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FINAL REPORT

A STUDY TO DEVELOP
A LOW TEMPATURE BATTERY
SUITABLE FOR SPACE PROBE APPLICATIONS

prepared for

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PREPARED UNDER CONTRACT NAS 3-6018 BY
TRW SYSTEMS GROUP
REDONDO BEACH, CALIFORNIA

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NASA FINAL REPORT

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TRW Systems Group

ABSTRACT

35406

This study examines in detail the practicability of heating spacecraft batteries to extend their capabilities into the ultra-low temperature space applications. Several acceptable design concepts are established using systematic evaluations of batteries, heat sources, thermal controls and mission interfaces. Potential applications are determined and new technology required for hardware design is defined. A development plan for new technology is presented.

Preliminary heated battery designs for two types of thermal requirements are investigated and model analyses are presented detailing technical performance.

Author

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1. INTRODUCTION

The purpose of this study was to investigate (without performing laboratory tests) the feasibility of heating batteries to extend their usefulness in space applications involving very low ambient temperatures and to establish the requirements of the system interface for low temperature batteries in space-probe systems. The study involved a systematic evaluation of all aspects of the problem including the types of batteries, heat sources, thermal controls, and mission interfaces.

Results and conclusions from this study, potential applications of the results, and missing technological links are reported; and recommendations for future development of new technology are presented.

1.1 STUDY DESCRIPTION AND WORK STATEMENT SUMMARY

The study performed under Contract NAS 3-6018 was a research program to develop a low temperature battery design for deep space probes.

The study was divided into three phases as follows:

Phase I - State-of-the-Art Survey

Phase II - Analysis

Phase III - Development Program Definition

The work statement summary for each phase of the study is as follows:

- Phase I - State-of-the-Art Survey
  - a) A literature search conducted to determine approaches and parameters previously investigated, was directed toward heated battery systems which enable battery operation to -73° C with potential to -143° C.
  - b) An industry survey, conducted to determine related areas where unreported work is being done, included isotope heater development, chemical heating system development, and other pertinent technology.
  - c) Feasible battery heating methods were isolated; the selection was based on a comparative analysis of space mission requirements.

d) Based on the above searches and analysis, a Phase II Analysis Plan was prepared.

- Phase II - Analysis

  a) Space systems requirements for batteries were investigated to determine energy requirements, mission durations, thermal system design restrictions, tentative mission temperature extremes, system and battery heat requirements, dynamic environments, and startup requirements.

  b) Basic battery heater properties were analyzed by comparing heat sources, thermal control methods, applicable batteries and battery packaging methods.

  c) Heated battery designs were integrated by empirical methods and by computer simulators to determine feasibility.

  d) The more practical heated battery systems were selected and further analyzed in depth to establish performance and design limitations in terms of mission duration, heat requirements, radiation shielding, electrical performance, and physical characteristics.

  e) Safety and logistics problems of handling and using battery heaters on the ground and in a space vehicle were investigated and defined.

  f) The battery designs and analysis missions selected for finalization were reduced to computer and empirical models and their constants fixed for purposes of preliminary design feasibility studies.

- Phase III - Development Program Definition

  a) The preliminary design concepts established in Phase II were evaluated for status of technology required to reduce them to working breadboards. The limits of technology were established for the general heated battery case, and the requirements for development of new technology were defined.

  b) A final report, submitted herein, was prepared.

1.2 SUMMARY OF WORK COMPLETED

The study of the feasibility of heating batteries for space applications was conducted under the guidelines of the above Work Statement Summary. The results of the Phase I state-of-the-art survey are reported in detail in Section 2, and the analysis of heated battery applications and designs is reported in detail in Section 3. Definition of a heated battery development program is presented in Section 4.

To briefly summarize the applications for heated batteries in space, the following comparison of various types of space missions versus potential battery heating methods was developed to enable selection of the more pertinent areas for detailed analysis.

a) Types of Space Missions for Batteries

  1) Nonoperative, unactivated, long-duration standby
  2) Nonoperative, activated, long-duration standby
  3) Continuous discharge at low rates
  4) Periodic discharge at high rates, open-circuit standby for short periods
  5) Periodic discharge at high rates, open-circuit standby for long periods
  6) Periodic discharge at high rates, trickle-charge standby for long periods
  7) Periodic discharge at high rates, high-rate recharge during short periods
  8) Nonoperative, unactivated, end-of-mission activation and short high-rate discharge
  9) Nonoperative, activated, open-circuit standby, end-of-mission low- to medium-rate discharge.

b) Methods of Heating Batteries

  1) Bootstrap heating
  2) Electrical heating - battery energy

      Self heat
      Heater

  3) Electrical heating - power system energy
  4) Radioisotope heating
  5) Cryogenically-stored fuels

6) Storable fuels and propellants
    - Monopropellants
    - Hypergolic propellants

7) Pyrotechnic and catalytic combustion heating.

Table 1-1 contains a comparison of the best theoretical methods of heating batteries for space missions. From the table it is apparent that there are, basically, two low temperature applications for batteries:

a) Long-term, continuous-heat, and

b) Short-term, rapid warmup with sustained heating for short periods.

Low temperature heated battery systems were, therefore, investigated in greater detail with the objective of establishing design concepts of heaters for both of the above applications by means of a study based upon the work statement.

1.3 RESULTS AND CONCLUSIONS

The study resulted in an accumulation of information and analyses which conclusively demonstrate the feasibility of operating present state-of-the-art battery cells in environments as low as -143°C, with potential of operating at even lower temperatures. Heaters are used to maintain the batteries at an acceptable operating temperature by one of two methods:

a) Maintaining batteries continuously at acceptable operating temperatures through the use of constant heat sources and active thermal controllers (most acceptable method found).

b) Warming batteries, stored at extremely cold temperatures, at the end of a long storage period. (The study has shown that this may be feasible if technology can be developed to support the assumptions that a battery can be stored in the frozen condition and successfully heated using chemical cartridge heaters or other high-rate heating methods.)

The study has further established the need for low temperature batteries in space missions and the technology required to develop the batteries to the prototype level.

1.3.1 Available Technology

In general, the basic technology required for battery design,

Table 1-1. Mission Versus Potential Heating Method in Space

	MISSION TYPE								
BATTERY HEATING METHODS	NONOPERATIVE, UNACTIVATED, LONG-DURATION, STANDBY	NONOPERATIVE, ACTIVATED, LONG-DURATION STANDBY	CONTINUOUS DISCHARGE AT LOW RATES	PERIODIC DISCHARGE (HIGH RATES) OPEN-CIRCUIT, STANDBY (SHORT PERIODS)	PERIODIC DISCHARGE (HIGH RATES) OPEN-CIRCUIT, STANDBY (LONG PERIODS)	PERIODIC DISCHARGE (HIGH RATES) TRICKLE-CHARGE, STANDBY (LONG PERIODS)	PERIODIC DISCHARGE (HIGH RATES) HIGH-RATE RECHARGE (SHORT PERIODS)	NONOPERATIVE, UNACTIVATED, END-OF-MISSION ACTIVATION, SHORT HIGH-RATE DISCHARGE	NONOPERATIVE, ACTIVATED, OPEN-CIRCUIT, STANDBY, END-OF-MISSION LOW- TO MEDIUM-RATE DISCHARGE
BOOTSTRAP HEATING	O	O	O	O	O	O	O	O	O
ELECTRICAL HEATING (BATTERY ENERGY)									
--SELF-HEAT	O	O	O	Y	O	O	X	X	O
--HEATER	O	O	Y	Y	Y	Y	X	Y	O
ELECTRICAL HEATING (POWER SYSTEM ENERGY)	X	X	X	X	X	X	X	X	X
RADIOISOTOPE HEATING	X	X	X	X	X	X	X	X	X
CYROGENICALLY-STORED FUELS	O	O	Y	Y	Y	O	Y	Y	Y
STORABLE FUELS AND PROPELLANTS									
--**MONOPROPELLANTS**	O	O	Y	Y	Y	O	Y	Y	Y
--HYPERGOLIC PROPELLANTS	O	O	Y	Y	Y	O	Y	Y	Y
PYROTECHNIC AND CATALYTIC COMBUSTION HEATING	X	X	O	Y	O	O	O	X	X

