat quantity to space. The large size is a disadvantage because the radiator must be stowed aboard the spacecraft, it must be passed in and out through the hatch, or require making and breaking of fluid lines in a hard vacuum, it must be maneuvered by the astronaut as he moves around during the EVA, and it must be kept directed to deep space. The coolant lines which transfer the LCG heat to the radiator surface are vulnerable to meteoroid penetration and the source of a fairly sizable heat leak.

Radiator improvement is possible through lowering the α/ϵ ratio by surface coatings that are durable and exhibit greater resistance to degradation with time. Selected metals and/or layered composites that resist meteoroid penetration and are low in mass also offer potential radiator improvement. Dimensional limitations of the spacecraft hatch may be partly overcome by folding or collapsible surfaces. Also, greater freedom of mobility for the astronaut might be realized from a deployable radiator that is self-aligning in space.

2. Heat Storage Sinks

a. Heat of fusion, ice

This system utilizes the heat of fusion of ice to store the heat generated by the astronaut. The coolant which absorbs the metabolic heat as it is pumped through the LCG is circulated through a packed bed of ice where the heat is transferred to the ice at $32^{o}F$ and the ice converts from a solid to a liquid at the rate of one pound of ice to water at the fusion temperature ($32^{o}F$) for each 143.4 Btu transferred. The ice is encapsulated in small spheres with a thin plastic membrane. When the ice is converted to water the volume reduces and the plastic enclosure dimples inward. The small spheres provide a relatively uniform cooling surface and a short distance for heat conduction so that the heat absorption rate is maintained nearly constant during a four-hour mission. The quantity of unusable ice can be minimized by placing several small diameter paper-clip-shaped heat conducting wires through the center of the spheres.

The encapsulation of the fusion material permits its reuse for multiple missions without deterioration. The fusion material is refrozen between missions on the spacecraft. However, an additive is required in the circulating coolant, assuming it is water, to lower its freezing point sufficiently below the fusion temperature of ice, $32^{o}F$, in order that the encapsulated water may be refrozen during the time available between missions onboard the spacecraft. A diagram of a heat of fusion heat storage unit employing encapsulated ice is shown in Figure 16.

The advantages of this method of heat storage are that the storage unit is regenerable aboard the spacecraft between missions, no expendables are required, the fusion material, water, is nontoxic, nonirritant, noncombustible, inexpensive

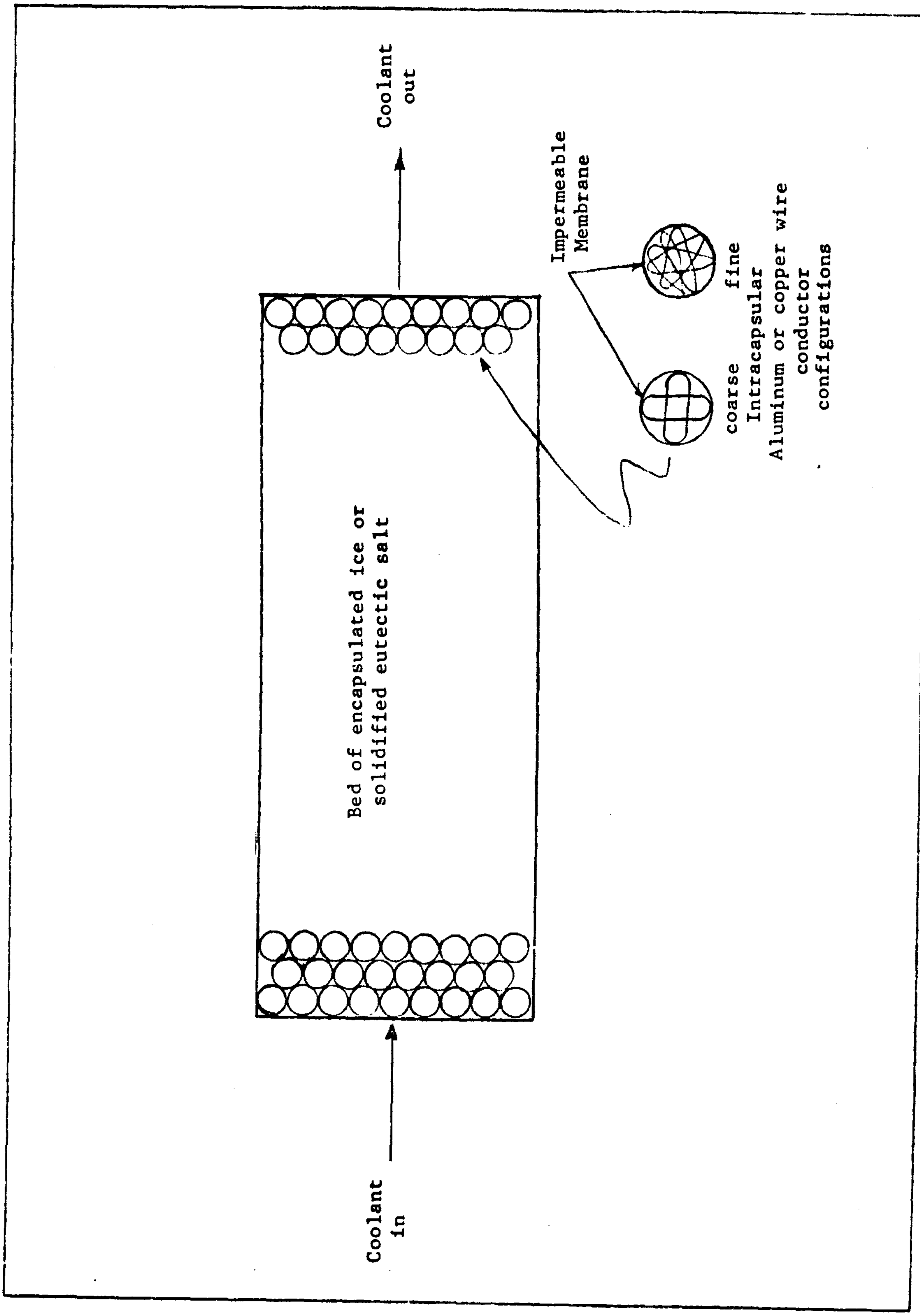


Figure 16 Heat of Fusion, Heat Storage Sink

and available, and the heat storage unit is simple in design.

The disadvantages of this method of heat storage are that a spacecraft cooling loop must be available to refreeze the encapsulated water between missions, and the regenerated encapsulated ice unit must be stored in a locker that is maintained below 32°F. The freezing point of the water in the coolant loop must be depressed with an additive to permit the ice regeneration through a heat exchanger interface.

b. Heat of fusion, eutectic salts

This approach is similar to the ice system except that a solid eutectic salt is encapsulated instead of ice. A candidate eutectic salt considered for this application is available from Melpar, Inc., Falls Church, Virginia and has a heat of fusion of 75 Btu/lb and a fusion temperature of 40°F which requires that the circulating coolant be at a temperature which is several degrees higher in order to transfer the heat load into the encapsulated eutectic salt. This higher coolant temperature is a slight disadvantage in comparison with ice in that ice can absorb slightly more sensible heat. The eutectic salt has a desirable characteristic, however, in that the coolant can not be chilled below 40°F, and therefore the coolant in turn cannot lower the skin temperature 39°F where frostbite damage can occur to tissue at points of suit impingement. The eutectic salt can be refrozen onboard the spacecraft by chilling the circulating coolant, water, without the need to use an additive to lower the coolant freezing point.

A schematic diagram of a heat of fusion encapsulated eutectic salt bed is the same as for ice and is shown in Figure 16. Advantages are similar to those given for ice. Disadvantages are also similar to those for encapsulated ice except that the coolant, water, can be used without additives to depress the freezing point, and the volume and weight of the eutectic salt required for one mission is larger than for ice.

c. Recoverable evaporant, water

This system utilizes water as the evaporant and lithium bromide as the absorbent. The coolant loop picks up the astronaut's metabolic heat when the coolant is pumped through the LCG and rejects it to the evaporator heat exchanger. The water vapor produced in the evaporator is absorbed in the absorber. A compressor is required to compress the water vapor to a slightly higher pressure than the pressure existing in the absorber which is the equilibrium pressure for the concentration and temperature of the aqueous solution of lithium bromide. The compressed water vapor is readily absorbed and the heat of absorption is conducted through heat exchange surfaces in the absorber to an integral space radiator and is radiated to space. The initial quantity of water in the evaporator/reservoir and lithium bromide solution in the absorber are sized for a four-hour mission.

Between missions the system is regenerated inside the spacecraft. To regenerate this heat sink, the evaporant supply valve is closed and the regeneration valve is opened. Heat is transferred into the absorber-heat exchanger from a regenerative heating loop causing the lithium bromide and water solution to boil thereby separating the evaporant from the lithium bromide as water vapor. The water vapor flows into the evaporant supply reservoir and is liquified. The heat of condensation is removed by circulating coolant through the reservoir. In this manner the evaporant is recovered and readied for reuse on the next mission. Thermal energy is both withdrawn from the spacecraft thermal control loop for heating the absorber and rejected to the spacecraft thermal control loop for cooling the evaporant reservoir.

A schematic diagram of a recoverable evaporant heat storage system utilizing water as the evaporant and lithium bromide as the absorbent is shown in Figure 17.

Advantages of this method are that the heat storage unit is regenerable, no additional equipment is required for additional mission hours beyond the first recharge, no expendables are required unless needed for the cooling loop aboard the spacecraft. The absorbent, lithium bromide, is nontoxic unless taken internally in large quantities, is chemically stable and has a very low vapor pressure. The evaporant, water, has broad

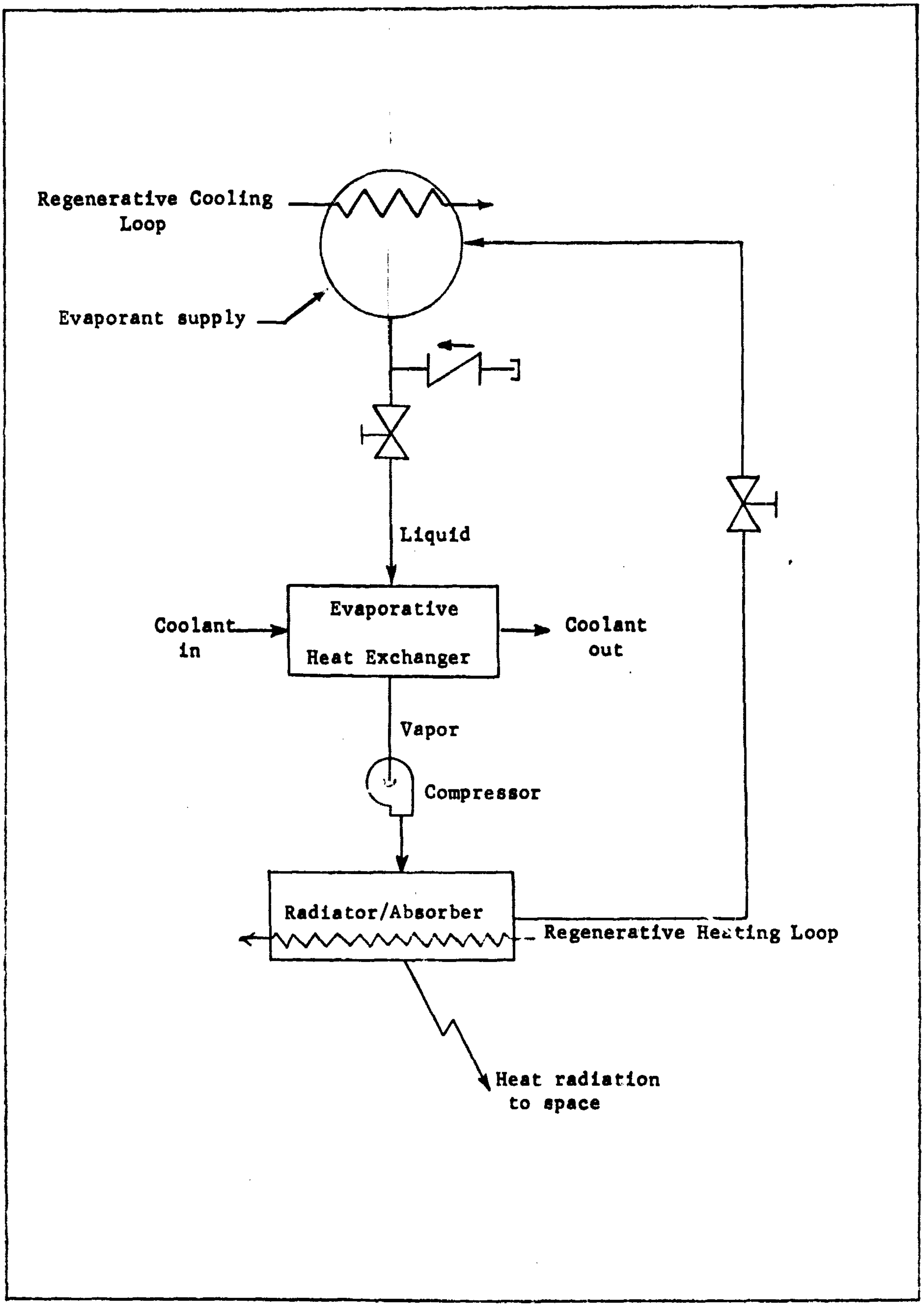


Figure 17 Recoverable Evaporant, Water, Heat Storage Sink

compatability and a high heat of evaporation.

Disadvantages are that a vapor compressor is required to increase the vapor pressure of the evaporant gas, water vapor, in the absorber so that the absorbent, lithium bromide, will absorb the water vapor at a high absorber/radiator temperature, compatible with efficient sizing of the absorber/radiator (for this study 200°F was used); the system must be regenerated onboard the spacecraft between missions (cooling and heating interfaces are required in the spacecraft for heat rejection and heat supply during regeneration); and power must be supplied by the spacecraft to operate the coolant pump during regeneration and recharge of the compressor and pump batteries. Spacecraft heating and cooling capacities should be available without requiring expendables. The volume and mass on the astronaut are large and a space radiator is needed to dispose of the heat of absorption. Controls required during the EVA periods and during the regeneration periods will add to the complexity of the system.

d. Recoverable evaporant, ammonia

This system functions in a similar manner to the recoverable evaporant, water, system except for two differences, 1) the concentration and temperature of the aqua ammonia solution in the absorber/radiator produces an equilibrium pressure below the ammonia pressure in the evaporator and the ammonia vapor flows into the absorber/radiator and is absorbed without the need of a compressor to boost its pressure and 2) the regeneration of the absorber inside the spacecraft between missions, boils off some water vapor with the ammonia vapor. This water vapor must be returned to the absorber radiator. This is accomplished by passing the boiled-off vapor mixture through a rectifier/reflux condenser.

The initial quantity of ammonia in the evaporator reservoir and water-ammonia solution in the absorber radiator are sized for a four-hour EVA mission. Between missions the system is regenerated inside the spacecraft in a similar manner to the recoverable evaporant, water, system. As a result the ammonia is recovered and ready for reuse on the next mission.

A schematic diagram of a recoverable evaporant heat storage system utilizing ammonia as the evaporant and water as the absorbent is shown in Figure 18. This diagram is similar to Figure 17 except that a gas compressor is not required between the evaporator and the absorber, and a rectifier/reflux condenser is required between the absorber and the evaporant storage unit (condenser).

Advantages are the same as listed under "recoverable evaporant, water" except that ammonia is the evaporant in this system. A gas compressor is not required in this system with its attendant volume, mass and power supply, however, a rectifier/reflux condenser is required.

Disadvantages are that ammonia, the evaporant, is toxic, ammonia is less compatible with materials of construction, the operating pressures are much higher in an ammonia-water system than in a water-lithium bromide system, and the evaporator/reservoir and absorber/radiator are larger and heavier in the ammonia-water system. The automatic controls required during EVA and during regeneration would add to the complexity of the system.

The toxicity of ammonia would tend to rule it out from further consideration, however, extra safety factors could be incorporated into the design to minimize the hazard. The size and weight of the evaporator/reservoir and absorber/generator could be further analyzed toward reducing their volume and weight if the heating and cooling interfaces in the spacecraft are compatible with the spacecraft energy system and heat balance.

