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Lunar Base Synthesis Study

FINAL REPORT

VOLUME III

APPENDIXES

Space Division
North American Rockwell

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APPROVED BY

J.M. MANSFIELD, PROGRAM MANAGER
LUNAR BASE SYNTHESIS

Space Division
North American Rockwell

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APPENDIX A. EGRESS/INGRESS OPERATIONS

EGRESS/INGRESS INTERFACES

The LSB functional flow analysis has provided a list of possible ingress/egress interfaces associated with the LSB mission, the resulting activities of concern are identified by Figure A-1. Basically, there exists three operational phases involving both recurring and nonrecurring ingress/egress operations. The first phase is that involving base facilities support which includes the initial base erection, periodic logistics, scheduled and unscheduled repair and maintenance, possible exploitation and contingencies.

The second phase involves base-dependent experimentation and exploration. Once the base is established, this phase is the next step in expanding the sphere of influence of the LSB. Local experimentation, exploration, and perhaps, travel to remote experiment modules will provide the operational proving ground for the next phase, remote sorties.

In essence, the sortie associated operations repeat, in miniature, the previous two phases: The vehicles must be repaired, modified, loaded and checked-out prior to the sortie. Several possible means are available to prepare the sortie (i.e., garaged equipment, base-docked vehicles, independent storage); however, these are all variations on the original base facilities operations. Next involves the actual sortie experimentation and exploration which, again, is a variation on (and operationally dependent upon) the techniques developed during the base exploration phase. Table A-1 details the above ingress/egress operations inherently including the several base configurations. Specific operations, procedures, and timelines are developed in the subsequent tasks.

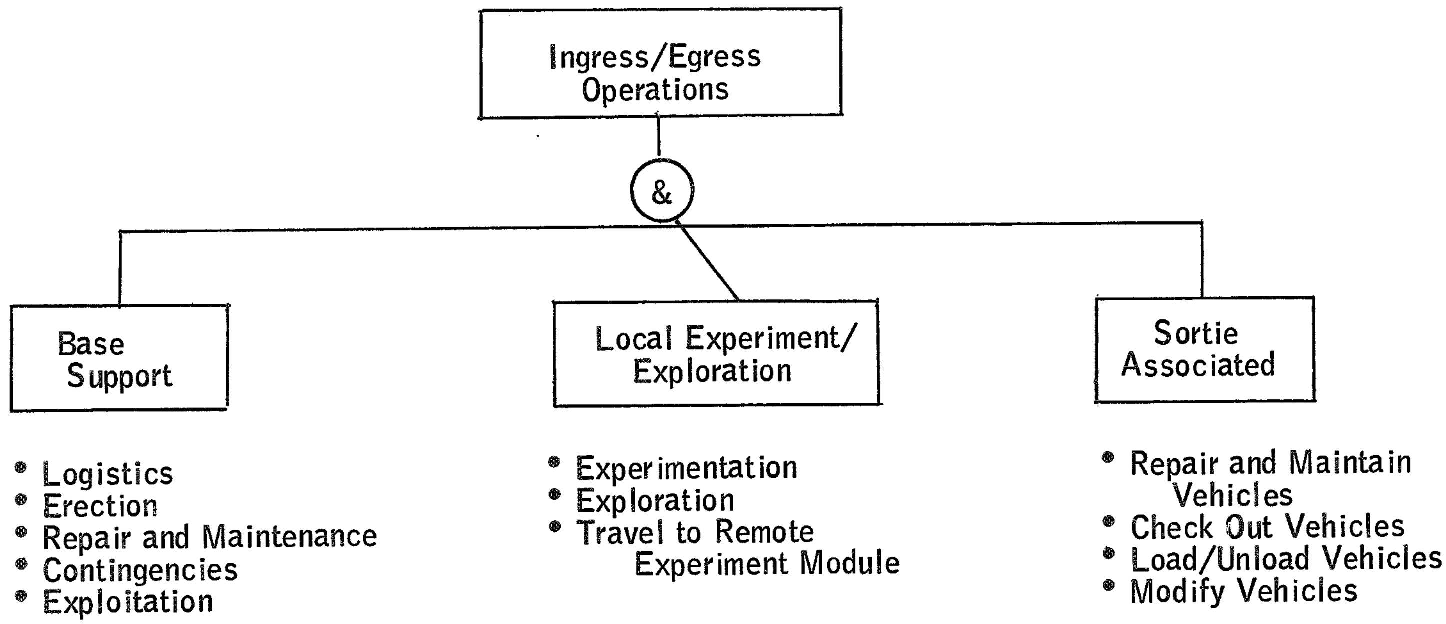

Ingress/Egress
Operations
&
Base
Support
• Logistics
• Erection
• Repair and Maintenance
• Contingencies
• Exploitation
Local Experiment/
Exploration
• Experimentation
• Exploration
• Travel to Remote
Experiment Module
Sortie
Associated
• Repair and Maintain
Vehicles
• Check Out Vehicles
• Load/Unload Vehicles
• Modify Vehicles

FIGURE A-1.

Table A-1. Base Ingress/Egress Operations

A. Base Support

1. Logistics (Cargo handling, transportation)

   a) With rover*
   b) Without rover

2. Erection

   a) With heavy machinery (attached to rover)*

      1) Main base
      2) Secondary modules
      3) Supporting equipment

   b) Without heavy machinery

      1)2)3) above

3. Repair and Maintenance

   a) With use of vehicle and heavy machinery*

      1) Periodic inspection, repair/replacement (if necessary) of base shelter module(s)

      2) Periodic inspection, repair/replacement (if necessary) of launch facilities

      3) Periodic inspection, repair/replacement (if necessary) of local sensors/science equipment

   b) Without use of vehicle

      1)2)3) above

4. Contingencies

   a) Evacuation:

      1) to aux. shelter
      2) to launch vehicle
      3) from disabled shelter
      4) from disabled vehicle

   b) Emergency encasement due to radiation, etc.

Table A-1. Base Ingress/Egress Operations (cont.)

A. Base Support

5. Exploitation - Mining

B. Local Experiment/Exploration

1. Perform local experiment (within one day's travel from base)

a) with vehicle*
b) without vehicle

2. Perform local exploration (within one day's travel from base)

a)b) above

3. Travel to remote experiment shelter module

a)b) above

C. Sortie Associated (multi-day independence from base)

1. Repair and maintain vehicles
2. Checkout vehicles
3. Load/unload vehicles
4. Modify vehicles (attachments)

*Requires R&M and checkout of vehicles periodically.

Important factors associated with Ingress/Egress:

Dust removal
Size of load (Vol., Dimensions)
Power required for A/L
Time for A/L
A/L system mass
Gas system
Cycles
Leakage

INGRESS/EGRESS REQUIREMENTS

The ingress/egress requirements applicable to the LSB, as determined by the Solar Powered Space Station and the Orbiting Lunar Station studies, are presented below:

General Requirements

1. An opening shall be provided which allows for the transfer of crew between two adjacent volumes in a shirt-sleeve environment. This opening shall also be capable of allowing the transfer between volumes of cargo of 5-foot diameter. The opening shall be capable of being sealed with or against a pressure differential rate of change between the two volumes of 0.38 psi/minute, and of providing a pressure-tight seal against the maximum atmospheric design pressure in either direction. Aisles and passageways for crew transfer only shall have a minimum width of 32 inches, with 36 inches to 42 inches preferred. The height shall be a minimum of 82 inches with 86 inches preferred. A height of 84 inches shall be considered as nominal. Tunnels for crew transfer only, which are less than seven feet in length, shall have a minimum diameter of 42 inches; tunnels which are greater than seven feet shall have a minimum diameter of 48 inches.

2. An internal airlock will be provided which allows for the transfer of IVA personnel between adjacent separately pressurizable volumes when a pressure differential exists or one volume is contaminated. It shall be capable of accommodating two pressure-suited men with backpacks or with umbilicals. One may be incapacitated. Transfer of the two men shall be possible unaided by other personnel. The airlock shall be capable of pressurized or depressurized operation with either of the two connecting

volumes depressurized. The airlock shall have a minimum height of 48 inches and a minimum diameter of 60 inches for a cylindrical airlock, or a minimum width and depth of 42 inches and 60 inches, respectively, for a rectangular airlock. Outward opening hatches and associated actuating mechanisms with access to and egress to each pressure volume shall be provided.

Use of the airlock shall not cause a rate of pressure drop of more than 0.5 psi/second in the connecting volumes for normal operations. Higher rates are acceptable for emergencies. Three uses of the airlock (entry and return into an unpressurized volume) shall not cause the atmospheric pressure in the pressurized volume to drop below 62 percent of the normal operating pressure.

3. The capability shall be provided which allows the transfer of EVA personnel from one of the pressurizable volumes to and from space. It shall be capable of accommodating two pressure-suited men with backpacks or with umbilicals. One EVA man may be incapacitated. Transfer of the two men shall be possible unaided by other personnel. The airlock shall be capable of pressurized or depressurized operation with the connecting volume(s) either pressurized or depressurized. Use of the airlock shall not cause a rate of pressure drop of more than 0.5 psi/second in the connecting volumes for normal operations. Higher rates are acceptable for emergencies. Three uses of the airlock (exit and return) shall not cause the atmospheric pressure in any pressurized volume to drop below 62 percent of the normal operating pressure.

4. A minimum distance of 10 feet shall separate the two openings on any one deck, edge to edge. The same minimum distance shall separate the opening of the internal airlock from the other opening of the same deck.

5. Staterooms, laboratories, toilets and other areas with restricted access shall provide two separate entry/egress paths for personnel. The two separate paths shall, where possible, lead to different areas on the deck. Where it is not practical to provide doors or normal access routes, the second entry/egress paths may be provided by knock-out panels for emergency use. These should be capable of being opened from either side.

These requirements define a minimum ingress/egress capability, the maximum being defined by the LSB operational objectives. Examining the base facilities ingress/egress interfaces it is noted that only certain equipment may exceed the above requirements for crew airlocks (i.e., vehicles, large scientific equipment, major base support systems). This equipment may require a modular component structure or, in the case of vehicles, separate garaging facilities. Ideally, this equipment is initially installed in the base modules and designed in modular form for the aforementioned ingress/egress capabilities.

Concerning experiment equipment: an inspection of the LSB experiment equipment list shows that all equipment except the following fall well within the guidelines. (Most EVA equipment is designed to be hand carried.) The exceptions: The various telescopes, electron microscope, antenna sets, RF noise survey system, the various laboratories, the 30-and 300-meter drills, the carbo-thermal processing unit. Each of the above exceptions is either extra-base equipment or incorporated in the base module design and, once emplaced, is stationary.

The ingress/egress interfaces are readily determined; however, the actual cycling required is somewhat undefined. The cycling associated with base erection, logistics, repair and maintenance and other base support operations has not been identified. A reasonable approach is that the main crew support module be deployed and activated as soon as possible. With this module as the base of operations, the other modules would be pressurized as connected, thus providing 1 cycle per two-man work crew (or four-man work crew if a four-man airlock is available) each day until module completion. After the base is fully activated, requirements on the observatory demand either a full crew at the distant module or 1 two-man cycle per day for maintenance. Vehicle preparation/repair may require additional cycling, as yet undefined. LSB experimentation ingress/egress requirements are presented in Table A-1. The cycling required should be treated as a conservative guideline until a more accurate, detailed, ingress/egress interface plan is available. Therefore, for the purpose of this report, it is not unreasonable to assume that at least one cycling daily is required, perhaps as many as four if several work crews are involved. Figure A-2 shows the effect of individual cyclings on a 10-hour work day for various cycling concepts: complete suit change; PLSS change only each cycle; one PLSS change only.

OPTION SYNTHESIS

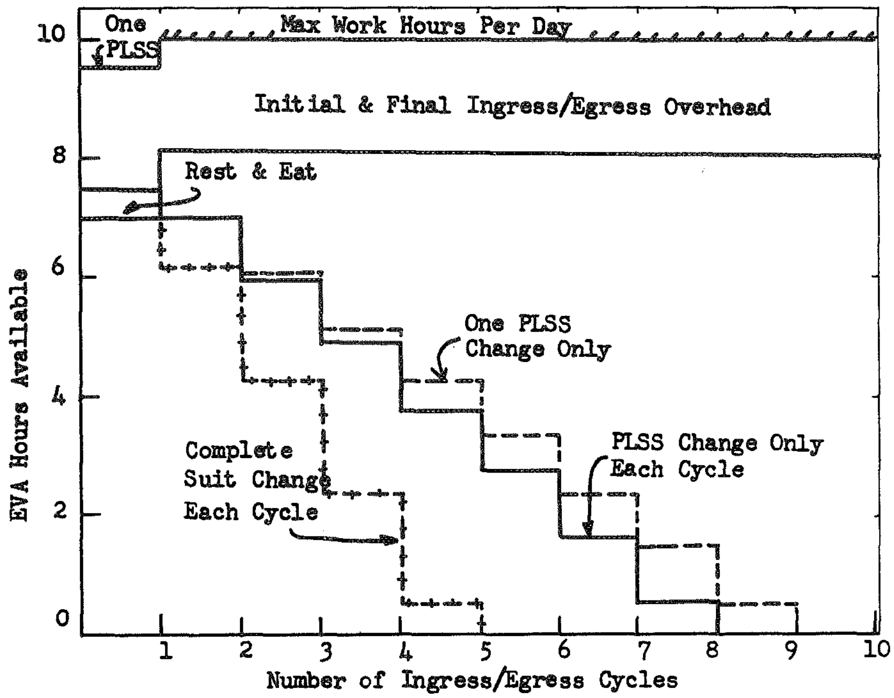
The major components of the fully operational LSB will consist of:

a) Two or more crew modules

b) One or two laboratory modules (excluding observatory)

c) Garaging facilities (with workshop)

d) Base operations module



Ingress/Egress Timeline

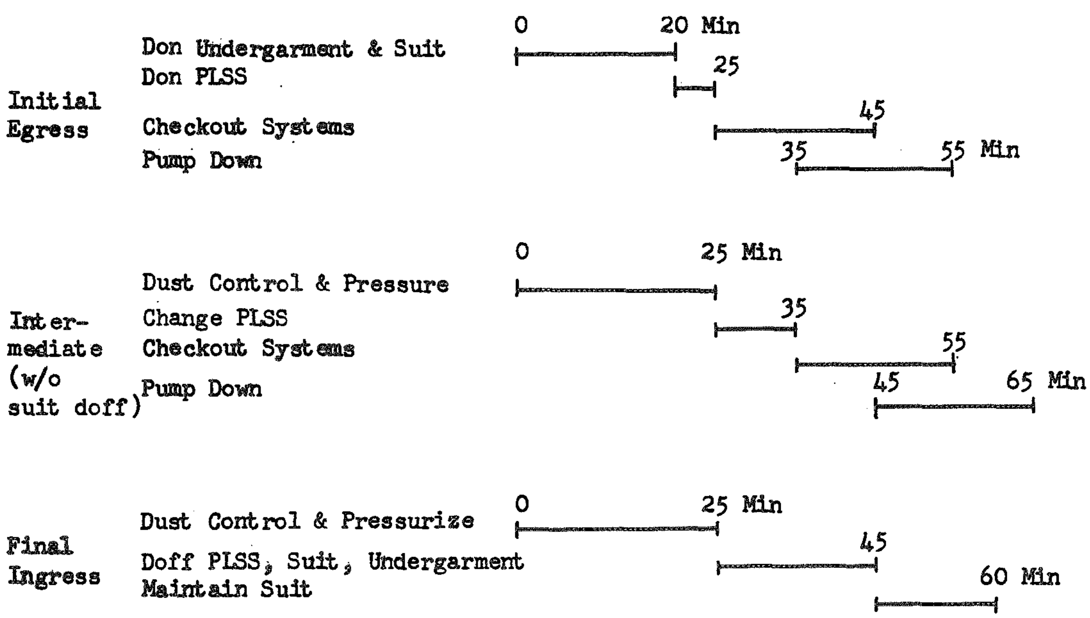


FIGURE A-2

e) Supporting systems module

f) Warehouse module

g) Observatory, drilling or other special science modules

Some of the above components may be functionally grouped into one module; an independent operation is assumed in this analysis.

EVA preparation and post-operation have been determined in Figure A-2. Activities include:

Preparation (45 min.): Donning under-garment
Suit and PLSS systems connection and checkout (pre-breathing not included - not necessary for cabin atmosphere <10 psi)

Post-Operation (60 min.): Dust control
Doffing PLSS, suit, and undergarment
Suit maintenance

Hence, each egress/ingress consumes at least 1-3/4 hours, not including any non-concurrent pumpdown time. Therefore, modules requiring frequent ingress/egress for the sole purpose of intermodule transfer should be connected for internal transfer, if physically possible.

Of the above list of components, certain modules may automatically be selected for joining to form the integral base complex from the requirement for frequent transfer:

a. Crew stations (intermodule transfer several times daily)

b. Base operations

c. Laboratory modules

In addition, one or more of the following may be joined to the above complex:

d. Garaging facilities (periodic transfer)

e. Warehouse

f. Supporting systems

The observatory will be placed at a distance from the central complex, and, since requiring frequent maintenance, may require on-location support personnel.

OPTION EVALUATION

In implementing the LSB mission, it is not unreasonable to estimate at least one airlock (AL) cycling per day. Assuming only a 100 ft^3 AL at 10 psi, an AL dump would require 5 lbm/cycle x 1 cycle/day x 180 days/resupply period or about 850 lbm of atmosphere per resupply period in addition to the normal leakage. On the overall three to five year LSB mission, this amounts to 5,100 to 8,500 lbm, far overshadowing a pumpdown (and power penalty) system. Developed in this section are the various factors, in parametric form, for an AL pumpdown system.

Equations were developed relating the physical properties of an airlock-pump-receiver system for an arbitrary gas system using a mechanical position displacement pumping model. A multi-stage pumping process operates on a similar manner; a 40 percent pumping efficiency (Reference A-1) was assumed to account for the additional work required. The process is as follows:

A. A simple model to be kept in mind describes the positive displacement pumping process. The major components are an airlock and receiver connected by a piston-cylinder pump with two check valves. Upon the cylinder expansion stroke the receiver valve is closed and the airlock opens, in effect, increasing the airlock volume adiabatically by the cylinder volume. (For computation purposes an incremental mass was chosen to be removed.) The new conditions in the airlock and cylinder are:

$m_a = m_{ai} - \Delta m,$	new airlock mass remaining (i denotes previous value)
$v_a = V_a/m_a,$	new specific volume
$\Delta V = v_a\, m_{ai} - V_A,$	volume removed
$P_a = P_{ai} \left(\frac{v_{ai}}{v_a}\right)^{\gamma}$	adiabatic pressure change
$T_a\, R_G = P_a\, v_a$	new temperature from the equation of state

The work required in this expansion is negligible.

B. On the compression stroke the airlock check valve closes and an adiabation compression occurs until the cylinder conditions match the receiver conditions and the receiver check valve opens. Empirically:

$$W^1 = \frac{\Delta m}{\eta_p(1-\gamma)} (T_{ki} R_G - T_a R_G)$$ work required for adiabatic compression to match receiver conditions

$$V_2 = \frac{\Delta m \; R_G \; T_{ki}}{P_{ki}}$$ volume of cylinder when receiver conditions are matched (i.e., ΔV is compressed to V_2 volume)

C. The receiver check valve opens and the remaining process involves a compression of the new effective receiver volume ($V_K + V_2$) to the actual receiver volume (V_K). Both isothermal (constant temperature) and adiabatic (no heat flux) processes were considered:

Isothermal

$$m_k = m_{ki} + \Delta m$$ new receiver mass

$$P_k = P_{ki} \left(\frac{V_K + V_2}{V_k}\right)$$ isothermal compression

$$W'' = \frac{P_k + P_{ki}}{2\eta_p} V_2$$ remaining compression work

$$W = W' + W''$$ total work required

$$Q = W$$ heat flux out of system (isothermal)

Adiabatic

$m_k = m_{ki} + \Delta m$ new receiver mass

$P_k = P_{ki} \left(\frac{V_k + V_2}{V_2}\right)^{\gamma}$ adiabatic compression

$R_G T_k = \frac{P_k V_k}{m_k}$ new receiver temperature

$W'' = m_k \frac{R_G T_k - R_G T_{ki}}{\eta_P (1 - \gamma)}$ compression work

$W = W' + W''$ total work required (no heat flux)

The above was calculated for varying airlock and receiver sizes and for different gas system. Chosen as baselines were a 100 ft^3 airlock volume pumped (free space volume) and a 10 psi atmosphere (3.5 P_{O_2} 6.5 P_{N_2}). The results are presented in parametric form.

Baseline:

Volume pumped vs. airlock mass remaining - Figure A-3

Receiver pressure vs. receiver volume -

Adiabatic - Figure A-4a

Isothermal - Figure A-4b

Receiver temperature vs. receiver volume - Figure A-5

Heat transfer rate (Btu's/ft^3 pumped) vs. receiver volume - Figure A-6

Work rate (kwhr/ft^3 pumped) vs. airlock mass remaining - Figure A-7

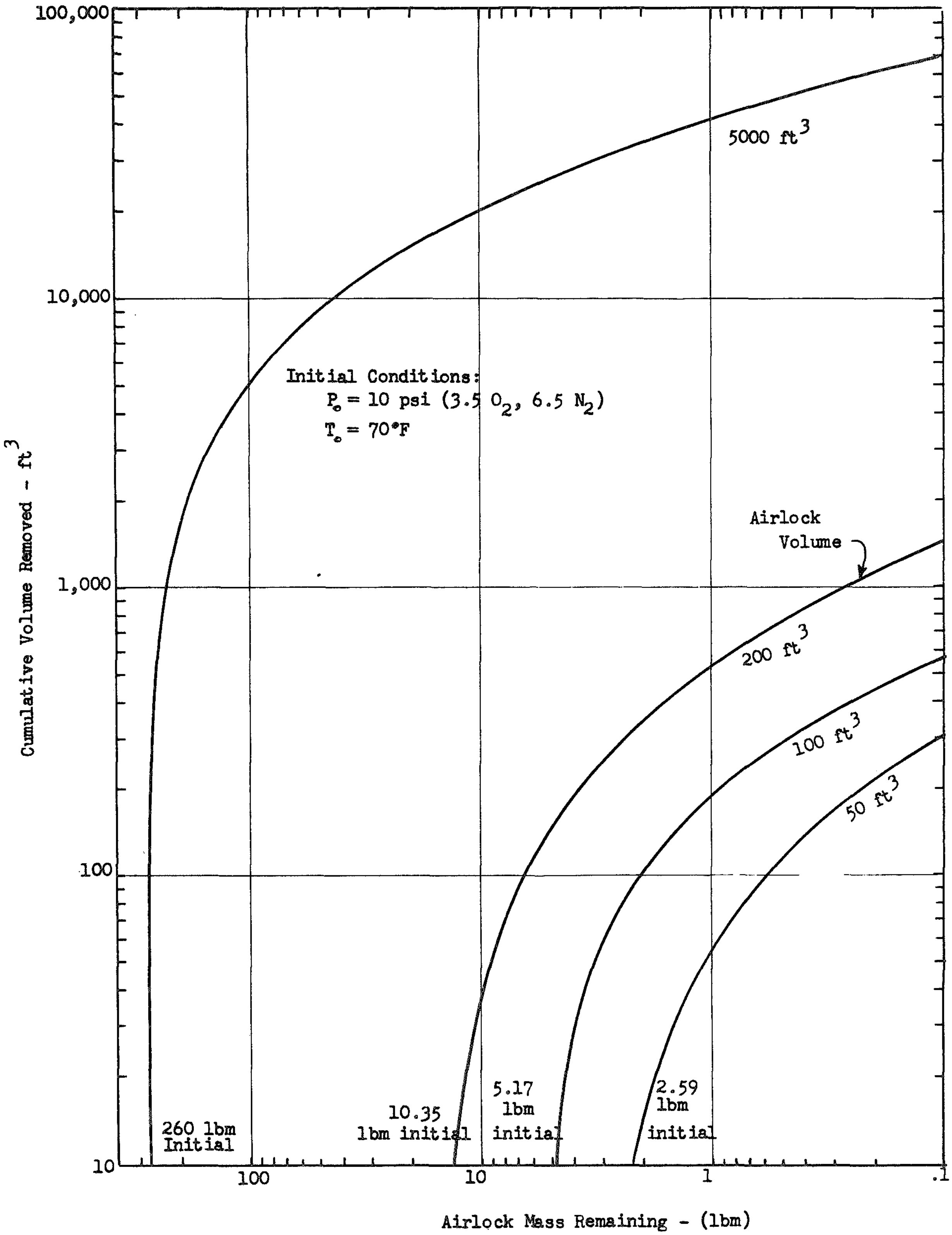

100,000
10,000
1,000
100
10
Cumulative Volume Removed - ft^3
5000 ft^3
Initial Conditions:
P_o = 10 psi (3.5 O_2, 6.5 N_2)
T_o = 70°F
Airlock Volume
200 ft^3
100 ft^3
50 ft^3
260 lbm Initial
10.35 lbm initial
5.17 lbm initial
2.59 lbm initial
100
10
1
.1
Airlock Mass Remaining - (lbm)

FIGURE A-3

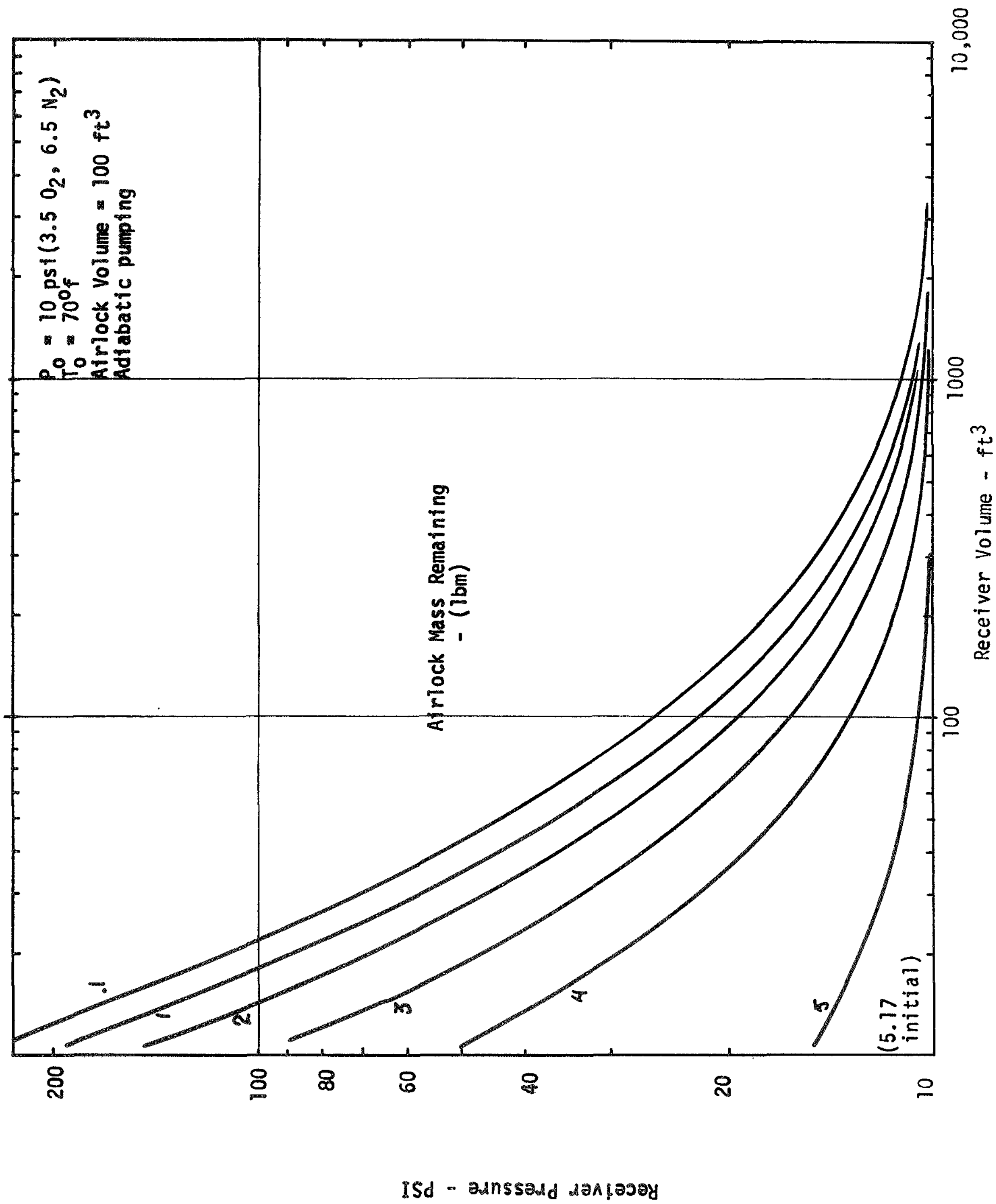

P_o = 10 psi (3.5 O_2, 6.5 N_2)
T_o = 70°f
Airlock Volume = 100 ft³
Adiabatic pumping
Airlock Mass Remaining
- (lbm)
.1
1
2
3
4
5
(5.17 initial)
200
100
80
60
40
20
10
100
1000
10,000
Receiver Volume - ft³
Receiver Pressure - PSI