X = THEORETICAL* ACCEPTANCE
Y = TIME-LIMITED MISSIONS ONLY
O = NOT APPLICABLE

*A DETAILED COMPARISON IS MADE IN SECTION 3.

heater design, and low temperature heated battery systems design is available, though the search of the literature and survey of the industry showed that the technology for 1) active thermal control, 2) storage of batteries at temperatures below -54° C, and 3) activation of high-rate heat sources at temperatures below -54° C is limited. Paragraphs 1.4 and 1.5 discuss the areas of technology which need to be developed before a battery can be designed for low temperature deep space probe missions.

The following paragraphs outline the available technology for batteries, heaters, thermal controls, and system designs.

1.3.1.1 Batteries

For typical space probe missions, it is apparent that only a few of the batteries presently in use or under development are applicable. The silver-zinc battery is applicable for 1-year missions, requiring limited cycling and has remote activation potential for extending the mission length significantly if means of storing the battery at low temperatures can be found. The silver-cadmium battery affords an excellent potential for 1 to 3 year missions. The nickel-cadmium battery is the most attractive for long duration missions of 3 to 5 years, and has a 10-year potential. The ammonia battery affords potential as a reserve for low temperature operation but is extremely difficult to handle and means must be found for remote activation. The zinc-mercury battery affords potential as a primary battery for mission durations of up to 3 years.

Batteries can be provided with a capability of supplying energy densities of 10 to 30 watt hours per pound (whr/lb) at high rates if operated in a temperature range of 4 to 38° C, and offer the potential for extending the energy density to 100 whr/lb for low-rate missions. From the study it is apparent that all of the batteries are capable of use in deep space probes; but the nickel-cadmium battery presently is the most acceptable, and the silver-cadmium battery is the second most acceptable. The question of sterilization was investigated briefly but no conclusive data were available which would cause the above conclusions to change.

1.3.1.2 Heaters

The investigation of potential methods of heating batteries for space applications resulted in the conclusion that the following three types of heaters were promising:

a) Electrical heaters deriving energy from a space vehicle primary power system.

b) Radioisotope heaters utilizing Pm-147 and Pu-238 radioisotopes.

c) Chemical cartridge heaters using combinations of metals and oxidizers in sequenced firings. (This type of heater would require new technology to extend ignition below -54° C and to develop methods of controlled sequential burning for time periods in excess of 10 to 100 hours.)

The heaters investigated were limited to types which could be operated in space and would be capable of supplying 5 to 50 watts of heat at controlled rates over periods of days to years. The general ground rules for selection of heaters were based on heating requirements to maintain batteries between 4 and 38° C in low temperature space environments of -40 to -184° C.

The technology for electrical heating with power supply energy is readily available. The following are typical:

a) Current technology: Present resistive heaters, insulation materials, and active thermostatic temperature controls are adequate, with minimum development, for space applications. Present heaters used in missile programs are available from most battery manufacturers.

b) Proposed designs: Electrical heaters would be incorporated into an egg-crate aluminum structure capable of providing uniform heat distribution (5.6° C gradients maximum) similar to the Orbiting Geophysical Observatory battery design (NASA Contract NAS 5-899). The design would utilize internal resistance heaters surrounding the cell blocks and external insulation and platform standoffs to isolate the battery from its environment. The weight and volume penalties for this approach are negligible compared to present spacecraft battery designs.

c) Operating time for electrical heaters is limited by the ability of the primary power system to provide the necessary energy.

d) The feasibility of applying electrical heaters to most space missions is poor. The added increment of the primary power system is costly to provide, and it may be impossible during eclipse or other low power periods when heating is most desirable. The use of electrical heating is the optimum choice if power system energy is available without increased cost and during periods in which the battery is exposed to low temperature. The technology presently available is sufficient and no new research is required to implement the concept.

The technology for radioisotope heating is presently being developed; however, new technology, summarized in paragraph 1.4 of this report, will be required for the specific application of radioisotope heating to batteries. The study developed the following conclusions about the application of radioisotope heaters to battery heating:

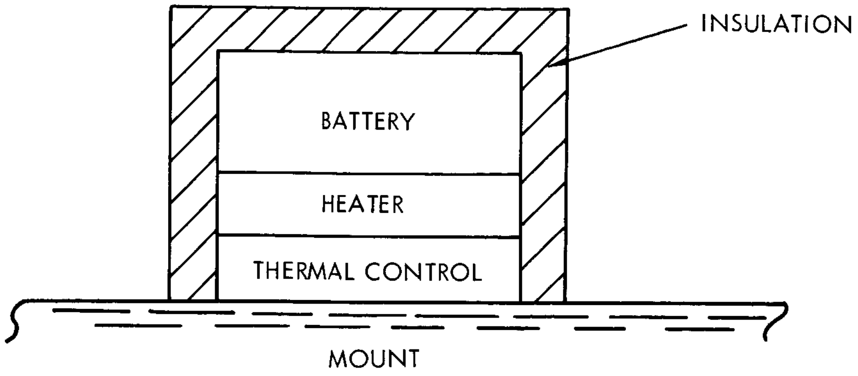
a) Current technology is in the development stage; the work remaining to develop battery heaters appears to be straightforward.

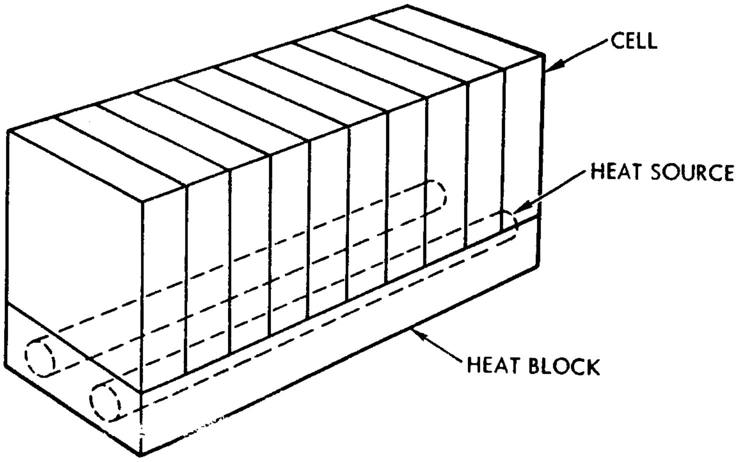
b) The proposed design developed by the study is essentially an internal isotope core mounted in a heat plate for heat distribution. The combination battery cell group and heater baseplate would be insulated from its environment with superinsulation and standoffs. An active thermal control will be required to regulate the heat flow from the battery to its environment to avoid overheating the battery during warmer temperature cycles. The weight and volume penalties for this design would be approximately 10 percent for a 600 watt-hour nickel-cadmium battery. The design concept is shown in Figure 1-1.