3. Heat Transport Mechanisms

a. Liquid loop

Candidate coolants for the liquid loop include:

(1) Water - This is the safest of liquids and has a high specific heat. The disadvantage is the 32°F freezing temperature, which sometimes requires an additive to depress the freezing point. Additives which are often suitable follow:

(a) Ethylene glycol

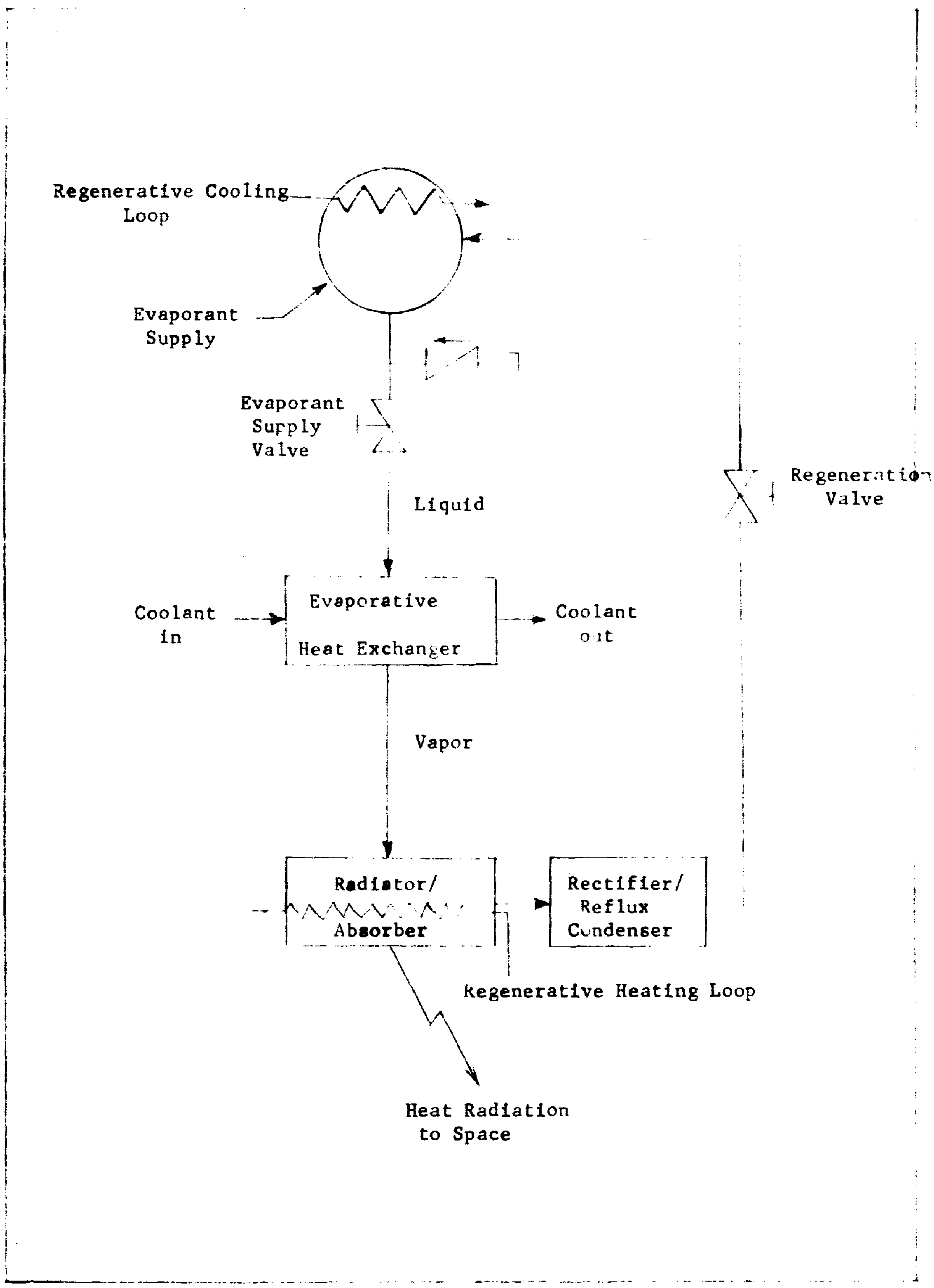


Figure 18 Recoverable Evaporant (Ammonia) Heat Storage Sink

(b) Propylene glycol

(c) Ethanol

(d) Methanol

(e) Calcium chloride

(f) Sodium chloride

(2) Freon 11, Freon 12, Freon 21, Freon 22, Freon 114, Refrig. 500 - These refrigerants are relatively safe. Their thermal conductivity is only one-fifth that of water and their heat capacity is one-fifth to one-fourth that of water, yet their freezing points are relatively low -137°F to -254°F. The use of some of low temperature heat sinks would require a secondary coolant with an equally low freezing point, or a temperature limiting device in the heat exchanger.

A schematic diagram of a liquid coolant loop using water as the coolant is shown in Figure 19. A schematic diagram of a liquid coolant loop with a secondary coolant is shown in Figure 20.

Advantages of the liquid coolant loop are that less power is required than for a gas coolant loop, water with a specific heat of 1.0 can be used as a coolant with most system concepts, additives with low toxicity are available which are miscible with water to depress the freezing point somewhat below 32°F and suitable freons are available for lower temperatures.

Disadvantages are that the liquid coolants must be kept above their freezing points, some liquids expand or contract upon freezing and can damage the hardware, the liquid coolants must be compatible with their coolant system, additives may lower the specific heat and/or thermal conductivity of the resulting coolant-mixture, and the freons are toxic and must not be released into the spacecraft.

b. Gas loop

Heat rejection to the gas loop must be considered a secondary function for PLSS systems because it is less efficient

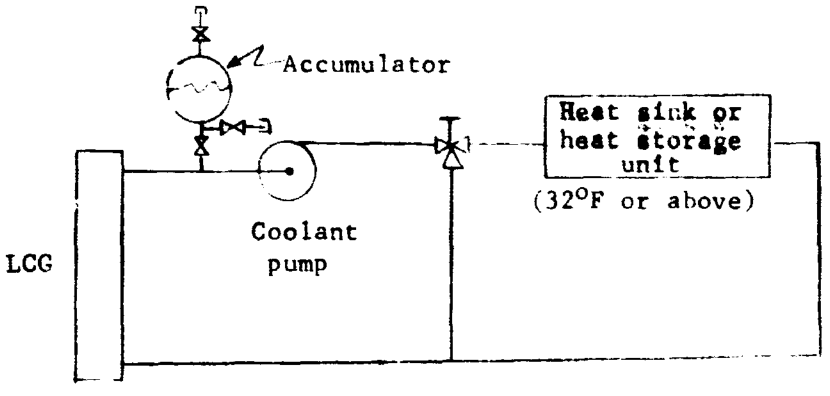


Figure 19 Liquid Coolant Loop Using Water

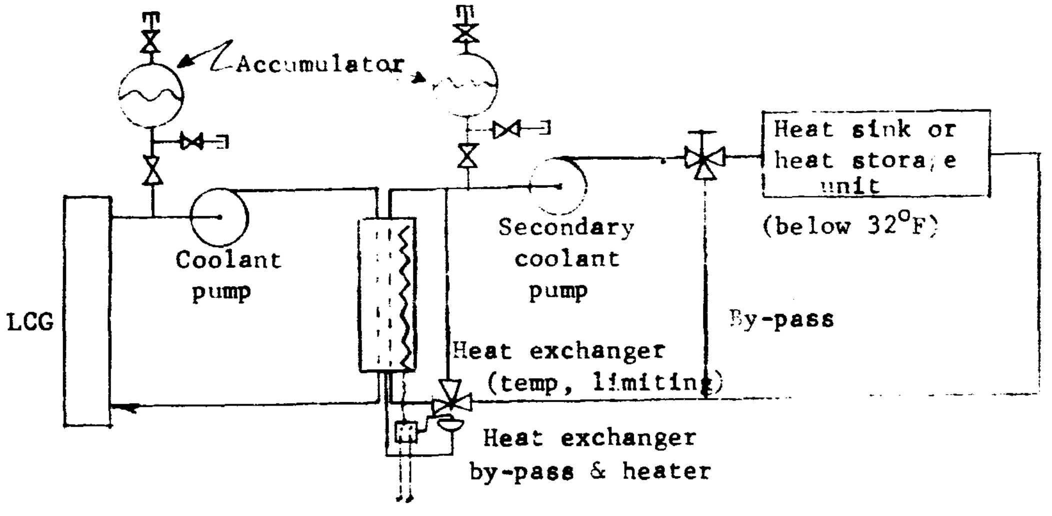


Figure 20 Liquid Coolant Loop with Secondary Coolant for Low Temperature Heat Sink

than liquid loops from the standpoint of the power required and the heat transfer per unit area. Since a minimum ventilation rate is required to remove carbon dioxide and sensible and insensible water from the suit, the consideration here is how to best reject latent and sensible heat from the gas loop. Two approaches can be used: (1) rejection of heat directly to the heat sink, and (2) rejection of heat to the liquid loop with subsequent rejection of all of the heat by the liquid loop to the heat sink. As a general rule it is better practice to reject all of the heat load through the liquid loop because design of the heat sink is greatly simplified. This simplification of design also generally results in economies in packaging that more than offset the volume required for the separate condensing gas loop heat exchanger.

c. Heat conduction through suit

This heat transport system utilizes a helium-porous insulation which can be "filled" by introducing low pressure helium gas which effuses through the insulation and increases the heat transmission through the suit. When the metabolic heat production decreases, the rate of heat transmission through the suit insulation is decreased by venting the helium from the insulation to space vacuum. A correlation between helium pressure within the insulation envelope and heat transfer rate permits the automatic regulation of the latter by a temperature-sensing, pressure regulating control. A cross-section of a space suit insulated with an open-cell, helium-fillable insulation is shown in Figure 21.

Advantages of this heat transport method are that this is a passive heat transport system, with the thermal resistance controllable by the helium gas pressure within the open-cell insulation. Tests on some insulation samples showed that the thermal conductivity K (Btu, inch/sq ft, ^{o}F, hour) varied from 0.06 to 0.7 when the helium pressure varied between 0.1 and 10 torr.

Disadvantages of this heat transport method are that expendable helium is required, that it must be vented to space whenever less heat is to be transferred away from the astronaut, that the space suit must be surface-coated for a low α/ϵ ratio, that the surface coating may degrade with time, that meteoroid puncture of the outer suit

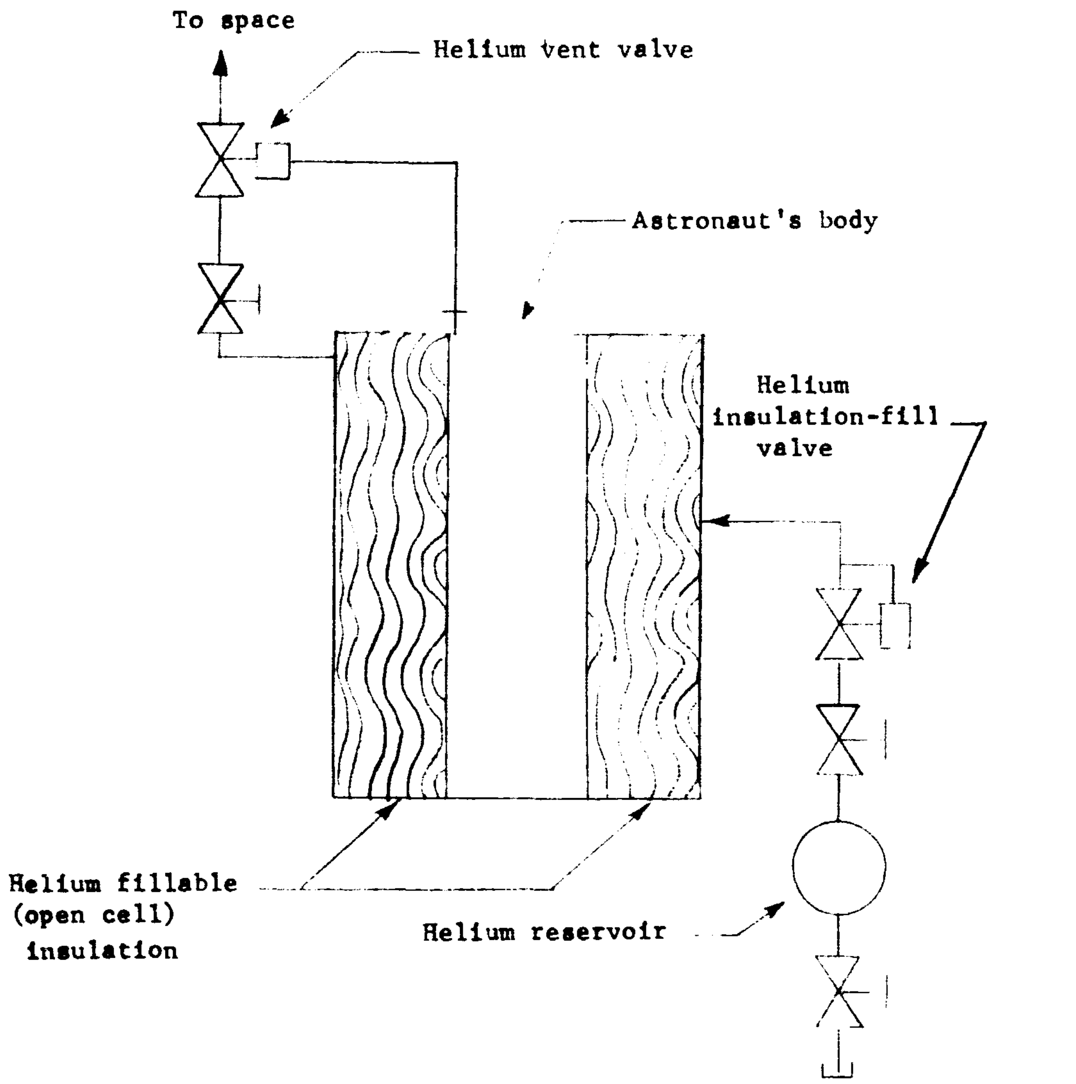


Figure 21 Cross-section of Helium Fillable Insulation in Space Suit Heat Transport Mechanism

cover will make it difficult to build the helium pressure sufficiently to dissipate the required heat quantity, and the insulation tested showed a three-fold increase in the heat transfer rate in earth orbit on the earth-sun line when the evacuated insulation was filled with helium (and a five fold increase in heat transfer rate in the earth umbra when the evacuated insulation was filled with helium). Hydrogen gas would increase the thermal conductivity of the insulation slightly more than helium, but it cannot be used because of its combustible nature.

d. Refrigeration

Use of the refrigeration or heat pump cycle allows the use of heat sinks that are at a higher temperature than the temperature of the coolant loop from which the heat is rejected. When used in conjunction with a radiator, a smaller radiator could be used to reject the metabolic load. However, the resultant increase in heat load quite often more than offsets the gain in temperature and a larger radiator is required. Note that all heat pump devices, including vapor refrigeration cycles, absorption cycles, and thermoelectric Peltier-type devices, are limited to a maximum efficiency as expressed by the Carnot cycle

$$\eta_{max} = \frac{T_r - T_e}{T_r}$$

where η = efficiency, T_r = radiator temperature, and T_e = temperature of fluid to the astronaut. This efficiency for a 200°F maximum radiator and 40°F equipment is only 24.3%. Thus, for a 2000 Btu/hr equipment load, the cycle power load would be 6250 Btu/hr, resulting in 8250 Btu/hr minimum for 200°F versus 2000 Btu/hr for 40°F, leading to a minimum 36% increase in radiator area. If a suitable endothermic chemical reaction could be found, a refrigeration cycle might be effectively utilized. However, the thermal load of the refrigeration cycle must be subtracted from the heat of reaction.

Refrigeration cycles considered in this study include: the vapor-compression cycle, the absorption cycle, Brayton cycle, fog cycle, Joule-Thompon cycle and the Vortex (Hilsch) tube cycle. A brief description of each refrigeration cycle is included in the following paragraphs.

(1) Vapor compression

This type of system would evaporate the working fluid by utilizing the astronaut-generated heat and compress the resulting vapor to a higher pressure and temperature at which temperature the heat will flow to a heat sink and liquify the vapor. The liquid pressure and temperature is then reduced sufficiently low so that heat will flow again from the astronaut and cause the liquid to boil and evaporate. An evaporator, gas compressor, condenser and expansion valve are required to accomplish the above together with controls which automatically provide for start-up, shut-down, load variations, temperature control, compressor protection and safety. A source of power must be provided to drive the gas compressor.

Disadvantages are that a power source must be provided, the system is complex when all components and controls are considered, and must be designed to be gravity-independent. A schematic of this system is shown in Figure 22.

(2) Absorption

This system is similar to the compression cycle except for the following differences. The refrigerant vapor is absorbed by a liquid or solid called an absorbent which has a high affinity for the vapor. The refrigerant is then separated from the absorbent by adding heat. The regenerated refrigerant vapor is now of a higher temperature and pressure and heat will flow from the vapor to a heat sink until the vapor liquifies. The liquid refrigerant is then reduced in pressure again by passing it through a restriction causing heat to flow from the astronaut causing the liquid to boil and evaporate. The hardware required to accomplish this includes an evaporator, absorber, liquid pump, heat exchanger, generator, condenser and expansion valve. If the absorbing liquid has a significant vapor pressure in the generator, a rectifier and reflux condenser must be added to prevent it from entering the condenser and evaporator. A source of heat must be provided to the generator to boil off and separate the refrigerant vapor from the absorbing liquid.