FIGURE A-4a

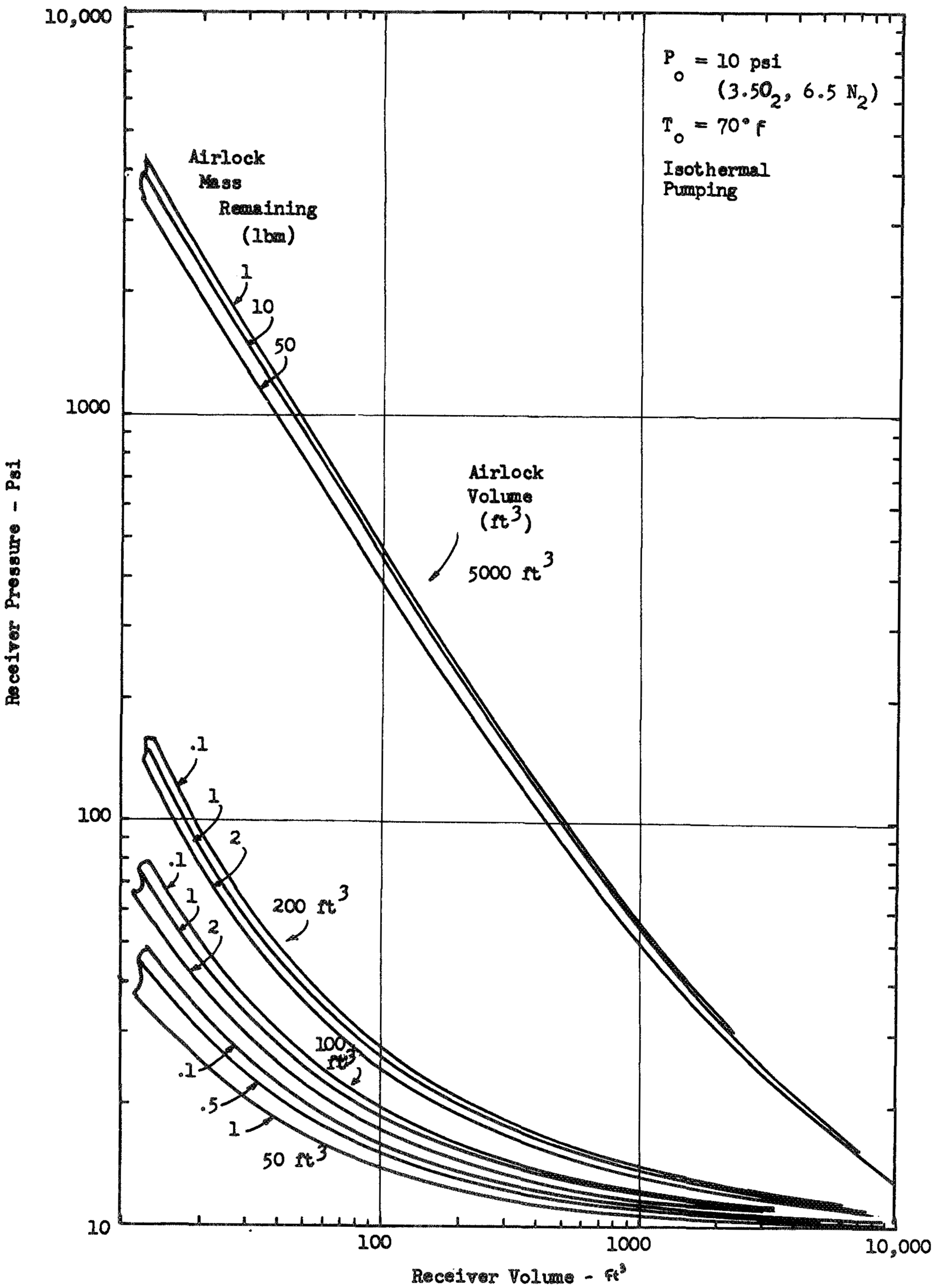

10,000
1000
100
10
Receiver Pressure - Psi
100
1000
10,000
Receiver Volume - ft³
P_o = 10 psi
(3.5 O_2, 6.5 N_2)
T_o = 70° f
Isothermal
Pumping
Airlock
Mass
Remaining
(lbm)
1
10
50
Airlock
Volume
(ft³)
5000 ft³
.1
1
2
200 ft³
.1
1
2
100 ft³
.1
.5
1
50 ft³

FIGURE A-4b

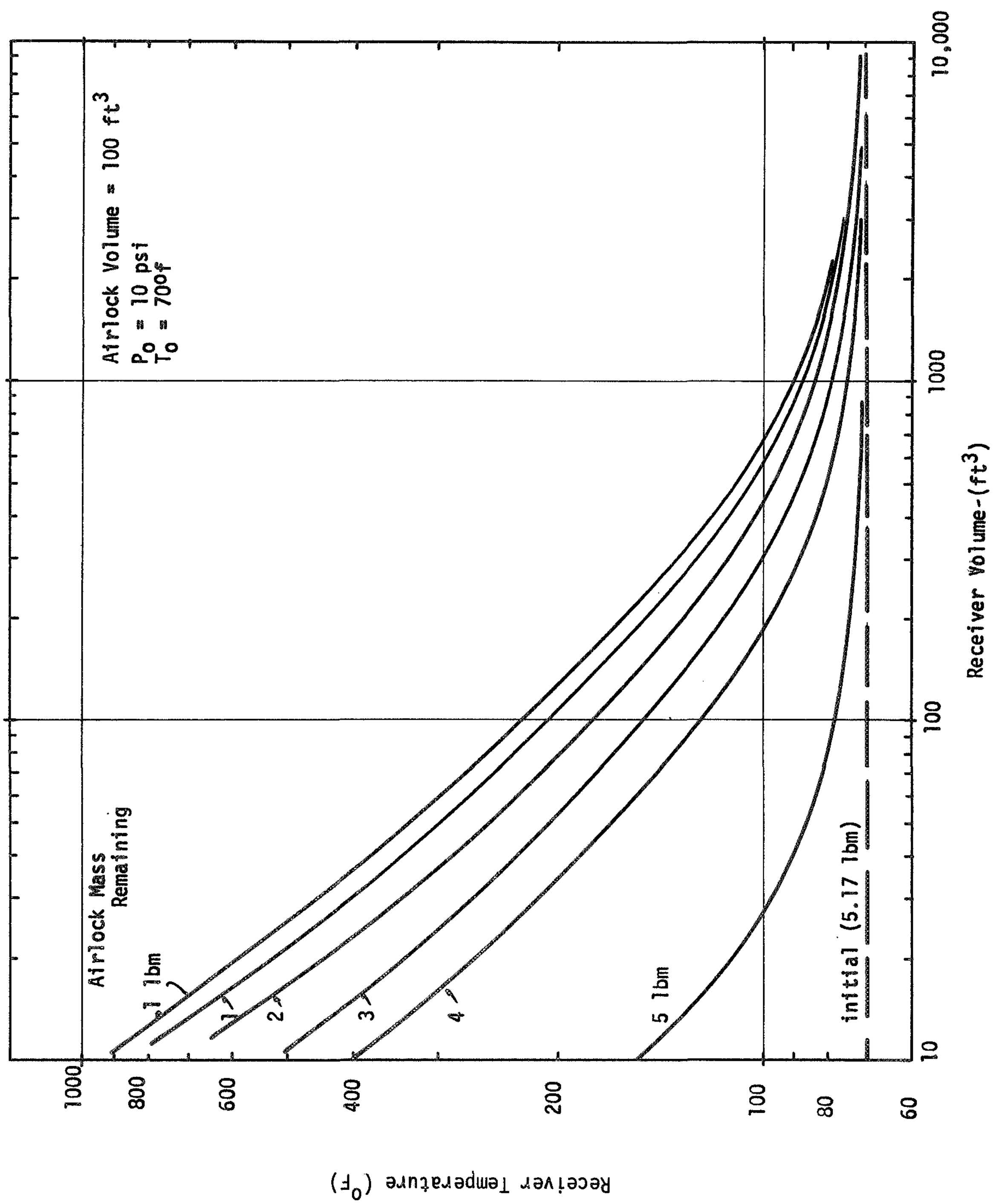

Airlock Volume = 100 ft^3
P_o = 10 psi
T_o = 70°f
Airlock Mass
Remaining
.1 lbm
1
2
3
4
5 lbm
initial (5.17 lbm)
10
100
1000
10,000
Receiver Volume-(ft^3)
1000
800
600
400
200
100
80
60
Receiver Temperature (°F)

FIGURE A-5

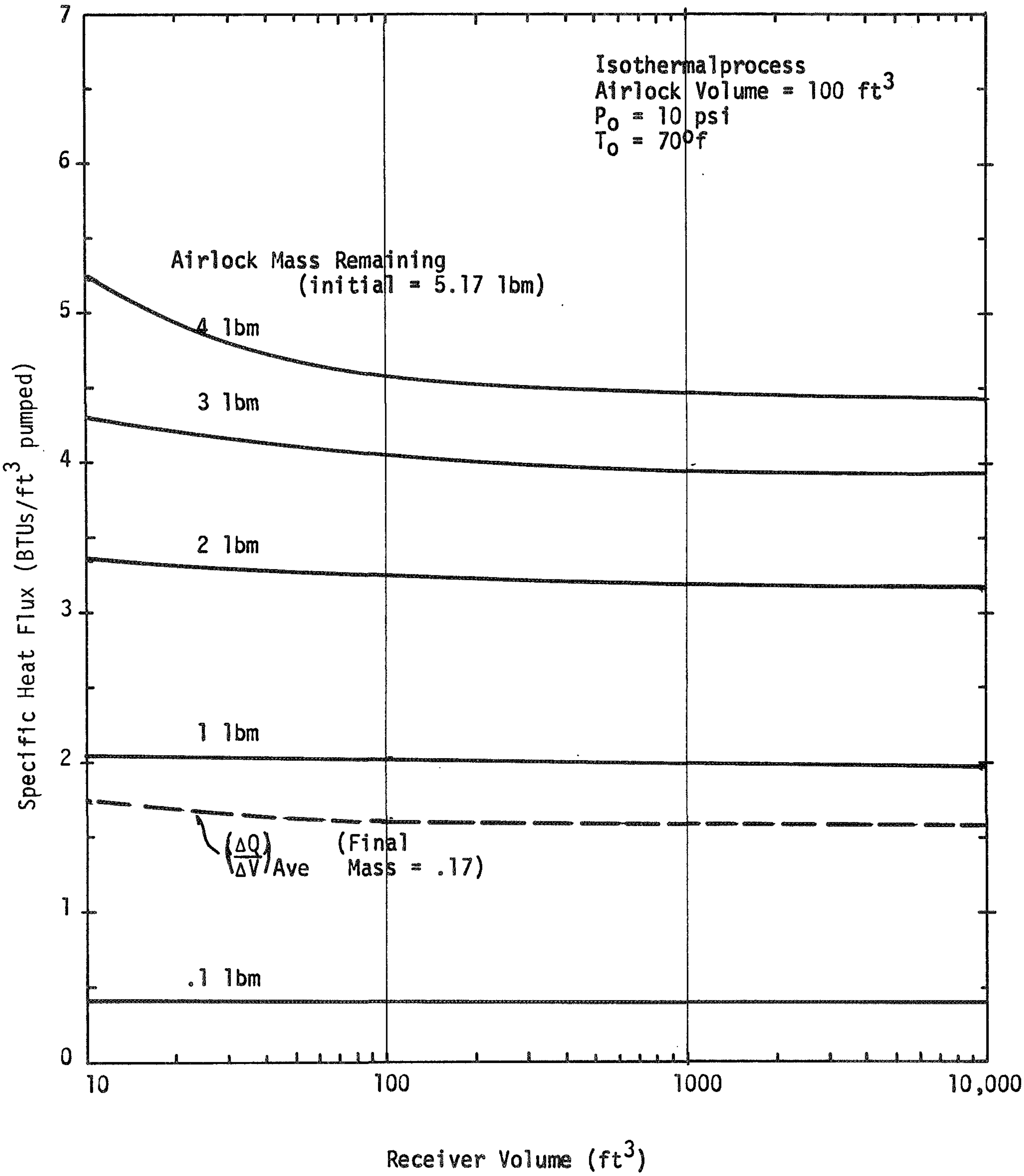

Isothermalprocess
Airlock Volume = 100 ft^3
P_o = 10 psi
T_o = 70^{o}f
Airlock Mass Remaining
(initial = 5.17 lbm)
4 lbm
3 lbm
2 lbm
1 lbm
$\left(\frac{\Delta Q}{\Delta V}\right)$Ave
(Final Mass = .17)
.1 lbm
Specific Heat Flux (BTUs/ft^3 pumped)
0
1
2
3
4
5
6
7
10
100
1000
10,000
Receiver Volume (ft^3)

FIGURE A-6

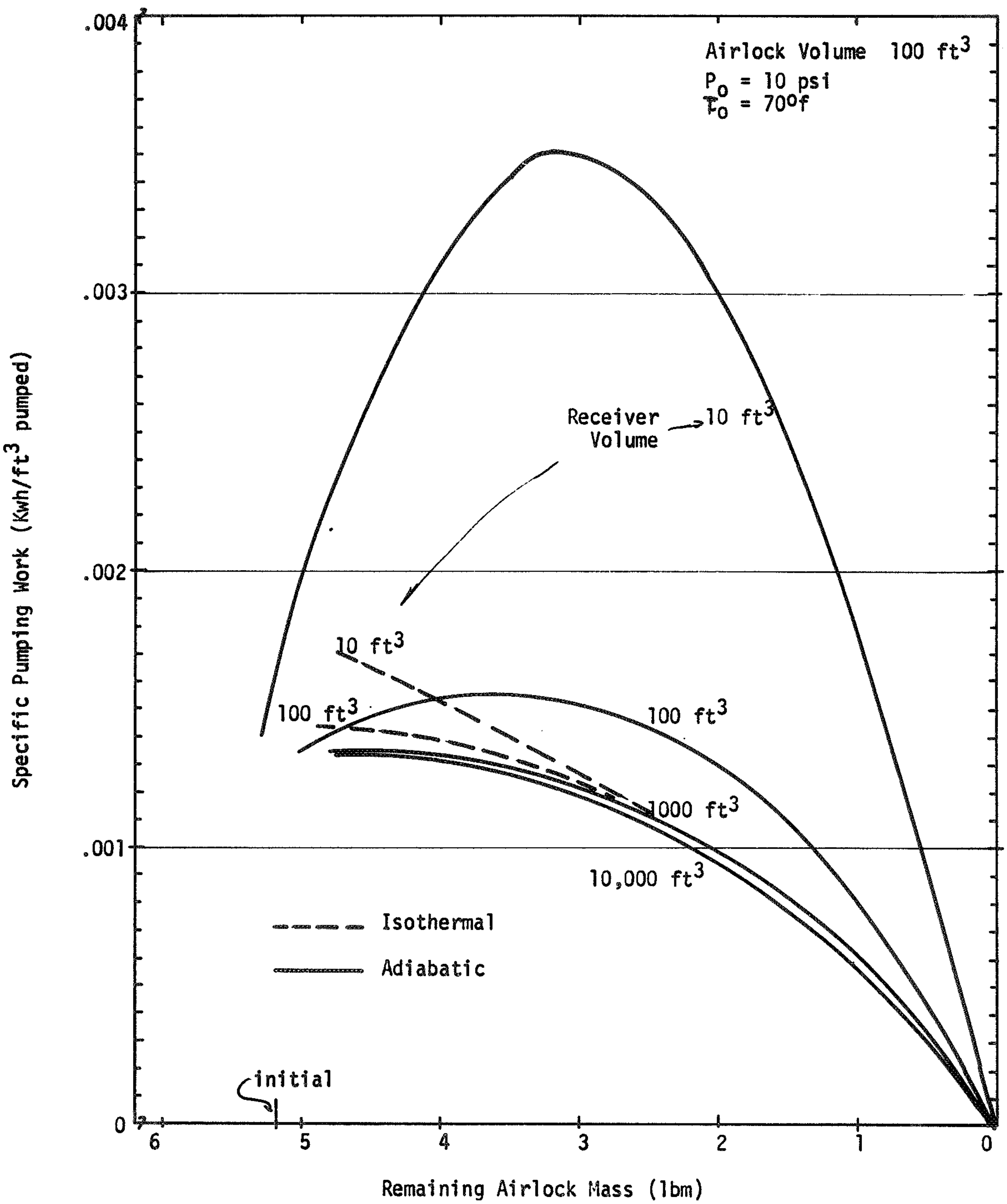

Airlock Volume 100 ft3
Po = 10 psi
To = 70°f
Receiver Volume
10 ft3
10 ft3
100 ft3
100 ft3
1000 ft3
10,000 ft3
Isothermal
Adiabatic
initial
.004
.003
.002
.001
0
6
5
4
3
2
1
0
Specific Pumping Work (Kwh/ft3 pumped)
Remaining Airlock Mass (lbm)

FIGURE A-7

Cumulative work required vs. receiver volume

Adiabatic - Figure A-8a

Isothermal - Figure A-8b

Average power required vs. pumpdown time

Adiabatic process, receiver volume 10 ft^3 - Figure A-9a

Isothermal process, receiver volume > 1000 ft^3 - Figure A-10a

Peak power required vs. pumpdown time

Adiabatic process, receiver volume 10 ft^3 - Figure A-9b

Isothermal process, receiver volume > 1000 ft^3 - Figure A-10b

Sensitivity:

Airlock volume - volumes of 50, 200, 5000 ft^3 were analyzed as follows:

Volume removed vs. airlock mass remaining

50 ft^3

200 ft^3 Figure A-3

5000 ft^3

Receiver pressure vs. receiver volume (isothermal)

50 ft^3

200 ft^3 Figure A-4b

5000 ft^3

Average heat flow rate (Btu's/ft^3 removed) vs. airlock volume, receiver volume > 1000 ft^3 - Figure A-11