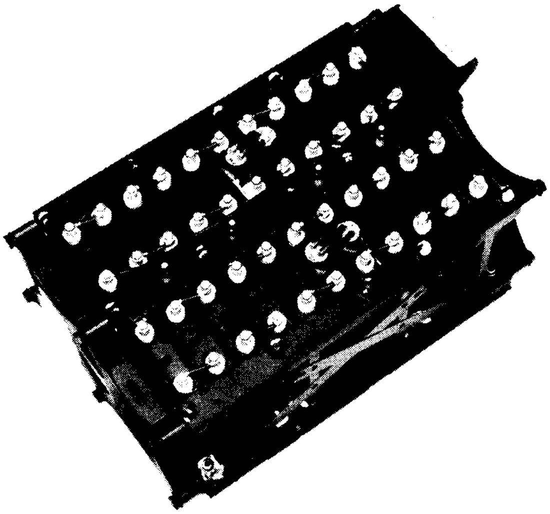
c) The operating time for the heater would be in excess of 10 years without significant degradation.

d) The most desirable method, found during the study, of heating batteries in space is by means of the radioisotope heater. The heater would permit full time control of battery temperature, never permitting the battery to be exposed to intolerably cold environments. The technology required to design the heater into a battery is under development, with a few problems still requiring research. (One such problem is to define the interfaces between the battery cells and the heater, and between the heated battery and sensitive radiation detection instrumentation.) Radioisotope capsules of the types recommended by the study have been evaluated and the materials necessary to incorporate the capsules into heaters are available. The feasibility of the use of radioisotope heaters to augment a spacecraft control system by providing heat to a battery is apparent. The feasibility of the use of such a heater

LOW TEMPERATURE BATTERY DESIGN







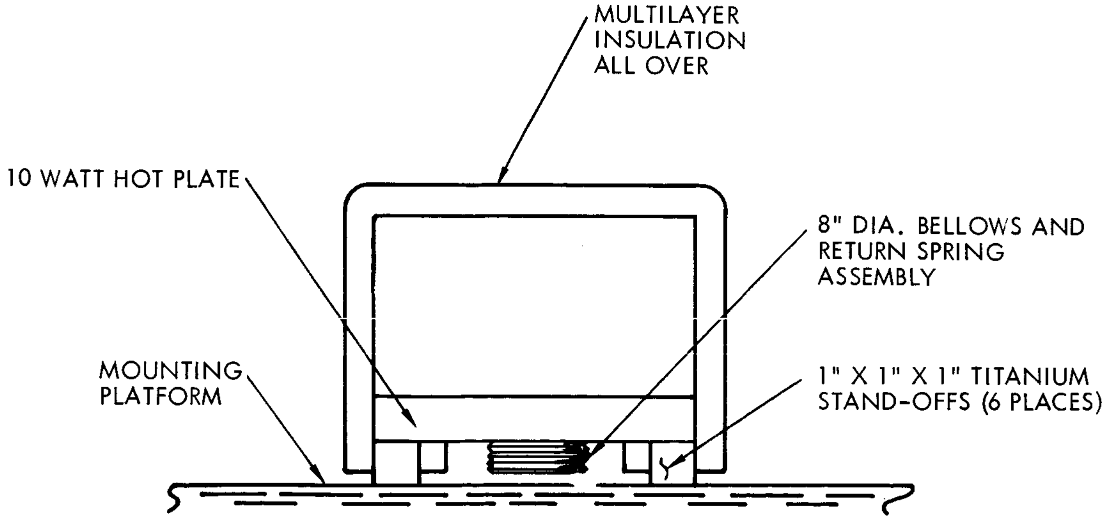


Figure 1-1 Conceptual Design of Battery

as a primary thermal control system is high for limited applications, including steady-state low temperature environments; cycling low temperature environments with short-period high temperature exposure, in which the period is not of sufficient duration to raise the overall system temperature above the maximum acceptable battery operating temperature.

The technology required to use chemical cartridge heaters as battery heaters in space applications has not been developed. Further investigation of the chemical cartridge as a battery heater resulted in the following conclusions:

a) Chemical cartridge heating in its current form is limited in scope and in application in space. It is limited to a minimum temperature of -54° C, a life of approximately 1 to 2 years of storage, and requires a significant improvement in technology to obtain controlled heating rates for periods in excess of 10 to 100 hours.

b) The proposed design for warmup application is an egg-crate core configuration with a matrix of cartridges and battery cells similar in concept to that shown in Figure 1-2. The configuration for steady state temperature control is essentially the same as the radioisotope concept, placing the cartridges on a baseplate to minimize temperature hot spots in the cell block. The design would incorporate sequential firing devices and a communication logic command system to control the rate of ignition and burning. Weight and volume penalties could not be specifically determined but were estimated to be approximately 12 percent to maintain a 600 whr nickel-cadmium battery at 4° C in a -27° C environment (31 percent in a -184° C environment) for 10 hours of heating. The penalty for warmup was comparable for a 4-hour warmup period.

c) The operating time for this device is limited to 10 to 100 hours maximum because the logic required to control the heating rate and the number of firing devices required prohibits long-life controlled heating, and the heater weight becomes prohibitive at 100 hours (50 to 60 pounds for 600 whr nickel-cadmium battery at 4° C in a -27° C environment).

d) The feasibility of using this device as a battery heater is poor. The method will require extensive basic research and development of heater materials, sequencing methods, battery to heater interface techniques and ignition.

