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PRELIMINARY STUDY OF ADVANCED LIFE-SUPPORT TECHNOLOGY FOR A MARS SURFACE MODULE

by R. E. Mitchell and M. Burns

Prepared by
IIT RESEARCH INSTITUTE
Chicago, Ill.

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PREFACE

This is the Final Report on IIT Research Institute (IITRI) Project No. K6101, originally entitled "Martian Shelter Technology." A new title that more accurately describes the material presented in the report has been selected. The research was performed for the National Aeronautics and Space Administration, Office of Advanced Research and Technology, under Contract No. NASr-65(15), by the Mechanical Engineering Division of IITRI. This report covers the work performed from March 28, 1966, to June 30, 1967.

The IITRI personnel who contributed to the program included R. Mitchell (Project Engineer), M. Burns, M. Lerner, S. Guzder, F. Iwatsuki, C. Cokeing, T. Owens, W. Langdon, and D. Klopp.

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PRELIMINARY STUDY OF ADVANCED LIFE-SUPPORT TECHNOLOGY FOR A MARS SURFACE MODULE

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IIT Research Institute

1. SUMMARY

This study represents an initial step in the direction of subsystem definition for extended manned missions on Mars, and is centered primarily around a review of the current literature and state-of-the-art techniques.

There is much information available on extended mission life-support technology. This information, however, is so strongly centered about zero-g and deep space applications that very little correlation can be made for an extension to the surface conditions on Mars. The fact that Mars has a gravity is one of the major differences. Some studies, considering a gravity field, have been performed. However, these studies were directed towards Lunar surface applications involving missions of very short duration, or concerned themselves primarily with gross power considerations rather than detailed subsystem technology.

Several studies for the manned Mars effort have been undertaken by Aeronutronics (MEM), North American Aviation (MMM), Douglas (Manned Mars Exploration), and Hamilton Standard (Manned Mars Subsystem Studies). These studies deal with Mars orbital and Mars entry subsystem analysis. Consequently, the life-support subsystems selected are not necessarily the best subsystems for the surface application.

A mission profile of four men for six months has been analyzed in this study. From this baseline assumption, we were able to project a Mars Surface Module (MSM) Configuration and assess on a preliminary basis those subsystems in life-support, power, and environmental control most candidate for the proposed mission.

2. INTRODUCTION

The problem of selecting optimum life-support subsystems for a given space mission has gained considerable attention in recent years. Mars appears to be the next natural step after the Lunar challenge has been met. Current Mars mission studies show that the technology will be available for landing a manned module on Mars by the mid-1980 time period. If the Apollo mission sequence is as successful as Project Gemini, this date could be shortened by as much as three to five years.

It is, therefore, desirable that we begin to carefully examine the alternative systems necessary to support man for extended periods on the surface of Mars. To achieve the necessary advances in the required technical disciplines, prior to the first manned Mars excursion, we must direct a major research effort toward those subsystem candidates most appropriate for the proposed mission. It is of interest to note that many of the basic subsystems necessary for a successful mission are presently under development. However, we must soon make a major commitment toward selecting and developing those subsystems from the various alternatives which have the best potential for realizing the Mars mission objectives.

In the Mars surface application, we are presented with a gravity field, an unbreathable atmosphere, a 24-hour day, very severe temperature variations at the poles and more moderate temperature fluctuations at the equator. These conditions limit the choice of subsystems for life support and do not permit a direct extension of the present studies involving other space applications. A strong influencing factor is that heat transfer external to the module is no longer purely by radiation to a constant sink temperature. The transfer now is also by convection with a varying atmospheric temperature.

It is for this reason that subsystems compatible with applications on the surface of Mars should be independently studied and evaluated. In making the final decision on manned landing systems, the effects of the Mars environment must be very accurately known. Certain systems that are operative in space could possibly be nonfunctional on the surface.

Temperature profiles and estimates of atmospheric constituents are presently available in various forms. As the space program continues, the available data will become much more accurate, and final modifications in calculations for the actual mission can be made. This will allow for a more precise selection of primary and back-up subsystems for the eventual missions.

3. MARS ENVIRONMENT STUDY

The most important characteristics of the Martian environment which affect the module design and the life-support system configurations are the temperature profile, atmospheric composition, length of Martian day, wind velocity, and soil conductivity. Presently, the order of magnitude of the Martian atmosphere variables are reasonably well known. Actually, this data has been available in rough form for at least 40 years. Although the measuring techniques have become more sophisticated and the data acquired more precise, no profound changes in the interpretation of the environmental constituents have occurred. The recent observations by the Mariner IV have certainly raised the level of confidence in previous predictions, and their particular benefit has been in establishing clear numerical values which could be used more directly by systems analysts and designers.

It is generally accepted that there is an abundance of carbon dioxide in the Martian atmosphere. However, there is much disagreement as to the exact value of the surface pressure; estimates range from three to 85 mb. Neither the module design nor the life-support systems selections are significantly affected throughout this small range of argument. The attempts to more accurately define the pressure are of interest to others primarily for the design of an aerodynamic reentry vehicle.

The temperature profile and atmospheric pressure affect the radiator design and the module wall design, and the latter affects the heat transfer from which the module Environmental Control System (ECS) is selected. The ECS includes the module primary air heat exchanger, and thus is part of the total Life-Support System.

In this study, we have used the diurnal variation of surface temperature substantiated experimentally by Sinton and Strong (ref. 1). Figure 1 was obtained for a range of soil conductivities from 3×10^{-5} to 1×10^{-4} cal/cm-sec-°K. Figure 2 shows the annual variation in temperature for varying depths below the Mars surface. It can be seen that the diurnal and annual variations dampen out as the depth below the surface increases. Annual variations in minimum diurnal temperature for various latitudes are shown in Figure 3. This graph shows quite readily why the equatorial areas are the most desirous for a manned Mars mission. The low seasonal temperature variations in this area considerably reduce the design problems associated with extended missions. This location also reduces the concern about exposure to very extreme temperatures, as would be the case for either the higher or lower latitudes.

The data on atmospheric pressure given in Figure 4 was substantiated by data from Mariner IV (ref. 2), and also by experiments performed by Kaplan, Münch and Spinrad (ref. 3) as indicated.

Tables I and II summarize the most recent thinking concerning the Planet Mars.

The Mariner IV probe landed November 1964 after traveling a total of 326,000,000 miles for 228 days. The magnetometer aboard failed to detect evidence of a Martian magnetic field.

The following items represent current opinions on other aspects of the Martian environment.

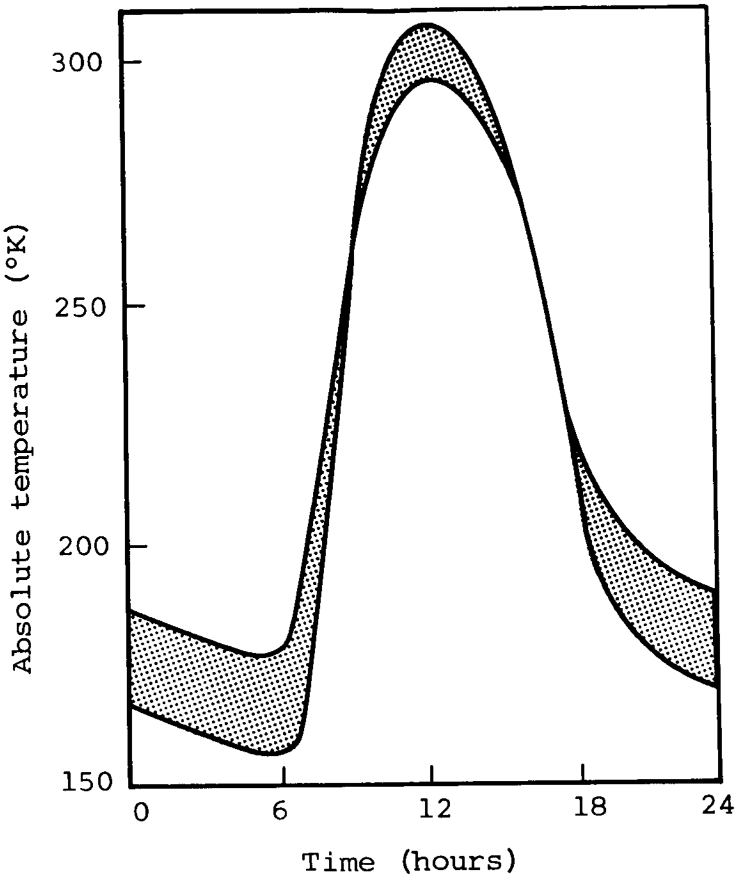


Figure 1 Diurnal Variation of the Surface Temperature of Mars
(From ref. 1)

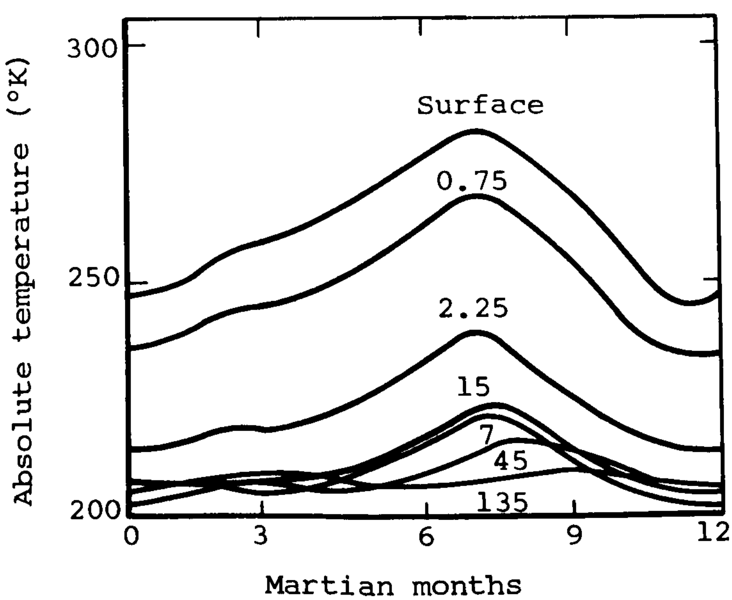


Figure 2 Annual Variation of Mars Average Soil Temperature
(From ref. 1)

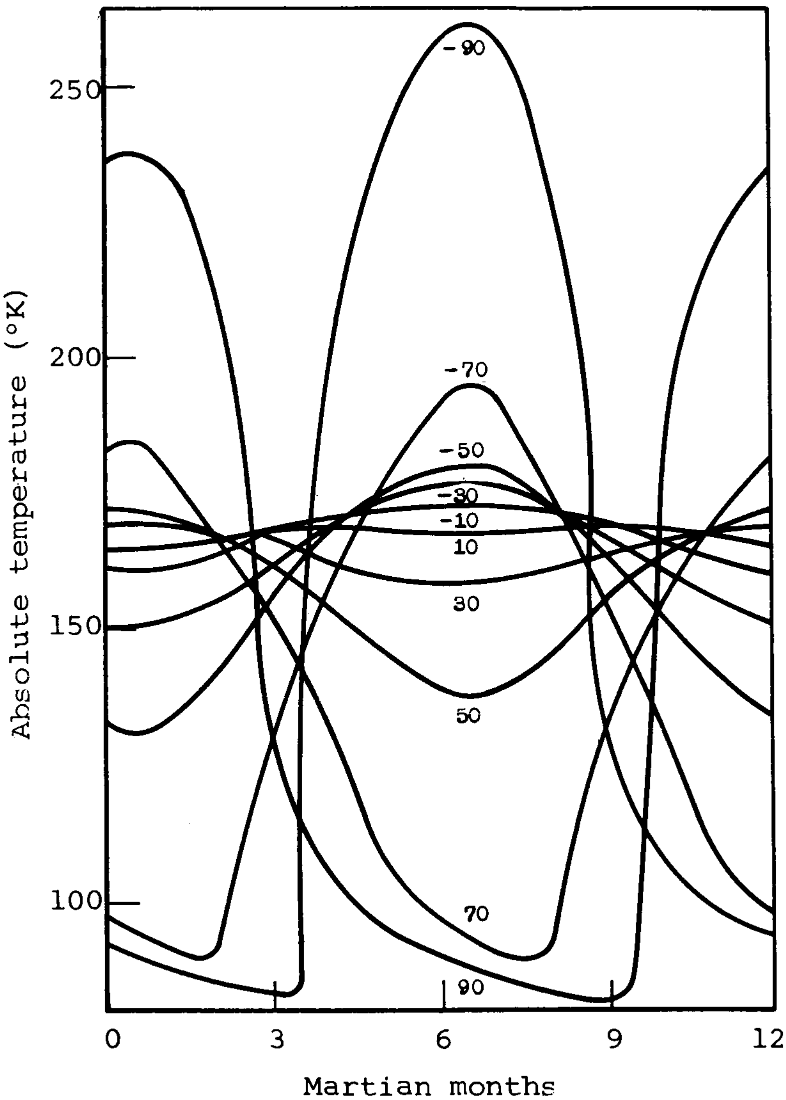


Figure 3 Annual Variation of Minimum Diurnal Temperature for Various Martian Latitudes

(From ref. 1)

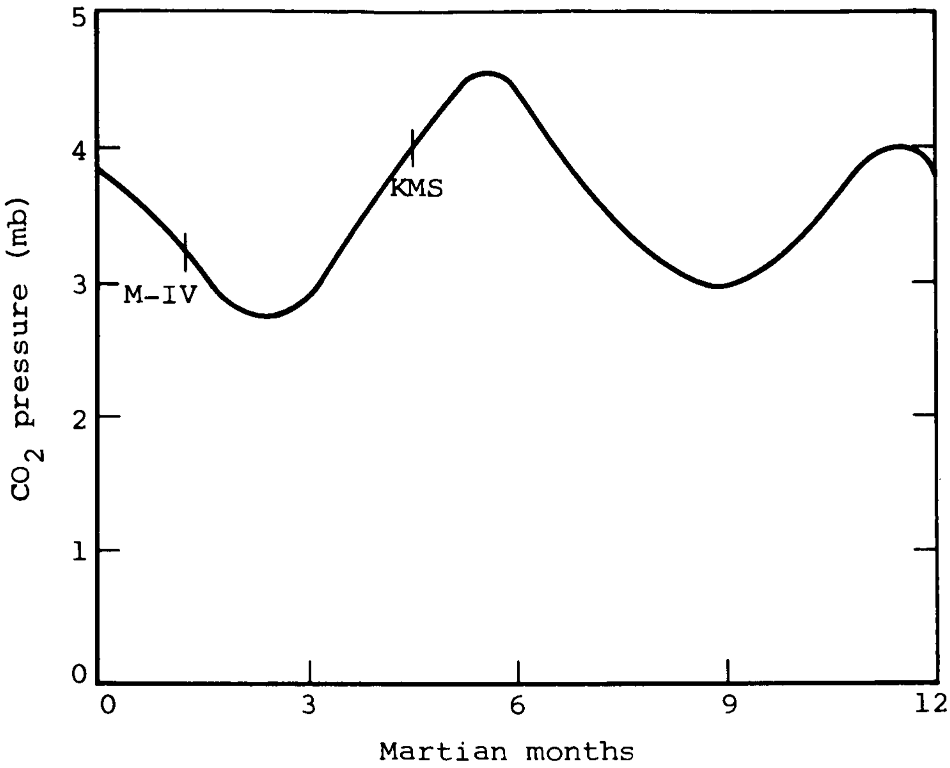


Figure 4 Variation in Carbon Dioxide Partial Pressure, Resulting from Seasonal Condensation and Evaporation

(From ref. 1)

TABLE I

MARS PLANETARY DATA

Measurement	Values
Composition	Mostly CO_2, small amount of $N_2 + A_2$
Temperature at Equator . . .	Day, max. 300°K Night, max. 180°K
Mass Ratio - Mars/Earth . .	0.107
Gravity	.39 of earth
Wind Velocity	37 mph
Equatorial Diameter	4.243 miles
Polar Diameter	4,198 miles
Escape Velocity	3.14 miles/sec
Martian Mean Solar Day . . .	24 hr, 39 min, 35 sec
Magnetic Moment	<0.001 of earth's
Martian Year	687 days
Solar Constant	Approximately 530 W/m^2 (Equator tilted 25° to orbital plane)
Soil Emissivity	.95 - 1.0
Surface Soil Conductivity. .	2.5×10^{-4} W cm^{-1} $°K^{-1}$
Surface Soil Specific Heat .	3.3 Joules g^{-1} $°K^{-1}$
Surface Soil Density	1.6 g-cm

TABLE II

MARS ORBITAL DATA

Measurement	Values
Eccentricity	0.093368
Inclination	1,849,910
Distance from Sun	
Mean	141,500,000 miles
Perihelion	128,300,000 miles
Aphelion	154,800,000 miles
Distance from Earth	
Mean Opposition	48,695,000 miles
Minimum Distance	34,670,000 miles
Maximum Distance	247,900,000 miles
Orbital Velocity	
Mean	15.0 miles/sec
Perihelion	16.4 miles/sec
Aphelion	13.6 miles/sec
Next Close Opposition	August 6, 1971
Again	September 28, 1988

- It is believed that the ground is hard with something similar to perma frost, which might be used as a heat sink. There is no liquid water available, and only a small amount of H_2O as vapor.
- The experts still have a difference of opinion as to whether some low forms of life exist on the surface.
- Dollfus found that when certain algae, small lichens, and minute mushrooms are sprinkled on pulverized limonite, they show a polarization curve similar to that of the Martian dark spots. Salisbury does not believe that plants grow on Mars.
- As to soil composition, Kuiper concluded the deserts of Mars consist of igneous rock similar to felsetic rhyolite. Dollfus concluded that it was powdered ferric oxides (limonite $HFeO_2$).
- Craters of Mars are believed to rise a few hundred feet above the surrounding surface and reach depths of several thousand feet below the crater rims.
- Mars has clouds which could interfere with solar panels, and Martian atmosphere is bombarded continuously by full flux of cosmic rays and full flurries of solar flare particles. The radioactive products of the disintegration present differ from those on earth and probably are somewhat more dense relative to the density of the atmosphere. Recent evidence indicates that ultraviolet rays penetrate the Martian atmosphere; however, since the total solar flux is low, this should not cause too great a problem.
- Radiation hazards on Mars should be in the same proportion as solar constants near earth (approximately) which is 520/1400, or 37 percent. Therefore, the radiation flux will be less than 37 percent due to attenuation of Martian atmosphere. This would be equivalent to a maximum flux of cosmic rays of 3 rads per year. Fluxes would be higher during solar flare activity.

4. PRELIMINARY TRADE-OFF CONSIDERATIONS

A. Life-Support System Summary

The life-support system must provide a shirtsleeve environment which is compatible with long-term habitation. The six systems needed in this effort are shown in Figure 5.

The items causing greatest concern are waste water management and regenerative atmospheric carbon dioxide content control.

1. Water management. - Through utilization of an efficient water reclamation system, maximum weight savings can be realized.

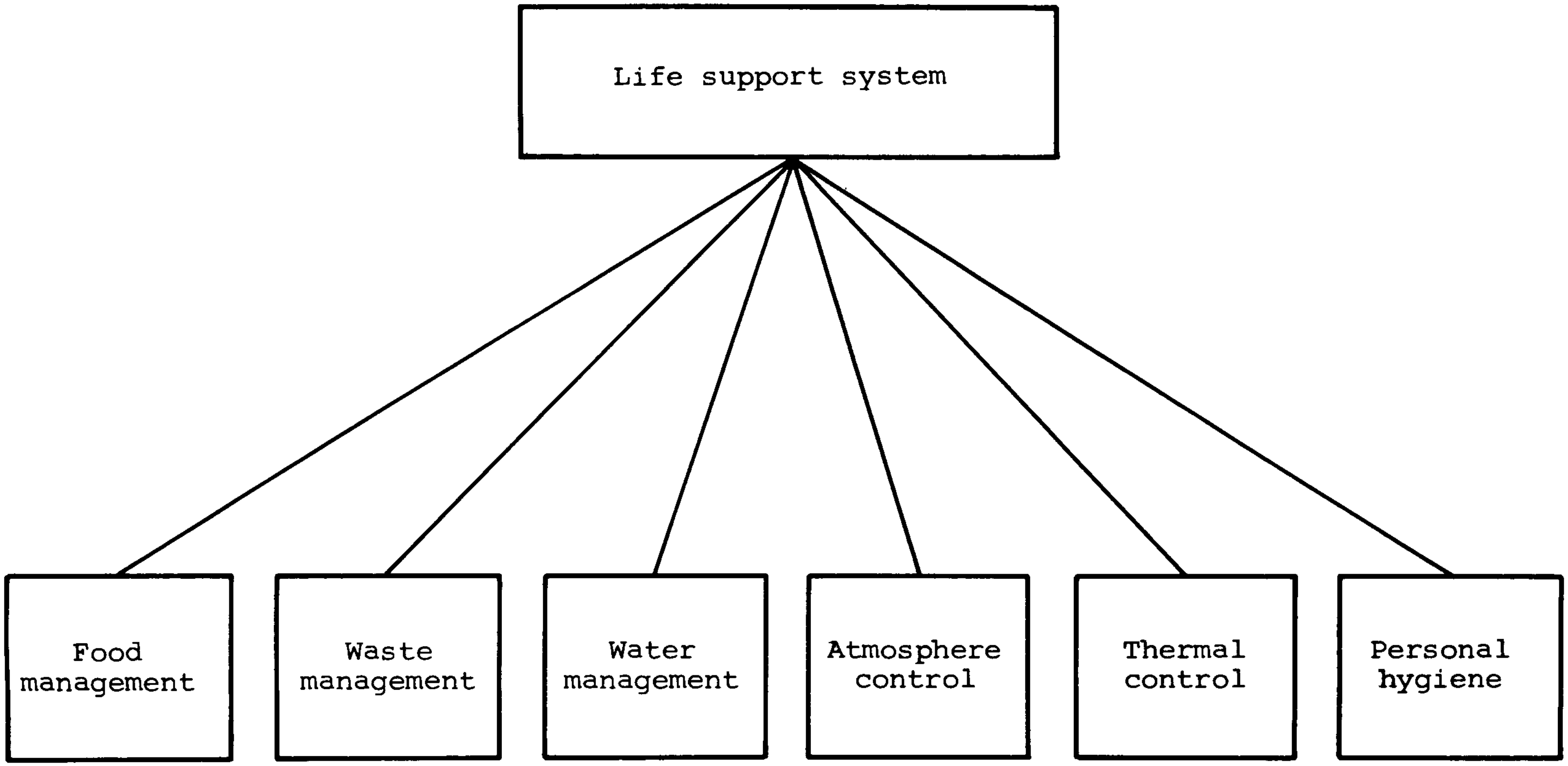


Figure 5 Life Support System Breakdown

Under an assumed mission condition of 720 man-days (four men, six months), a nonregenerative system would require about eight tons of water. This far exceeds the weight penalties imposed by all the other systems combined. Therefore, either complete or possibly partial water regeneration is a must to any manned Mars mission.

The most easily closed loop is the (respiration-perspiration) water cycle. This involves about 3.0 lb/man-day and is perhaps the purest water to be handled. Next would be the water used for washing and personal hygiene; this involves about 15.0 lb/man-day. For extended missions, it is almost certain that a urine-water reclamation system will have to be employed. Any weight savings to be realized by reclamation of fecal water is far offset at the present by the difficulties involved in handling this material. It is not ancicipated at this point in the study that fecal water recycling will be available or necessary by mission time.

Waste water and urine will probably be made available for reuse by the processes indicated in Figures 6 and 7. The processes are similar except for pyrolysis of the urine vapor to remove volatile constituents (ref. 4).

The man-day water requirement baseline to be used in the report is shown in Figure 8; water requirement quantities have been expanded so as to be more realistic for extended mission conditions (refs. 5 and 6).

2. <u>Carbon dioxide management</u>. - The second largest weight savings is to be found in the area of a CO_2 removal and oxygen regeneration system. This will not only reduce the amount of oxygen that has to be stored, but it will also eliminate the large amounts of CO_2 absorption material that would have to be stored for an open loop CO_2 dump cycle. The most promising processes for CO_2 reduction are the Bosch and the Sabatier cycles shown in Figures 9 and 10; both, however, require a very pure CO_2 feed (ref. 7).

In the Bosch cycle, CO_2 is combined with hydrogen over a hot iron catalyst to produce water and solid carbon. The water is then electrolyzed, yielding hydrogen and oxygen. The Bosch reactor requires 921 Btu/lb of CO_2 reacted at a temperature of about 1100°F. At this temperature level, conversion efficiency is about 30 percent per pass, so that recycling is necessary. The major disadvantage of this cycle is the difficulty in removing the carbon build-up from the catalyst (ref. 8).

In the Sabatier cycle, CO_2 is combined with hydrogen to produce water and methane. The Sabatier reaction is exothermic and takes place at only 550°F. This reaction temperature is much lower than that of the Bosch reactor and results in a much lower power requirement for start-up. The problem of carbon

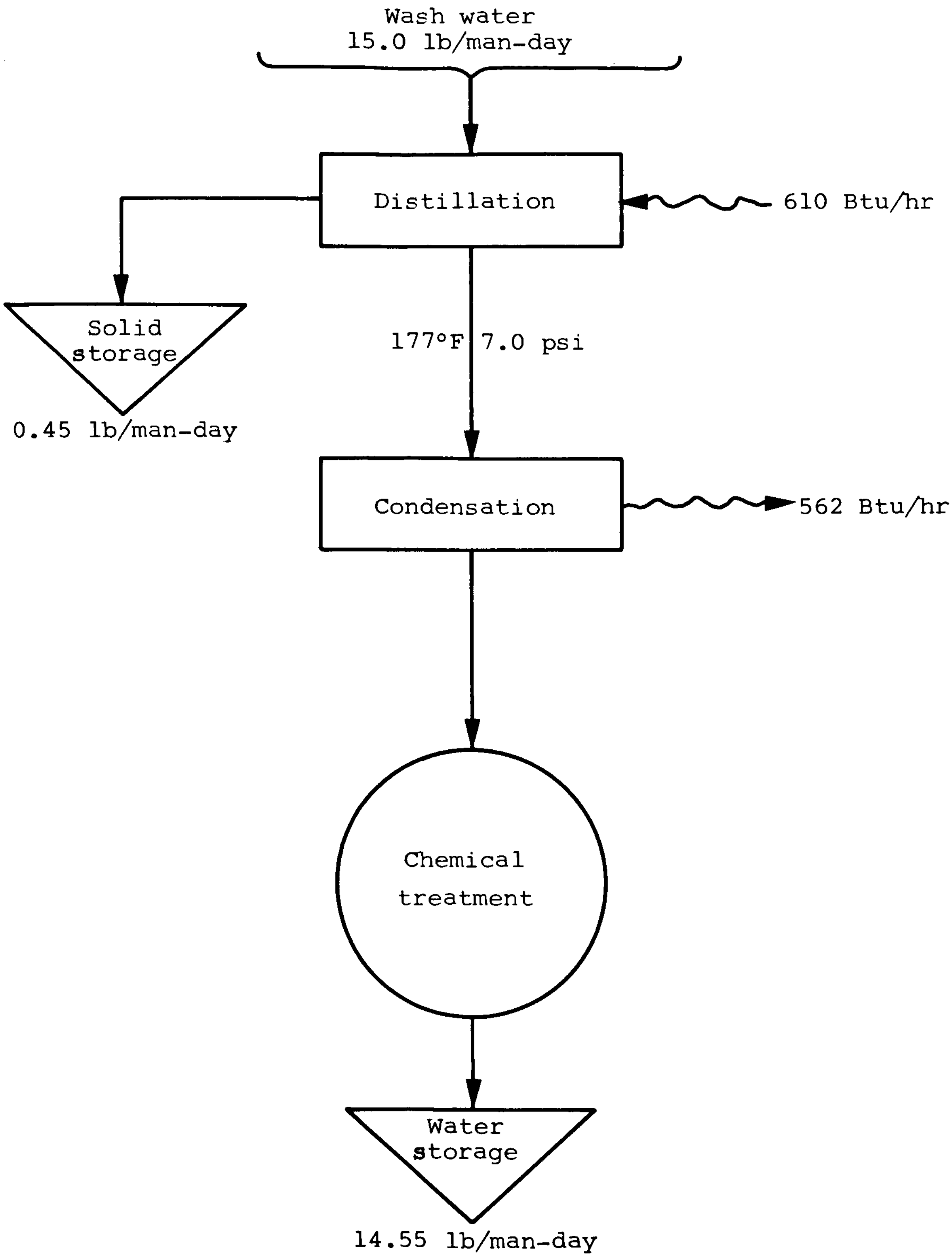


Figure 6 Wash Water Distillation Regeneration Cycle

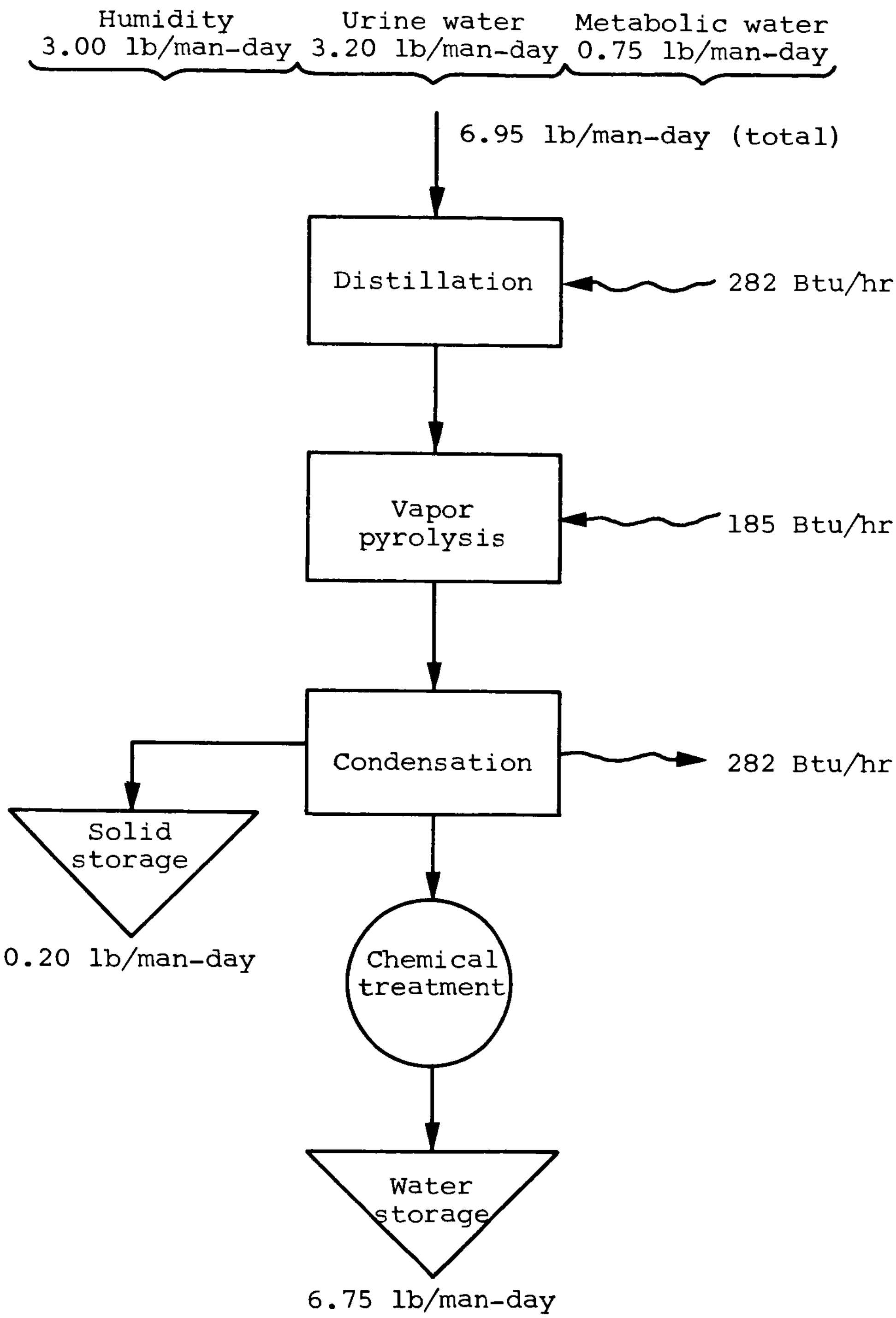


Figure 7 Distillation/Pyrolysis Water Regeneration Cycle

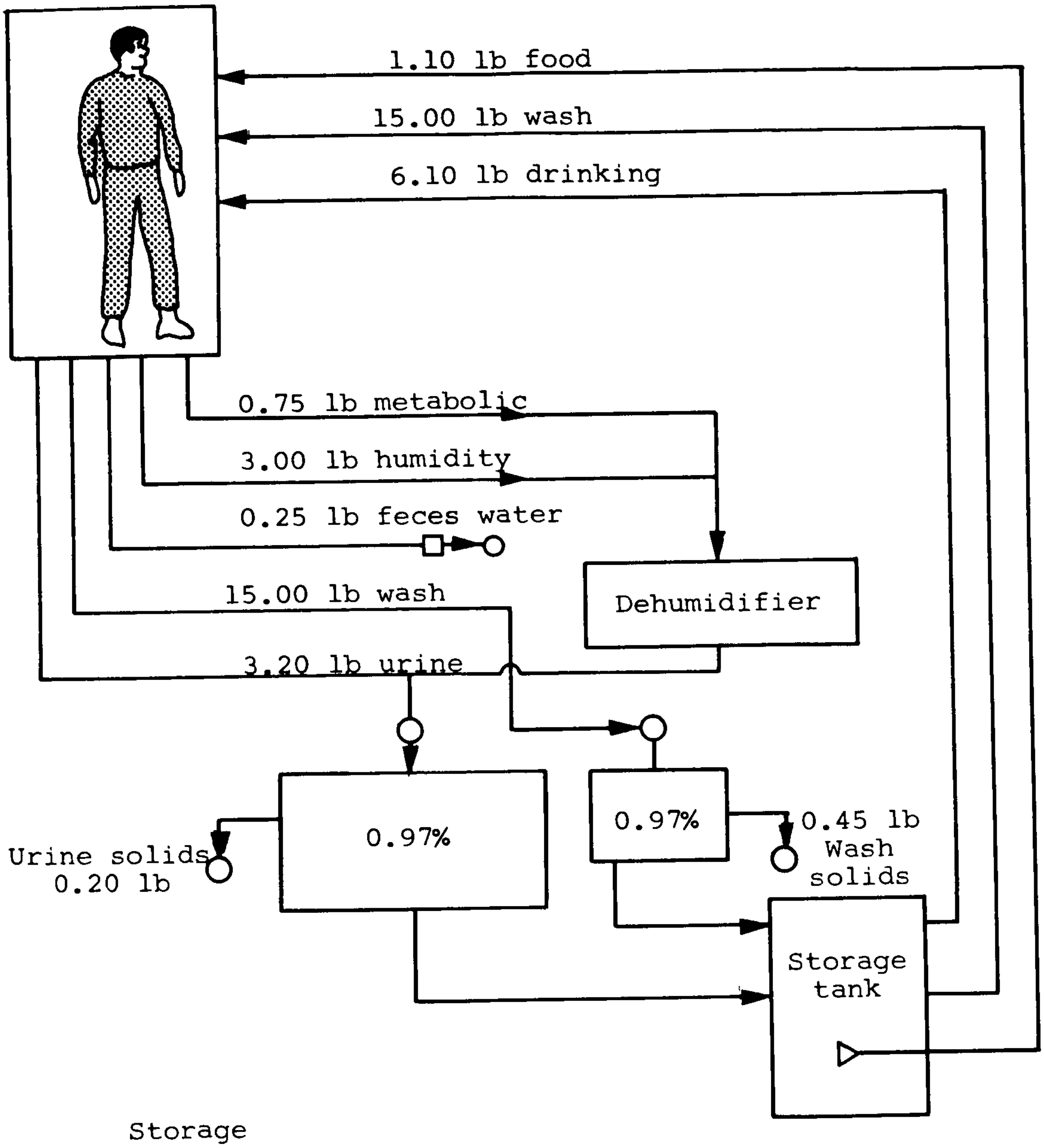


Figure 8 Man-Day Water Balance

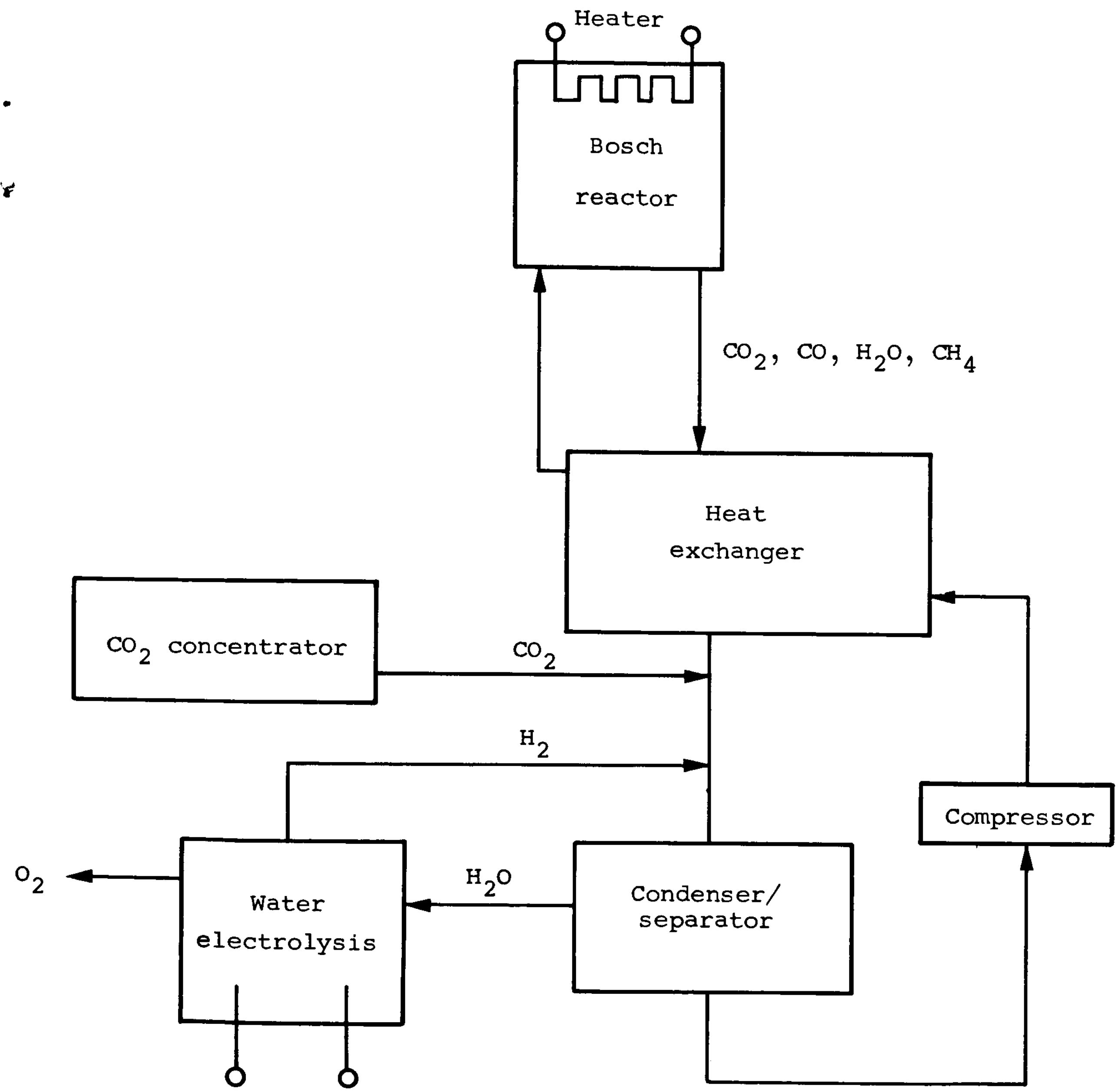


Figure 9 Bosch System for Carbon Dioxide Reduction

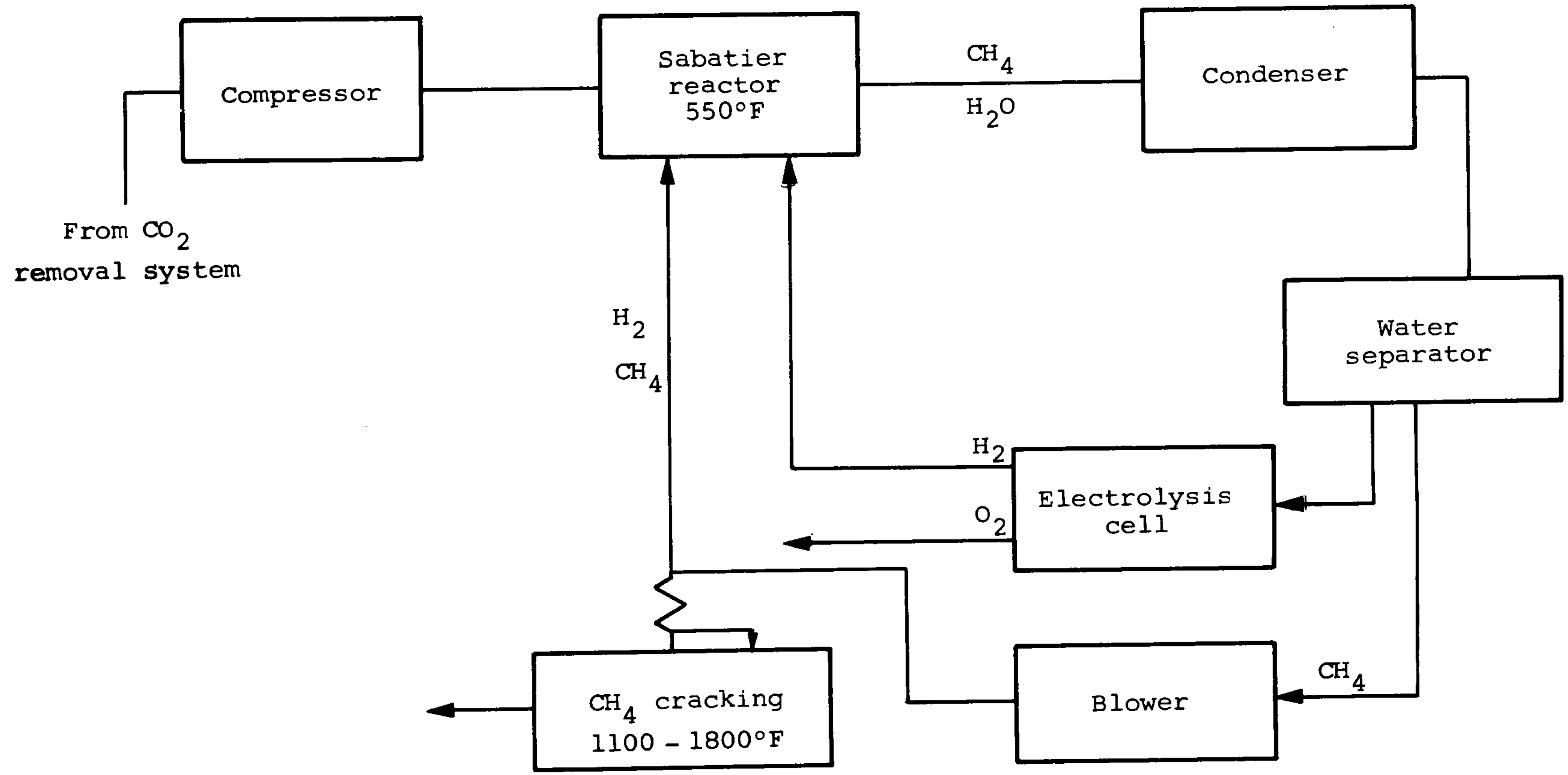


Figure 10   Sabatier System for Carbon Dioxide Reduction

removal is eliminated, since there is no carbon involved in the reaction. However, to produce the same amount of water, the Sabatier reaction uses twice the hydrogen needed for the Bosch reaction.

For the present, the Sabatier system appears attractive because of its low power penalty, high reaction efficiency, and the absence of any carbon removal problems.

Several alternate methods of CO_2 reduction are compared in Table III.

There are several processes for obtaining pure CO_2 from the cabin air. They fall into three basic categories; solid adsorption, liquid adsorption, and electrodialysis. The most promising approach is through the use of a molecular sieve composed of synthetic zeolite. The method has one drawback in that the zeolite preferentially absorbs water, and hence, entering air must be reasonably dry. A system has been developed which uses the zeolite in combination with silica gel drying beds; this scheme is described in section 6.A.

More advanced concepts, such as the molten electrolyte, which eliminate the need for CO_2 purification, will also be considered in section 6.A.

B. Power System Summary

The two primary factors affecting the selection of the power system are the mission duration and the total amount of power required. We assumed in this analysis a mission duration of 6 months and estimated a power level of approximately 5 kW. The exact power level is calculated in section 7. There are certain factors involving the environment of Mars which must also be considered.

- 24 day-night cycle
- External temperature which varies from 150° to 300°
- Solar density approximately 530 W/m^2 or less depending on amount of cloud coverage
- Possibility of meteoroid shower
- Gravity .39 of earth
- Availability of CO_2 and small amounts of N_2 or A_2
- Possibility of exposure to radiation

It must, of course, be remembered that we are considering a manned mission and that the system to be selected is for surface operation on Mars.

TABLE III

COMPARISON OF CARBON DIOXIDE REDUCTION PROCESSES

[From ref. 9]

Process	Over-All System Reaction	Carbon Deposited	Separate H_2O Electrolysis	Recycle of Process Gases
Sabatier	(a)	No	Yes	No
Sabatier with methane cracking	(b)	In reactor	Yes	Yes
Bosch	(c)	In reactor	Yes	Yes
Molten electrolyte	(d)	In reactor	No	No
Solid electrolyte	(e)	In reactor	No	Yes
Thermal decomposition	(f)	In reactor	No	Yes

[a] $CO_2 + 4H_2 \rightarrow CH_4 + 2H_2O$

[b] $CO_2 + 4H_2 \rightarrow C + 2H_2 + 2H_2O$

[c] $CO_2 + 2H_2 \rightarrow C + 2H_2O$

[d] $Li_2O + CO_2 \rightarrow Li_2O + C + O_2$

[e] $2CO_2 \rightarrow C + CO_2 + O_2$

[f] $CO_2 \rightarrow C + O_2$

Based on these general assumptions, the selection of a suitable space power system considering the present and projected state-of-the-art is limited to only a few of the following systems being studied. These systems are shown in Figure 11. Figure 12 indicates the range of primary utilization for the space power systems under consideration.

1. Thermoelectric conversion system. - Thermoelectric systems are now being used or considered for applications where power requirements are from a few watts to several kilowatts. Thermoelectric conversion devices can be used with both solar or nuclear energy sources depending on such factors as cost, availability, solar intensity, and application. Long life, low cost, radiation resistance and advanced state of development are some of the major advantages of this type of system.

Several thermoelectric materials have been investigated, and typical performance characteristics are shown in Table IV. However, alloys of lead and tellurium (PbTe) have received the most attention in the past. They have a comparatively high "figure of merit" (see section 5.C.1.), but are quite brittle and very sensitive to impurities. Alloys of germanium and silicon (GeSi) have recently been receiving serious consideration because of higher mechanical strength, higher resistance to oxidation and little or no degradation. These materials have also shown promise of higher operating temperature. However, the "figure of merit" for GeSi is only one-half of that of PbTe alloys. Nevertheless, the possibility of longer useful life and reduced weight of heat radiators (due to operation at higher temperature) might make GeSi alloys quite competitive in future space applications.

Thermoelectric devices can be made to achieve efficiencies of about 12% (ref. 10). However, the overall efficiency of the conversion system varies from about 2 to 5% since it depends on so many factors such as method of heat transport, merit of design, temperature of operation and method of handling waste heat. Nevertheless, cost and weight of the system are very low and, therefore, the system deserves serious consideration for long duration missions.

2. Thermionic conversion. - The thermionic system is a static heat-to-electrical energy conversion system that looks attractive for future space applications. The projected power output levels are as high as several megawatts.