Cumulative pumping work (isothermal vs. airlock volume - Figure A-12

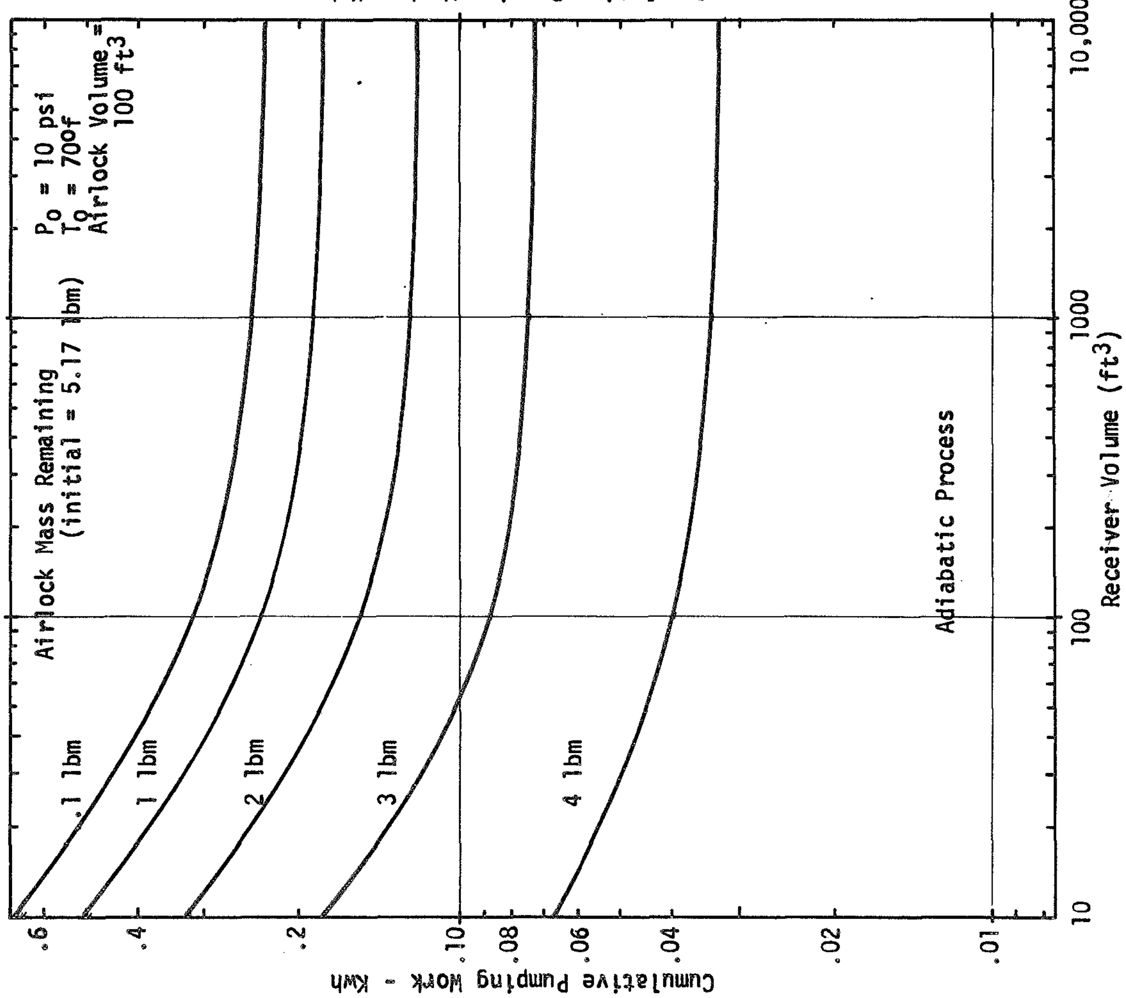


FIGURE A-8a

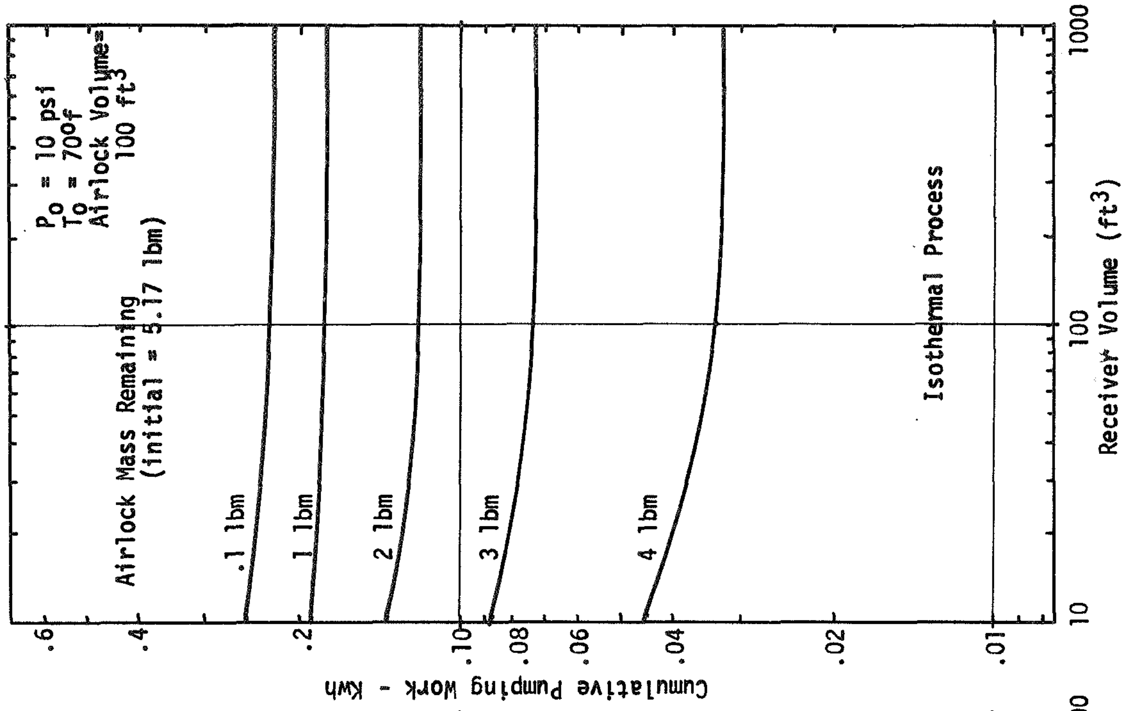


FIGURE A-8b

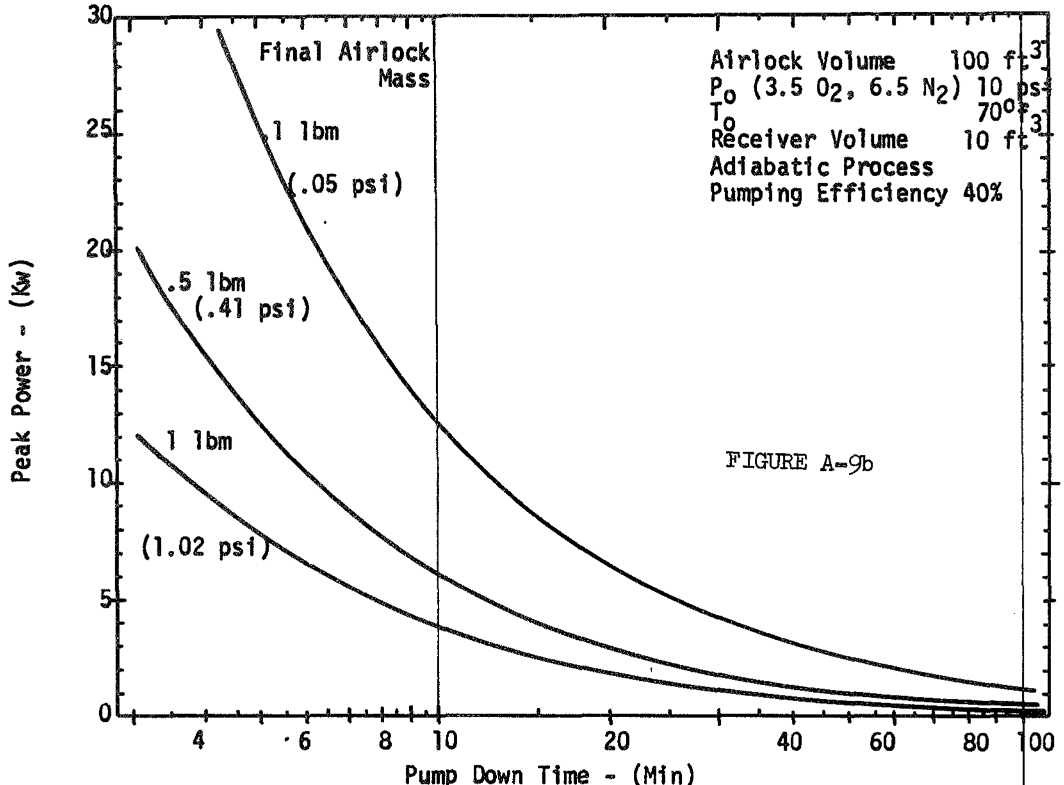


FIGURE A-9b

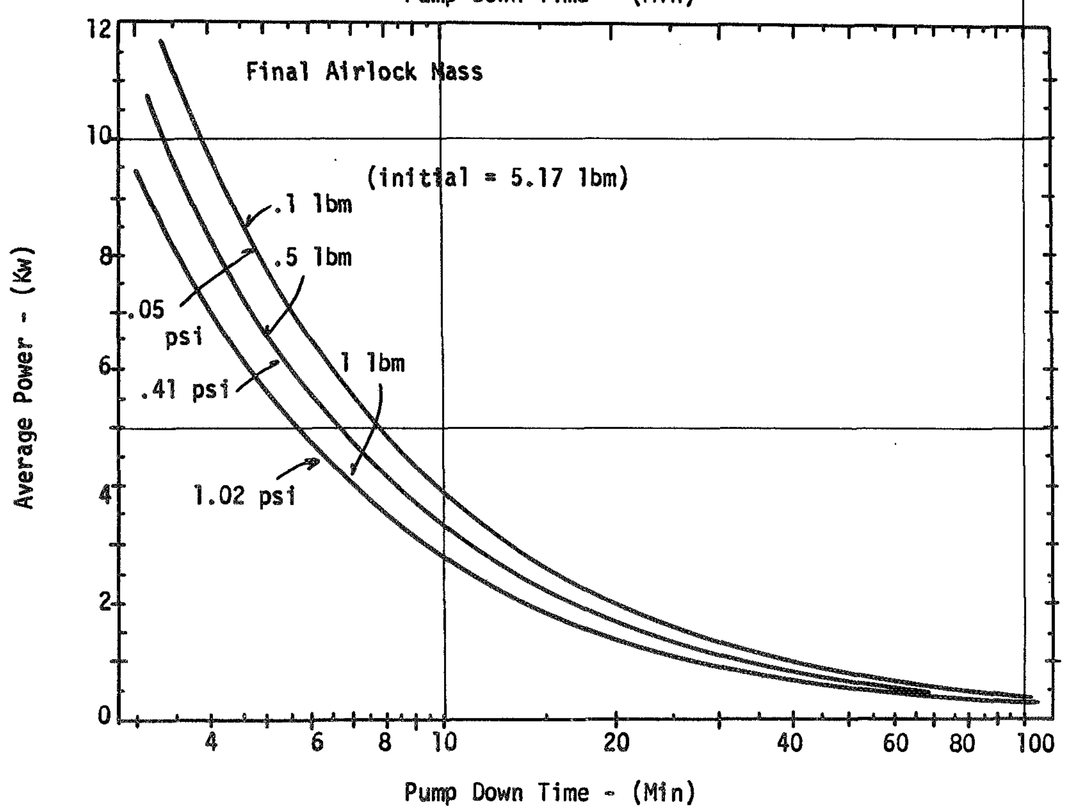


FIGURE A-9a

Temp. Fm. AP-86, Exp. 5-30-71

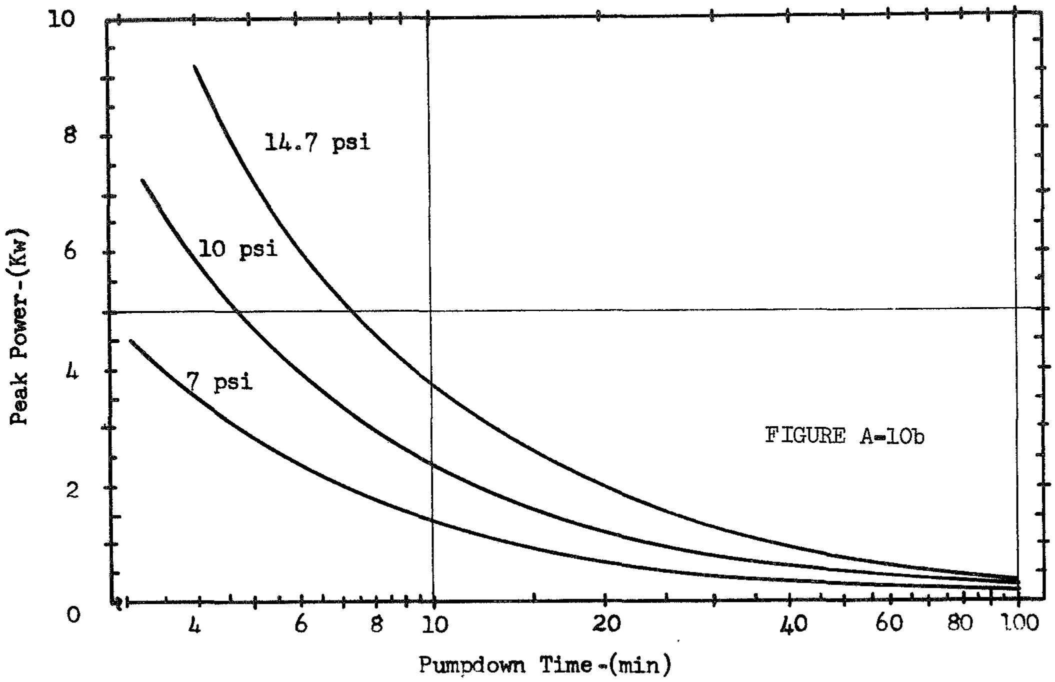


FIGURE A-10b

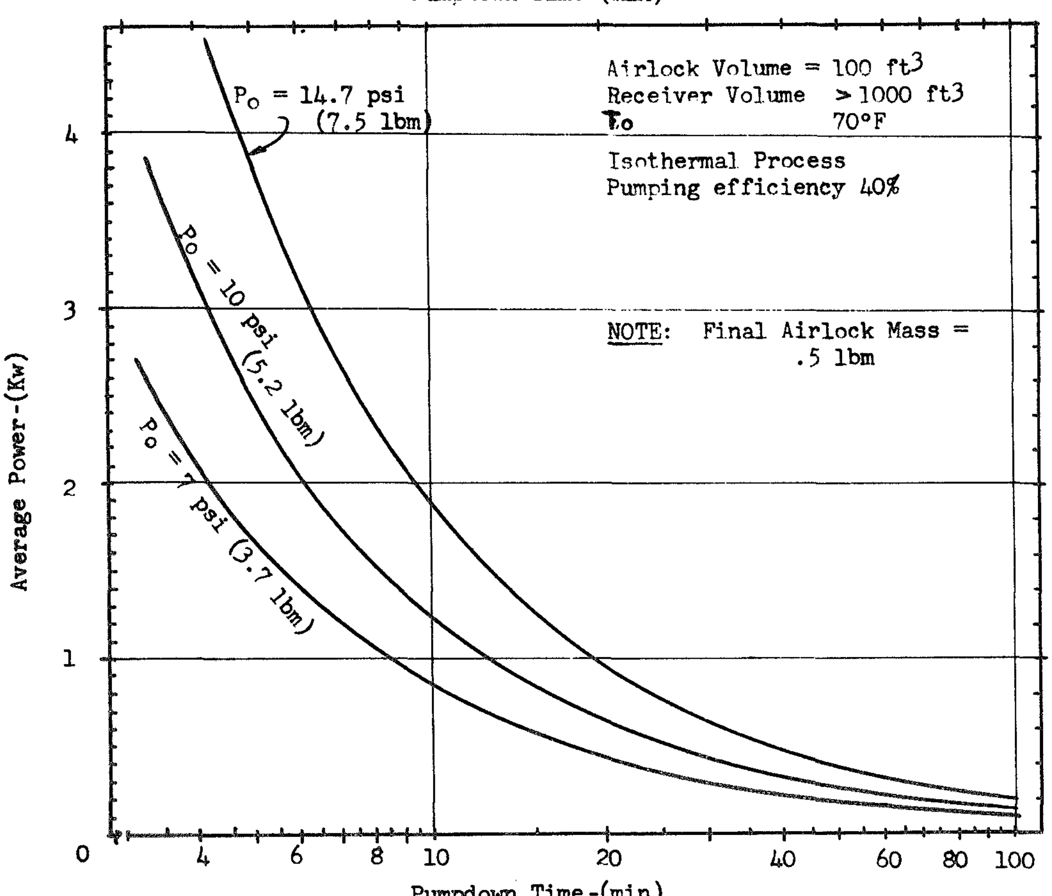


FIGURE A-10a

Temp. Fm. AP-86, Exp. 5-30-71

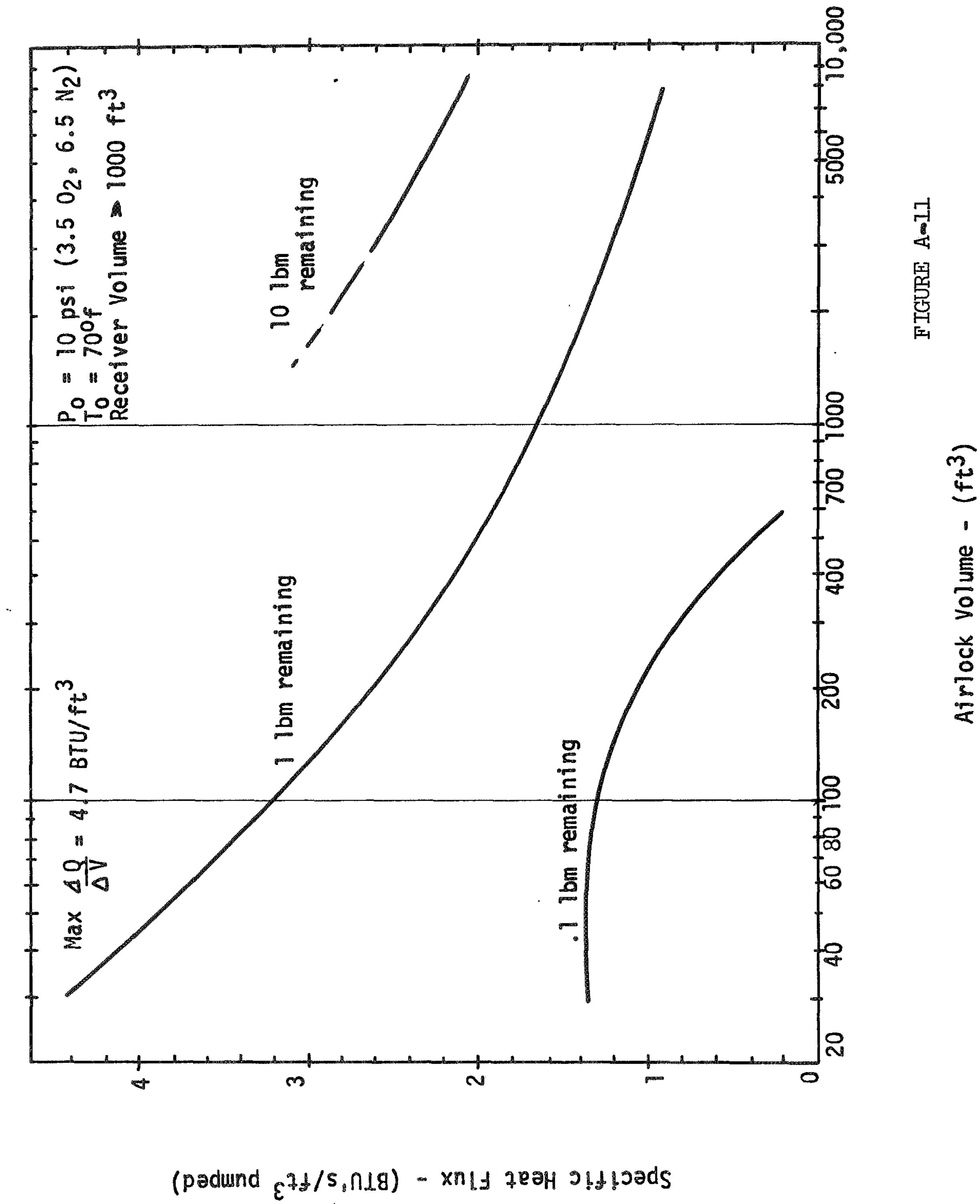

Max ΔQ/ΔV = 4.7 BTU/ft³
Pₒ = 10 psi (3.5 O₂, 6.5 N₂)
Tₒ = 70°f
Receiver Volume ≫ 1000 ft³
10 lbm remaining
1 lbm remaining
.1 lbm remaining
Specific Heat Flux - (BTU's/ft³ pumped)
Airlock Volume - (ft³)
0
1
2
3
4
20
40
60
80
100
200
400
700
1000
5000
10,000

FIGURE A-11

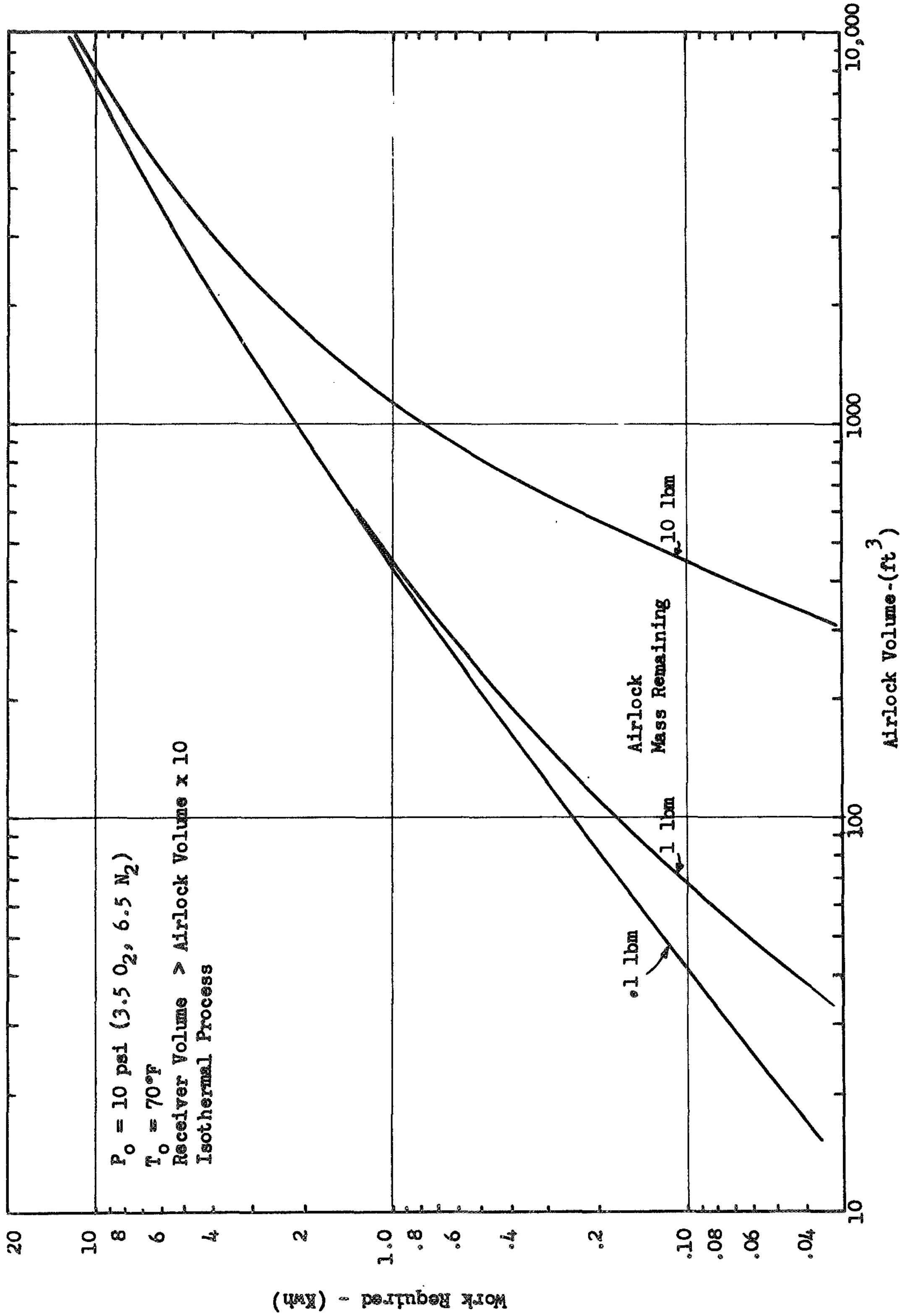

P_o = 10 psi (3.5 O_2, 6.5 N_2)
T_o = 70°F
Receiver Volume > Airlock Volume x 10
Isothermal Process
Airlock
Mass Remaining
10 lbm
1 lbm
.1 lbm
Airlock Volume-(ft^3)
10
100
1000
10,000
Work Required - (Kwh)
20
10
8
6
4
2
1.0
.8
.6
.4
.2
.10
.08
.06
.04

FIGURE A-12

Average pumping power vs. airlock volume (parameterized with respect to pump down times and pumping rates) - Figure A-13

Peak pumping power vs. airlock volume (above) - Figure A-14

Atmosphere selection - pressures of 7 psi (3.5 P_{O_2}, 3.5 P_{N_2}) and 14.7 psi (standard atmosphere) were analyzed as follows:

Average pumping power vs. pump down time, airlock volume 100 ft^3, receiver > 1000 ft^3 - Figure A-15

Peak pumping power vs. pump down time - Figure A-16

<u>Sample Calculation:</u>

Airlock Pump Down Performance

1. Assumptions:

   Airlock volume = 100 ft^3

   Air mass remaining in airlock = 1 lbm

   Pump down time = 10 minutes

   To Determine:

   Volume of air pumped

   Volume rate pumped

   Work required

   The volume of air pumped can be determined from Figure A-3. Using the parameter of air mass remaining of 1 lbm and an airlock volume of 100 ft^3, read volume of air pumped of 190 ft^3. From Figure A-12 using the same parameter of 100 ft^3 and 1 lbm, read power required of 0.17 KWh. Assuming a pumpdown

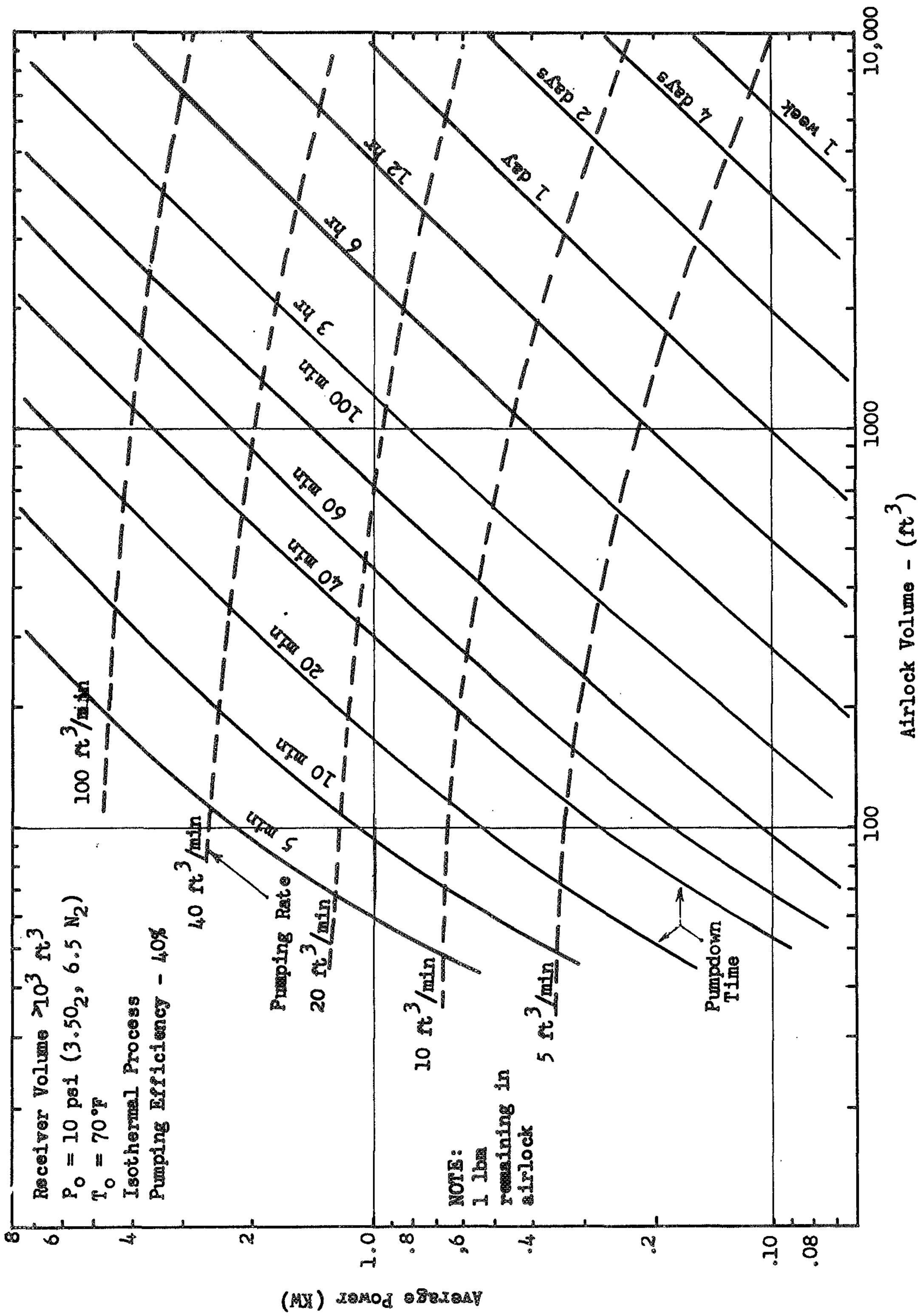

Receiver Volume >10^3 ft^3
P_o = 10 psi (3.5O_2, 6.5 N_2)
T_o = 70°F
Isothermal Process
Pumping Efficiency - 40%
NOTE:
1 lbm
remaining in
airlock
Pumping Rate
100 ft^3/min
40 ft^3/min
20 ft^3/min
10 ft^3/min
5 ft^3/min
Pumpdown
Time
5 min
10 min
20 min
40 min
60 min
100 min
3 hr
6 hr
12 hr
1 day
2 days
4 days
1 week
Airlock Volume - (ft^3)
100
1000
10,000
Average Power (KW)
8
6
4
2
1.0
.8
.6
.4
.2
.10
.08

FIGURE A-13

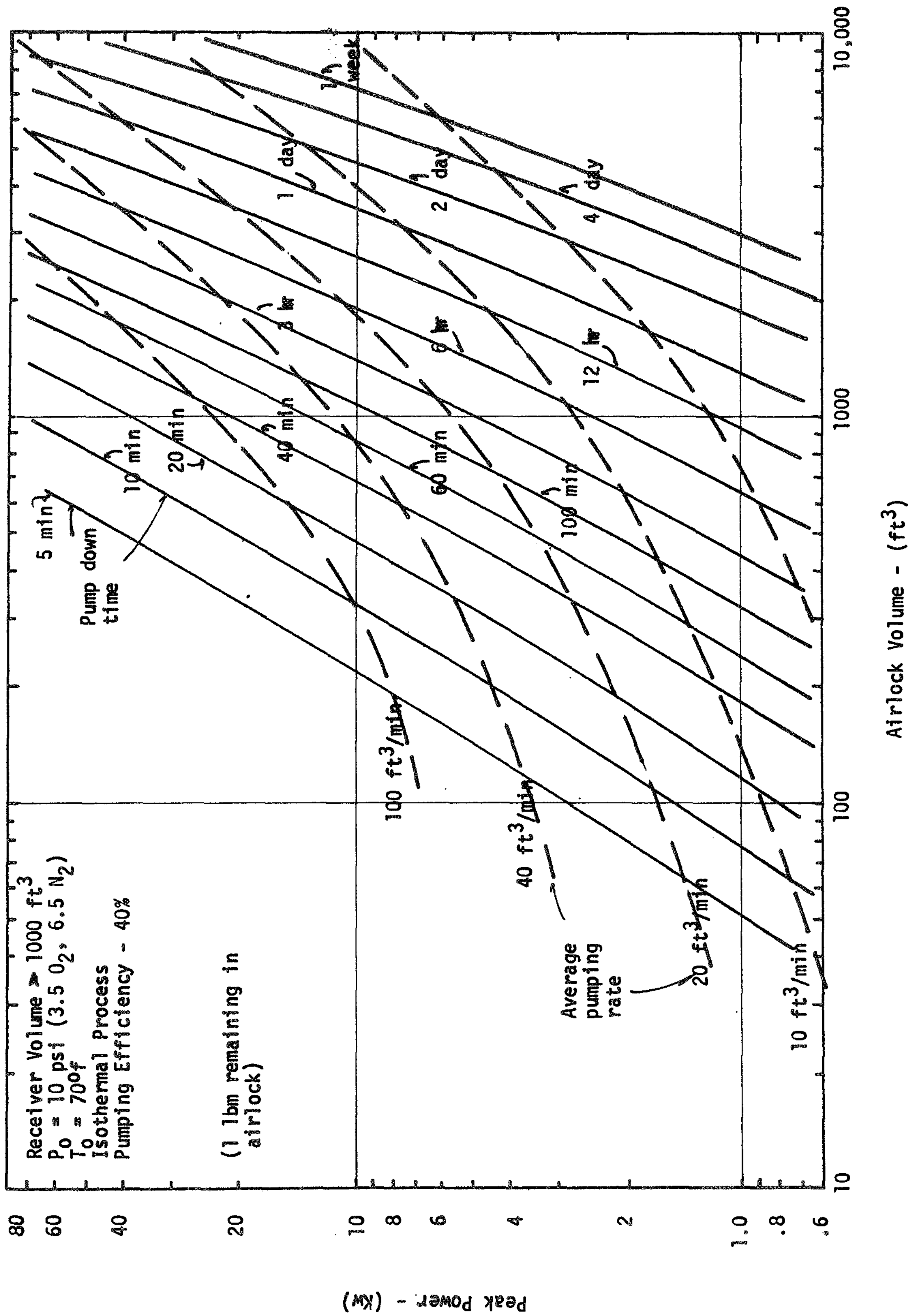
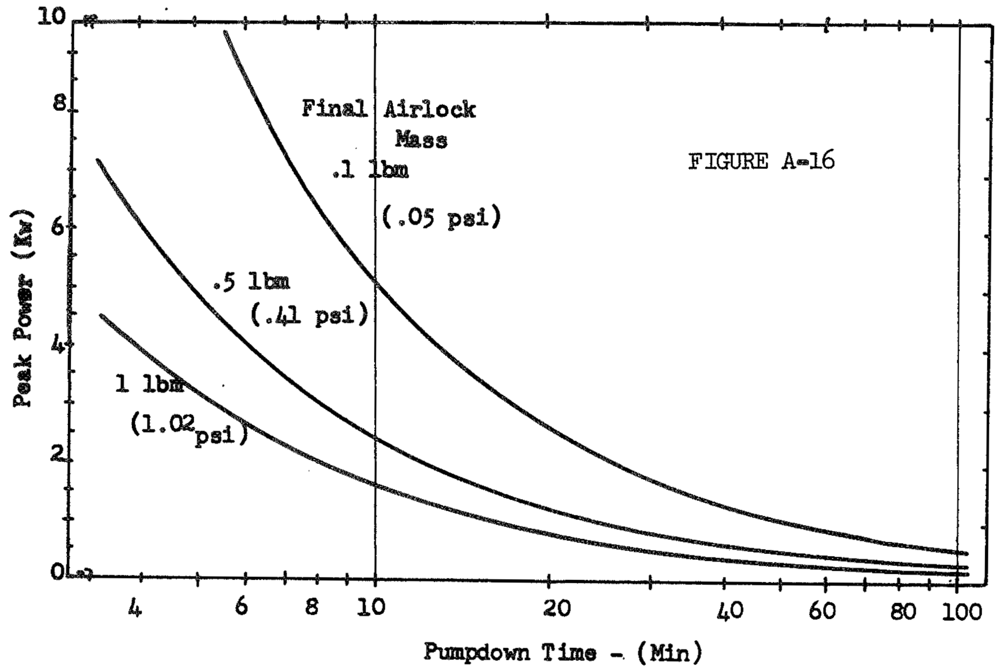

Receiver Volume ≫ 1000 ft^3
P_o = 10 psi (3.5 O_2, 6.5 N_2)
T_o = 70°F
Isothermal Process
Pumping Efficiency - 40%
(1 lbm remaining in airlock)
Pump down time
5 min
10 min
20 min
40 min
60 min
100 min
3 hr
6 hr
12 hr
1 day
2 day
4 day
1 week
Average pumping rate
100 ft^3/min
40 ft^3/min
20 ft^3/min
10 ft^3/min
Peak Power - (KW)
80
60
40
20
10
8
6
4
2
1.0
.8
.6
Airlock Volume - (ft^3)
10
100
1000
10,000

FIGURE A-14


10
8
6
4
2
0
Peak Power (Kw)
Final Airlock
Mass
.1 lbm
(.05 psi)
FIGURE A-16
.5 lbm
(.41 psi)
1 lbm
(1.02 psi)
4
6
8
10
20
40
60
80
100
Pumpdown Time - (Min)