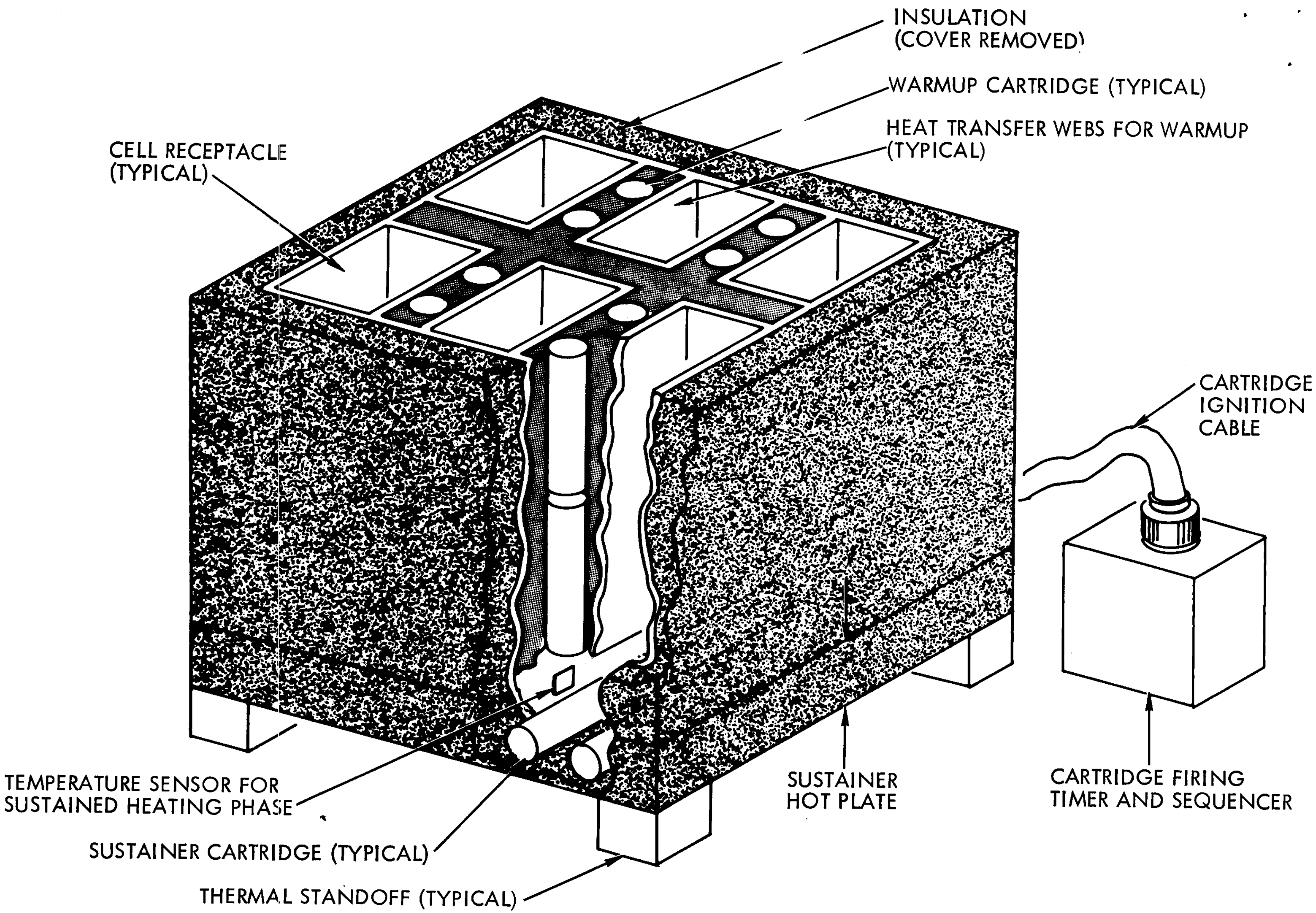


Figure 1-2 Battery Assembly for Pyrotechnic Cartridge Warmup and Sustained Heating

It was obvious from discussions in the literature and from the results of the study that extensive new technology will be required to apply this type of heater with controlled rates to a temperature sensitive battery. (The study could not conclusively show that this type of design would work nor could it prove that it would not.) Section 4 contains a development program which would further define the capability of the design and determine if it is feasible to use it for space applications.

It was concluded from the above results that the most desirable method of heating batteries in space is to use surplus electrical power from the primary power system if it is available. The power system increment required will be 5 to 50-watts. The second choice is to use a 5 to 50-watt radioisotope heat source with an active thermal control element. The second choice was found to offer excellent potential for long-life heating requirements as well as for applications in which the battery must always be in a condition to deliver transient energy demands. The last choice is to use chemical cartridge heating if it can be developed, if batteries can be safely frozen, and if adequate thermal control can be achieved during a short duration warmup.

1.3.1.3 Thermal Controllers

Numerous investigations were conducted to find passive and active thermal control devices which have been developed and could be used with low temperature batteries. The search of the literature and survey of industry did not disclose any devices which would meet all of the requirements. Several devices were investigated extensively and the following conclusions were reached:

a) The technology required to insulate batteries has been developed but the simple use of insulation for the application in question results in a single-point design which is incompatible with nearly every conceivable space mission, since most of the missions involve temperature cycling and a varying spacecraft environment.

b) Heat pipes have possible merit in the future. It was concluded, however, that the technology of such devices is still in the development stage and no conclusions can be drawn at this time for the application.

c) Bimetallic thermal switches were investigated at some length. It was concluded that the devices do not offer potential for the present application because the thermal conductivity through the device is low and the conducting area which can be provided using the devices on a battery package did not meet the requirements for controlling battery thermal inputs and dissipation. The devices also suffer acute thermal conductivity problems in vacuum; a reliable mechanism for making the interfaces thermally conductive has not been developed.

d) Because the study of existing technology provided no significantly useful devices for low temperature battery application, several new ideas were generated as part of the study. The devices were investigated using mathematical models and their feasibility was demonstrated with expected performance far superior to that of any other device studied. Details of the devices are disclosed in Section 5. It was concluded from the investigation that bellows-actuated thermal switches offer the best potential method of controlling battery heat input and dissipation to maintain acceptable battery temperatures in ambient temperatures as low as -240° C and as high as +260° C.

The available technology on thermal controllers which could be used for low temperature battery applications is very limited and will have to be developed further before a practical low temperature heated battery design can be developed. The recommended approach is to investigate and develop either, or both, the use of bellows combined with materials whose change of state at high temperature results in expansion of the bellows, and a high pressure mechanical contact with high thermal conductivity or whose change of state results in increased thermal conductivity of the filler medium in conjunction with a heat switching action of the bellows thermal switch. Such a device would isolate the battery from its mounting surface when the environment is extremely cold and would tie the battery directly to the mounting platform when the battery temperature becomes excessively hot.

1.3.1.4 Low Temperature Battery Systems

No technology (other than the ammonia battery system) could be found on a low temperature battery system for space applications. The ammonia battery system offers some potential but has not been fully developed for space systems requirements.

Technology developed for heating batteries for terrestial applications is limited to a minimum temperature of -54° C. The technology is not directly applicable to space missions because the heat transfer medium in space is primarily conduction whereas on Earth it is primarily convection. In addition, the packaging technology used on Earth cannot be used on a spacecraft or in a space environment because of dynamic environment requirements which prevent complete thermal isolation of the battery package from its environment with insulation.

The study proceeded with further investigation into low temperature systems, and into development of the technology, to define the requirements for a heated low temperature battery system for space missions. The technology was drawn from existing space applications and projected estimates of system designs for future missions. Details are presented in Sections 2 and 3 of this report.

1.3.2 System Requirements for a Low Temperature Space Battery

The system requirements for a low temperature space battery are summarized in the form of design specifications. The detailed analysis and requirements leading to the specifications are presented in Section 3. The requirements are referenced to the following general ground rules developed in the study:

1.3.2.1 System Definitions

For cold environments, use a heated battery system consisting of a heat source compatible with specific mission requirements, an active thermal control, and an integrated high-conductivity cell block configuration and thermally insulated battery package design.

For hot environments, rely on a spacecraft thermal control system design. This approach is not a part of the battery problem considered in this study and is excluded from consideration in the following specifications.

1.3.2.2 General Temperature Requirements

Spacecraft surface temperatures range from -240 to +260° C. This general environment is applicable for all missions considered.

Local battery environments within a spacecraft or experiment package range from -40 to +38° C. (Certain exceptions would require a range of -240 to +38° C.) The study found this range to be typical of that which a spacecraft thermal control system would be required to provide for electronic equipment in general.

Battery operating temperatures range from 4 to 38° C. The study found that outside of this range, the performance of standard space batteries dropped sharply.

1.3.2.3 Heating Requirements

The study found that, in general, heat supplied at 5 to 50-watts would be sufficient to maintain a battery in its normal operating temperature range when exposed to the temperature specified above.

The specifications which follow are essentially battery design criteria which are intended to form an outline for accumulating technology required to design a low temperature battery based upon the above ground rules. It is general in nature and defines the extremes to be anticipated.

1.3.2.3.1 Capacity. The battery system in general will provide 50 to 1,500 whr of energy at discharge rates of C/100 to C over the temperature ranges specified below. The voltages which can be expected as a requirement for power system loads should be between 20 and 50 volts. Voltage regulation would normally be between ±10 and ±30 percent from nominal bus voltage to the end-of-discharge voltage over the temperature range specified.

1.3.2.3.2 Size and Weight. Battery weight should be 10 whr/lb nominal, with design goals of 30 to 50 whr/lb for end-of-mission primary batteries and on-the-line standby batteries having less than 1 year of life. Figure 1-3 presents a summary of nominal size and weight of standard space batteries. A low temperature battery should meet such requirements.