Although, relatively speaking, this system is still in the early development stage, laboratory tests performed to date show high promise because of inherent advantages such as:

- Light weight
- No radiation damage
- High efficiency

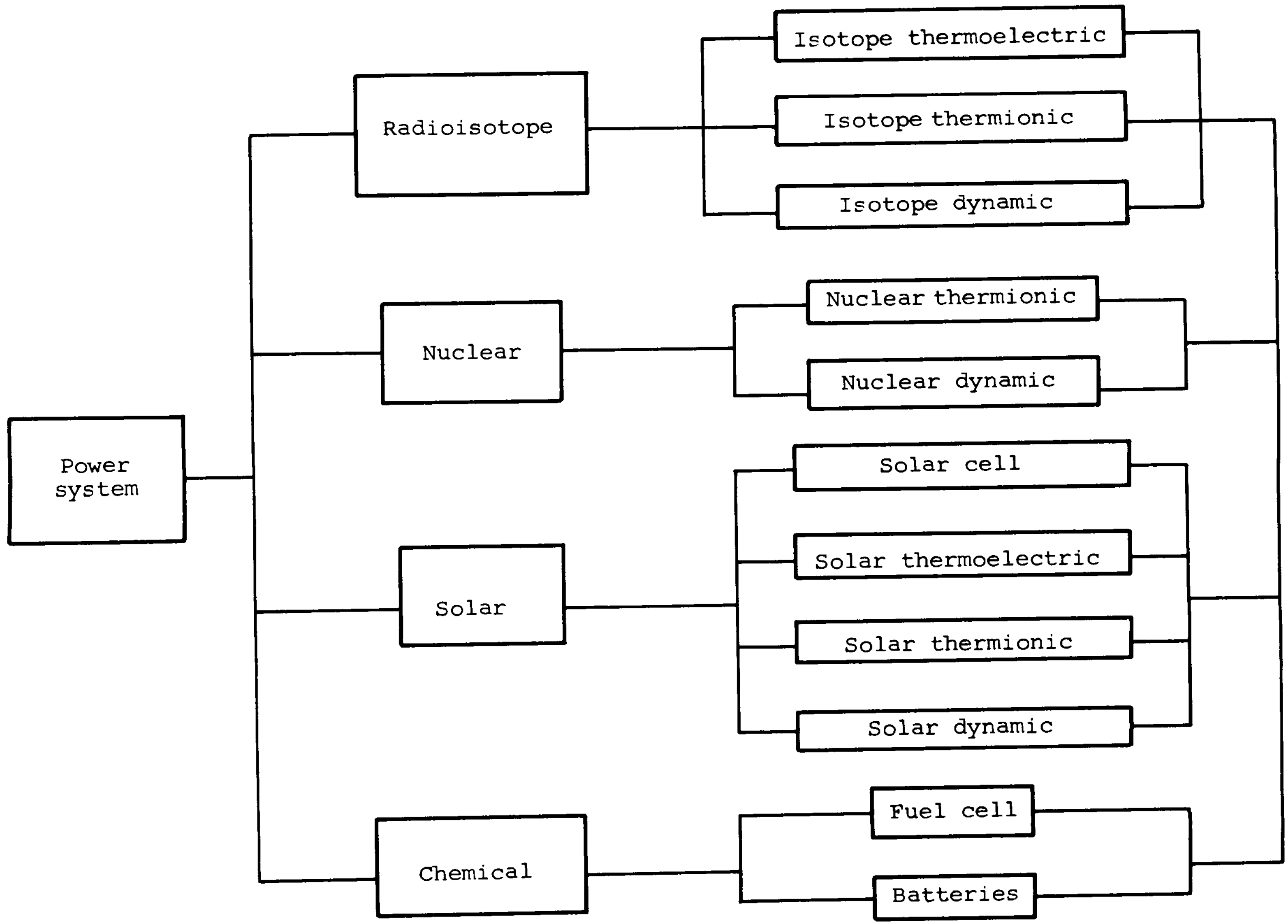


Figure 11 Power System Alternatives

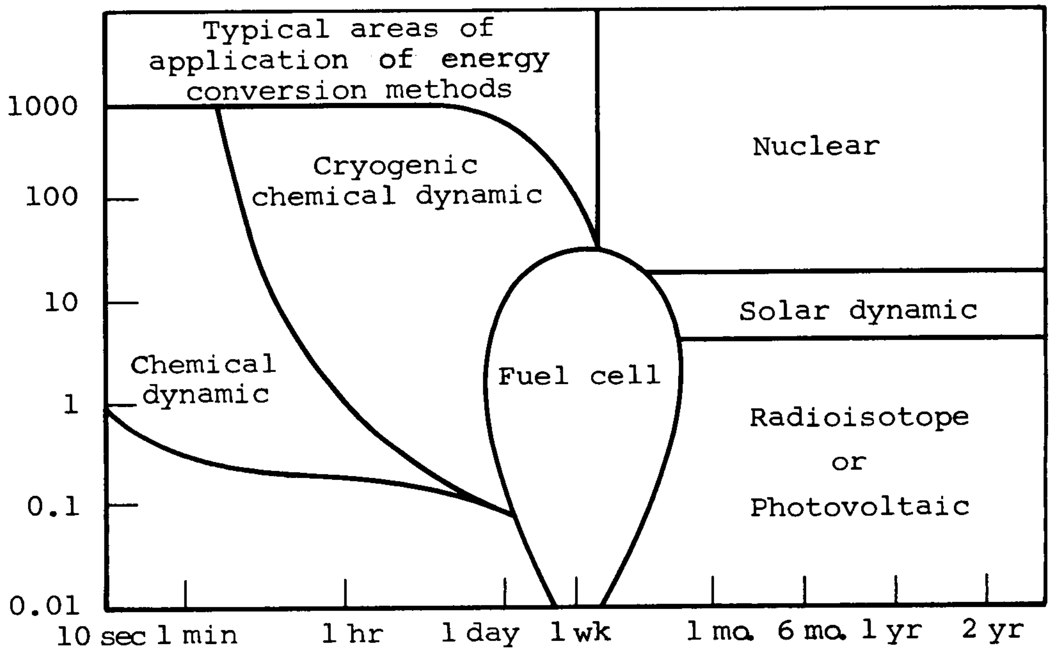


Figure 12 Primary Operational Ranges of Space Power Systems
(From ref. 9)

TABLE IV

TYPICAL PERFORMANCE CHARACTERISTICS OF THERMOELECTRIC DEVICES

[From ref. 11]

Item	Present Performance			Projected Performance		
Material	Bi-Te	Pb-Te	Si-Ge	Bi-Te	Pb-Te	Si-Ge
Hot junction temperature (°C)	250	450	850	250	450	950
Cold junction temperature (°C)	80	180	350	80	180	350
Module efficiency (percent)	4.5	5.0	6.5	5.0	7.0	8.0
Module specific power (W/lb)	15 to 20	15 to 20	15 to 20	15 to 20	30	40
Module specific cost ($/W) (excludes isotope)	70	60	100	15 to 20	20 to 30	10 to 20

- High operating temperature, and light weight radiators

- High redundancy achievable with modular construction

- High power density resulting in compact size

- Very long life

Heat-to-electric conversion efficiencies (for the diode) of over 19% at moderate emitter temperature has already been demonstrated, and efficiencies of up to 30% in the future are predicted. Overall system efficiency will, of course, be less, depending on mission application and design merit. Power density measurements have been approximately 10-20 W/cm^2 and life of several thousand hours has been demonstrated in many laboratories to date.

As mentioned earlier, thermionic converters are still in a development stage and many design problems are yet to be overcome. The predicted performance looks very attractive and the system deserves consideration for the Mars module application, particularly with the use of a radioisotope as an energy source.

3. Dynamic energy conversion systems. - Low specific weight, high efficiency, and advanced state of development have broadened interest in dynamic energy conversion systems. Additional interest in dynamic components has been generated because some of the large static power systems also require rotating parts to provide the function of heat transport. For power requirments greater than 1 kW, dynamic systems also offer a distinct advantage over static systems from the standpoint of overall system weight.

The two dynamic systems that are receiving most attention are the Brayton cycle system and Rankine cycle system. A comparison of general characteristics is presented in Table V.

In a Brayton system (note Figure 13) using argon, the heat generated by the source (solar, radioisotope or nuclear) is transferred to a heat exchanger and then removed by the argon fluid. The argon gas then expands in a turbine which drives an alternator and compressor. The turbine exhaust passes through a recuperator, and transfers heat to the compressor discharge gas. The gas then flows through the radiator heat exchanger where it rejects heat to a separate liquid heat transport fluid. The heat transport fluid then rejects the heat to space through a radiator. The gas from the radiator heat exchanger then flows to the compressor where it is compressed and flows to the heat source through the recuperator, and the cycle is then repeated.

The Rankine system differs from Brayton in that the working fluid is used in two phases, liquid and gaseous. The working fluid in the liquid phase is vaporized by the heat source, and

TABLE V

COMPARISON OF ENERGY CONVERSION SYSTEMS

[From ref. 10]

System	Advantages	Disadvantages	Major Problem Areas
Brayton Cycle	1) Uses single phase working fluid. 2) Working fluid is nontoxic & not corrosive.	1) High system efficiency only obtained by high turbine inlet temperature & low compressor inlet temperature. 2) Low heat transfer coefficients & non-isothermal heat transfer for heat addition & heat rejection, requires large heat exchangers. 3) Performance greatly affected by changes in turbine & compressor efficiencies.	1) Development of turbo compressor-alternator gas bearings. 2) Development of turbines & compressors to operate at high efficiencies.
Rankine Cycle	1) Minimum radiator area. 2) Low system weight.	1) Requires boiling, condensing & separation of two phase fluid flow in the zero g environment. 2) Temperature and radiation breakdown of Dowtherm A.	1) Development of a radiator condenser to operate in space. 2) Prevent freezing of the fluid in the radiator.
Thermo-electric	1) No rotating parts. 2) Minimum development effort required relative to other systems.	1) High system weight. 2) Available radiator area dictates high cold junction temperatures and results in low system efficiency.	1) Sublimation of element at high operating temperature. 2) Development of a good electrical contact between the element & the conductor. 3) Mechanical properties of the semi-conductors create problems in designing an integrated system.

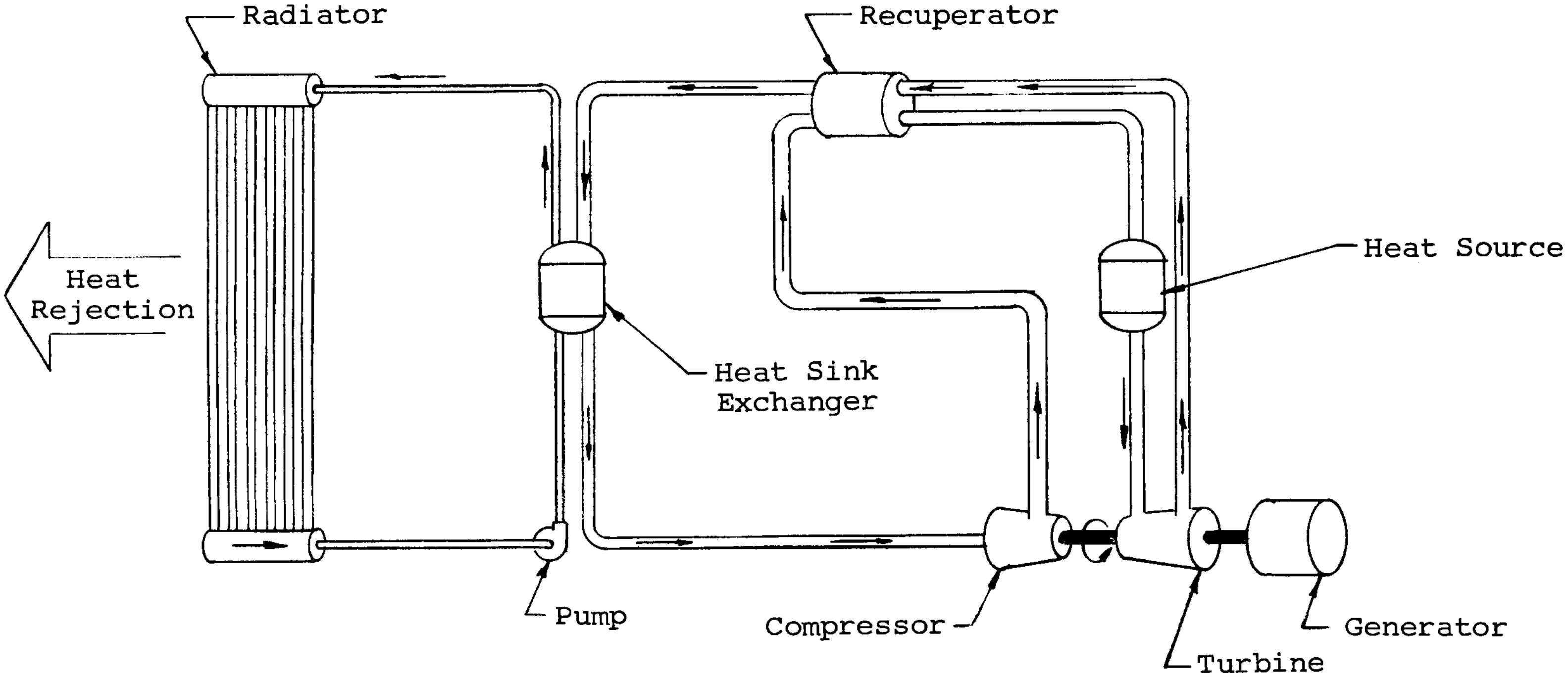


Figure 13 Brayton Cycle Space Power System
From ref. 11

the vapor expands through a turbine which drives an alternator and condensate pump. The exhaust gas is condensed by a heat rejection loop, and condensed liquid is vaporized again by the heat source repeating the cycle.

The Brayton cycle has distinct advantages over the Rankine cycle from the standpoint of the single phase working fluid, reduced corrosion problems, and better cycle temperature ratio. The Brayton cycle efficiency is higher than that of Rankine cycle although in both cases, the cycle efficiency is greatly dependent on cycle temperature ratio. The specific weight and the compressor work are higher for Brayton cycle than the Rankine cycle. The Brayton cycle also requires much larger radiator area.

4. Nuclear system. - Nuclear systems, as defined in this report, use a reactor as the heat source and would not require orientation or elaborate energy storage systems.

From Figure 12, it can be seen that the range of application for nuclear power systems is from tens to hundreds of kilowatts. The estimated power demand for the MSM is below the ten kW level. The SNAP-8 reactor, which is liquid-metal cooled, is the only reactor power system fully committed to development at this time, and it appears that it could be the late 1980's before any of the SNAP series might appear in mission use (ref. 11).

A reactor system, based on the available technology, would be quite heavy, because of the shielding required. Some reduction in shielding weight could be achieved by the employment of the shadow shielding concept. This, however, requires that the reactor be located remotely, which creates the necessity for long heat transport fluid lines with the attendant pumping power penalty and risk of meteorite penetration (ref. 12).

5. Radioisotope system. - Considerable information and design data have been available for the application of radioisotopes to space power generating systems. A summary listing of advantages and disadvantages of these systems is presented in Table VI, and a comparison of characteristics of specific isotopes is presented in Table VII. Radioisotopes as a source of energy have been strongly considered for missions such as unmanned interplanetary probes, lunar orbiting vehicles, communication satellites, and manned space laboratories, where electric power requirements are as high as 10 kW. The radioisotopes receiving most attention for use as a power source are polonium-210, plutonium-238, strontium-90, cerium-144 and curium-244.

The primary advantages of radioisotope power systems for the Mars Module application are that (1) the system is not subject to radiation degradation, (2) it is not susceptible to degradation from meteoroid damage, (3) it does not require orientation, and (4) it is not affected by cloud cover or dust storms.

TABLE VI

ADVANTAGES AND DISADVANTAGES OF ISOTOPIC POWER SOURCES

[From ref. 10]

Advantages	Disadvantages
Can be used in any environment	Safety requirements
No power storage requirement	Availability
Flexible in operation	Possible costs
Long operational use	Unfamiliarity to users
Compact	Possible interference with experiments
Heat available to maintain equipment thermal balance	
Highly reliable	

TABLE VII

CHARACTERISTICS OF RADIOISOTOPES FOR POWER PRODUCTION

[From ref. 10]

Isotope	Chemical Form	Half-Life	Power Density, W/cm^3	Type of Radiation Emitted	Cost, $/W
Pu^{210}	Metal	138 days	1210	α	188
Pu^{238}	PuO_2	89.6 yr	3.5	α	894
Sr^{90}	SrO	28.0 yr	1.4	β	19
Ce^{144}	Ce_2O_3	285 days	20	β,γ	1.0
Cm^{244}	Cm_2O_3	18.4 yr	26.4	α,n	357

The basic properties that greatly influence selection of isotopes for specific missions are

- Power density and source volume
- Radiation emission and shielding requirements
- Half-life
- Cost
- Availability of required amounts of the isotope in the future

Of the properties listed above, the most significant for long term manned missions are shielding requirements, cost and availability.

Polonium-210 requires very little shielding, has high power density, will be readily available at reasonable cost, but has a half-life of only 138 days.

Plutonium-238 has a very long half-life (approximately 89 years), requires little shielding, but exhibits a low power density. Its availability is also limited and cost is very high.

Strontium-90 has 28 years of half-life, is very cheap and readily available. However, it requires very heavy shielding.

Cerium-144 has a short half-life (285 days) and requires heavy shielding. However, it is cheapest, and is very readily available.

Curium-244 has a very long half-life (189 years) and will be readily available, but requires heavy shielding and is quite expensive.

6. Secondary power systems. The following systems are more well developed and traditional for space power applications, however, ultimate power output levels for extended missions preclude their consideration as primary sources. It may be reasonable, however, to utilize them for peak loads or auxiliary power in conjunction with one of the other systems discussed.

a. Solar cells: It has been common to state in the past that solar cells are applicable to nearly all space power requirements up to 1 kW of electric power, but beyond that, the physical size, cost, orientation and other problems make them impractical. However, as space power requirements increase and exceed 1 kW, and solar cell development continues, they may become feasible for multikilowatt systems in certain applications.

Although solar cells have been proved to be very advantageous for use in many earth orbiting missions, there are certain problems associated with them which make their performance unpredictable in our application. Some of the major problems are:

- Light intensity. - Not only is the output dependent on the solar density, but the efficiency of the cell changes with solar intensity. There is also a lack of solar energy in periods of night or shade and additional energy storage systems must be considered a part of an entire system, imposing weight penalties, depending on the duration of shade periods.

- Temperature. - Efficiency of solar cell decreases greatly with increase in operating temperature.

- Orientation. - The cell must be oriented for optimum utilization of solar energy.

- Large surface area. - Due to low efficiency of energy conversion and lack of sufficient energy density, very large surface areas are generally required.

b. Solar concentrators: Solar concentrators for use in systems requiring energy conversion by thermionic, thermoelectric and dynamic systems are usually a paraboloid structure available as one-piece structures up to 10 ft diameter, and as unfolding petal-type structures for larger diameters. Major considerations in the proper design of solar concentrators are high reflectivity, good geometrical accuracy, light weight, high collector-absorber efficiency of the mirror-cavity combination, and precise orientation.

Maximum collector-absorber efficiencies are dependent on the cavity temperature, collector diameter, mirror geometry, distance from sun and mirror efficiency. The specific weight of solar collectors ranges anywhere from 0.2 lb/ft^2 to 1.0 lb/ft^2 (ref. 13). Since solar collectors can be used with one of many conversion systems listed earlier, the collector area requirement will vary depending on the cavity temperature requirements and the efficiency of the conversion system.

Solar collector requirements for the Mars Module application are subject to the same difficulties discussed previously for solar cell systems. Their performance, however, is not affected by temperature or particle radiation as in the case of solar cells.

c. Batteries: The following types of batteries have normally been considered most applicable to space use.

- Nickel-Cadmium (Ni-Cd)
- Silver-Cadmium (Ag-Cd)
- Silver-Zinc (Ag-Zn)

Ni-Cd and Ag-Cd have a greater cycle life than Ag-Zn cells, and are used generally as secondary cells. Ag-Zn cells have the highest energy density, but cannot sustain as many charge discharge cycles.

All sealed cells appear to have some of the following shortcomings:

- Limited ability to withstand repeated deep discharges
- Limited capability for rapid recharge
- Possible damage by overcharging
- Narrow working temperature range
- Poor energy density
- Limited shelf life
- Potential separator failures
- Potential seal failures

Temperature has a strong effect on the performance of all batteries. Charge-discharge characteristics, shelf life, voltage and current characteristics, and cycle life are all greatly dependent on temperature.

Operating temperature range for Ni-Cd and Ag-Cd is from about 30°F to 100°F, whereas some experimental Ag-Zn batteries are good for temperatures from about 40°F to as high as 250°F.

The cycle life utilizing 25% depth of discharge is maximum for Ni-Cd and is approximately 8000 cycles at room temperature (ref. 11). Ag-Cd, under the same conditions, has a cycle life of about 3000. High temperature, experimental Ag-Zn cells have demonstrated a cycle life of 2500 at 120°F, using 25% depth of discharge.

Energy density of all batteries would also vary depending on the mission requirements and on number of charge-discharge cycles.

Ni-Cd is the most limited of the three from an energy density consideration (2 to 10 W-hr/lb, depending on cycle life). Ag-Cd batteries vary from 6 to 20 W-hr/lb, whereas the experimental Ag-Zn batteries have demonstrated an energy density of 10 W-hr/lb at a cycle life of approximately 2000 cycles; increasing to as much as 40 W-hr/lb at zero cycle life.

Although batteries are not suitable as a primary power system for missions of any appreciable duration, they can provide power in shade periods, in emergency, and for peak load requirements.

d. Fuel cells: Fuel cells have been successfully used for many missions, particularly of short duration. Fuel cells like all chemical power systems, have the inherent disadvantage of requiring excessive amounts of fuel for long duration missions. Presently, the fuel cell is limited in application to missions of one month duration. The most significant problem associated with fuel cells for mission durations greater than 30 days is the increase in weight and volume of the system due to fuel storage requirements. Other problems, such as degradation of the cells, electrodes, electrolyte and seals have also limited system applications to short missions.

Major advantages that have led to the use of fuel cell systems on many missions are:

- High energy efficiency
- Very low power dissipation at idling resulting in almost zero fuel consumption in this mode
- Silent operation and reduced crew fatigue
- Very few moving parts and increased reliability
- Production of potable water
- Not susceptible to degradation by radiation exposure

Among all advantages, the most important one, particularly for manned missions, is the production of potable water as a by-product. As much as 1.0 lb of water is produced per kW-hr of electrical energy output (ref. 14).

7. Concluding remarks. - The foregoing background discussion of the state-of-the-art in space electric power systems has presented general information describing the advantages and disadvantages of the various candidate systems. More detailed subsystem considerations are presented in section 6.

5. MODULE DESIGN CONSIDERATIONS

A. Summary

This phase of the study was directed toward developing a feasible configuration for the module, as well as performing a structural analysis and layout of components. It enabled us to assess some of the interrelated factors associated with the module subsystems and extended mission requirements. This approach also permitted us to develop some practical values which could be directly applied to our studies in subsystem integration.

The results of this phase of study were utilized in the following areas of the overall plan:

- Definition of a physical module configuration consistent with the environment of Mars, and suitable for extended missions
- Derivation of volume and weight information for the various system components to assure compatible module volume and construction
- Establishment of more specific design requirements and operating characteristics for the various subsystems
- Determination of physical constraints affecting major subsystems, including life support and energy conversion, and establishment of a design-oriented rather than purely study-oriented base for assessing advanced technology

In the section to follow, the factors affecting the selection of the final configuration are considered in detail, as are the design assumptions made in its development. The module design details are presented in Appendix A.

B. Design Guidelines

To enable the module to fulfill its intended function, the following physical requirements and general design guidelines have been adopted.

1. <u>Structural integrity</u>. - The module structure has been designed to withstand the extremes of the Mars environmental conditions. The materials chosen will maintain their structural integrity for the duration of the mission.

2. <u>Safety</u>. - Design techniques which would economize on structure weight and volume were given secondary consideration to the safety of the personnel involved, and only high reliability design approaches were considered.

3. Life support. - The module has the capability of supporting four men for six months. The personnel will be able to live in the structure without the aid of space suits or similar protective devices. Adequate space is also provided for servicing the necessary life-support equipment.

4. Ingress and egress. - Due to the low absolute pressure and adverse compositon of the Martian atmosphere, an airlock has been provided at one end of the module for ingress or egress of personnel.

5. Interior design. - The interior of the module has been designed to satisfy the physical and psychological needs of the crew for a stay time of six months duration. Areas have been included for work, sleep, eating, hygiene, recreation, exercise, and communications.

6. Size and weight. - The cargo envelope has not been fully defined, however, from recent published studies regarding Mars mission booster development, it appears that the module will fit quite easily into the envelope which is currently being envisioned.

7. Instrumentation. - This area has been designed to house equipment for Earth communications and comprises the monitoring station for all of the module systems. A storage area is also provided for most of the experiments to be performed.

8. Power. - The module will use a thermally integrated isotope-Rankine power supply. It will be integrally mounted at the end of the module, maintaining a safe distance from the personnel.

C. Design Philosophy

The primary objective of this configuration study is to identify and assess those areas of advanced technology which have the greatest potential for effectively supporting a Mars mission of six months' duration in the 1980 period and beyond. The value of this type of study will be significantly enhanced if we can logically select candidate subsystems (for life support, power generation, and other essential requirements) and then show how these subsystems can be functionally integrated in a Mars Surface Module (MSM).

The larger system weights necessary for the MSM require that expanded structures and fabric construction techniques be considered very cautiously. The very high surface drag forces, small expansion pressures, and high transportation and landing stresses involved, cause the detail design considerations to be quite extensive and essentially beyond the scope of the present study. However, it is necessary that the present study in module design be broad enough to include the primary considerations.

The factors considered and assumptions made in arriving at the proposed MSM configuration were as follows.

1. Considerations. -

- A minimum amount of surface preparation required
- Adequate living and equipment space with room for equipment repair
- Stabilization devices integrally mounted on the module framework
- A minimum amount of ground contact area to reduce the module heat load
- A low wind drag and stability profile due to expected high surface winds

2. Assumptions. -

- A means of lowering the module to the surface of Mars will be available at the time of landing.
- Personnel will work in a shirtsleeve environment, and will wear space suits only upon leaving the module. Personnel will work outside the module only for vital experiments and in emergency conditions.
- The module will house four men for six months.
- Seismic effects are not critical.
- The surface material on Mars will not present a radiological hazard to men or materials.
- The men will be able to occupy the module fairly soon after landing on the planet.

The configuration selected from the above considerations does not purport to be the optimum configuration for the final Martian effort, since the total mission remains very much undefined.

It does, however, possess many of the basic attributes required of a typical extended mission MSM, primarily from the standpoint of volume and weight. It is hoped that through this reduction of abstract thoughts to physical dimensions, many system design considerations that would otherwise have been overlooked will be accounted for.

D. Subsystems Considerations

The following discussion of the power source and the life-support subsystem characteristics was undertaken to tentatively identify candidate subsystems and to assure the availability of space for each subsystem needed to control the module environment. The vital requisite of these subsystems is to maintain a habitable shirtsleeve environment for the duration of the mission.

Whenever a final selection of a specific component or process is selected for analysis, the detailed tradeoff summary will be presented in a later section. The components, processes and/or subsystems to be outlined have been evaluated in this context for weight and volume in order that a reasonable subsystem layout could be made.

The functions to be performed by each of the life-support and environmental control subsystems are summarized in Table VIII. The astronaut comfort level was selected from Figure 14.

For the advanced state-of-the-art components, weight and volume projections were made to the 1980 time period. These advanced processes are in constant and rapid evolution, hence, the updating of the preliminary tradeoffs is a continuing effort.

1. Power supply. - The estimated power demand for the module is approximately 5 kW of electrical power, which will be furnished by an isotope-Rankine power supply. It is anticipated that plutonium-238 will be the source isotope and that Dowtherm A will be the working fluid in the Rankine cycle. This power system, when used in a thermally integrated configuration with the life-support subsystems, will weigh 2,076 lbs and occupy a volume of approximately 19 ft^3 (refs. 15 and 16).

2. Food supply. - The food management subsystem is based on the utilization of dehydrated (primarily freeze-dried) and other room-temperature stable foods, to eliminate requirements for freezers, refrigerators, and ovens or other cooking equipment (ref. 17). All that is needed in the way of facilities and equipment is a food storage volume, a packaging technique, and a water heater and cooler with a metered (1.1 lb/man-day) dispensing device. Considering a food requirement of 1.4 lb/man-day, a mission duration of 180 days, and a crew size of four, the total food storage parameters would be as follows:

$$\text{Food Weight} = \text{Specific food consumption} \times \text{crew size} \times \text{mission duration}$$

$$= 1.4 \times 4 \times 180 = 1{,}010 \text{ lb}$$

The food supply is assumed to be packaged with an average density of 17.5 lb/ft^3. Based on this allocation, the required storage volume is 57.6 ft^3.

TABLE VIII

LIFE SUPPORT REQUIREMENTS

Requirement	Design
Metabolic Input	
Oxygen	1.84 lb/man-day
Drinking Water	6.10 lb/man-day
Food (including mixing water) . . .	2.50 lb/man-day (3,000 calories)
Metabolic Output	
Carbon Dioxide	2.20 lb/man-day
Feces	0.45 lb/man-day
Urine	3.20 lb/man-day
Perspiration and Respiration	3.00 lb/man-day
Heat	11,000 Btu/man-day
Environmental Control	
Cabin Temperature	75°F
Relative Humidity	50 %
Total Pressure	7.0 psia
Oxygen Partial Pressure . . .	3.5 psia
Nitrogen Partial Pressure . .	3.5 psia
Carbon Dioxide Partial Pressure	Less than 5.0 mm Hg
Atmospheric Circulation . . .	50 ft^3/min
Wash Water	15.0 lb/man-day
Total Atmospheric Leakage . . .	4.00 lb/day
Make Up: Oxygen	2.13 lb/day
Nitrogen	1.72 lb/day
Water	0.06 lb/day
Carbon Dioxide . . .	0.09 lb/day

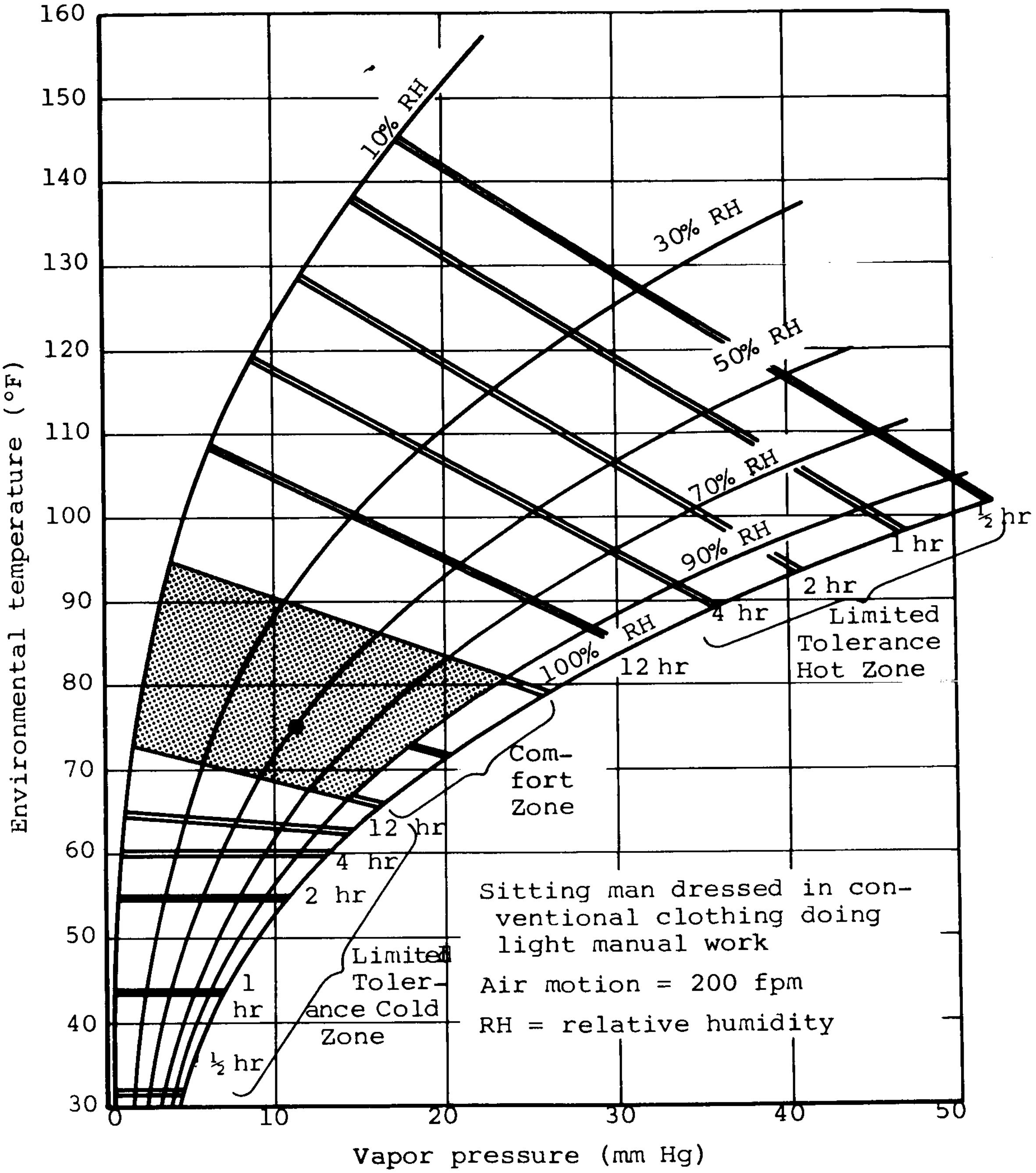


Figure 14 Thermal Requirements for Tolerance and Comfort in Crew Compartments

(From ref. 18)

3. Water reclamation. - The human fluid wastes (wash water, urine, and condensate) would be converted to potable water by a distillation process. An additional vapor-pyrolysis step would be needed for the urine purification cycle. A separate collection of all water sources is recommended because of the added pyrolysis step, and because the purifying agents added to each cycle vary in degree and type (ref. 19).

After processing, the potable water can be pumped to a combined storage point at the top of the module.

For comparative purposes, if no water reclamation subsystems were used, the total water required would be as follows:

$$\text{Open Water Supply} = 22.2 \text{ lb/man-day} \times 4 \text{ men} \times 180 \text{ days}$$

$$= 16{,}000 \text{ lb}$$

It is recommended that at least one-sixth (one month's demand) of the open water supply be carried in storage. This will include the make-up requirement of 0.9 lb/man-day, or 648 lb total, and will aid in maintaining a low bacteria count and contaminant level within the storage volume, as well as allow a reasonable margin of safety from water processing equipment malfunction.

The storage volume for the water supply was derived as follows:

$$\text{Storage Volume} = \frac{1}{6} \times \frac{16{,}000}{62.4} = 42.7 \text{ ft}^3$$

The total atmospheric-distillation water reclamation system will weigh 171 lb and occupy a volume of about 8 ft^3.

4. Carbon dioxide removal. - The system chosen for analysis was the molecular sieve composed of synthetic-zeolite. The sieve cannot be used alone since it has a higher affinity for water than it has for CO_2, thus a drying step must be added which uses silica-gel to remove the moisture from the incoming air.

Two silica-gel molecular sieve branches are connected in parallel so that when one is absorbing CO_2, the other branch is desorbing CO_2.

The anticipated weight and volume of the system are 60 lb, and 3.0 ft^3 respectively (ref. 20).

5. Oxygen recovery. - The system chosen for analysis is a methanation reaction defined as follows:

$$CO_2 + 4H_2 \rightarrow CH_4 + 2H_2O$$

This reaction is called a Sabatier reaction, whereby CO_2 is combined with hydrogen to produce water and methane. The water is then electrolyzed to produce O_2 and the methane is either burned or dumped overboard.

This method has several advantages:

- High overall reaction efficiency (99%)
- Low temperature reaction (550°F), and hence a low start-up power requirement
- Low weight (64 lb)
- Requires no carbon removal from the reactor

The methane produced in the reaction could be further reduced to carbon and hydrogen, however, this would bring back the problem of carbon removal.

It would be more efficient to burn off the methane and use the reaction as incinerator for waste materials such as paper, tissue, whiskers, etc.

6. <u>Waste management</u>. - Urine will be handled as described in section 5.D.3. There will, however, be an additional temporary storage area for direct collection and later processing by the urine water-reclamation system.

No attempt will be made to obtain water from the fecal wastes. It is not anticipated that this process will be available or necessary by mission time.

The fecal wastes will be stored temporarily, then transferred to a drying unit before final storage. The total feces to be stored is derived as follows:

$$\text{Feces Storage Weight} = (0.45\ \text{lb/man-day})(4\ \text{men})(180\ \text{days})$$

$$= 324\ \text{lb}$$

This can be stored at a density of 50 lb/ft^3, hence:

$$\text{Feces Storage Volume} = \frac{324}{50} = 6.5\ ft^3$$

The equipment necessary to accomplish the transfer and drying will weigh approximately 185 lb and occupy 6 ft^3.

7. <u>Thermal control</u>. - The selection of a temperature control method is dependent on the module heat loading. This heat loading is itself a function of the equipment being selected. For this reason the sizing calculations for the thermal control subsystem are presented in a later section on thermal control which takes into account the factors mentioned.

8. <u>Humidity control</u>. - To maintain the relative humidity level within the module at a comfortable level of 50%, moisture generated within the module must be removed.

The water comes from three sources:

- Crew activity (respiration and perspiration)
- Control processes (water reclamation, thermal control, etc.)
- Crew functions (showers, food preparations, etc.)

The method selected is the simplest method of moisture removal, that of a cooler-condenser operating at approximately 45°F. The system weighs about 20 lb and occupies about 2 ft^3 of space.

9. Contaminant control. - The atmospheric contaminant control subsystem must be made up of several components to remove different types of contamination.

An electrostatic precipitator will be used to remove suspended particulate matter such as dust. An activated charcoal filter will be used for removal of odors and noxious or toxic gases. For bacteriological contaminants, an "absolute filter" will be used in combination with ultraviolet lamps to destroy the spores and bacteria trapped in the filter.

The total system will weigh approximately 2.50 lb and occupy 1.5 ft^3 of space.

10. Atmospheric circulation. - In addition to normal ducting, space must be allocated for a blower to handle approximately 50 ft^3/min of module air. A duplicate of this blower will be installed in case of a malfunction. The two blowers will weigh about 32 lb and occupy 4 ft^3.

11. Atmospheric supply. - A preliminary analysis was made to obtain baseline weight and volume figures for oxygen and nitrogen storage.

The water vapor and carbon dioxide concentrations are neglected for this evaluation.

It was assumed that the airlock would be used once a week for the duration of the mission or a total of 24 times. One egress and ingress cycle requires 21.5 lb of oxygen and 18.8 lb of nitrogen (ref. 21). It was further assumed that gaseous storage in spherical tanks will be used. From the standpoint of weight, this may appear to be a poor choice. However, it does present potential advantages over the cryogenic method for long term storage, since pressure decay and boiling vent losses do not occur, and the storage is at higher pressure allowing instantaneous and rapid use. For this study, we are not attempting to make a firm selection in this area. We do know that by using either supercritical or subcritical cryogenic storage, the total weight of the gaseous storage technique could possibly be compensated by the added reliability.

To determine the tankage weight and volume for gaseous storage, a pressure of 8,000 psi was chosen for the storage of oxygen (note Figure 15), and a pressure of 3,000 psi was chosen for the storage of nitrogen (note Figure 16). With the information derived from Figures 15 and 16, plots for oxygen and nitrogen tank weight were made. These are shown in Figures 17 and 18.

The oxygen balance is as follows:

Metabolic Input	+7.36 lb/day
Atmospheric Leakage	+2.13 lb/day
Average Airlock Losses	+2.86 lb/day
O_2 Recovery System	-5.76 lb/day
Storage	+6.59 lb/day

Oxygen Storage Weight = 6.59 x 180 = 1,190 lb

From Figure 15 Oxygen Storage Volume = 31.0 ft^3

From Figure 17 Oxygen Tankage Weight = 1,810 lb

Nitrogen is not consumed in the metabolic process, hence the only supply needed is for leakage and airlock losses. The nitrogen balance is as follows:

Atmospheric Leakage	+1.72 lb/day
Average Airlock Losses	+2.51 lb/day
Storage	+4.23 lb/day

Nitrogen Storage Weight = 4.23 x 180 = 762 lb

From Figure 16 Nitrogen Storage Volume = 57.4 ft^3

From Figure 18 Nitrogran Tankage Weight= 945 lb

The weights and volumes for each of the subsystems discussed in the preceding paragraphs is summarized in Table IX.

E. Concluding Remarks

The information presented in this section, along with that of Appendix A, has permitted us to develop a general configuration for the module and to identify and assess, on a preliminary basis,

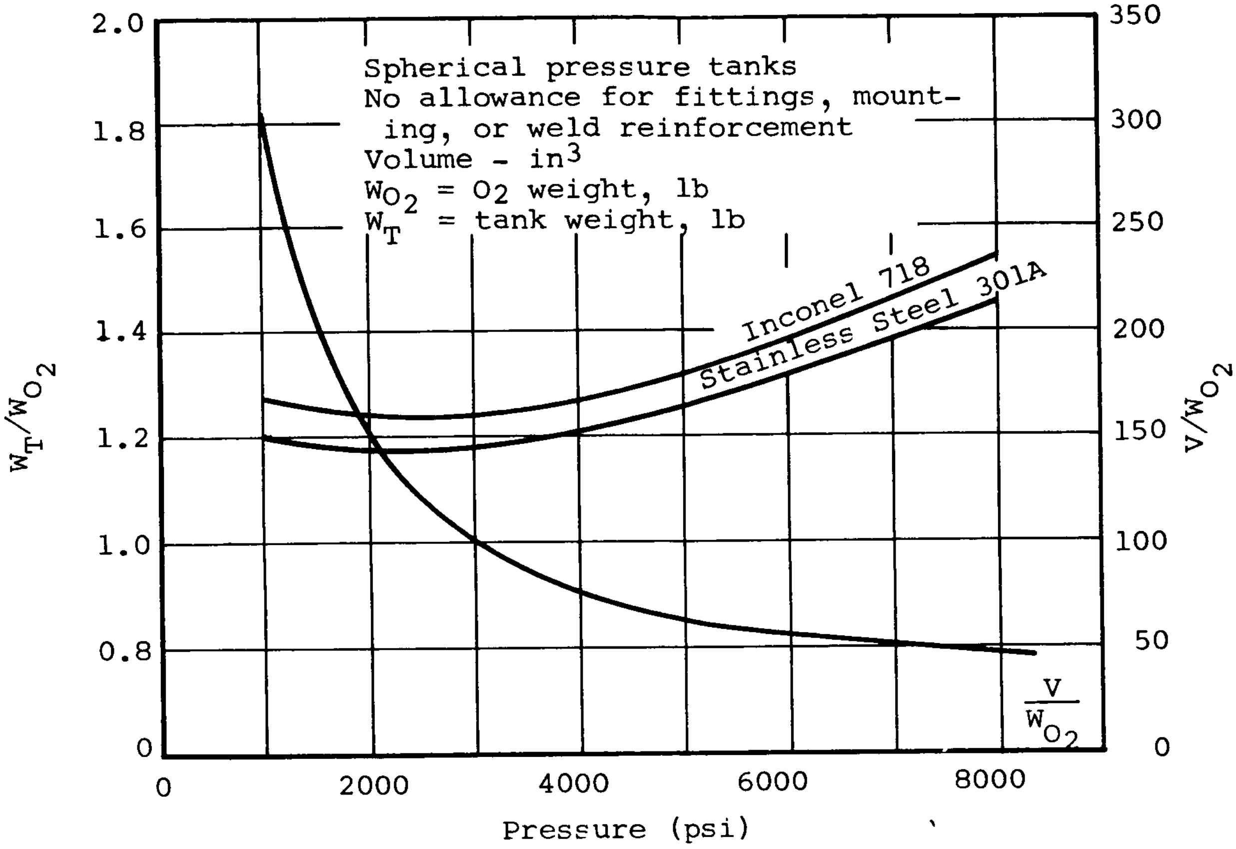


Figure 15 Oxygen Tankage Required for Gaseous Storage (From ref. 22)

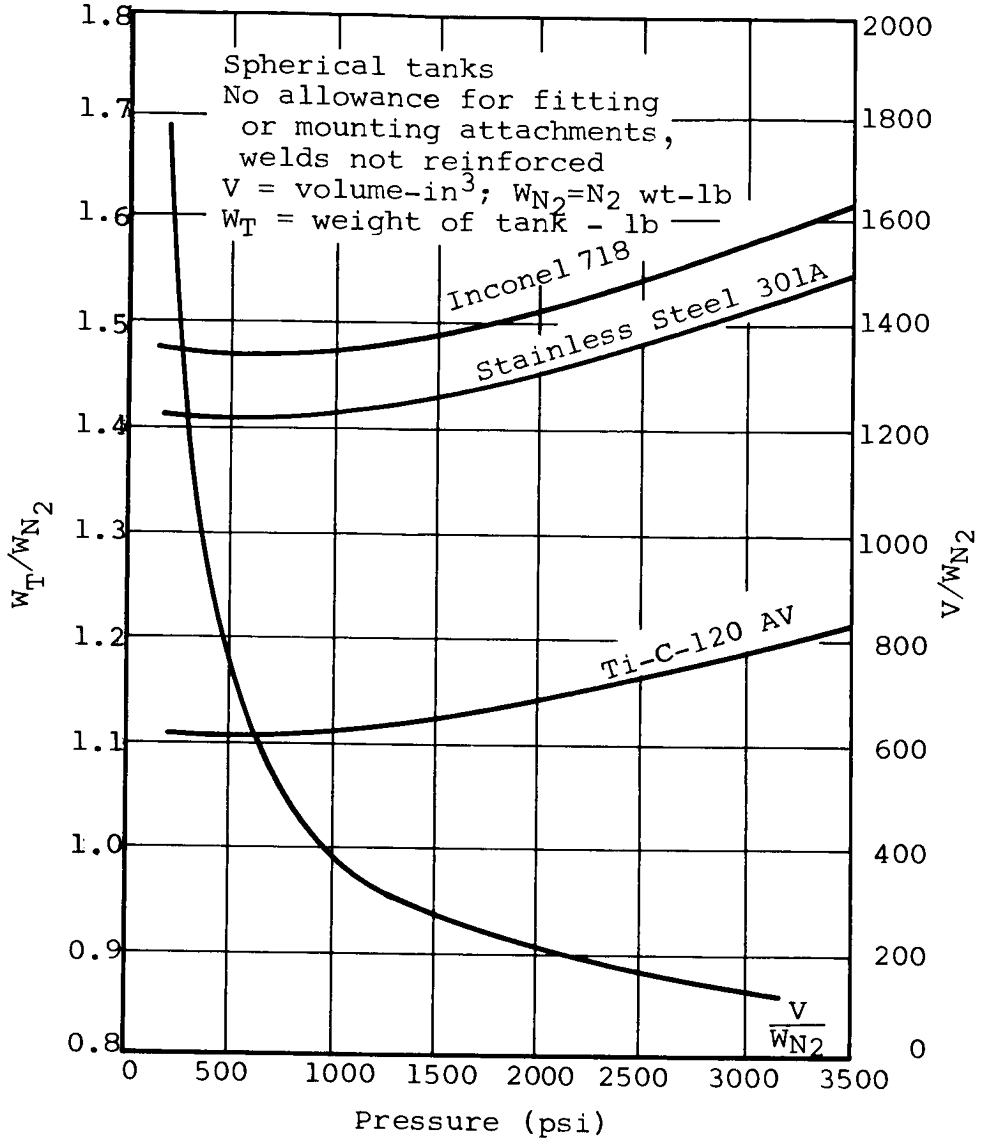


Figure 16 Nitrogen Tankage Required for Gaseous Storage

(From ref. 22)

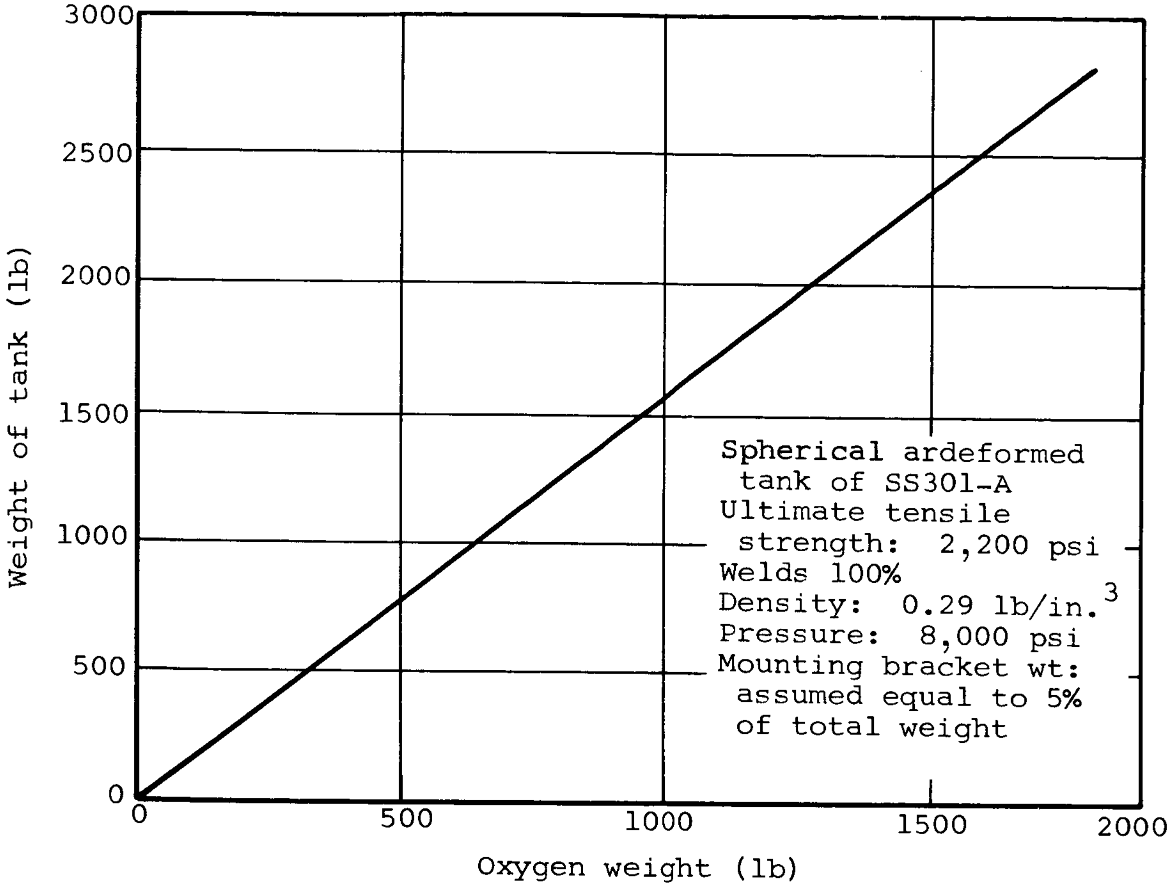


Figure 17 Tankage Weight as a Function of Fluid Weight, Gaseous Oxygen Storage

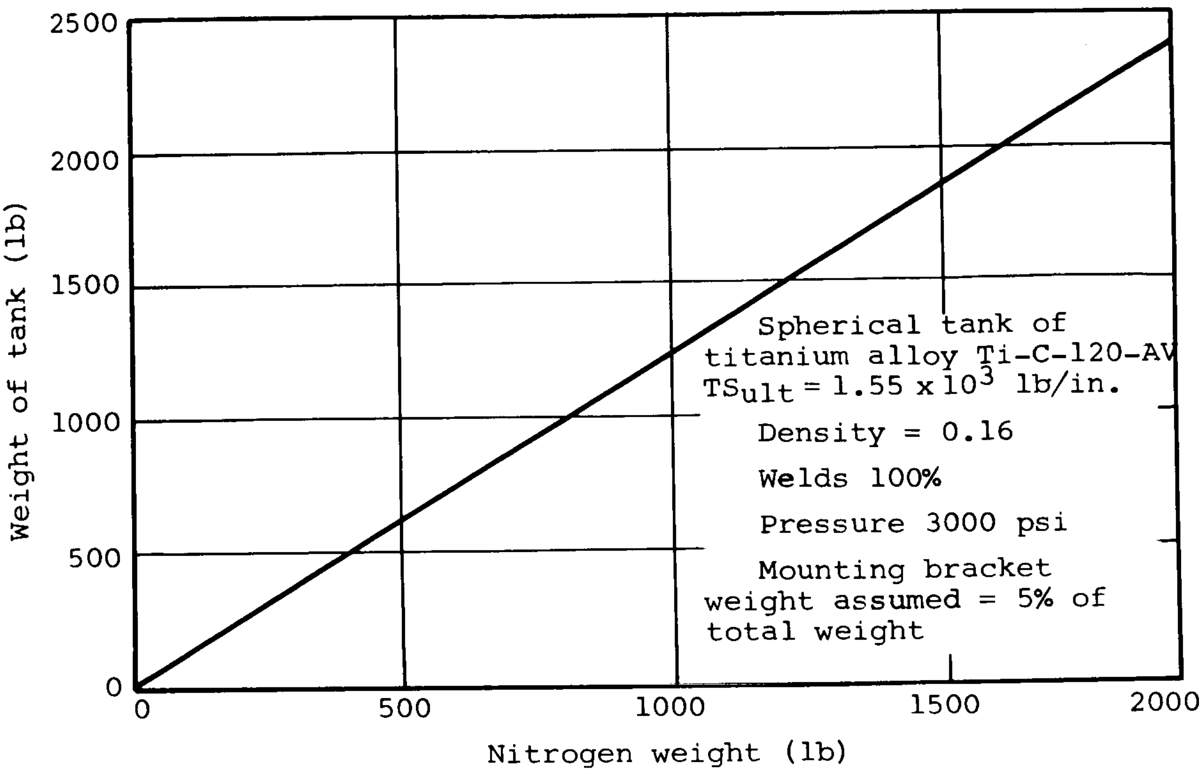


Figure 18 Tankage Weight as a Function of Fluid Weight, Gaseous Nitrogen Storage

TABLE IX

SUBSYSTEM VOLUME AND WEIGHT SUMMARY

Subsystem	Volume, ft^3	Weight, lb
Power Supply	28.0	3,080
Food Supply	57.6	1,152
Water Supply	42.3	2,640
Water Reclamation	8.0	171
CO_2 Removal	3.0	60
CO_2 Reduction and O_2 Recovery. .	5.0	64
Feces Processing	6.0	185
Thermal Control	6.0	700
Humidity Control	2.0	20
Contaminant Control	1.5	25
Atmospheric Circulation	4.0	32
Oxygen (includes tankage).	31.0	3,000
Nitrogen (includes tankage). . .	57.4	1,707

the subsystems essential in meeting the extended mission requirements. The scope of this section has been confined to the development of broad guidelines and basic system characteristics pertinent to module design.