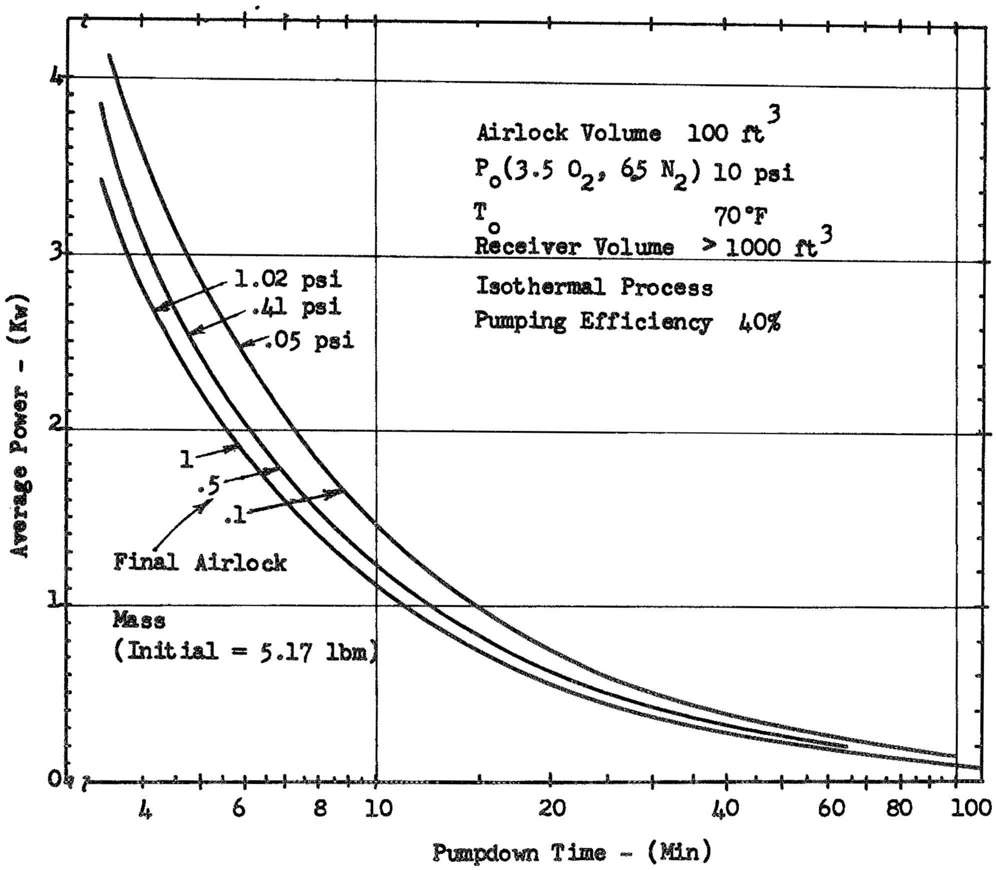
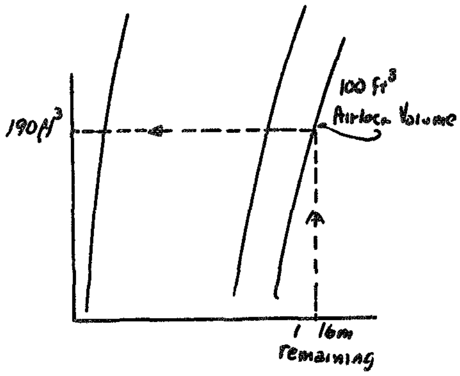
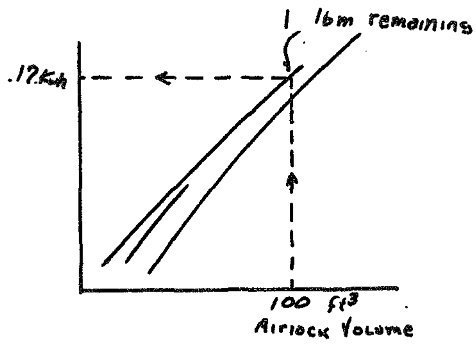

4
3
2
1
0
Average Power - (Kw)
Airlock Volume 100 ft³
Po(3.5 O2, 6.5 N2) 10 psi
To 70°F
Receiver Volume > 1000 ft³
Isothermal Process
Pumping Efficiency 40%
1.02 psi
.41 psi
.05 psi
1
.5
.1
Final Airlock
Mass
(Initial = 5.17 lbm)
4
6
8
10
20
40
60
80
100
Pumpdown Time - (Min)

FIGURE A-15

time of 10 minutes, the volume rate pumped is 190 ft^3/10 min. = 19 ft^3/min. The power required in (KW) = 0.17 KWhr/1/6 hr = 1.02 KW. These values can be checked by using Figure A-13; however, this figure is only good for the condition where the airlock is pumped down to 1 lbm remaining.



From Figure A-3



From Figure A-12

Work = .17 KWH

Volume = 190 ft^3

If pumpdown time = 10 min. (1/6 hr.)

Then:

19 ft^3/min. pump rate

1.02 KW of power

2. If airlock volume = 200 ft^3 and 2 lbm remaining:

Then:

350 ft^3 pumped

.26 KWH power required

If 30 ft^3/min pumped

Then:

Time = 12 min.(1/5 hr.)

Average power = .26 x 5 = 1.3 KW

SYSTEM EVALUATION

For option evaluation, airlock atmosphere, positive displacement and scavenge pump weights have been determined and presented in Figure A-17.

Cryogenic tankage penalties have been identified (Reference A-2) to be:

O_2 = .310 lbm/(lbm O_2)

N_2 = .75 lbm/(lbm N_2)

H_e = 3.00 lbm/(lbm H_e)

Power system evaluation determines the cost of energy and power estimated to be:

500 lbm/average daily kw's of power

The structural penalties for tunnels (based on 1 lbm/ft^2) are:

82" x 40" x length = 17 lbm/ft length tunnel

60" x 40" x length = 14 lbm/ft length tunnel

Hatches are estimated at:

150 lbm/hatch

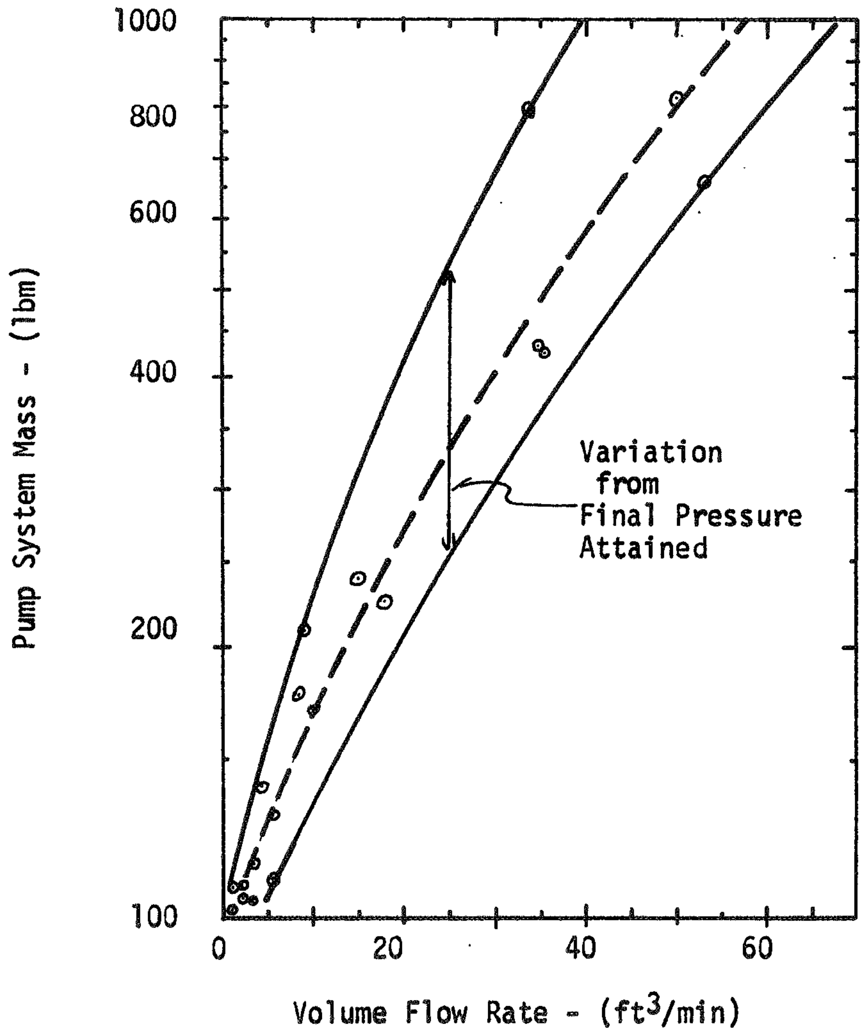
Airlocks, a combination of the above, are estimated at:

84" x 60" x length; 2-hatch = 300 lbm + 24 lbm/ft length

3-hatch = 450 lbm + 24 lbm/ft length-1.4 ft

60" x 60" x length; 2-hatch = 300 lbm + 20 lbm/ft length

3-hatch = 450 lbm + 20 lbm/ft length-1.25 ft



Laboratory Vacuum Pump Weights

FIGURE A-17

The following example shows the effect of internal passages vs. EVA transfer:

Consider two modules requiring one cycling of two crew members daily (e.g., crew module to base operations and return). For EVA transfer through an 84" x 60" x 60" airlock (similar apparatus at the second module) with a twenty-minute pumpdown, a minimum of fifty-five minutes is required for each transfer (10 minutes of pumpdown concurrent with suit checkout) for a total of one hour, 50 minutes daily required for a complete cycle. The weight penalties are:

Systems

2 two-hatch airlocks	840 lbm
Airlock pump & motor	100 lbm
Pumping power	
(40 min/day @ .7 Kw AV)	10 lbm
	950 lbm

Consumables

Manpower (@ 12.5 lbm/manday)	1.9 lbm/day
Airlock dump (.1 lbm each)	.2 lbm/day
	2.1 lbm/day

For just the basic systems weight an equivalent tunnel length can be determined:

950 lbm - 300 lbm hatches = 650 lbm for tunnel,
@ 17 lbm/ft gives
38 ft. length (82" x 40")

In addition, for each day of operation, the tunnel length can be extended .12 feet or:

Equivalent tunnel length = 38 ft + .12 ft/day

RECOMMENDED CONCEPT

The fully integrated concept, is recommended as being the better operational and minimum weight concept. The various physical properties are:

Main internal hatches, access ways: 80" x 34"

Large airlock hatches: 80" x 60"

Small airlock hatches: 80" x 40"

Tunnels: 82" x 40" x length

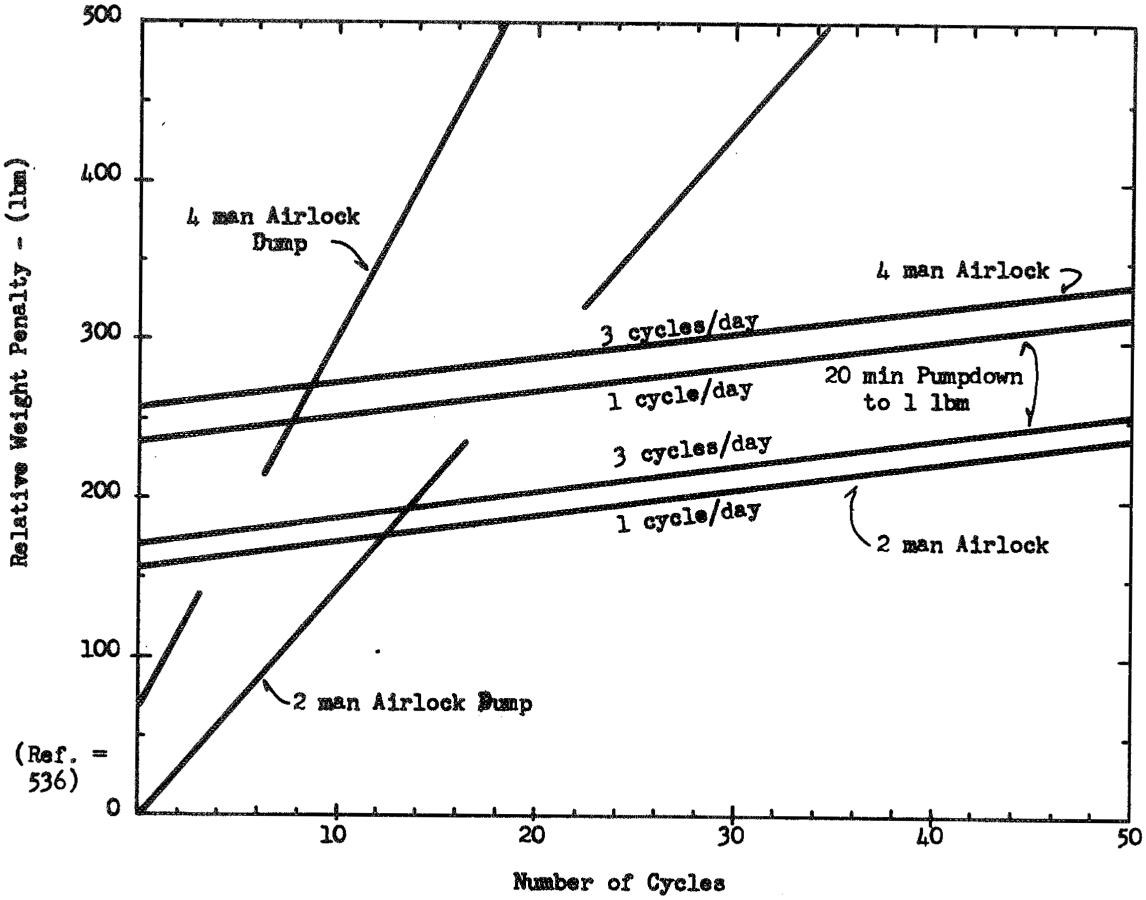
Large airlock (4-man): 84" x 60" x 96" (280 ft^3)

Small airlock (2-man): 84" x 60" x 60" (175 ft^3)

The advantages of such a configuration are:

1. Integrated Concept: Allows inter-module transfer between those modules frequently visited with the need of EVA, thus saving the weight associated with module air dumping (either total or after partial pumpdown), pumping and airlock equipment, and suit wear, and the time associated with suit donning, checkout, airlock pumpdown, dust control doffing and suit maintenance. Hatches will be required anyway and by placing modules in close proximity the tunnel or connecting chamber weight is minimized.

2. Airlocks: The two airlock configurations of varying sized airlocks allows the choice of size of airlock to be pumped; naturally, the smaller of the two should be used if its capacity is sufficient. Figure A-18 shows



Assumptions	Volume	3 Match Airlock Mass	Mass of Air Contained	Pumping Power Penalty
4 man Airlock:	280 ft^3	608 lbm	14.5 lbm	12.5 lbm/cycle/day
2 man Airlock:	175 ft^3	536 lbm	9.05 lbm	7.5 lbm/cycle/day
Pump and motor:	150 lbm			
Tankage Penalty:	O_2 - .3 lbm/lbm O_2 N_2 - .75 lbm/lbm N_2			

$\Rightarrow$.6 lbm/lbm air (10 psi)

FIGURE A-18

the weight advantage of airlock pumpdown vs. direct dump as a function of cycling. This allows the egress/ingress of six men (4 + 2) daily with only two pumpdown cycles. If just one four-man airlock is available, then three pumpdown cycles are required (two to egress, one to ingress after the first crew has entered). A six-man airlock would alleviate this problem, but then all six must egress/ingress simultaneously - not an unreal constraint in the first stages of base activation/operation but in the latter stages it is doubtful six men will be required to egress/ingress. For supplies and vehicle maintenance the garage and, perhaps, warehouse will be evacuated for cargo and vehicle transfer.

3. <u>Contingencies</u>: With the integrated concept only the incapacitation of multiple modules will divide the complex - in all other cases there exists either a primary or secondary route available for internal transfer.

4. <u>Vehicle Integration</u>: To pump down the garage facilities for daily vehicle usage would place unjustifiable requirements on both power and pumpdown time. Likewise, an EVA transfer would impose similar requirements on vehicle pumpdown and dust control in addition to that on the LSB. It is recommended that a docking port be utilized (similar to current airport facilities) to allow immediate internal transfer, thus avoiding these mentioned problems. Minimum vehicle modification and/or cargo handling could be accomplished by another returning EVA crew, if necessary; sortie preparation will still be accomplished in the garage facilities.

REFERENCES

A-1. Engineering Criteria for Spacecraft Cabin Atmosphere Selection, Douglas Missiles and Space Systems Division Report DAC-59169, November 1966.

A-2. LESA: Human Factors and Environmental Control/Life Support Systems, Vol. 6, Garrett Airesearch Report SS 3243-6.

APPENDIX B. ACTIVE THERMAL CONTROL

Thermal control of the lunar base complex consisting of the semi-permanent base shelter and the sortie convoy has been investigated briefly to identify possible or promising methods to maintain the design temperatures for the internal environment of the personnel shelters and the equipment located within these shelters.

The semi-permanent base shelter complex was assumed to be adequately insulated from the lunar environment so there would be negligible heat leak in or out of the shelter. It was also assumed that the base would be located at or near the equator which would result in extreme thermal environmental conditions. An isotopic power system is being considered for the base complex, but it was assumed to be located away from the personnel shelter so that it would not be practical to utilize the power system waste heat to help achieve the required thermal control.

In contrast to a fixed base shelter, the sortie convoy would consist of mobile units, such as the motorized personnel units which would not only be used to propel the convoy but also serve as a short-term living quarters and a mobile shelter which would serve as the living quarters for the crew when they are away from the main base complex. These mobile units would be subjected to the varying thermal environment, and it was assumed that a combination of insulation and surface coating would be utilized to minimize the heat leak in or out of these personnel units. Previous studies, such as the MOBEV study conducted by Bendix Corporation (Reference B-1) indicate a heat leak range of +1000 to -1700 Btu/hr for a four-man unit. No attempt was made during this brief and preliminary study to estimate the heat leak or to identify the type of insulation and surface coating. This would be done after more design details have been established. Similar to the base complex, an isotopic power system is being considered for the sortie convoy, and the close proximity of the power system offers the possibility of utilizing its waste heat to help achieve thermal control by providing heat to the personnel quarters or some process associated with thermal control.

In addition to the above considerations and assumptions, it was further assumed that the gross type and nature of the equipment are similar for both the base complex and the sortie convoy, so the following temperature ranges would be applicable:

1. Electronic and scientific equipment = -30 to 120 F
2. Humidity control and water chiller = 45 F
3. Personnel shelter or quarters = 70 + 5 F

The heat rejection load from each of the three sources would be expected to vary considerably. For example, for the 12-man Space Station, the on-board equipment waste heat load ranges from 10,000 to 20,000 Btu/hr, the humidity control and water chiller, about 2000 Btu/hr, and the cabin temperature control, about 4000 Btu/hr. A similar ratio or proportion of waste heat from the three sources for the lunar base complex and sortie convoy would be expected. Further, the waste heat load is expected to vary from the design or maximum value to a lower or minimum value, which would depend upon the operational mode of the base complex.

For the purpose of this preliminary study, the maximum or minimum heat loads from the three areas indicated above were assumed on a gross basis and treated as a parameter. This was considered desirable to provide data for a parametric form for greater utilization.

Table B-1 summarizes the general requirements that are applicable for thermal control and were used as a general guideline for this brief study.

Table B-1. General Requirements

Base Complex	
Total life	2-5 years
Resupply period	180 days
Crew size	2 to 12
Base design	modular, 3-man size
Heat rejection load	variable
Energy sources	electrical-isotopic power solar
Sortie Convoy	
Total life	2 years
Sortie duration	10 to 120 days
Crew size	2 to 4
Personnel shelter	4 man size
Heat rejection load	variable
Energy source	electrical - isotopic power solar waste heat (power system)
Base Location	At or near equator
Sortie Convoy Location	Within 250 miles radius of base.

The results of this study are presented in two sections. The first covers the analysis of the space radiator to establish the desired operating range and limitations of the radiator. This is followed by a discussion of various thermal control concepts which are applicable for the base complex and for the personnel units of the sortie convoy.

Space Radiator

The space radiator was investigated in some detail since this major component of a thermal control system would be subjected to a widely varying environmental condition and, in addition, to varying heat rejection loads. A radiator designed for one condition could be over- or under-designed for another condition so that the radiator temperature and thus the coolant outlet temperature could be either too high or too low. Further, during the lunar night the radiator temperature could drop to a point for which the viscosity could become sufficiently high to require excessively high pumping power.

This investigation was conducted to provide data that could be used to establish the design limits. The results have been presented in parametric plots or curves.

For this analysis, the radiator was assumed to be in a horizontal position, since this results in minimum absorbed energy from direct solar and lunar surface at the subsolar point.

The heat rejection capability of a radiator is dependent upon the radiator temperature and the optical properties of the radiator surface for a given environmental condition. This is illustrated in Figure B-1 for the case of a horizontal radiator at the subsolar condition. As shown, it is readily apparent that low values for the α_s/ϵ ratios and high temperatures are necessary to achieve high heat rejection capability. However, practical considerations and limitations of available materials, such as surface coatings and coolants, would limit the specific heat rejection rate that could be achieved. The radiator operating temperature would be limited by the temperature limits of the available coolants or working fluid and the radiator inlet and outlet temperatures as specified by the equipment and processes that require cooling. The values for the α_s/ϵ ratios, particularly in the lower range depend upon the coatings or finishes that are currently available or will be available and are suitable for the lunar surface base application.

Zinc oxide (Z-93), a radiator coating developed for the Apollo program, would result in a heat rejection rate of about 20 Btu/(hr)(sq.ft.) for a 60 F radiator. This heat rejection rate could be significantly improved by utilizing a relatively recent development which is identified as the optical solar reflector which is described in Reference B-2. For this particular surface finish, the heat rejection rate would be about 80 Btu/(hr)(sq.ft.) for a 60 F radiator. Thus, the radiator area would be about one-fourth the size as compared to one that utilizes Z-93. This is particularly significant for the sortie convoy, since a minimum area is desirable.

The optical solar reflector has demonstrated under simulated space conditions to be completely stable under ultraviolet and particulate irradiation (Reference B-2). However, this type of surface could be extremely susceptible to dust or contaminants and thus lose some heat rejection capability. For the purposes of this study, this particular surface will be considered as the first choice and the Z-93 as the alternate.

For these two surface finishes or coatings, the radiator temperatures at the lunar night condition and reduced heat rejection loads were established for a radiator sized for the lunar day condition at the subsolar point. In essence, this represents a fixed area radiator which would have more than ample heat rejection capability at the lunar night condition and for reduced heat rejection load. This is illustrated in Figure B-2 for the case of the optical solar reflector, $\alpha_s/\epsilon = .06$, and a radiator effectiveness of 0.90.

For example, if the radiator was sized or designed for an average temperature of 60 F, at the subsolar point (the maximum lunar day condition), the average radiator temperature would be about 25 F at the lunar night condition for the heat rejection load at the design value; that is, a heat load ratio of 100%. As the heat load is reduced from the design value, the radiator temperature at the lunar night period would drop correspondingly. For the same 60F radiator, if the heat load is reduced to 25%, the radiator temperature would reach about -105 F for a radiator effectiveness of .80. Whether or not this low temperature is permissible will depend upon the coolant or working fluid that is used and the pumping power that would be required. Figure B-3, from Reference B-3and modified to include hydrogen gas data, illustrates the rapid rise in pumping power, as indicated by the pumping power parameter, with lower temperatures for a number of common coolants. Among the liquid coolants, only Dow-Corning 200 appears feasible at -100 F although its pumping power is relatively higher than others at higher operating temperatures. Another possible working fluid is hydrogen gas, which can be used at extremely low temperatures, which would be beyond any of the common types indicated in Figure B-3. Although hydrogen could be hazardous, much of the risk could be reduced through proper design, such as utilizing a dual loop whereby the hydrogen gas loop would be located externally and a second loop with non-hazardous coolant located inside the personnel shelter or areas.

If Dow-Corning 200 is the only suitable coolant, then any reduction in heat rejection load would be limited to about 25% of the design load. If the heat rejection load must be reduced below 25%, then additional heat must be applied to keep the radiator from dropping below the -100 F limit, or the radiator must be sized initially at a higher temperature; that is, higher than 60 F.

The effect of operating at reduced loads at the subsolar point also results in lower radiator temperature. This is illustrated in Figure B-4.

If the zinc oxide coating is used, which has an α_s/ϵ = .20, the average radiator temperature at the lunar night condition would be considerably lower than for the case of the optical solar reflector. This is apparent by comparing Figures B-5 and B-3. If, for example, a radiator is designed for an average temperature of 60 F at the subsolar point, then at the lunar night condition this same radiator would have an average temperature of about -100 F at the design or 100% heat rejection load. If the heat rejection load was reduced to 25%, the average radiator temperature would drop to about -200 F. For this condition, the Dow-Corning 200 would not be a suitable coolant, and hydrogen gas would be the only possible candidate.

The foregoing discussion has been based on sizing the radiator at the subsolar condition, that is, the maximum condition of the lunar day and then determining the resultant radiator temperatures at the lunar night condition.

As an alternate possibility, an analysis was made to establish the radiator temperature at the subsolar point for a radiator sized at the lunar night condition for specified or desired radiator temperatures. In this case, the radiator would be undersized for the lunar day so that the radiator temperature would be considerably higher than the desired temperature. This is illustrated in Figure B-6 for several values for the ratio, α_s/ϵ . For example, if the radiator temperature is to be maintained between 40 and 60 F at the lunar day condition and between -20 and 0 F at the lunar night condition, the surface coating or finish must have optical properties with

α_s/ϵ = .10 or less. This can be satisfied by the optical solar reflector. Also, there would be several possible coolants that could be considered for the temperature ranges indicated. However, these temperature ranges are for the design or 100% heat rejection load. For reduced loads, the minimum temperature during the lunar night condition could drop below -100 F (a 25% load condition).

On the other hand, if the radiator was sized initially for lunar night condition and at a reduced load condition so that the coolant temperature would never drop below a design value, then the radiator temperature at the lunar day condition and for the design heat load must be raised to a relatively high value in order to reject all the waste heat. For example, if the radiator were sized at the lunar night condition for a minimum temperature of -20 F and a reduced heat load of 25%, then at the subsolar point of the lunar day, the radiator must operate at about 190 F to reject the design or total heat load for a surface coating of α_s/ϵ = .10. This would then require an active system such as a vapor compression refrigeration system to be operating during the lunar day period in order to raise the radiator or heat rejection temperature to the 190 F level.

An alternative to a fixed area radiator would be a variable area radiator so that only the necessary heat rejection area would be utilized. This approach would allow a wider range for the type of surface coating and the coolant that could be used; however, some added complexity would be involved in achieving the variable area with varying environmental condition and heat loads.

The amount of complexity would depend on the means or method of achieving the variable area. Two possible approaches which are relatively simple are discussed. One method or approach would be to divide the radiator into several panels with each panel having a flow control valve so that, as the environment and/or the heat rejection load changes, the panels would be shut off or on and only the required panels would be operating. The non-operating panels with no flow would be allowed to freeze or reach a condition in which the coolant would not flow freely. These panels would not be operative until they are unfrozen or the fluid reaches an operating temperature by the environmental heat or a combination of electric heaters and environmental heat. This means of varying the heat rejection area has been used successfully on the Apollo spacecraft and generally referred to as selective stagnation.

The second approach and perhaps the simplest would be to provide an insulated, sliding cover over the radiator surface to achieve the variable area. This would not only eliminate any freezing, but would result in nearly a constant radiator temperature regardless of the environmental condition and/or the heat rejection load. The required radiator area would be established for the maximum heat rejection load and at the sub-solar condition. Thus, for power-down condition or reduced heat load and for the lunar night condition, the radiator area would be reduced to maintain the desired or design outlet temperature for the coolant. For example, if the radiator is designed for an inlet temperature of 80 F and an outlet temperature of 40 F, the reduction in radiator area for the lunar night condition for variation in the heat rejection load is illustrated in Figure B-7 for three different ratios of α_s/ϵ. This shows that the area must be reduced significantly with reduction in heat load in order to maintain the desired outlet condition.

An additional advantage of a variable area radiator is the possibility of simple controls for the coolant circuit or loop since the coolant outlet temperature from the radiator can be maintained at or near the design value for all operating conditions.