1.3.2.3.3 Operating Temperature. A thermal environment for a low temperature space battery is expected to fall within the range of -240 to +38° C for the external battery package. The range should be expected to be extended to +260° C using multiple active thermal controls. When the battery is operating on the line in a discharge or charge mode, the temperature differential within the battery cell groups should not

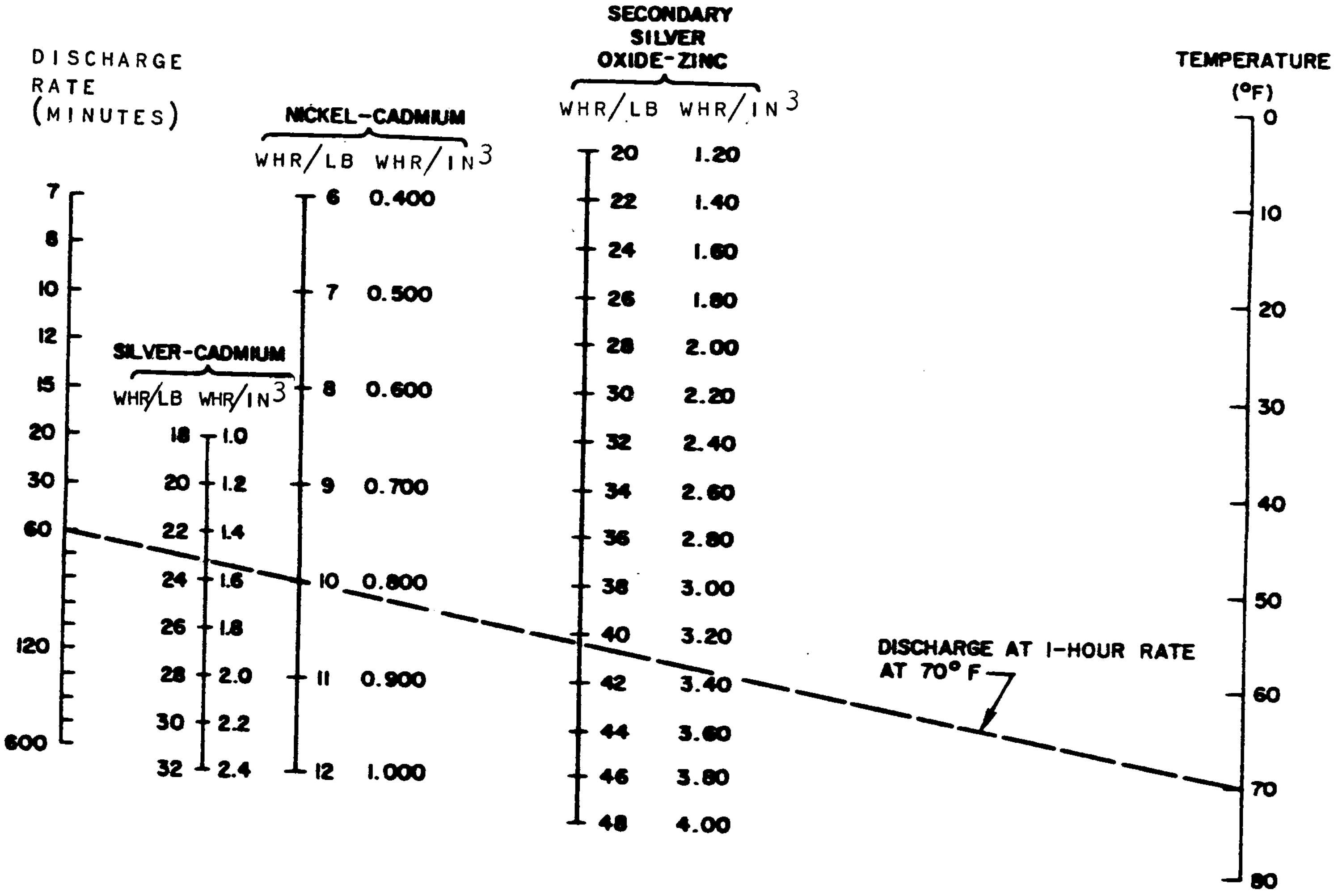


Figure 1-3. Comparison of Specific Energy of Secondary Silver Oxide-Zinc, Silver-Cadmium, and Nickel-Cadmium Systems

exceed 5.6° C in the above ranges to provide for adequate cell balance. The study has shown that the typical lower temperature for a battery in a spacecraft would be approximately -40° C and that a 5 to 50-watt heater would have to be provided in order to maintain the battery in a useful temperature range (4 to 38° C). In addition, any heated battery design using radioisotopes must be safe for re-entry and mission-abort conditions.

1.3.2.3.4 Nonoperating Temperature. A battery system in a non-operating condition should be capable of being stored at a temperature as low as -184 or -240° C. If it is to be operated in a mission which will not require power until some extended future date, the battery system must be capable of either being warmed from the storage condition to an operating temperature of 4 to 38° C within a period of 4 to 8 hours or of continuously maintaining the cell block at an acceptable operating temperature.

1.3.2.3.5 Dynamic Environment. The following dynamic environment is typical for a low temperature battery system:

a) Complex Motion Vibration:

	Sinusoidal		Gaussian Random	
Axis	Frequency (cps)	Acceleration (g, 0-peak)	Frequency (cps)	PSD Level (g^2/csp)
All	5 to 250	3 to 5	20 to 2000	0.1
	250 to 400	5 to 6.5		
	400 to 3000	10 to 13		

Vibration displacement should be limited to 0.4-inch double amplitude.

b) Spacecraft Mode Response Vibration: Same general requirements as above except as follows for sinusoidal vibration:

Axis	Frequency (cps)	Acceleration (g, 0-peak)
All	50 to 250	20

c) Linear Acceleration: Fifty g should be considered in the design.

d) Shock: The following should be considered in the design:

Thrust Axis	50 ±5 g 2 to 6 msec pulse period
Transverse Axes	25 g 4 msec pulse period

For certain hard lander missions, shocks on the order of 100 to 200 g of the same form described above should be expected.

1.3.2.3.6 Radiation. If a heated battery design uses a radioisotope heat source, the combination of radiation from the source and the attenuation of the battery materials should provide a radiation-free window which is compatible with the low sensitivity radiation detectors which are anticipated for the mission. (Paragraph 3.2 outlines some typical detectors.)

1.3.2.3.7 Packaging. The heated low temperature battery system should be designed as an integrated package which could be securely mounted to a typical spacecraft platform and which would be capable of operation in any environment specified above.

1.3.2.3.8 Activation and Startup. Activation and startup of a low temperature battery in space should be automatic upon command enable signals from an external source of command. Activation must be initiated over the full range of temperature specified above for any battery stored in a nonoperating condition. A battery normally maintained in the operating temperature range and in on-line standby should be capable of immediately supplying energy for transient load demands and would require no heating startup functions (except for thermostatic controls provided to regulate the temperature of battery heating blankets using power system energy).

1.3.2.3.9 Life. Life of a low temperature battery system in space should be 3 to 5 years of operation with a goal of 10 to 20 years. Certain specific missions will require an operational life as low as 1 year; it is anticipated that such missions will be in the minority.

The above specifications summarize the capabilities required for a low temperature battery in space. They are by no means all-inclusive in detail but they do establish the ground rules from which preliminary designs can be established and toward which new technology can be developed. It was found in the study that new technology will have to become available before the above requirements can be met.

1.3.3 Potential Applications

The potential applications for heated low temperature batteries in space probes were examined in the study. It was conclusively shown that requirements exist for a heated low temperature battery with an active controller and may exist for a storable low temperature battery capable of being warmed up to operating temperature. A brief listing of the applications is shown in Table 1-2.

1.3.4 Design Concepts

Two basic design concepts were developed in the study:

a) A long-term continuous-duty battery which would remain continuously on the bus and would be maintained within its normal operating temperature range at all times.

b) A long-term low-temperature storable battery capable of being warmed up at end-of-mission, followed by a short duty cycle while the battery is maintained within its normal operating temperature range.