In the next section, we will provide a more detailed justification of the subsystems used in this section of the report.

6. SUBSYSTEM TRADEOFF STUDIES

A. Life-Support System

1. Oxygen recovery methods. - All recent manned earth orbiting missions have had to contend with the removal of metabolic CO_2 contents in the cabin atmosphere. These past missions, however, have been comparatively short in duration and the CO_2 production rate of 2.20 lb/man-day could be handled simply by carrying an amount of LiOH on board equal to the amount of CO_2 produced during the mission. LiOH combines with CO_2 to produce Li_2CO_3 and H_2O, which then must be stored throughout the mission.

For extended mission conditions, large amounts of CO_2 are generated, and large quantities of oxygen are required for the basic metabolic process over the duration of the mission. In current applications such as nuclear long range submarines, where no particular oxygen storage weight problem exists, the CO_2 is removed by using monoethanolamine which is then freed of CO_2 by boiling. This process does not lend itself to space applications since the CO_2 has to either be stored or dumped which, in either case, represents a loss in oxygen. It becomes advantageous in extended mission technology to not only remove the CO_2 produced inside the cabins, but to find a means of regenerating the O_2 consumed in the metabolic process, preferably by reduction of CO_2.

Water is also given off in the metabolic waste products and could be used as a source of oxygen production. However, the collection of respiration and perspiration involves a certain amount of purification and, thus, lends itself more suitably to use in the potable water regenerating system. Thus, the best way of closing the oxygen loop is through the reduction of carbon dioxide.

There are many methods of generating oxygen through the reduction of carbon dioxide. They all depend on first having obtained reasonably pure CO_2 from the cabin air. The CO_2 concentration technique that has had the most development and, consequently, which holds the most promise, is the solid adsorbent technique using regenerable synthetic zeolite. Zeolite has a higher affinity for water than it does for CO_2, thus, a separate water-adsorbent bed must be used upstream. Either zeolite or silica-gel could be used, however, silica-gel is preferred because it has a lower desorption energy.

The basic four-bed system, shown schematically in Figure 19, operates as follows. The air to be processed is first dried in a desiccant bed; the flow is then cooled since the adsorption characteristics of synthetic zeolites are poor for warm beds; it next enters the molecular sieve bed, where CO_2 is removed. The process gas is then circulated through the second silica-gel bed, where it picks up the water adsorbed during the previous cycle. Finally, the air from which the CO_2 has been removed is exhausted from the system and returned to the module air. While one molecular sieve bed is adsorbing, the second is regenerated by thermal desorption. To envision the alternate mode of operation, simply rotate all valves 90° clockwise. Under actual operating conditions, this takes place at approximately 40 minute intervals (ref. 23).

All the CO_2 reduction systems to be considered must be used in combinations with a CO_2 concentration unit that provides a reasonably pure CO_2 feed.

Six basic systems that involve current study for space application were evaluated, using as a basis for comparison the following factors in their order of relative importance: (1) reliability, (2) weight, (3) power, (4) volume and (5) maintenance.

The reactions evaluated were:

- The Sabatier reaction
- The Sabatier reaction with methane decomposition
- The Bosch reaction
- The molten electrolyte cell
- The solid electrolyte cell
- Thermal decomposition

The first three are indirect conversion processes, and the last three are direct.

Table X shows a comparison of the processes including basic reaction data. Table XI gives a general summary of the advantages and disadvantages of each subsystem type.

a. Sabatier reaction: The basic Sabatier cycle, with ideal flow rates, is shown schematically in Figure 20. CO_2 is reduced by hydrogenation over a catalyst of nickel deposited on Kieselguhr to form methane and water. Note that no carbon is produced in this reaction. The water is then electrolyzed to generate oxygen and produce more hydrogen to support the reactor. The methane can be dumped overboard or used for the incineration of various waste materials. As can be seen from Figure 20, the electrolysis of water produces only one-half of the hydrogen required for the

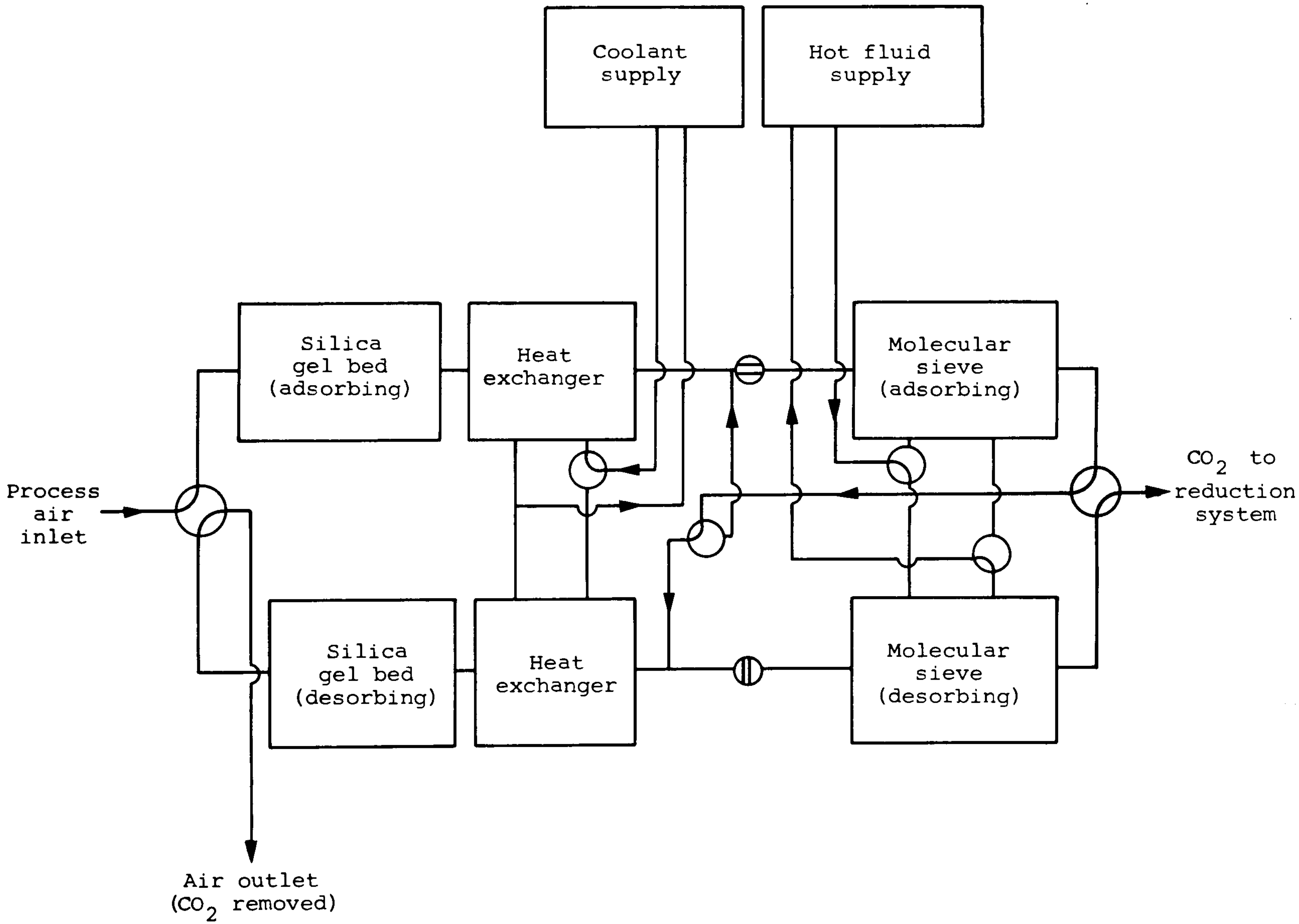


Figure 19 Thermally Desorbed Four-Bed Molecular Sieve System

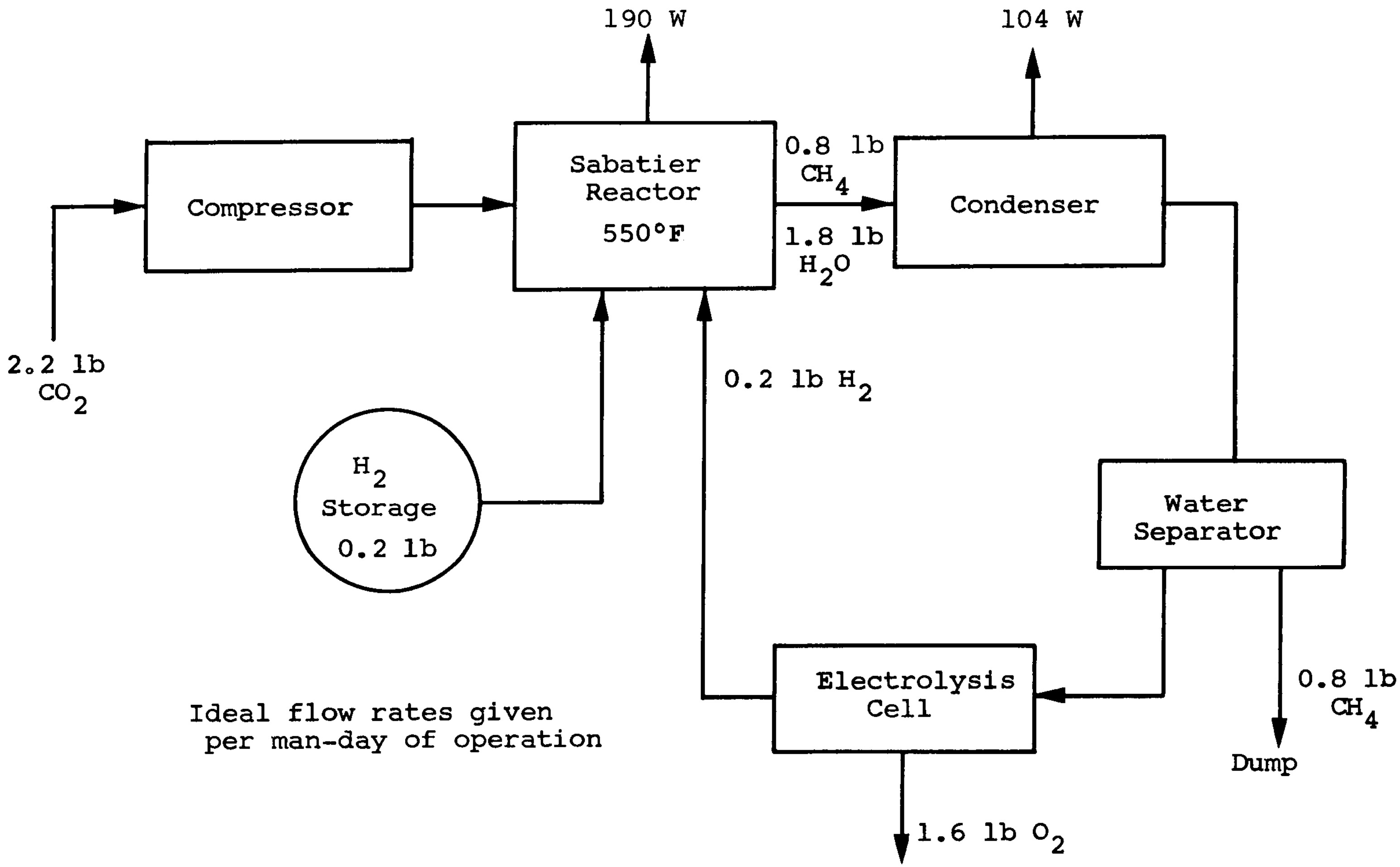


Figure 20 Sabatier System

TABLE X

COMPARISON OF CARBON DIOXIDE REDUCTION PROCESSES

[From ref. 23]

Subsystem	Volume, ft^3	Subsystem Weight, lb	Expendable Weight, lb/day	Power, W
Bosch				
1. Tapco[a] (includes electrolysis)	3.3	100	1.64 H_2O	1205
2. MRD[b] (no electrolysis included)		125	1.64 H_2O	500
Sabatier				
1. MRD	2.2	54	0.64 H_2 2.00 H_2O	75
2. Sabatier reaction only	0.6	8		0
3. Sabatier system estimate, including acetylene formation from CH_4		239	1.64 H_2O	2265
Solid electrolyte				
1. Estimate		194	0.23 (catalyst) 1.45 O_2	1275
2. Isomet[c] system	36	200	0.14 (catalyst)	3000
Molten electrolyte				
Lithium carbonate system	8	200	1.45 O_2	1500

[a]Division of Thompson Ramo Wooldridge
[b]Mechanics Research Division of General American Transportation Company
[c]Isomet Corporation

TABLE XI

ADVANTAGES AND DISADVANTAGES OF CARBON DIOXIDE PROCESSES

Process	Advantages	Disadvantages
Sabatier	No carbon deposited in reactor. Self-sustaining low temperature reaction. Very high conversion efficiency. Relatively low component weight.	Make-up hydrogen required.
Sabatier with methane decomposition	Closed loop system.	Carbon deposited in decomposition reactor. Material problem in decomposition reactor. Difficult to prevent carbon buildup in Sabatier reactor.
Bosch	One step process. Exothermic reaction.	Carbon deposited in reactor. Low conversion efficiency due to number of probable reactions. Recycle gas drying.
Molten electrolyte	No electrolysis. Direct conversion.	Carbon deposited on cathode. High power required. Low conversion efficiency.
Solid electrolyte	No electrolysis. Oxygen produced in single step.	Carbon deposited in reactor. High power required. High electrode contact resistance.
Thermal decomposition	No electrolysis. Oxygen produced in single step.	Extremely high temperature required. Carbon deposited in reactor. Low conversion efficiency. Material problems in reactor.

Sabatier reactor, hence, storage of the other one-half is required. The Bosch reaction is exothermic and takes place at 550°F. It is now well established that nickel catalysts promote the reaction at efficiencies as high as 100 percent if the temperature is maintained at this level. Figure 21 shows the variation in the amount of CO_2 reacted with the temperature of the reaction. The Sabatier reaction gives off 1770 Btu per lb of carbon dioxide processed and is self-supporting.

b. Sabatier with methane decomposition: A candidate reaction cycle is shown in Figure 22 which makes use of the methane that would be dumped by use of the pure Sabatier transfer. The methane is decomposed by a high temperature process described as methane "cracking." The hydrogen obtained is used to fulfill the total requirements of the Sabatier reactor, thus forming a closed loop system. On the surface, this apprears to solve the major disadvantage of the pure Sabatier reaction, but there is a new problem of carbon build-up in the reactor which reduces the "cracking" efficiency. Catalyst activity deteriorates rapidly as the carbon-to-catalyst ratio increases above a value of 2:1. In addition to this, the added power to perform the cracking operation is quite high.

If the carbon were to remain in the cracking reactor (100% of CH_4 is reacted), then the build-up could be handled more effectively since the primary Sabatier reaction would not be affected. However, the methane cracking will not be complete, and some methane will thus be carried over to the Sabatier reactor which will then reduce the conversion efficiency of the Sabatier reactor, and subsequently through the following reactions,

$$CO_2 + CH_4 \rightarrow 2\,CO + 2H_2$$

$$2CO + 2H_2 \rightarrow CO_2 + C + 2H_2$$

carbon will be deposited in the Sabatier reactor. This reactor is highly sensitive to carbon build-up on the catalyst, and plugging occurs at a carbon-to-catalyst ratio of about 0.5. Another item of concern is the material problem surrounding the cracking reactor. Recent tests have shown that metal alloys such as columbium, titanium, Hastelloy and Waspaloy are degraded at 2000°F when subjected to the environment of methane pyrolysis (ref. 12). This suggests the possibility that a ceramic material may have to be used in the reaction zone, resulting in a very severe weight penalty.

There are various other schemes suggested for accomplishing the methane decomposition. One scheme, by Isomet Corporation, forms acetylene (C_2H_2) and H_2 in an electric arc furnace. The H_2 is separated by a palladium filter and is recycled to the Sabatier reactor. Carbon formed in the arc process presents a handling problem and also tends to short out the arc. Power consumption was estimated to be 400 watts, which is quite high. In addition, the system has not been developed to any extent.

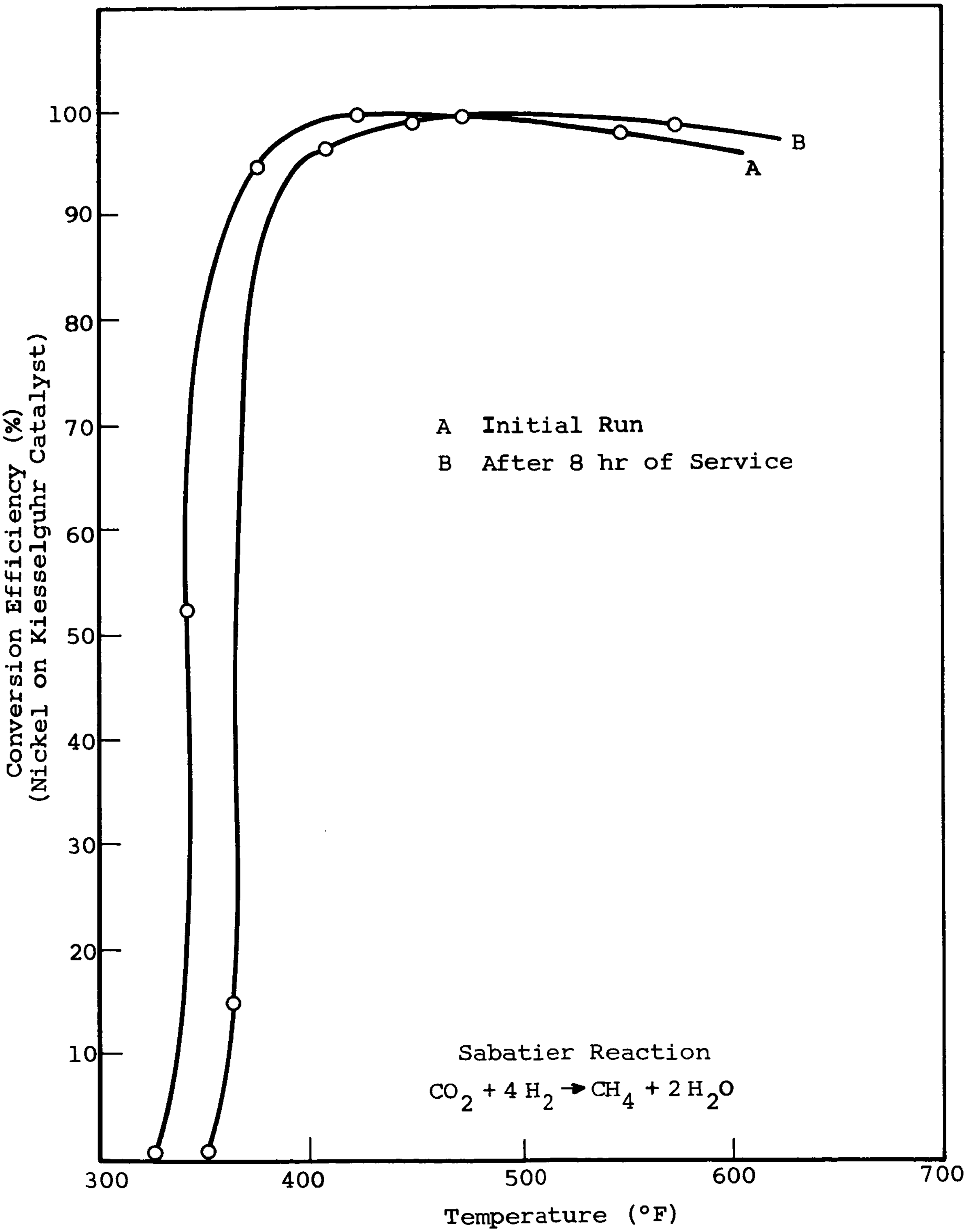


Figure 21 Sabatier Reaction Efficiency
(From ref. 12)

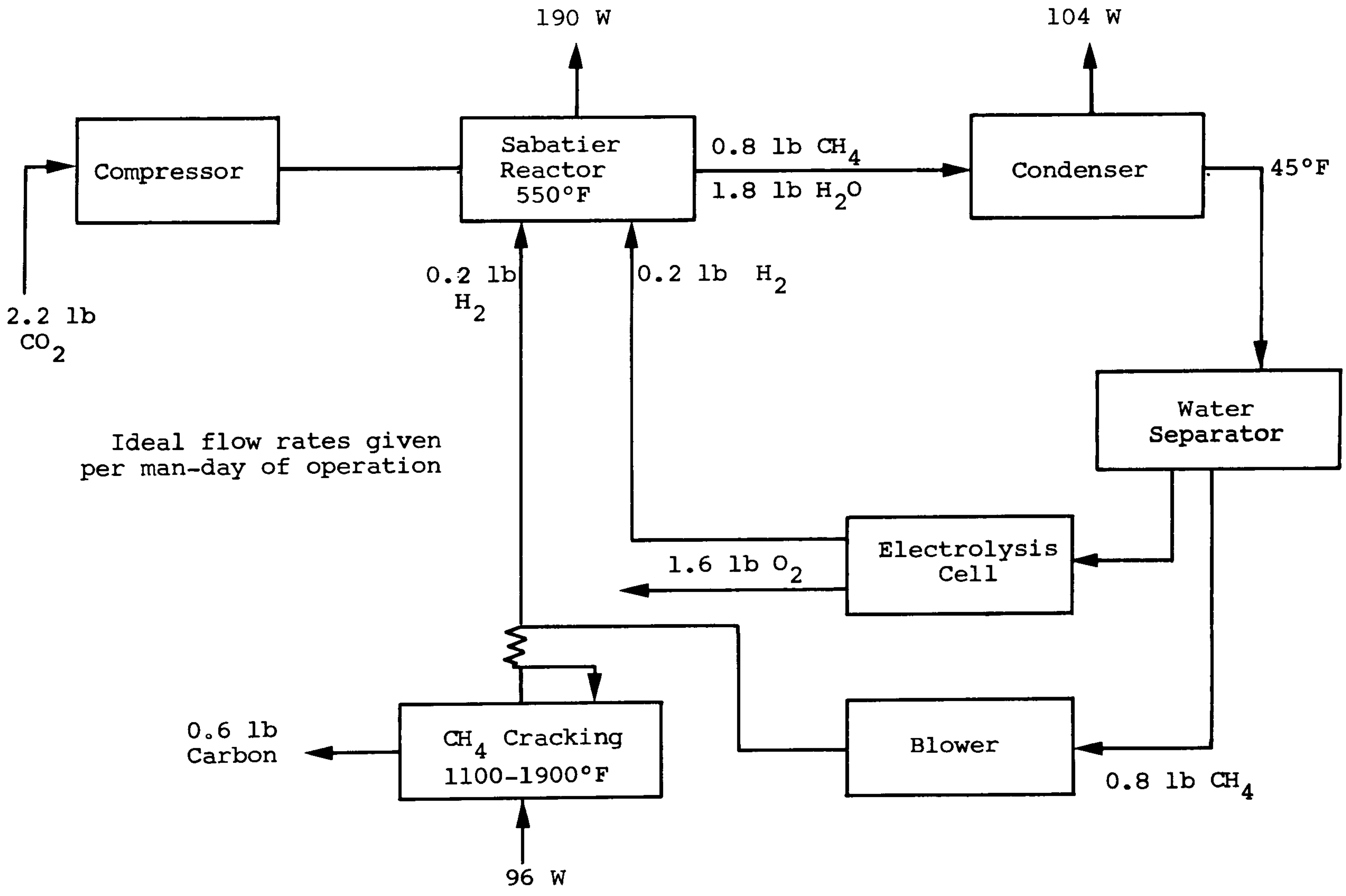


Figure 22 Sabatier System with Methane Decomposition

This method has all the disadvantages of the methane cracking procedure just described, plus the fact that material balance is not achieved since hydrogen is expelled in the form of acetylene.

Another method suggests reacting the by-products of the Sabatier reactor (after cooling and water separation) with CO_2 in a catalytic reactor to produce carbon, water and CO_2. By processing part of the CO_2 in the methanation reactor and CO_2-CH_4 reactor, the load on the methanation reactor is reduced. The reaction by-product with the water removed, may be recirculated into the CO_2-CH_4 reactor to obtain higher efficiencies. This method also has the same carbon removal and catalyst consumption problem, and again defeats the primary objective of obtaining a material balance.

It, therefore, appears that the methane cracking method is preferable to the formation of acetylene or further reaction of the Sabatier by-products with CO_2.

c. Bosch reaction: The one-step Bosch cycle is shown in Figure 23. The complete reduction of CO_2 to C and H_2O is unfortunately not achieved in one step. Several other reactions take place in the reactor, namely:

$$CO_2 + 4H_2 \rightarrow CH_4 + 2H_2O$$

$$CO_2 + H_2 \rightarrow CO + H_2O$$

$$CO + 3H_2 \rightarrow CH_4 + H_2O$$

The above reactions may represent from 30 to 98 percent of the recycle load necessary for complete conversion. It is also known that the water formation reactions proceed at low efficiency if methane is not present. Thus, the one-pass conversion efficiencies are low, and recycling of the product gases after removal of water is necessary. Also, the recycle gases must be dried to low dew points since the Bosch reactor is very sensitive to the presence of water in the inlet stream. The basic reaction is exothermic at approximately 27 W/man and a temperature of 1100°F.

The removal of carbon is currently the major problem confronting designers of systems operating on this cycle. Advanced methods of handling carbon removal make use of iron catalysts in the form of rotating disks which after carbon collection are then passed by stationary scrapers. The dislodged carbon is then blown downstream by the recycle gas flow to a regenerable stainless steel filter. Another advanced method attempts to blow carbon off the catalyst, charge it electrostatically, and then hold the carbon in the reactor by means of an electromagnet (ref. 23).

Carbon-to-catalyst ratios on the order of 20 to 1 have been reported when the catalyst was present in the form of steel wool.

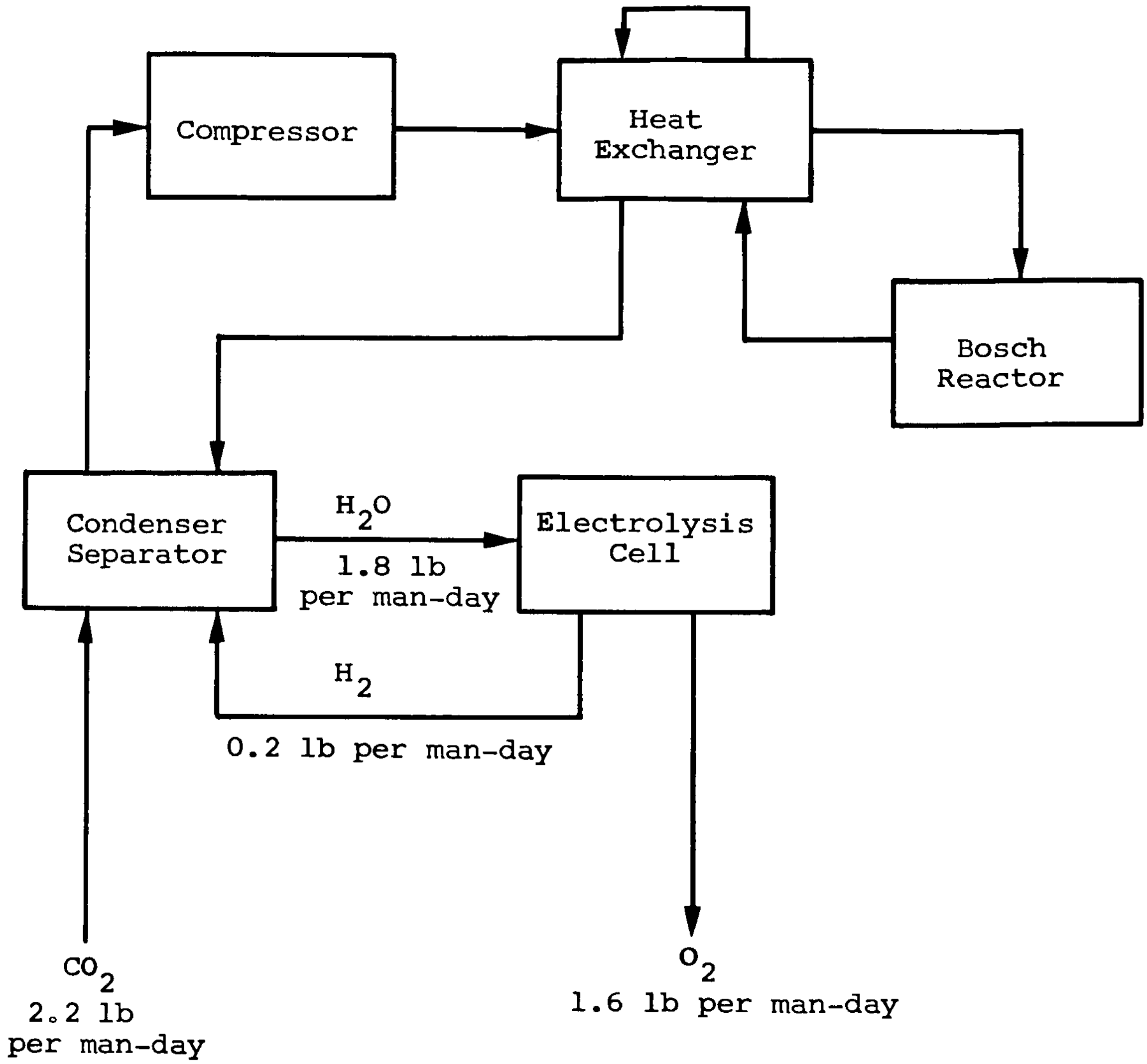


Figure 23 Bosch System (Ideal Flow Rates)

It seems reasonable to conclude that for missions of 6 months duration, several changes of the catalysts will have to be made, and will probably require shutdown of the Bosch reactor.

d. Molten carbonate cell: The molten electrolyte systems are in a very early stage of development and currently are being demonstrated in the laboratory only. The basic process is shown in Figure 24. Carbon dioxide is introduced into an electrolytic cell containing a eutectic mixture of LiCl and Li_2CO_3 at about 940°F; oxygen is collected at the anode and carbon is deposited on the cathode. A major problem is the removal of the hard carbon from the cathode, which could necessitate the periodic replacement of the complete cathode element.

The primary advantage of the system is that oxygen is recovered in a direct single step. The penalty we pay is that a substantial power input (530 W/man) is required and the electrolytic efficiency is only about 20 percent. Either electrical or thermal power is also required to maintain the mixture at the eutectic temperature. The problems of mixing and separation in a zero -g field were not considered for the Mars surface application.

e. Solid electrolyte cell: This system, too, is in a very early stage of development and hence, for purposes of comparison, must be projected operationally to the 1980 time period. The basic electrolysis reaction is

$$2CO_2 \xrightarrow{e} 2CO + O_2$$

A solid electrolyte such as ZrO_2 - Y_2O_3 is impermeable to molecular gas flow and thus, the electrolyte acts like a charged catalyst relative to the above reaction.

An operating temperature of about 1800°F is required to permit oxygen migration (ref. 12).

The advantage of this type of system is that oxygen is produced in a single step without the need for electrolysis. This reaction is unlike the molten electrolyte in that half of the oxygen produced appears in the form of carbon monoxide. Thus, in order to recover all of the oxygen available, the carbon monoxide can be converted to oxygen and carbon dioxide by the use of a catalytic reaction. If the reactor is not included, oxygen lost in the form of CO would result in a large weight penalty. The disadvantages of using this cycle can now be shown through the use of Figure 25.

With the reactor now included, the old problems of carbon removal and catalyst consumption are back again. The system now has acquired many of the disadvantages of the Bosch cycle or the Sabatier cycle with methane decomposition.

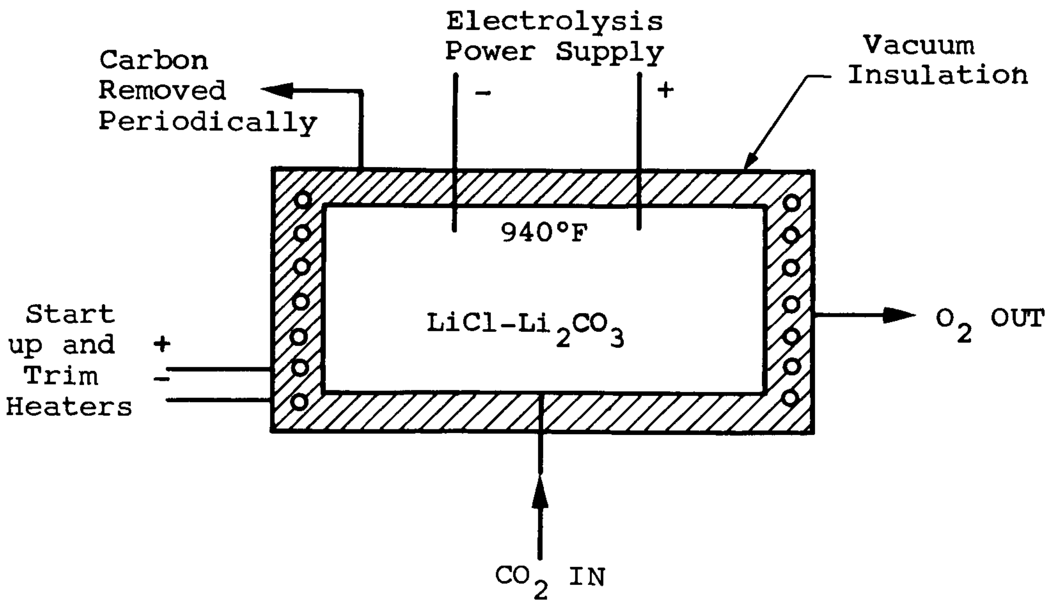


Figure 24 The Molten Carbonate Cell
(From ref. 22)

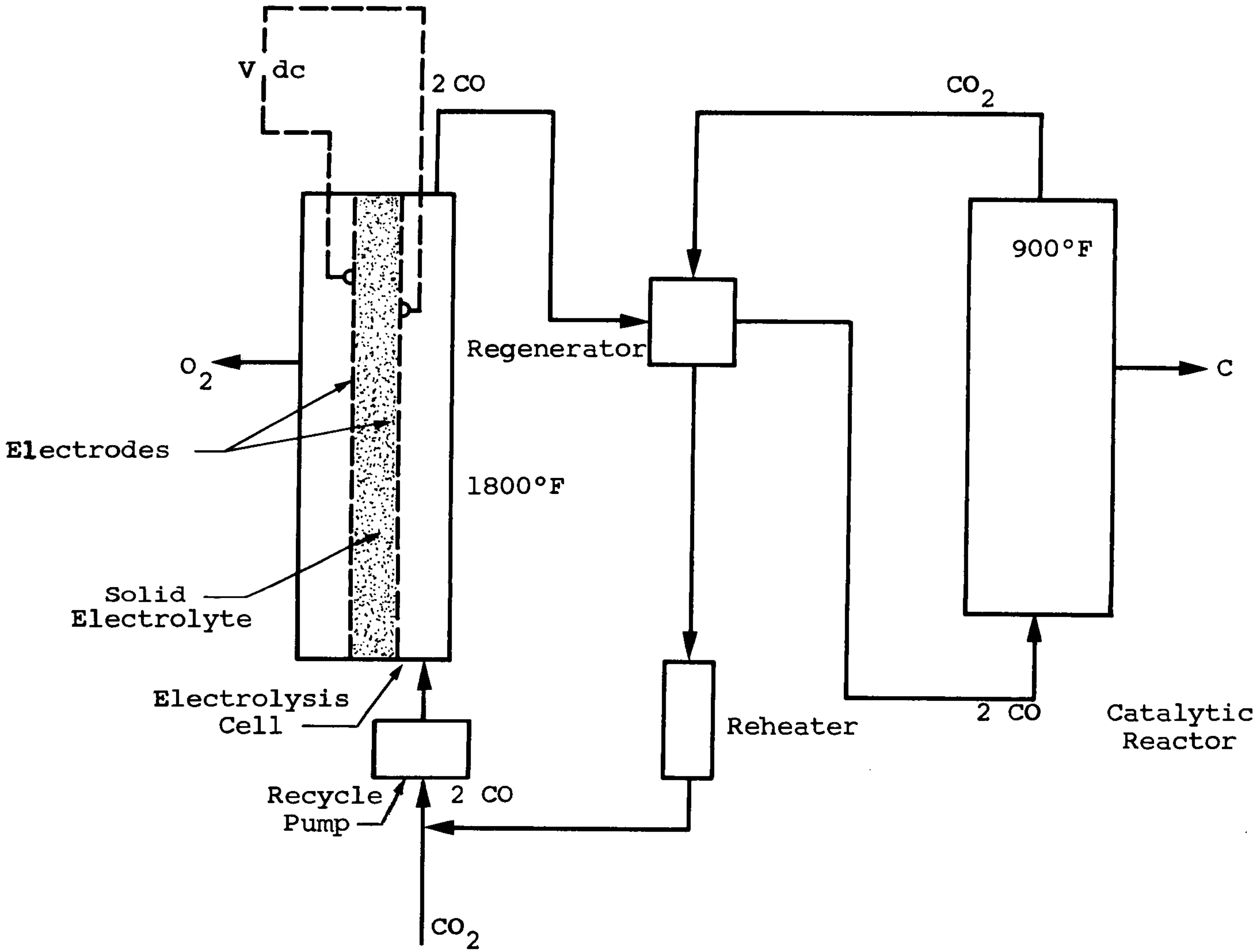


Figure 25 The Solid Electrolyte Cell
(From ref. 12)

The system requires a high power level both for maintaining the high temperature electrolytic cell and the necessary dc charge.

Another very critical problem is that of improving contact between the electrodes and the electrolyte to reduce the electrical resistance of the cell.

f. Thermal decomposition: The molecular structure of carbon dioxide has a very high thermal stability. As a result, direct thermal decomposition of carbon dioxide into carbon and oxygen requires about 3900 Btu/lb of CO_2 at a temperature of 6760°F.

Figure 26 shows that even at a temperature of 10,000°F, only 20% of the CO_2 is decomposed and this goes into carbon monoxide and oxygen rather than carbon and oxygen. Thus, we are forced to add an additional reactor as was done in the case of the solid electrolyte to convert carbon monoxide to carbon and carbon dioxide. The flow diagram for this system is shown in Figure 27. It can be seen that the problems of the solid electrolyte system are derived again, namely, carbon deposit and catalyst consumption. Refractory metals, such as Mo, Ta and W, would be oxidized in the decomposition process, thus forcing us to use ceramic material. This excessive weight penalty, along with the associated high temperature and power required, makes this system very undesirable for our application.

g. Concluding remarks: Although the three direct processes for CO_2 reduction discussed are each in a very early stage of development, analysis of these systems shows that the main problem of excessive power consumption will probably not be solved in time for the manned Mars surface mission.

The three indirect processes compared in Table XII, will be highly competitive for most manned missions within this time period.

The Bosch system and the Sabatier system with methane cracking, both have the problem of carbon removal, which will probably necessitate scheduled shutdown periods. In addition to this, they have a larger component weight, volume, and power requirement than the open loop Sabatier process.

Because of the low weight of hydrogen gas needed to balance the cycle, it is not felt that the storage of the amount required for the six month mission, poses a serious restriction on the use of the pure Sabatier cycle. Continuous, highly efficient, low power operation can be obtained with excellent reliability for the duration of the mission. Also, the methane can be used for the incineration of waste material before dumping. For these reasons, the Sabatier cycle without methane decomposition was chosen for the Mars surface mission.

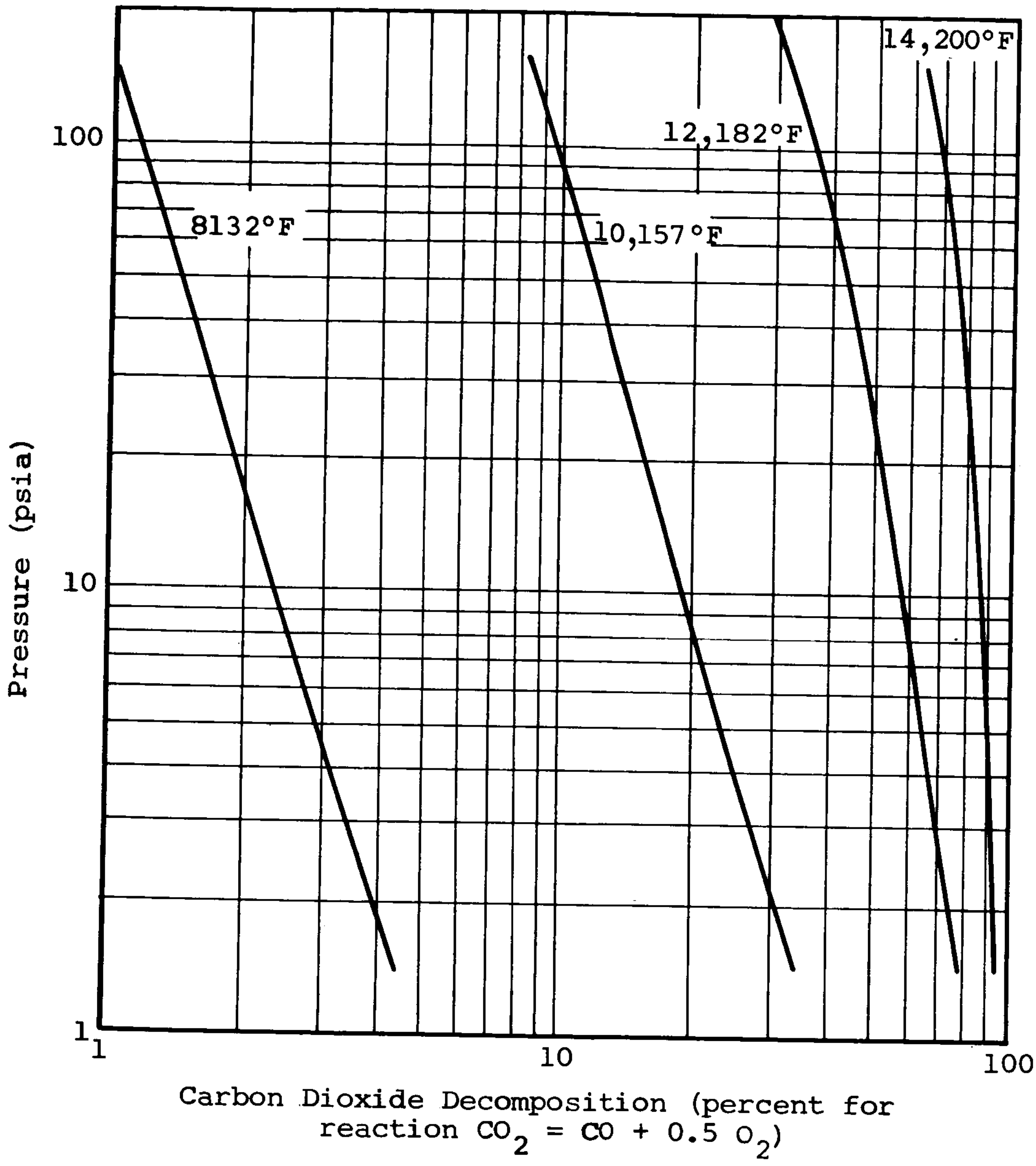


Figure 26 Equilibrium Carbon Dioxide Decomposition (From ref. 12)

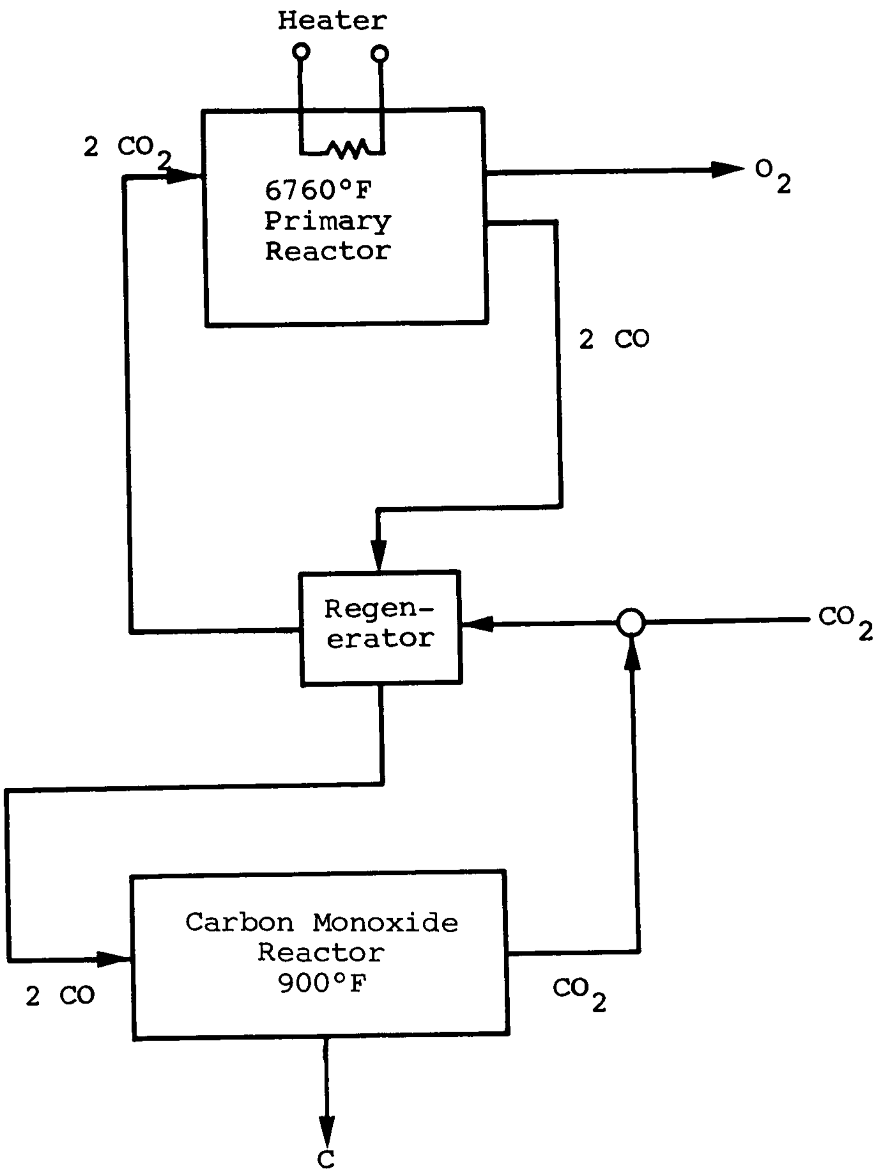


Figure 27 Thermal Decomposition

TABLE XII

COMPARISON OF INDIRECT METHODS OF CARBON DIOXIDE REDUCTION AND OXYGEN RECOVERY[a]

Parameters	Sabatier	Sabatier with Methane Cracking	Bosch
Hardware weight, lb	54.0	80.0	100.0
Hardware volume, ft^3.	2.2	3.0	3.3
Input power, W	1000.0	1100.0	1205.0
Heat load to temperature control loop coolant, Btu/hr . .	520.0	630.0	530.0
Heat load to module atmosphere, Btu/hr	800.0	1000.0	810.0
CO_2 reaction, %	98.0	65.0	30.0
Methane cracking efficiency, %		60.0	
Hydrogen requirements, lb/man-day.	0.8	Balance	Balance

[a]Based on four-man crew CO_2 reduction rate of 8.8 lb/day.