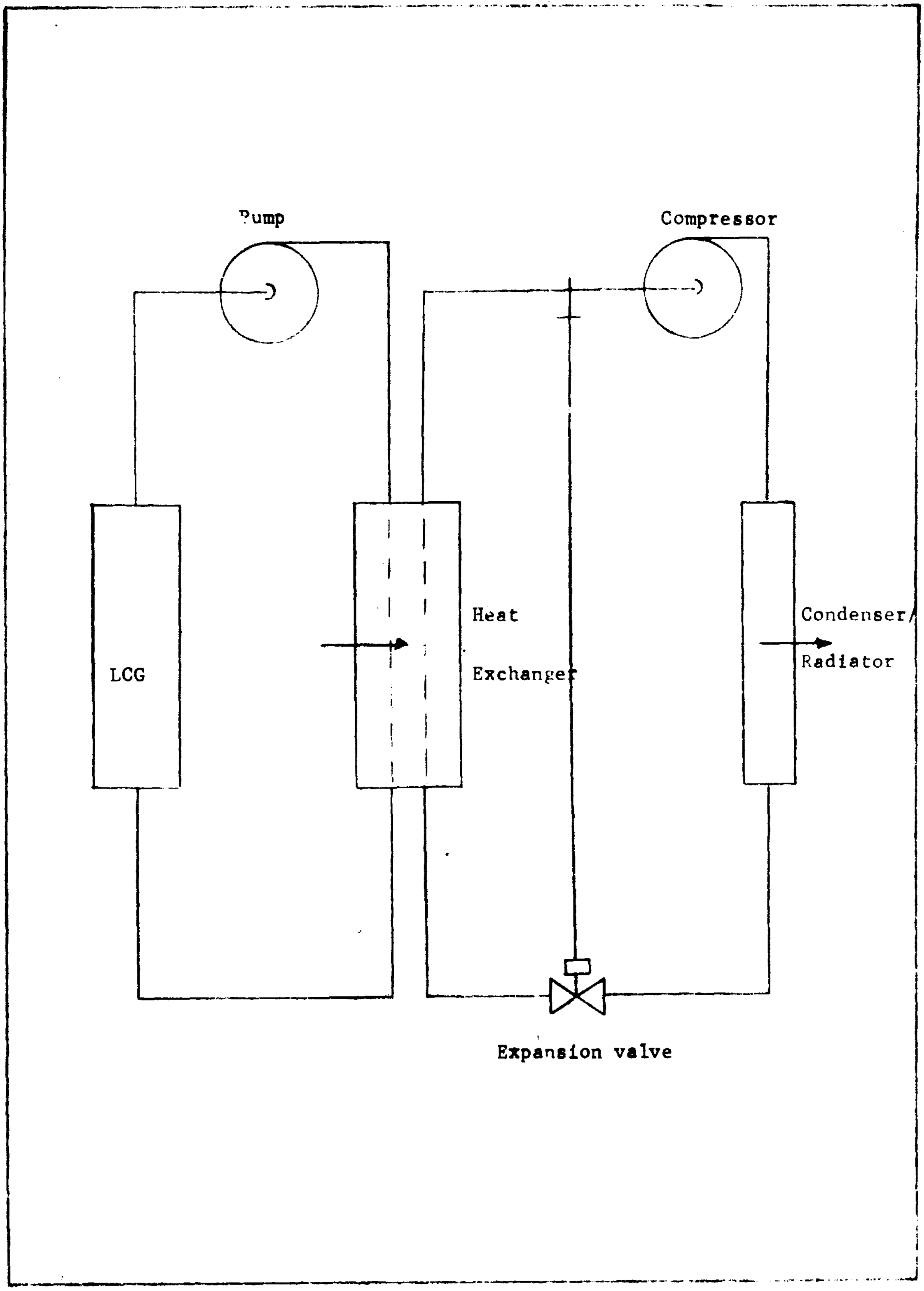


Figure 22 Refrigeration, Vapor Compression

Disadvantages are that a power source must be provided, that too much power is required, that there is a large volume and weight penalty, heat of absorption plus heat of condensation must be rejected, that the system must be designed to be designed to be gravity independent and that the system will be complex. The advantage of a system which has no moving parts such as is the case with the Servel refrigerator is lost in zero gravity because the Servel cycle utilizes gravity to produce convective currents for circulation. The Servel cycle is also extremely attitude sensitive and would not be applicable to portable systems without considerable development effort. A schematic of this system is illustrated in Figure 23.

(3) Brayton Cycle

The Brayton cycle receives heat generated by the astronaut and transfers it into the working fluid, a gas, which is then compressed. The hot compressed gas is then expanded in a turbine to a lower temperature and absorbs heat transfered from the astronaut. The energy absorbed by the expander is used toward driving the gas compressor. However, some additional energy must be supplied to the gas compressor to complete the gas compression since the heat source, the astronaut, is at a lower temperature level than the heat sink, such as a space radiator. A schematic diagram of the Brayton cycle applied to the coolant loop of a space suit heat rejection system is shown in Figure 24.

Disadvantages are that a gas compressor, two heat exchangers and a gas expander are required, power is required for the compressor, all the components and controls make the system complex, the volume and weight additions are large and gas expansion irreversibility in the expander results in additional work to drive the compressor, which requires electric power.

(4) Fog Cycle

This concept is based on using a liquifiable vapor as a heat transport medium. The vapor is circulated by means of a vapor pump between the LCG where it

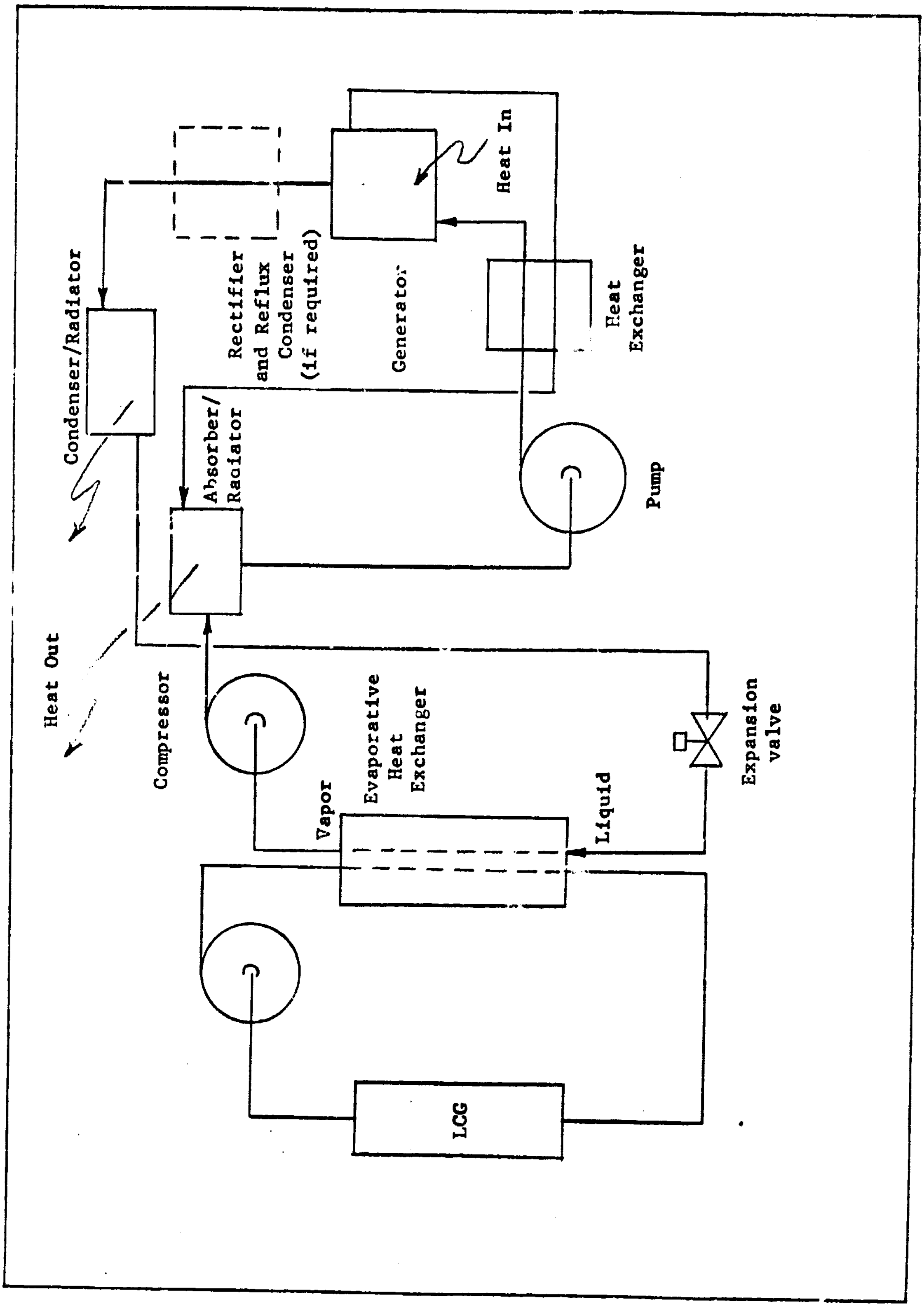


Figure 23 Refrigeration, Absorption

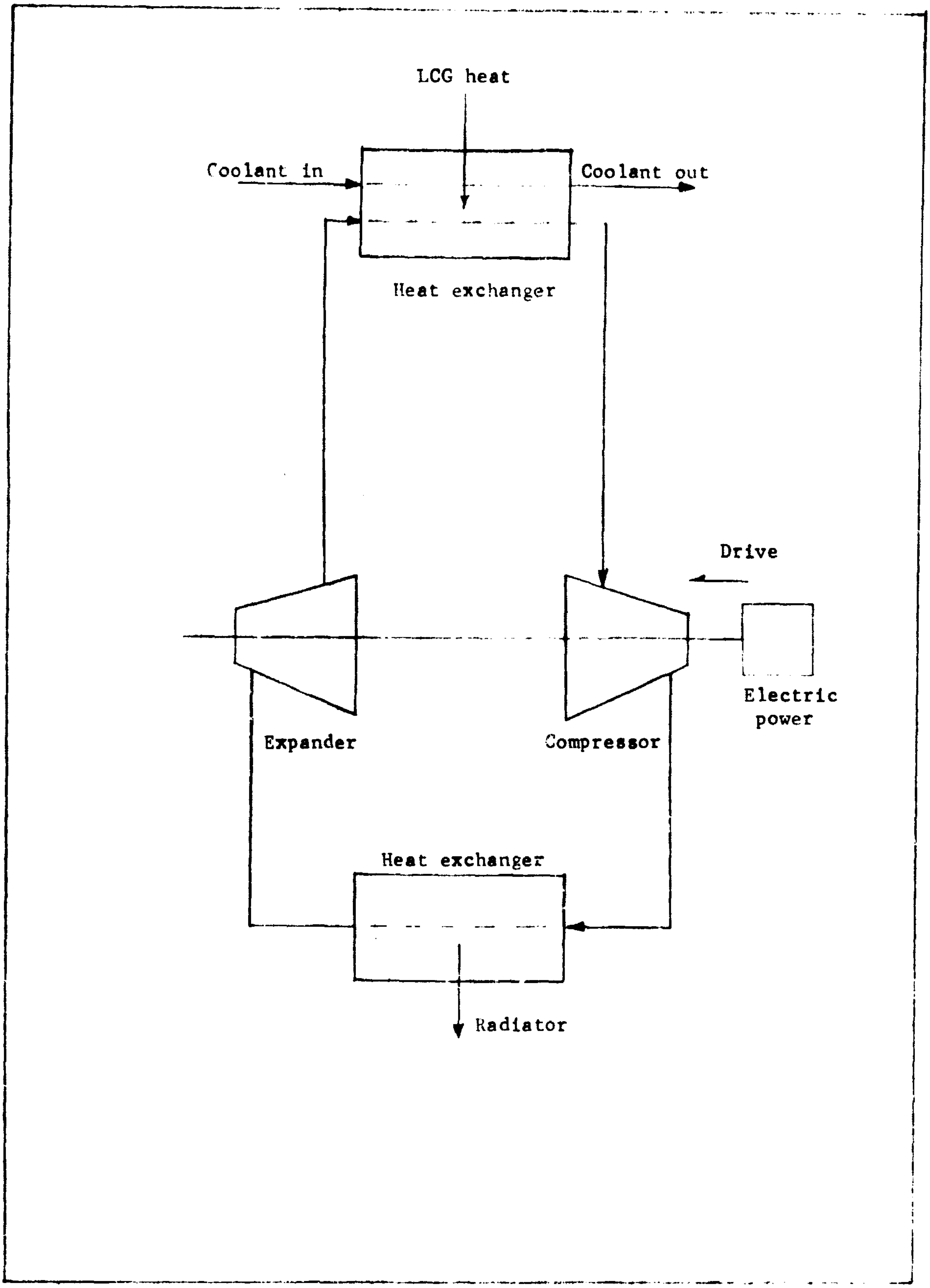


Figure 24 Refrigeration, Brayton Cycle

absorbs the astronaut's generated heat and gives up the absorbed heat. The quality of the fog increases in the LCG as some of the suspended liquid is converted into vapor and the quality decreases as the fog is passed through the heat sink and some of the vapor is converted back to suspended liquid. A schematic diagram of the fog cycle heat transport method applied to a space suit heat rejection system is shown in Figure 25.

Disadvantages are that a vapor pump and two heat exchangers are required, the system becomes complex with components and controls, and the volume and weight additions must be considered.

(5) Joule-Thomson Cycle

Heat generated by the astronaut is transferred into the cool circulating gas in a heat exchanger. The gas is then compressed and heat from the compressed gas transfers into a heat sink. The gas is then allowed to expand which cools it as it enters the heat exchanger to again receive heat generated by the astronaut. Power must be supplied to drive the gas compressor. A schematic diagram of the Joule-Thomson cycle applied to the coolant loop of a space suit to reject the astronaut heat to a heat sink is shown in Figure 26.

Disadvantages are that a gas compressor, two heat exchangers and a gas expansion valve are required, all the components and the controls make the system complex, the volume and weight are large, the gas expansion irreversibility in the expansion valve is large and results in a low system efficiency, and electric power is required to drive the compressor.

(6) Vortex (Hilsch) Tube Cycle

This system utilizes the principle of the vortex or Hilsch Tube. Heat from the astronaut LCG is transferred by heat exchanger into a circulating gas which is then compressed to a higher pressure and temperature level. Heat flows from this hot gas to a heat sink. The gas is then directed circumferentially into a tube at high velocity. The resulting spinning turbulence transfers heat within the gas causing the