The conceptual model for the long-term, continuous-duty battery is shown in Figure 1-1. The design envisioned for the long-term storable battery is similar except that the source of heat for warmup will have to be interspersed between the cells as shown in Figure 1-2. The entire battery system in each case would be insulated from the surrounding environment with superinsulation and low thermal conductivity standoffs. Figure 1-4 shows the performance of the conceptual model of a long-term, continuous-duty battery used in a Mars Lander model. The performance shown demonstrates the feasibility of the concept.

Several similar models were developed using the above design concepts in potential space applications including a deep space probe, and Mars soft lander and hard lander models. The technology was

Table 1-2. Potential Mission Applications for Low Temperature Batteries

	Estimated Mission Date
PLANETARY MISSIONS	
Lunar Landers	1967 - 1970
Lunar Beacons and Lander Experiments	1969 - 1971
Mars Orbiter	1969
Mars Landers	1975 - 1980
Small Space Laboratories	1970
Large Space Stations	1975
Earth Orbiters with Local Cold Spots	1970
PROBES	
Galactic Probes	1971
Advanced Pioneer Missions	1970
Near-Asteroid Probes	1972
Mercury Probes	1972
Deep-Space Planetary Probes	1973
Far-Asteroid and Comet Probes	1974
Jupiter Probes	1975
Jupiter Flyby	1979
Saturn Flyby	1980 - 1985

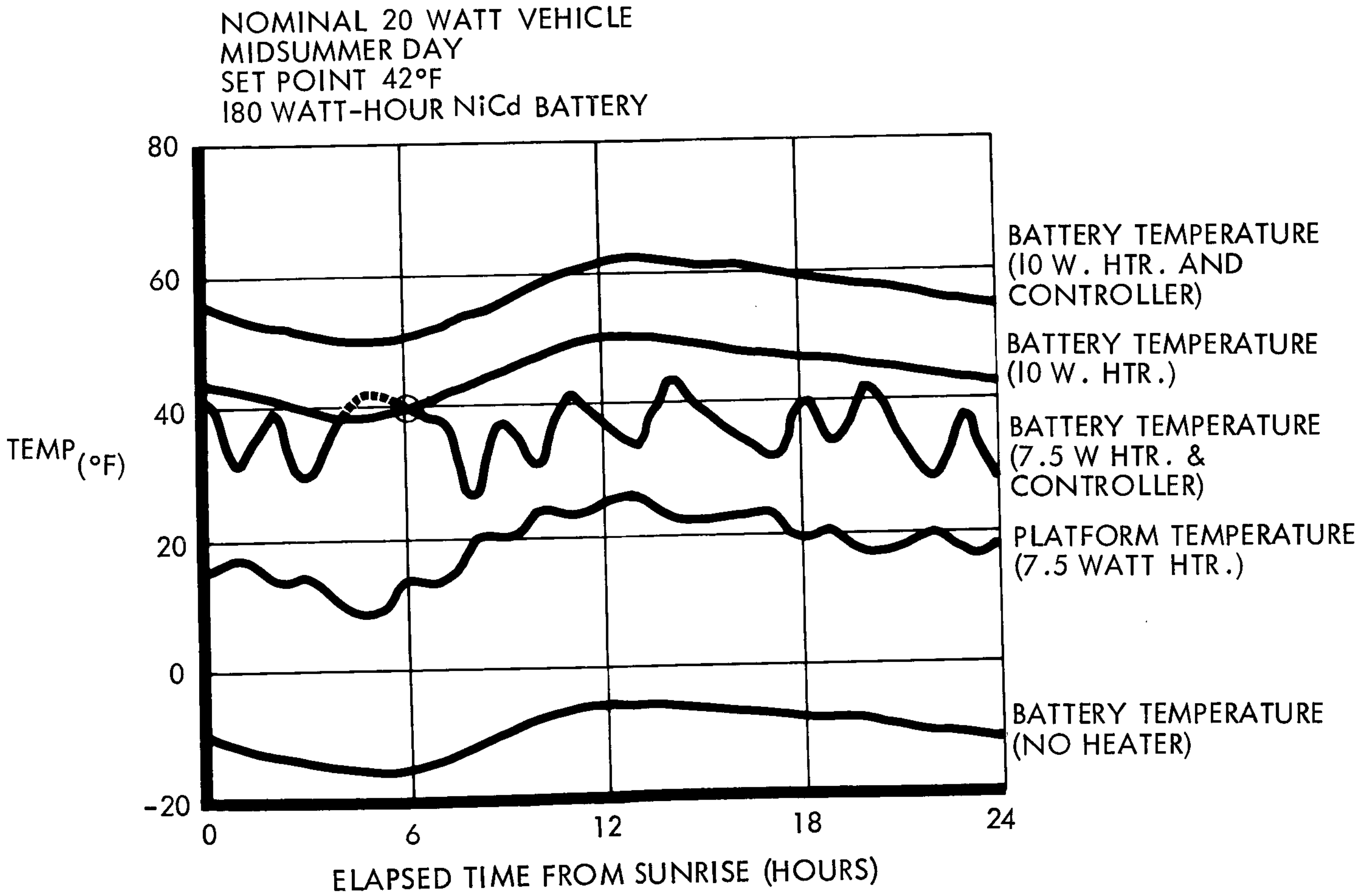


Figure 1-4. Mars Lander Battery Model Performance Using Insulated Battery With Active Thermal Controller And Steady-State Heat Source

available to thoroughly examine the long-term, continuous-duty design concept which was proven to be theoretically feasible. The long-term storable design could not be proven feasible because the technology required to investigate the design in detail is not available.

To summarize, the study has shown that the use of standard, available, sealed, space batteries with heat sources in extremely low temperature environments is feasible and is also compatible with near-Earth high-temperature environments. Computer models of basic design concepts are defined in detail in Appendix C (the final input program for the computer models has been submitted to NASA Lewis Research Center under separate cover). It can be concluded from the study that there is a technological need for a heated low temperature battery by 1967 or 1968.

1.4 NEW TECHNOLOGY REQUIRED

The new technology required to provide the necessary background for designing a low temperature battery for a specific mission is presented in detail in Section 4. The study found the following technology to be required before a final flight design can be completed.

a) Long Term, Continuous Operation Battery

1. The radiation attenuation characteristics of various battery geometrics and materials are not available for analysis and comparison with acceptable spacecraft-instrumentation tolerance levels.

2) Thermal control has not been developed for providing active control of battery temperature in cycling or changing environments.

3) Battery packaging concepts capable of structurally withstanding extremely low temperatures with nearly constant, room ambient internal temperatures have not been developed.

4) The design concept for a low temperature, continuous operation battery (developed as part of the study) will require a laboratory feasibility demonstration. The computer simulation feasibility demonstration was positive, but the assumed constants must be verified experimentally.

b) Long Period, Cold Storage, End of Mission, Warmup Battery

1) The effects of very low temperature storage on batteries are not adequately known to determine feasibility of this design concept.

2) Short term, warmup devices which are compatible with batteries and which can be used at low temperature have not been developed. Such devices must heat a battery from a low temperature (-29 to -184° C) to the operating temperature (4 to 38° C) in 4 to 8 hours uniformly and without high battery temperature gradients. The devices must be capable of reliable initiation and control to -184° C.

3) Active control of heat and battery temperature when internal dissipation or environmental temperatures cause the temperature of the battery to rise beyond its upper operating temperature limit is not available. This technology is missing for both types of battery systems and for electronics packages in general.