B. Environmental Control System

1. Summary. - The environmental control system primarily involves temperature control and moisture control. Secondary requirements are for odor and contaminant control (see section 5.D.9.). The selection of an adequate thermal control system initially requires that a reasonably accurate heat balance be performed for the module, taking into account such factors as the heat transfer through the module wall and the heat given off interior to the module by the various life-support components. Moisture control will be accomplished as part of the thermal control system.

2. External heat load. - The heat, Q, dissipated from the interior of the module can be defined by the following general formula:

$$Q = UA\Delta T$$

where U is defined as the overall coefficient of heat transfer, or heat transferred per unit time per unit area, per degree. Thus, the heat transferred through any specific wall configuration is completely defined once the interior and exterior design temperatures are selected. To be completely rigorous, the wall is defined as an infinitely flat composite structure. The wall can be made up of several different materials with different values of conductivity. If the material thicknesses are given by x_1, x_2, x_3, etc., and the conductivities k_1, k_2, k_3, etc., U is then defined by the following general expression:

$$U = \frac{1}{\frac{1}{f_i} + \frac{x_1}{k_1} + \frac{x_2}{k_2} + \cdots + \frac{1}{f_o}}$$

The terms f_o and f_i are the surface conductances of the inside and outside surfaces respectively, and are in general, a function of surface roughness, wind velocity and mean wall temperature. The terms in the denominator, $1/f_i$ and $1/f_o$, produce very small numbers compared to the primary insulation terms x_1/k_1 and x_2/k_2 and, for all practical purposes, can be neglected. Also, for high wind velocities as is anticipated on Mars, the surface conductance terms are much smaller, due to a thinner gaseous film layer. Thus, U can be simply written as:

$$U = \frac{1}{\frac{x_1}{k_1} + \frac{x_2}{k_2}}$$

There is no appreciable temperature drop across the three aluminum shell layers shown in Figure 28, due to the high conductivity and small thickness of aluminum. These terms were also neglected in the above formula. The major and minor diameters of the proposed ellipsoidal module configuration are

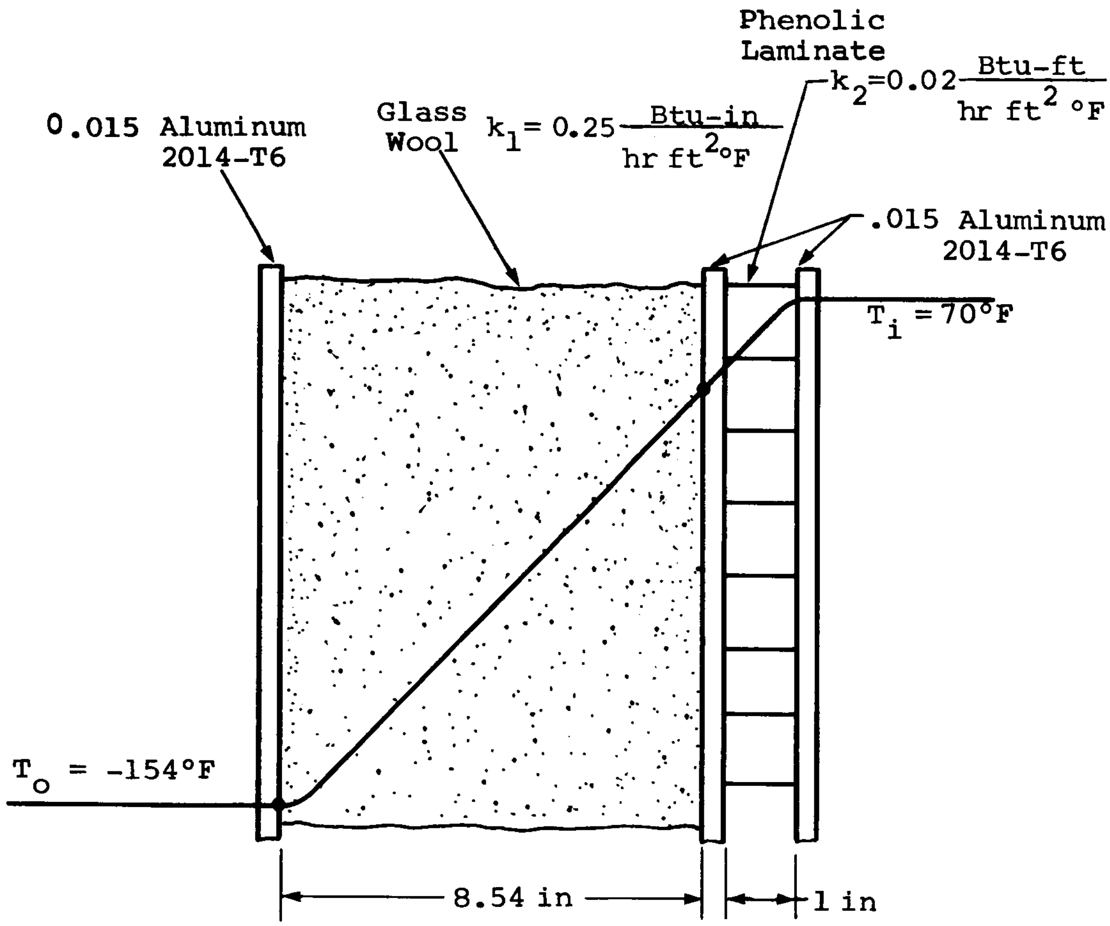


Figure 28 Module Wall Section Design Conditions

very large as compared to the wall thickness. Thus, plane wall transfer can be assumed with the inclusion of only a slight error. The total surface area was obtained in the appendix calculations.

The design ΔT considers the minimum equatorial temperature estimate (ref. 1).

$$Q = \frac{1}{\frac{8.54}{0.25} + \frac{1}{12(0.02)}} A\Delta T$$

$$Q = 0.0261(1186)(224)$$

$$Q = 6930 \text{ Btu/hr}$$

This figure represent the maximum transfer through the module wall. The daily heat transfer fluctuation is shown in Figure 29.

3. Interior heat load. - For complete heat balance, the heat given off by the processes, personnel and other items interior to the module, must be considered. These elements are summarized in Table XIII. It can be seen that they contribute about 14,885 Btu/hr on the average to the interior of the module.

4. Thermal control requirements. - From this gross analysis, it can be seen that the thermal control system must remove heat from the interior to maintain the 70°F nominal temperature level. Thus, the worst condition for cooling occurs when the shelter is dissipating the least amount of heat, or in other words, when the temperature outside the module reaches a maximum. This occurs at approximately 300°K (80°F) at this temperature 310 Btu/hr add to the interior loading. Thus, the cooling loop must be able to handle at least 15,195 Btu/hr.

5. Thermal control system. - It has been shown through the heat balance calculations that the basic function of the thermal control system will always be to cool the module, the highest cooling condition occurring when the external temperature reaches 80°F. The most moderate cooling condition occurs when the outside temperature reaches -150°F at night. This suggests that the basic cooling system should neither be completely active nor passive, but rather employ both an active peak cycle transfer loop and a passive external loop, taking advantage of the low external temperature at night. Hence, the active system would only have to operate while the outside temperature was above approximately 20°F, or about 8 hours out of the 24-hour day. Transfer could then be made to an external radiator-convector system requiring only pump power. The basic configuration would be as shown in Figure 30 with valves A and B set for time-cycle operation corresponding to an external temperature of 20°F. The module cooling would be accomplished by means of a heat exchanger in the primary module air circuit. This is the ideal transfer point since this air should be cooled before entry into the CO_2 concentration unit for maximum efficiency.

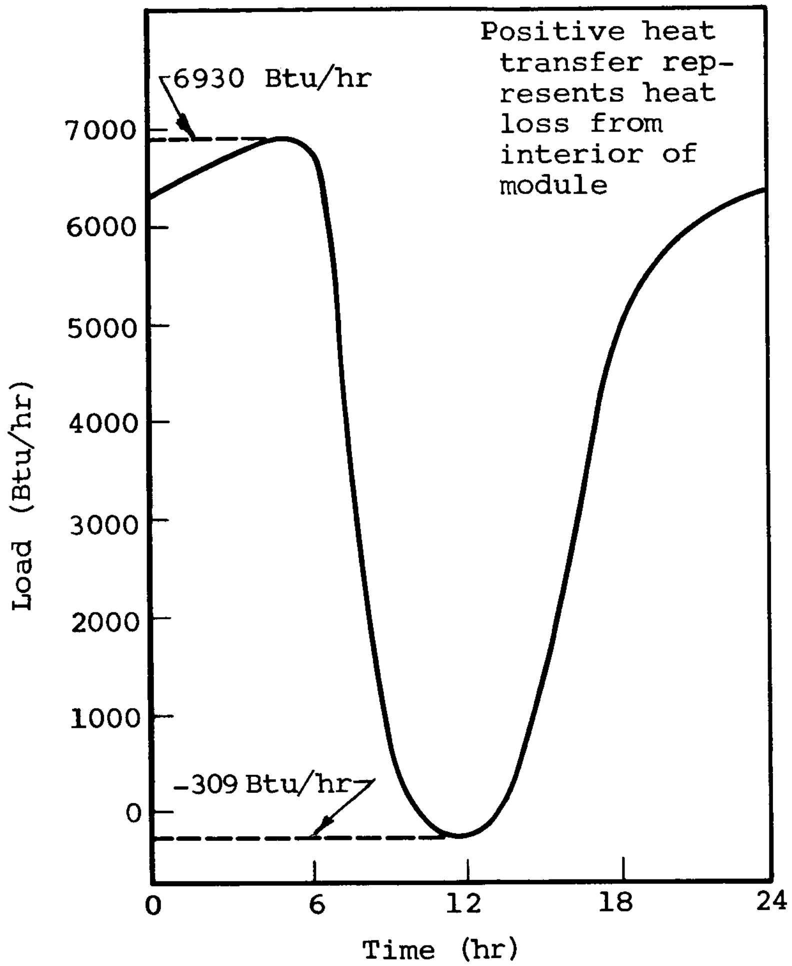


Figure 29 Daily Heat Transfer through Module Wall

TABLE XIII

MODULE INTERIOR HEAT LOADING

Element	Heat Load, Btu/hr
CO_2 Concentration	5,083
CO_2 Reduction - O_2 Generation	1,333
Contaminant Control	273
Atmospheric Circulation	1,020
Thermal Control	1,360
Wash Water Recovery	48
Metabolic & Urine Water Recovery . .	185
Feces Processing	584
Food Preparation	144
Hygiene	205
Equipment and Lighting	2,530
Instrumentation	290
Personnel (4)	1,830
TOTAL	14,885

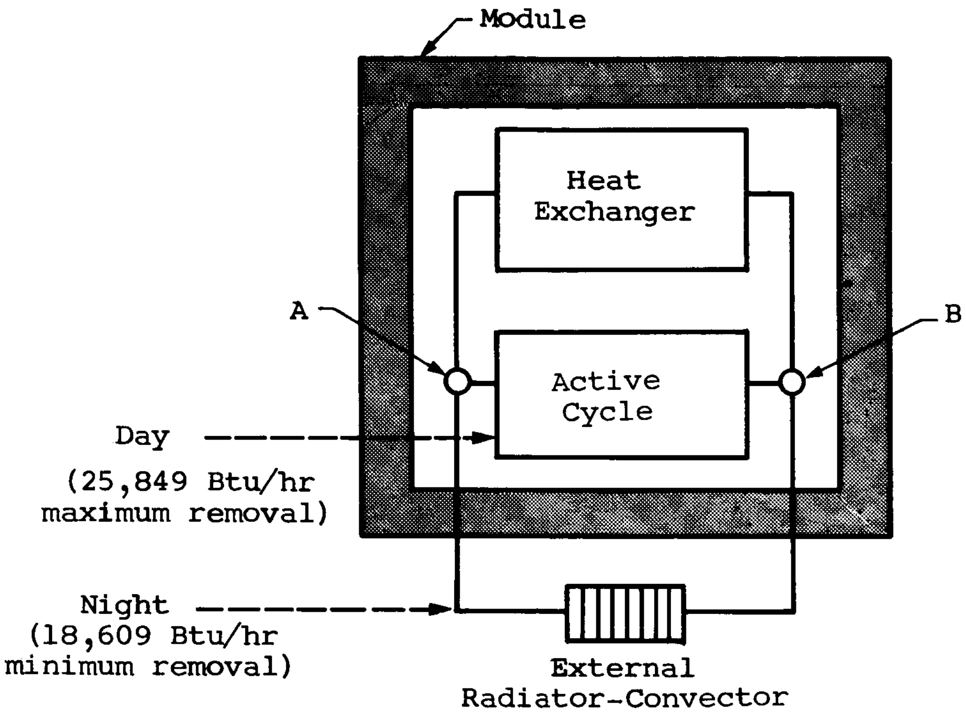


Figure 30 Module Cooling Concept

The air temperature leaving the module air-heat exchanger must be kept above 32°F to prevent the condensate from freezing in the heat exchanger or the water separator unit. The heat exchanger thus acts as the dehumidifier in the respiration-perspiration metabolic water regeneration cycle shown in Figure 31.

It is desirable to keep the exit air temperature below 50°F to prevent excessively moist air from entering the CO_2 concentration unit since the first step is one of drying under passage through silica-gel beds.

The basic active-refrigeration cycle is shown in Figure 32. The cycle is made up of four steps: compressor work, condensation, expansion, and evaporation. Evaporation takes place in the primary air heat exchange unit and condensation is at atmospheric temperature. Freon 11 is a common fluid used in this cycle with $T_d \cong 150°F$.

The system to be used in conjunction with the active cycle is shown in Figure 33. The velocity of the fluid (which should ideally be Freon 11 to reduce the complexity of the primary air heat exchanger) must be varied to compensate for the variation in external temperature and prevent an excessively low exit air temperature. The final semi-active vapor-liquid cycle is shown in Figure 34.

The atmospheric condenser should be located external to the module to reduce the heat loading and take advantage of the high wind velocity and low temperature. This same external exchanger could then be used for passive cooling.

Figure 35 shows the diurnal cooling profile to be handled by the thermal control system. Coolant is also needed in areas other than the primary air heat exchanger as shown in Figure 36. Coolant loops should be provided for the Sabatier reactor, the Sabatier cycle condenser, the condensers used in the two water purification cycles, the silica-gel after coolers, and drinking water. A summary of the equipment cooling requirement is shown in Table XIV, and amounts to an additional 10,654 Btu/hr superimposed on the basic cooling requirements indicated in Figure 35. The final design profile shown in Figure 37 is the total required cooling load for the active portion of the thermal control loop.

The active portion of the thermal control system must be capable of handling approximately 2,000 Btu/hr at normal peak conditions. This amounts to 7.63 kW of thermal energy. The subsystem curves are presented in Figure 38. This shows a system weight of approximately 700 lb, a radiator area of 250 ft^2, and an electrical power input of 2.5 kW for a compressor outlet temperature of 150°F. This value of 2.5 kW represents the normal peak load condition for the active portion of the thermal control cycle throughout the Martian day. The passive mode of operation requires an average of only 114 W of pump power. This gives an

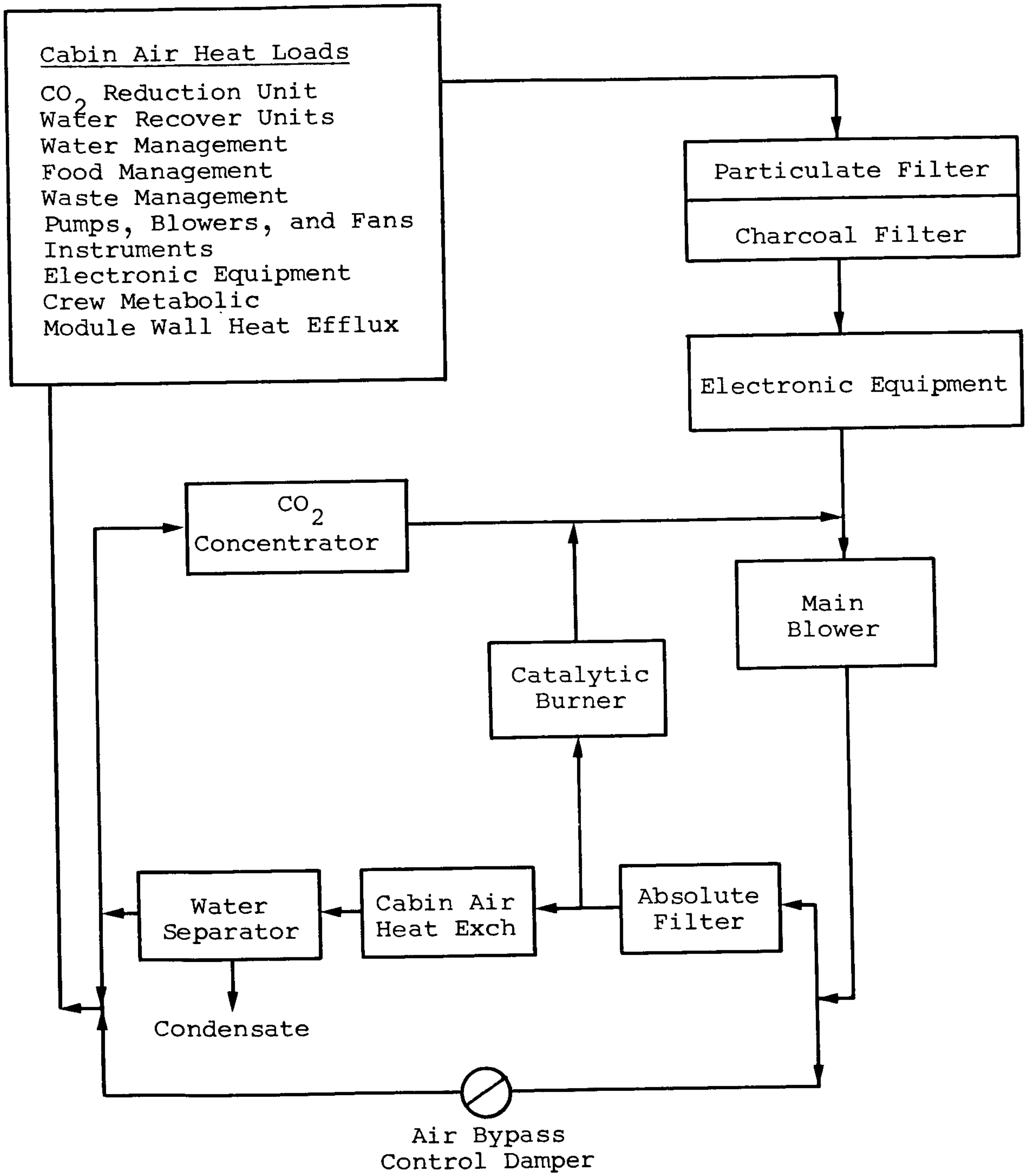
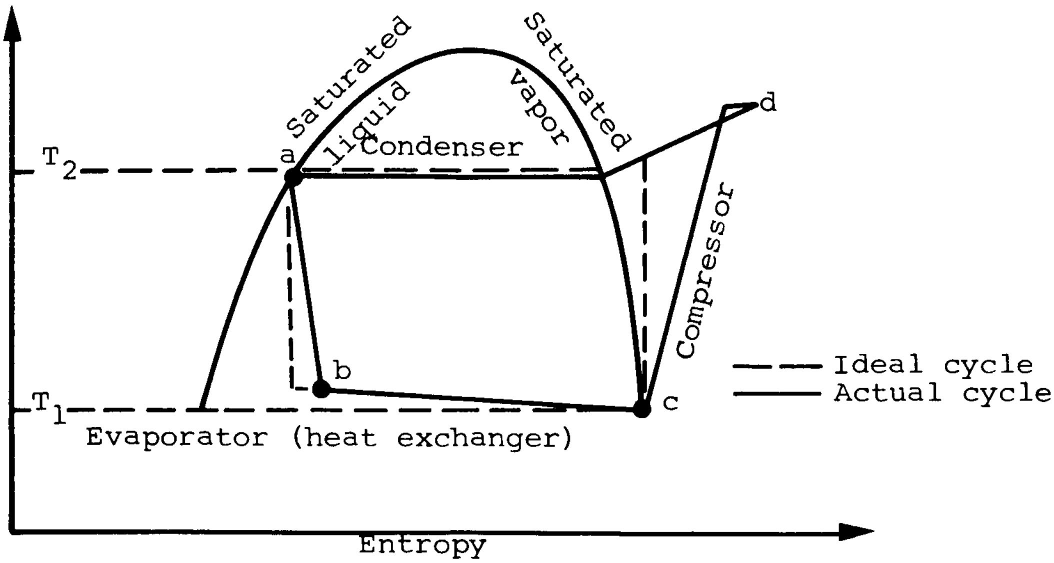
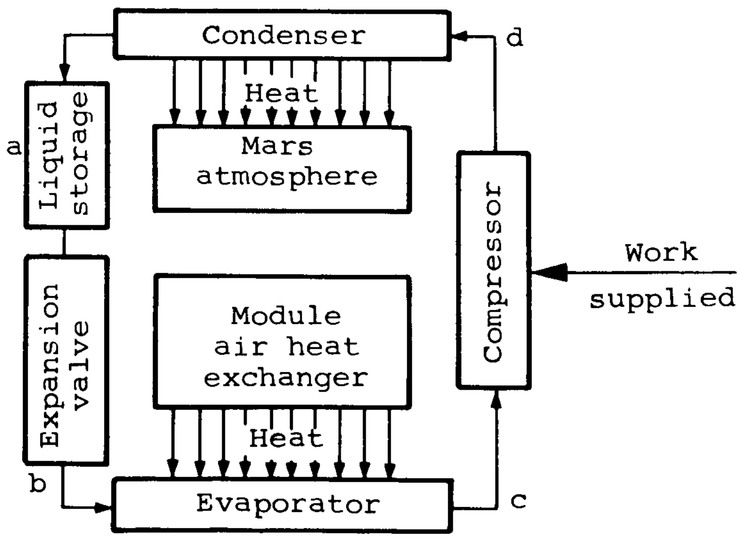


Figure 31 Thermal Control Air Circuit
(From ref. 23)



(a) Entropy as a function of temperature



(b) Elements of vapor-compression refrigeration cycle

Figure 32 Vapor-Compression Refrigeration Cycle
(From ref. 23)

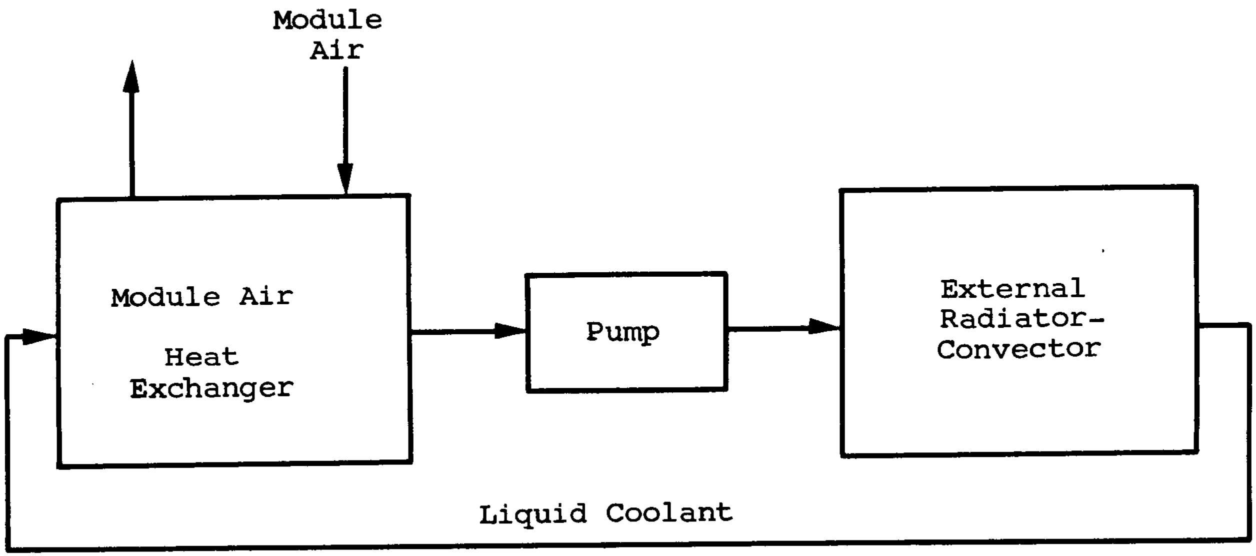


Figure 33 Passive Liquid Loop

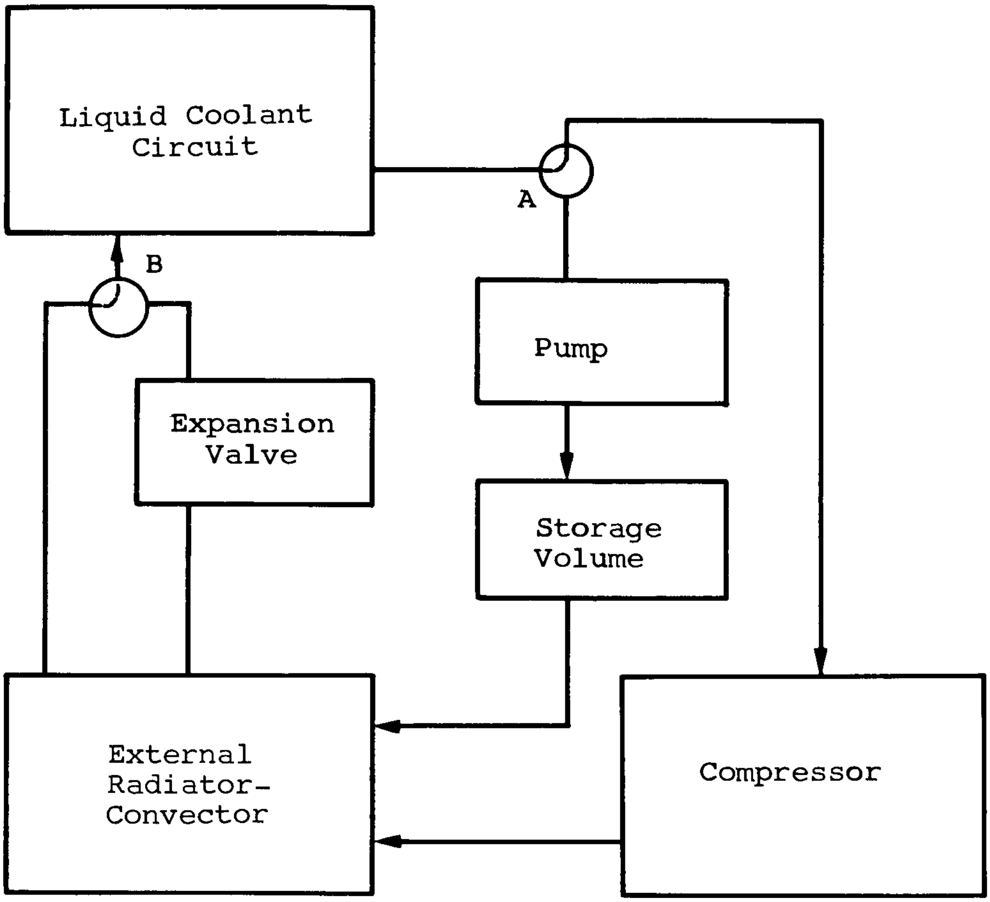


Figure 34 Semi-active Vapor Cycle

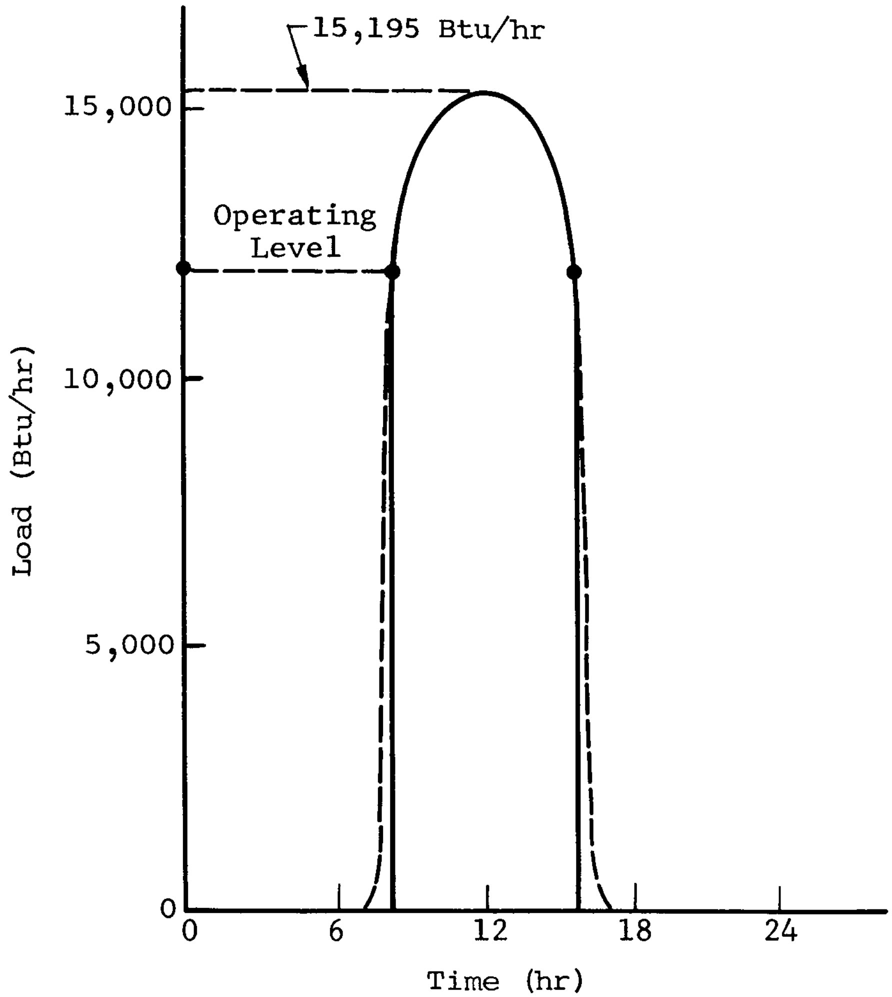


Figure 35 Active Cycle Capacity Needed for Heat Exchanger

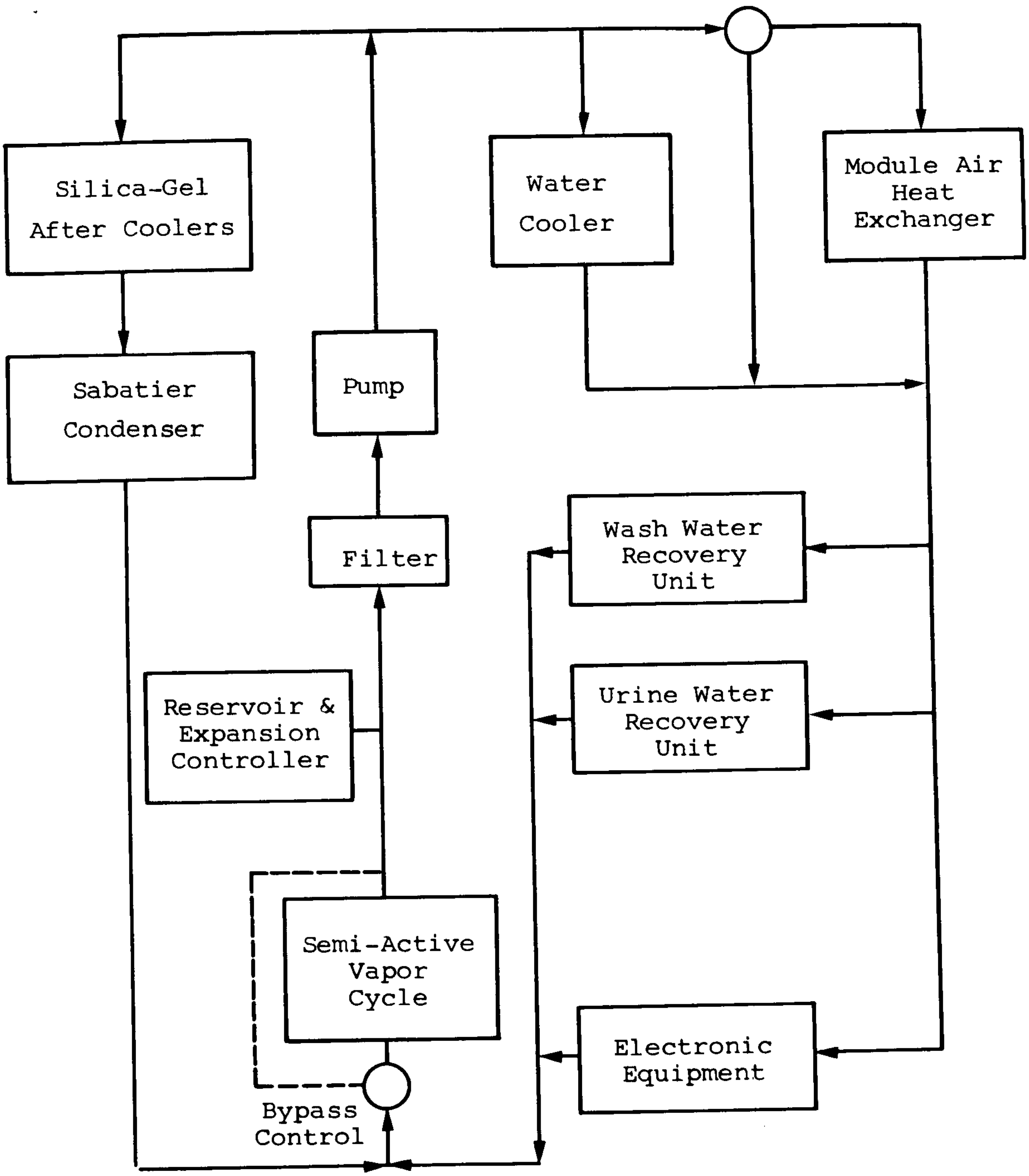


Figure 36 Liquid Coolant (Flow Diagram)

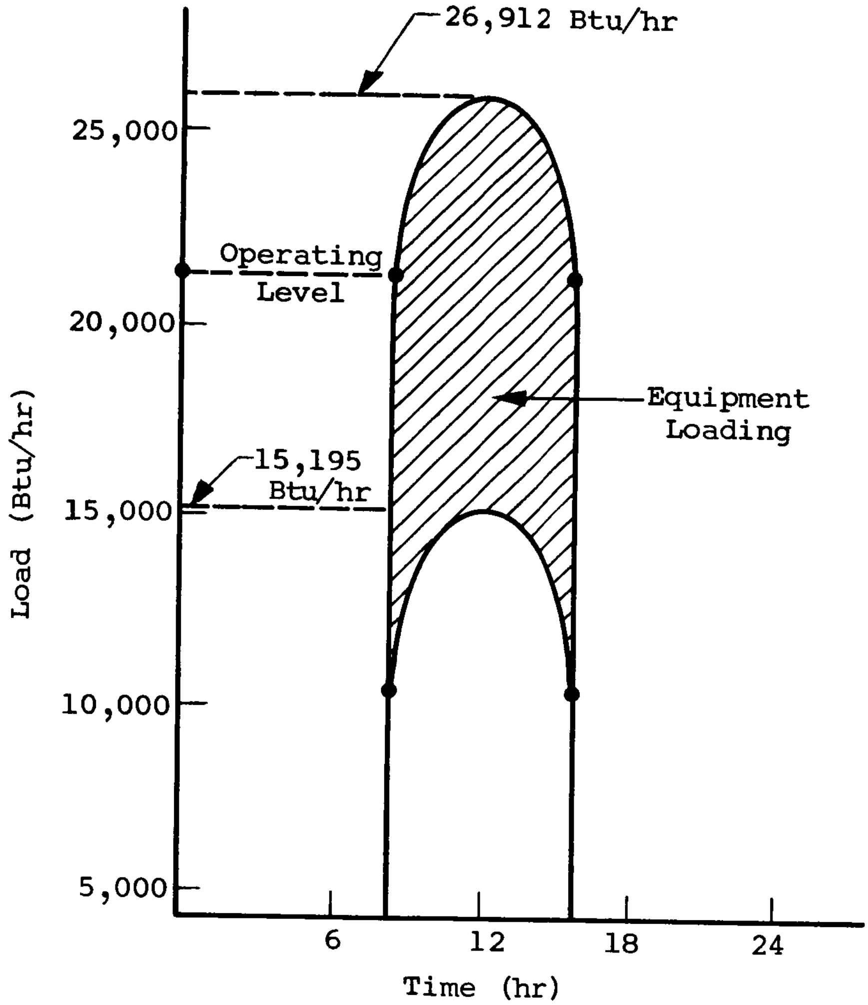


Figure 37 Total Active Cycle Capacity

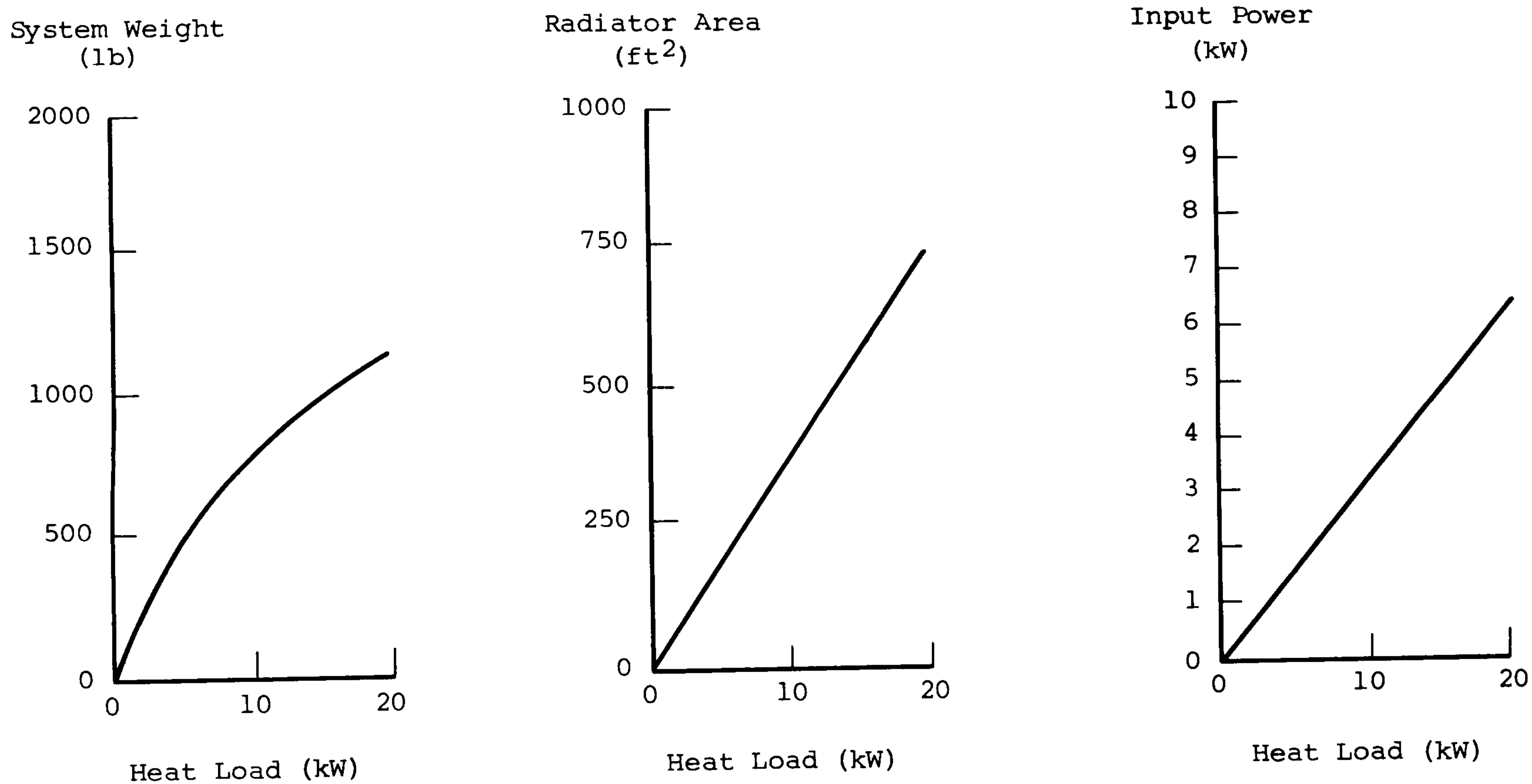


Figure 38 Active Cycle Characteristics
(From ref. 6)

TABLE XIV

EQUIPMENT COOLING REQUIREMENTS

Equipment	Heat Removal, Btu/hr
Sabatier reactor	648
Sabatier condenser	352
Condenser I (wash)	562
Condenser II (urine)	282
Silica-gel after coolers . .	4,510
Water cooler	30
Electronic equipment and lighting	4,270
TOTAL	10,654

overall average of 1.2 kW of power used for the purpose of thermal control which was obtained from the total thermal profile presented in Figure 39.

C. Power System

1. Thermoelectric conversion system. - Thermoelectric systems are now being used or considered for applications where power requirements are from a few watts to several kilowatts. Thermoelectric conversion devices can be used with either solar or nuclear energy sources, depending on factors such as cost, availability, solar intensity, and application. Long life, low cost, radiation resistance and advanced state of development are some of the major advantages of the thermoelectric system.

The thermoelectric (Peltier) effect has been known for many years. Until recently, the development of thermoelectric devices has been handicapped by the low mechanical strength and poor chemical stability of the thermoelectric materials. Through continuous development of semi-conductor materials, these problems have been solved. However, the overall efficiency of all state-of-the-art thermoelectric power systems remains significantly low. Of the thermoelectric materials being investigated, alloys of lead and tellurium (PbTe) have received the most attention. They have a comparatively high figure of merit, but are quite brittle and very sensitive to impurities. The figure of merit, Z, for thermoelectric elements is expressed as:

$$Z = \frac{S^2}{\rho k}$$

where

Z = figure of merit ($°C^{-1}$)

S = Seebeck voltage coefficient (V/°C)

ρ = electrical resistivity (ohm-cm)

k = thermal conductivity (W/cm-°C)

An alloy of germanium and silicon (GeSi) is a newer material and is receiving considerable attention for space application because of its higher mechanical strength, higher resistance to oxidation, very slight degradation and promise of higher operating temperature. The figure of merit for GeSi is, however, only one-half of that of PbTe alloys; however, the possibility of longer useful life and reduction of weight of waste heat radiators due to higher operating temperature might make GeSi alloys quite competitive in space applications.

Figures 40 and 41 show the figures of merit for PbTe and GeSi alloys respectively for various operating temperatures. A comparison between these and other materials for thermoelectric properties is given in Table XV.

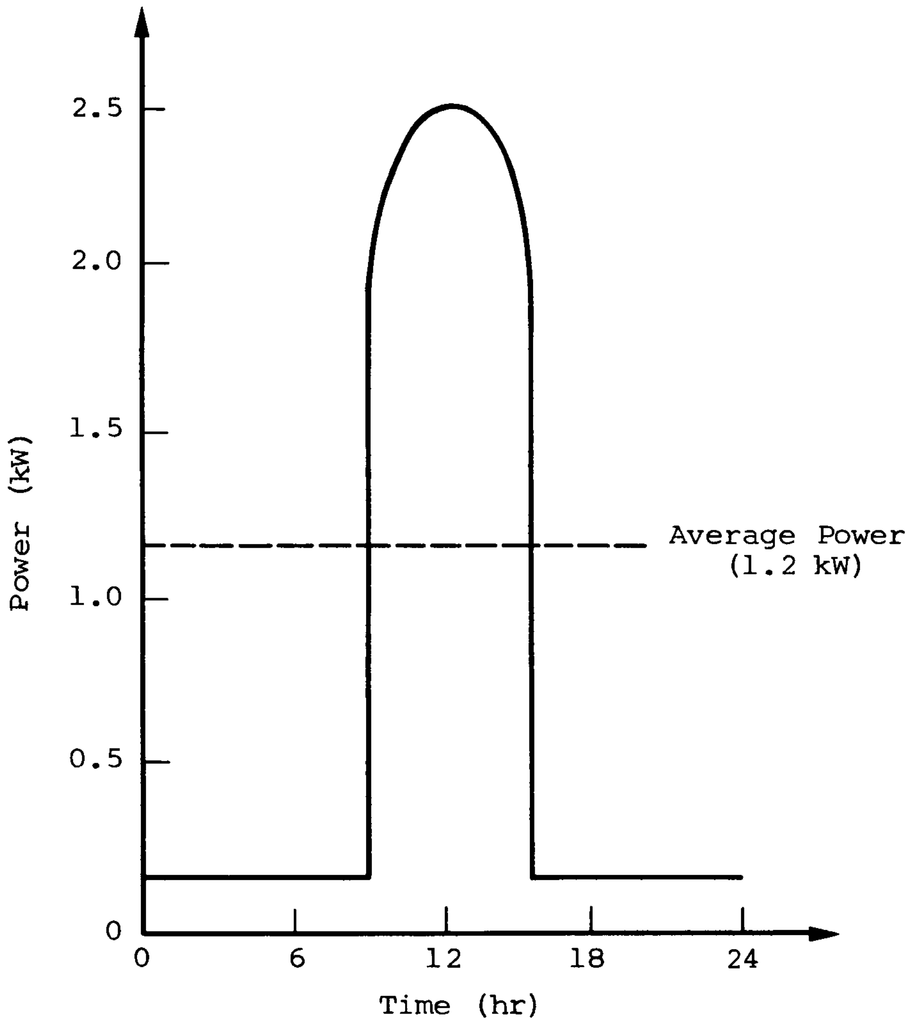


Figure 39 Thermal Control Electrical Input Profile

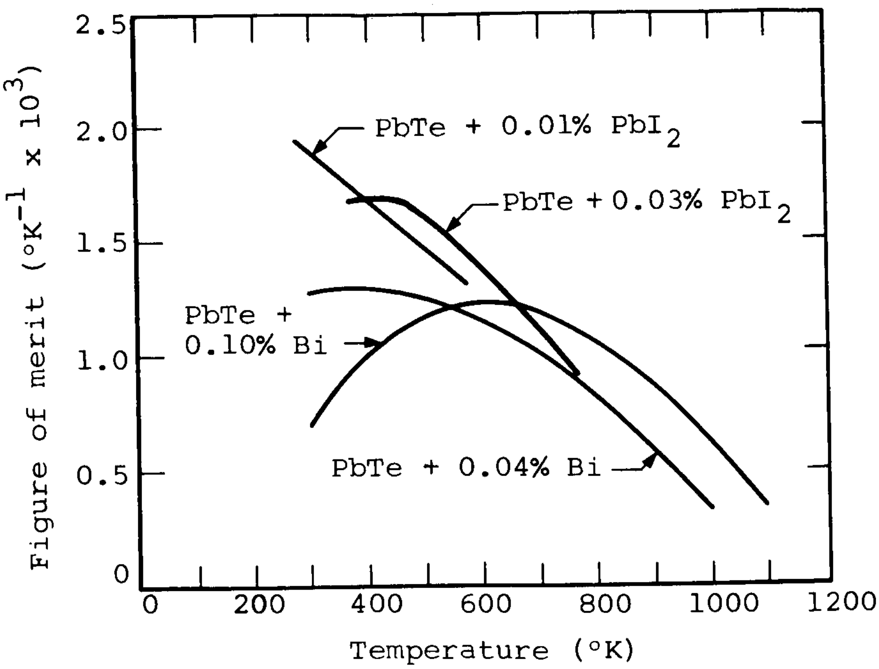


Figure 40 Figure of Merit of N-Type Lead Telluride (From ref. 11)

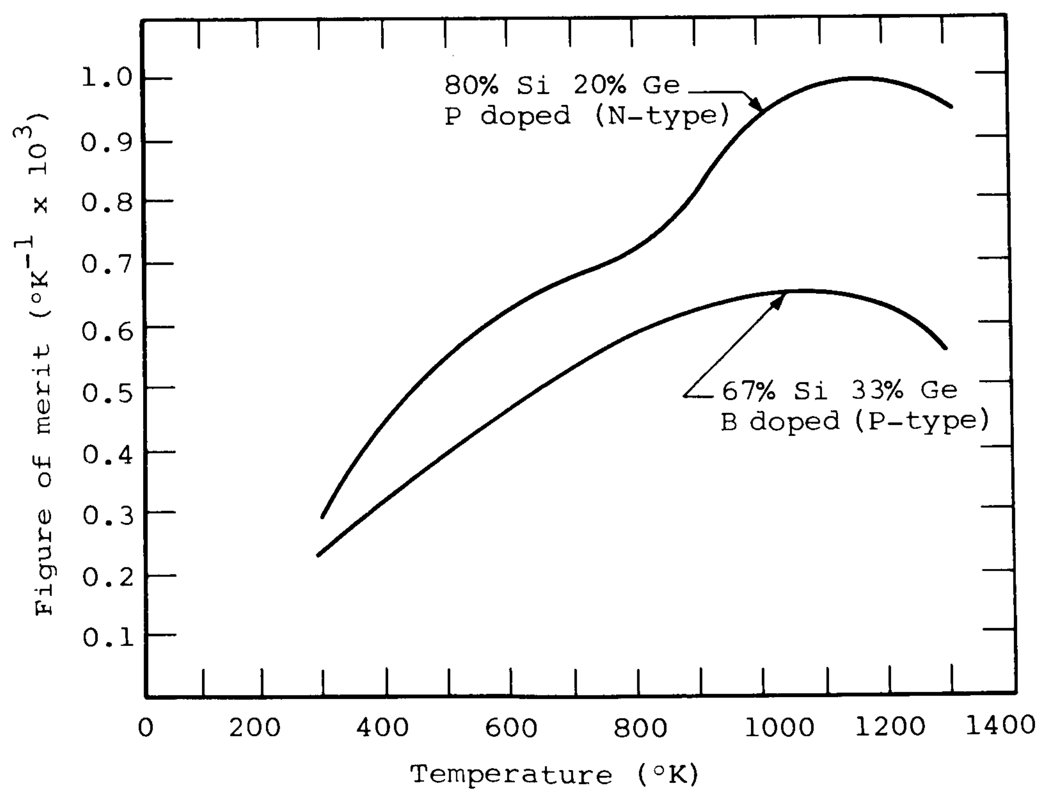


Figure 41 Figure of Merit of Optimum Silicon Germanium Alloys (From ref. 11)

TABLE XV

COMPARISON OF FIGURES OF MERIT[a]

[From ref. 24]

Material	Maximum Temp., °C	ΔT °C	Average Figure-of-Merit over Temperature Range, $°C^{-1}$	$\frac{1}{4}Z\Delta T$
AgSbTe	600	450	0.0015	0.17
SiGe	1100	950	0.0006	0.14
PbTe	600	450	0.0010	0.11
PbSnTe	550	400	0.0010	0.10
BeBiTe	600	450	0.0010	0.11
CeS	1200	1050	0.0002	0.05
BiTe	250	100	0.0018	0.04

[a]Maximum operating temperature and $\frac{1}{4}Z\Delta T$ for a cold junction temperature of 150°C.