The radiator surface area can be readily estimated for various heat rejection loads and radiator average temperature from Figures B-8 and B-9 at the sub-solar condition. Figure B-8 is based on a radiator surface with an optical solar reflector and an assumed radiator effectiveness, $\varphi = 0.90$, and Figure B-8 is based on Z-93 coating and an assumed radiator effectiveness, $\varphi = 0.90$.

As an example, in establishing the radiator area, consider a crew compartment or shelter designed for a four-man crew and a radiator with an inlet of 80 F and an outlet of 40 F to give an average radiator temperature of 60 F with the following assumed heat rejection load:

1. Electronic and scientific equipment = 15,000 Btu/hr
2. Cabin or shelter environment = 2000 Btu/hr; and
3. Humidity control and water chiller = 1000 Btu/hr

for a total heat rejection load of 18,000 Btu/hr. If the radiator is designed with an optical solar reflector surface, then from Figure B-8 the area would be about 260 square feet. If Z-93 is used for the radiator surface, the radiator area would be about 825 square feet, from Figure B-9. This example shows the advantage of using the optical solar reflector for the radiator surface.

Active Thermal Control Concepts

Various active thermal control concepts are discussed which include the simple semi-passive concept to the more complex which incorporates both semi-passive and active methods for thermal control. These concepts cover approaches to achieve minimum size radiator as well as means to compensate for the wide variation in the thermal environment and the heat rejection loads. Further, these concepts take into consideration the practical limitations of the coolant (or working fluid) and the surface coating or finish of the radiator.

Figure B-10 gives the classification guide for the various concepts that are illustrated schematically in Figures B-11 through B-15. Concept classification has been arranged on the basis of fixed area radiator with single and multiple coolant loops and variable area radiators. Under these two general categories, the concepts are further classified as separate thermal control, partially, and fully integrated control. The separate thermal concept permits designing the system to meet the specific requirements of the equipment or process. For the lunar base, this concept would mean three separate systems, one for the electronic equipment, one for the cabin heat exchanger, and one for the humidity control and water chiller. This is based on the allowable or design temperature range and the amount of heat rejection load, which was discussed earlier. Instead of separate thermal control systems, the alternative would be one system or an integrated system to provide the necessary thermal control for all the equipment and the personnel areas.

The simplest thermal control concept is the one with a single coolant loop in which the coolant flows between the radiator and the component which requires temperature control. Concepts A, B, and C of Figures B-11 and B-12 illustrate this simple concept for the separate through the fully integrated thermal control. This concept would be feasible if a radiator surface finish or coating with low absorptivity value and high emissivity value, such as the optical solar reflector, and a coolant which has low viscosity (or pumping power requirement) at low temperatures, such as hydrogen gas, were available and suitable for the lunar surface base and the sortie convoy. Previous discussion in the section on space radiators has pointed out the possible low temperatures during the lunar night condition, particularly for the low heat rejection loads.

If it is considered objectionable to having hydrogen gas as a coolant inside the personnel shelters, it would be possible to use two coolant loops. One would be located inside the personnel shelters and the coolant selected would be safe or would be non-hazardous to the personnel. A number of common types of coolants could be used. This loop would be thermally connected to a second loop by a heat exchanger. The second loop, containing the hydrogen gas as the coolant, would be outside the personnel areas and would contain the radiator for heat rejection to space. This two-loop concept is illustrated by Concepts D and E of Figure B-12 and B-13, respectively. The two loops increase the number of components and can be considered to be more complex than the single loop, but still this approach is relatively simple.

The concepts discussed up to this point require a coolant that remains highly fluid; that is, it must have low viscosity at temperatures in the -200 F range. This requirement limits the possible choice of coolants to gaseous hydrogen. It is possible to broaden the selection or choice of coolants by the incorporation of an active thermal control method such as the vapor compression refrigeration system or by utilizing waste heat from another system or process. The use of the vapor compression refrigeration system makes it possible for high coolant inlet and outlet temperature from the radiator, that is, a high temperature radiator, so that the radiator temperature during the lunar night period would remain in a range which is suitable for the commonly available coolants. For example, if Freon 21 is selected as the coolant, the minimum temperature would be limited to about -30 to -40 F, as indicated in Figure B-3. Thus, for a radiator designed at 60 F, the heat load reduction at lunar night condition would be limited to about 50% of the design load as indicated in Figure B-2 for a radiator with an optical solar reflector surface. If a vapor compression refrigeration unit is incorporated into the system so that the radiator is designed to operate at 120 F during lunar day, the heat load reduction at lunar night could be lowered to about 30% of the design load. In essence, this represents a dual system in which the semi-passive coolant loop would be operational during the lunar night period and the vapor compression refrigeration system would be operational during the lunar day period.

Lower heat load limits, that is less than 30%, during the lunar night period, could be achieved by higher radiator design temperatures; however, a practical upper limit would be reached as the power required for the refrigeration cycle increases rapidly for higher operating temperatures. This is illustrated in Figure B-16 for a number of common refrigerants and based on an assumed heat rejection load of 20,000 Btu/hr and an evaporator temperature of 40 F.

Concept F of Figure B-13 illustrates one possible way to incorporate a vapor compression refrigeration system into a basically semi-passive coolant loop. An additional advantage in employing the vapor compression refrigeration system is that it will result in a minimum area radiator because the radiator would be designed to operate at high temperature.

A possible simpler approach would be to utilize waste heat from another system, such as the isotopic power system. If this system were designed for heat rejection in the 300 F range, and a waste heat load of 5 to 10 kw, it would be more than adequate to supply the necessary heat to the thermal control system to keep the radiator temperature at the design or a suitable temperature during the lunar night and low heat rejection load conditions. The use of the waste heat will require the addition of a suitable heat transport loop and also require the waste heat source to be reasonably close to the thermal control system. This approach or concept could be utilized for the thermal control associated with the sortie convoy since the isotopic power system could be in close proximity to the personnel shelter and the motorized cabs. This concept would not be feasible for the lunar base since the power system is to be located too far from the personnel and equipment shelters or modules. Concept G of Figure B-14 illustrates a system which incorporates the means to utilize the waste heat.

Another approach that merits serious consideration is the variable area radiator which would be designed to vary the radiating surface area to compensate for the change in the thermal environment and/or the heat rejection load. As mentioned earlier, there are various possible ways to design a variable area radiator. The one that appears to be the simplest is the insulated, sliding door or cover (Concepts H and I) which would be actuated automatically to maintain the required radiator area. A simple, automatic control system with a sensor located in the coolant outlet from the radiator could be used to achieve a nearly constant coolant outlet for all varying conditions. This approach not only allows a wider selection of coolants, but it also permits a wider selection for the surface coating. Further, this approach could be used to compensate for degradation of the surface coating.

The variable area radiator concept merits serious consideration for the lunar surface base complex since there appears to be no serious limitations as to allowable or available area for the radiator. However, for the sortie convoy, there would be area limitations as well as weight limitations that make this method unfeasible.

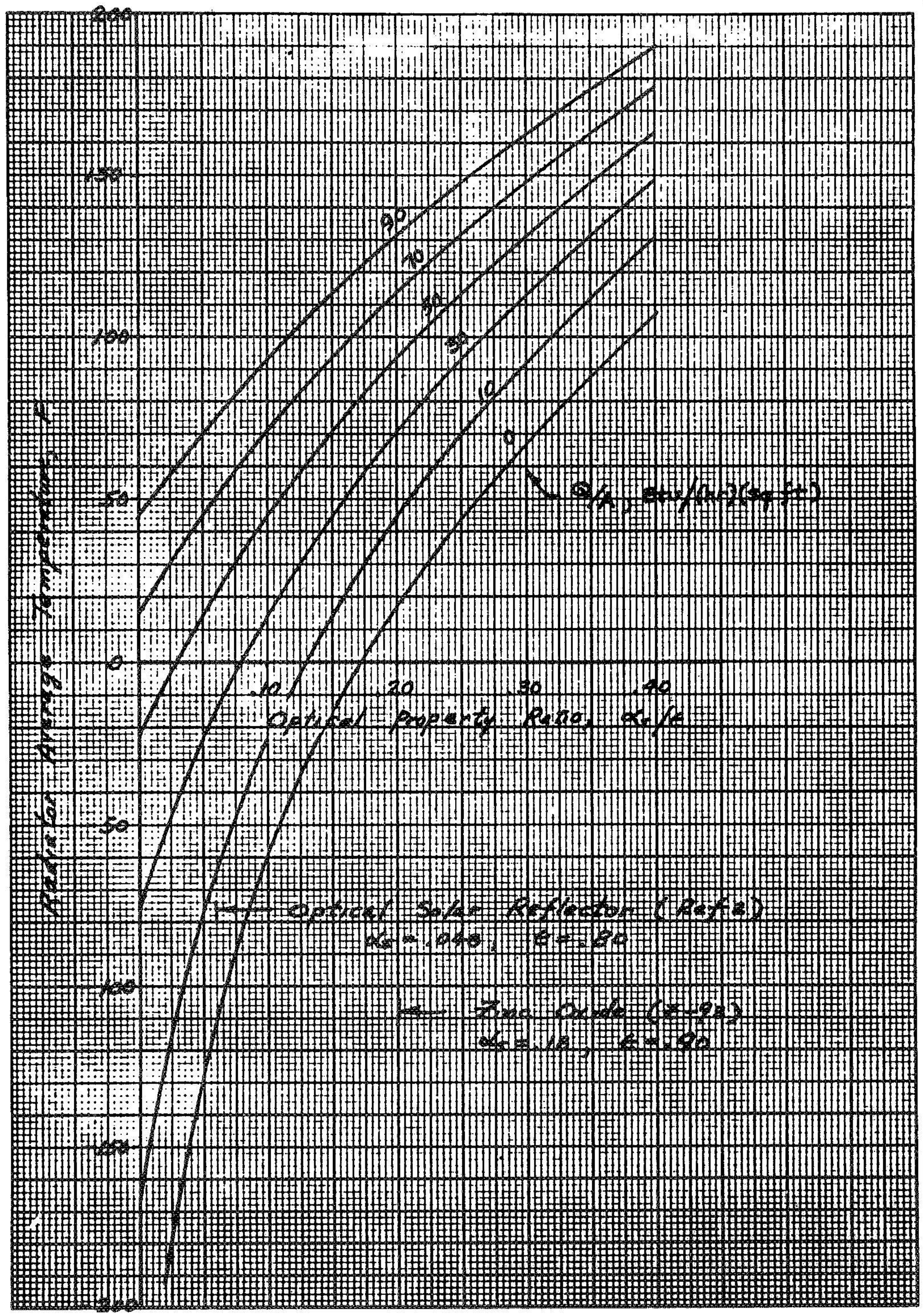


Figure B-1. Radiator Heat Rejection Rate per Unit Area at Subsolar Point

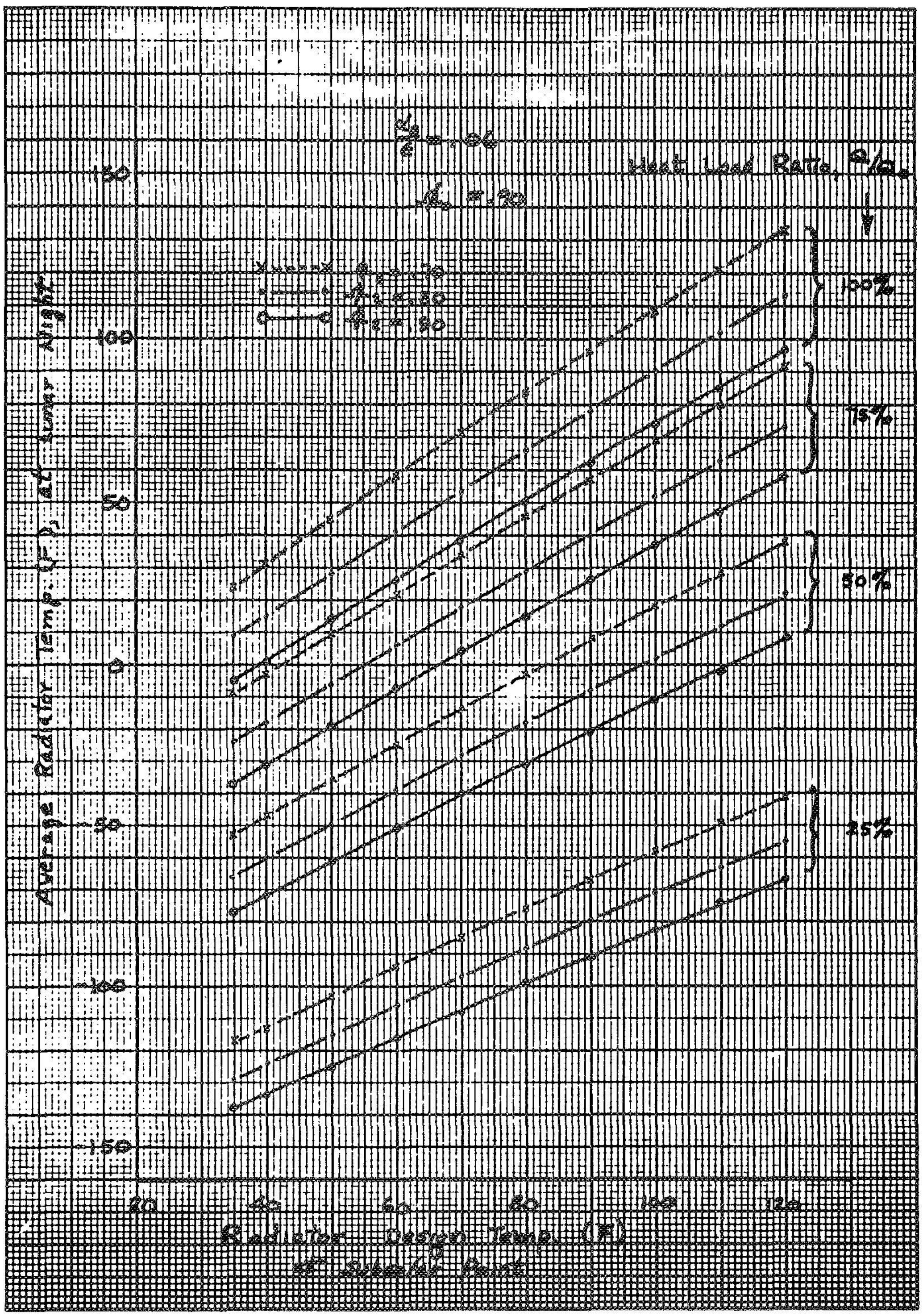


Figure B-2. Average Radiator Temperature at Lunar Night Condition for Optical Solar Reflector Surface

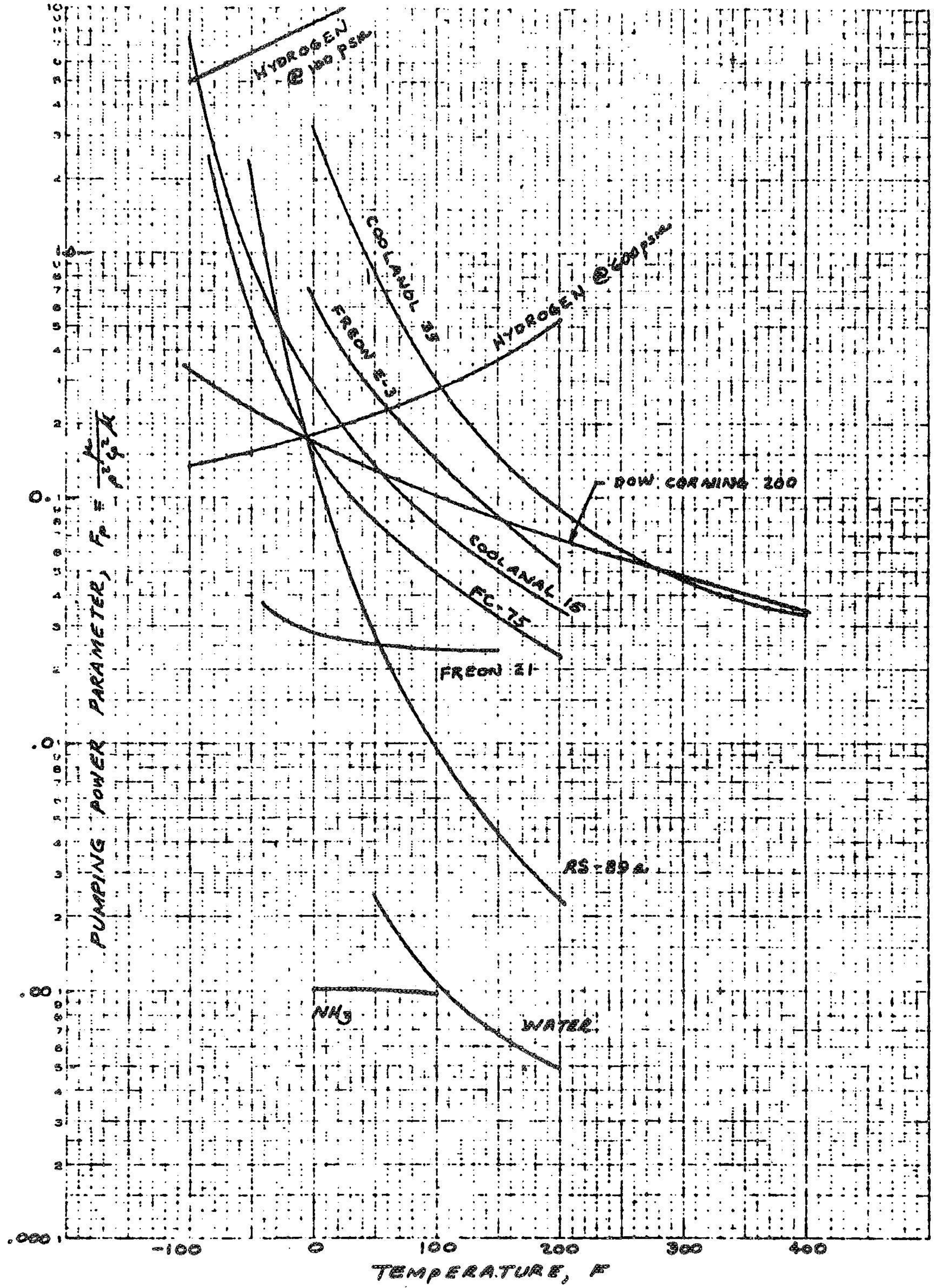


Figure B-3. Pumping Power Parameter vs. Temperature for Coolant

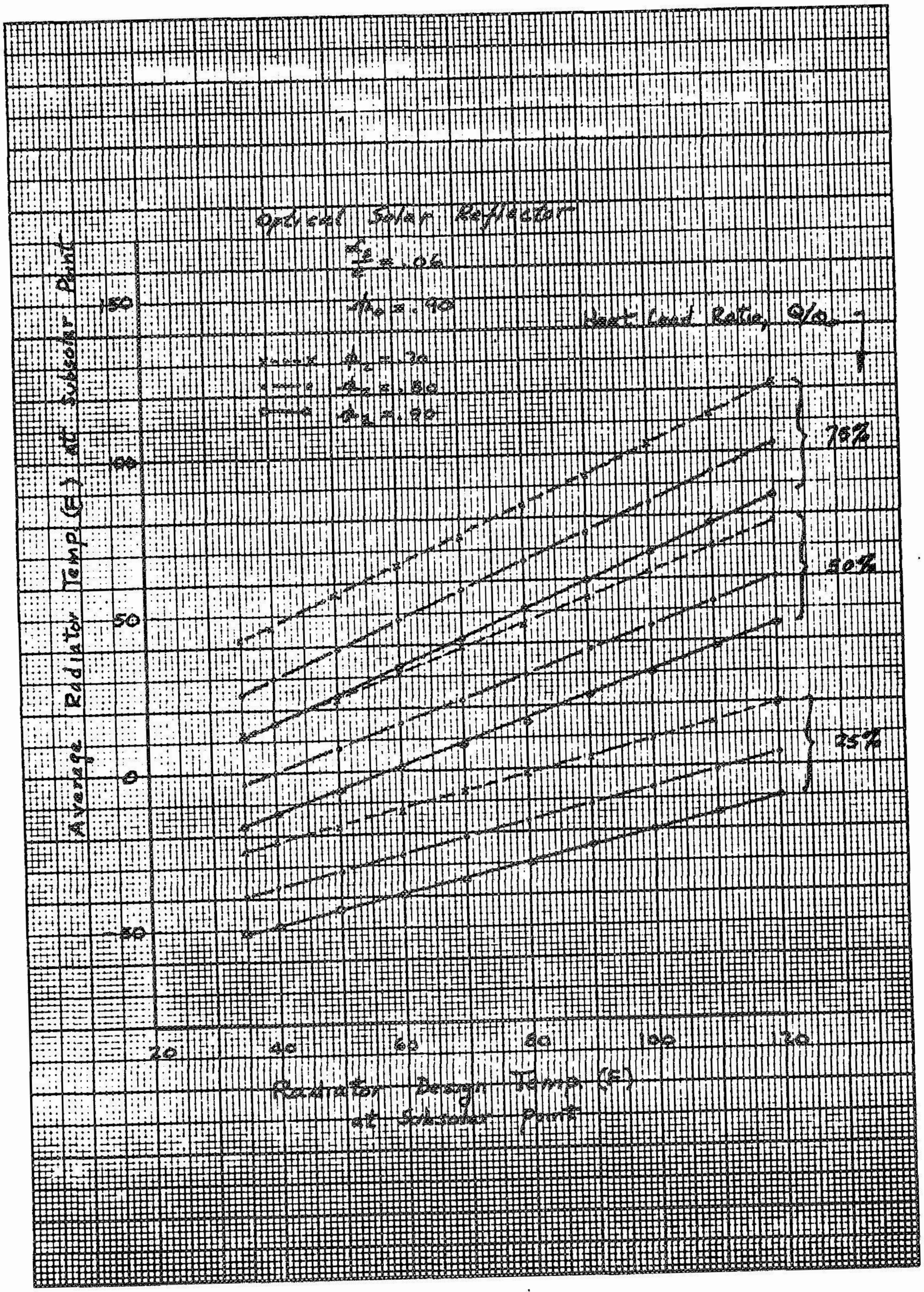


Figure B-4. Average Radiator Temperature for Powered Down Condition at Subsolar Point

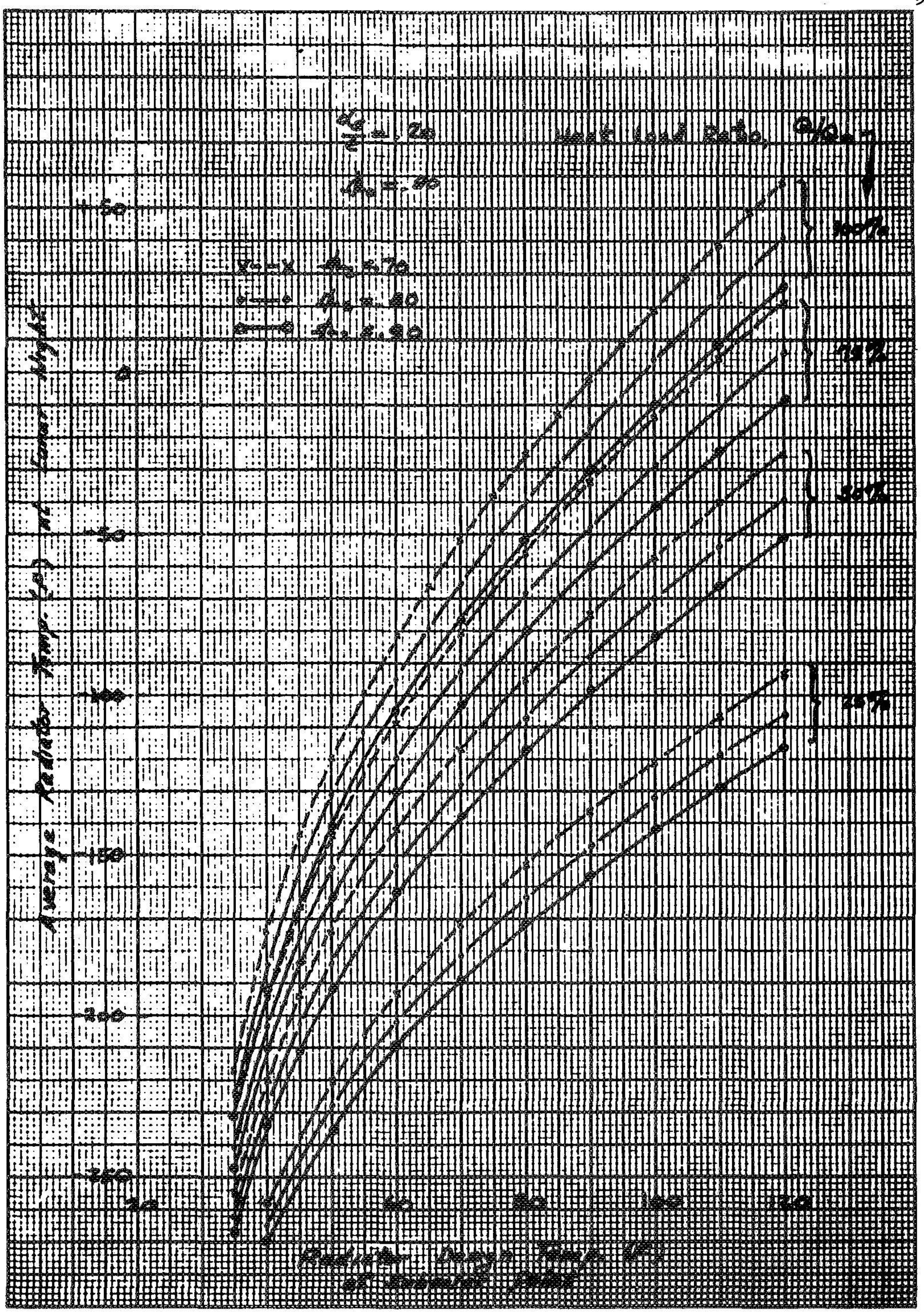


Figure B-5. Average Radiator Temperature at Lunar Night Condition for Z-93 Surface Coating