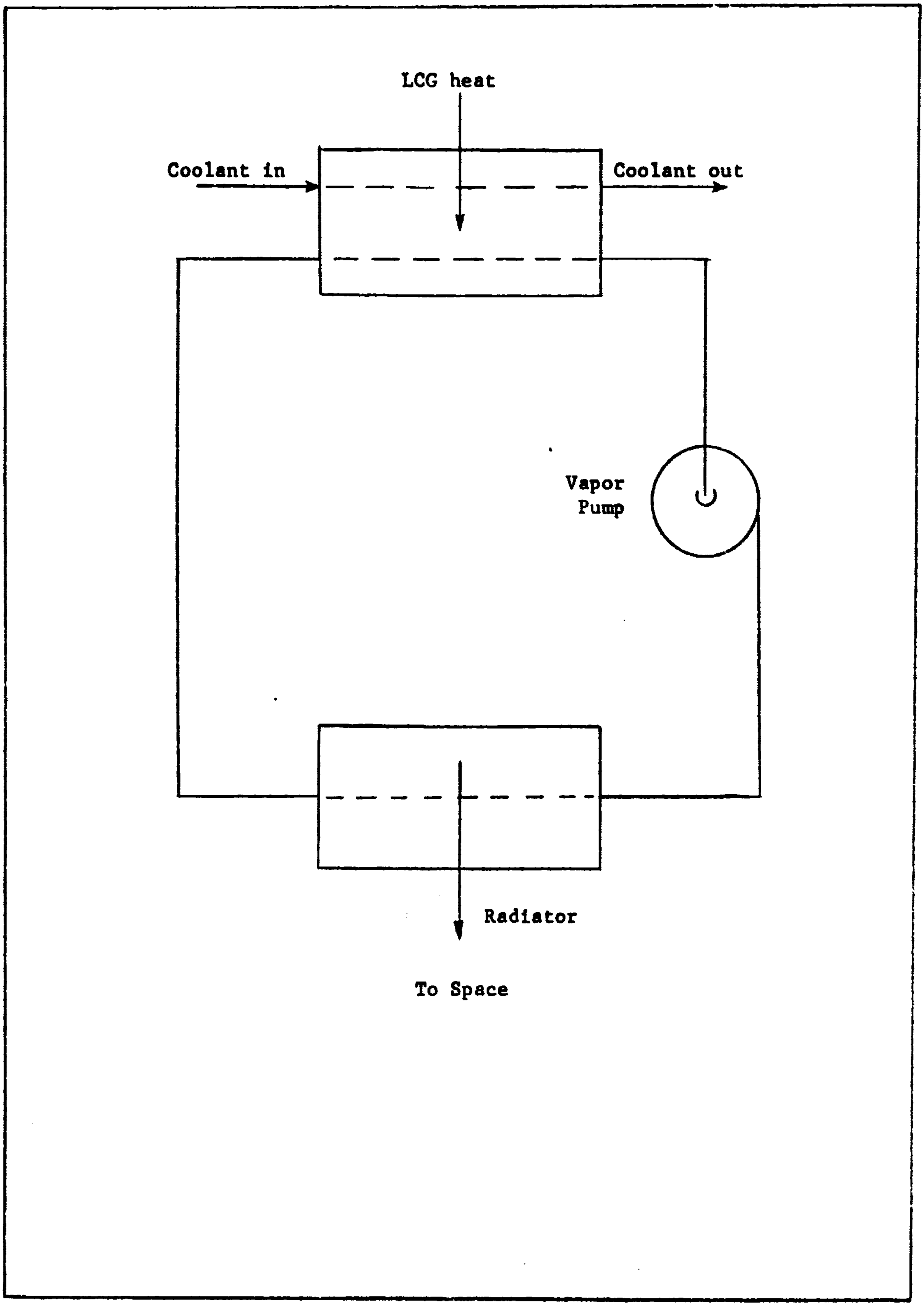


Figure 25 Refrigeration, Fog Cycle

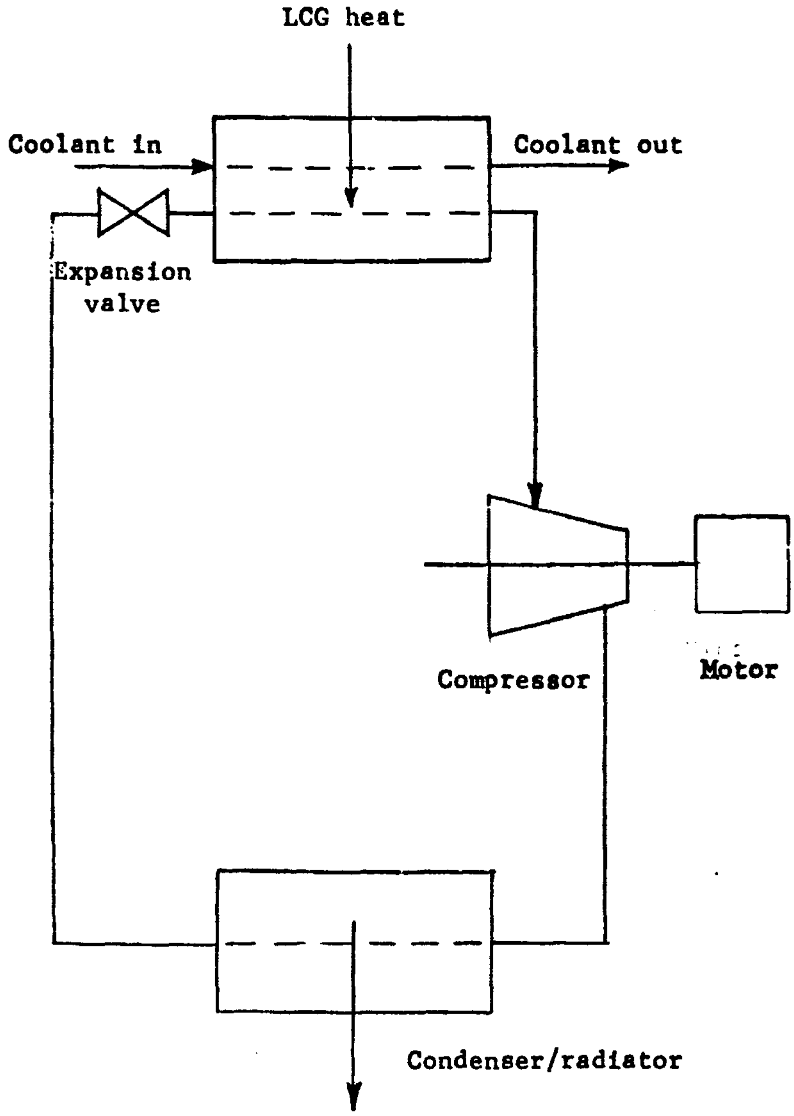


Figure 26 Refrigeration, Joule-Thomson Cycle

temperature to drop in the gas at the center of the vortex. The cool gas flows through the heat exchanger where it again receives heat from the LCG. Since only a portion of the gas is cooled in the vortex tube, the balance of the (warm) gas is bled from the vortex tube to a heat exchanger where it is cooled by heat transfer into a heat sink. This cooled gas is then mixed with the gas that has picked up the LCG heat and the gas mixture is recompressed for flow through the vortex again. A schematic diagram of the vortex tube applied to a space suit rejection system is shown in Figure 27.

Disadvantages are that a gas compressor, three heat exchangers, a vortex tube and controls are required, the system is complex, the volume and weight are large, the irreversibility of the vortex tube results in a low system efficiency and high power requirements, and the vortex tube acoustic levels may be objectionable to the astronaut.

7. Peltier thermoelectric devices

This system utilizes two different materials connected by a cold junction. The opposite sides of the materials, called A and B, are connected to a hot junction into which a direct current is introduced. The heat transfer between the cold and hot junctions in watts is the product of the current in amperes introduced between the two legs of the hot junction and the Peltier coefficient for materials A and B.

In spacesuit application, the cold junctions of the multiple Peltier units would be in contact with the astronaut's skin and the heat to be rejected would flow to the hot junction where it would be dissipated by a space radiator or other heat sink. A schematic diagram of a concept for the application of a Peltier thermoelectric device to space suit heat rejection is shown in Figure 28.

Advantages of a Peltier heat transport device are that it is a simple, passive system with no moving parts, requires no expendable fluids, and is gravity independent.

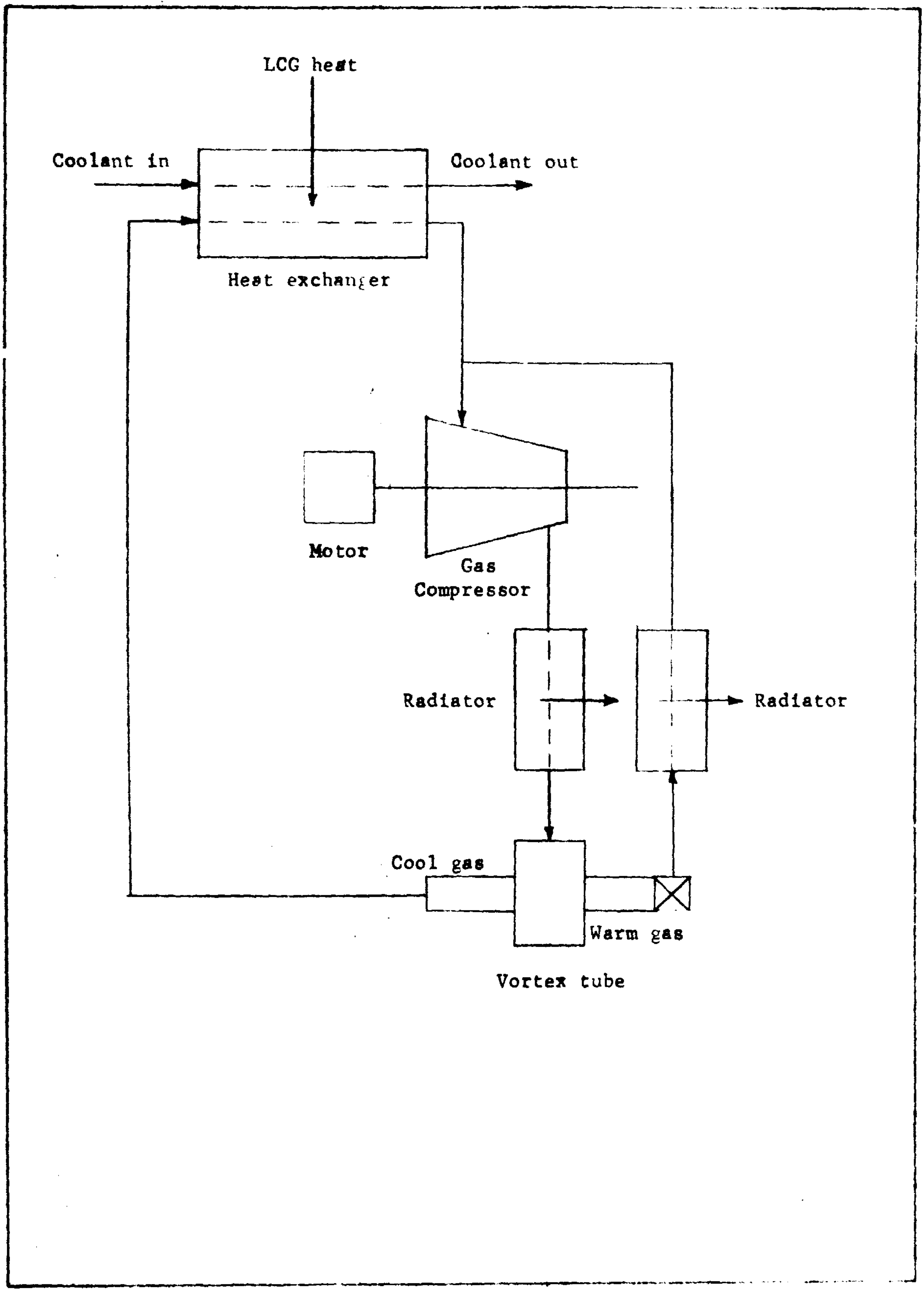


Figure 27 Refrigeration Vortex (Hilsch) Tube Cycle

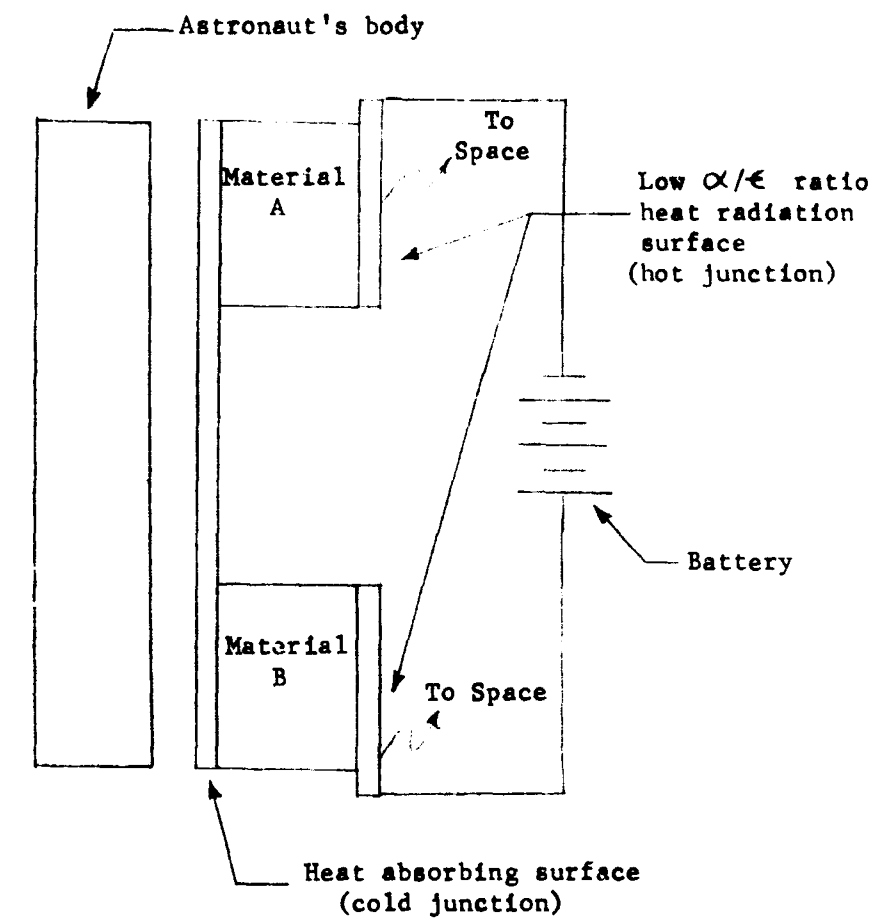


Figure 28 Refrigeration Peltier Thermoelectric Cycle

Disadvantages of this heat transport method are that electric power is required and efficiency is very low.

e. Wick heat pipe

This system utilizes the principle of evaporation in one end and condensation in the other end of a sealed enclosure which is evacuated and then charged with a refrigerant which will vapor-phase upon the application of heat and which will liquid-phase upon the removal of heat, simultaneously, within the same enclosure. The vapor produced flows from the evaporator end to the condenser end due to a small pressure differential caused by the condenser surface being cooler than the evaporator surface. The condensed liquid returns to the evaporator surface through the capillary action of a wick or capillary screen which connects the condensing surface with the evaporating surface. Choice of the working fluid is largely determined by vapor pressure at operating temperatures, latent heat of vaporization, surface temperature and safety. Surface contamination and the formation of noncondensable gases must be prevented inside the heat pipe during its operation. A schematic diagram of a concept for the application of a heat pipe for the rejection of heat from a space suit is shown in Figure 29.

Advantages of the heat pipe are that it is a simple, passive system, without moving parts, requires no expendables, can be sealed to keep out contaminants, can be designed to operate with a small temperature difference between its evaporating surface and its condensing surface, and can be designed to transport large heat quantities.

Disadvantages of the wick heat pipe are that it requires a wick or screen which is sensitive to contamination, requires evacuation before charging, must be free of contamination, requires hermetic sealing, and when the pipes are integral with the space suit, they may constrain astronaut mobility.

f. Grooved heat pipe

This heat pipe utilizes grooves in place of the wick or capillary screen. The grooves are designed to reduce resistance to liquid flow and to improve heat transfer on the evaporating and condensing surfaces. Optimization procedures have been developed by Martin Marietta

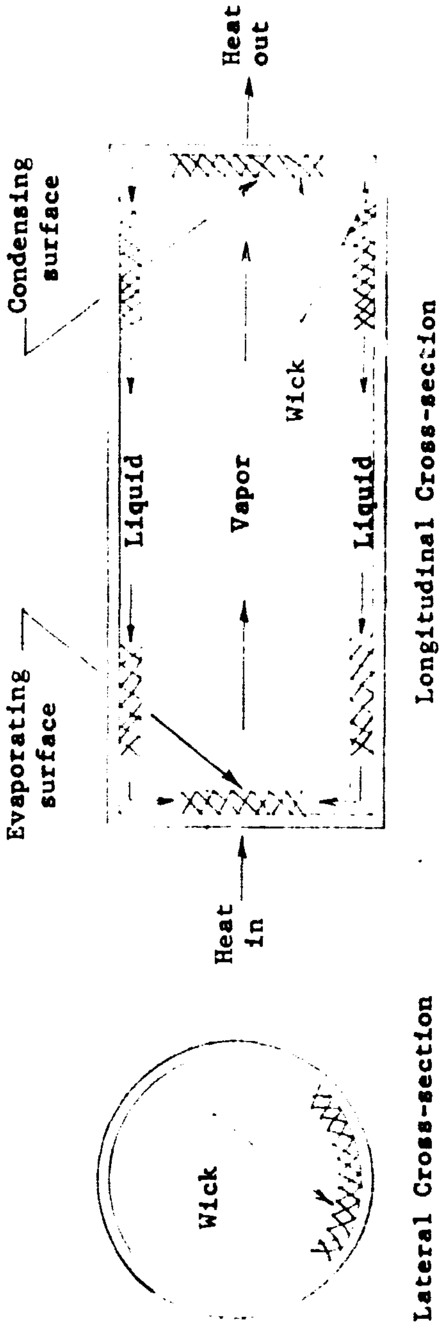


Figure 29 Wick Heat Pipe

Corporation for determining the width and depth of rectangular grooves in a tubular pipe for maximum heat transport rate or the minimum pipe diameter for the required heat transport rate. Analysis and tests shown that a capillary screen placed over the grooves to separate the liquid flow channels from the vapor flow passage further improves the results. For proper operation, contaminants which will cause internal surface deterioration and the formation of noncondensable gases must be excluded during manufacture of the heat pipe. Sketches of the grooved heat pipes are th be found in Figure 30.

Advantages are the same as given for wick and screen type heat pipes except that this configuration has a lower resistance to fluid flow, is not dependent upon snug wick contact with the inner wall, has less entrapment of vapor bubbles than a wick, and will transport liquid faster and therefore more heat between ends than a wick-type heat pipe.