Thermoelectric elements can be made to achieve efficiencies of about 12%. However, the overall efficiency of the conversion system (which depends on many factors such as method of heat transport, merit of design, temperature of operation and waste heat) varies from about 2 to 5%. The efficiency and specific power (W/lb) of thermoelectric conversion is dependent on thermal power density (W/cm^2) as depicted in Figure 42.

The overall system efficiency is also significantly lower than other energy conversion systems under study, and we do not feel that these systems will be competitive for the Mars surface application.

For the Mars surface application, the most effective use of thermoelectric energy conversion would be in combination with either a Radioisotope (RTG) or a reactor power source. The difficulty, however, is that the RTG devices, although very reliable, have a very low power output level. The SNAP-9A, a 25 W RTG, has been flown on a Navy navigation satellite. The first NASA use of RTG's is on Nimbus B; two SNAP-19 units, 30 W each, are used for secondary power. One reactor system, the SNAP-10A although not used in an actual mission, has been test flown. The power output, however, is still only 500 W. Larger systems are now being developed, but it does not appear that they will be mission capable by the mid 1980's.

2. Radioisotope system. - A considerable amount of information and design data is available for application of radioisotopes to space power generating systems. Radioisotopes as a source of energy have been strongly considered for missions such as unmanned interplanetary probes, lunar orbiting vehicles, communication satellites, and manned space laboratories and other long duration space missions requiring electrical power outputs of a few kilowatts.

The energy source of the radioisotope power system is heat that results from the natural decay of radioactive material. Radioisotopes, as found in nature, are obtained as by-products from a fission reactor or can be produced by artificial transmutation with neutron irradiation. They decay by emitting energetic particles (α or β) or electromagnetic radiations; these are subsequently converted to thermal energy during absorption.

The major advantages offered by radioisotopic fuels are:

- Can be used in any environment
- No power storage requirement
- Flexible in operation
- Long operational use
- Compact

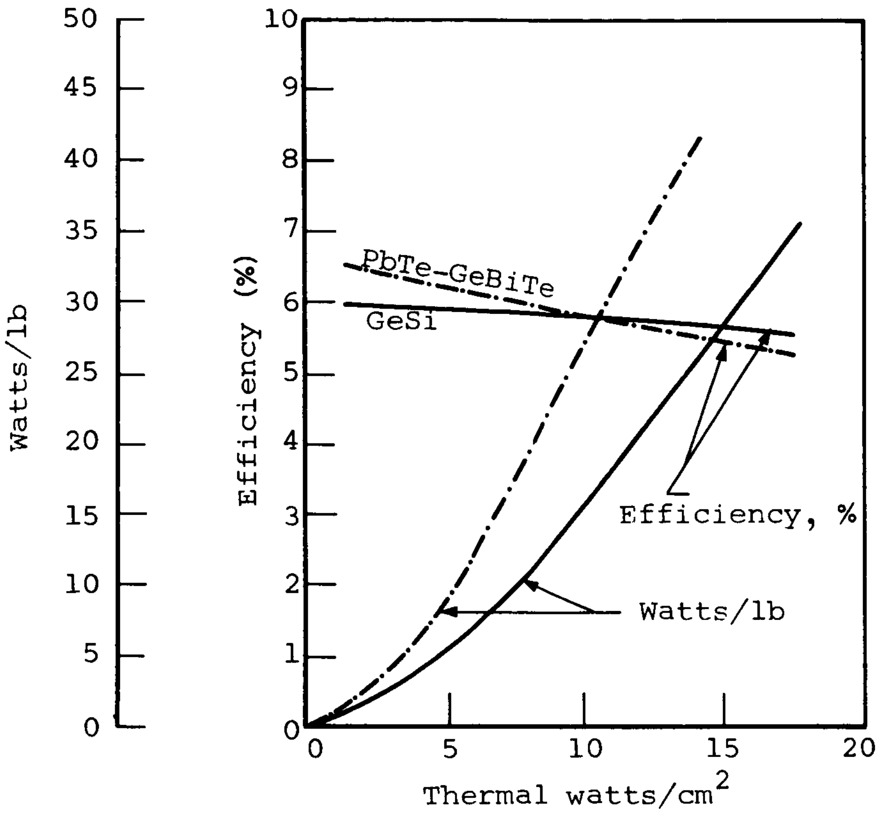


Figure 42 Efficiency and Specific Power of Thermoelectric Conversion (From ref. 25)

- Heat available to maintain equipment thermal balance

- Highly reliable

The disadvantages associated with radioisotopic system usage are:

- Safety requirements

- Availability

- Possible costs

- Unfamiliarity to users

- Possible interference with experiments

Polonium-210, plutonium-238, promethium-147, strontium-90, and cerium-144 are the five most studied isotopes for space applications. The selection of isotopes for a particular system is normally based on the following criteria:

- Fuel power density must be sufficient to match the requirements of the power system

- Decay rate should be compatible with the required operating time of the system

- Must be available in sufficient quantities to meet inventory requirements

- Emitted radiation and shielding requirements must be compatible with maximum crew dose and ground handling requirements

a. Fuel power density: Power density is determined by the choice of the energy conversion system. In general, a thermoelectric converter, or any dynamic system can operate with any isotope regardless of its power density. However, low system efficiency and high weight results when low power density fuels are used. In contrast, thermionic converters operating at temperatures higher than 1200°C require isotopes with power densities greater than 5 W/cm^3. As shown in Table XVI, Po^{210} has the highest power density (1210 W/cm^3) and is highly considered for short mission duration from the standpoint of cost, compactness, and radiation emission as will be discussed later.

b. Isotope life: As an isotope decays and emits energy, the source strength decreases in an exponential manner. To maintain a certain minimum power level, the heat source is normally oversized to allow for this degradation. If considerable excess energy is available at the start of the mission (100% excess energy is considered the maximum), it is usually rejected by dumping the heat directly from the source. The half life of the five isotopes under discussion vary from as low as 138 days for Po^{210} to as high as 89.6 years for Pu^{238} as shown also in Table XVI.

c. Fuel availability: Fuel availability is fundamental and very important. Many uses have been already planned for the more desirable radioisotopes, and priorities have to be established for their use. Production rates by fiscal year are shown in Table XVII. The maximum availability of any of these fuels is ultimately limited by the raw material except in the case of Po^{210} where there is an abundant supply of its raw material, bismuth. Table XVIII shows the total potential isotope capacity (kW_t/yr). The minimum and maximum predicted availability of isotope fuels is also depicted in Figures 43 and 44.

It is obvious from these figures that the planned production rates for all fuels except strontium and cerium are either inadequate or marginal. In particular, the cumulative availability of Pu^{238} will only be 400 kW_t by 1980, as shown in Figure 45. Twice this amount, however, is feasible with additional facilities, and proper planning (ref. 26).

d. Radiation emission: The safety philosophy for a radioisotope space power system, manned or unmanned, is basically concerned with the degree of containment and accountability of the fuel and its radiation. The potential safety problem areas and their solutions are strongly dependent upon the choice of isotope and shielding requirements.

Shielding must be provided about the radioisotope heat source to limit the radiation dose to the crew members. A total allowable integrated whole body dose of 54 rem has been established as an upper limit. This amount should include contributions by radiation in the environment, if present. Since the total system weight is governed by the amount of shielding required, it is necessary that an optimum selection of isotope fuel and shield be made.

Po^{210} and Pu^{238} are alpha emitters and require very little shielding. In contrast, Sr^{90}, Pm^{147} and Ce^{144} are beta emitters accompanied by energetic gamma radiations requiring heavy shielding, as shown in Figure 46. Comparison of radiation dosages for alpha and beta emitters can be seen in Tables XIX and XX. Weight requirements for a Manned Orbiting Laboratory (4 kW_e power), using different radioisotope fuels with various conversion systems, are shown in Table XXI. For example, a radioisotope-Brayton system would require 2479 lb of shield weight using Sr^{90}, whereas the same system using Pu^{238} would only require 36 lb of shield weight.

e. Isotope fuel cost: Another major consideration is the estimated cost of production related to limited availability of isotope fuels, as shown in Table XXII. The estimated cost for

TABLE XVI

CHARACTERISTICS OF ISOTOPES

[From ref. 9]

Isotope	Isotope fuel form	$\frac{W}{cm^3}$	Half-life	Melting point, °C	Weight density, g/cm^3	Specific activity, w/kc	Decay particle
Po^{210}	Metal	1210	138 days	254	9.3	31.7	Alpha
Pm^{147}	Pm_2O_3	1.5	2.6 years	2300	5.55	0.37	Beta
Sr^{90}	SrO_2	0.93	28 years	2430	2.65	6.5	Beta
Pu^{238}	PuO_2	5.0	89.6 years	2250	11.46	34.5	Alpha
Ce^{144}	CeO_2	13.8	285 days	2680	7	7.9	Beta

TABLE XVII

RADIOISOTOPE PRODUCTION CAPABILITIES

[From ref. 9]

Isotope	FISCAL YEAR					
	1964	1965	1966	1967	1968	1969
Po^{210}						
g	50	100	100	1000	1000	1000
kW_t	7.0	14	14	140	140	140
Pu^{238}						
kg	6.0	13	18	24	32	36
kW_t	2.6	5.65	7.8	10.4	13.9	15.65
Pm^{147}						
mC	0.5	0.5	0.5	0.5	15[a]	30
kW_t	0.18	0.18	0.18	0.18	5.5	11.0
Sr^{90}						
mC	5	5	5	5	10[a]	10
kW_t	32.5	32.5	32.5	32.5	65	65
Ce^{144}						
mC	3.5	3.5	3.5	3.5	50[a]	100
kW_t	27.7	27.7	27.7	27.7	350	790

[a]Based on construction of HIP (Hanford Isotope Plant) fission product process plant.

TABLE XVIII

TOTAL POTENTIAL ISOTOPE CAPACITY[a]

[From ref. 9]

Isotope	Potential capacity[b] kW(t)/yr, fiscal year							
	1967	1968	1969	1970	1971	1972	1974	1980
Ce^{144}	1060	1150	1285	1420	1630	1840	2395	4990
Pm^{147}	16.5	18.7	21.4	25	30	35	49	111
Sr^{90}	93	111	133	159	189	225	308	695
Pu^{238}	12.6	17	19.7	25.8	28	32	39	73

[a]Includes government and private sources.

[b]Values indicate total quantity of raw material available, based on Atomic Energy Commission estimates from both government and private reactors.

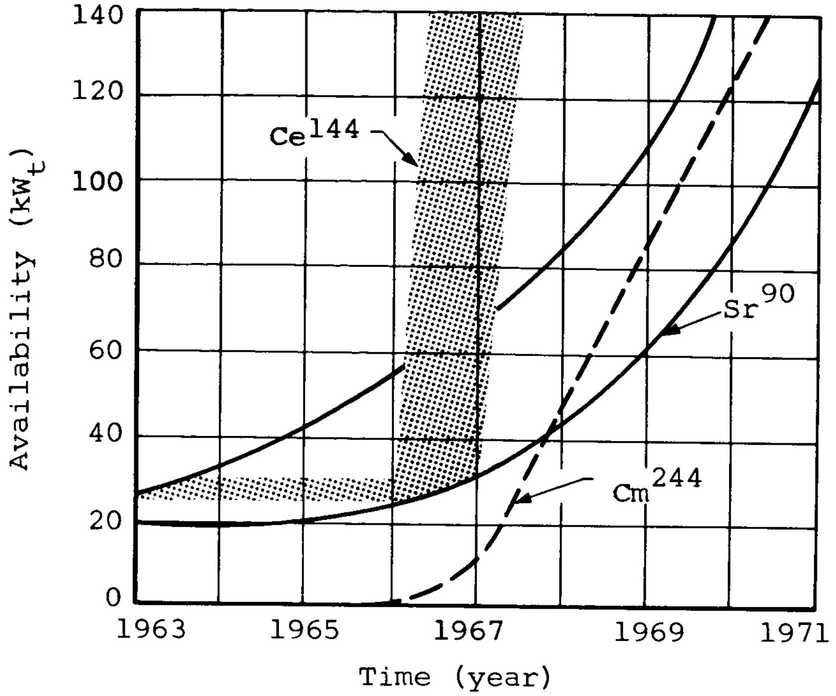


Figure 43 Isotope Availability (Ce, Sr, Cm)
(From ref. 16)

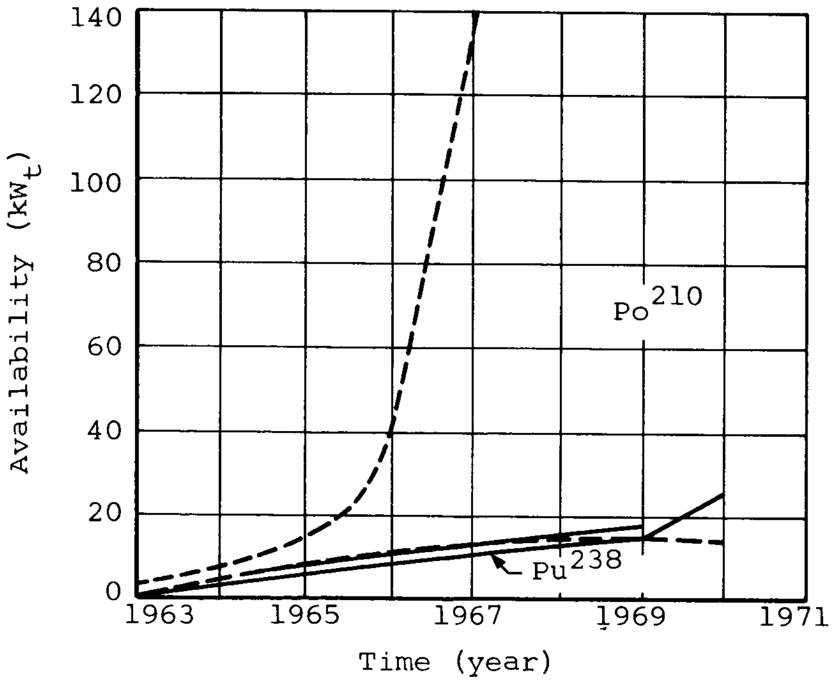


Figure 44 Isotope Availability (Po, Pu)
(From ref. 16)

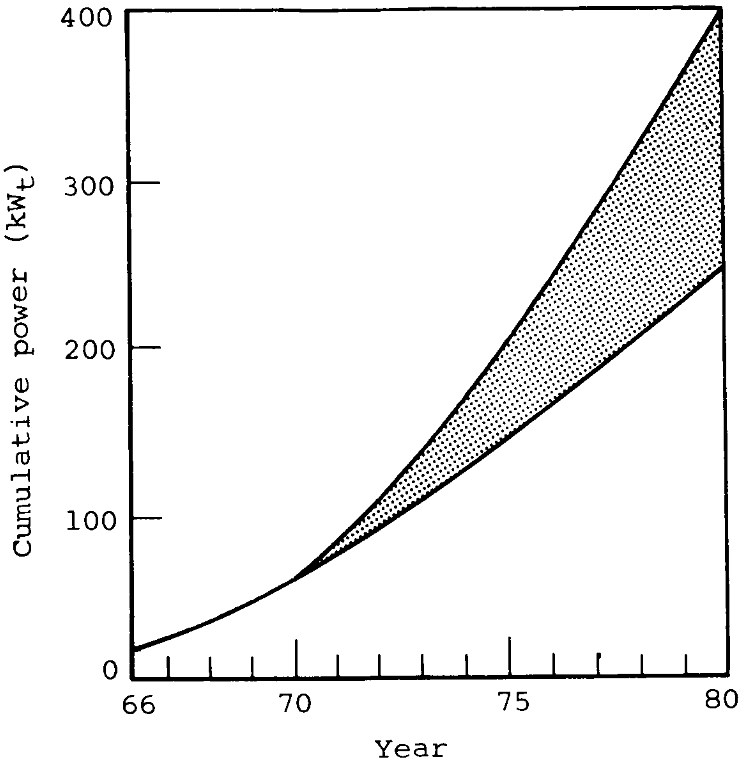


Figure 45 Estimated AEC Plutonium-238 Production
(From ref. 25)

TABLE XIX

CHARACTERISTICS OF ALPHA EMITTERS FOR POWER PRODUCTION

[From ref. 33]

Isotope	Half Life	Compound	Compound Power, W/g	Dose Rate, mR/hr[a]	
				Bare	3 cm U
Po^{210}	138 day	Metal Matrix	17.6	760	1.8
Pu^{238}	86 yr	PuO_2	0.35	5	0.03
Cm^{242}	163 day	Cm_2O_3 In Metal Matrix	15.5	280	2.0
Cm^{244}	18.4 yr	Cm_2O_3	2.5	600	32.0

[a]At 1 meter from 5 kW_t source.

TABLE XX

CHARACTERISTICS OF BETA EMITTERS FOR POWER PRODUCTION

[From ref. 26]

Isotope	Half Life	Compound	Compound Power, W/g	Dose Rate, mR/hr[a]	
				Bare	3 cm U
Co^{60}	5.27 yr	Metal	3.0[b]	$3x10^8$	$6x10^6$
Sr^{90}	28 yr	$SrTiO_3$	0.2	$6x10^6$	10^4
Pm^{147}(c)	2.67 yr	Pm_2O_3	0.3	$1x10^5$	1.0
Tm^{170}	127 day	Tm_2O_3	1.75	$4x10^6$	50

[a]At 1 meter from 5 kW_t source.

[b]200 curies/g metal.

[c]Aged - 1 half life.

TABLE XXI

WEIGHT SUMMARY (POWER SYSTEM/SHIELD)

[From ref. 9]

	Sr^{90}	Pm^{147}	Ce^{144}	Po^{210}	Pu^{238}
Thermoelectric direct heat removal	2505/7350	3220/1030	1070/19000	835/356	2150/50
Thermoelectric indirect heat removal	1830/5400	2420/635	925/9600	750/200	1660/25
Thermionic	-	-	550/30000	550/1340	-
Brayton	1085/2479	1180/177	915/5725	900/138	1105/36
Stirling	1000/2479	1090/177	870/5725	880/138	1050/36
Rankine mercury	1135/3029	1270/262	890/6538	865/182	1105/62
Rankine organic	1070/3029	1195/262	845/6538	825/182	1045/62

TABLE XXII

ISOTOPE COST DATA

[From ref. 9]

Isotope	Projected costs, $\$/W_t$	Yearly production rate for projected costs
Po^{210}	188	100 g
Pu^{238}	894	50 kg
Pm^{147}	486	10-30 mc
Sr^{90}	78	10 mc
Ce^{144}	5	100 mc

various isotopes varies considerably. However, it is quite apparent that Pu^{238}, which seems to be the best choice for manned missions, is by far the most expensive as compared to low cost Sr^{90}, which requires very heavy shielding for manned missions.

Although the anticipated cost figure for Pu^{238} is quite high, the consideration that it is an alpha emitter (which keeps the shield weight down) and the very long half life (which yields a reasonably constant power level throughout the mission) makes it the most attractive candidate.

We feel that the isotopic power source gives the most desirable properties for the anticipated Mars surface conditions, considering such factors as cloud cover and dust storms which would affect solar cells. It also has distinct advantages over a nuclear reactor power source from the standpoint of weight and accessibility.

3. <u>Solar concentrators</u>. - Although solar concentrators are studied for space power applications utilizing various static and dynamic power conversion techniques, the solar-thermionic system approach appears most promising, particularly from the standpoint of cost, weight, and size with greatly minimized radiation damage problems.

Pertinent factors which must be realized or considered in the design and material selection for the different solar concentrator configurations are:

- Operating temperature of the energy conversion device or system
- Efficiency
- Weight per unit projected area
- Specific power: thermal energy per unit weight at a specific temperature
- Prelaunch storage volume and deployment method
- Optical degradation due to thermal gradients and space environment effects
- Meteoroid damage to mirror surface
- Dust collection on mirror surface

Among its important characteristics, a collector must gather solar radiation efficiently and provide the desired degree of concentration of the radiation commensurate with the ability of the conversion system to use the heat.

Figure 47 shows efficiency as a function of concentration ratio for six different collector concepts. The concentration ratio is the ratio of the projected mirror area of the collector to the aperture area of a cavity heat absorber. Efficiencies as high as 91% will be attainable by 1970 (ref. 9).

Solid one-piece concentrators, in 5 and 10 ft sizes, have been made with high mirror efficiencies. Concentrators with 25 and 50 ft diameters and high reflectivities will also be available by 1970.

The collector-absorber efficiency which can be obtained from any mirror-cavity combination, is a function of the geometrical and reflective efficiency of the concentrator and the design temperature of the cavity. At each cavity temperature, an optimum entrance diameter can be determined which will result in the best combination of reradiation losses from the cavity entrance and losses due to the solar radiation not entering the entrance. Figure 48 shows a typical family of curves where the temperature of the cavity ranges from 1600° to 1800°C, and the mirror efficiency corresponds to that obtained from concentrators at mirror's distance from the sun. The curves show maximum collector-absorber efficiency and indicate quite clearly that for large concentrators, the obtainable system efficiency is lower. Calculations have been made for determining the effect of vehicle distance from the sun on the mirror efficiency (ref. 27); these curves presented in Figure 49 show that good collector-absorber efficiencies can be obtained at Mars.

One of the principal aims of development to date has been to construct practical collectors of minimum weight. For six different collector concepts shown in Figure 47, the specific weights are plotted in Figure 50, and the combined collector-absorber efficiencies are given in Figure 51. The specific weights vary from 0.40 to 1.04 lb/ft^2, with the collectors having the lower weights also having poorer geometries.

The selection of a collector for a given power conversion system could be based on the specific power, the ratio of power to unit weight. Figure 52 shows specific power values for different cavity temperatures considering solar constant of 130 W/ft^2. In the temperature range of 1500 to 3500°R, the inflatable collector delivers the most power per pound because of its extreme low unit weight.

The mirror weight and diameter requirements for missions at different distances from the sun are well represented in Figure 53 for different electrical power outputs using thermionic generator.

Solar collector requirements for Mars surface application are, however, subject to the same difficulties as solar cell systems. The most significant is low solar density, and possible loss of energy due to cloud coverage or dust storms. However, the effects of temperature and radiation exposure are much less significant as compared to solar cells.

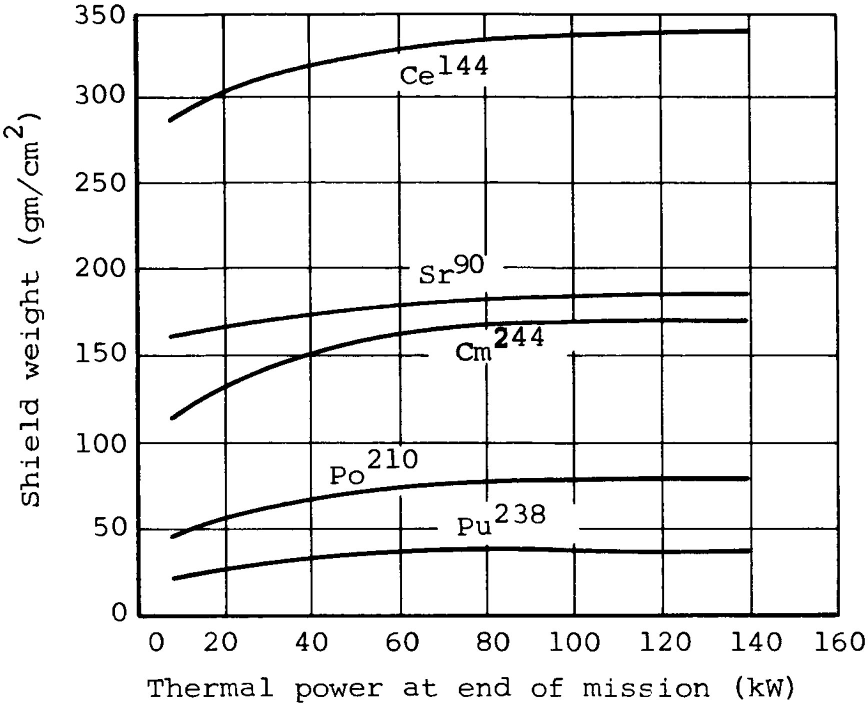


Figure 46 Shield Weights
(From ref. 16)

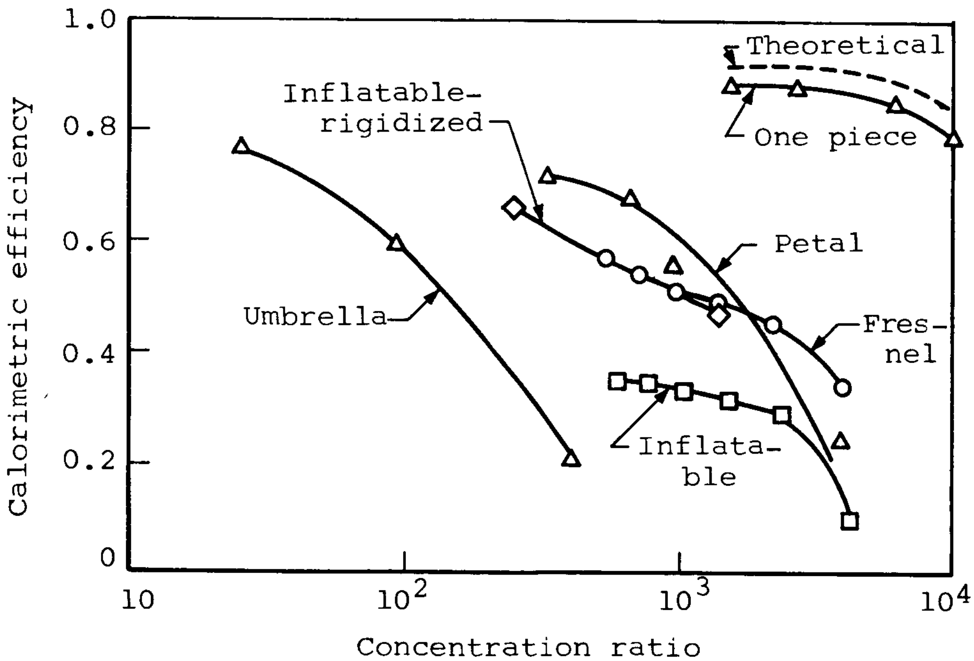


Figure 47 Collector Efficiency
(From ref. 27)

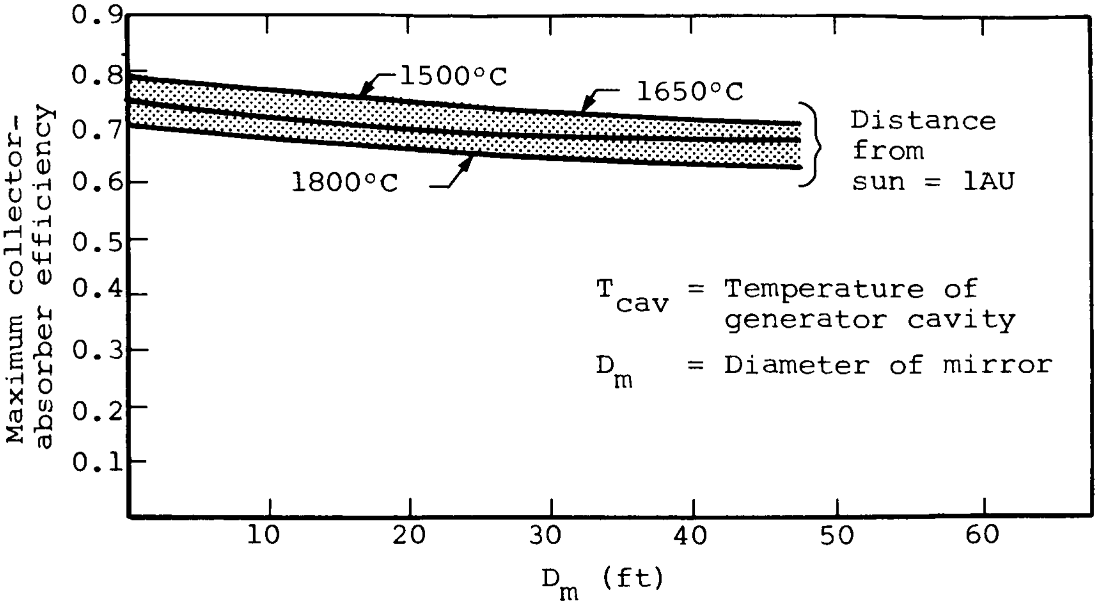


Figure 48 Maximum Collector-Absorber Efficiency for Mirrors at Earth's Distance from Earth

(From ref. 9)

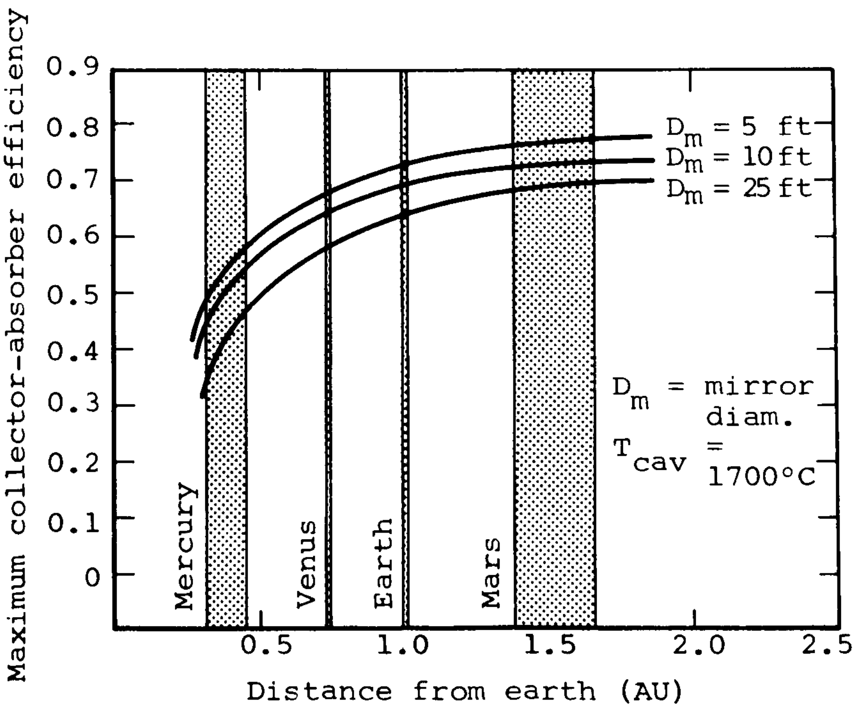


Figure 49 Maximum Collector-Absorber Efficiency for Mirrors at T_{CAV} = 1700°C

(From ref. 9)

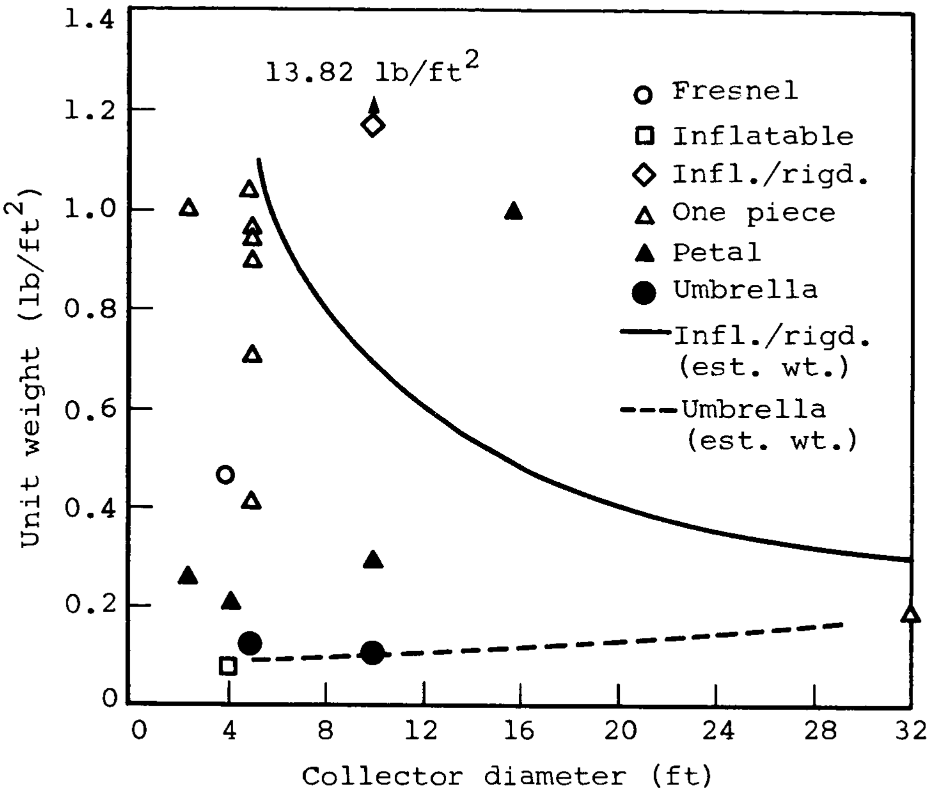


Figure 50 Collector Unit Weight
(From ref. 27)

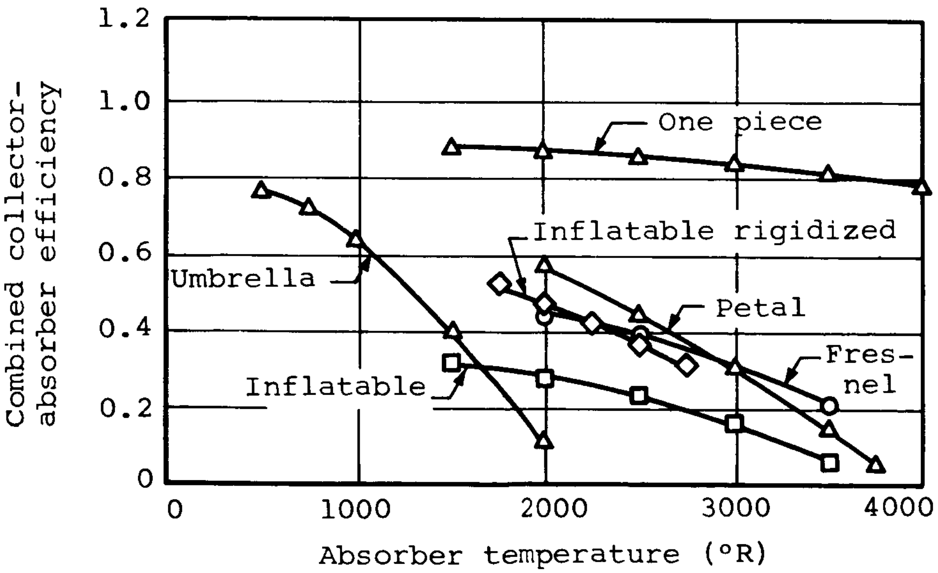


Figure 51 Combined Collector-Absorber Efficiency
(From ref. 27)

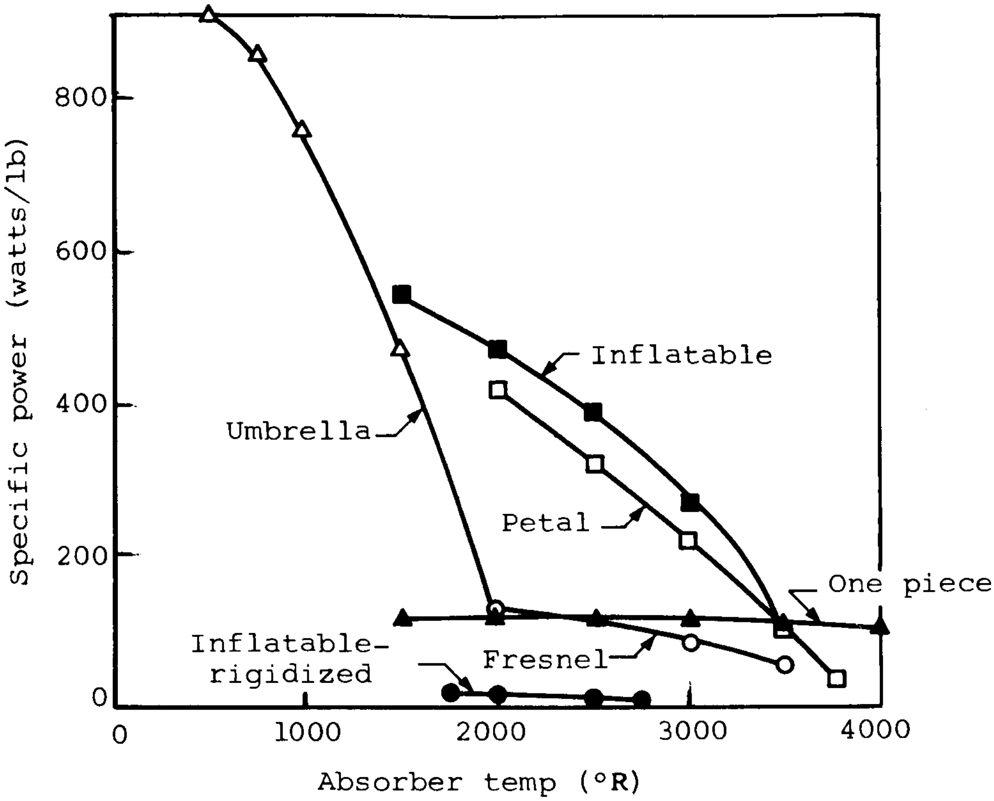


Figure 52 Collector Specific Power
(From ref. 27)

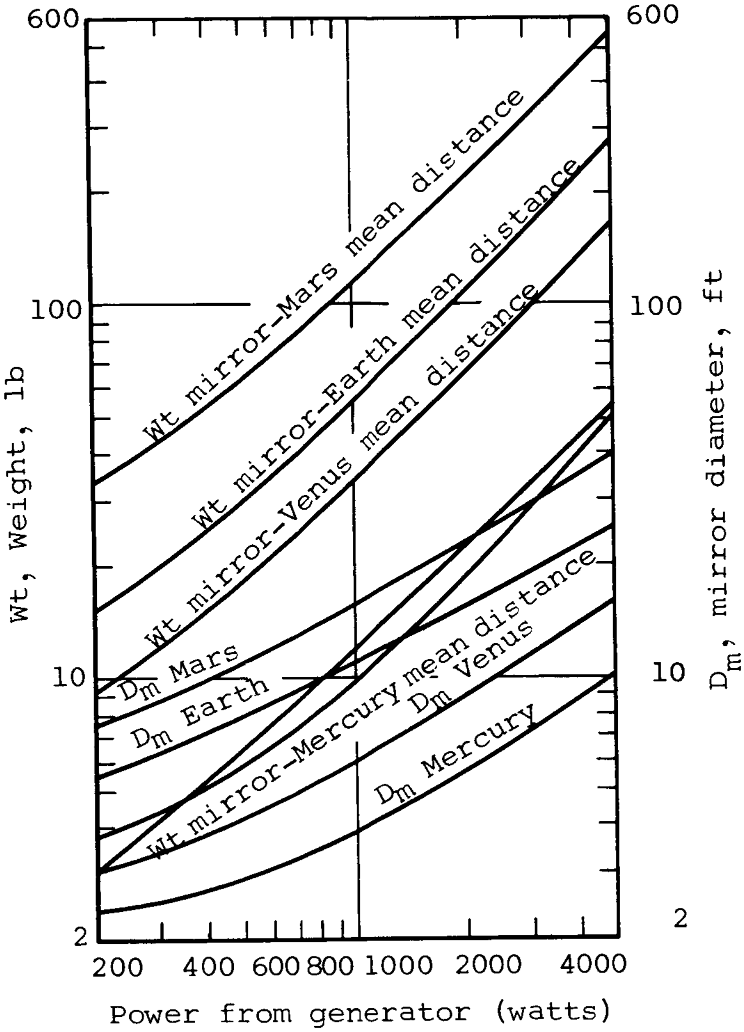


Figure 53 Predicted Weight and Diameter of Concentrator at Various Power Levels

(From ref. 9)

4. Thermionic power conversion. - A thermionic converter is a device which converts heat into electricity by utilizing the thermionic emission of electrons. It is basically a simple device consisting of a hot (typically 1400-2200°K) emitter electrode spaced a short distance from a cooler (typically 600-1200°K) collector electrode. Electrons flow from the emitter across the gap to the collector and then return to the emitter through an external load where they do useful work.

Thermionic conversion appears promising for space power because of light weight, lack of moving parts, and because the high collector temperature reduces the size of the radiator necessary for getting rid of waste heat.

Thermionic systems involving many diodes have a definite redundancy advantage compared to single pieces of rotating machinery. Thermionic converters have been studied for future space applications with sources of energies such as solar concentrators, isotope heat, and nuclear reactors.

Power output, efficiency, and lifetime have been measured by many organizations, using an electrically heated Cesium thermionic converter (ref. 28).

Figure 54 gives a typical family of current-voltage characteristics observed at various cesium reservoir temperatures with T_e (emitter temperature) held fixed. If one optimizes both T_c (collector temperature) and the Cs reservoir temperature, a maximum efficiency and a maximum power point can be found for a set of curves such as in Figure 54. The maximum power point occurs at a current density of 34 amp/cm^2 and electrode potential of 0.7 volts, whereas the maximum efficiency occurs at 16 amp/cm^2 and 1 volt.

Heat-to-electrical conversion efficiencies (for the diode alone) of over 19% have been demonstrated with a corresponding power density of 10-20 W/cm^2 (ref. 28). Long operating lives at moderate emitter temperatures have also been demonstrated in many laboratories.

Because of high efficiency (almost twice that of thermoelectric converters), thermionic conversion is highly considered in conjunction with radioisotope heat for power loads of up to 10 kW_e.

Studies of power systems using solar energy with thermionic conversion indicate that a complete system, solar collector, thermal storage, thermionic converter and supporting structure, would weigh about 80 lb/kW (ref. 28), which is quite low compared to other systems utilizing dynamic energy conversion.

For a system larger than 10 kW, nuclear reactors appear more attractive as heat sources for thermionic converters. Studies by G. E. Vallecitos Laboratory (ref. 29) indicate that such a system in the power range of 10 kW to 40 kW could be developed.

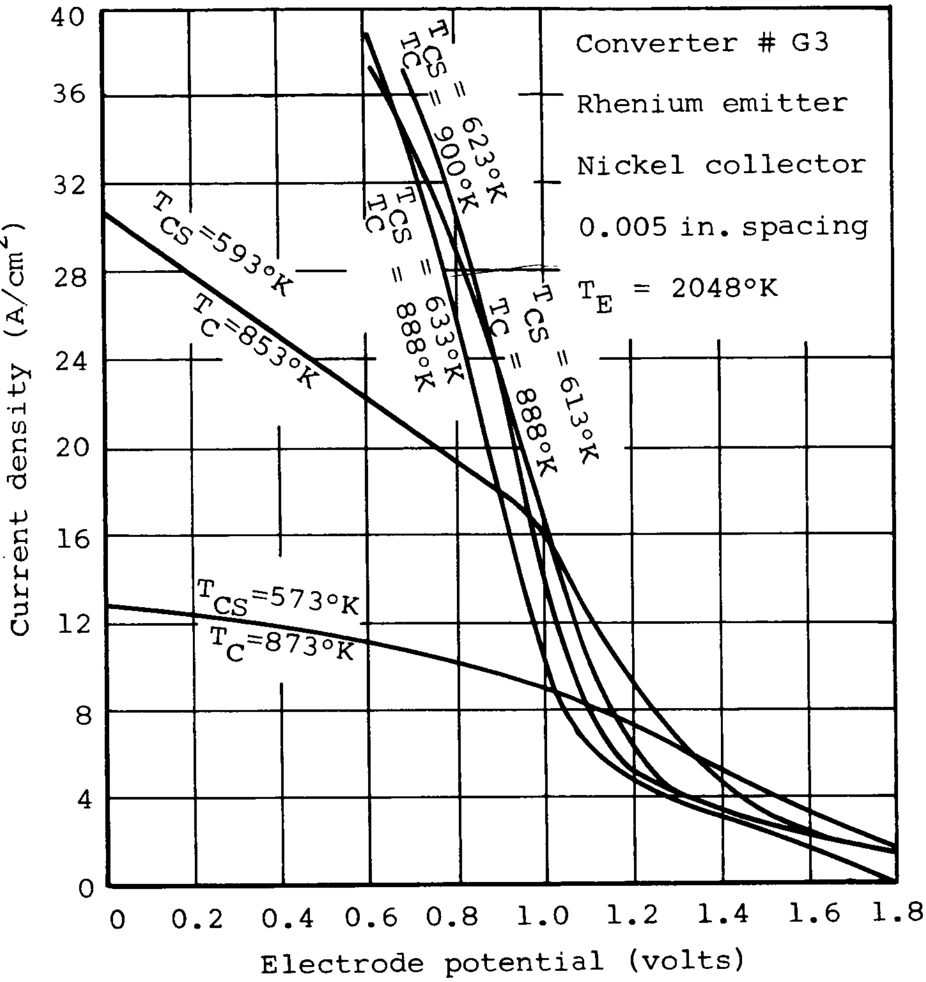


Figure 54 Thermionic Current Voltage Characteristics
(From ref. 30)

Thermionic converters are still in the early development stages and have many associated design problems which have yet to be overcome.

One major problem is the fact that a large thermal power level is required for operation which in turn means a very heavy nuclear shielding requirement. It is very unlikely that these systems will be mission capable by the mid 1980's.

5. Brayton cycle. - Low specific weight, high efficiency and advanced state of development have broadened interest in dynamic energy conversion systems. Additional interest in dynamic components has been generated because some of the large static power systems also require rotating parts to provide the function of heat transport. The weight advantage over static systems becomes particularly noticeable as the power requirements exceed 1 kW.

Two dynamic systems that are receiving the most investigation are the Brayton cycle system and Rankine cycle system.

In a Brayton system using argon, the heat generated by the source (solar, radioisotope or nuclear) is transferred to a heat exchanger and then removed by the argon fluid (see Figure 13). The argon gas then expands in a turbine which drives the alternator and compressor. The turbine exhaust passes through the recuperator and transfers heat to the compressor discharge gas. The gas then flows through the radiator heat exchanger where it rejects heat to a separate liquid heat transport fluid. The heat transport fluid then rejects the heat through a radiator. The gas from the radiator heat exchanger then flows to the compressor, and returns to the heat source through the recuperator before repeating the cycle.

The Brayton cycle efficiency is relatively high. Figure 55 shows a plot of Brayton cycle efficiency and optimum compressor ratio versus the ratio of compressor and turbine inlet temperatures. For these same temperature ratios, Figure 56 gives the radiator area and inlet and outlet temperature.

6. Rankine cycle. - The Rankine system differs from the Brayton system in that it uses a two phase working fluid. The working fluid in the liquid phase is vaporized by the heat source. The vapor expands through a turbine which drives the alternator and condensor pump. The exhaust gas is condensed by a heat rejection loop and condensed liquid is vaporized again by the heat source repeating the cycle.

The overall efficiency of the Rankine cycle is somewhat lower than that of the Brayton and is plotted for different values of heat rejection temperature in Figure 57. For most space applications, the Brayton cycle is generally preferred over the Rankine cycle due to the problem of phase separation in a zero-g field. Since we are now dealing with a gravity condition, the Rankine system becomes highly competitive. The primary advantage is the improved heat transfer properties of a fluid over a gas.