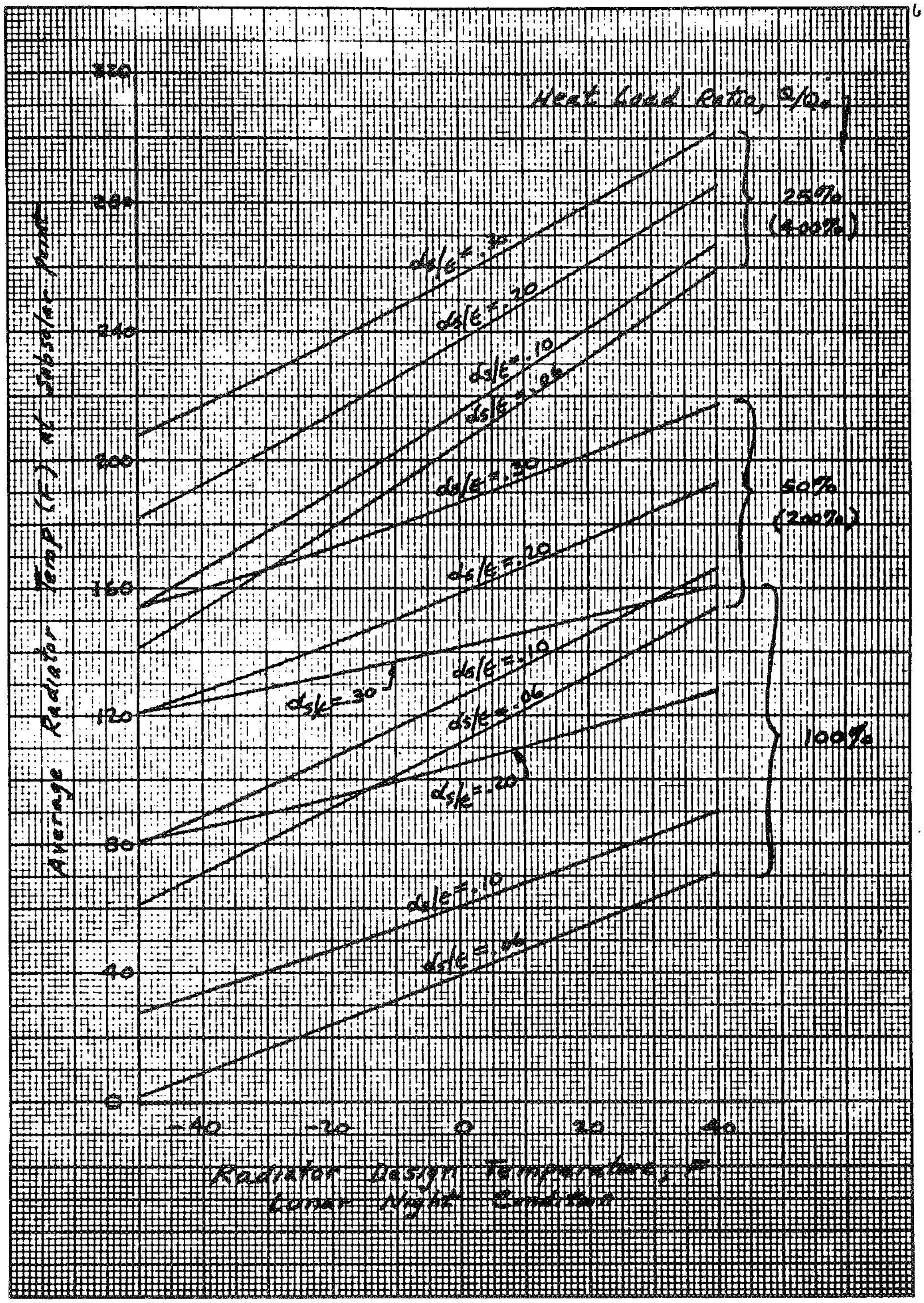


Figure B-6. Average Radiator Temperature at Subsolar Point

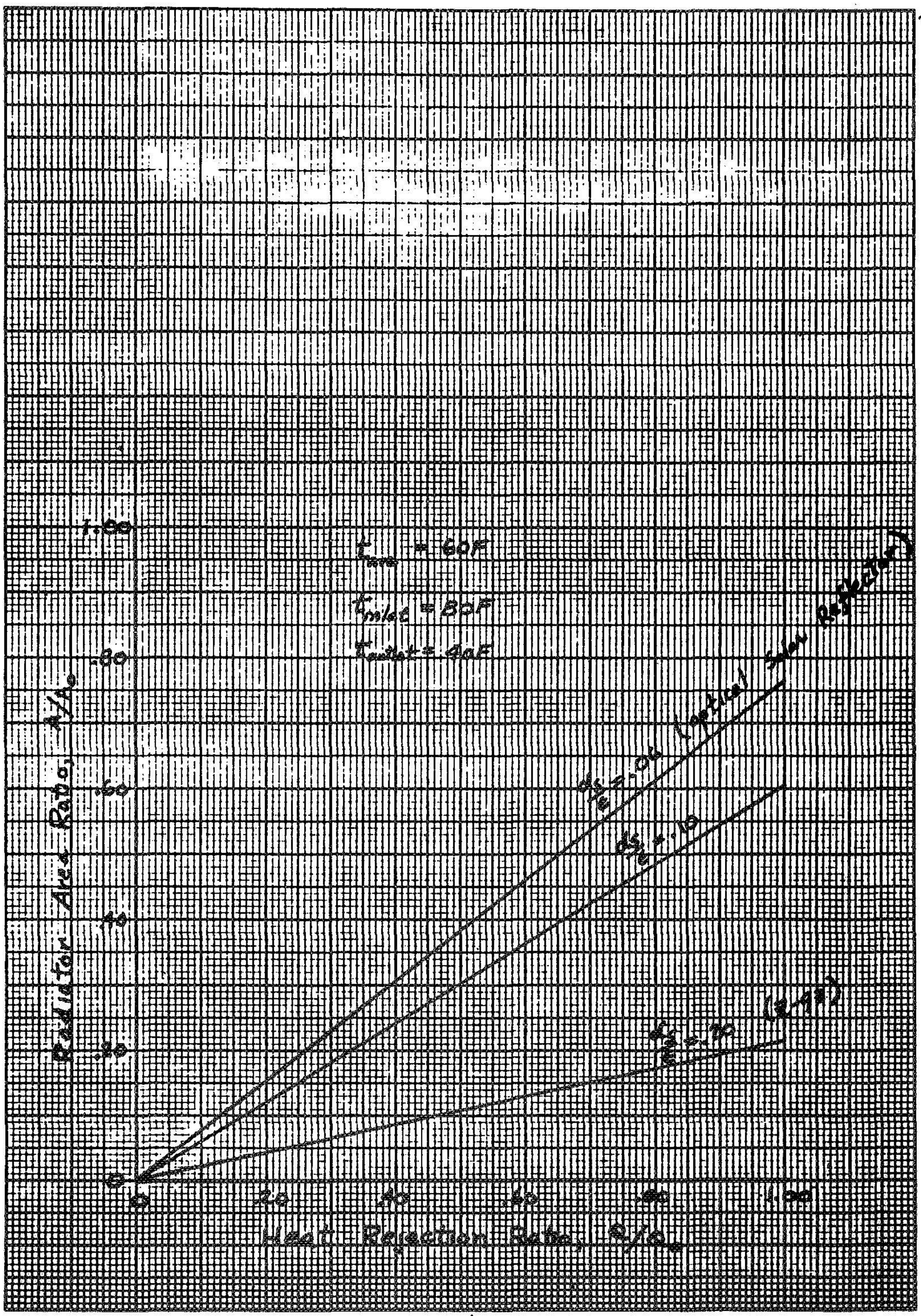


Figure B-7. Radiator Area Ratio vs. Heat Rejection Rates at Lunar Night Condition

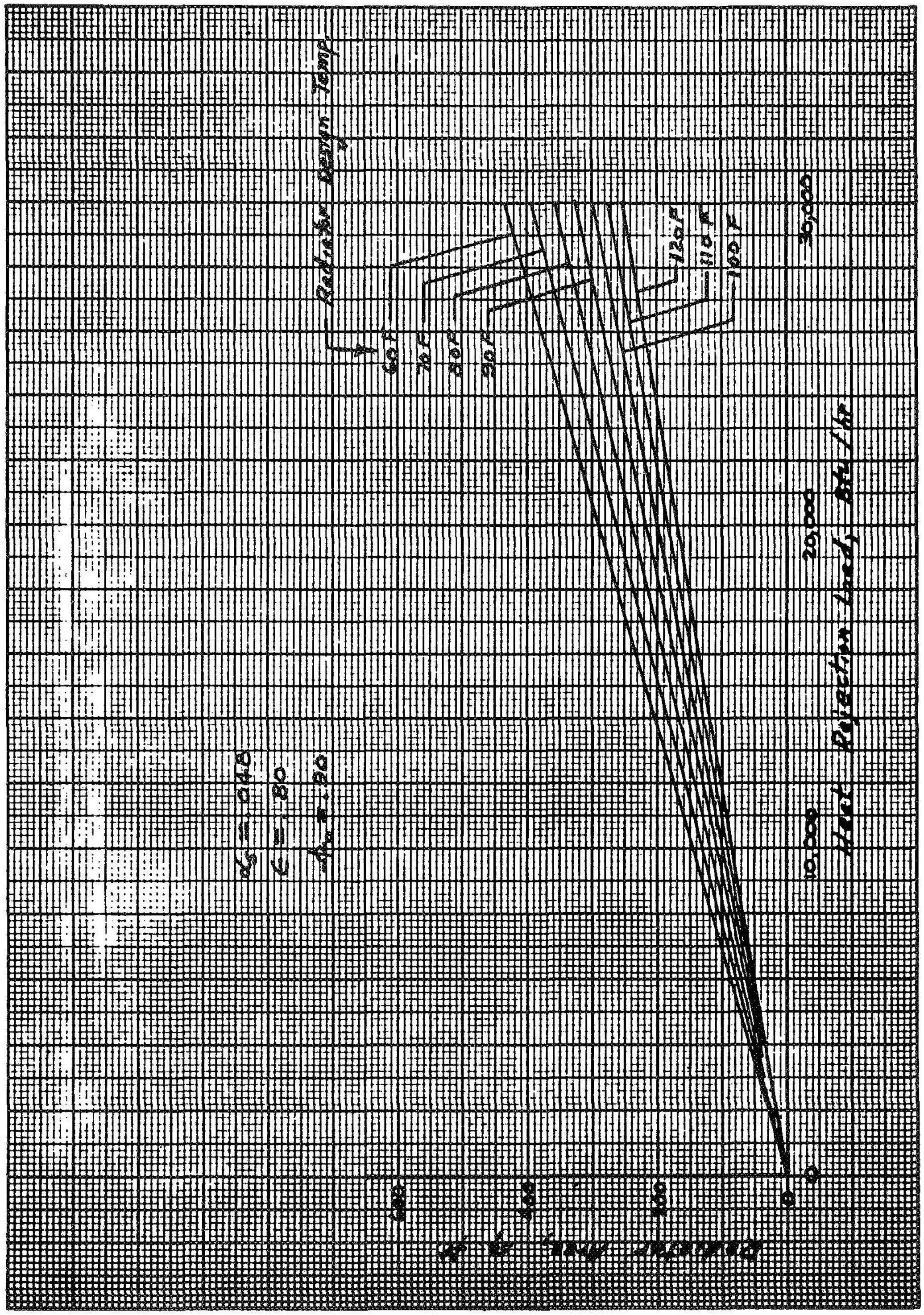


Figure B-8. Area Requirement for Radiator with Optical Solar Reflector Surface

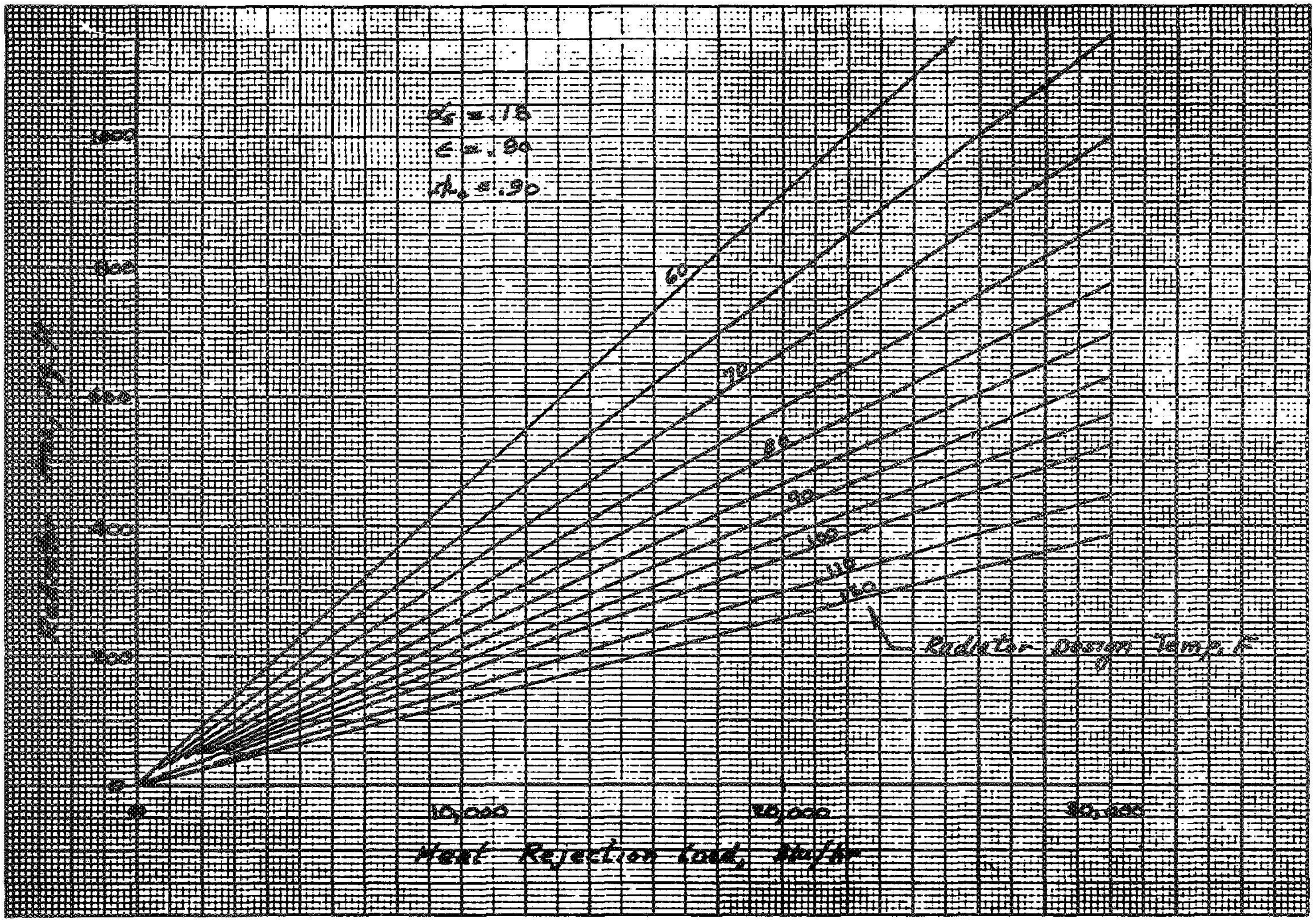
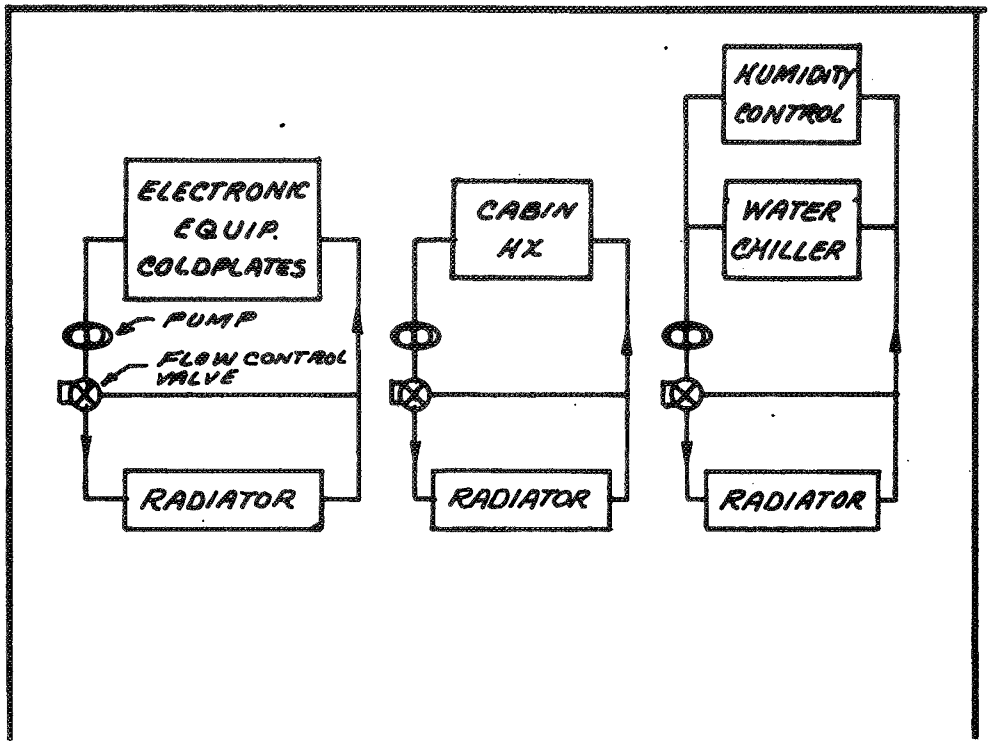


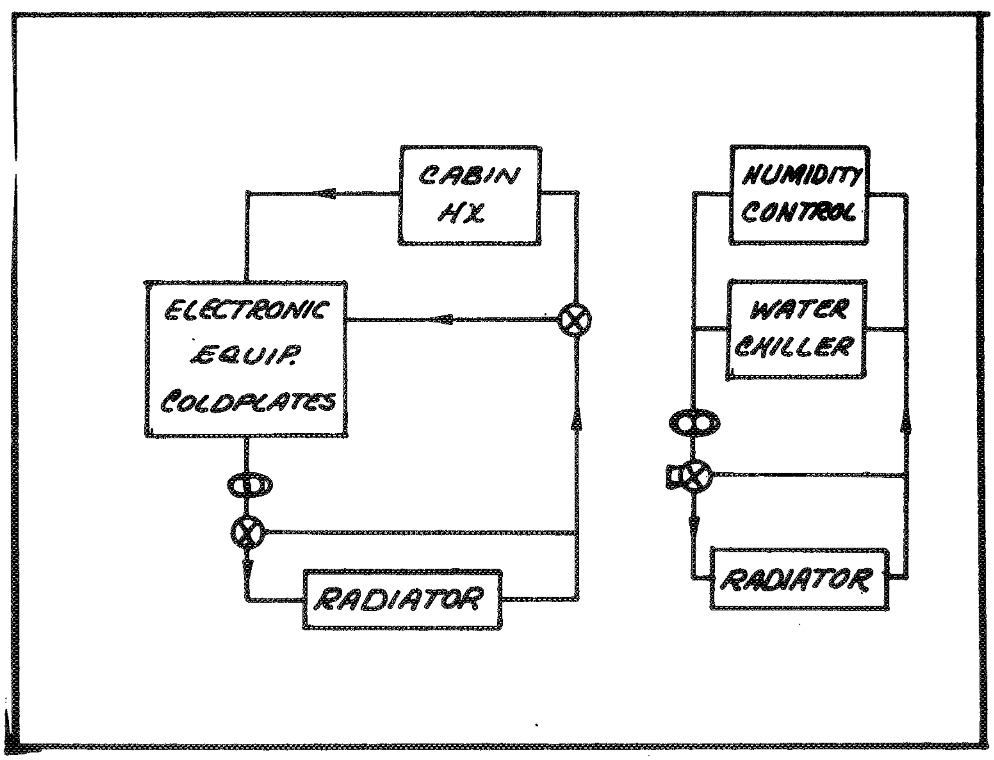
Figure B-9. Area Requirement for Radiator with Z-93 Surface Coating

	Fixed Area Radiator			Variable Area Radiator
	Single Coolant Loop	Multiple Coolant (Working Fluid) Loops		
Separate Thermal Control	Concept A (Fig, B-11)	Concept D (Fig. B-12)		Concept H (Fig. B-14)
Partially Integrated Thermal Control	Concept B (Fig. B-11)			
Totally Integrated Thermal Control	Concept C (Fig. B-12)	Concept E (Fig. B-13)	Concept G (Fig. B-14)	Concept I (Fig. B-15)
		Concept F (Fig. B-13)		