4) No technology on the performance of this system concept is available. Computer models cannot be developed until the basic technology above has been developed in order to provide basic assumptions for analysis. The final system concept technology must be acquired before a flight design can be initiated in order to define interfaces, performance, and design limits.

The acquisition of the basic technology discussed above is prerequisite to proceeding further with low temperature battery designs for future missions. The development program recommended for acquisition of the necessary technology is summarized in the following paragraph of this report.

1.5 RECOMMENDED PROGRAMS TO DEVELOP NEW TECHNOLOGY

The programs recommended to develop the new technology necessary for future design and application of heated low temperature batteries for space missions are defined in Section 4.

The programs recommended are outlined below:

a) Long Term, Continuous Operation Battery

1) Constant heat source radiation and configuration studies using batteries as attenuators.

    2) Development of constant heat source.

    3) Development of thermal controller for heat rate control.

    4) Studies of material compatibility for battery packaging.

    5) Fabrication of prototype battery and test for continuous operation in cycling and/or steady-state low-temperature environments.

b) Long Term, Cold Storage, End of Mission, Warmup Battery

    1) Investigation of effect of low temperature storage and warmup on standard battery designs.

    2) Development of a storable low temperature battery with warmup capability.

    3) Development of a thermal controller for heat rate control and distribution.

    4) Fabrication of prototype battery and test for low temperature storage and end-of-mission warmup in a low temperature environment.

The schedule for development of new technology required to meet projected need dates is shown in Table 4-2. The design goals for the programs should be completion of prototype evaluation by the end of 1967 for the storable battery and the end of 1968 for the continuous operation battery.

2. LITERATURE SEARCH AND INDUSTRY REVIEW

During Phase I, survey trips to various industrial and Governmental organizations were made to obtain the latest data available as related to low temperature battery and heater technology. Appendix A and the classified Appendix D contain a listing of these trips with a brief summary of the results obtained. Details of the information pertaining to this study is discussed within the appropriate technical sections.

The literature search conducted was aimed at four main areas of study--1) battery technology, 2) battery heating methods, 3) thermal control technology and 4) published space mission requirements. A bibliography of those documents relating to this study is given in Appendix B.

The results of the literature search and industry survey are given in the following paragraphs.

2.1 BATTERY TECHNOLOGY

A literature search and analysis of the state-of-the-art of battery performance as a function of temperature was conducted. The results of this work established the typical ranges of operation which a heated battery would have to meet in order to operate satisfactorily in low temperature space environments. A brief discussion of theoretical effects of temperature on batteries and a detailed discussion of different types of batteries, battery characteristics, and heater compatibility follows.

2.1.1 Temperature Effects on Electrochemical Battery Systems

Temperature has several independent effects upon the operation of any electrochemical battery system. The effects are divided, for convenience, into three categories:

Effect upon emf
Effect upon electrolyte conductivity
Effect upon electrode kinetics

2.1.1.1 Temperature Effect upon EMF

The effect of temperature upon emf is accurately expressed by the following thermodynamic relationship (Reference 1) :

$$\Delta S = -N_e F \left(\frac{\partial E_o}{\partial T} \right)_P \quad . \qquad (1)$$

This relationship is used in the determination of entropies of reaction. In most working battery systems,

$$\left(\frac{\partial E_o}{\partial T} \right)_P$$

is found to be negative, so that a reduction in temperature results in small increases in emf .

2.1.1.2 Temperature Effect upon Electrolyte Conductivity

The effect of temperature upon electrolyte conductivity has been treated theoretically for dilute aqueous solutions. Although some effort has been made to extend this treatment to concentrated solutions, theoretical predictions of conductivity have not been accurate.

An approximate empirical relationship (Reference 2) between conductivity of aqueous solutions and temperature which may be useful in obtaining estimates of temperature-induced changes in conductivity is:

$$K_T = K_{18^oC} \left[(1 + \beta(T - 18^oC) \right] \qquad (2)$$

where

K_T = Electrical conductivity at temperature T.

β = Constant approximately equal to 0.025 for bases and and 0.016 for acids.

This relationship holds only when the composition of the electrolyte remains unchanged. As the electrolyte approaches the freezing point, composition changes occur which cause further changes in electrolyte conductivity. Upon total freezing the electrolyte conductivity is, for all practical purposes, negligible.

2.1.1.3 Temperature Effect upon Electrode Kinetics

The effect of total electrolyte freezing upon the performance of the battery after it is reheated will vary with battery construction and type. In batteries not specifically designed to accommodate electrolyte freezing, physical deterioration of the cell cases, separators, and electrodes is likely to occur due to the expansion of water upon freezing. If large amounts of free electrolyte are available, freezing may rupture cell cases. If all of the electrolyte is immobilized in the separators and porous electrodes, freezing may result in perforation of the separators by ice crystals and will cause partial or total disintegration of the electrode structure. The extent of damage will vary according to the structural strength of the electrodes and the rate of freezing, which affects the size of crystals formed.

No test data could be provided to substantiate this hypothesis.

2.1.2 Thermal Characteristics of Existing Battery Systems and Battery/Heater Compatibility

Virtually all batteries display a variation in capacity, load voltage, and current delivery capability with temperature. This relationship is a function of the discharge rate, minimum acceptable end-of-charge voltage, and regulation required.

Temperature values used in this text are defined as follows:

a) Minimum Safe Temperature: The temperature at which the battery will deliver only negligible currents at normal operating voltages and below which permanent damage may occur to battery structure.

b) Minimum Practical Temperature: The temperature at which some capacity deterioration occurs at practical discharge currents. The battery still delivers more than half of its true capacity.

c) Minimum Recommended Temperature: The temperature at which negligible deterioration of capacity occurs at practical discharge currents.

d) Maximum Extended Temperature: The maximum temperature at which the battery may be stored or operated for long periods (days, weeks, etc.)

e) Maximum Safe Temperature: The maximum temperature at which the battery may be stored or operated for short periods of time (one hour or less).

2.1.3 Zinc-Silver Oxide Batteries

Zinc-silver oxide cells are sufficiently variable from type to type and from manufacturer to manufacturer that there is no single relationship between performance and temperature general enough to express the behavior of the cells.

These variations are due to differences in plate thickness, number of plates, separator layer thickness, number of separator layers, and quantity of electrolyte used. The differences in design and construction are due to differences in the performance, storage and cycle life, etc., required to accomplish specific missions.

In general, primary silver-zinc batteries are designed for short time use. By storing short life batteries in the activated condition better performance will be achieved in the low temperature region than will be achieved with silver-zinc batteries designed for longer storage and service life. However, the short-use batteries are more sensitive to over-temperature. Batteries designed for long cycle-life secondary use are usually constructed with multiple layers of separators and reduced in electrolyte content so that their performance at low temperature is degraded. Data on low temperature performance and high temperature deterioration are shown in Figures 2-1 through 2-3.

For a preliminary evaluation of the usefulness of manually-activated primary silver-zinc batteries in a specific thermal environment, the following data are presented with the understanding that these are representative of only one of the many types of silver-zinc batteries.

	°F
Minimum Safe Temperature	-65
Minimum Practical Temperature	0
Minimum Recommended Temperature	+50
Maximum Extended Temperature	+100
Maximum Safe Temperature	+165

For accurate information about specific battery properties, it is necessary to test specific batteries with a knowledge of the time-temperature and service-life profile since thermal degradation factors are cumulative with time.