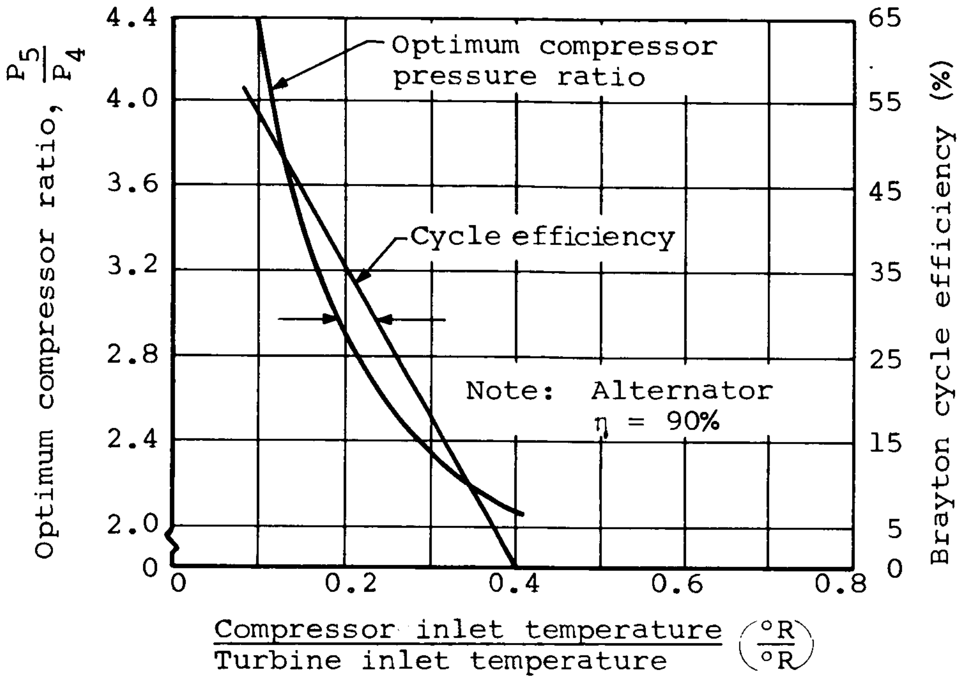


Figure 55 Overall Brayton Cycle Efficiency and Optimum Compressor Pressure Ratio

(From ref. 16)

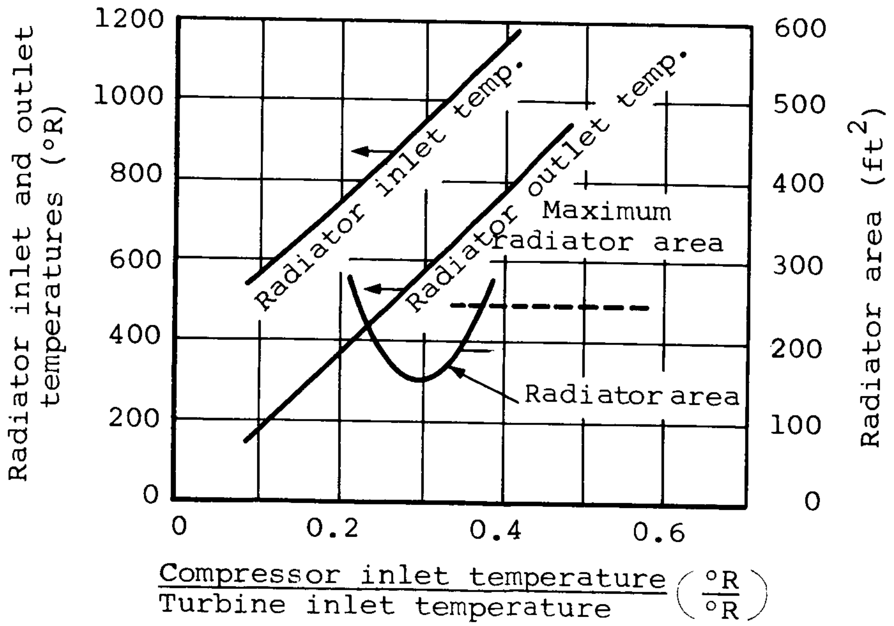


Figure 56 Brayton Cycle Radiator Temperature and Area

(From ref. 16)

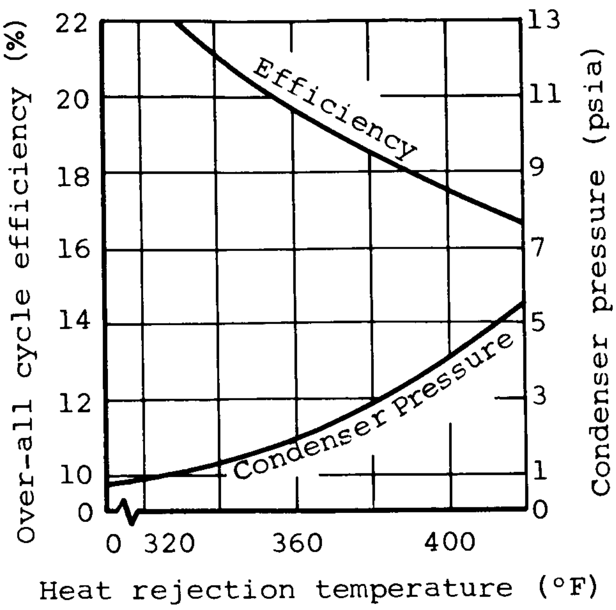


Figure 57 Overall Rankine Cycle Efficiency and Condenser Pressure

(From ref. 16)

a. Working fluid: Water, mercury, and organic fluids have been compared for use as the working fluid in the Rankine cycle, and it has been concluded that the organic fluids are preferable for the Mars surface application.

For ground applications, water is the most common working fluid used in the Rankine cycle. It has the advantage of a high latent heat, but the disadvantage of a comparatively low critical temperature. This reduces the possibility of obtaining high efficiency with the simple Rankine cycle. In addition, the very high critical pressure increases the weight of the power plant because of the need for a high pressure boiler.

For Rankine cycle space applications, the use of mercury as the working fluid has been under investigation for about eight years, and during this time many problems have been solved. The major concern has been operation in a zero g field which causes phase separation difficulties with the two-phase working fluid. Since a gravity field does exist on the Mars surface, there should not be a problem of phase separation.

Mercury has a very high critical temperature and moderate vapor pressures in the high temperature range, and has been contained quite successfully for moderate time intervals under steady state conditions. The expected high system weights using mercury (due to its high specific gravity) is not reflected in increased system weight, since mercury has such excellent heat transfer properties. This reduces the heat transfer area required, and the corresponding weight of working fluid which must be accommodated. The major drawback for the Mars surface application is potential leakage and corrosion under extended mission conditions.

An organic Rankine system has been developed more recently utilizing biphenyl or Dow-Therm-A. These fluids exhibit the unique property of a positive slope of the saturated vapor line. The peculiar shape of the liquid phase region on the temperature-entropy diagram shown in Figure 58, means that the turbine is always operated with superheated vapor. Conventional working fluids exhibit the normal vapor dome in Figure 59, indicating that the turbine inlet gas must be superheated initially to achieve the same effect. Thus, the shape of the Dow-Therm-A vapor dome results in a lower turbine inlet temperature, greatly simplifying the development of the radioisotope heat source. The expansion of the fluid through the turbine takes place completely in the superheated region and prevents condensation in the last stages of the turbine. The temperature at the end of the turbine expansion process is higher than the condensing temperature, hence a regenerator can be used to recover some of the waste heat from the turbine exhaust. This waste heat can be used to preheat the boiler feed to obtain a higher cycle efficiency. This cycle of operation is shown in Figure 60.

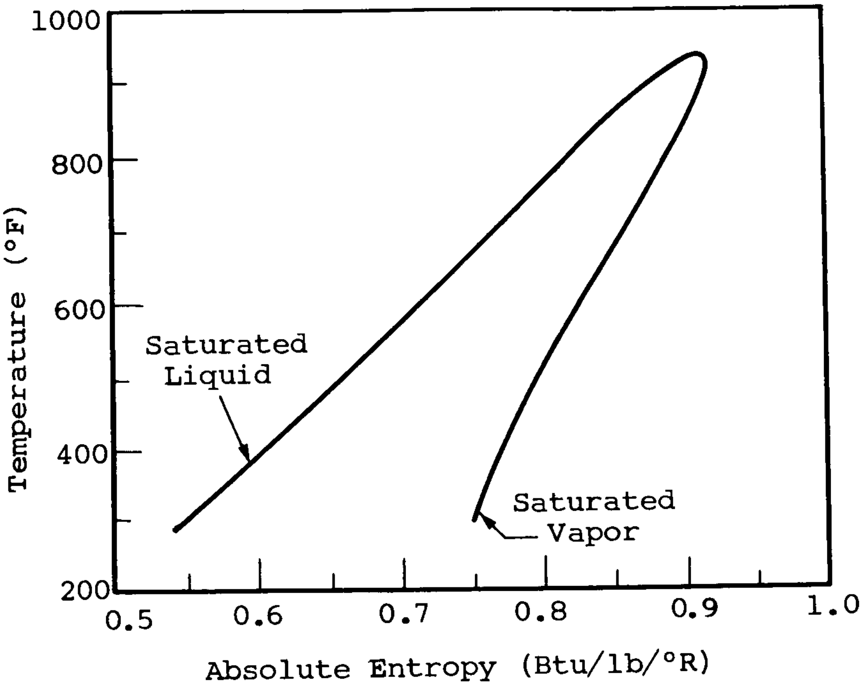


Figure 58 Temperature-Entropy Diagram for Dowtherm A

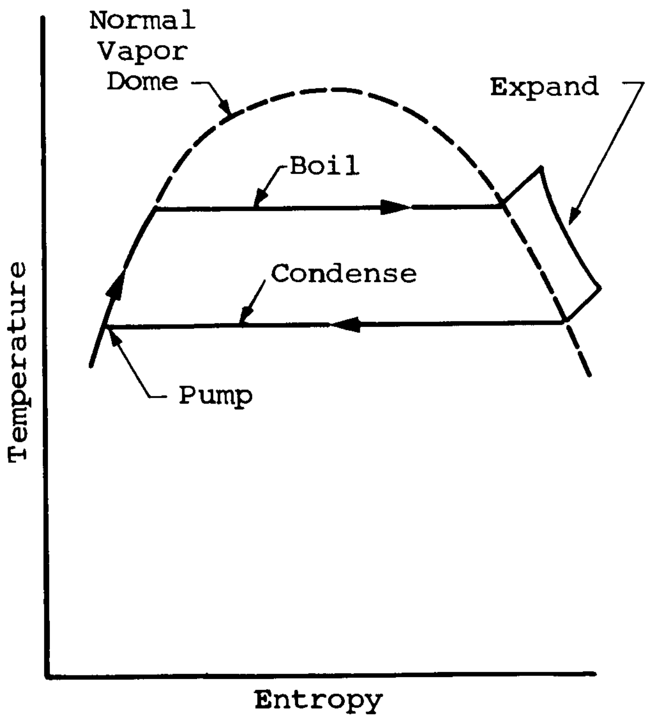


Figure 59 Rankine Thermodynamic Cycle

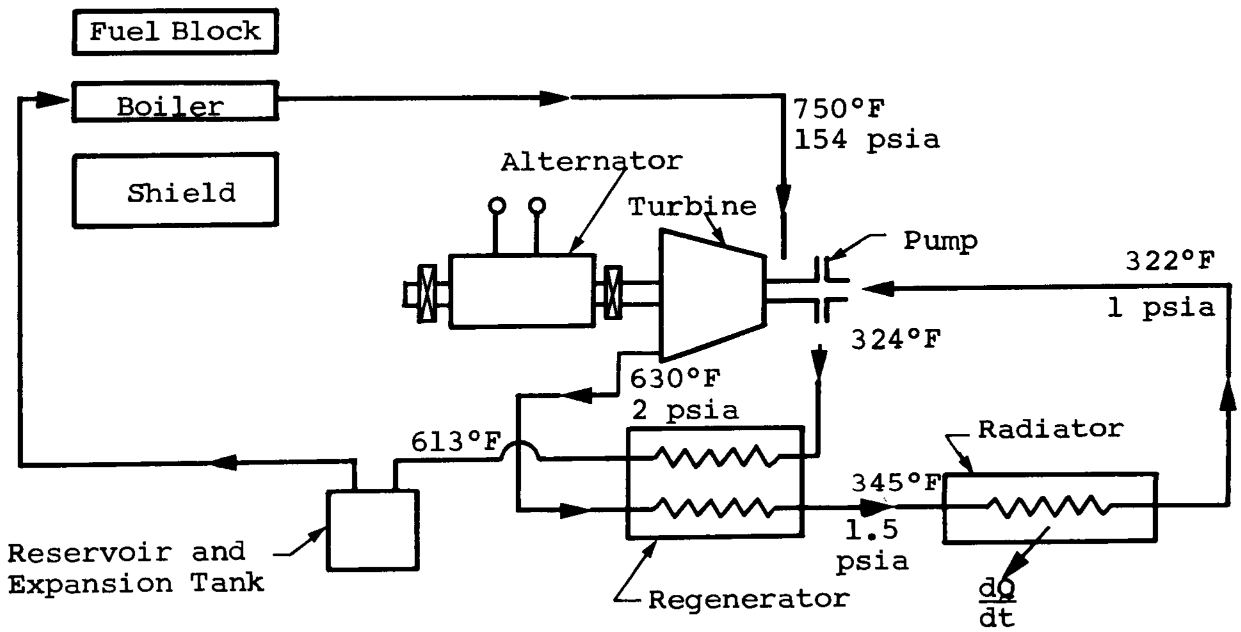


Figure 60 Rankine Cycle Unit
(From ref. 16)

The major feasibility question for the Dow-Therm-A working fluid is that of long-term temperature and radiation breakdown. The Dow-Therm-A working fluid would receive an integrated dosage from an isotope heat source of approximately 2×10^3 rem from neutrons, and 3×10^4 rem from gammas, for a 120-day mission. Data obtained from Sundstrand Aviation indicates that these doses will result in less than one-tenth of one percent degradation of the Dow-Therm-A. If necessary, this could be overcome by removing tars and volatiles as they are formed and by providing make-up fluid to account for the loss.

7. Brayton and Rankine cycle comparison. - A summary of the basic system characteristics just described is presented in Table XXIII for ease of reference.

The Rankine cycle demonstrates distinct advantages over the Brayton cycle for the Mars surface application. The major advantage of the Rankine system is the lower total system weight. The two systems are compared on the basis of weight in Table XXIV. The largest contributing item is the reduced weight of the isotope source since the Brayton system requires fuel capsule-to-gas heat exchangers resulting in larger and heavier radiation shields. Another item is the reduction in required radiator area for the Rankine system. Figure 61 shows a comparison of radiator areas for various cycle temperature ratios; the difference is quite significant. The reduction in required heat transfer area is also significant from the standpoint of thermal integration of the power source waste heat and the life support endothermic processes. The Rankine cycle efficiency is only slightly lower than the Brayton cycle efficiency, as can be seen in Figure 62.

An additional advantage is found under thermally integrated system performance. The Brayton system is much more sensitive to interactions, since the efficiency is decreased more rapidly by any increase in compressor inlet temperature. This increase would result from a decrease in the life support endothermic process heat demand. The effects on the Rankine cycle are less because of the larger amount of heat being rejected (smaller proportion being removed) and its higher temperature. There is a large amount of waste heat at a relatively high temperature in the Rankine cycle so that decreased electrical output would not affect the endothermic processes (ref. 31).

Since Pu^{238} is the selected isotope, it is significant to note that both the Brayton and Rankine cycles exhibit a lower weight penalty when used with this source as compared with the next closest competitor, Po^{210}. The weight reduction is shown in Figures 63 and 64 for 4 kW systems.

From the foregoing preliminary analysis, we have concluded that the Pu^{238} isotope-Rankine system using Dow-Therm-A as the working fluid is the most desirable candidate for the extended mission Mars surface application under consideration.

TABLE XXIII

BRAYTON AND RANKINE POWER SYSTEM CHARACTERISTICS

[From ref. 14]

	Rankine	Brayton
Working Fluid	Liquid Metal Two Phase	Inert Gas Single Phase
Zero-g Effects	Boiling and Condensing	None
Corrosion Problems	Liquid Metal Corrosion	None
Compressor or Pump Work	Low	High
Cycle Temp. Ratio T_{min}/T_{max}	High	Low

TABLE XXIV

BRAYTON AND RANKINE ISOTOPE POWER SYSTEM COMPARISON

[From ref. 31]

Function	Mercury Rankine Power System		Brayton Power System	
	Nonintegrated	Integrated	Nonintegrated	Integrated
Electric Power to Station Bus, kW_e	5.00	3.79	5.00	3.81
Generator Output, kW_e	6.07	4.62	5.77	4.46
Isotope Source Size, kW_t	50.3	42.7	29.0	24.0
Isotope Source Weight, lb	1072	989	2105[a]	1861[a]
Turbomachinery Weight, lb	55	50	166	146
Heat Rejection System Weight, lb	142	121	367	305
Waste Heat Rejection, kW_t	41.8	35.3	19.5	16.1
Power System Weight, lb	1775	1612	3444	2989

[a]Does not include thermal control

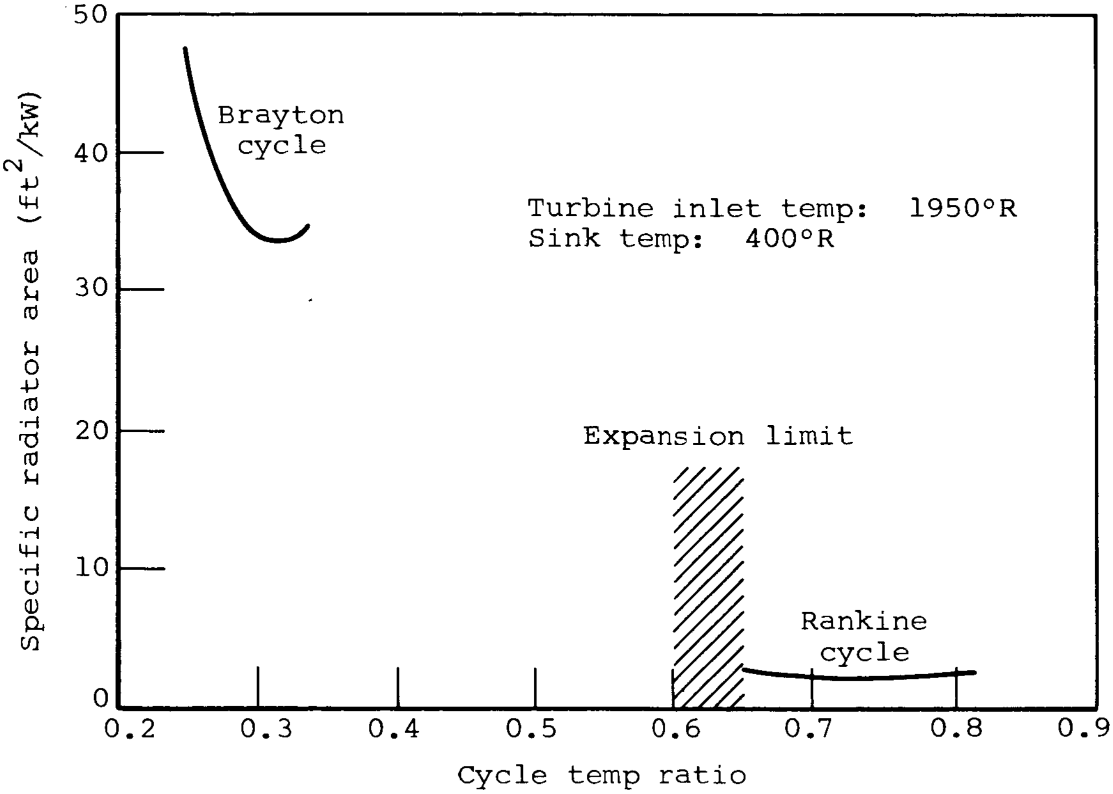


Figure 61 Brayton and Rankine Radiator Area Comparison
(From ref. 11)

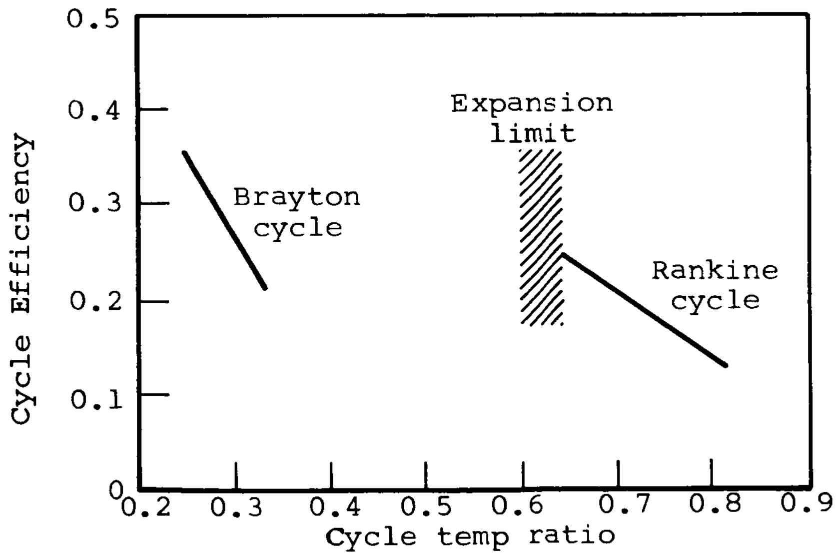


Figure 62 Brayton over Rankine Cycle Efficiency Comparison
(From ref. 11)

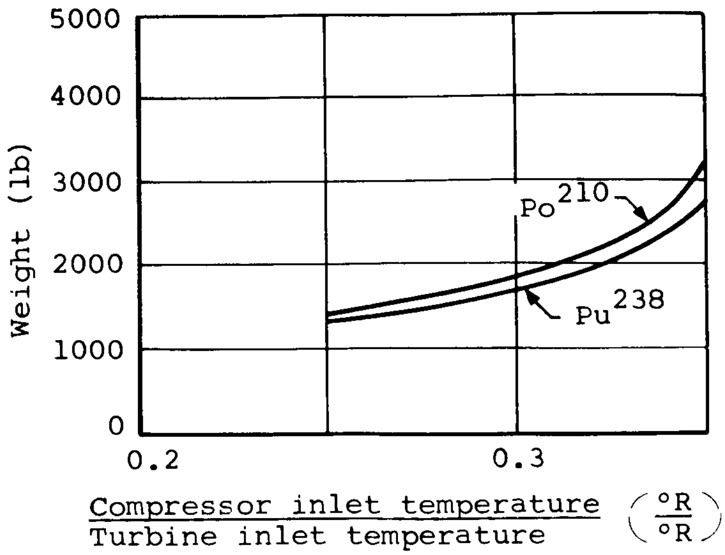


Figure 63 Brayton Cycle System Weight
(From ref. 16)

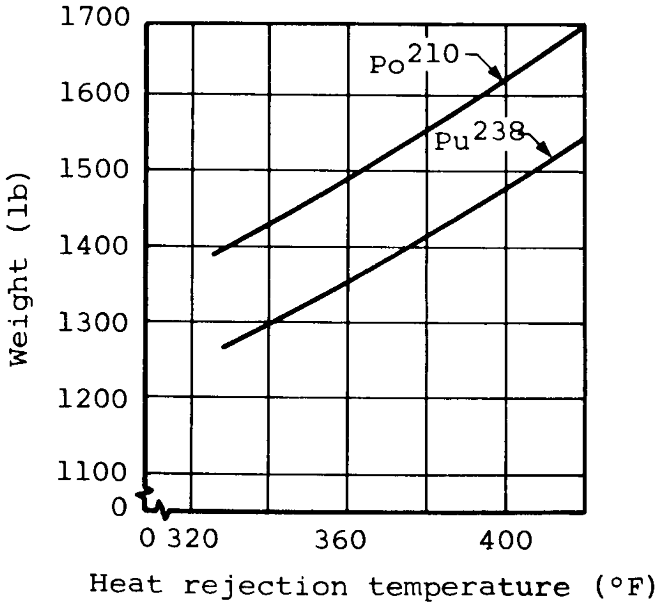


Figure 64 Rankine Cycle System Weight
(From ref. 16)

8. <u>Solar cells</u>. - It has been common to state, in the past, that solar cells are applicable to space power requirements up to 1 kW of electric power. Beyond that, the area, cost, orientation and other problems make them impractical. However, research is progressing towards increasing the practical capacity of solar cells in the range beyond 1 kW. Theoretically, the larger capacity systems might be quite feasible for certain applications, although none are yet in operation.

Considerable research effort has been devoted toward increasing the resistance of solar cells to particle radiation, a problem associated with all semi-conductor devices. Degradation under radiation is logarithmic with time as illustrated by Figure 65, and it is possible to derate the power supply and extend the useful life. Relative radiation damage resistance is greater for N/P cells than it is for P/N cells. It is also greater for CdS thin film cells than it is for conventional silicon cells.

Other problem areas associated with solar cells, making performance prediction very complex, are:

a. Light intensity: For most of the Earth orbit missions, this is not a serious problem because there is fairly adequate light intensity available for the major portion of the orbit, and the durations in shade are short enough that storage battery requirements are not too serious. However, for interplanetary missions, two factors must be considered, (1) the intensity of light during day period, and (2) the duration of shade period (requiring stored energy). The voltage current and resistance characteristics of the typical one ohm N/P cell are shown in Figures 66 and 67 for different values of illumination. Current is generally a linear function of intensity, whereas voltage begins to saturate at very low illumination and increases only slightly with increasing intensity. This results in a power output level that is directly proportional to the illumination.

Silicon cells have an energy efficiency of approximately 12% at 25°C, and the present state-of-the-art in thin films cells has shown consistent efficiencies of around 3.5% although efficiencies up to 6 and 7% are expected in future.

b. Temperature effect: Performance of solar cells is greatly dependent on temperature, since the energy efficiency at 125°C drops to about half its value at room temperature. Voltage and current characteristics are also seriously affected by temperature, as shown by Figure 68. Open circuit voltage drops with increasing temperature at a rate of approximately .9925 V/C, whereas the short circuit current increases only slightly with temperature. The percent change in efficiency is approximately -56% per °C.

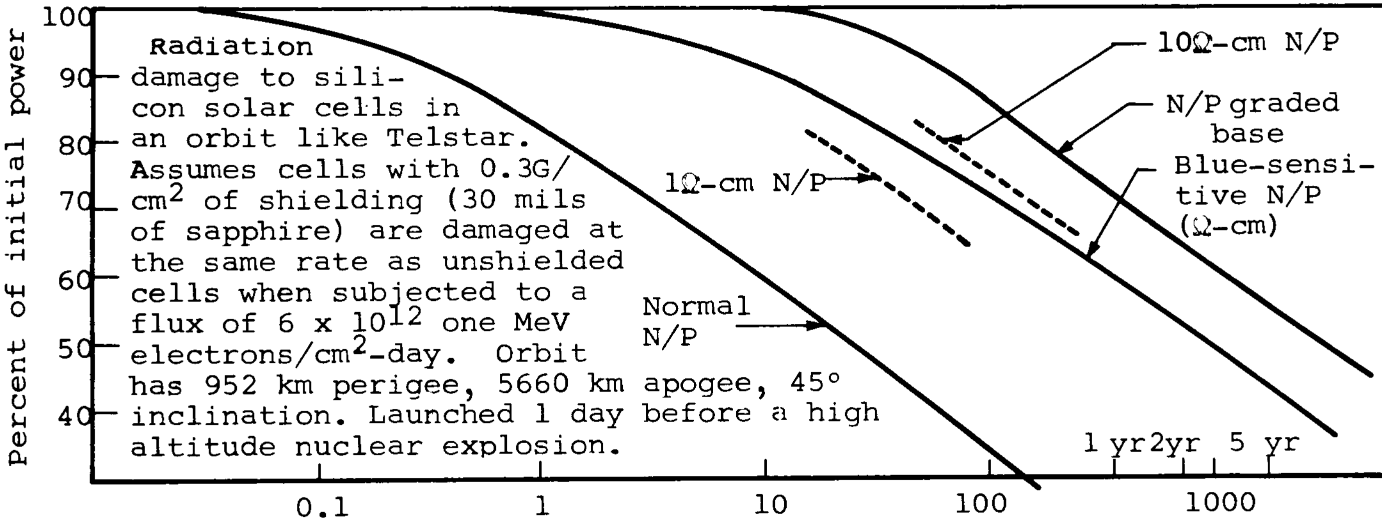


Figure 65 Radiation Damage to Silicon Cells
(From ref. 29)

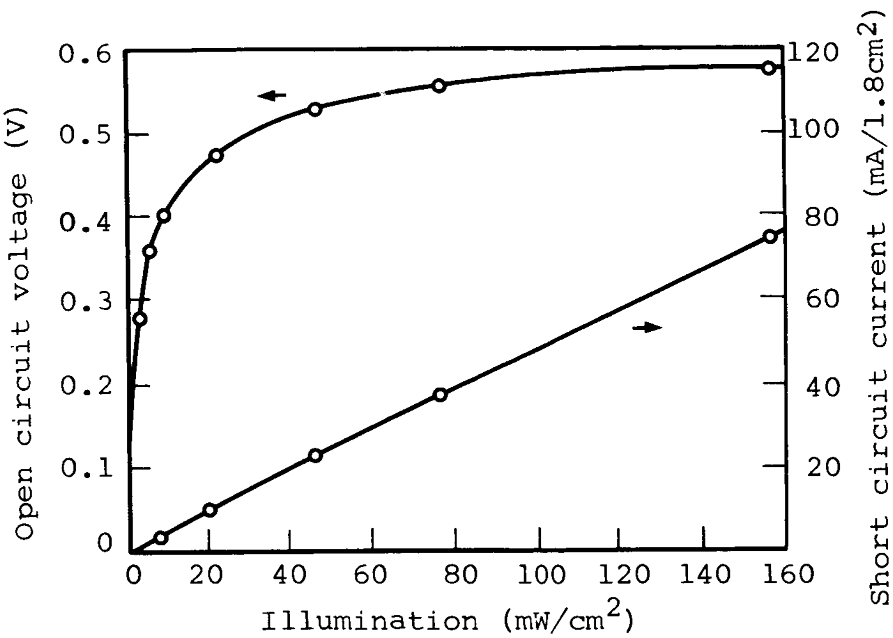


Figure 66 Typical Open Circuit Characteristics of Typical 1-ohm-cm N/P Cells

(From ref. 32)

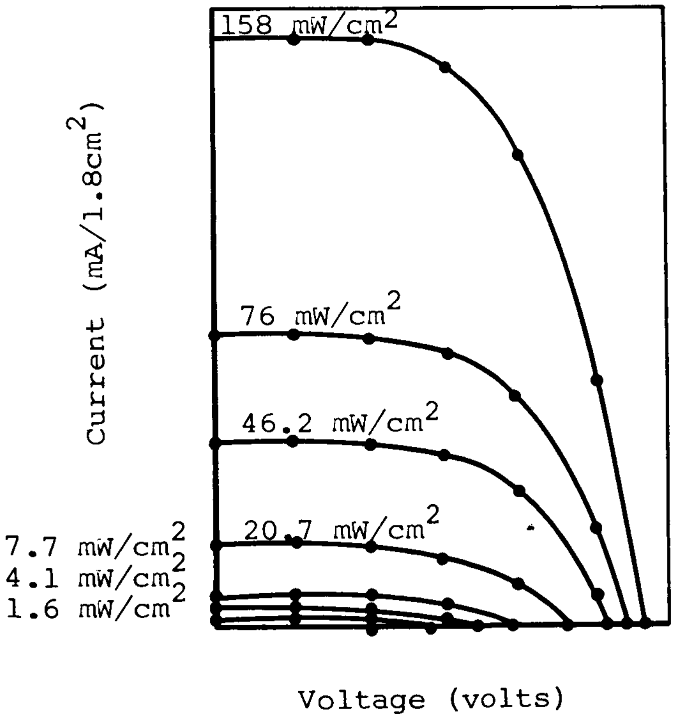


Figure 67 Typical Resistance Characteristics of 1-ohm-cm N/P Cells

(From ref. 32)

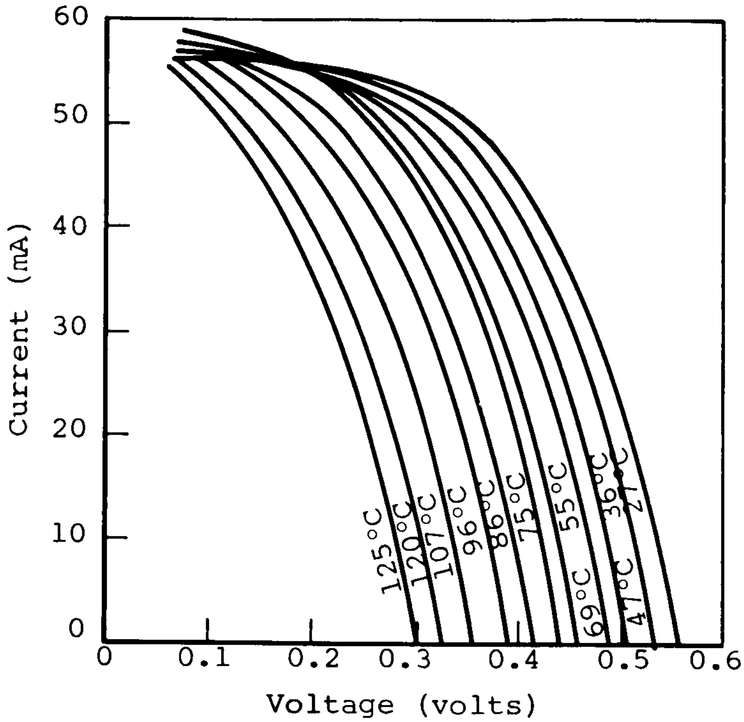


Figure 68 Temperature Characteristics of 1-ohm-cm N/P Cells

(From ref. 32)

c. Surface area: Solar cells in the multikilowatt range would certainly require very large areas. Considering operating conditions of 50°C, and sunlight at 1 A.U., present state-of-the-art silicon cells (8-mil thick, 4-cm^2 cell with 6-mil cover glass 10-5%AMZ eff) would require roughly 87 ft^2 of area per kW of cell power output, whereas CdS cell (Plastic substrate, 50-cm^2 cell, 5% AMI eff) would require roughly 220 ft^2/kW. Much larger surface area would, of course, be required if poor illumination conditions prevail and long shade duration exist (requiring additional power for storage systems). One approach to reducing large area requirements would be to use special reflectors to increase the intensity of illumination per unit area.

One of the major disadvantages of having very large surface area is inconvenience of packaging into a launch vehicle envelope. Orientation problems of solar cells would also increase significantly.

d. Degradation: This can be caused by any one of the following conditions:

- Storage
- Humidity damage
- Ultraviolet light damage
- Meteoroid damage

e. Losses in power equipment: From a weight standpoint, a solar cell power system would be one of the lightest for any mission if storage energy requirements for shade period were small. The overall weight of a solar cell power system is highly dependent on the amount of batteries required, and would vary considerably from one mission to another. Neglecting any additional power requirement for battery charging, system weights as low as 30 lb/kW have been estimated for future Earth orbital missions.

For Mars surface missions where shade periods are as long as 12 hours, and assuming Ni-Cd batteries were used for night power, the additional weight requirements for night power alone would be in excess of 2,500 lb/kW (based on an optimistic estimate of kW-hr/lb for future long mission Ni-Cd batteries). This would impose additional weight requirements on day power for battery charging.

The environmental conditions on Mars pose other serious restrictions on solar cell power sources, namely the expected cloud cover and the anticipated dust storms. Considering these factors along with the very large panel areas required for larger systems, we do not feel that the solar cell will be candidate for the Mars surface mission under consideration.

9. <u>Fuel cells</u>. - Fuel cells have made their appearance in previous space applications and will continue to be used on many short duration missions. There are many advantages to be derived from the use of fuel cells:

- Production of potable water without further processing
- Few moving parts in auxiliary equipment, resulting in high overall reliability
- Increased efficiency with decreasing power, resulting in almost zero fuel consumption at idling
- Low temperature cell operation, allowing them to be incorporated into cabins occupied by man
- Nontoxic exhaust products
- High energy efficiency
- Silent operation, reducing crew fatigue
- Modular design and construction, permitting a variety of output voltages, and the possibility of changing connections to suit the load conditions

Of the various types of fuel cells available, the hydrogen-oxygen cells have gained the most interest for manned missions, since potable water is given off as a by-product of the reaction.

Presently there are two basic types of hydrogen-oxygen fuel cells available. One is the high temperature Bacon-type system, and the other is the lower temperature membrane-type system. The Bacon-type lends itself more readily to applications with high temperature environments, whereas the membrane-type system operates below the boiling temperature of water.

Many power system application studies in the past have resulted in curves (such as Figure 12) which suggest an optimum mission duration time for fuel cell systems of approximately one week. Very little work has been done towards optimization of the performance of fuel cells for longer duration missions. These extended missions require lower specific propellant consumption, long life without loss of performance or efficiency, and ability to use standby units that can be started in space.

There are several problems associated with fuel cells for mission durations longer than Gemini or Apollo:

a. Parasitic losses: Many peripheral components, which are part of the overall power system, cause parasitic loads on the power source, reducing the thermodynamic efficiency. Components responsible for such parasitic losses are auxiliary pumps, dynamic power systems and converters and inverters necessary to produce the right kind of electrical power.

b. Heat dissipation: The radiator temperature is dependent on the type of the fuel cell used. The heat removal from the radiator depends on the environmental condition and may require specially oriented and specially designed radiators.

An additional problem is created by the extremely low temperatures during night operation, which might require external heating to maintain the cell's operating temperature.

c. Service life: The fuel cell life is affected not only by the endurance of dynamic components, but also by degradation of the cell, electrodes, electrolyte and seals.

The life of oxygen electrodes are very dependent on operating temperature, and degrade rapidly at higher temperature. On the other hand, at lower operating temperatures, the specific fuel consumption is high.

d. Fuel storage and system weight: This is the most significant problem associated with the fuel cell system for a long duration mission, since the storage system weight becomes prohibitive. Normally, either supercritical gas or cryogenic liquid fuel is used for fuel cell systems, and the use of superinsulation for fuel containers is necessary to reduce the amount of heat absorption through the container walls.

Typical cryogenic storage weights of hydrogen and oxygen in liquid form are depicted in Figure 69, and clearly indicate that storage weight penalties increase rapidly beyond a period of one month. The ratio of total storage system weight to available reactant weight is smaller for larger volumes. This is because the losses due to heat absorbed is proportional to surface area and not the volume, and for larger volumes, the ratio of surface area to volume is smaller. These figures are based on spherical storage containers and heat leak rate into storage container of 5 Btu/hr.

In addition to the storage weight requirements, the estimated total power system weight (excluding storage) for different occupancy times are shown in Figure 70 for different system output requirements. It is shown that a 4 kW system, weighing approximately 1400 lb for zero occupancy, requires additional weight of 13,600 lb for occupancy time of 120 earth days. The approximate weight of available reactants (ref. 9) can be estimated to be

$$4 \text{ kW} \times 120 \times 24 \text{ hr} \times 1.0 \text{ lb/hr} = 11{,}520 \text{ lb}$$

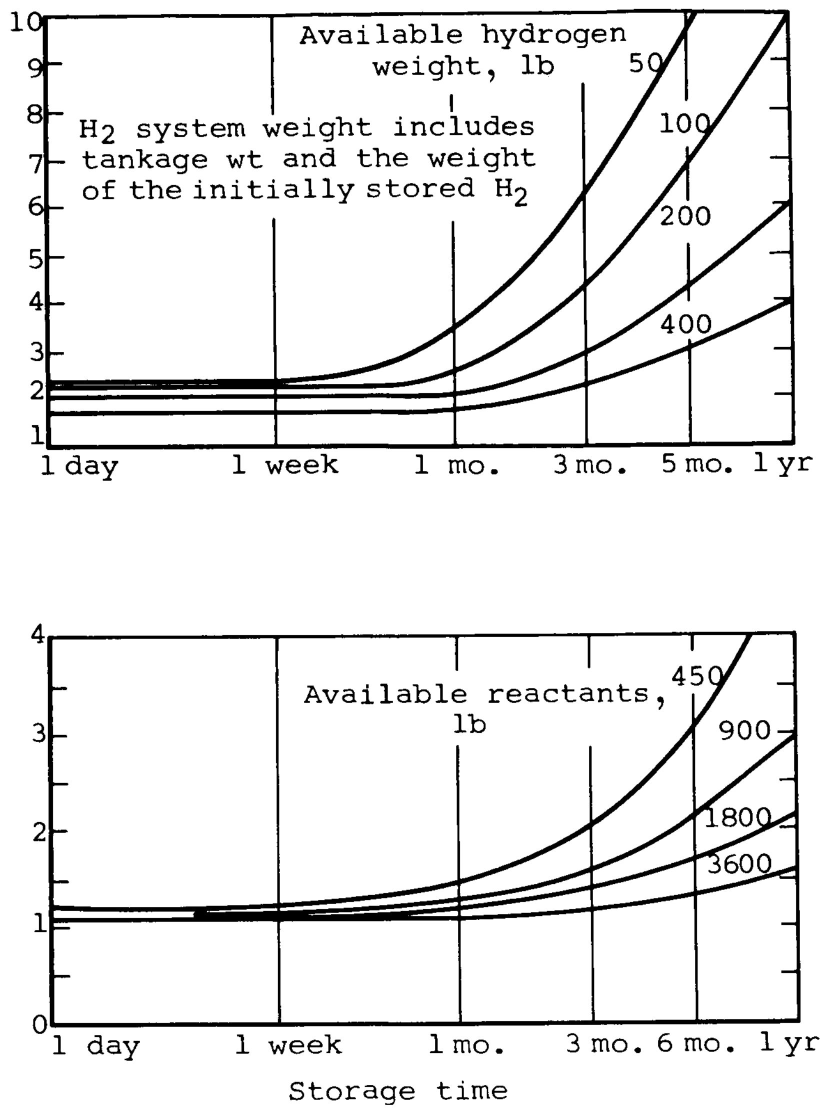


Figure 69 Cryogenic Storage Weights
(From ref. 9)

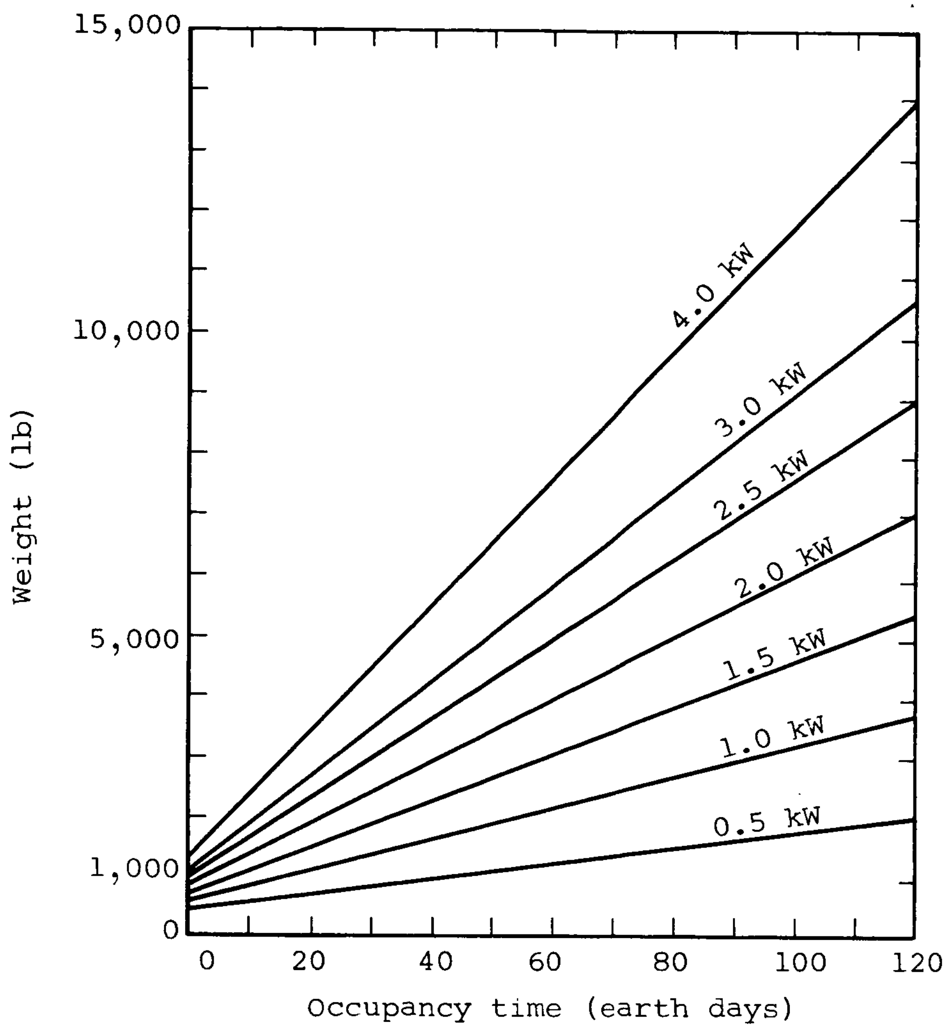


Figure 70 Fuel Cell Electric Power System Weight
(From ref. 9)

Furthermore, additional increases in weight due to storage losses of reactants prior to occupancy time, must also be considered in evaluating overall system weight requirements. It is interesting to note, however, that if such a system were available for use over a period of 120 Earth days, as much as 11,000 lb of potable water could be available as a by-product for astronaut usage.

Although a large amount of potable water is produced for extended missions, the initial system weights (fuel storage system weight plus weight of stored fuel) for a 6 month mission are much too high. For mission durations in excess of 60 days, the fuel cell will continue to be unattractive when compared with dynamic systems.

10. Batteries. - The battery is the main energy storage and conversion device used in today's space missions. The silver-zinc primary battery (the main one used today) delivers from 30 to 90 W-hr/lb depending on its discharge rate. This means that approximately 10 to 35 lb of batteries must be supplied for each kW-hr delivered. This is much too high to be considered as a primary power source for the Mars mission under consideration. The silver-zinc battery has very poor charge-discharge characteristics and a low cycle life (as shown by Figure 71), and as a result is not seriously considered for secondary or rechargeable use. For these applications, the less energetic silver-cadmium and nickel-cadmium systems must be used. They both exhibit good charge-discharge characteristics, however, Figure 72 shows that if long cycle life at low energy densities is required, the nickel-cadmium battery should be used.

The relationship between battery life and severity of service is extremely important. For long duration missions, the demands on the battery must be relaxed accordingly, whereas on short term missions, battery service conditions may be more stringent.

In choosing a suitable battery, another very important factor is the operating temperature. The nickel-cadmium cell has the best cycle life, but just as the other cells, it is limited as to the range of temperatures over which it can operate; this fact is shown in Figure 71. It can bee seen that the cycle life of the nickel-cadmium cells drops off very sharply for operating temperatures above 100°F or below 30°F.

Specific test data for the three cells under consideration is presented in Figures 73, 74 and 75. Figure 73 shows again the excellent cycle life possible at about 75°F with nickel-cadmium cells. A fairly high capacity (12 A-hr) is obtained for a low discharge rate (25%). For higher temperatures and higher discharge rates (assuming the same cell capacity), the number of cycles possible drops off very sharply.