Figure B-10. Active Thermal Control Concept Classification Guide

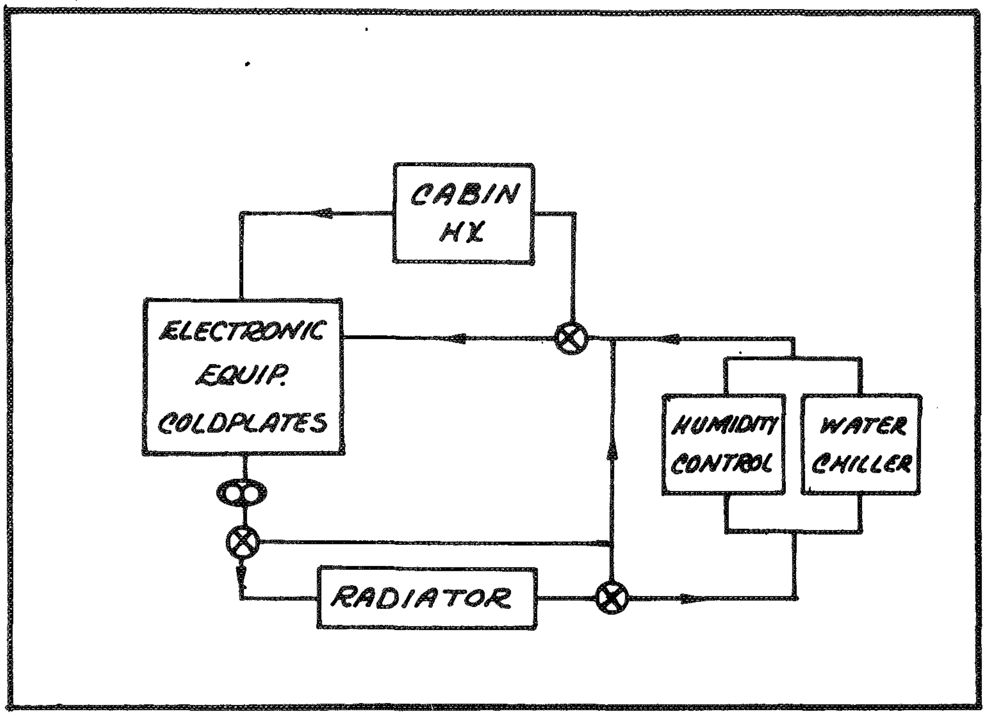


Concept A

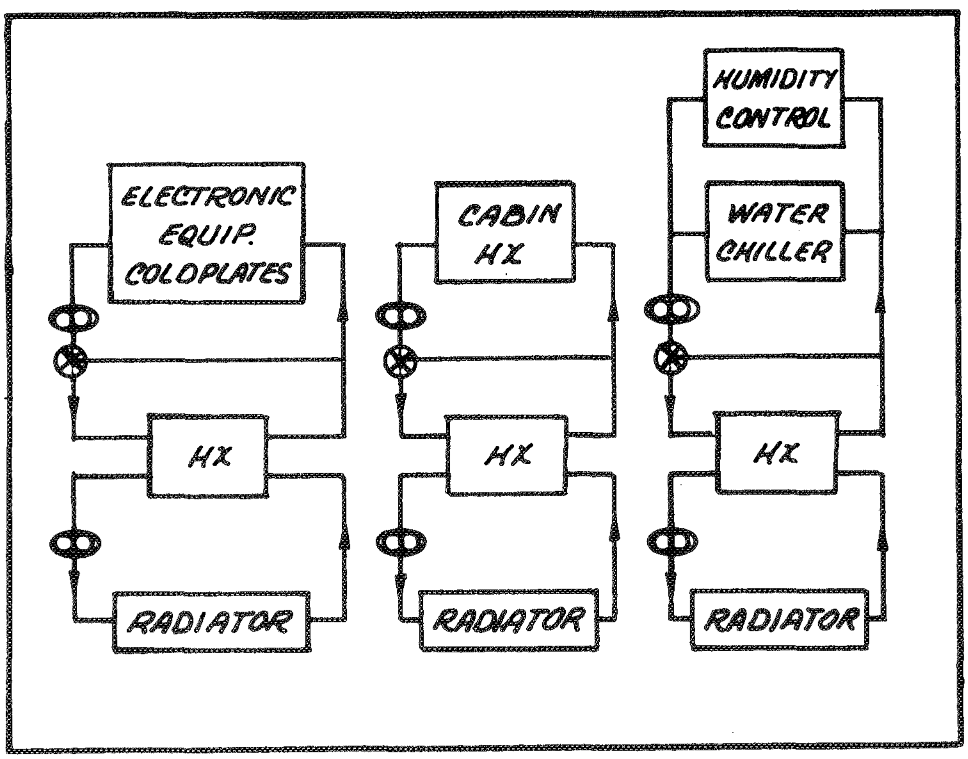


Concept B

Figure B-11. Active Thermal Control Concepts A and B

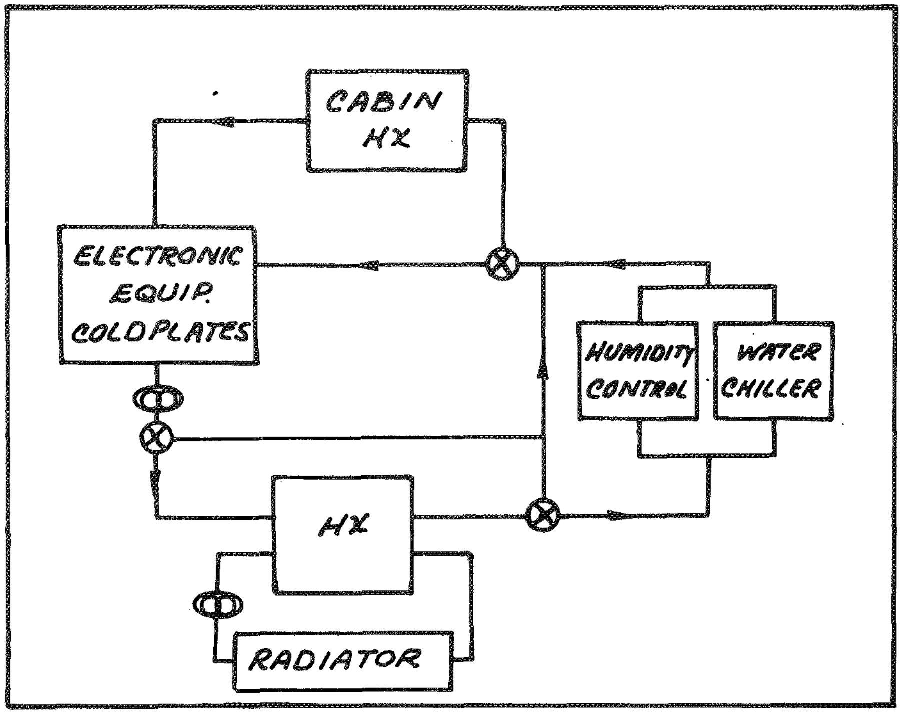


Concept C

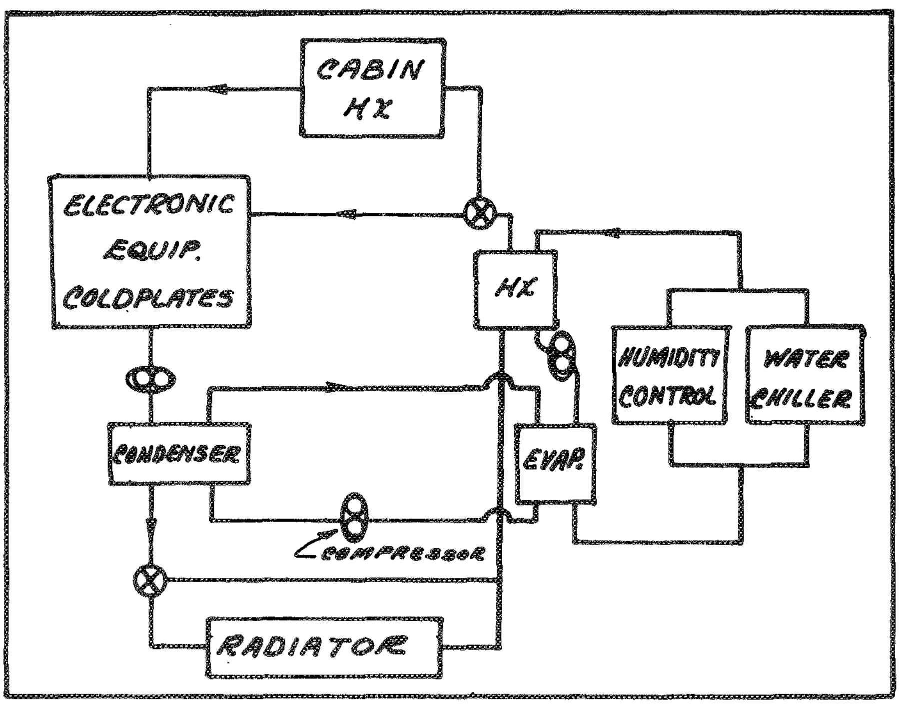


Concept D

Figure B-12. Active Thermal Control Concepts C and D

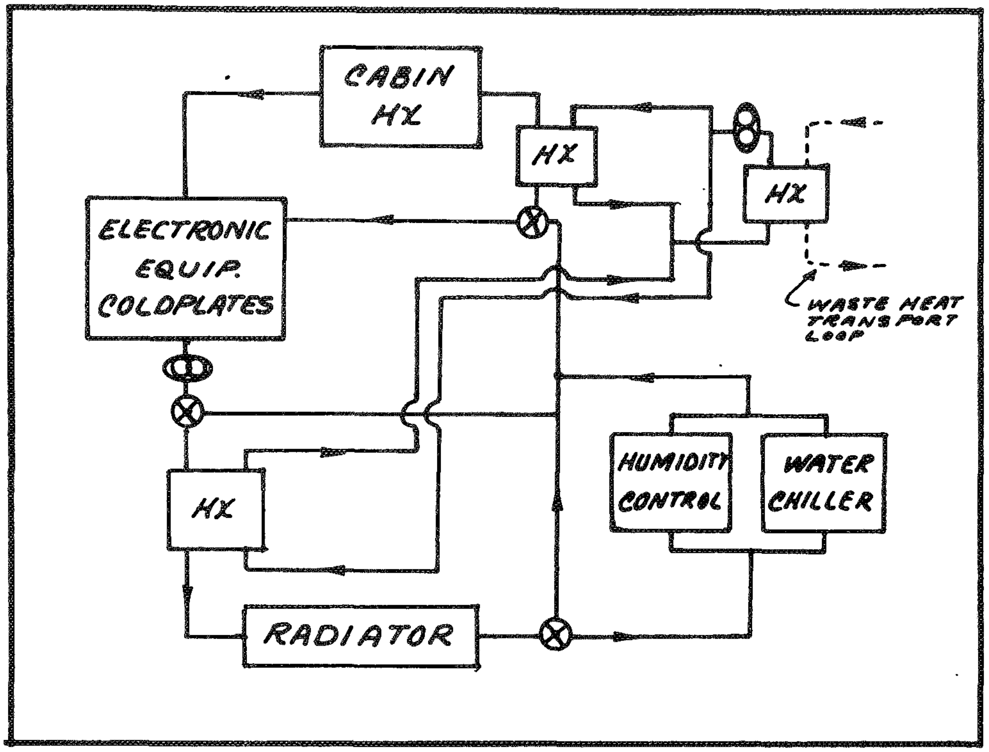


Concept E

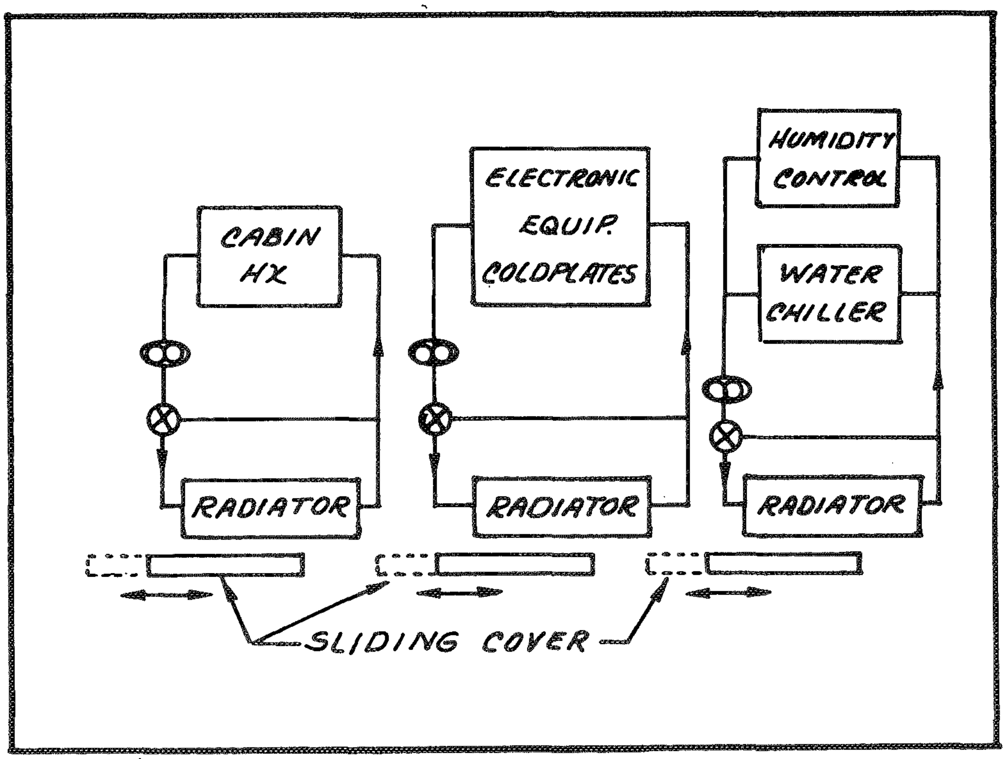


Concept F

Figure B-13. Active Thermal Control Concepts E and F

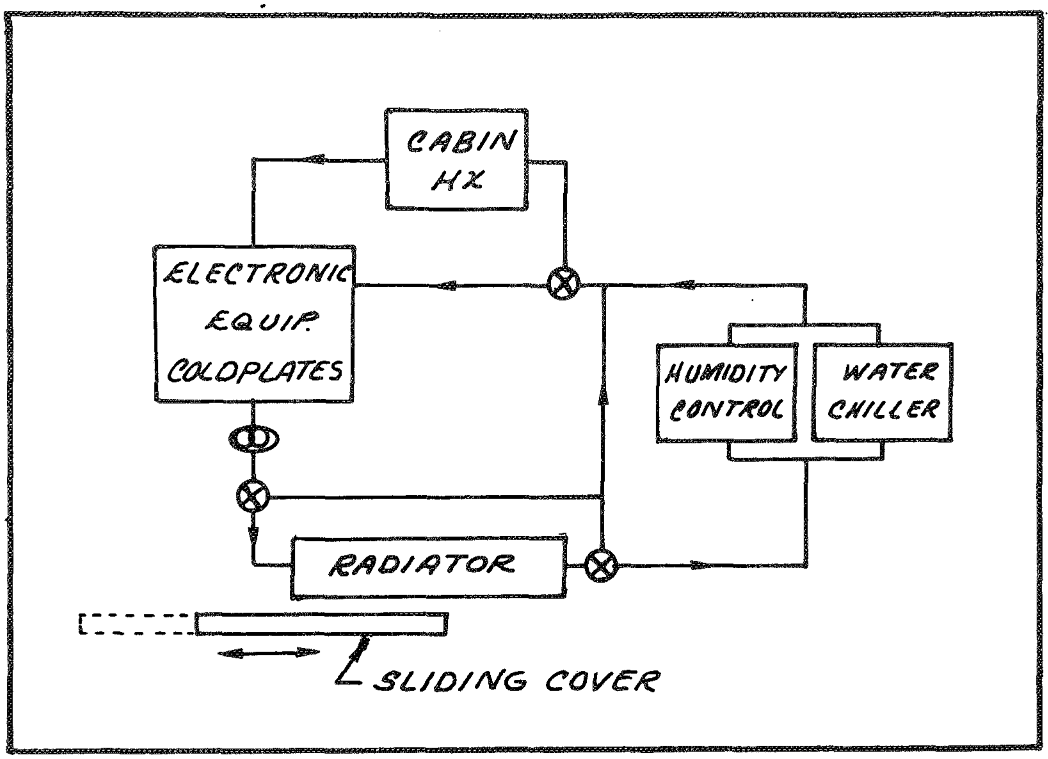


Concept G



Concept H

Figure B-14. Active Thermal Control Concepts G and H



Concept I

Figure B-15. Active Thermal Control Concept I

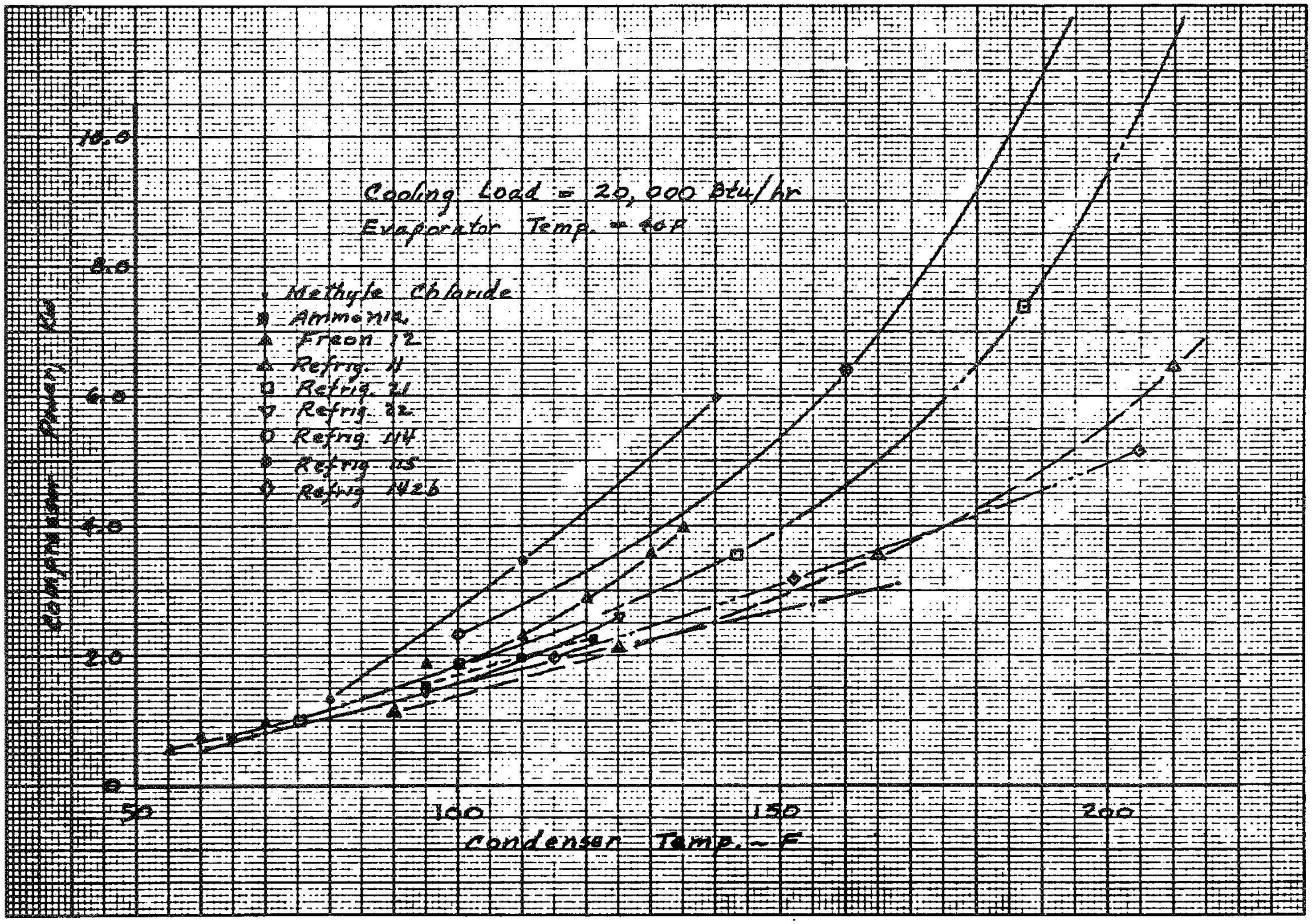


Figure B-16. Power Required for Vapor Compression Refrigeration Cycle

REFERENCES

B-1. Lunar Surface Mobility Systems Comparison and Evolution (MOBEV), BSR 1426, Volume II, Book 3, November 1966.

B-2. Marshall, K. N. and R. A. Breuch, "Optical Solar Reflector: A Highly Stable, Low Spacecraft Thermal Control Surface, J. of Spacecraft and Rockets, Vol. 5, No. 9, September 1968, pp. 1051 to 1056.

B-3. Solar-Powered Space Station Thermal Concept Formulation, SD 70-535 (in preparation).

APPENDIX C. SPECIFICATION FOR A MOBILE ELECTRICAL POWER SYSTEM FOR A LUNAR SURFACE BASE

1. SCOPE

This specification sets forth the general requirements for an Electrical Power System (EPS) for the generation, conditioning, and distribution of prime mobility power for a Lunar Surface Mobility Vehicle (LSMV); and for the support of certain EVA activities during field exploration sorties.

2. APPLICABLE DOCUMENTS

Documents which are relevant to this specification shall be delineated in order of precedence at a later date.

3. REQUIREMENTS

3.1 Design and Construction - The EPS shall comprise the following six elements:

(1) A radioisotope heat source

(2) A heat-to electrical energy conversion device

(3) A waste heat rejection system

(4) A power conditioning device(s)

(5) A battery system for the storage of electrical energy

(6) A power distribution system.

A major consideration in the design and construction of the EPS shall be that the radioisotope heat source shall be readily installable in the primary energy portion of the EPS of either the LSB or the LSMV. Therefore commonality shall exist between the power systems of the LSB and LSMV to the extent that interchangeability of heat source modules is facilitated, and that individual modules may be employed in either application. With the exception of the batteries, the remaining elements of the LSMV EPS shall be an integral part of the vehicle.

3.2 Performance

3.2.1 Radioisotope Heat Source - The radioisotope fuel shall be Polonium-210, having a BOL specific thermal power of 100 watts/gram. The aggregate source shall be sized to provide 25 kwt at the end of one half-life. This requirement ensures the delivery of 3.5 kwe from a 15% efficient EPS throughout a 138-day utilization period. Power-flattening from BOL to the end of the first half-life period shall be a requirement of the heat rejection system.

3.2.1.1 Heat Source Modularization - The radioisotope heat source shall be comprised of an aggregate of modules. A module shall be designed in accordance with the commonality and interchangeability requirements stated in 3.1. A module shall be designed in accordance with the further requirements stated below:

(1) The fuel material shall be reliably contained to permit safe handling by personnel, and to prevent accidental contamination of the environment

(2) The modules shall be designed to minimize radiation hazards during interchanging operations, and to accommodate the generation of helium gas within the fuel containers.

3.2.1.2 Shielding Requirements - The radiation fluence emanating from the heat source shall be attenuated to a level not to exceed 150 millirems per day at a radial distance of 10 feet from the source location in the LSMV. Full advantage shall be taken of LSMV geometry and construction, and supplementary shadow shielding may be used in regions of the fuel modules and hot-loop transfer regions as necessary to achieve this attenuation. Figure C-1 presents shadow shield weights for Pu-238, which are sufficiently close to the requirements for Po-210 to serve for preliminary supplementary shielding estimation purposes.

3.2.2 Energy Conversion - The conversion of radioisotope heat energy into electrical energy shall be accomplished by means of a turboelectric device. Consideration shall be given to both the organic-Rankine and the Brayton thermodynamic cycles. Whereas the Brayton cycle can be made to operate with a system efficiency in excess of 20%, its low heat rejection temperature necessitates a relatively large radiator area (roughly 230 sq. ft.). Inasmuch as power-flattening must also be performed by the cooling system, the total radiator area and weight may be excessive for mobile application. The organic-Rankine cycle is less efficient (12 - 18%), but requires approximately 20% less radiator area (roughly 175 sq. ft.) than the Brayton cycle. Selection of the thermodynamic cycle shall be made following trade and interface studies relative to the identification of suitable heat rejection areas on, and near the LSMV structure.

3.2.3 Heat Rejection - A heat rejection system shall be provided to maintain the design cold junction temperature of the thermodynamic cycle selected for the energy conversion device. The system shall employ radiators, and auxiliary extended surfaces if required. Coolant loops with working fluids compatible with the external environment described in this specification, shall be used to transport waste heat from cold-junction points to the rejection areas. Maintenance

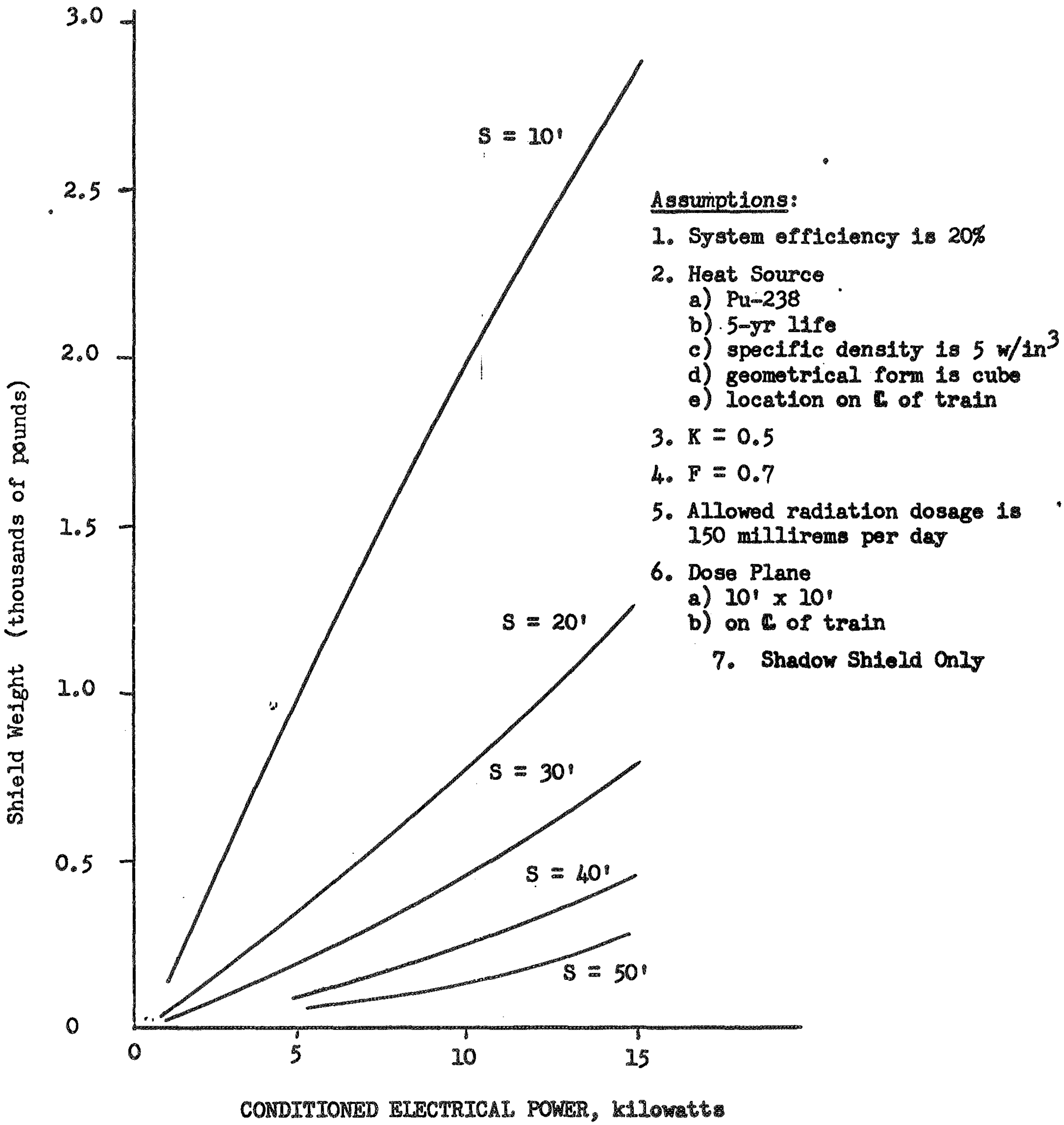


Figure C-1. Radioisotope Shield Weights

of the design cold-junction temperature implies that sufficient radiator capacity must be provided to also dissipate excess energy (power flattening) during early portions of the radioisotope half-life.

3.2.3.1 Coatings - Any necessary coatings shall be provided to protect the radiator surfaces from UV and micrometeoroid degradation, and to enhance heat rejection performance.

3.2.4 Power Conditioning - A power conditioning system shall be provided to perform the following functions for an average electrical load of 3.5 kilowatts:

(1) Provide voltage regulation for the turboalternator

(2) Convert the generator output to electrical energy of the forms and quality required by the LSMV subsystems

(3) To charge silver oxide cadmium batteries.

3.2.5 Battery System - Energy storage shall be performed by a silver-oxide-cadmium battery set having a 10,000 ampere-hour capacity at the one-hour rate. The batteries shall have a total volume of not more than 13 ft^3, and shall have a total weight of not more than 700 pounds.

3.2.6 Power Distribution - One a.c. buss and at least two d.c. busses shall be provided as initial points of power distribution. The number of busses required shall be a function of the nature of user loads and the degree of electromagnetic compatibility which must be achieved.

3.2.6.1 Motive Power Distribution - Power for the drive motors shall be distributed from an independent buss. All motive power distribution lines shall be capable of handling 100 amperes for 5-hours.

3.2.6.2 Distribution of Power to External Loads - The EPS shall be capable of transferring power to lunar peripheral equipment for the support of field activities. Additionally, the EPS shall be capable of providing emergency power to the LBS should the need arise. Accordingly, suitable receptacles for such external loads shall be provided as part of the distribution system.

3.3 Natural Environment

3.3.1 Thermal Environment - During the 354-hour lunar day, the LSMV will receive heat from direct sunlight, sunlight reflected from the lunar surface, and from IR energy emitted by the lunar surface. During the lunar night, the only source of external heat will be the surface IR radiation which declines to very low values just before sunrise. The resulting thermal environment shall be taken as that which is presented as Figure C-2.

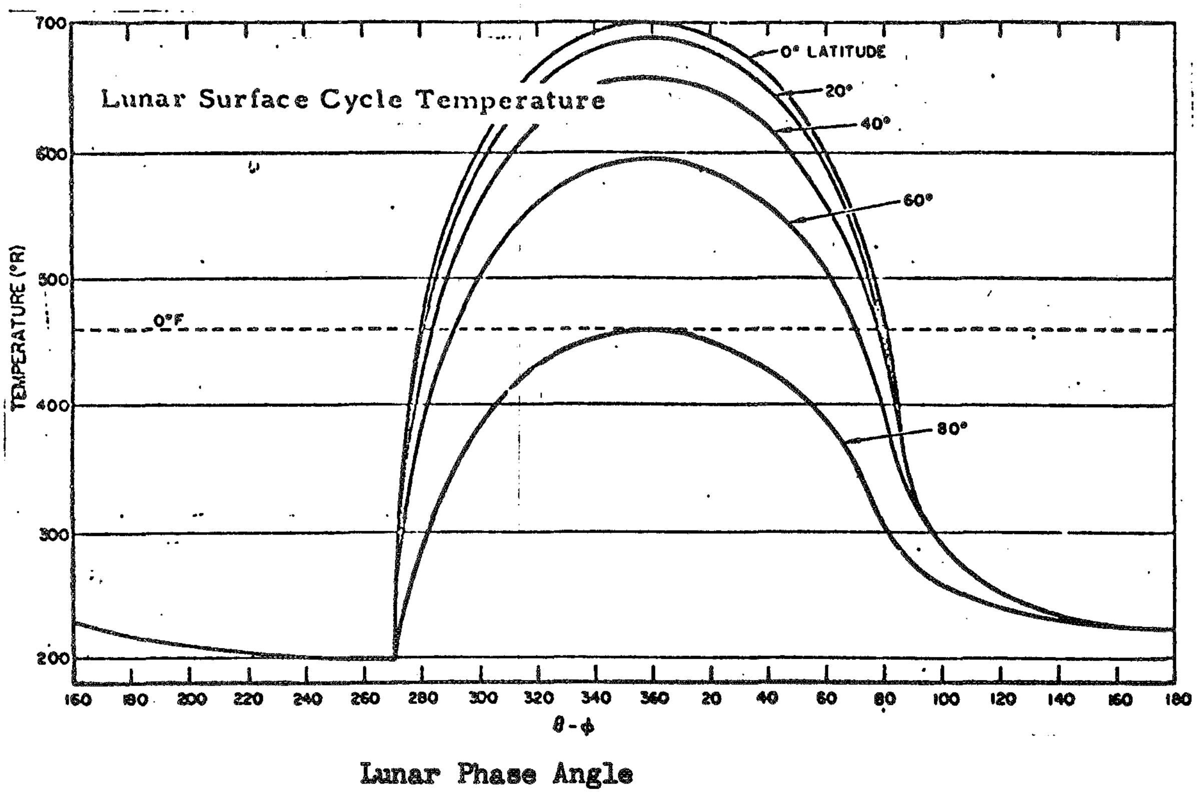


Figure C-2. Lunar External Thermal Environment

3.3.2 Natural Radiation Environment - The background natural radiation environment shall be considered to be principally the electromagnetic and particulate (solar wind) radiations from the sun. Radiation resulting from solar flare activity shall be superimposed on the background radiation environment to determine the total fluences.

3.3.3 Environmental Conditions Induced by Mobility

3.3.3.1 Dust - Apollo experience has shown that lunar dust may be readily deposited on surfaces as a result of activity on the lunar ground. Consideration shall be given to the performance of the EPS heat rejection system under probable dust deposition conditions.

3.3.3.2 Vibration - The EPS shall be designed to operate reliably under all conditions of road shock and vibration which are anticipated during operation of the LSMV.

3.4 Maintenance and Repair - The EPS shall be designed for ease of maintenance and repair. Provisions shall be made for carrying a survival kit of hardware spare parts on all sorties.

3.4.1 Resupply - All maintenance schedules, including radioisotope module exchange due to half-life depletion, shall be based on a resupply cycle of 180 days.

3.5 Service Life - The service life of the EPS (exclusive of radioisotope fuel modules) shall be from 2 to 5 years.

APPENDIX D. MSS/LSB FUNCTIONAL REQUIREMENTS COMPARISON

The NR Modular Space Station (MSS) was defined for use in earth orbit, whereas the LSB modules were defined to satisfy the LSB mission which involves different conditions. Use of MSS modules may appear attractive, but the practicality depends on the modification requirements. This analysis was performed to identify any major functional differences which could result in design differences.

Table D-1 presents a top level comparison which indicates one major MSS function is not required for LSB application.

Tables D-2 through D-7 provide a detail breakdown by subsystem, for all those that apply.