Disadvantages are that the inner surfaces of the pipe and the screen are sensitive to contamination, wetting characteristics of the inner surfaces and the heat pipe performance can degrade with time, the pipe must be completely evacuated before it is charged, the pipe requires hermetic sealing and when the pipes are integral with the space suit they may constrain astronaut mobility.

g. Capillary-pumped liquid-vapor loop

This system incorporates phase change into the functional requirements for the heat transport liquid in the LCG which encircles the astronaut's body. The LCG is designed with closely spaced loops of small diameter tubing. The volume of vapor formed by heat from the astronaut's body fills most of the tubing except for a liquid film on the inside wall. The expanding vapor moves rapidly toward the cooler portion of the tubing causing the liquid wall film to flow along with the vapor though at a little slower velocity due to tube wall drag. The vapor condenses on the inside walls in the cooler portion of the tubing, or the heat rejection unit, and as the liquid volume increases, the liquid bridges across the tubing due to surface tension and forms liquid slugs. The slug size increases as the vapor trapped between the slugs condenses until the tubing is filled with liquid. The liquid moves through the last portion of the tubing loop

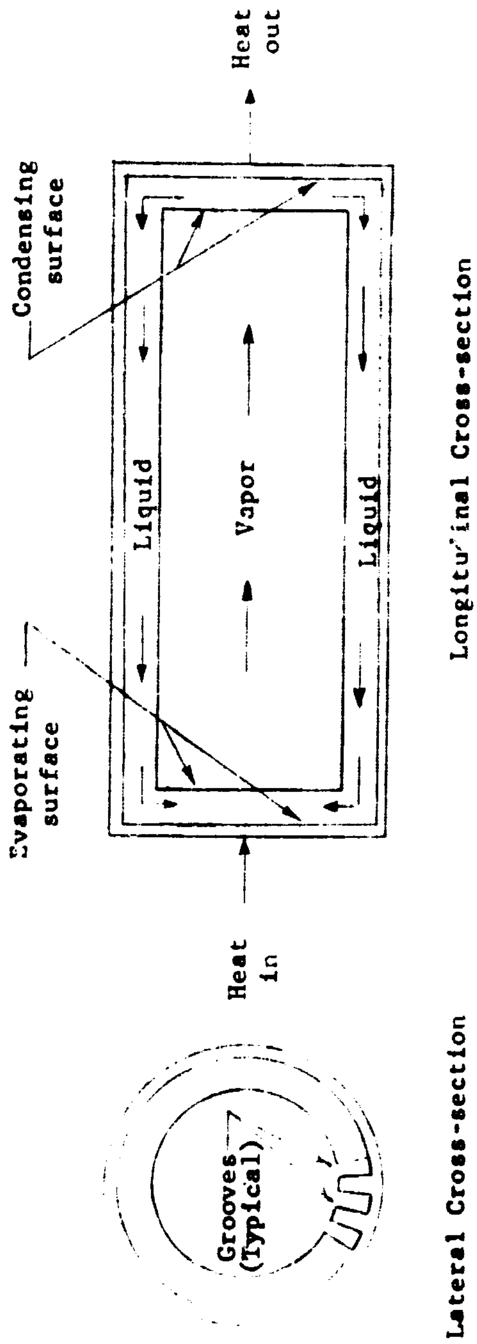


Figure 30 Grooved Heat Pipe

back to the LCG.

Near the LCG the liquid enters a capillary wick which continues inside the tubing while the liquid is absorbing heat from the astronaut's body. The liquid has absorbed sufficient heat that a portion of the liquid vapor phases at it emerges from the wick. The wall film continues to vaporize until all the LCG heat is absorbed. The surface tension of the liquid causes it to continue to feed from the wick into the vapor-forming space in the tubing. The higher pressure produced by the forming vapor provides the force for moving the vapor away from the LCG and through the tubing. A schematic diagram of a capillary-pumped heat transfer loop is shown in Figure 31.

Advantages of this method of transporting heat are that it is a simple, passive system with no moving parts, uses no expendable liquids and functions in both zero-g and one-g.

Disadvantages are that the heat transport tubing loop must be free of noncondensable gases, the tubing size must be small enough so that the accumulating condensed liquid will bridge across the tubing in the condenser section and form liquid slugs, the capillary section must be short to avoid significant pressure drops, the capillary pumping section is sensitive to contamination during manufacture, and performance may degrade with time as the result of breakdown of the monomolecular surfaces of the capillaries. The size limitation of the tubing will limit the quantity of heat that can be transported by one network of tubing. Therefore, several parallel networks may be required.

h. Integral heat pipes/suit radiator

This heat transport mechanism is a passive design made up of two layers. The layer next to the astronaut's skin is a trapezoidal corrugated suit shell with each trapezoid containing a constant conductance heat pipe. The outside layer is made up of variable conductance heat pipes. The conductance of heat through the variable pipes is controlled by separating the evaporating surface from the condensing surface and placing a vapor flow control valve between the two areas or chambers. The liquid condenses in the condensing chamber, returns to the heat pipe surface adjacent to the corrugated suit shell and flows through the heat pipe vapor valve. This

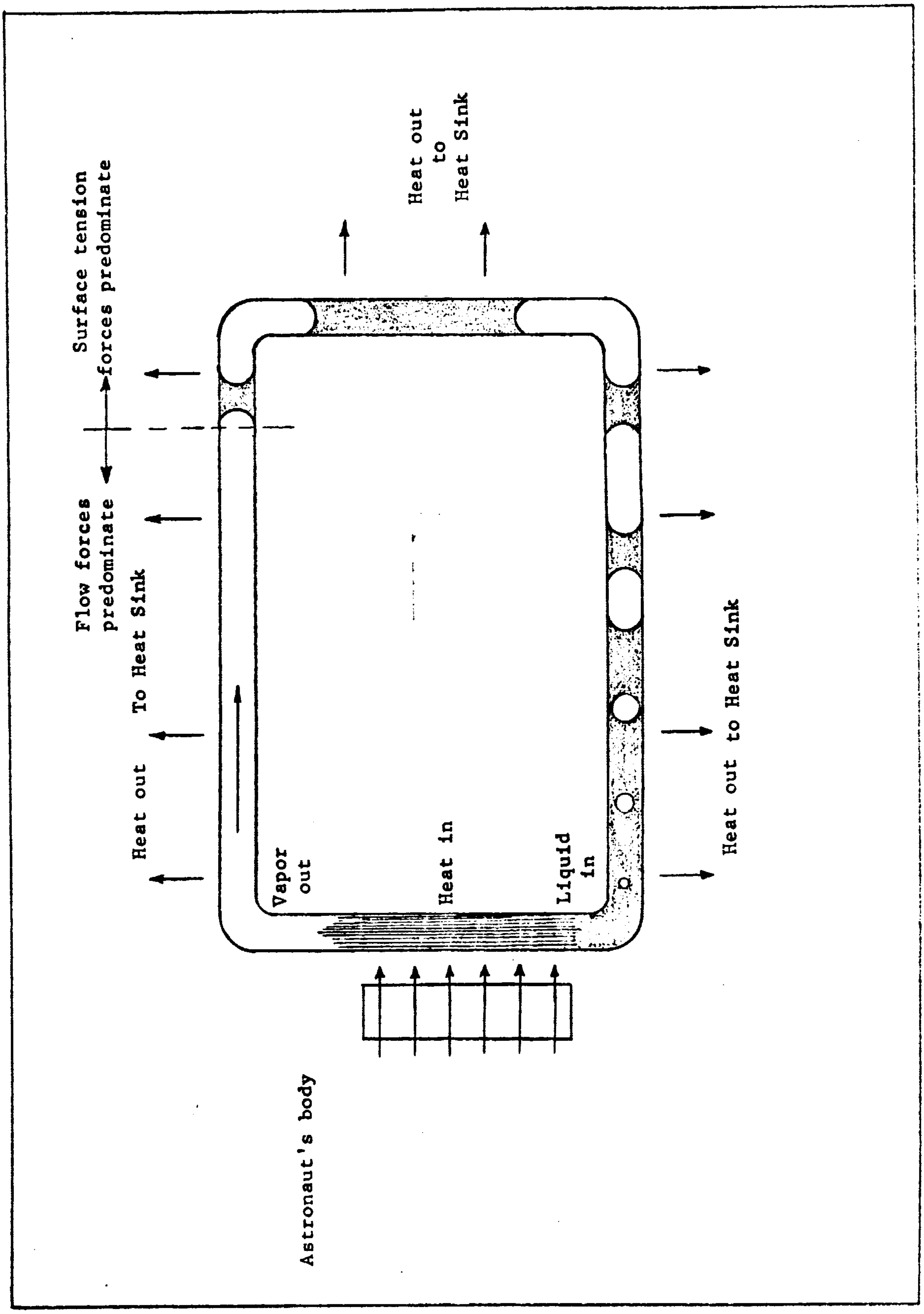


Figure 31 Capillary-Pumped Liquid/Vapor Loop

valve is controlled by a temperature sensor and bellows. When the heat quantity generated by the astronaut decreases, or when the condensing/radiating surface in the heat pipe is warmer than the evaporating surface, the valve restricts or closes thus matching the heat radiation quantity to the heat generated quantity and maintains the suit shell at a comfortable temperature. schematic cross-section of this heat transport mechanism is shown in Figure 32.

Advantages of this heat transport mechanism are that it is passive and requires no expendable fluids.

Disadvantages of this mechanism are: a) the capillary wicks are sensitive to contamination and may degrade with time, and b) that the condensing/radiating surface must see deep space to be effective.

B. CANDIDATE SYSTEM SELECTION

The foregoing concepts were qualitatively evaluated and the less promising concepts were disqualified. System approaches were then synthesized from the more promising concepts, and the more promising approaches were selected for further consideration.

1. Selection Criteria

For a system to be selected for further consideration an advantage must be apparent in at least one of the following categories:

1. The candidate system must have equal or smaller total launch weight than state of the art heat rejection systems.
2. The candidate system must require a smaller volume on the man.
3. The candidate system must possess characteristics which make possible the development of more reliable and less sensitive flight hardware.

2. Selected Systems

As a result of preliminary screening and technical reviews, eleven candidate systems were selected for further consideration. Nine of the candidate system consisted of a liquid-cooled-garment

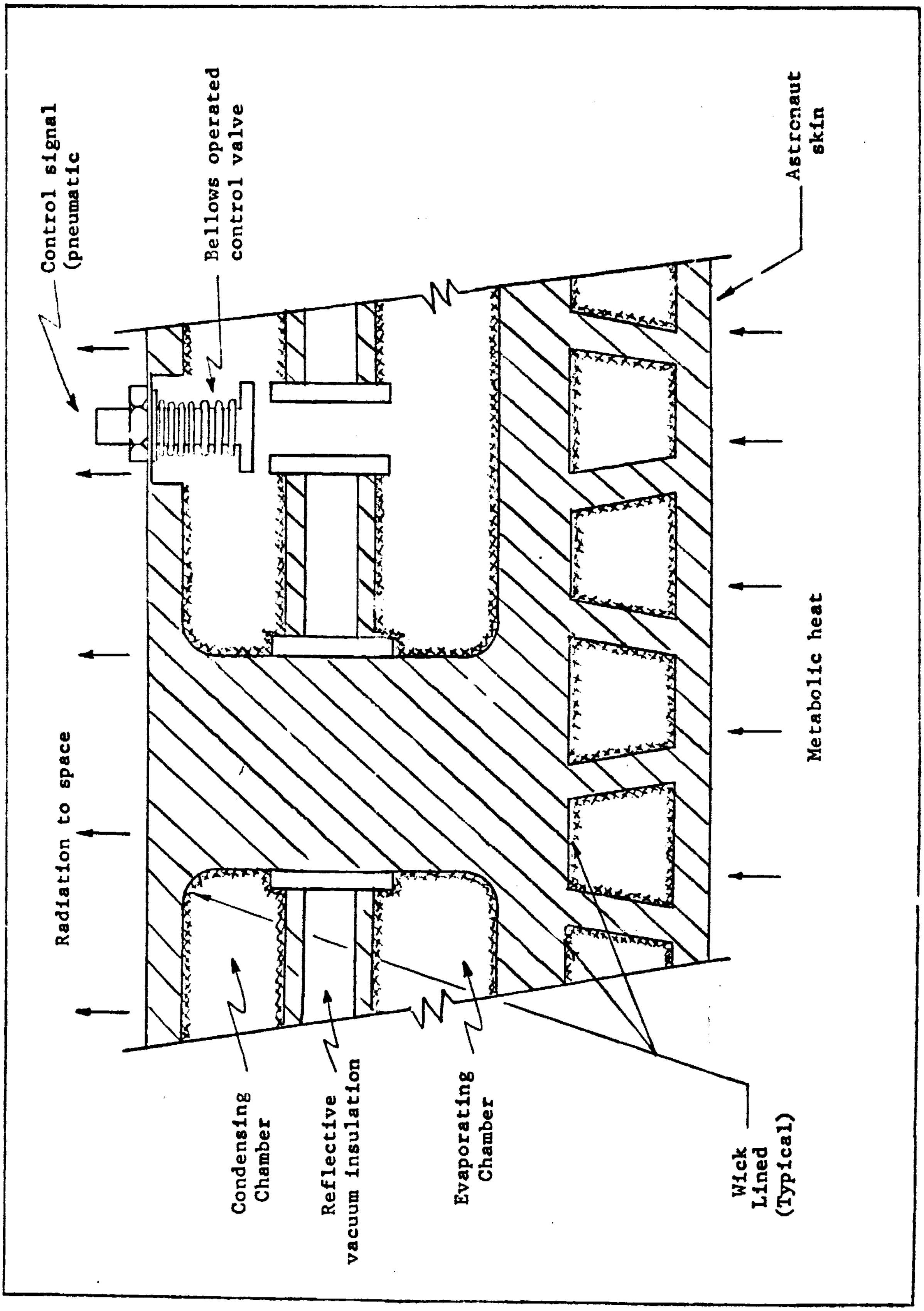


Figure 32 Cross-Section of Integral Variable-Conductance Heat Pipes/Suit Radiator

heat-transport loop and one of the following sinks:

a. Wick-fed evaporator

b. Forced vortex boiling evaporator

c. Porous plate sublimator

d. Gelled water sublimator

e. Heat of fusion, ice

f. Heat of fusion, eutectic salt

g. Recoverable evaporant, water

h. Recoverable evaporant, ammonia

i. Space radiators

The other two systems also selected were:

j. Integral vaporization-diffusion

k. Integral heat pipes/suit radiator

3. Considerations Pertinent to Each System

a. Wick-fed evaporator

The system incorporating the wick-fed evaporator was considered essentially to be the advanced Portable Environmental Control System (PECS) configuration. However, it is recognized that additional development effort on the wicks would be desirable to decrease their sensitivity to contamination.

b. Forced vortex boiling evaporator

This system is similar to the previous one except that a water injection shell-and-tube core with tube turbulators is substituted for the wick-fed core. The primary advantage to this system is that it eliminates the necessity for capillary devices and therefore may be less sensitive to contamination.

c. Porous plate sublimator

The system considered is essentially that of the Apollo PLSS. However, like the wick-fed evaporator system additional development effort appears desirable to decrease the sensitivity of the capillary device, in this case, the porous plate, to contamination.

d. Gelled-water sublimator

This system is very similar to the porous plate sublimator except that a fine-mesh screen, or screens is substituted for the porous plate and a gelling agent is added to the water. Based on Martin Marietta's past experience with gelled propellants, this system appears feasible. This approach would have the advantage of eliminating the porous plate, but it would require special flight support equipment for between mission recharging.

e. Heat of fusion

Systems incorporating heat of fusion sinks are another approach which has the advantage of being regenerable. Two systems, each containing different fusion materials were examined, these are water and an eutectic salt from Melpar, Inc. In both cases the design was based upon encapsulating beads of the heat of fusion material in an impermeable membrane and then packing the beads into a bed. Heat transfer is then accomplished by passing the coolant through the bed. Using water as the heat of fusion material provides a significant volume advantage as compared to the eutectic salt, however, a compatible, non-toxic, additive must be added to the coolant to depress its freezing point. Provision must be made in the spacecraft for heat sink resolidification.

f. Recoverable evaporant

The recoverable evaporant heat sinks combine either the wick-fed evaporator or the forced vortex boiling evaporator with an absorption bed/radiator and a compressor. While both systems are regenerable, the volume on the astronaut is large and spacecraft interfaces for both cooling the evaporator reservoir and heating the absorbent bed during regeneration are required.

g. Space radiators

The advantage of a space radiator is that no expendables are required. The large envelope size, volume and mass make the radiator difficult to manage in passing through the spacecraft hatch, difficult to maneuver during EVAs, and difficult to keep pointed toward deep space for heat rejection. The latter disadvantage could be overcome by deep space sensing and automatic orientation aligning controls, however, these controls will add to the volume and weight of the radiator assembly.