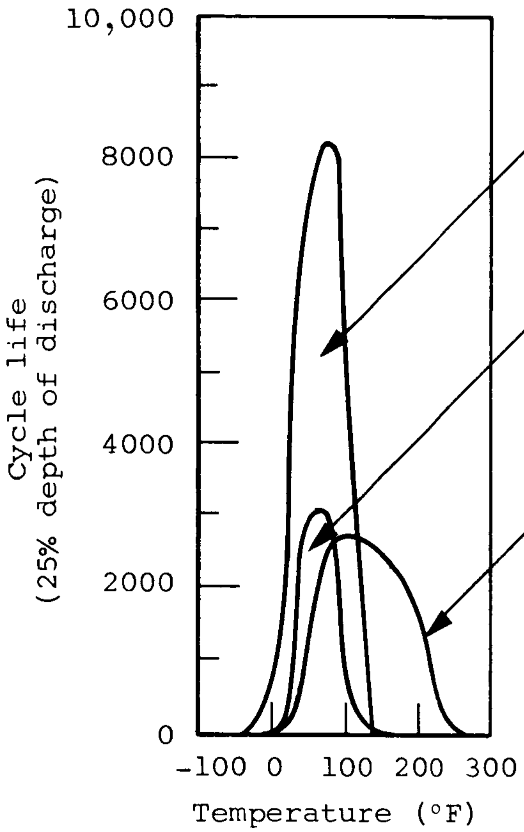


Figure 71 Comparison of Cycle Life of Candidate Batteries

(From ref. 11)

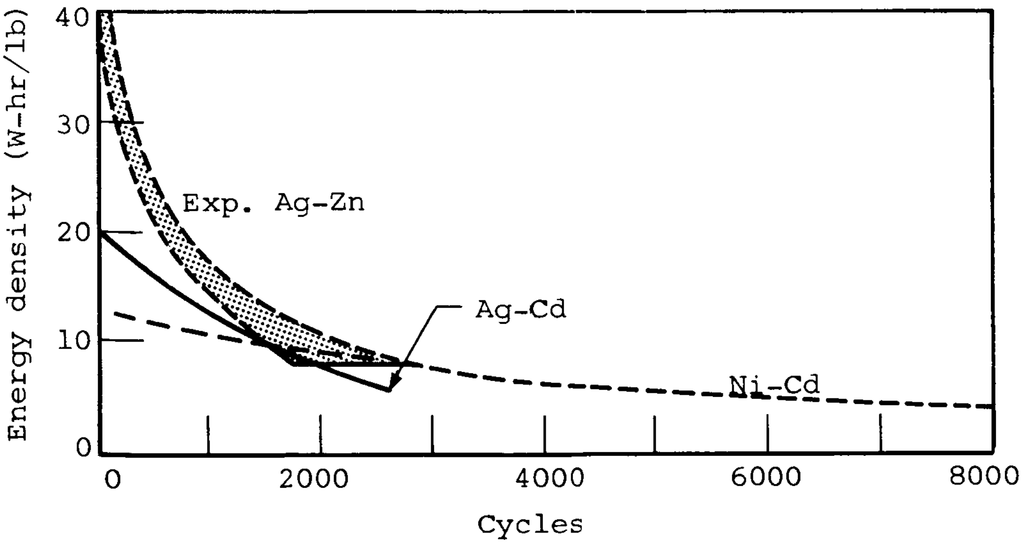


Figure 72 Comparison of Energy Density vs. Cycle Life

(From ref. 11)

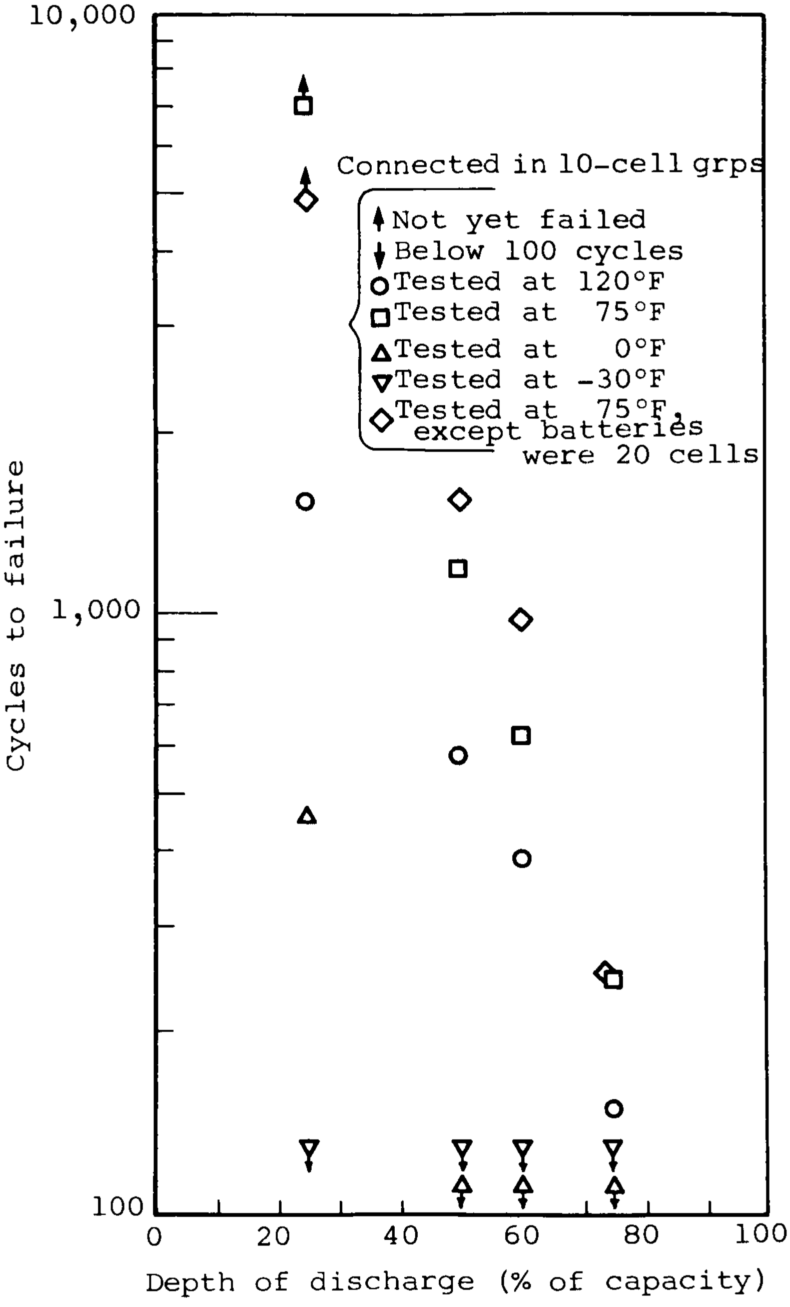


Figure 73 Cycle Life of Nickel-Cadmium Cells
(From ref. 33)

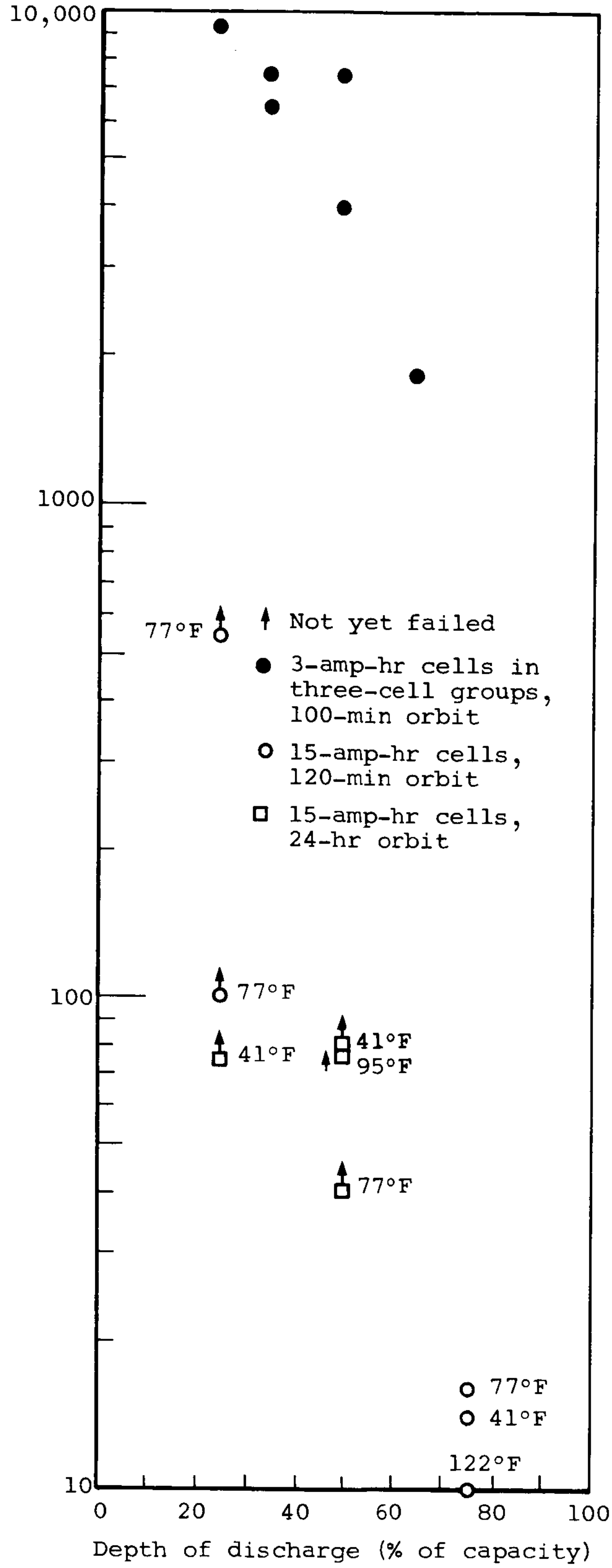


Figure 74 Cycle Life of Silver-Cadmium Cells
(From ref. 33)

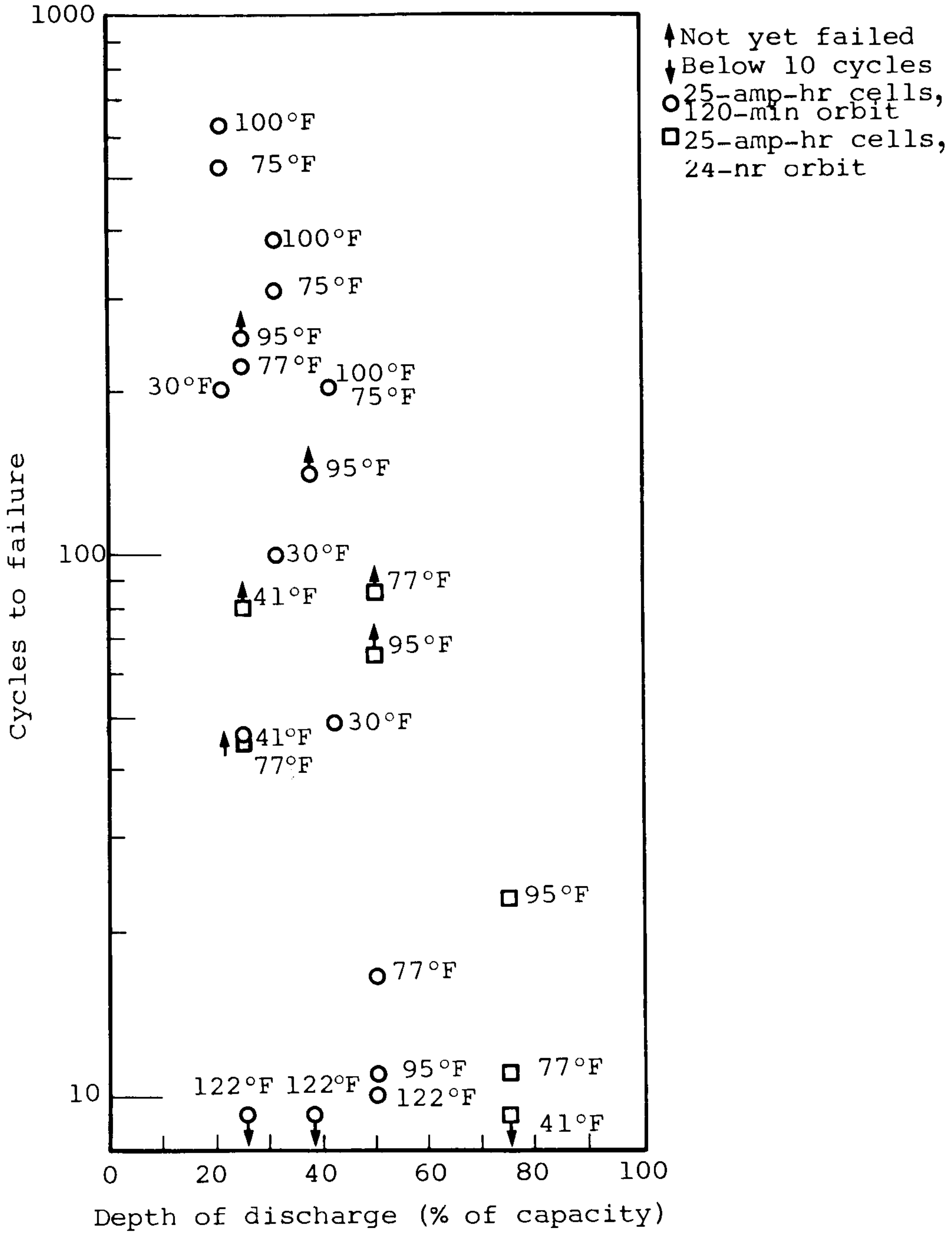


Figure 75 Cycle Life of Silver-Zinc Cells
(From ref. 33)

Figure 74 points out the fact that an excellent cycle life can be obtained for silver-cadmium cells for high discharge rates (50%) provided the cell capacities are low (3 A-hr). For a cell capacity equivalent to that under which the nickel-cadmium cells were tested (12 A-hr), the cycle life for a silver-cadmium cell is reduced by a factor of ten.

The very poor cycle characteristics of the silver-zinc cells can be seen in Figure 75.

Although the battery cannot be considered as a primary source of power, it still remains a better candidate than fuel cells for secondary and emergency power, since no cryogenic storage or start-up problems are involved. The nickel-cadmium battery is the one proven reliable battery system for long-term service with rapid recharge capability, therefore, we recommend its use for peak load and emergency conditions for the Mars surface mission being considered.

7. THERMAL INTEGRATION

The thermal energy requirements for the life-support subsystems can be supplied by electrical power from the module electrical power system, but it is evident that thermal to electrical to thermal power conversion is expensive and inefficient.

Although electrical heating is a straightforward method of obtaining the required process heat, the resulting increase in the electrical power requirements would increase the size of the total power system. Since the life-support processes are little affected by the source of the heat, the use of heat from the basic power system for the endothermic life-support processes would improve the performance of the total system.

Life-support systems for manned spacecraft operate nearly continuously for the entire mission, and many life-support system functions use thermal energy directly. Others would use thermal energy if it were available at high temperature, or low penalty, or both. It is usually inefficient, and costly in weight to generate energy in other forms and then convert it to thermal energy. Thus, it appears advisable to investigate all life-support functions to determine whether the processes could be accomplished by the direct use of thermal energy.

A good case can be made for the substitution of direct thermal energy wherever moderate temperature electrical heaters are presently used in life-support systems. Typical of these applications would be:

- Thermal cycling of molecular sieve beds to desorb CO_2
- Water recovery from urine and wash water
- Decomposition of solid wastes
- Heating of the spacecraft

The following are some of the advantages that can be achieved through the use of thermal integration:

- Reduction in electrical power required
- Reduction in heat required by the power system
- Reduction in radiator size and weight
- Reduction in isotope quantity

It would be very desirable to use heat from the life-support system exothermic processes as thermal inputs to other endothermic life-support processes, however, there is only one exothermic process; the Sabatier reaction. The control of the Sabatier reactor temperature is very critical for achieving high conversion efficiency. Thus, variations in reactor heat loading could not be tolerated and, hence, the use of this reaction heat is not recommended.

Another possible source of thermal energy would be from the electronic components. This, however, would be at a low temperature and in a nonconcentrated form, thus requiring large amounts of hardware at very low efficiencies to collect and redistribute this energy. Hence, this source was not given serious consideration.

The best source of thermal power can be found in the power generating system. A model power generating system with an overall efficiency of η is presented in Figure 76 as three elements: the thermal source, the power converter, and the radiator. If the electric output of power converter is kW_e, then the amounts of thermal power available from the source and at the radiator are kW_e/η and $[(1-\eta)/\eta]\,kW_e$, respectively. Thus, there are two possible sources of life-support process heat; these are designated by points A and B in Figure 76.

It is more desirable to utilize the thermal energy that would be dissipated by the radiator than to utilize heat directly from the thermal source, since this would result in an increase in the source weight, volume, power and cost. We will show later that the electrical power generating system, sized to perform only functions other than life support, provides enough waste heat at point B to run the required life-support systems.

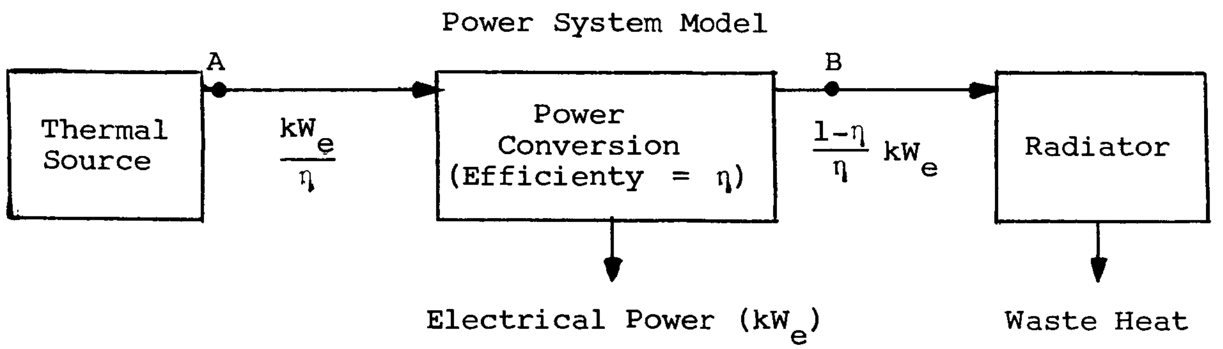


Figure 76 Thermal Energy Sources

A. Nonintegrated Life-Support System

If no thermal power were used in the life-support system, the electrical power output of the energy conversion system would have to provide 8.22 kW of thermal and electrical energy. This figure is derived by summarizing the power required for each module subsystem, as indicated in Table XXV. It can be noted here that of all the life-support processes, the regeneration of O_2 from CO_2 requires by far the largest power input. Over half of the total power required is needed for this purpose alone.

B. Integrated Life-Support System

Consideration of a thermally integrated life-support system first requires the selection of points in the life-support system to which thermal integration with the waste power loop can be accomplished.

A thermodynamic analysis of the Rankine cycle configuration selected shows that an air inlet temperature of 345°F will be available at the waste heat exchange point. Processes that operate at a temperature level much higher than approximately 300°F should, therefore, not be considered for thermal integration with the waste heat loop.

Table XXVI shows the operating temperatures of the various life-support functions. It can be seen here that the processes to be considered are:

- CO_2 concentrator desorbing operations
- Both water recovery evaporators
- Feces dryer unit
- Food preparation
- Hygiene water heating

The pyrolysis step in the metabolic-urine water recovery cycle is at much too high a temperature for thermal heat input from the waste heat loop.

The final thermally integrated configuration would be as shown in Figure 77. Several heat transport fluids are available for prolonged operation without degradation. Of these, ethylene glycol exhibits the best properties; however, it is toxic if ingested. Since the transport fluid is exposed to the water supply, another suitable fluid must be used. NASA Langley has found that DC-331, a silicone fluid produced by the Dow Corning Company, exhibits the most suitable characteristics for heat transport purposes.

TABLE XXV

NONINTEGRATED POWER REQUIREMENT

Process	Input Power, Btu/hr	Input Power, W
CO_2 Concentration	9,593	2,810
CO_2 Reduction - O_2 Generation	3,755	1,100
Contaminant Control	273	80
Atmospheric Circulation	1,020	296
Thermal Control	4,090	1,200
Wash Water Recovery	610	179
Metabolic & Urine Water Recovery . .	672	197
Feces Processing	584	170
Food Preparation	154	45
Hygiene	225	66
Electronic Equipment and Lighting . .	6,800	1,990
Module Control Instrumentation . . .	290	85
TOTAL	27,656	8,218

TABLE XXVI

LIFE-SUPPORT SYSTEM - ENDOTHERMIC POWER REQUIREMENT

Subsystem	Operation	Temperature Level, °F	Average Endothermic Power, Btu/hr
CO_2 Concentration	Desiccant and Sieve Desorbing	250	8,363
Metabolic & Urine Water Recovery	Evaporator Pyrolysis	120 1,800	467 225
Wash Water Recovery	Evaporator	120	610
Solid Waste Management	Dryer	200	584
Food Preparation	Food Preparation	180	130
Hygiene	Water Heater	180	205
		TOTAL	10,584

Using the figures just presented, the overall energy balance for the MSM is shown in Table XXVII. These figures represent nominal design conditions; maximum and minimum design conditions were not considered in our analysis. The electrical power requirement after thermal integration of the life-support subsystems is shown in Table XXVIII. Table XXIX shows the effects of thermal integration on the power system requirements.

The isotope source power required was obtained by using a heat block transfer efficiency of 92.8 percent and an overall Rankine conversion efficiency of 22 percent. Using this conversion efficiency, the formula for waste heat is given in Figure 76. For the integrated configuration, the waste heat available is 56,200 Btu/hr. Only 10,584 Btu/hr are needed in the life-support processes. The remaining heat must be dissipated external to the module as shown in Figure 77.

The analysis presented shows the desirability of employing thermal integration for the life-support functions, and stresses the importance of utilizing this approach in all future extended manned missions to optimize system power and weight.

8. CONCLUSIONS

The primary objective of this study in advanced life-support technology was to examine the effects of the Mars surface conditions on the selection and application of life-support and related systems. For study purposes, a hypothetical four-man, six-month mission profile was chosen, which we feel closely approximates an eventual mission, since the total travel time involved is over one year, and the number of functions to be performed requires at least four men. The imposition of these typical mission constraints allowed us to examine the various disciplines involved in much greater depth. Our assessment of the current research and technology was projected into the 1980 period and was particularly concerned with the extended mission characteristics of the systems considered, and their ability to function in the Mars environment.

The subsystem selections were made using the following general order of importance for the tradeoff variables: (1) reliability, (2) weight, (3) power, (4) volume, and (5) maintainability. Since the applied evaluation technique was purely qualitative, the order of importance assigned to the variables could not be followed explicitly nor could the amount of emphasis on each variable be consistently applied for each particular tradeoff. We feel that all the individual

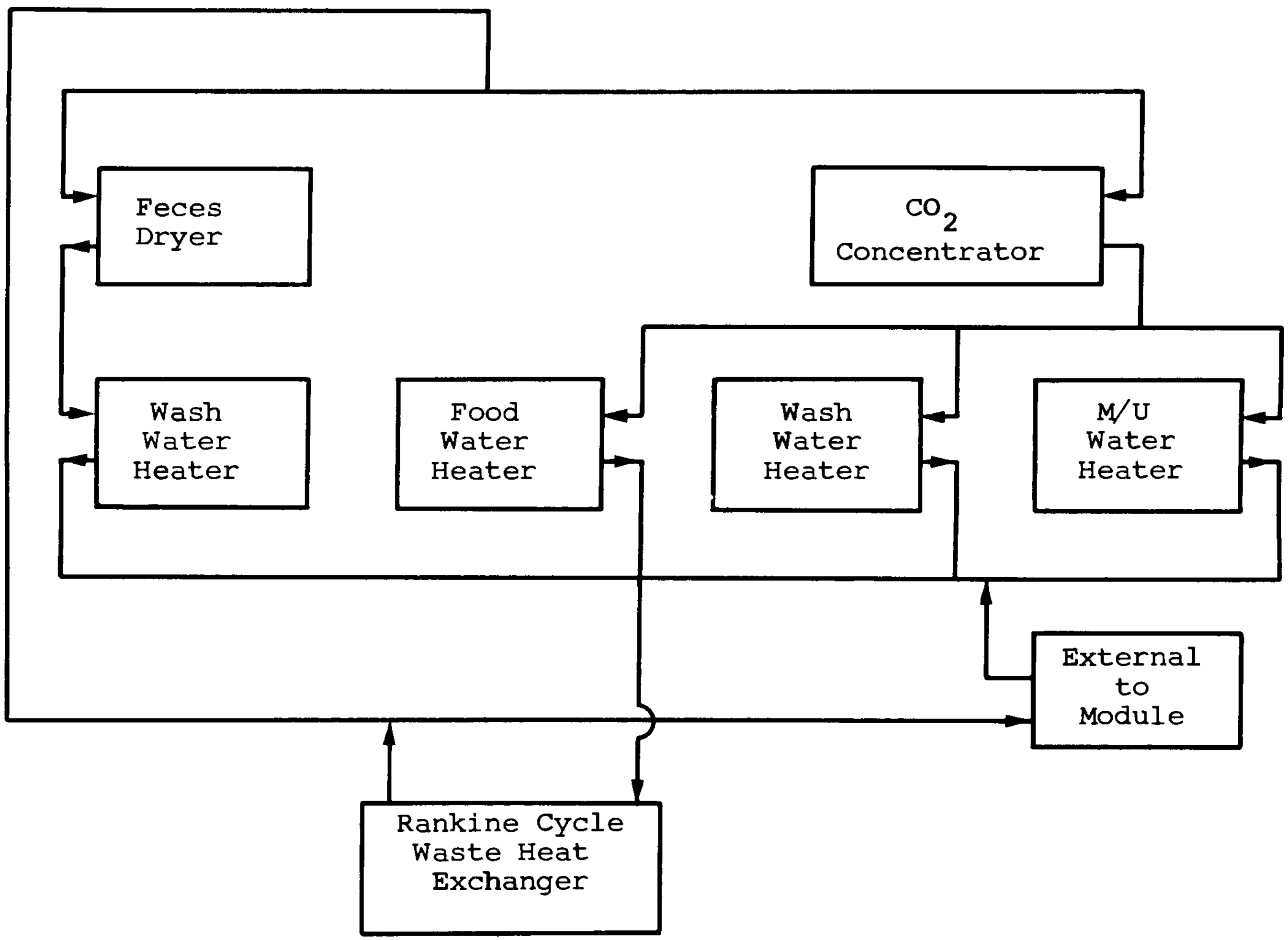


Figure 77 Process Heat Flow Diagram

TABLE XXVII

HEAT LOAD CHART - INTEGRATED SYSTEM

	Heat Source, Btu/hr			Process[c]	Heat Transport, Btu/hr		
	Metabolic Heat	Electrical Heat	Power System Waste Heat	Heat of Reaction	Air (ECS) Circuit	Coolant Circuit	Mars Atmos
1		1,230	8,363	CO_2 Concentration	5,083	4,510	
2		3,755		+648 CO_2 Reduction - O_2 Generation -2070	1,333	1,000	
3		273		Contaminant Control	273		
4		1,020		Atmospheric Circulation	1,020		
5		4,090		Thermal Control	1,360		2,730
6			610	[a]Wash Water Recovery	48	562	
7		205	467	[b]Metabolic & Urine Water Recovery -205	185	282	
8			584	Feces Processing	584		
9		24	130	Food Preparation	144	10	
10			225	Hygiene	205	20	
11		6,800		Electronic Equipment and Lighting	2,530	4,270	
12		290		Module Control Instrumentation	290		
13	1,830			Personnel	1,830		
TOT.	1,830	17,687	10,379	+648 -2,275	14,885	10,654	2,730

[a]4.1 Hours Operations

[b]7.0 Hours Operations

[c]Heats of reaction for the appropriate processes are given in Btu/hr: plus(+) for exothermic reactions and minus (-) for endothermic reactions.

TABLE XXVIII

MSM AVERAGE ELECTRICAL POWER REQUIREMENT

Unit	Average Electrical Power,	
	W	Btu/hr
CO_2 Concentration	361	1,230
CO_2 Reduction - O_2 Generation . . .	1,100	3,755
Contaminant Removal	80	273
Atmospheric Circulation	300	1,020
Metabolic & Urine Water Recovery	60	205
Food Preparation	7	24
Module Control Instrumentation . .	85	290
Thermal Control	1,200	4,090
Electronic Equipment & Lighting . .	1,990	6,800
TOTAL	5,183	17,687

TABLE XXIX

MSM POWER SYSTEM COMPARISON

Function	Dow A - Rankine Power System	
	Nonintegrated	Integrated
Electric Power . . . . W	8,218	5,183
Isotope Source Power . . . . kW_t	40.3	25.4
Isotope Source Weight . . . . lb	614	368
Power Generating System Weight . . . . lb	2,700	1,707
Waste Heat Rejection . . . . kW_t	29.1	18.4
Power System Weight . . . . . . lb	3,315	2,076
Life-Support System Electrical Power . . . . W	5,110	2,080

systems selected represent very good choices based on the imposed mission constraints; however, we are not able to say, in general, that the selected total subsystem configuration represents an optimum choice. To do this, we would have to consider all candidate subsystem combinations, which is much beyond the scope and purpose of this study.

A serious attempt was made to utilize any and all available Mars natural resources that would result in a significant material savings. There are three major functions that could be performed by extraterrestrial raw materials:

- rocket fuel source materials
- construction materials
- raw materials for life support

We have concluded that the information presently available is much too limited and inconclusive to project the availability of either of the above raw material possibilities. We expect the information to become much more conclusive in the coming years of Mars exploration. If raw materials are found to perform either of the functions mentioned, this will require a complete reevaluation of the problem.

The major conclusion drawn from this study is that the life-support and power system configuration which might be applicable for Mars orbital and interplanetary phases of the Mars exploration plan, will not be that most suitable for the proposed Mars surface operation. This important theme results from our analysis and current knowledge of the Mars environment and its effects on the performance of appropriate life-support and power equipment. The following conclusions relate more specifically to the major life-support subsystem areas, and indicate the main factors considered in choosing particular subsystem elements.

A. Power Source

An isotope power source has the most desirable properties for the mission envisioned. It has distinct advantages over solar cells or solar collectors, since it is anticipated that Mars will have a cloud cover and dust storms, which reduce the power output of these systems.

1. Isotope. - Plutonium-238 is considered the best of the isotope candidates, primarily because of its low shielding requirement and long half-life which results in a relatively constant power supply for the duration of the mission.

We strongly feel that all necessary steps toward increased production of Pu^{238} should be taken immediately to assure adequate availability.

2. Nuclear. - The nuclear (reactor) power source is highly competitive with the isotopic source for extended missions. For the power level (5 kW) and mission duration considered (six months), the isotope source maintains a slight advantage from the standpoint of weight. However, as the required power level approaches 10 kW, the nuclear source becomes highly competitive and would have to be considered more carefully.

B. Energy Conversion System

1. Dynamic. - The Rankine cycle is selected as the energy conversion system most compatible with Mars surface use. In general, the Brayton cycle is preferred over the Rankine cycle for orbital and deep space applications, primarily because of the phase separation problems associated with a two-phase working fluid in a zero-g field. This problem is of little concern for the surface application, and as a result, the other advantages of the Rankine system become significant. These are:

- Reduced weight of the isotope source; in comparison, the Brayton system requires fuel capsule-to-gas heat exchangers resulting in larger and heavier radiation shields

- Reduced radiator area for fluid heat transfer as compared with gaseous heat transfer

- Reduced integrated system weight for fluid heat transfer as compared with gaseous transfer

Dow-Therm-A is selected as the working fluid in the Rankine cycle, primarily because of the unique positive slope of its saturated vapor line on the temperature-entropy plot. This characteristic assures turbine operation in the superheated region at lower turbine inlet temperatures.

2. Static. - The static conversion technique that holds the most promise for future use in Mars exploration is the thermionic converter. These devices are in a very early stage of development, and as a result, it is very difficult to project their availability. If a major breakthrough can be made toward increased efficiency, this system will become highly competitive with dynamic conversion techniques.

C. Secondary Power

The required weight of batteries or fuel cells is far too high for consideration as a prime power source. The battery exhibits much better charge-discharge characteristics than does the fuel cell, and thus, is highly compatible with intermittent power demands. Of the battery candidates, the Ni-Cd cell has the longest cycle capacity, and thus is selected as a secondary power source for peak loads and emergency power.

D. Life Support

1. Degree of Closure. - For the mission considered in this study, a fully closed ecological cycle is highly impractical. It also seems very unlikely that a process for generating food from bacteria or feces will be available or necessary by the mid 1980's, since the conventional freeze-dry food packaging techniques are well developed and well suited for this application.

It is also impractical to consider reclamation of the water present in the form of feces, because of the low water quantities present and the numerous difficulties involved in handling this material. We do feel, however, that the wash water and the water present in the form of urine will have to be reclaimed to avoid excessive weight penalties.

In addition, oxygen will have to be reclaimed from the carbon dioxide that will be generated throughout the mission to avoid excessive oxygen storage weight and volume penalties.

2. Water Management. - Through utilization of an efficient water reclamation system, maximum weight savings can be realized. Under the assumed mission conditions (720 man-days), a non-regenerative system would require

about eight tons of water. This far exceeds the weight penalties of all the other life-support systems combined.

Because of low system weight and simplicity, a distillation process is selected to convert the astronauts' fluid wastes (wash water, urine, and condensate) to potable water. The wash water collection sources should be separate from the urine and condensate since an additional vapor pyrolysis step is required for urine.

3. Carbon Dioxide Management. - From the standpoint of power utilization, carbon dioxide reduction and oxygen regeneration are the major considerations under life-support processes. The total function of concentrating and reducing carbon dioxide involves nearly half of the total power required for the module, and as a result, is considered quite carefully in this study. The pure Sabatier cycle is selected as most candidate for carbon dioxide reduction because of the very high conversion efficiencies, low power requirement, and continuous operation capability.

The Sabatier system would be used in combination with a four-bed molecular sieve-synthetic zeolite carbon dioxide concentration unit. This unit has undergone extensive testing and has performed very well for extended periods.

a. Alternate indirect processes. - The major problem associated with alternate indirect carbon dioxide reduction processes such as the Bosch cycle or the Sabatier cycle with methane decomposition, is that of carbon build-up on the catalyst. To make these cycles candidate for the mission envisioned would require the development of a carbon handling technique that would permit continuous and efficient system operation.

b. Direct processes. - Direct processes for carbon dioxide reduction, such as thermal decomposition, the molten carbonate cell, and the solid electrolyte cell, are expected to have too large a power penalty in comparison with the pure Sabatier cycle. Hence they are not as desirable for the proposed mission.

E. Module Design

The module configuration evolved in this study is used primarily as an aid in giving a better sense of the volumes and weights of specific subsystems relative to the total module volume and weight. As an outgrowth of this study, two very important conclusions can be drawn:

- The total module weight and volume derived in this study is well within the state-of-the-art in rocket development and is capable of being launched by a post-Saturn booster.
- From the standpoint of volumetric efficiency, a very heavy emphasis should be placed on low profile designs in life-support equipment; a package height of approximately two feet would be desirable.

9. RECOMMENDATIONS

Although the interrelating factors were analyzed for one arbitrary mission profile, we feel that there are several general recommendations that can be made for the total Mars exploration plan:

- Necessary steps to increase production of Pu^{238} should be taken immediately to insure adequate availability.
- More extensive development of the Rankine cycle, using Dow-Therm-A as the working fluid under simulated Mars surface conditions, should be started immediately.
- Thermionic systems development should be increased in the hope that a major technological breakthrough will occur.
- In general, life-support component design should be directed toward lower profiles; a package height of approximately two feet would be desirable.
- An increased effort in the area of water management such as shower suit development for minimal water quantities would result in a very large weight savings.

This study represents an initial and somewhat limited step in the area of subsystem definition for extended Mars missions. The next phase of study should be appropriately expanded so as to permit a quantitative approach to the tradeoff process. The objectives of the next phase of study should include the following:

- Analyze all candidate life-support subsystem configurations to make an optimum choice
- Establish order of candidacy for various life-support subsystem configurations
- Examine all probable Mars surface mission profiles
- Examine variations in the order of preference and importance of the various tradeoff variables

The objectives could be accomplished by the development of a numerical search technique that would permit rapid selection of optimum life-support components by a digital computer.

Although the present preliminary study is not completely comprehensive, we feel that the technological base formed will serve as a primary source of input to future research programs involving manned Mars surface operations, and will permit NASA to review, in advance, the proposed areas of future activity that will materially enhance progress in this field.

APPENDIX A. MODULE DESIGN DETAILS

1. Geometric Shape

Since the structure is to be a pressurized enclosure, practical considerations dictate that its exterior surfaces be curved. The shapes discussed in the following paragraphs are presented in Figure 78.

A complete sphere is the most efficient pressurized shape and offers a maximum of volume with a minimum of surface area. The sphere also makes single-point contact with the supporting surface, and hence, very little site preparation is required. The internally-concentrated loads on a sphere must either be supported by the surface material or by the internal framework which then transmits the loads to the area of ground contact. Although the sphere has the highest ratio of internal volume-to-surface area, the interior shape does not permit the volume to be used very efficiently from anthropomorphic considerations (ref. 20).

The hemispheric cylinder has a much more efficient internal configuration, and loads could be transmitted directly to the surface; however, it requires very extensive site preparation. A circular cylinder has a nominal line contact on the ground, so the amount of site preparation would be small, and the number of degrees of instability is also less than the sphere. The circular cylinder, however, suffers from the same deficiency as the sphere in that the interior volume cannot be used efficiently. When adequate floor and ceiling area are obtained, large amounts of wasted volume exist above and below the living volume. All other characteristics of the circular cylinder are ideal for module construction, hence the problem is reduced to one of finding a shape that possesses all the attributes of the circular cylinder, but has a more efficient interior volume.

This line of reasoning leads quite naturally to the elliptic cylinder where large floor and ceiling areas can be obtained with very little wasted volume. Since access must be provided to the equipment above and below the living volume, the so-called wasted volume becomes even smaller.

The scalloped cylinder appears to have the same advantages as the elliptic cylinder, however, there is an added line contact area, and the equipment volume is seriously reduced.

As a result of this analysis, the elliptic cylinder was selected as the candidate geometric shape.

The bulkhead configuration was established on the basis of locating the power supply a safe distance from the astronauts,

(1) Sphere

(2) Hemispheric Cylinder

(3) Circular Cylinder

(4) Scalloped Cylinder

(5) Elliptic Cylinder

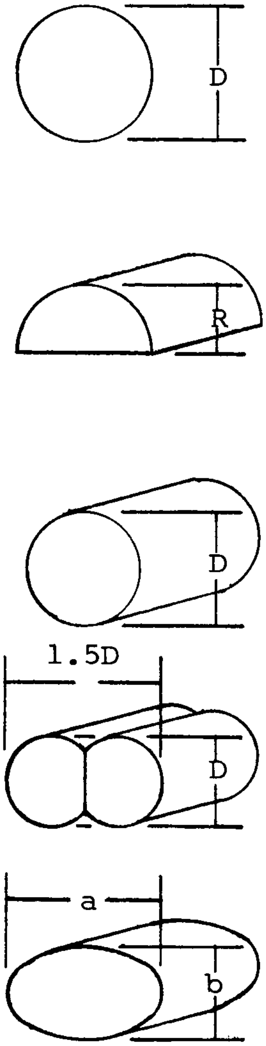


Figure 78 Module Geometry

and preventing the airlock and entrance from occupying a large portion of the living volume.

2. Structural Configuration

The structural weight of the elliptic cylinder configuration is based on cabin pressurization stresses, thermal balance, and structural capacity.

Meteoroid impact penetration does not present a problem on the surface of Mars (ref. 19). The Martian atmosphere will dissipate these particles much as the Earth's atmosphere does. The radiation hazards on Mars should also be in the same proportion as those on the Earth. Mars has lower atmospheric density, but is a greater distance from the Sun.

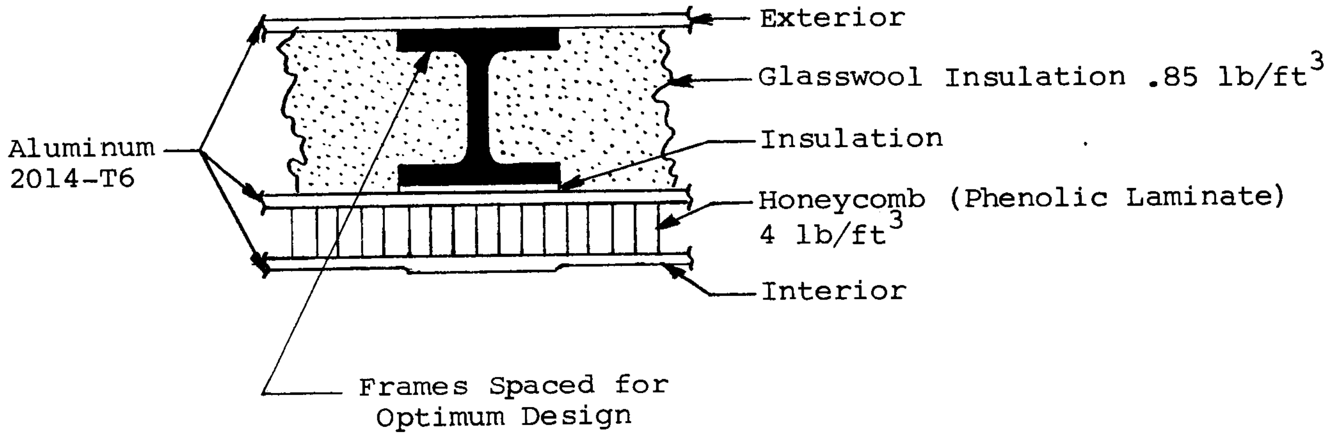
A weldable aluminum alloy, 2014-T6, was selected for the structure. In addition to ease of fabrication, this material is readily available, low in cost, and has an excellent strength-to-density ratio. A honeycomb sandwich material was selected for analysis, since it is very efficient for pressurization, body bending, and axial loads (ref. 20).

The circumferential moment sustained by the elliptic cylinder cross section must be carried by frames to obtain an optimum design. The module wall shown in Figure 79 was designed to carry an interior pressure of 10 psi (ref. 34). A relief valving system between the airlock and the interior volume will assure maintenance of a normal operating pressure of 7 psi.

All calculations made in arriving at the final configurations are presented in Appendix B.

The shell of the structure serves the dual function of a thermal and vacuum protection. It also acts as the structural system from which all stationary interior furnishings are mounted. In this manner, the interior partitions become essentially nonload bearing and thus, are held to a minimum in thickness and weight (ref. 35). Hence, the walls, floors and ceilings are sandwich sections made up of aluminum skins with a filler of high-density polyurethane foam. Edges are reinforced with channels for loadbearing capacity, and the floor sections have a soft vinyl coating.

The final module configuration is detailed in Figures 80 through 83.



Scale: None

Figure 79 Basic Wall Structure

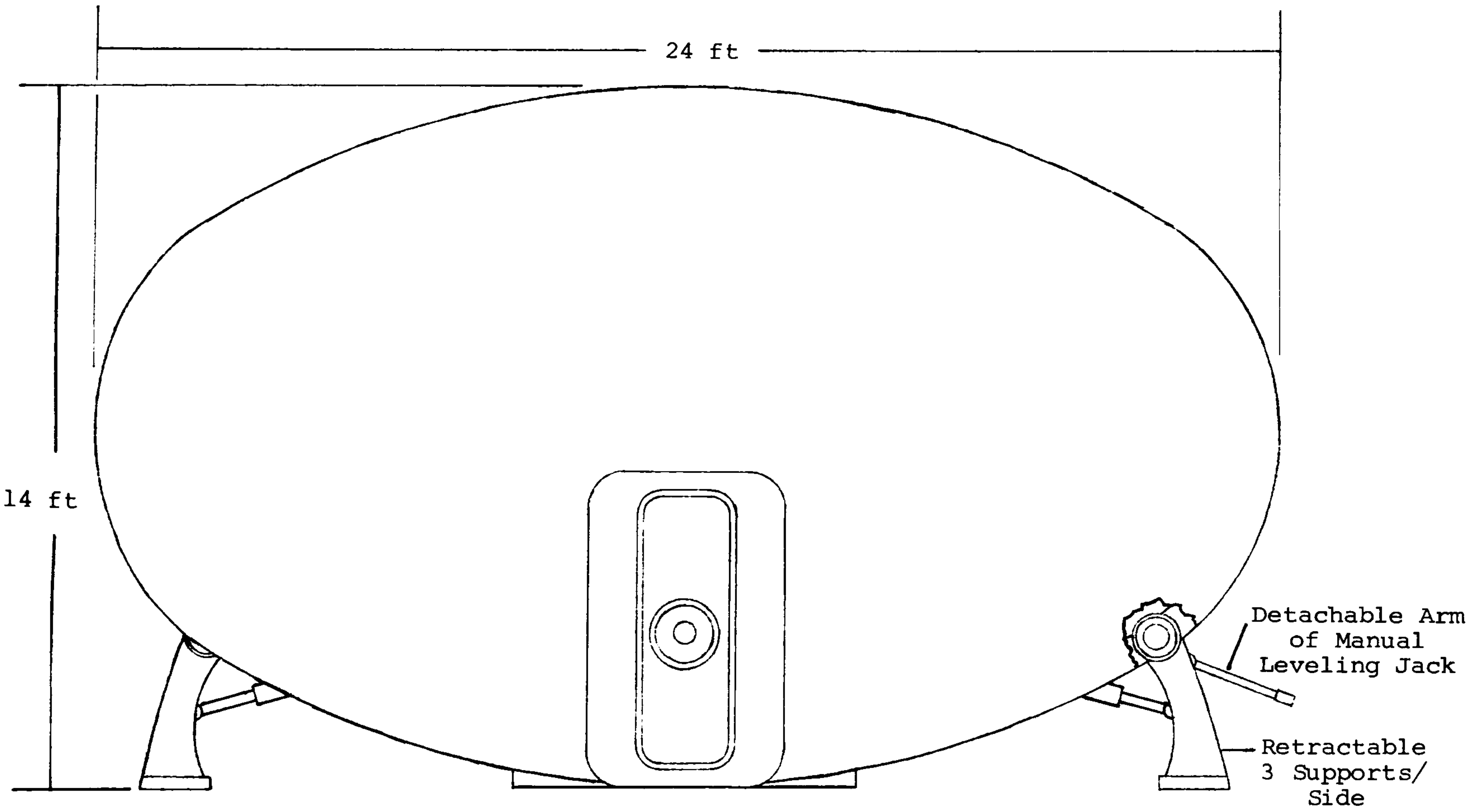


Figure 80   Front Elevation

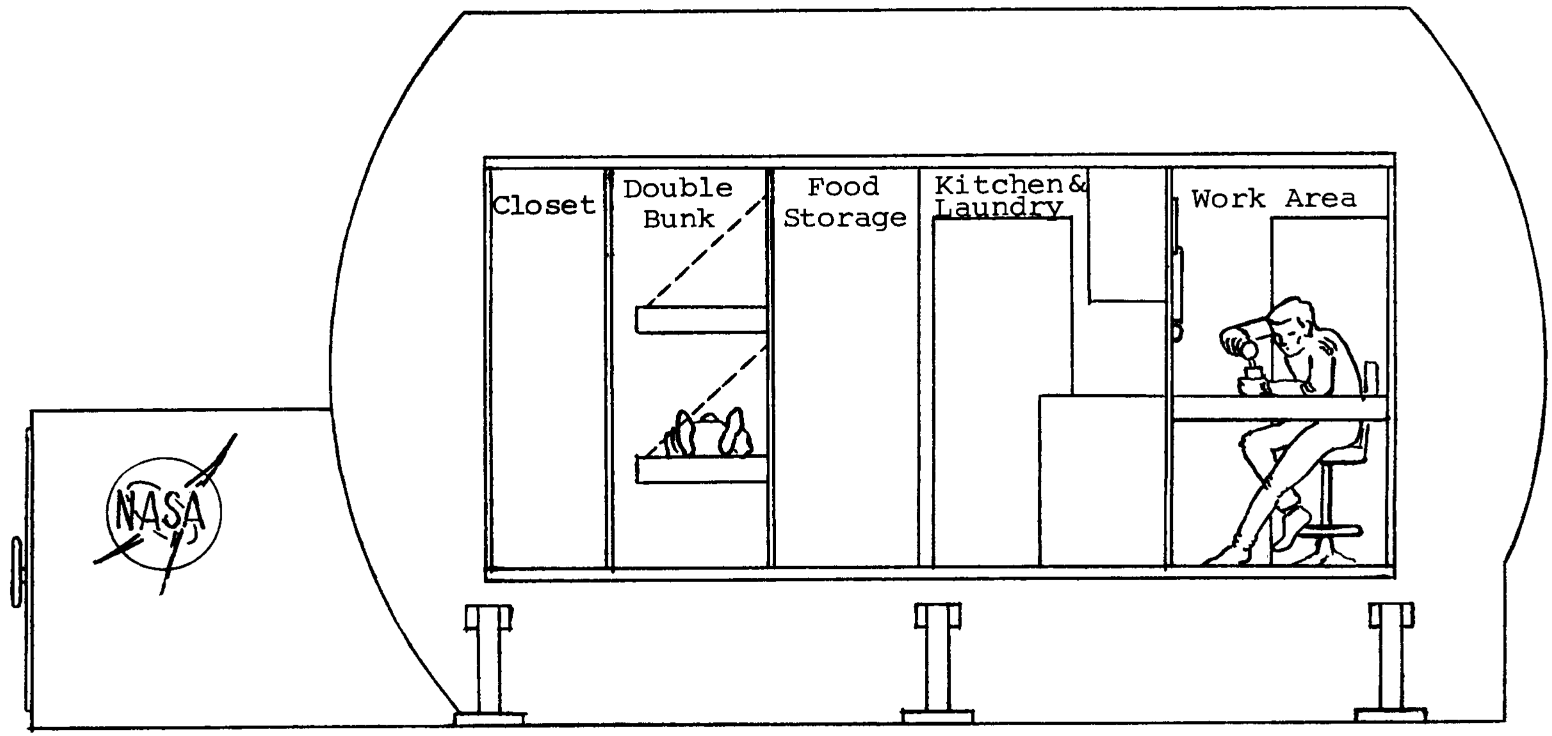


Figure 81 Right Elevation (Right Compartment Cutaway)

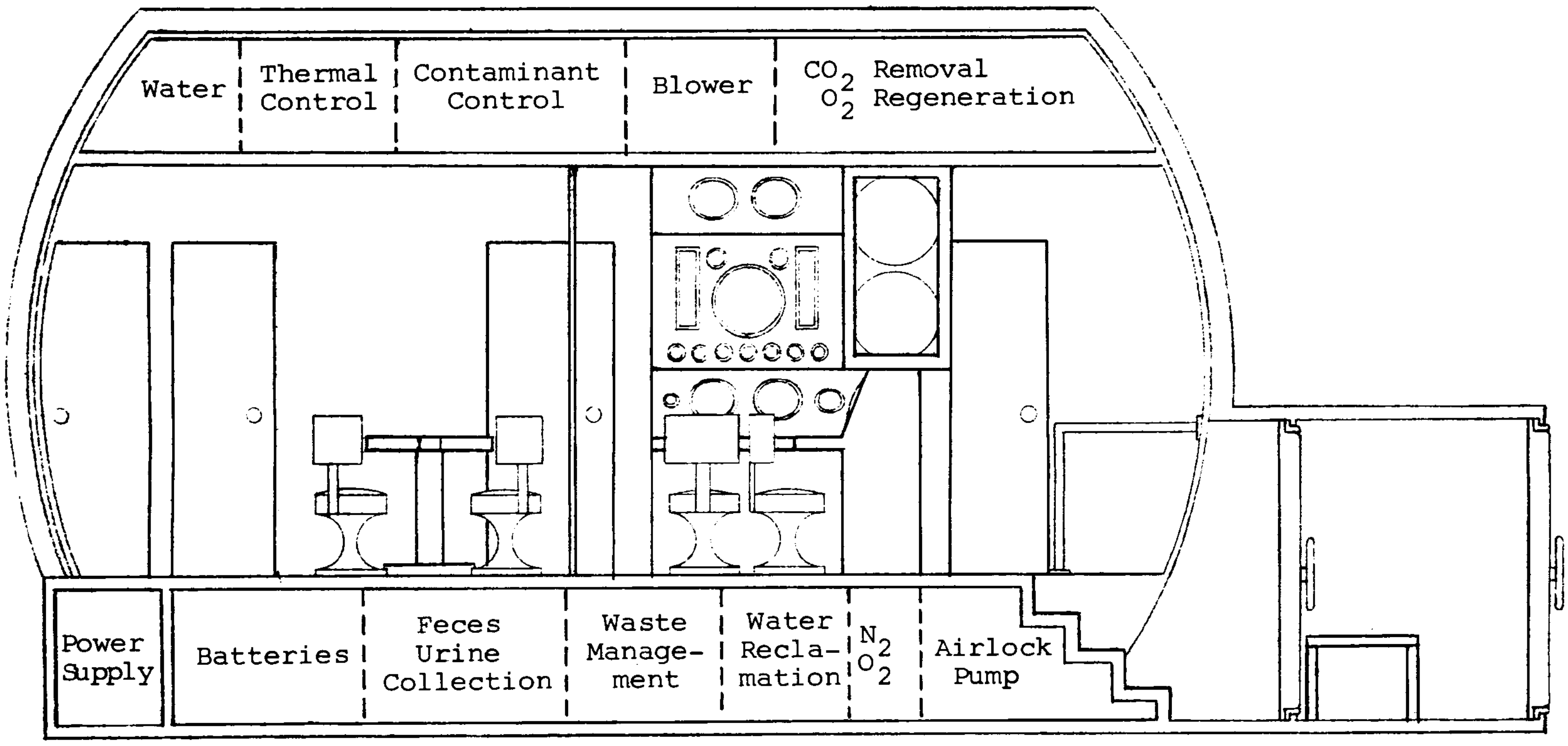


Figure 82 Left Elevation (Center Compartment Cutaway)

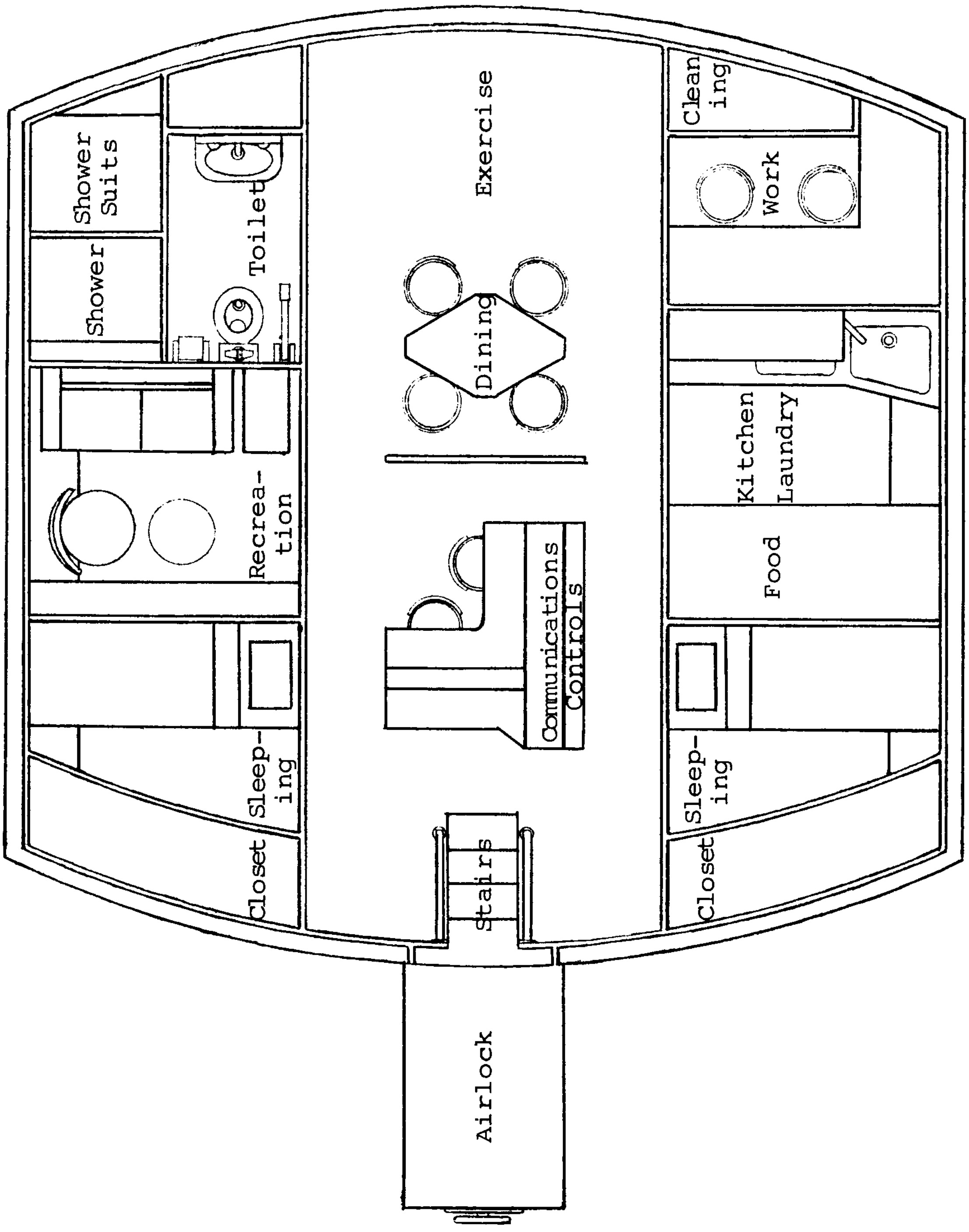


Figure 83 Floor Plan

3. Floor Plan Analysis

The cylindrical volumes, shown in cross section in Figures 84 and 85 (above and below the living volume), contain all the necessary life support equipment. Figures 86 and 87 show how the equipment will be arranged above and below the living volume. There is space allotted for piping, ducting, conduit, accessibility to the equipment, and possible cadaver storage.