Table D-1. LSB-MSS Functional Requirements Analysis

Subsystem Required	Commonality	
	LSB	MSS
1. Atmospheric Management	X	X
2. Crew Services Management	X	X
3. Electrical Power and Distribution	X	X
4. Communications	X	X
5. Information Management	X	X
6. Structural	X	X
7. Position and Stability Control		X

Table D-2. Atmospheric Management, Part 1

SUBSYSTEM ITEM	LSB PERFORMANCE REQUIREMENT	MSS PERFORMANCE CAPABILITY
Atmospheric Storage		
O_2 Supply	2.20 lb/man-day metabolic storage for 20 days 4.46 lb/day leakage tbd pumpdown	1.84 lb/man-day metabolic 6.0 lb/day leakage
N_2 Supply	7.60 lb/day leakage storage tbd pumpdown for 20 days	24.0 lb/day leakage
Pumpdown	Pump to cabin - 2 A/L, 1-4 man, 1-2 man	Pump to cabin - External A/L
CO_2 Management		
Concentration	5.0 mm Hg nom; 7.6 min Hg max, 15.0 mm Hg for 2 hrs	3.0 mm Hg nom; 7.6 mm Hg max (7 days) 15.0 mm Hg energ (2 hrs)
Removal	2.57 lb/man-day	1.98 - 3.0 lb/man-day
Thermal Control		
Metabolic Heat	13,000 BTU/man-day	10,300 - 13,600 BTU/man-day (650 BTU/man-hr)
ECS Equipment Heat	47,400 BTU/hr	-
Environmental Heat	Negligible (soil insulation	Note: (Average heat load/module = 17,750 BTU/hr Total)
Radiator Heat Rejection	76,000 BTU/hr	
Temperature	65° to 75° F	65° to 75°

Table D-2. Atmospheric Management, Part 2

SUBSYSTEM ITEM	LSB PERFORMANCE REQUIREMENT	MCS PERFORMANCE CAPABILITY
Atmospheric Control		
Ventilation	40 ft/min nom. 15 to 100 ft/min range	40 ft/min nom. 15 to 100 ft/in range
Pressure		
Total	10 psia	14.7 psia nom (10 psia min)
O_2 Partial	3.5 psia nom; 3.7 psia max	3.1 psia nom. (3.5 psia max)
Humidity		
Absolute	8 to 12 mm Hg pp H_2O	8 to 12 mm Hg pp H_2O
Relative	30 to 50%	30 to 50%
Trace Contaminants	Aerospace Standards	Aerospace Standards
Leakage	12.24 lb/day	30 lb/day

Table D-3. Crew Services Management, Part 1

SUBSYSTEM ITEM	LSB PERFORMANCE REQUIREMENT	MCS PERFORMANCE CAPABILITY
Water Management		
Potable		
Req'd - Food Prep	5.59 lb/man-day	1.66 lb/man-day
Drink		4.14 lb/man-day
Avail. - Food	1.04 lb/man-day	.64 lb/man-day
Total Intake	6.63 lb/man-day	6.44 lb/man-day
Personal Hygiene		
Washing	4.0 lb/man-day	4.0 lb/man-day
Shower	16.6 lb/man-day	16.6 lb/man-day
Flushing	3.5 lb/man-day (recycled)	3.5 lb/man-day
Housekeeping		
Laundry	None	None
Dishwasher	3.0 lb/man-day	2.7 lb/man-day (inc. leak)
Temperature Control	Hot - 160° F Cold - Room Temp.	Hot $155^{\circ} \pm 5^{\circ}F$ Cold $50^{\circ} \pm 5^{\circ}F$
Experiments & Lab	4.6 lb/day	18 - 35 lb/day (recycled)

Table D-3. Crew Services Management, Part 2

SUBSYSTEM ITEM	LSB PERFORMANCE REQUIREMENT	MSS PERFORMANCE CAPABILITY
Food Management		
Supply	Dry - 1.04; Frozen - .64 = 1.68 lb/man-day	1.68 lb/man-day
Storage	180 days initial	120 days nom + 30 days cont.
Preparation	Primary & Backup galleys - microwave ovens, hot plates, refrig, freezers	Primary & Backup galleys
Servicing & Cleanup	Water fountains, sinks, dishwasher, tables, benches	Same facilities
Inventory Control	Central processor status assessment	Central processor status assessment

Table D-3. Crew Services Management, Part 3

SUBSYSTEM ITEM	LSB PERFORMANCE REQUIREMENT	MCS PERFORMANCE CAPABILITY
Personnel Health, Hygiene, Habitability		
Medical/Dental		
Supplies	.098 lb/man-day	.139 lb/man-day
Facilities	Medical Lab & Treatment	Medical Labs
Clothing	Disposable - .13 lb/man-day	0.47 lb/man-day
Personal Equip.	.20 lb/man-day	.104 lb/man-day
Housekeeping & Hygienic Facilities	Complete Hygienic Facility/4 men	Numerous Hygienic Stations
Supplies	.25 lb/man-day	(Accounted for)
Facilities Areas		
Staterooms	(12) 480 ft^2 floor area	(12) 800 ft^2
Galleys	(2) 175 ft^2 (est)	210
Medical	(1) 70 ft^2 (est)	(1) 150
Command Center	(1) 70 ft^2	(4) 225
Labs	(5) 390 ft^2	(5) 755
Airlocks	(2) 170 ft^2	10 ft^2
Hygienic	(3) 60 ft^2	390
Gar. Warehouse	(2) 780 ft^2	-
Suit & Veh. Maint.	(1) 290 ft^2	-
Rec, Assembly, Dining	(2) 220 ft^2	400
Other	(Aisle) 415 ft^2	(Crew Serv) 20
Total	3120 ft^2 floor area	3130 ft^2 floor area

Table D-3. Crew Services Management, Part 4

SUBSYSTEM ITEM	LSB PERFORMANCE REQUIREMENT	MSS PERFORMANCE CAPABILITY
Waste Management		
Metabolic Wastes		
Urine	3.62 lb/man-day	3.45 lb/man-day 2.54 to 4.48 range
Feces	0.21 lb/man-day	0.38 lb/man-day
Trash		
Food Wastes	Negligibles	
Food Pack.	1.18 lb/man-day	15 lb/day (wet)
Miscellaneous	5 lb/day (est)	
Disposal	Collection, dry, sterilizing, compaction, storage	Same

Table D-3. Crew Services Management, Part 5

SUBSYSTEM ITEM	LSB PERFORMANCE REQUIREMENT	MSS PERFORMANCE CAPABILITY
Special Life Support IVA/EVA PLSS	Concept - A-PLSS - fully rechangeable CO_2 by Z_nO, H_2O recovered by $LiBr^2$	IVA - 8 mn hrs O_2 EVA - 1/month for 2 men
Emergency Supplies	20 days open loop supplies/man 60 days contingency over normal resupply	1 month contgency
Fire Control	Conflagration - Evacuation/pressure dump Small (Local) - O_2 supply by face masks, CO_2 extinguishers	Nuclei counter sensor with CO_2 extinguisher
Dust Management	EVA-IVA, Brush & air shower	N/A

Table D-4. Electrical Power System

SUBSYSTEM ITEM	LSB PERFORMANCE REQUIREMENT	MSS PERFORMANCE CAPABILITY
Power/Energy Requirements		
-ECLSS	13.5 kwe (12 men at base) avg.	11.4 kwe (12 men)
-Communication	1.7 kwe avg	1.8 kwe
-Lighting	1.4 kwe avg	2.0 kwe
-Logistics Veh. Support	1.1 - 5 kwe (reliquefaction)	
-Sortie Vehicles	3.6 kwe avg	N/A
-Science		
Deep drilling	3.4 kwe avg (non-concurrent)	N/A
Astronomy	4.05 kwe	
Exp. and Lab analysis	1.2 kwe	> 6.0 kwe
-Other		
-Power Loss (1%)	.3 kwe	
Energy Storage	ten, 10 k amp hr batteries (on vehicles) eight, 40 amp hr batteries (in modules)	at least 672 NiCd amp hr batteries
Power Distribution	dual bus	dual bus
Power Conditioning and Control	24-30v dc/203v ac - 400 cps	56 v dc/120-208v ac - 400 cps

Table D-5. Communications and Data Comparison Summary

ORIGIN / Terminal	Base		MSFN		Log. Vehicles		LFU		Sortie Veh.		EVA		Fixed Sites		(OLS)	
	LSB	MSS	LSB	MSS	LSB	MSS	LSB	MSS	LSB	MSS	LSB	MSS	LSB	MSS	LSB	MSS
Base	Internal Audio Video Data	X X X X	Voice Data Video	X X X	Voice Command Guidance (Radar)	X X X	Voice Command Guidance	X X X	Voice Data Command Guidance (LF) Video	aug aug aug aug aug	Voice Guidance (LF)	X X	Voice Commands	aug aug	Relay Voice Data Cmd. Video	X X X X
MSFN	Voice Data Cmds. Video	X X X X		N/A	Voice Command	N/A		N/A	Voice Command Data	N/A	Voice	N/A	Voice Command	N/A		N/A
Logistics Vehicle	Voice Data Beacon interrog.	X X X	Voice Data	N/A		N/A		N/A		N/A	Voice	N/A		N/A		N/A
LFU	Voice Data Beacon interrog.	X X aug		N/A		N/A		N/A	Voice Data Beacon interrog.	N/A	Voice	N/A		N/A		N/A
Sortie Vehicles	Voice Data Video	aug aug aug		N/A		N/A	Voice	N/A	Voice Comd. Data	N/A	Voice	N/A	Voice	N/A	Voice Data	N/A
EVA	Voice Biomed. Data	X X		N/A	Voice	N/A	Voice	N/A	Voice	N/A	Voice	N/A	Voice	N/A		N/A
Fixed Sites	Voice Data Video	aug aug aug	Voice Data Video	N/A		N/A		N/A	Voice	N/A	Voice	N/A		N/A	Voice Data Video	N/A
	Relay	aug	Relay	N/A	Relay	N/A		N/A	Voice Command	N/A		N/A	Relay	N/A		N/A

X = same capability, minor changes in antenna system

aug = augmentation required

Table D-6. Information Management

SUBSYSTEM ITEM	LSB PERFORMANCE REQUIREMENT	MCS PERFORMANCE CAPABILITY
Software	Supervisory programs	Same
	Application programs	
	Support programs	
	Data base	
Data Processing		
Experiments	Central Processor	Same
Laboratory Anal.	Local Processors	
Bio-Medical	Pre-processors	
Mobilescience Support	Data Bus	
	RACU	
	Central timing	
	Archival Memory	
Displays & Controls		
Systems Status	Command Console	Same
Systems Control	Remote Terminal Unit	
Maintenance Support	Hard Copy Printer	
Mission Status	Microfilm Projector	
	CMDR's Stateroom Console	
	Local M/A Displays	
	Local Clock Displays	

Table D-7. Structure, Part 1

SUBSYSTEM ITEM	LSB PERFORMANCE REQUIREMENT	MSS PERFORMANCE CAPABILITY
Structure Subsystems		
Bulkheads	Ellipsoidal-weight/volume efficient	Conical-transfer docking and orbit maintenance loads
Floor	Longitudinal deck - one habitable level	Longitudinal deck - main floor access from central corridor under floor
Aisles, Passageways	Center passageway	Beneath floor passage
Hatches	Separating pressurable volumes	Separating pressurable volumes
Docking	EOS-RNS-Tug transfer loads - docking at either ends and at one point on side	EOS-MSS to transfer loads - docking at either end
Ingress/Egress	Two airlocks; pumpdown system for drivein garage, warehouse; vehicle docking capability with base; side docking ports available for emergency egress	Externally attached airlock, normal transfer through docked vehicle.
Skin	Skin & stringer, lunar soil provides meteoroid & radiation protection.	Skin & stringer, meteoroid & radiation protection.

Table D-7. Structure, Part 2

SUBSYSTEM ITEM	LSB PERFORMANCE REQUIREMENT	MSS PERFORMANCE CAPABILITY
Module Configuration		
Overall Complex Layout	Circular floor plan; docking and airlocks at exposed ends for detail see Drawing 7B	Circular floor plan for crew, lab garage & warehouse modules; galleys & back-up CM are single entry - see drawing
Individual Module Comparison		
Garage Module (WBS-07)	One module devoted to IVA repair and maintenance of vehicles has pumpdown capability for Ingress/Egress or vehicles may dock for IVA transfer - Tools & machines in adjacent workshop module	New module (not MSS) containing A/L and garage, adjacent workshop
Crew & Medical (WBS-01)	Four crew staterooms medical facility personnel hygiene, shower, lavatory dry john	CCM #3 (pg 3-83) crew command stateroom 3 staterooms/no medical (in CCM#2) Personnel hygiene facilities

Table D-7. Structure, Part 3

SUBSYSTEM ITEM	LSB PERFORMANCE REQUIREMENT	MCS PERFORMANCE CAPABILITY
Module Comparison (Cont.)		
Crew & Operations Module (WBS-02)		CCM 1 (pg 3-85
	Crew Commander stateroom	No staterooms
	Three crew staterooms	"
	Command control & field communications center	Command control & field commun. center
	Personnel hygiene	No personnel hygiene also: data lab; photo lab; physics & material science lab (geochem, biosciences)
Assembly & Recreation Module (WBS-05)	Main galley - food prep, stor, work area, laundry airlock assembly/recreation area	Galley Module-growth version (pg 3-88) No airlock (on garage or warehouse) Small rec & assembly area also: hygiene facilities
Drive-in Warehouse Module (WBS-08)	Storage area for intact cargo module Pumpdown capability of module for cargo module transfer Docking capability, available with vehicles	New module (similar to drive in garage module) with airlock

Table D-7. Structure, Part 4

SUBSYSTEM ITEM	LSB PERFORMANCE REQUIREMENT	MSS PERFORMANCE CAPABILITY
Module Comparison (Cont.)		
Lab & Backup	Geochemistry Lab	CCM #2 (pg 3-85)
Command Post Module (WBS-04)	Photo Lab Data Analysis Lab Bio-science Lab Backup command & communication center	Labs with modified (geochem, bioscience) others in CCM #1
Sortie & Transient Crew Module (WBS-03)	Four staterooms Backup galley Personnel hygiene facilities	Four staterooms Backup galley Personnel hygiene facilities also: none
Base Maintenance Module (WBS-06)	Repair station - shelter, mobility suits airlock	part of CCM #2 No airlock - part of garage/ warehouse

APPENDIX E. LSB LANDING SITE CONSIDERATIONS

A principal activity of the LSB will be the routine landing and launching of logistic spacecraft from and to lunar orbit. This function will require a designated flight-pad facility having acceptable approach characteristics and necessary ground support equipment, and at least minimal improvements to facilitate the handling and movement of passengers and freight.

The direction of landing approach from orbit will determine the major axis of an ellipse-shaped landing area for the LSB. This area should have a reasonably smooth and uncluttered surface at least two miles wide and five to ten miles long, with an unobstructed low-angle approach path to favor early visual spotting and afford maximum line-of-sight range to aligned flight communication and beacon transmitters. The flight approach direction should tend to be perpendicular to the prevailing contours, but no determination has yet been made as to gradient constraints or whether the approach should preferably be made toward an upsloped or downsloped surface.

A circle approximately 300 feet in diameter near the center of the elliptical landing area will be designated the Spacecraft Landing Site for definition, flight targeting and improvement purposes. A suggested design for such a facility is indicated on Figure E-1, entitled "LSB Landing Site". The uprange and downrange ends of the area will be reserved for possible overshot and undershot landings, with vehicular access via the service roads routed to the navigational aid stations.

The installation of a homing beacon (RF and visual) and a fan marker beacon is suggested as a practical minimum for navigational and landing aid purposes. Although the antennae would probably be permanently installed, the use of portable transmitters and/or self-contained power sources should be considered, and especially for the more distant fan marker.

The shelter will contain the various protective structures and most of the items of support and operational equipment required for the operation of a manned lunar surface base. This complex will be located cross-range from the landing site to minimize the landing hazard and the gradient along the main connecting road, and approximately 6,000 feet distance from the landing site to minimize the dust problem at the complex and still be within practical walking and power distribution limits. The shelter and the landing site will be connected by a single lane main road which will loop the functional area at each end.

In order to function as a target for precise landings the pad must be visually identifiable to incoming spacecraft pilots during their final approach across the relatively featureless lunarscape. A permanently installed array of site markers is suggested for this purpose, consisting of white panels arranged in a radial sunburst pattern. The radial arrangement will minimize

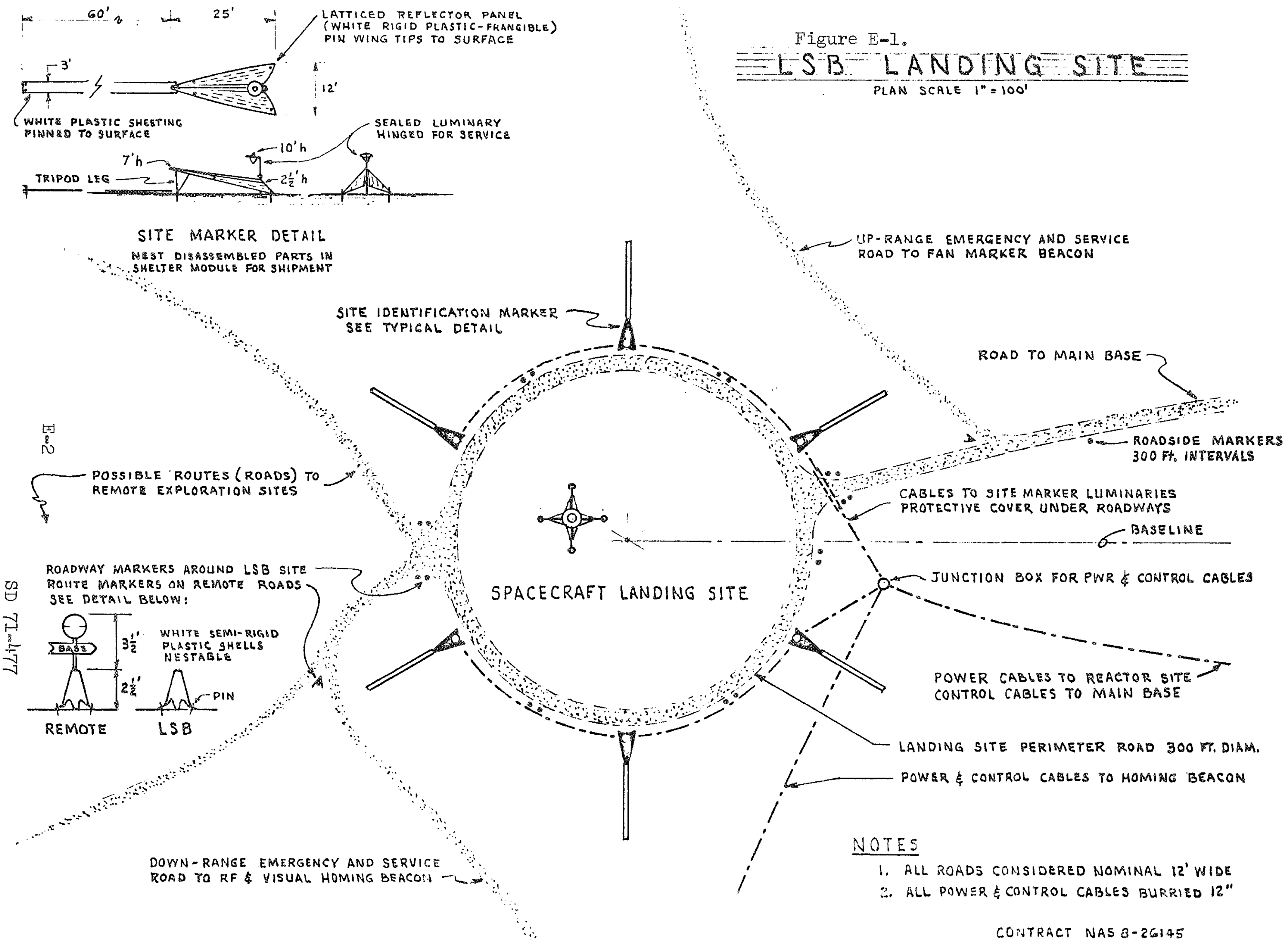


Figure E-1.
LSB LANDING SITE
PLAN SCALE 1" = 100'

the destructive and dust deposition aspects of rocket exhaust plumes and at the same time create a distinctive image on the lunar surface.

Each panel assembly would consist of a frangible rigid-plastic reflector panel located at the innermost end of a ribbon of flexible sheet plastic, and all would be pinned down to the surface to maintain their proper positions and alignment. The rigid-plastic panels would be designed with a folded shape and a latticed surface to maximize reflection and visibility from all directions above, and at the same time minimize dust retention and any uplift associated with rocket exhaust. The frangible aspect of the design is a safety feature in recognition of the fact that a spacecraft might at some time land directly on top of the panel.

A suggested shape and set of dimensions is shown on Figure E-1. The dimensions of the rigid panel are constrained by logistic considerations the sizes suggested are intended for two-piece shipment within a 15 foot diameter by 30 foot long shelter module, assuming the 5 foot 4 inch net width of the half-panel would pass through a minimum 5 foot 6 inch high door opening in the end of the module. Of course, other assembly plans and shipment concepts are quite possible.

The suggested white color is to achieve a maximum contrast against the bland low albedo lunar surface. As a further aid to visual spotting by an incoming pilot (or even an orbiting pilot) it is recommended that each white reflector panel be equipped with a luminary unit positioned directly above the broad end of the panel. This luminary would be designed as a sealed unit having a plastic enclosure under a reflector plate to project white light down upon the reflecting surface, and having a red or green high-intensity flasher light positioned above the reflector plate. The entire unit would be mounted atop a hinged arm which could be swung down to allow convenient servicing by suited personnel.

Although perhaps most essential for possible lunar night landings, the luminaries (especially the colored flasher units) would nonetheless also serve very well during normal daytime landings. The lights could be operated intermittently or flashed, together or in any sequence, to (1) conserve power drain, (2) enhance visual spotting, and (3) transmit information by visual signal to landing or orbiting spacecraft.

Power supply and control cables leading to the pad will serve the alternate and dual purposes of powering the luminaries and beacons during landing (and possibly take-off) operations, and thereafter supporting the spacecraft during its stay on the pad. These cables would terminate at a junction box, which would distribute the power and also provide a connector for the spacecraft umbilical cable which would be laid over the surface after landing. These cables should be installed with a minimum bury of at least 12 inches and so located as to minimize the possibility of damage from vehicular traffic and possible surface erosion at the pad. The soil cover over buried cables will also provide insulation protection against solar radiation.

It is noted that the range marker and homing beacon facilities will be located respectively in the uprange and downrange emergency landing areas, and that the roads into these areas will serve the dual purpose of facility service access and a measure of improved road access to emergency landing incidents. It is noted that these service roads should be arranged so as to compliment the road/route requirements to remote exploration areas, but at the same time avoid unnecessary crossings of the cable system.

Figure E-1 also shows a typical detail of the standard roadway marker proposed for use throughout the main LSB area, and a detail of accessory additions for use as route markers in more remote locations. The large white plastic sphere on top (actually two nestable hemispheres) is to facilitate the spotting of a route or control point from a distance by the driver of a lunar surface vehicle. Typical locations for route markers might be outlying base facilities (i.e., antenna station), or forks along frequently used exploration routes, or as trail markers dropped off by exploration parties.

The problem of possible soil erosion within the central pad area of the landing site deserves some mention. Although the gusting forces associated with the rocket exhaust plumes of lunar landing vehicles are reported much less severe than is usually associated with rocket and jet engine exhausts on earth, this aspect is largely offset by the lack of an atmosphere and the 1/6 gravity factors on the Moon. The natural result of this combination of factors would indicate a severe dust and erosion problem if the landing pad is located in an area which has a thick mantle of unconsolidated dust and rock debris, and especially in view of the almost certain lack of a heavy construction capability during at least the initial and early periods of LSB activity.

No really good solution to the landing pad erosion problem is yet apparent. It has been noted that the exhaust impingement temperature/pressure proviles for lunar landing type rocket engines is such that something in the nature of a special tarpaulin could serve as a take-off pad, however, a reasonable sized tarpaulin would constitute quite a problem for a landing pilot (or the ground crew), and a full sized (300 foot diameter) tarpaulin would be prohibitively heavy and of doubtful durability on a soft surface.

Other alternatives for solution of the dust problem would therefore seem to be, to select a pad-site where the surface is either devoid of significant dust cover, or covered by not more than one or two feet of dust so that several landings during the initial exploration or construction phases could blow the surface clean and expose a reasonably firm subgrade surface (hopefully smooth and nearly level). A similar alternate proposal would be to clean the surface with a series of uniquely designed explosive charges - this technique might be especially appropriate and necessary if the mantle is up to two, three or four feet thick and contains rock debris along with the fine dust.

APPENDIX F. LUNAR SURFACE BASE MAINTENANCE PLAN

GENERAL

Lunar Surface Base (LSB) mission success and crew safety over the long duration of the mission depend upon continuous system operation. The requirement for sustained, long-life equipment operation establishes the need for maintainability because maintenance reduces the need for extensive development to maintain system reliability. All elements of the LSB program, including experiments and support equipment, shall be designed for complete maintainability. Maintenance is the restoration of preservation of an equipment to its designed operational condition. By the optimum utilization of maintenance during mission operation, the impact of damage and failures can be greatly reduced.

REFERENCE DOCUMENTS

S&E-QUAL-69-9, NASA/MSC Mission Maintainability and Maintenance Guidelines, 1 May 1969.

MAINTENANCE CATEGORIES

Mission maintenance activities include fault detection, fault isolation, servicing, calibration, adjustment, repair, modification, replacement, update and refurbishment, and functional verification. Maintenance activities are categorized either as scheduled or unscheduled.

Scheduled maintenance (preventive maintenance) is any planned maintenance activity deemed necessary to retain or enhance the functional capability of the equipment.

Unscheduled maintenance (corrective maintenance) is the activity required to restore normal operating capability after a random failure, loss of calibration, drift, damage, etc., of functional and non-functional equipments. Unscheduled maintenance stems from discovery of a failed, damaged, or degraded equipment, and is the restoration of an equipment to its designed operational condition or replacement of the malfunctioning or damaged item.

ANALYSIS

A maintenance analysis shall be performed on all elements of the LSB mission. Maintenance estimates and predictions will be developed for each subsystem to serve as maintainability goals. The analysis will assist in trade-off studies with redundancy and hardware reliability, and will evaluate mission maintenance action in relation to mission constraints and limitations; e.g., weight and volume.

The analysis shall begin during the design concept phase of the program and shall be continually updated as the program progresses. The analysis shall result in the following:

1) Maintenance criticality category list.
2) Maintenance priority list.
3) Recommendation as to the items that shall be maintained and any design modifications and/or supporting development required to allow maintenance of these items.
4) List of any items for which maintenance was not recommended, giving the reason why the item is not to be maintained.
5) Tools, equipment, and facilities required.
6) Weight and volume of maintenance equipment requirements.
7) Recommendation of time allocations for all maintenance actions.

MAINTAINABILITY CRITERIA

Maintainability is the design characteristic that facilitates the preservation or restoration of an element to its designed operational state with a minimum of time, skill, and resources under the planned maintenance environment.

Maintainability shall not be an isolated requirement that is considered separately, but shall be developed as an integral part of the total design requirements.

Maintainability shall be incorporated into each element utilized to accomplish the LSB mission such as spacecraft, landers, shelters, rovers, flyers, experiments, and equipment and tools for fabrication, assembly, and test of various elements. Not only shall element maintainability be attained, but also mission configuration maintainability shall be provided.

DESIGN CHARACTERISTICS

During the conceptual phase, the following design factors should be considered:

Replaceable Unit (RU) - Subsystem maintenance and repair will be accomplished normally by removal and replacement at an RU level. The removed RU may than subsequently be repaired at the piece part level. Important characteristics of an RU are:

1) It should be a logical, functional unit.
2) It must be capable of performance measurement and failure indication by means of an onboard checkout system.
3) It should be capable of replacement by a single crew member.
4) Removal and replacement of the RU must not create an unsafe condition and should be accomplished with minimum impact to subsystem operation.
5) Maintenance tools and techniques should be standardized, and time to perform maintenance should be known.
6) Removal and replacement techniques should be identical for both planned and unplanned maintenance.

Monitor/Alarm - Selected measurements reflecting the subsystem operation/performance should be cued by the onboard checkout system to determine operational status. When a performance anomaly occurs, the fault area and detection mode shall be immediately available at the control panel.

Fault Detection - Detection of suspect malfunction is accomplished either by readout from the onboard checkout system or by manual and visual methods. The method(s) selected should be determined based on time element, skill levels, repairability and maintainability, proximity to LSB facilities, and equipment available.

Isolation and Verification - Fault isolation of a verified malfunction is accomplished by crew action consisting of data evaluation, inspection, or test stimulus of the failed assembly.

Accessibility - Equipment and equipment support shall be accessible and so arranged to permit passage of units, tools, and crew for repair. The route shall be safely traversable in a pressurized suit.

Crew and Equipment Safety - Rough surfaces, sharp corners, and uninsulated power leads shall be eliminated. Limit stops shall be provided on roll-out racks and drawers to prevent their being dropped. Capability shall be provided to isolate those segments of the subsystem which will require repair, replacement, or servicing, such that the critical subsystem functions remain active, and hazard to personnel and contamination of the environment or subsystem is prevented.

Electrical Connectors and Cables - Connections shall be possible only in the correct orientation and cables shall be attached to remain at the site of the disconnect.

SPARES

From the mission maintenance analysis, assistance will be provided for the reliability and safety determination of items to be repaired. These items will establish the kind of spares required. The spares requirement for all major mission elements will be examined for equipment duplication and storage requirements. The individual lists will be integrated to minimize the number and types of spares while retaining the required reliability.

During mission operation, a means shall be available for checking spares to determine if a spare is in operating condition prior to installing the spare in the system. Preferably, the condition of the spare shall be ascertained prior to removal from the storage location.

TOOLS, EQUIPMENT, AND FACILITIES

The tools, equipment, and facilities shall be defined that must be aboard either the primary mission or resupply mission if mission maintenance is to be successfully accomplished. Requirements should consider items that are portable, can be utilized in more than one location, and can be utilized to check out more than one item or one system.

Tools, equipment, and facilities shall be minimized to the extent feasible. Coordination with design engineering will be initiated during preliminary design. The preliminary design of all major elements shall be analyzed to determine common or multiple usage of items among major elements.

To reduce weight and complexity and to increase safety, hand operated tools are preferred to power operated.

All maintenance equipment shall be easily and safely transportable.

To energize a power operated item shall require an overt conscious act of the repairman. Design shall prevent accidental energization.

INVENTORY SYSTEM

An inventory system shall be established to account for and control the disposition of spares and maintenance tools, equipment, and facilities. The system shall provide for quick, easy, and accurate updating of data needed for repair resources of each individual element and integrated mission hardware.

PROCEDURES AND DISPLAYS

A step-by-step procedure shall be developed for each repair task. If automatic isolation to the level of repair is not provided, the procedure shall include description for manual isolation, repair resources required, equipment operating instructions, test points to be checked, expected outputs, and each step in the actual repair or maintenance.

Procedures shall be brief, understandable, and easy to read and follow during anticipated normal and emergency environmental conditions.

A method of indexing and display shall be developed which will allow immediate availability of failure information and ready selection of repair mode. A common medium shall be used for all information display. The medium shall be highly reliable and useable under adverse environmental conditions. If the display is an operational device, the medium shall be repairable during the mission.