Efforts to find a suitable application fot the heat rejection radiator evolved into the lunar ECS radiator cart. The lunar surface with its one-sixth gravity provides easy mobility for the mass of a wheel-mounted radiator. The irregular moon surface suggests a two-wheel cart with a retractable parking strut. The cart-mounted radiator could be folded as necessary for passage through the spacecraft hatch and storage onboard. The size and mass of the radiator could be minimized by a pivot mount to keep the two radiator surfaces facing deep space. For details of the ECS lunar cart, see Radiators under Concept Descriptions.

h. Integral vaporization-diffusion

This approach is the approach being developed by McDonnel Douglas, Santa Monica, California for the NASA Manned Spacecraft Center.

i. Integral heat pipe/suit radiator

For this analysis the utilization of the variable conductance heat pipe and suit radiation system being developed for NASA/Ames was considered. Heat rejection is limited by available suit surface area and is insufficient to reject the entire heat load. From the interim program report, NASA-CR-73168, the maximum heat rejection rate is given as approximately 2000 Btu/hr. Because this method cannot reject the total required heat load, it was later deleted.

4. Parametric Analysis

A parametric analysis was performed to determine volume, weight and power variance between the selected candidate systems for EVA hours ranging from one four-hour mission up to 50 four-hour multiple missions or 200 mission hours total for one man. If an EVA mission of four-hours duration is made by two astronauts, this would equal twice the parameter required for one astronaut for four hours (i.e., the data point for four hours for one man must be selected and multiplied by two).

Analyses were performed on the previously listed candidate systems, with the exception of the integral heat pipe/suit radiator system which was included in the volume and weight table but not considered further in the parametric study because of insufficient heat rejection capability. In addition, a 50/50 combination of a heat rejection sink and a heat storage sink, (wick fed evaporation and ice fusion) was also included. The 50/50 ratio was chosen because it is half-way between the 100 percent heat rejection system and the 100 percent heat storage system, and illustrates the parametric advantages and disadvantages of combined systems. A special design to meet specific requirements would likely result in the selection of a different ratio.

The schematic diagrams in the component description section show the arrangement and identify the necessary components for functioning systems. The thermophysical and heat transfer analyses determined the approximate volumes, pressures and surface areas required. These values provided the basis for estimating the typical envelope size, volume, dry weight and operating (wet) weight for each component. Where possible, the size, volume, and weight was based on components previously qualified for space. For each system having a gas loop and liquid loop, the assumption was made that all gas loop heat is rejected to the liquid loop with subsequent rejection of the combined liquid and gas loop heat load to the heat sink.

For the systems employing a coolant loop to transport the heat rejected from the LCG to the heat sink, the hardware required to make up the coolant transport loop, its volume, dry weight and wet weight are summarized in Table 1. The total volume and weights for the heat transport loop and the required auxiliaries such as evaporant pressure control valve and evaporant reservoir valves are added to the volume and weights for each heat sink to obtain the total volume and weight on the astronaut less power supplies. Volume and weights on the astronaut, less batteries, are summarized in the Table 2.

The power required for one astronaut for one four-hour EVA mission is assumed to be supplied by four-hour batteries for each heat rejection system. Silver-zinc batteries with a volume of

Table 1

COOLANT TRANSPORT LOOP

Item	Volume Cu. In.	Dry Weight lbs.	Wet Weight lbs.
Pump and motor	30.5	2.0	2.2
Coolant by-pass valve	4.5	1.0	1.1
Coolant accumulator	16.0	.5	1.0
Coolant tubing	28.0	2.9	3.9
TOTAL	71.0	6.4	8.2

Table 2
Volumes and Weights on Astronaut of Selected
Candidate Heat Rejection Systems
(Less Batteries)

System	Active Agent	Component	Typical Envelope	Volume cu. in.	Weight - lbs dry	Weight - lbs wet	Remarks
Wick Fed Evaporation	Water	Evaporator	6.0 x 6.0 x 3.0	105.0	4.0	6.0	
		Evap press control valve	2.25 x 2.25 x 6.0	30.5	1.3	1.5	
		Evaporant reservoir	6.0 x 6.0 x 10.5	370.0	4.5	16.5	
		Evap reservoir valves	(3) 1.0x1.0x2.0	6.0	1.0	1.5	
		Subtotal		511.5	10.8	25.5	
		Heat transport loop		71.0	6.4	8.2	
		Total		582.5	17.2	33.7	
Forced vortex boiling evaporation	Water	Evaporator	5.6 dia x 12 in	165	6.0	8.0	
		Evap press control valve	2.25 x 2.25 x 6.0	30.5	1.3	1.5	
		Evaporant reservoir	6.0 x 6.0 x 10.5	370.0	4.5	16.5	
		Evap reservoir valves	(3) 1.0x1.0x2.0	6.0	1.0	1.5	
		Subtotal		571.5	12.8	27.5	
		Heat transport loop		71.0	6.4	8.2	
		Total		642.5	19.2	35.7	
Sublimation	Water	Sublimator	6.0 x 12.0 x 3.0	216.0	6.4	7.7	
		Evaporant reservoir	6.0 x 6.0 x 10.5	370.0	4.5	16.5	
		Evap reservoir valves	(3) 1.0x1.0x2.0	6.0	1.0	1.5	
		Subtotal		592.0	11.9	25.7	
		Heat transport loop		71.0	6.4	8.2	
		Total		663.0	18.3	33.9	
Sublimation	Gelled Water	Sublimator	6.0 x 12.0 x 3.0	216.0	6.4	7.7	
		Reservoir	6.0 x 6.0 x 10.8	390.0	4.8	17.0	
		Control valves	(3) 1.0x1.0x2.0	6.0	1.0	1.5	
		Subtotal		612.0	12.2	26.2	
		Heat transport loop		71.0	6.4	8.2	
		Total		683.0	18.6	34.4	
Recoverable evaporant	Ammonia and water	Evaporative heat exchanger and reservoir	18.0 x 18.0 x 3.6	1162	14.6	35.6	Spacecraft regeneration is required
		Absorber/radiator	36 x 36 x 2.5	2437	82.2	82.2	
		Valves	2.0 x 2.0 x 2.0	8	1.2	1.2	
		Subtotal		3607	98.0	119.0	
		Heat transport loop		71	6.4	8.2	
		Total		3678	104.4	127.2	

Table 2 (Continued)

System	Active Agent	Component	Typical Envelope	Volume cu. in.	Weight - lbs dry	Weight - lbs wet	Remarks
Recoverable evaporant	Water and lithium bromide	Evaporative heat exchanger and reservoir	18.0 x 18.0 x 1.5	486	8.0	21.0	Spacecraft regeneration is required
		Absorber/radiator	30.0 x 38.4 x 3.2	3260	165.0	165.0	
		Valves	2.0 x 2.0 x 2.0	8	1.2	1.2	
		Compressor/motor	3.0 x 3.0 x 6.0	54	4.0	4.0	
		Subtotal		3808	178.2	191.2	
		Heat transport loop		71	6.4	8.2	
		Total		3879	185.1	199.4	
Heat of fusion	Water/ice	Fusion bed	18.0 x 18.0 x 8.3	3200.0	102.0	122.0	Resolidification in spacecraft is required. Freezing point in coolant loop must be depressed with an additive
		No auxiliaries reqd.					
		Heat transport loop		71.0	6.4	8.2	
		Total		3271.0	108.4	130.2	
		(Freezing point 32°F)					
Heat of fusion	Eutectic salt	Fusion bed	18 x 18 x 10.9	3826.0	185.0	209.0	Resolidification in spacecraft is required
		No auxiliaries reqd.					
		Heat transport loop		71.0	6.4	8.2	
		Total		3897.0	191.4	217.2	
		(Freezing point 40°F)					
Lunar radiator	Radiator	Cart mounted radiator		6400	136.0	138.0	(includes 20 ft. umbilical)
		Heat transport loop		381.0	37.0	42.8	
		Total		6781	173.0	180.8	
Integral Heat pipes/Suit radiator	Radiator	Radiator/heat pipes integrated into hard shell of suit		3040		60	2000 BTU/hr maximum heat rejection rate. Volume and weight estimated from NASA Report NASA-CR-73168
Integral diffusion-vaporization	Water	Evaporative cooled garment	1.0 cu ft, stowed	436	17.7*	31.2	Actual prototype vol. and dry wt. All other values are estimated.
		Water feed control	(6) 1 x 1 x 1.5	9	1.5	2.1	*Contractor estimates that flight model would weight under 10 lbs. (wet weight would reduce accordingly)
		Vent press. control	2.25 x 2.25 x 6.0	30.5	1.3	1.5	
		Reservoir valves	(3) 1 x 1 [illegible]	6	1.0	1.5	
		Reservoir (incl. in garment)					
		Total		481.5	21.5	36.3	

Table 2 (Continued)

System	Active Agent	Component	Typical Envelope	Volume cu. in.	Weight - lbs dry	Weight - lbs wet	Remarks
Combination 50/50 evaporation/fusion	Water/ice	Evaporator		52.5	2.0	3.0	
		Evap. press. control	2.25 x 2.25 x 6.0	30.5	1.3	1.5	
		Liquid reservoir	6 x 6 x 5.25	185.0	2.3	8.4	
		Liquid reservoir valves	(3) 1 x 1 x 2	6.0	1.0	1.5	
		Subtotal		274.0	6.6	14.4	
		Coolant heat transport loop		71.0	6.4	8.2	
		Subtotal		345.0	13.0	22.6	
		Fusion Bed		1740.0	51.3	66.4	
		Total		2085.0	64.3	89.0	

0.33 cubic inches per watt and 0.29 pounds per watt were assumed. The watts of power required by each system were multiplied by these volume and weight factors to determine each systems power-volume and power-weight penalty. The results of these determinations are listed in Table 3.

Battery volumes and weights for four EVA hours were added to those listed in Table 2, to arrive at the total volume and total weight on one astronaut for one four-hour EVA mission. These total values are shown on two bar charts, Figures 33 and 34.

Next, multiple mission usage up to a total of 50 four-hour missions for one man was considered. It was assumed that the multiple missions would take place during a single spacecraft flight and that a period of time equal to each prior mission would be available inside the spacecraft to prepare the heat rejection system of the portable life support system package for the next mission. It was further assumed that the candidate system was charged on the ground prior to launch for the first four-hour EVA mission, therefore, spacecraft support is required for the second and subsequent EVA missions. Spacecraft support power and thermal heat transfer requirements for regeneration of each hour of EVA are listed in Table 4. To determine the total support power and thermal heat transfer requirements for a mission having N hours of EVA, the spacecraft support power and thermal heat transfer requirements for one hour are multiplied by N-4.

Fuel cells are assumed to supply the spacecraft power during regeneration and the fuel cell power penalty imposed is 250 cubic inches and 1.33 pounds per kilowatt hour. This penalty is charged against each system requiring spacecraft regeneration power in accordance with the values shown in Table 4.

The water supplied to the evaporation and sublimation systems is assumed to be transferred from the spacecraft storage tank at average spacecraft temperature by utilizing available gas pressure to provide the transfer force.

The systems utilizing the heat of fusion of ice and eutectic salt would be resolidified in the spacecraft giving up latent heat via a heat exchanger to a spacecraft "cooling loop" operating below the ice or eutectic salt freezing temperatures.

TABLE 3

BATTERY REQUIREMENTS FOR ONE ASTRONAUT FOR ONE EVA HOUR AND FOR FOUR EVA HOURS

	Battery Equivalent					
	One EVA Hour			Four EVA Hours		
SYSTEM	Power Watts	Cubic Inches	Pounds	Power Watts	Cubic Inches	Pounds
Evaporation, wick fed, water	6	2	0.2	24	8	.69
Evaporation, forced vortex, water	6	2	0.2	24	8	.69
Sublimation, water	6	2	0.2	24	8	.69
Sublimation, gelled water	6	2	0.2	24	8	.69
Heat of fusion, ice	6	2	0.2	24	8	.69
Heat of fusion, eutectic salt	6	2	0.2	24	8	.69
Recoverable evaporant, ammonia	10	3	0.3	40	13.3	1.14
Recoverable evaporant, water	162	54	4.6	648	214	18.50
Combination, 50/50, evap./fusion	6	2	0.2	24	8	.69
Lunar radiator cart	10	3	0.3	40	13.3	1.14
Integral heat pipes/suit radiator	0	0	0	0	0	0
Integral diffusion-vaporization	0	0	0	0	0	0