Very generous access volumes are allowed because it is believed that the ability to make rapid equipment repair will be vital to the mission. The calculated volumetric allocations for each piece of equipment are presented in Table XXX. The various areas are located to maximize the time-distance profile of the astronauts with respect to the isotope power source.

Partitions are provided for each area so that privacy from noise and human activity can be maintained. In case of sickness, the recreation area will serve as an emergency sick bay. Space is provided for a very complete exercise program in order to keep the personnel physically fit. Extended periods in the reduced gravitational field could otherwise make reacclimation to Earth gravity most difficult. The recreation unit will be used to show movies, read books, and play mildly active games.

Floor space and volume allotments for each area are presented in Table XXXI.

For conservation of the water supply, it is anticipated that some type of shower suit will be developed to cleanse the body with minimum water quantities.

In lieu of windows, the shelter will utilize a 360 degree television scanner system. This will eliminate the problems of sealing and breakage associated with glass. This system is provided not so much for observation, but rather as a visible link with Mars to aid in maintaining morale, and to prevent the fear of a cave-like existence. This one-way surveillance system will also eliminate the sensation of being watched.

4. Interior Furnishings and Equipment

At the time of fabrication on Earth, all equipment will be bolted to the frames. The furnishings will also be integrally built as part of the units, and the bath fixtures will be one piece molded plastic for reduced weight.

The interior color selection is an important consideration primarily for psychological and morale factors, rather than aesthetics. The overall appearance of the interior will be bright with an individual area color scheme. Indirect incandescent lighting will be used throughout, with intensities adjustable from zero to forty foot-candles (ref. 35). The total module concept is shown in Figure 88.

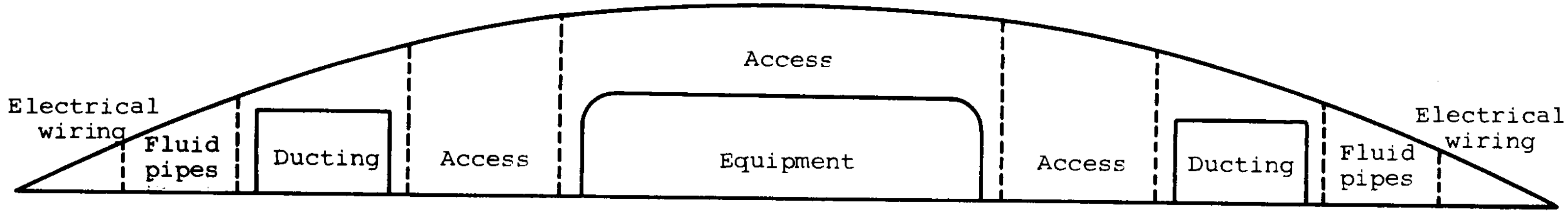


Figure 84   Volume Utilization above Ceiling

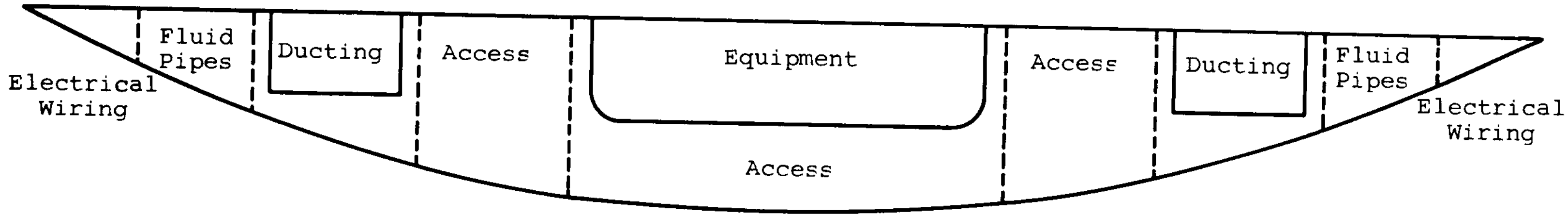


Figure 85   Volume Utilization below Floor Level

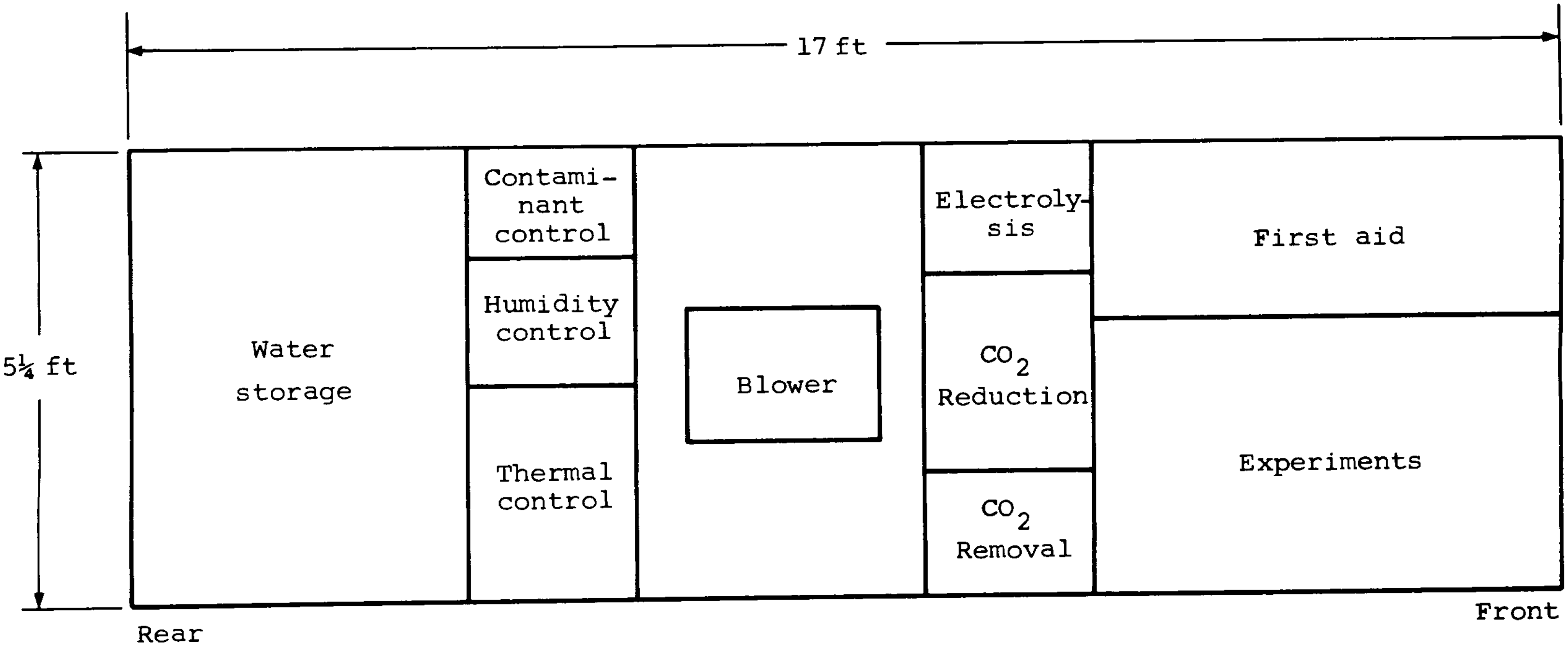


Figure 86 Equipment Layout Above Ceiling Level

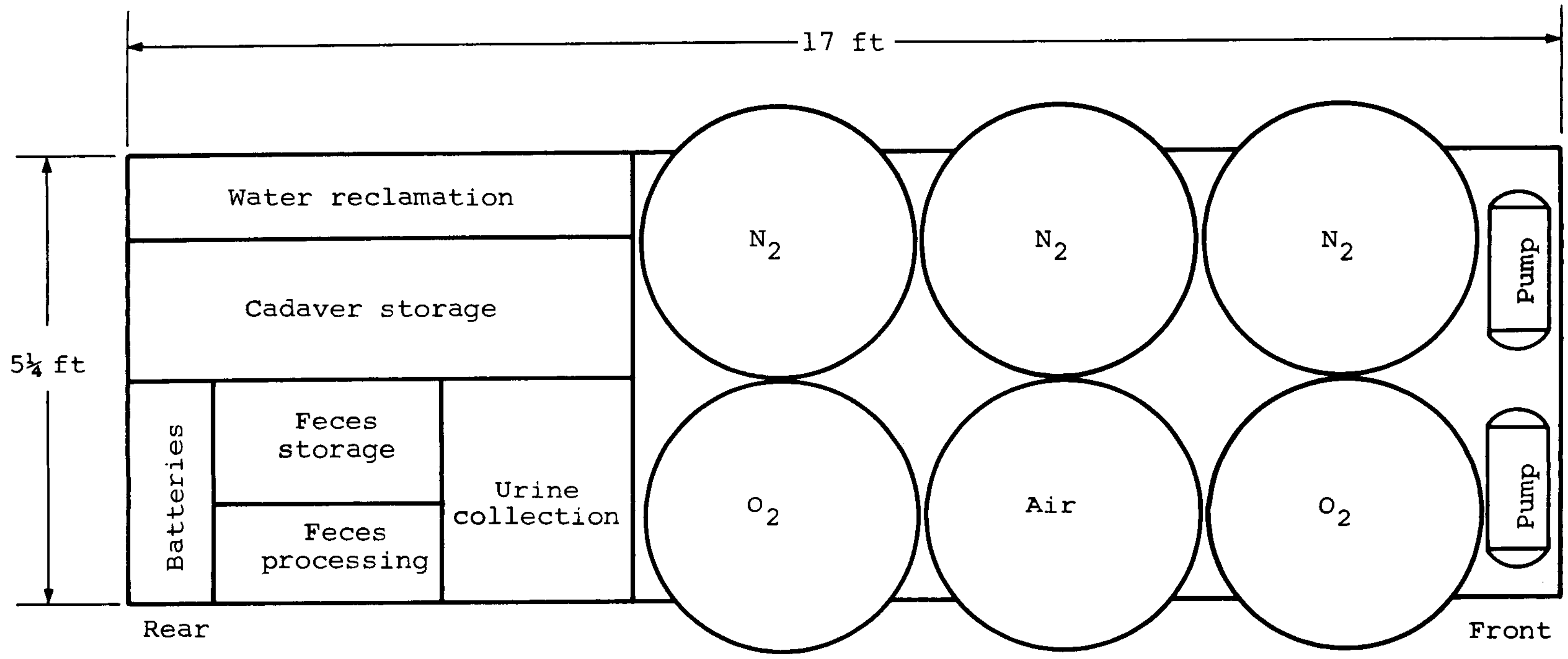


Figure 87 Equipment Layout Below Floor Level

TABLE XXX

ALLOCATED EQUIPMENT VOLUMES

Unit	Dimensions, ft	Allocated Volume, ft^3
Water Storage	$5\frac{1}{4} \times 4 \times 2$	42.00
Contaminant Control	$2 \times 1\frac{1}{4} \times 1\frac{3}{8}$	2.75
Humidity Control	$2 \times 1\frac{1}{2} \times 1\frac{3}{8}$	3.44
Thermal Control	$2 \times 2\frac{1}{2} \times 1\frac{3}{8}$	6.87
Blower	$1\frac{1}{2} \times 2\frac{1}{4} \times 1\frac{3}{8}$	3.61
Electrolysis	$2 \times 1\frac{1}{2} \times 1\frac{3}{8}$	4.12
CO_2 Reduction	$2 \times 2\frac{1}{4} \times 1\frac{3}{8}$	6.18
CO_2 Removal	$2 \times 1\frac{1}{2} \times 1\frac{3}{8}$	4.12
First Aid	$5\frac{1}{2} \times 2 \times 1\frac{3}{8}$	15.10
Experiments	$5\frac{1}{2} \times 3\frac{1}{4} \times 1\frac{3}{8}$	24.60
Feces Storage	$3\frac{3}{8} \times 1\frac{1}{2} \times 1\frac{3}{8}$	8.00
Feces Processing	$3\frac{3}{8} \times 1\frac{1}{8} \times 1\frac{3}{8}$	6.00
Water Reclamation	$6 \times 1 \times 1\frac{3}{8}$	8.25
Urine Collection	$1\frac{5}{8} \times 2\frac{5}{8} \times 1\frac{3}{8}$	5.85
Batteries	$2\frac{1}{2} \times 1 \times 1\frac{3}{8}$	3.44
Cadaver Storage	$6 \times 1\frac{1}{2} \times 1\frac{3}{8}$	21.30
Food Storage	$3 \times 6\frac{1}{2} \times 8$	156.00
Power Supply	$2 \times 2\frac{1}{2} \times 7$	35.00
Oxygen	2 (spherical)	38.60
Nitrogen	3 (spherical)	58.00
Air	1 (spherical)	19.30
Pumps	2 (cylindrical)	1.32

TABLE XXXI

STATISTICAL INFORMATION

Area	ft	ft^2	ft^3
Work	4 x 7	28.0	224
Laundry Kitchen	7½ x 7	52.5	420
Bedrooms (2)	4½ x 7	31.5	252
Closets (2)	2 x 7	14.0	112
Communication	6 x 5	30.0	240
Dining	5 x 5	25.0	200
Exercise	5 x 8	40.0	320
Bath	6 x 7	42.0	336
Incinerator	1½ x 3	4.5	36
Recreation	6 x 7	42.0	336
Utility Closet	1½ x 4½	6.8	54
Airlock	6 x 6 x 4	24.0	144

5. Packaging

The total estimated weight for the proposed Mars module configuration is 20,000 lb with a volume of about 5,944 ft^3.

A recent study by Douglas (ref. 36) indicates that in the late 1970's, a post Saturn rocket will be capable of launching a Mars Excursion Module (MEM) of about 13,800 ft^3 total volume and weighing approximately 55,000 lb. The top half of the 482 ft long launch vehicle is shown in Figure 89 along with the packaging arrangement for the MEM. Figure 90 gives a visual comparison of the MEM and surface module configurations. It seems reasonable to conclude that the launch capability will be sufficient to carry the MSM as a unit.

The primary meteoroid shielding will be furnished by the skin of the rocket. Although meteoroids are not a consideration on the surface of Mars, the need for good insulation qualities has resulted in a wall that will also act as an ample meteoroid barrier.

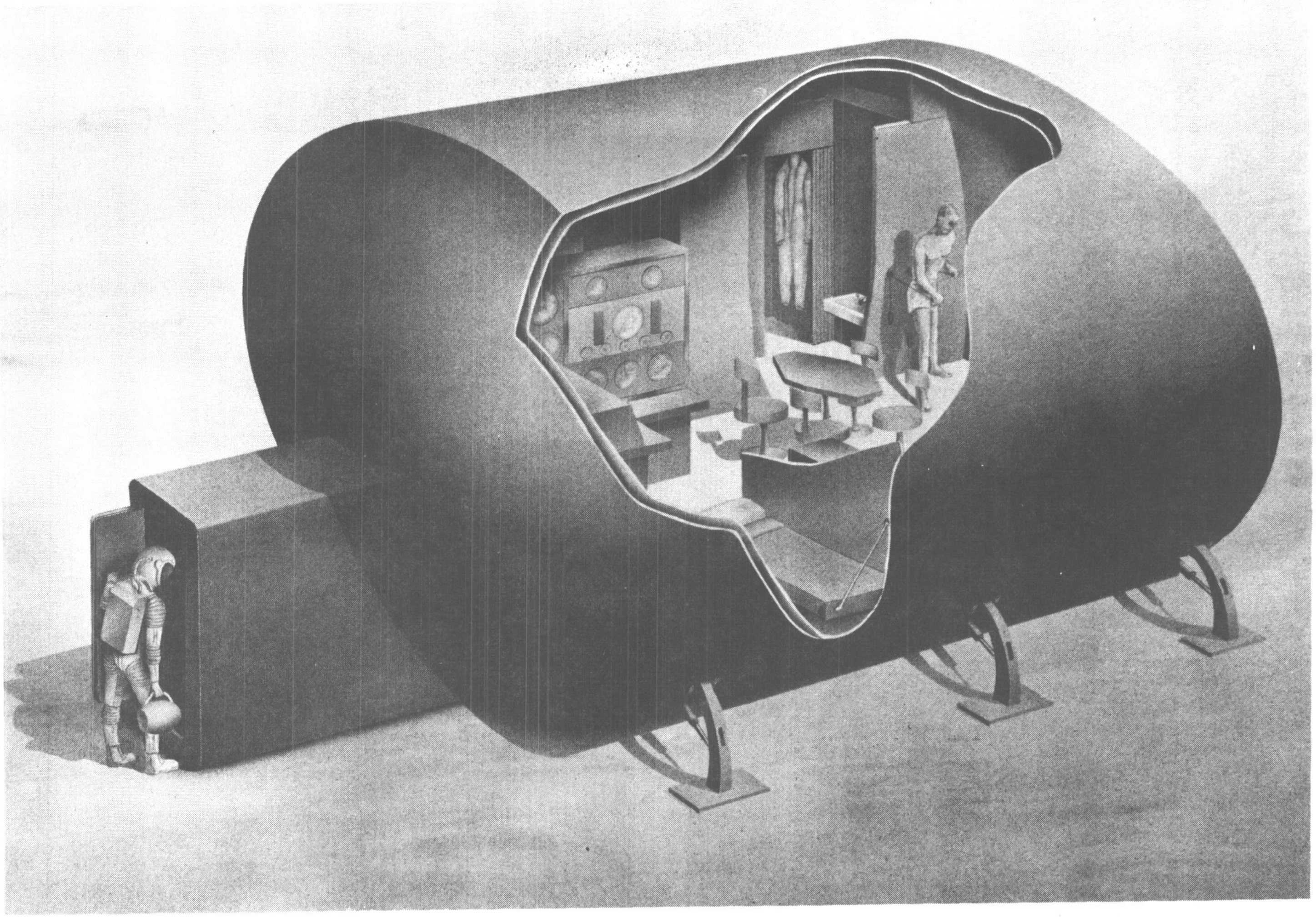


Figure 88 Module Concept

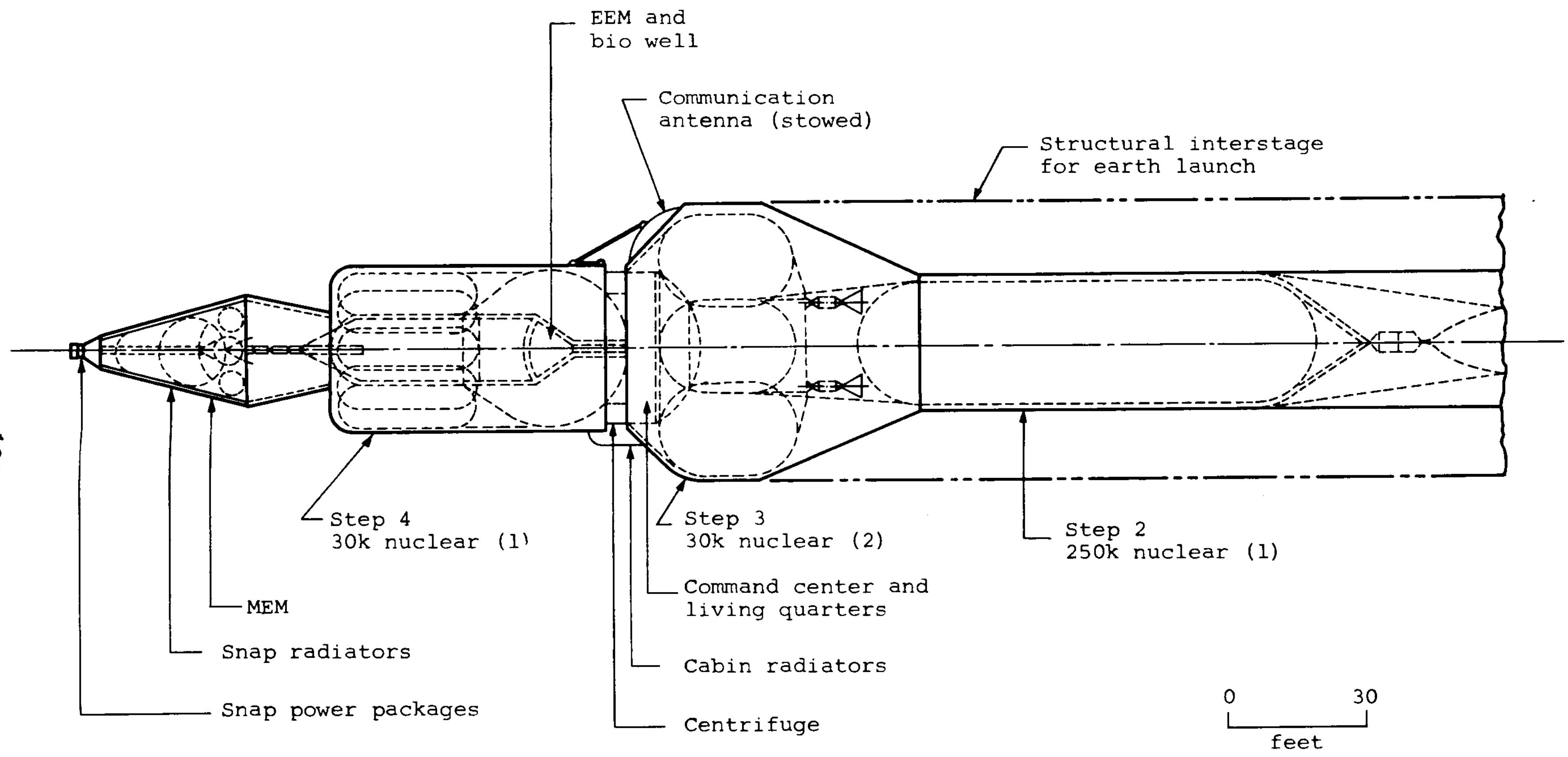


Figure 89 Transport Concept for MEM

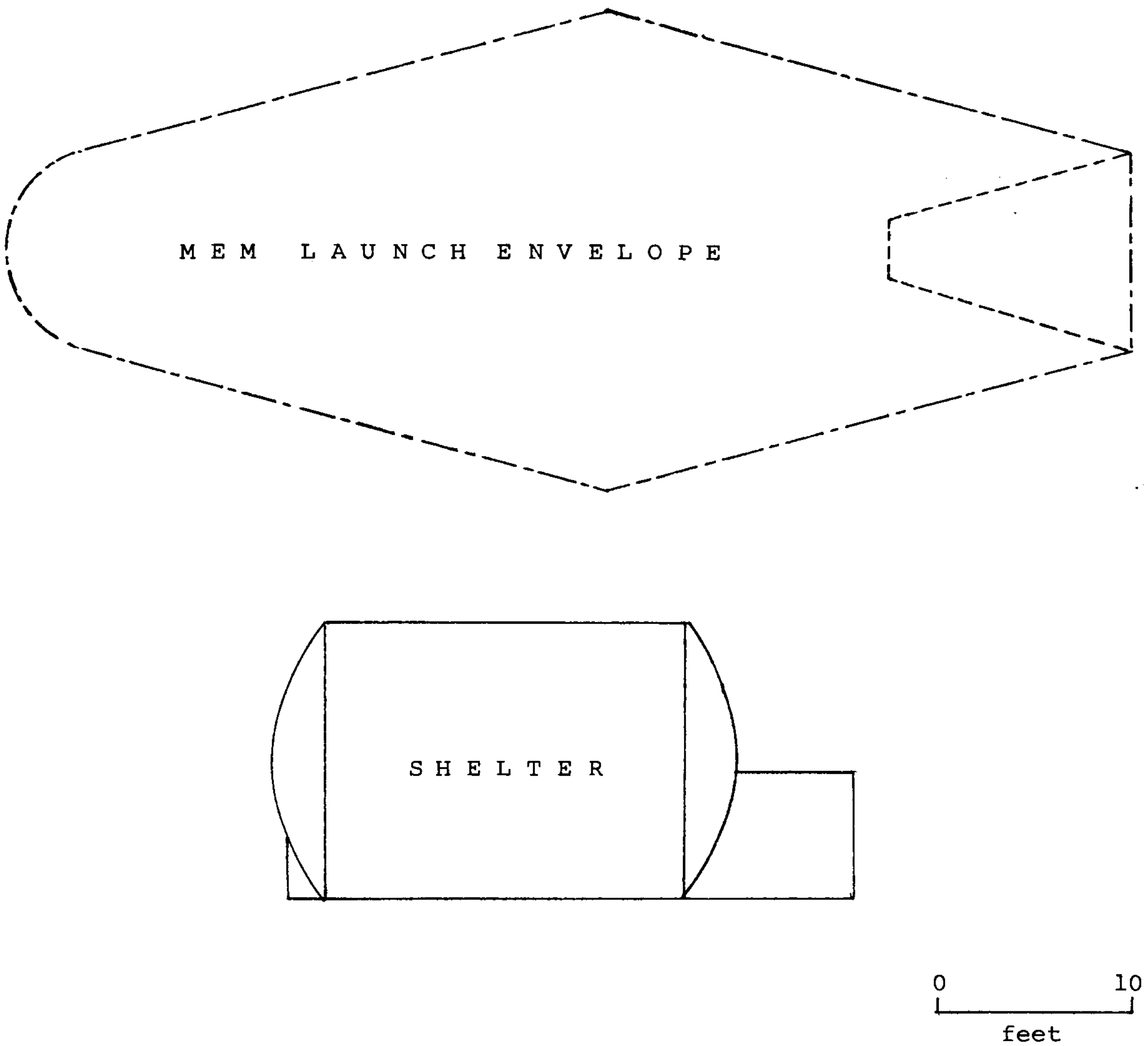


Figure 90 Mars Surface Module and MEM Size Comparison

APPENDIX B. CALCULATIONS

This appendix details the calculations made in selecting the module wall and finding the total volume and weight. Predicted 1970 material properties were used for the ultimate tensile strength and allowable compressive stress.

1. Glossary of Terms

h - sandwich height - 1 in

t - aluminum sheet thickness - 0.015 in

P - design cabin pressure - 10 psi

S_T - ultimate tensile strength - 72,000 psi

S_C - allowable compressive stress - 65,000 psi

ρ_c - core density - 4 lb/ft^3

ρ_f - frame density - 0.098 lb/in^3

a - ellipse major axis - 24 ft

b - ellipse minor axis - 14 ft

d - frame depth

n - number of frames

$\frac{d}{35}$ - frame thickness

f - frame spacing center to center

A_f - total frame cross-sectional area

A - area of an I-beam flange

p - design perimeter of ellipse

W_E - ellipsoid shell weight

W_A - airlock shell weight

T - total frame weight

V_s - shelter volume

V_A - airlock volume

S - total surface area

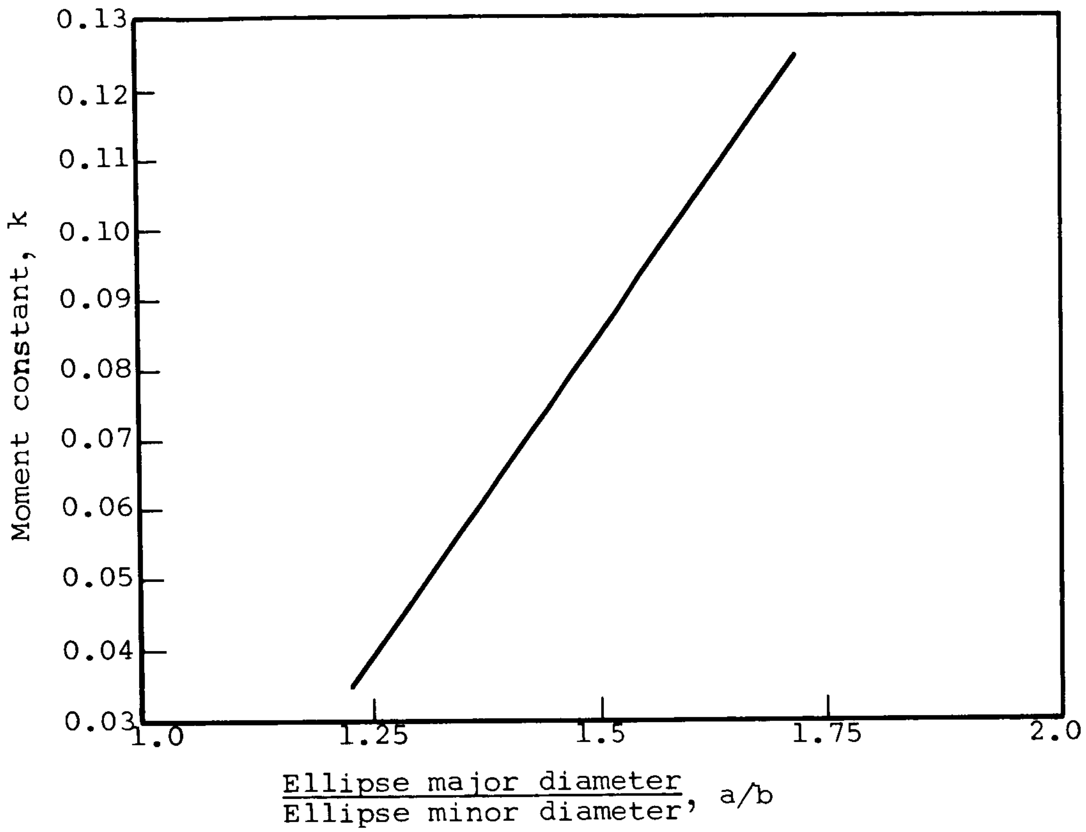


Figure 91 Moment Constant Variation with a/b
(From ref. 20)

2. Frame Spacing

Optimum design of the sandwich for varied frame spacings is obtained by the following equation:

$$2t = Pf^2/6hS_c$$

$$0.030 = 10f^2/6(65{,}000)$$

$$f^2 = (.018)(65{,}000) = 1170$$

$$f = 34.2 \text{ in}$$

for which

$$n = 7$$

3. Frame Depth

The equation governing the frame design is $M = kfPb^2$ where $k = .12$ from Figure 91. The frame section is assumed to be comprised of two flanges of equal area, and joined together by a web of depth d and equivalent thickness d/35. The total frame area A_f is, therefore:

$$A_f = 2A + \frac{1}{35} d^2$$

$$A_f = \frac{2kfPb^2}{dS_c} + \frac{1}{35} d^2$$

for minimum weight,

$$\frac{dA_f}{dd} = 0$$

then

$$d = \left[\frac{35kfPb^2}{S_c}\right]^{1/3}$$

$$d = \left[\frac{35(.12)(34.2)(10)(168)^2}{65{,}000}\right]^{1/3}$$

$$d = \left[622\right]^{1/3}$$

$$d = 8.54 \text{ in}$$

4. Aluminum Shell Thickness In Ellipsoidal Domes

$$t = \frac{Pb}{4S_T}\left[1 - (\frac{b}{a})^2\right]$$

$$t = \frac{10(14)(12)}{4(72,000)}\left[1 - (\frac{14}{24})^2\right]$$

$$t = \frac{42(.658)}{7200} = .00384$$

Material thicknesses of this order are impractical. This, however, indicates that use of the .015 material in the ellipsoidal domes will be adequate.

5. Design Perimeter

$$p = \pi(a+b)\ \frac{64-3R^4}{14-16R^2}$$

where

$$R = \frac{a-b}{a+b}$$

$$R = \frac{24-14}{24+14} = \frac{10}{38} = .263$$

$$p = \pi(38)\ \frac{64-3(.0048)}{14-16(.0691)}$$

$$p = \pi(38)\ \frac{64}{12.9}$$

$$p = 592 \text{ in}$$

6. Total Frame Cross-Sectional Area

$$A_f = \frac{2M}{dS_C} + \frac{d^2}{35}$$

$$A_f = \frac{2}{8.54}\ (17.8) + \frac{73}{35}$$

$$A_f = 4.18 + 2.08$$

$$A_f = 6.25 \text{ in}^2$$

7. Total Frame Weight

$$T = npA_f \rho_f$$

$$T = 7(592)(.098)(6.26)$$

$$T = 2540 \text{ lb}$$

8. Volume Calculations

$$V_s \simeq \pi \frac{ab}{4} 18 + \frac{4}{3} \; \frac{ab}{4} 3$$

$$V_s \simeq \pi \; ab \; (\frac{18}{4} + 1)$$

$$V_s \simeq \pi(14)(24)(4.5+1)$$

$$V_s \simeq 5800 \text{ ft}^3 \qquad V_A = (6)(6)(4) = 144 \text{ ft}^3$$

9. Total Surface Area

$$S = p \; \frac{18}{12} + \pi(\frac{24^2}{4} + 9) - \pi(\frac{14^2}{4} + 9)$$

$$S = \frac{592}{12} 18 + \frac{\pi}{4} (24^2 - 14^2)$$

$$S = 888 + 298$$

$$S = 1186 \text{ ft}^2$$

10. Ellipsoid Shell Weight

a. Aluminum. -

$$W_S = (\frac{.045}{12})(1186)(1728)(.098) = 753 \text{ lb}$$

b. Honeycomb. -

$$W_H = (\frac{1}{12})(1186)(4) = 395 \text{ lb}$$

c. Fiberglass. -

$$W_F = (\frac{9}{12}) \; (1186)(.85) = 756 \text{ lb}$$

$$W_E = 753 + 395 + 756 = 1914 \text{ lb}$$

11. Airlock Shell Weight

Figures obtained by a ratio of volumes.

$$W_S' = \frac{144}{5800}(753) = 18.65 \text{ lb}$$

$$W_H' = \frac{144}{5800}(395) = 9.78 \text{ lb}$$

$$W_F' = \frac{144}{5800}(756) = 18.73 \text{ lb}$$

$$\text{Frames} = \frac{2540}{1914}(47.16) = 63.10 \text{ lb}$$

$$W_A = 18.65 + 9.78 + 18.73 + 63.10$$

$$W_A = 110.26 \text{ lb}$$

12. Floor and Ceiling Allowance

The floor and ceiling are sandwich layers of .031 aluminum with 3 in of low density (3 lb/ft^3) polyurethane fill material.

a. Aluminum. -

$$(2)(20)(22.5)(144)(.062)(.098) = 79 \text{ lb}$$

b. Core. -

$$(2)(20)(22.5)(3)\left(\frac{3}{12}\right) = \underline{675 \text{ lb}}$$

TOTAL 1465 lb

13. Supports and Joints

There are seven spans of 6 in^2 cross-section running the width of the module in the floor and ceiling. These are connected to the main ellipse supports. The supporting struts are aluminum and spaced on approximately 3 ft centers. An area of 3 in^2 is assumed for the struts with an average height of 2 ft.

Joints were figured at 10% of the total support weight.

a. <u>Struts</u>. -

(2)(56)(3)(2)(12)(.098) = 712 lb

b. <u>Spans</u>. -

(2)(7)(6)(20)(12)(.098) = 1975 lb

c. <u>Joints</u>. -

(.1)(2687) = 269 lb

TOTAL 2956 lb

14. Total Module Structure Weight

Frame	2540 lb
Shell	1914 lb
Airlock	110 lb
Floor and Ceiling	1465 lb
Supports and Joints	2956 lb
TOTAL	8985 lb

REFERENCES

1. Sinton, W.M.; and Strong, J.: Astrophys. J. vol. 131, 1960.

2. Kliore, A.; Cain, D.L.; Levy, G.S.; Eshleman, V.R.; Fjeldiso, G.; and Drake, F.O.: Science, vol. 149, 1965, p. 1243.

3. Kaplan, L.D.; Münch, G.; and Spinrad, H.: Astrophys. J., vol. 139, 1964, 1.

4. Hanson, L.L.: Preliminary Study of Thermal Integration of Electrical Power and Life Support Systems for Manned Space Vehicles. Space Power Systems Engineering, July, 1966.

5. Goldbaum, G.C.; Manned Mars Exploration in the Unfavorable (1975-1985) Time Period. Douglas Aircraft Co., Report 45576, January, 1964.

6. Jones, A.L.; and Nissim, W.: Study of Subsystems Required for a Mars Mission Module. Final Report, Contract NAS-9-1748, January, 1964.

7. Cawton, R.W.: Staying Alive in Spacecraft. Chem. Eng. July, 1966.

8. Mattson, H.W.: Keeping Astronauts Alive. International Science and Technology, June, 1966.

9. Szego, George C.; and Taylor, Edward J.: Space Power Systems Engineering. vol. 16. Academic Press, 1966.

10. Schuh, N.F.: Thermoelectric Power Systems. Astronaut. Aerospace, May, 1963.

11. Silverstein, A.; Publication of the Space Power Systems Advanced Technology Conference. Lewis Research Center, Cleveland, Ohio. August 23-24, 1966.

12. O'Reilly, W.J.; Application of Radiosotopes to Manned Spacecraft Life-Support Systems. SAN-575-12. vol. 1. AiResearch Manufacturing Company, July 1965.

13. Heath, A.R. Jr.; and Maxwell, P.T.: Solar Collector Development. Astronaut. Aerospace Eng. May, 1963.

14. Szego, G.C.; and Cohn, E.M.: Fuel Cells for Aerospace Application. Astronaut. Aerospace Eng. May, 1963.

15. Hanson, K.L.: Thermal Integration of Electrical Power and Life Support Systems for Manned Space Stations. General Electric Report on Contract NAS3-2799, Nov. 1965.

16. Tonelli, A.D.; and Regnier, E.P.: Radiosotope Power-Generating System for a Manned Space Laboratory. Journal of Engineering for Industry, May 1966.

17. Helvey, T.C. NASA Conference on Nutrition in Space and Related Waste Problems. University of South Florida, NASA SP-70, Tampa, Florida, Apr. 27-30, 1964.

18. Ferguson, James: Handbook of Instructions for Aerospace Personnel Subsystems Design. AFSC Manual No. 80-3, Dec. 1, 1966.

19. Mitchell, R.E.: Martian Shelter Technology. IIT Research Institute Interim Technical Report 1 on Contract NASr-65(15), Oct. 7, 1966.

20. Jones, A.L.; and Nissim, W.: Study of Subsystems Required for a Mars Mission Module. North American Aviation Inc., Report No. SID64-1-3 on Contract NAS-9-1748. vol 3. Jan. 2, 1964.

21. Campbell, J.A.: Design Feasibility Study for Portable Lunar Survival Shelter. IIT Research Institute Report on Contract NAS-9-1753, 1963.

22. Hower, K.L.: Mars Landing and Reconnaissance Mission Environmental Control and Life Support System Study. Fourth Monthly Progress Report on Contract NAS-9-1701, 1963.

23. Armstrong, R.C.: Life Support System for Space Flights of Extended Time Periods. NASA CR-614, 1966.

24. Klem, R.L.; and Dingwall, A.G.F.: Energetics 3: Thermoelectric Power. Mech. Eng., Aug. 1966.

25. Shuh, N.F.: Thermoelectric Power Systems. Astronaut. Aerospace Eng. May 1963.

26. Schulman, Fred: Isotopes and Isotope Thermoelectric Generators. Space Power Systems Advanced Technology Conference, Aug. 23-24, 1966.

27. Heath, A.R. Jr.; and Maxwell, Preston T.: Solar Collector Development. Astronaut. Aerospace Eng. May 1963, pp 58-61.

28. Wilson, Volney C.; and Hamilton, Robert C.: Thermionic Convertors for Space Power. Astronaut. Aerospace Eng. May 1963.

29. Houston, John M.: Energetics 4: Thermionic Power. Mech. Eng. Sep. 1966.

30. McCellland, D.; and Pichel, M.A.; and Springer, L.M.: Solar Concentrator Development for Space Power Systems. AIEE Paper 62-1314, 1962.

31. Woods, R.W.; and Erlanson, E.P.: Thermal Integration of Electrical Power and Life Support Systems for Manned Space Stations. NASA CR-543, 1966.

32. Koerner, T.W.: Static Power Conversion for Spacecraft. Astronautics and Aerospace Engineering, May, 1963.

33. Space Batteries. Technology Handbook, NASA SP-5004.

34. Lampert, S.; and Younger, D.G.: Multiwall Structures for Space Vehicles. Aeronutronic, Report No. U-910. Contract No. AF33(616)-6641, May 1960.

35. Alexander, J; and Merkel, K.H.: Lunar Shelter Program. University of Cincinnati, November 1963.

36. Koprowski, E.F.: Manned Mars Exploration in the Unfavorable (1975-1985) Time Period. Douglas Aircraft Report No. SM-45584 on Contract NAS8-11005, February 1964.

BIBLIOGRAPHY

Baier, W.H.: Lunar Shelter. Armour Research Foundation (IIT Research Institute) Proposal No. 62-421K, December 27, 1961.

Cann, M.W.P.; Davies, W.O.; Greenspan, J.A.; and Owens, T.C.: A Review of Recent Determinations of the Composition and Surface Pressure of the Atmosphere of Mars. NASA CR-298, 1965.

Chandler, H.: Carbon Dioxide Reduction System. Technical Doc. Report No. (AMRL-TDR-64-42), Isomet Corp., Palisades Park, N.J., May 1964.

Coe, C.S.; and Rousseau, J.: Oxygen Recovery System Integration for Long-Range Space Missions. ASD-TDR-63-260.

Dieckamp, Herman: Technical papers presented at the Intersociety Energy Conversion Engineering Conference. American Institute Aeronautics and Astronauts, Sep. 26-28, 1966.

Goldbaum, G.C.: Manned Mars Exploration in the Unfavorable (1975-1985) Time Period. Douglas Aircraft Report No. SM-45577, Contract NAS8-11005, February 1964.

Goldbaum, G.C.: Manned Mars Exploration in the Unfavorable (1975-1985) Time Period. Douglas Aircraft Report No. SM-45581, Contract NAS8-11005, February 1964.

Gross, F.R.: Life Support System for Stay Time Extension Module (STEM). Goodyear Aerospace Corp., Akron, Ohio, Jan., 1965.

Hanson, K.L.: Thermal Integration of Electrical Power and Life Support Systems for Manned Space Stations. NASA CR-316, 1966.

Hooper, J.R.: Steady-State and Dynamic Operating Characteristics of a Simulated Three-Loop Space Rankine-Cycle Powerplant. NASA CR-625, 1966.

Houston, J.M.: Energetics 4: Thermionic Power. Mech. Eng. Sep. 1966.

Koelle, H.H.: Handbook of Astronautical Engineering. McGraw-Hill, New York, 1961.

Kovacik, Victor P.: Dynamic Energy Conversion. Astronaut. Aerospace Eng. May 1963.

Kuiper, G.P.: The Atmospheres of the Earth and Planets. University of Chicago Press, Chicago, 1952.

Leighton, R.B.; and Murray, B.C.: Behavior of Carbon Dioxide and Other Volatiles on Mars. Science, vol. 153. July, 1966.

Mills, E.S.: Manned Mars Exploration in the Unfavorable (1975-1985) Time Period. Douglas Aircraft, Report No. SM-45582, Contract NAS8-11005, February 1964.

NASA; The Physical Ephemeris of Mars. Memo: RM-3999-NASA, January, 1964.

NASA; Study of Heat Shielding Requirements for Manned Mars Landing and Return Missions. NASA CR-308, 1965.

Owen, T.: The Composition and Surface Pressure of the Martian Atmosphere: Results from the 1965 Opposition. IIT Research Institute, Chicago, Illinois.

Salisbury, Frank: Martian Biology. Science, vol. 136, no. 3510, Apr. 1962.

Vancouleurs, G. de: Geometric and Photometric Parameters of the Terrestrial Planets. Memo: RM-4000-NASA, Mar. 1964.

Warnecke, P.; and Marmo, F.F.: II: No. 2 In the Martian Atmosphere. NASA CR-5, 1963.

Wasko, P.E.: Manned Mars Exploration in the Unfavorable (1975-1985) Time Period. Douglas Aircraft Report No. SM-45579, Contract NAS8-11005, Feb. 1964.

Weissbart, J.: and Smart, W.H.: Study of Electrolytic Dissociation of CO_2-H_2O Using a Solid Oxide Electrolyte. NASA CR-680, 1967.

Woods, R.W.; and Erlanson, E.P.: Thermal Integration of Electric Power and Life Support Systems for Manned Space Stations. General Electric, Contract NAS3-6478, Sep. 1966.