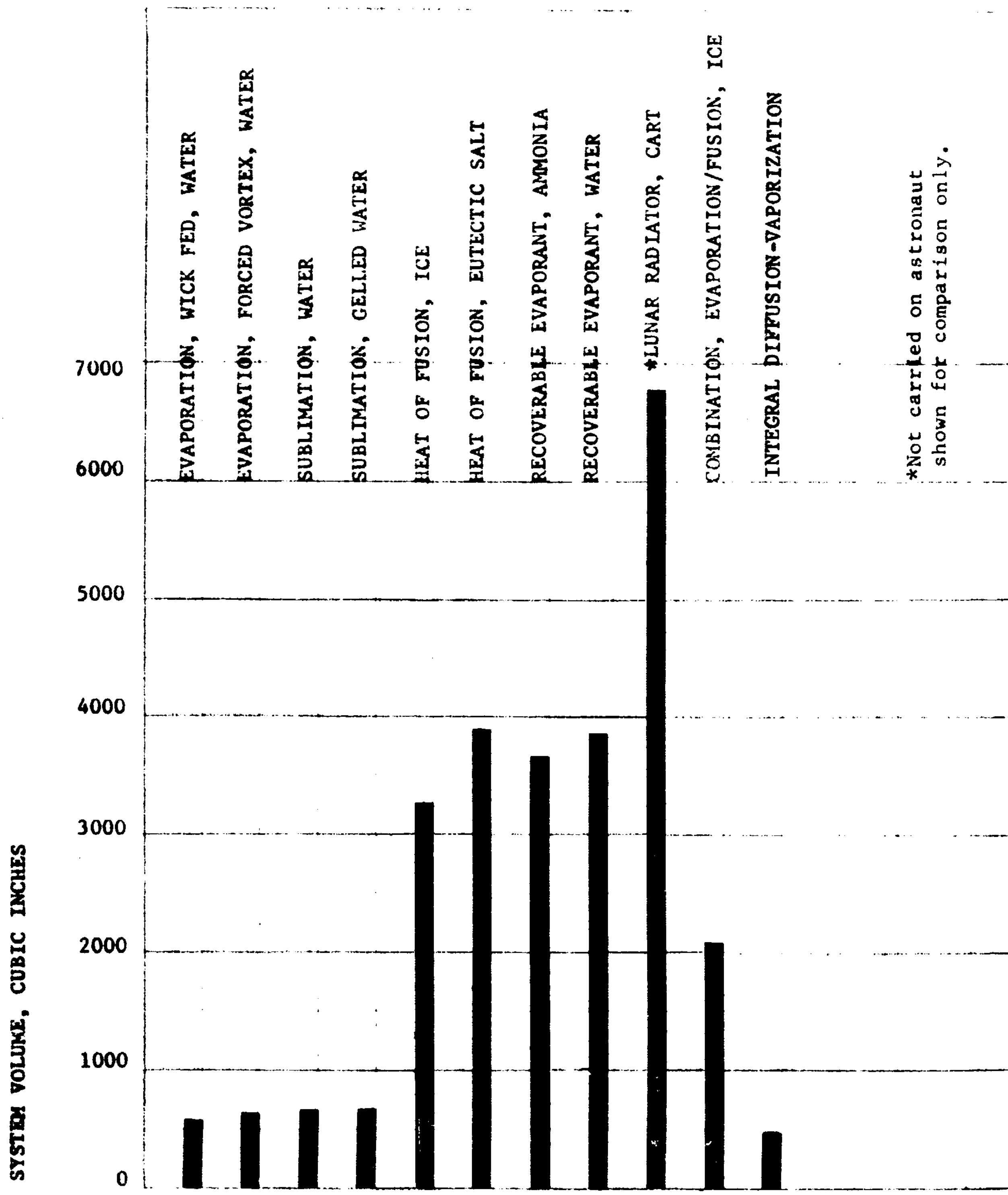


FIGURE 33 VOLUME ON ONE ASTRONAUT FOR ONE FOUR-HOUR MISSION

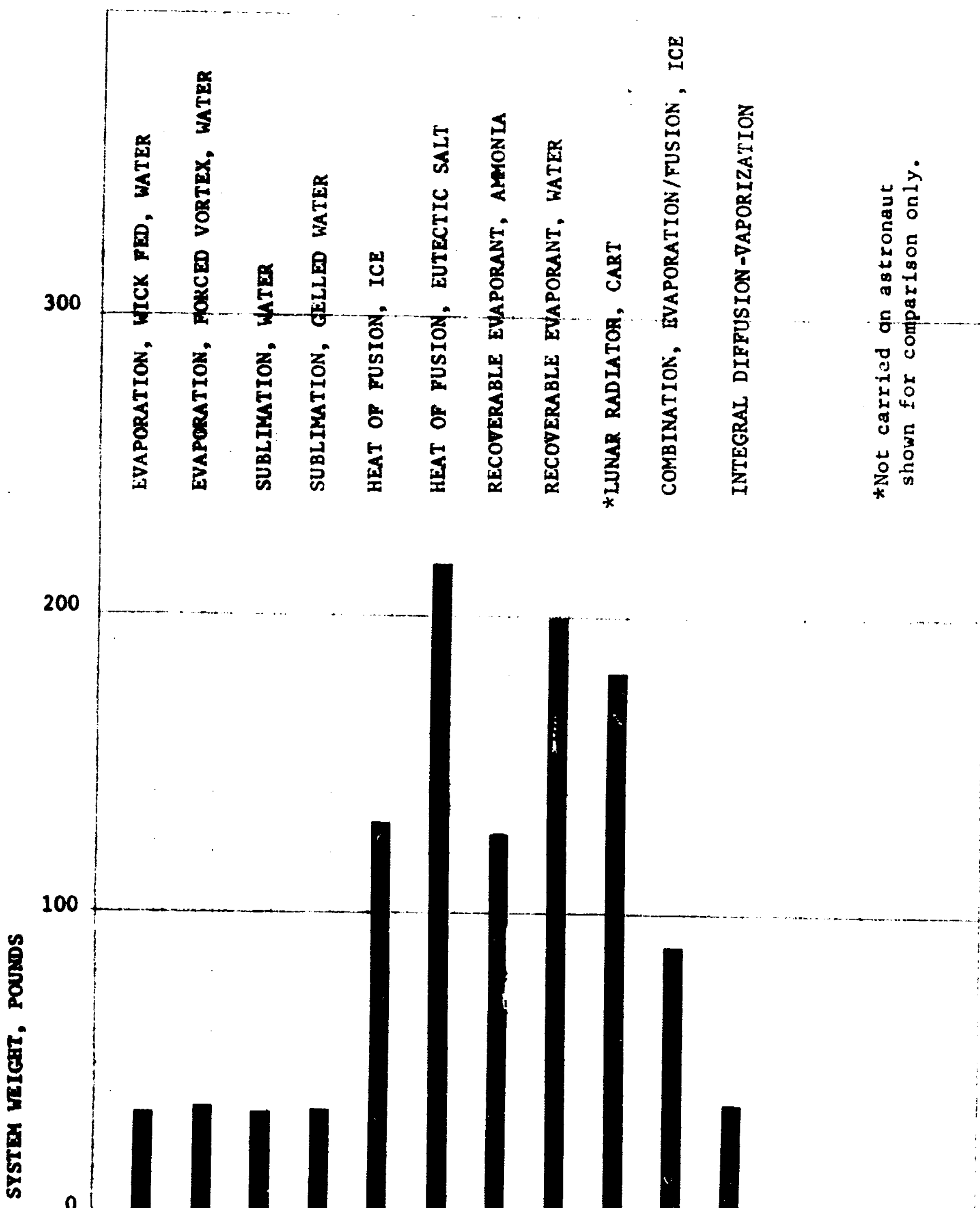


FIGURE 34 WEIGHT (AT ONE-G) FOR ONE ASTRONAUT FOR ONE FOUR-HOUR MISSION

TABLE 4

SPACECRAFT POWER AND HEAT TRANSFER REGENERATION REQUIREMENTS PER MAN HOUR OF EVA

SYSTEM	POWER REQUIRED - Watts	HEAT REQUIRED - BTU	HEAT TO BE REJECTED - BTU
Evaporation, wick fed	None	None	None
Evaporation, forced vortex	None	None	None
Sublimation, water	None	None	None
Sublimation, gelled water	None	None	None
Heat of fusion, ice	6	None	4170
Heat of fusion, eutectic salt	6	None	4240
Recoverable evaporant, ammonia	10	3550	2880
Recoverable evaporant, water	10	4235	3415
Combination, evap/fusion	6	None	2140
Lunar radiator, cart	None	None	None
Integral diffusion-vaporization	None	None	None
Heat pipes/suit radiator	None	None	None

The ammonia recoverable evaporant system during regeneration after four EVA hours of maximum heat rejection rate, requires heat, 14,200 Btu from a spacecraft "heating loop" for separating the ammonia vapor from the aqua ammonia solution while simultaneously rejecting ammonia vapor. The heat exchangers required for heat transfer in this system are assumed to be rated similar to a qualified liquid/gas heat exchanger at 6.15 cubic inches and 0.77 pounds per 1000 Btu and are added to the spacecraft support requirements. Since the regeneration of a candidate heat rejection system was assumed to be accomplished during periods when spacecraft heating and cooling loads are low, no equivalent volume and weight penalties were assessed for heat required or for heat rejected.

The recoverable evaporant, water, system has its water vapor absorbed by concentrated lithium bromide. Spacecraft power volume and weight penalties are added to this system for regeneration as later discussed. During regeneration, the vapor compressor is not utilized, but the LCG circulating pump is utilized. The pump operated on spacecraft power during regeneration. Heat exchangers required for heat transfer during regeneration (61.5 cubic inches and 0.77 pounds per 1000 Btu) are added to the spacecraft volume and weight support burden.

The heat of fusion systems, ice, and eutectic salt, also utilize heat exchangers to transfer heat to the spacecraft cooling loop during resolidification of the ice and the eutectic salt. The same volume factor per 1000 Btu, 61.5 cubic inches, and weight factor, 0.77 pounds, were used in determining the spacecraft support burden.

The heat quantities to be supplied by the spacecraft and rejected to the spacecraft versus total EVA mission hours for one man are shown in Figures 35 and 36, and the spacecraft power requirements versus total EVA mission hours is shown in Figure 37.

The candidate systems are enclosed in a protective insulated stowage container after recharging. These containers are sized to enclose each system except the lunar cart and have one inch wall thickness. For this study, each container is assumed to be made of flexible, protective insulation which is collapsible to leave no internal space void when the candidate system is removed. The insulation is assumed to have a density of 1.6 pounds per cubic foot.

The total launch volume and total launch weight versus total number of EVA mission hours for one man is plotted in Figures 38 and 39 respectively. For a given number of EVA man-hours the total

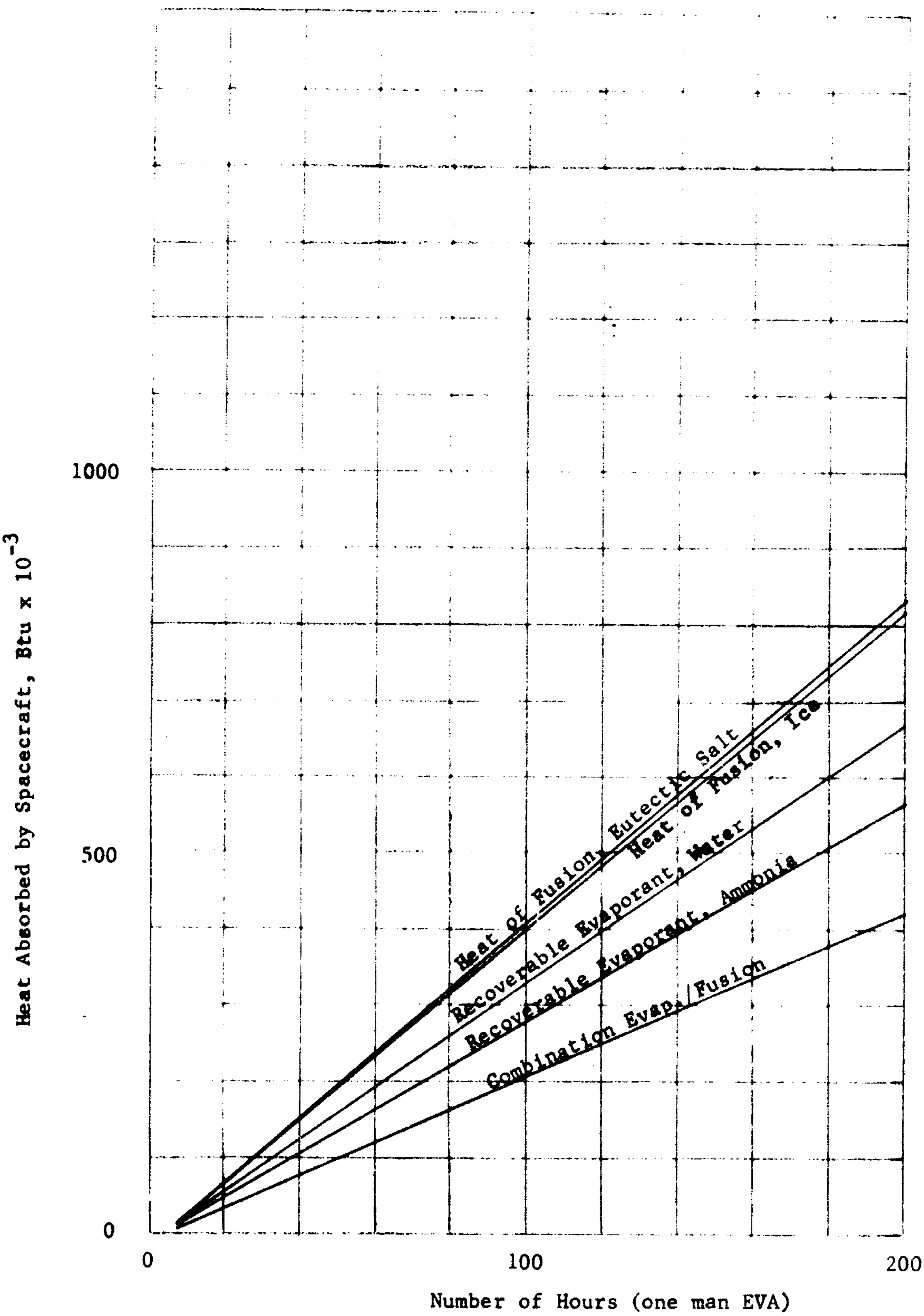


Figure 35 Heat Absorbed by the Spacecraft During Recharge

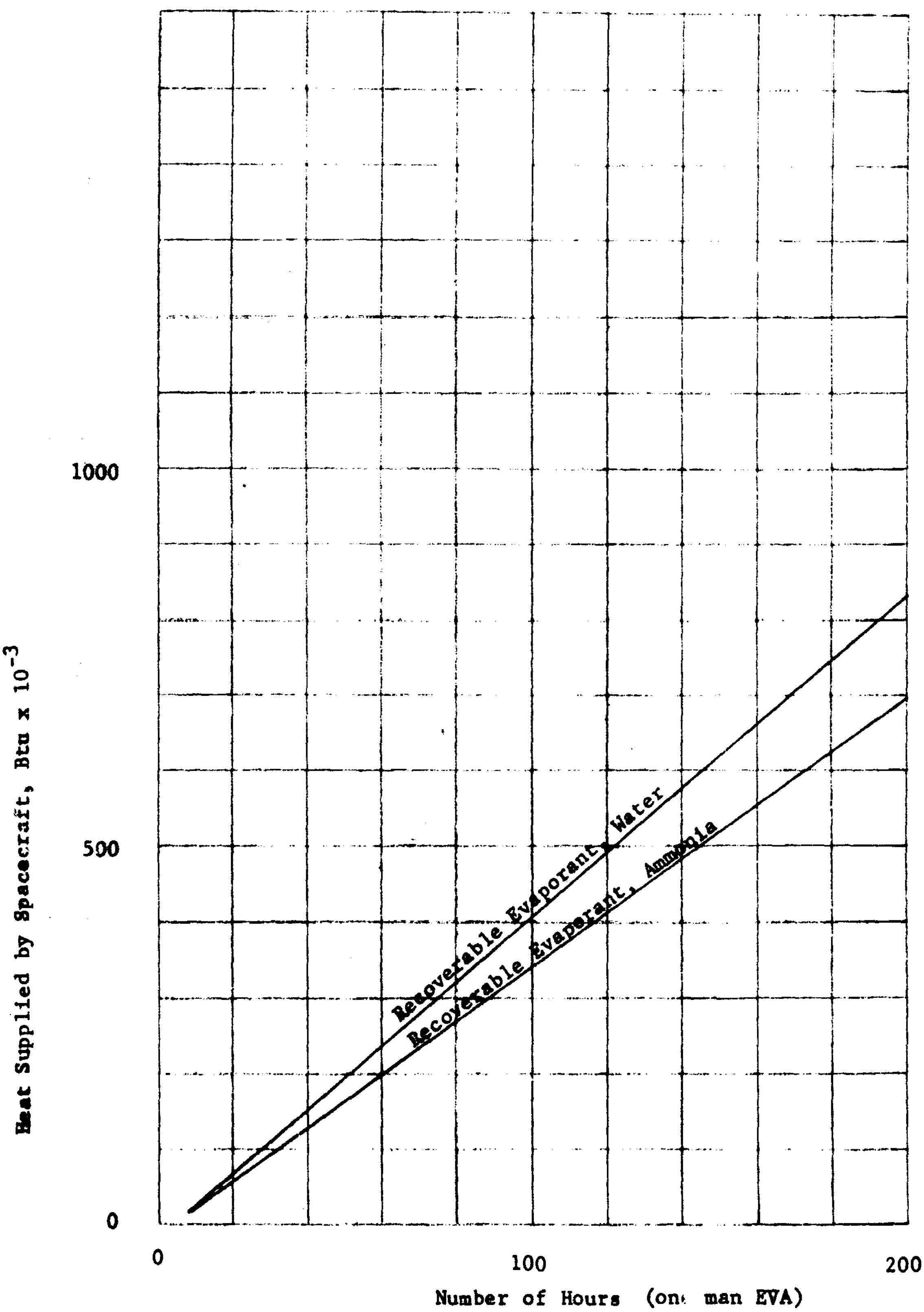


Figure 36. Heat Supplied by the Spacecraft During Recharge

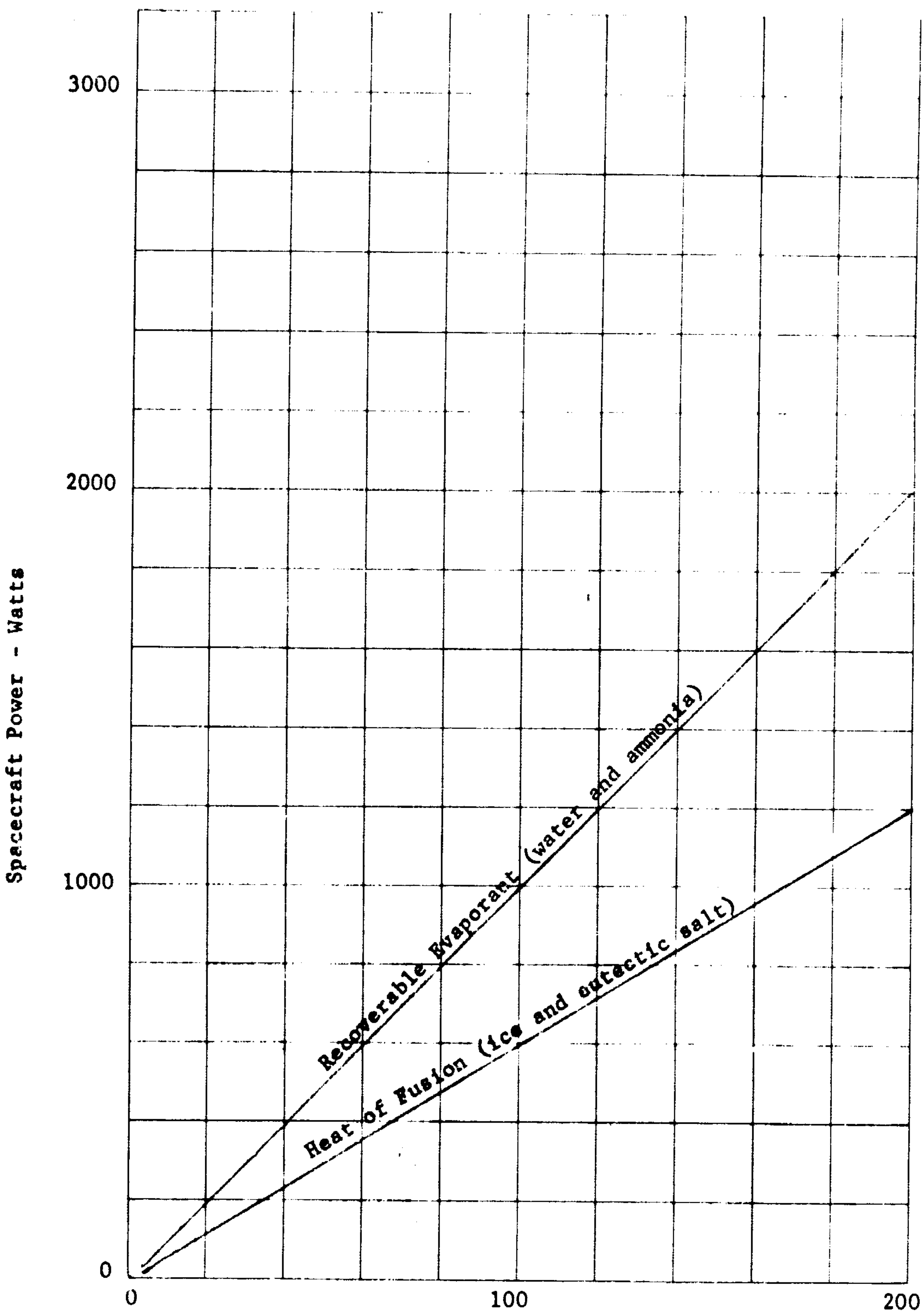


Figure 37 Spacecraft Power for Recharge

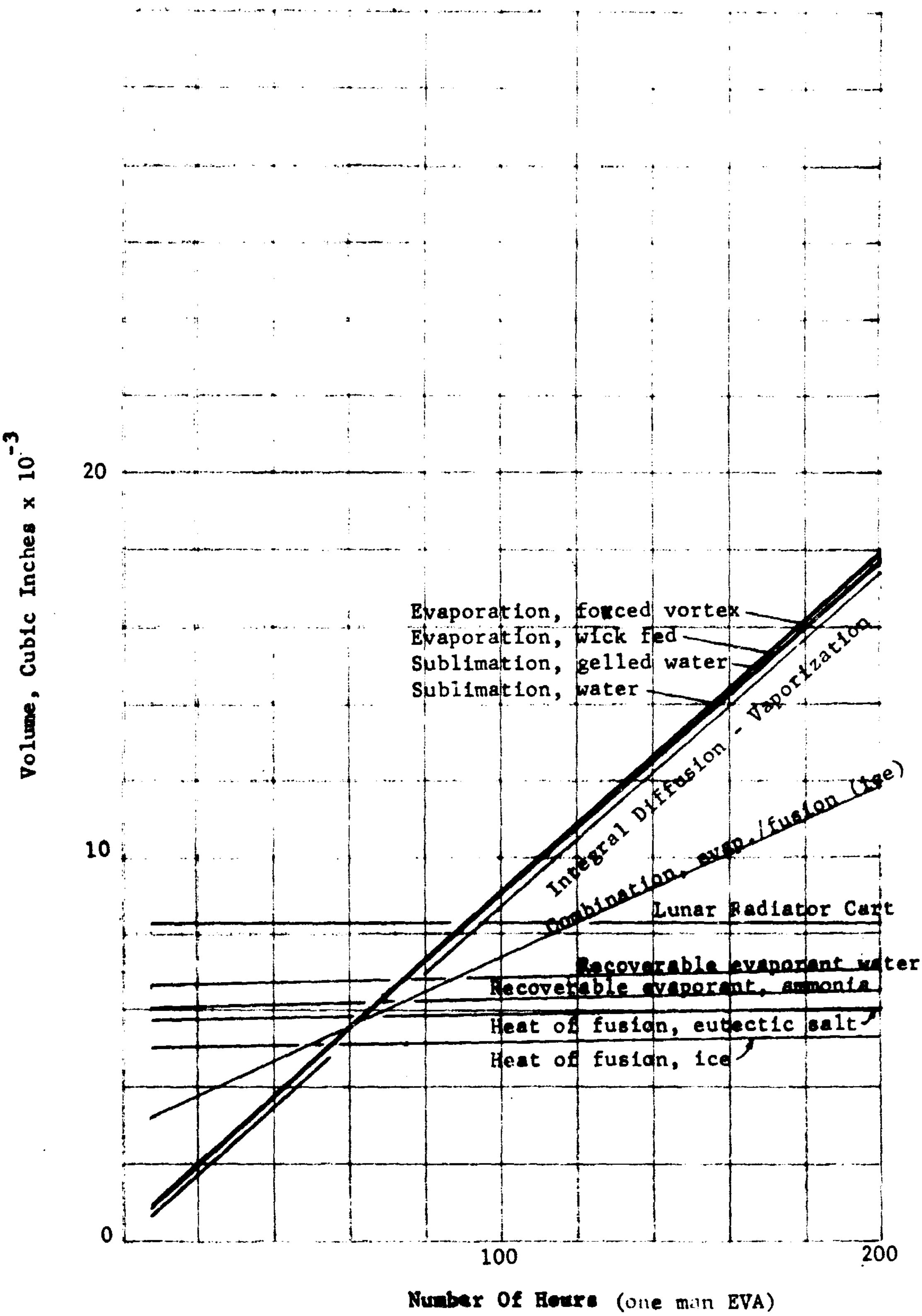


Figure 38 Total Launch Volume

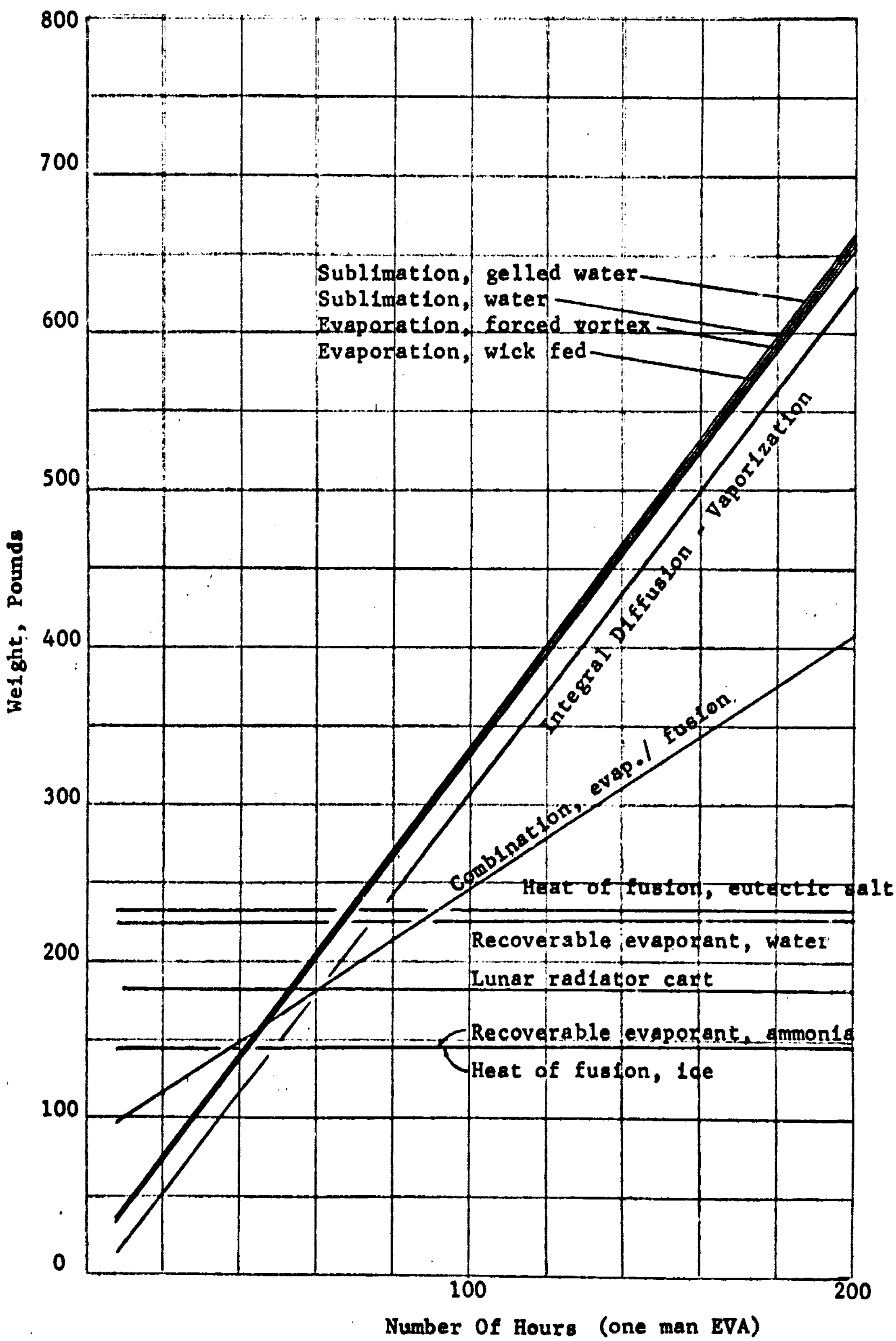


Figure 39 Total Launch Weight

launch weight or volume is the sum of the volume or weight on the man plus the volume or weight imposed by expendables, spacecraft power, spacecraft regeneration hardware, and stowage container.

The foregoing parametric analysis can be utilized by the designer in selection of the proper system for a specific mission. To aid designers in this endeavor, the following procedure is provided:

a. To allow for changes in the state-of-the-art since publication of this report, check the values used to make up the tables and curves and also check on guidelines and assumptions, and update as required.

b. Incorporate data on any new systems under consideration into the tables.

c. Revise the bar charts and curves as required to incorporate all updated information.

d. From the bar charts select candidate systems which provide the lowest volume and weight on the man.

e. From the curves, select the point on each curve for each candidate system which corresponds to the number of man-hours of EVA per man.

f. Multiply each value obtained in steps d and e by the number of men to be performing EVA.

g. Evaluate the resulting data based on mission design requirements and constraints.

IV. CONCLUSIONS

The selection of heat rejection systems for next generation PLSS is based primarily upon the parametric analysis. However, it must be tempered by a number of important considerations such as safety, reliability, inflight maintainability, crew time lines for servicing, potential failure modes, spacecraft interfaces, probable impact of the system on astronaut center of mass, and development risk.

In evaluating the parametric analysis, volume on the man and total launch weight were overriding considerations. Based on all of the aforementioned considerations and the study ground rules as outlined in the Discussion, the following are our conclusions.

1. As shown in Figure 39, the integral diffusion vaporization system, or water evaporators or sublimators used in conjunction with a liquid cooled garment, impose the minimum weight penalty when the total mission usage for one man will be forty hours or less. Above forty hours per man regenerable heat rejection methods are advantageous from a launch weight standpoint.

2. The only PLSS flight qualified hardware capable of recharge is the Apollo PLSS sublimator. While it is satisfactory, two areas of improvement should be considered: 1) decreasing the sensitivity of the capillary transport device, the porous plate, to contamination, and 2) development of hardware and/or procedures which reduce or eliminate the evaporant supply aeration problem.

3. The integral diffusion vaporization system provides the minimum volume on the man. While only slightly smaller than the evaporator or sublimator/LCG systems, the features of being integrated into the suit and requiring no pump power offer a significant potential advantage. Offsetting this advantage is the fact that considerable development effort is still required to improve mobility, to improve water feed control, and to optimize hardware design. This approach also suffers from the same problem inherent with any new system: the lack of any operational experience.

4. The problem of the sensitivity of capillary transport devices to contamination can be dealt with in one of two ways 1) improving capillary transport technology or 2) eliminating capillary transport devices altogether. With respect to the

former, much improvement appears possible. An approach for investigating this possibility is set forth in Recommendation No. 1 of this report. The second approach, the total elimination of capillary devices, also appears feasible - two examples are the forced vortex boiling evaporator and gelled water sublimator. Several versions of the forced vortex evaporator have been tested and their feasibility demonstrated. In the category of non-capillary evaporators/sublimators the forced vortex evaporator offers recharge advantages over the gelled water sublimator because the gelled water requires processing the gel for each recharge, whereas the forced vortex evaporator can be recharged with more conventional equipment.

5. Where total mission usage is in excess of 40 hours for one man, the macroencapsulated ice heat of fusion bed with the liquid cooled garment appears to be the best choice for earth orbital operations. This is supported from the standpoints of the smallest volume penalty on the man of any of the regenerable systems and total launch weight. It also has the advantage of being adaptable to a wide variety of packaging configurations including dispersion to integrated zones throughout the suit. Encapsulation of the ice/fusion medium in paralyne or a similar impermeable material and the inclusion of metallic thermal conducting wires within the medium will require development. Also, the water coolant loop presents a problem because a means of depressing its freezing point must be identified, and resulting additions must be both non-toxic to the man and non-reactive with the liquid cooled garment assembly. Two candidate additives are ethyl alcohol and propylene glycol.

6. A compromise approach which has merit for earth orbital missions is a combination evaporator and fusion bed heat sink used with a liquid cooled garment. For this study a water evaporator was placed in series with an ice heat of fusion bed with each designed to take fifty percent of the heat load. This approach results in a reduction of one-third in the volume on the man; however, it results in an increase in mission weight and in the length and complexity of the recharge procedure over the totally regenerable heat of fusion bed/LCG system.

7. For lunar exploration, the lunar radiator cart offers many advantages. The radiator requires no regeneration and is remote from the astronaut. One obstacle to overcome with this approach is the design of an umbilical assembly which will have an acceptable heat leak and which will be light

weight and flexible enough to permit the astronaut good maneuverability.

8. Each of the candidate approaches discussed can be utilized for either earth orbital or lunar exploration missions with the exception of the lunar radiator cart. A deployable radiator might be utilized in earth orbit, but this appears to be undesirable because problems associated with radiator position are a serious impediment to astronaut maneuverability and appears to be difficult to overcome.

9. Both recoverable evaporant concepts should be dropped from further consideration because they do not appear to offer any significant advantages to overcome the substantial disadvantages of 1) development risk, and 2) regeneration procedure complexity.

10. The eutectic salt heat of fusion bed system is larger and heavier than the ice heat of fusion system and is worthy of further consideration only if a suitable coolant can not be identified for the ice heat of fusion system.

11. Refrigeration cycles cannot be utilized for thermal transport within the constraints of earth orbital or lunar surface mission requirements. Advantages to be obtained by utilizing a higher temperature heat sink are more than offset by the much larger heat load introduced by the refrigeration cycle. For a refrigeration cycle to be practical it would have to reject heat to a high temperature heat sink which is capable of receiving the normal heat load in addition to the refrigeration heat load (which would be three to four times the normal heat load) more efficiently than a heat sink operating at a compatible temperature and rejecting only the required heat load. The practicality of such a prospect appears highly unlikely.

12. Methods of removing heat from the astronaut need further consideration. The state of the art approach of utilizing a liquid cooled garment is not totally desirable because it fights man's thermoregulatory processes. To reject all heat at maximum sustained metabolic rates requires a coolant temperature of 41 +4,-1° next to the skin. As a result the surface tissue reacts to the cold by shutting off effusion of blood which retards heat transfer. In this condition the heat transfer from the tissue is greatly reduced from the heat transfer from the same tissue in the mode of maximum conduction (maximum effusion of blood into surface tissue).

The astronaut feels cold and uncomfortable; yet in extreme conditions it is theoretically quite possible that he may be actually storing heat. Such a condition is of course undesirable but to date no other approaches offer promise of much improvement. More effort is desirable in this area which is beyond the scope of this study.

V. RECOMMENDATIONS

1. For missions requiring less than forty hours of EVA per man, an improved state-of-the-art evaporator or sublimator used with a liquid cooled garment (LCG) is the best approach for the next generation of PLSS (for both earth orbital and lunar surface operations). Improvement in currently developed evaporators and sublimators should be undertaken to reduce or eliminate capillary-transport-device performance degradation and to eliminate aeration of the evaporant water. In the first area an investigation covering (1) methods for preclusion of contaminants, (2) determination of optimum water pH (7 is not necessarily desirable), (3) procedures for control of water pH, (4) modification of contaminants to weaken propensity for bonding, (5) methods for increasing solubility of contaminants in water (such as by the addition of a phospholipid), and (6) modification of the monomolecular capillary surface layer should be undertaken. In the second area a task should be initiated to develop hardware and/or procedures which eliminate the evaporant supply aeration problem. Ultimate selection of either an evaporator or sublimator should be based upon the results of an improvement program which includes the aforementioned tasks.

2. The integral diffusion vaporization system should be considered as a potential successor to evaporator/LCG and sublimator/LCG systems. Therefore, continued development effort in this area is recommended.

3. For missions in excess of forty hours of EVA per man, a macro-encapsulated water/ice heat of fusion bed should be developed. This unit could then be used to provide total heat sink capability or can be used to share the heat load, in conjunction with an evaporator or sublimator. Selection of a total heat of fusion bed sink or a hybrid heat-of-fusion bed/evaporator (or sublimator) sink is dependent upon mission requirements. The tradeoff is between volume on the man and total liftoff weight.

4. Though the lunar radiator cart violates the constraint of this study requiring the heat rejection system be mounted on the man, the attractiveness of such a no-expendable approach makes it worthy of further consideration. Therefore, for missions requiring in excess of fifty hours total of lunar surface EVA per man, the development of a lunar radiator cart, used in conjunction with an LCG, is recommended.

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