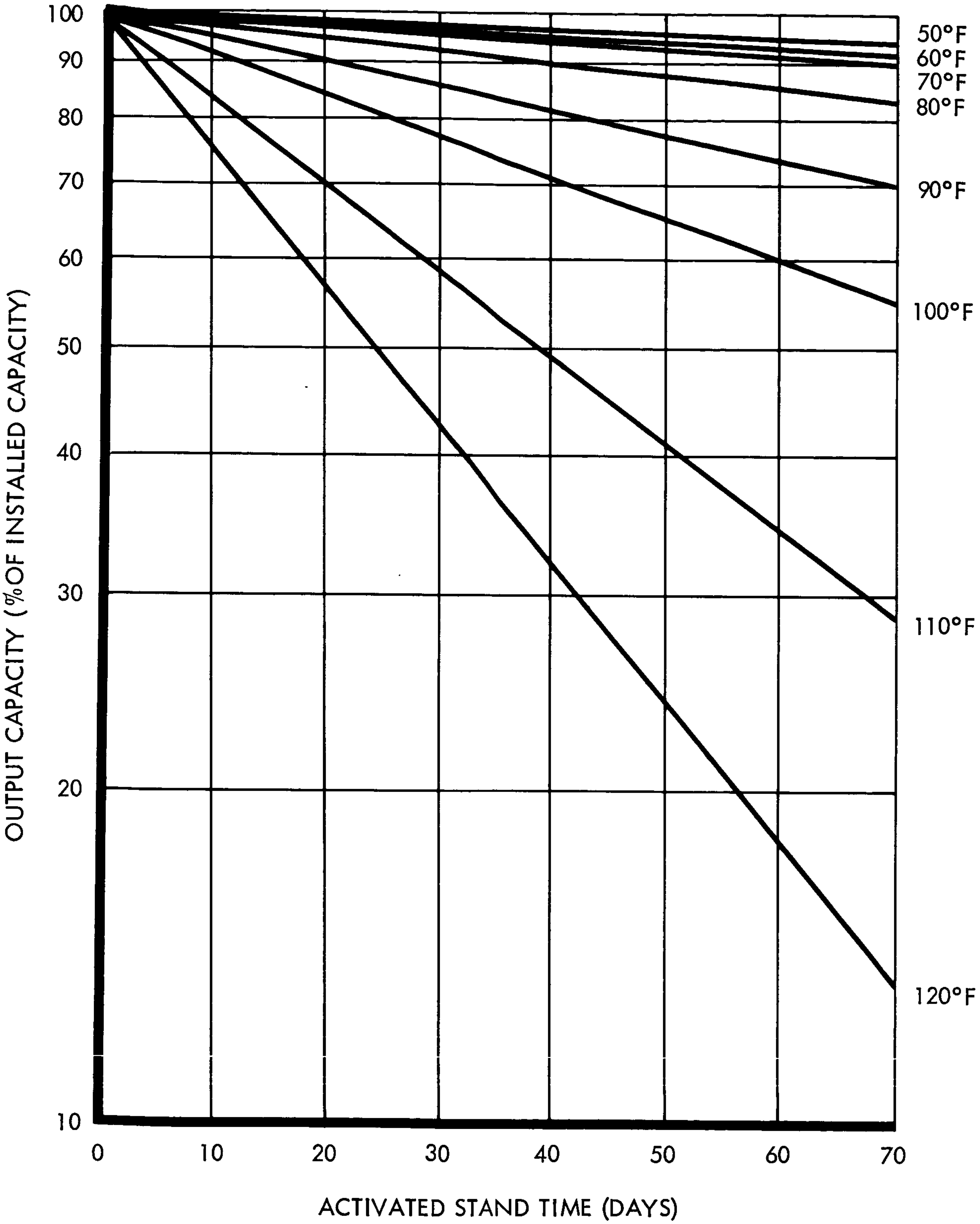


Figure 2-1. Silver-Zinc Primary Battery Capacity Available as a Function of Activated Stand Temperature and Time (1963 Capability - ESB DATA (Unpublished)

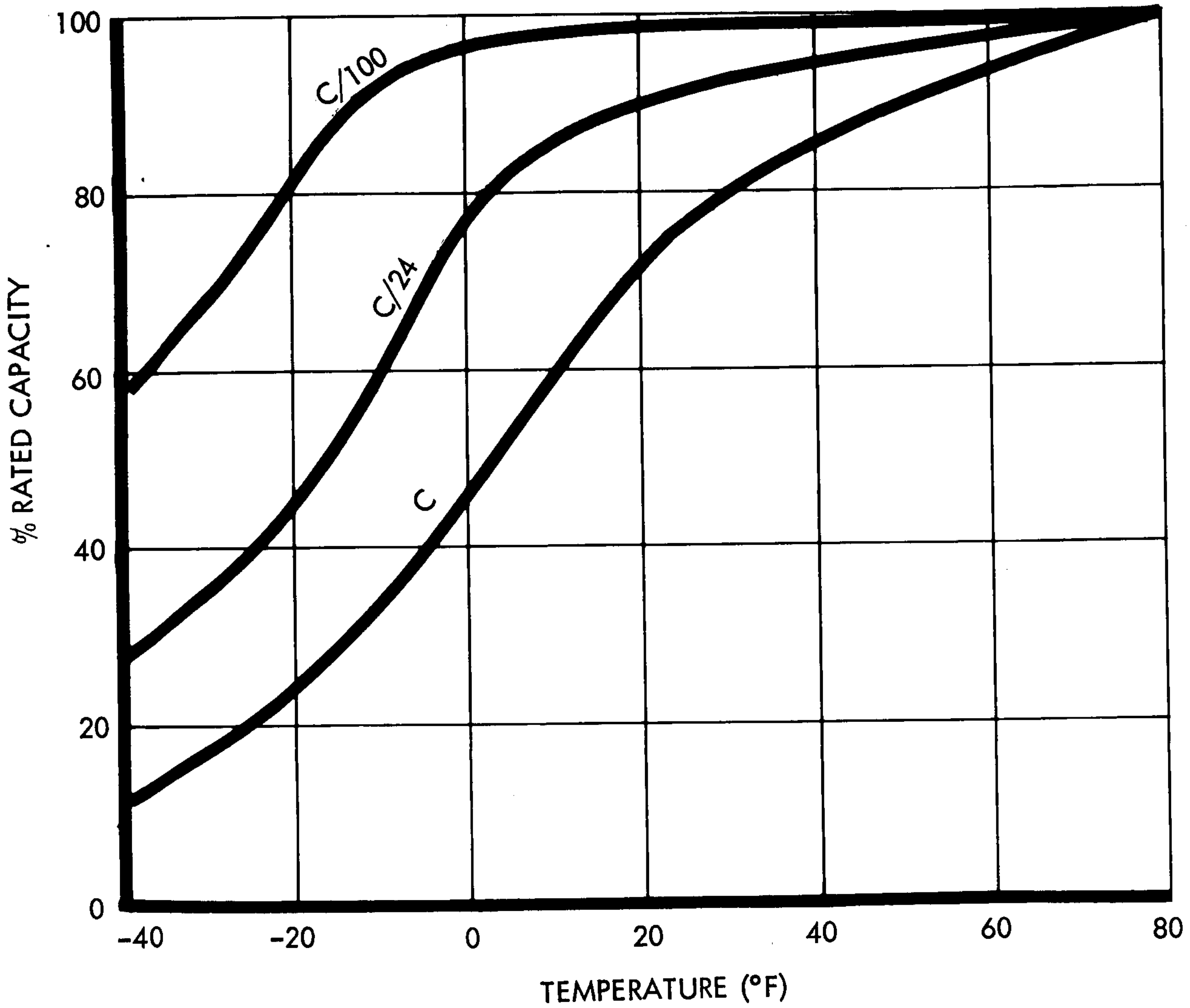


Figure 2-2. Silver-Zinc Cells, Effect of Temperature on Capacity to 1.0 Volt (Discharge Rate is the variable parameter)

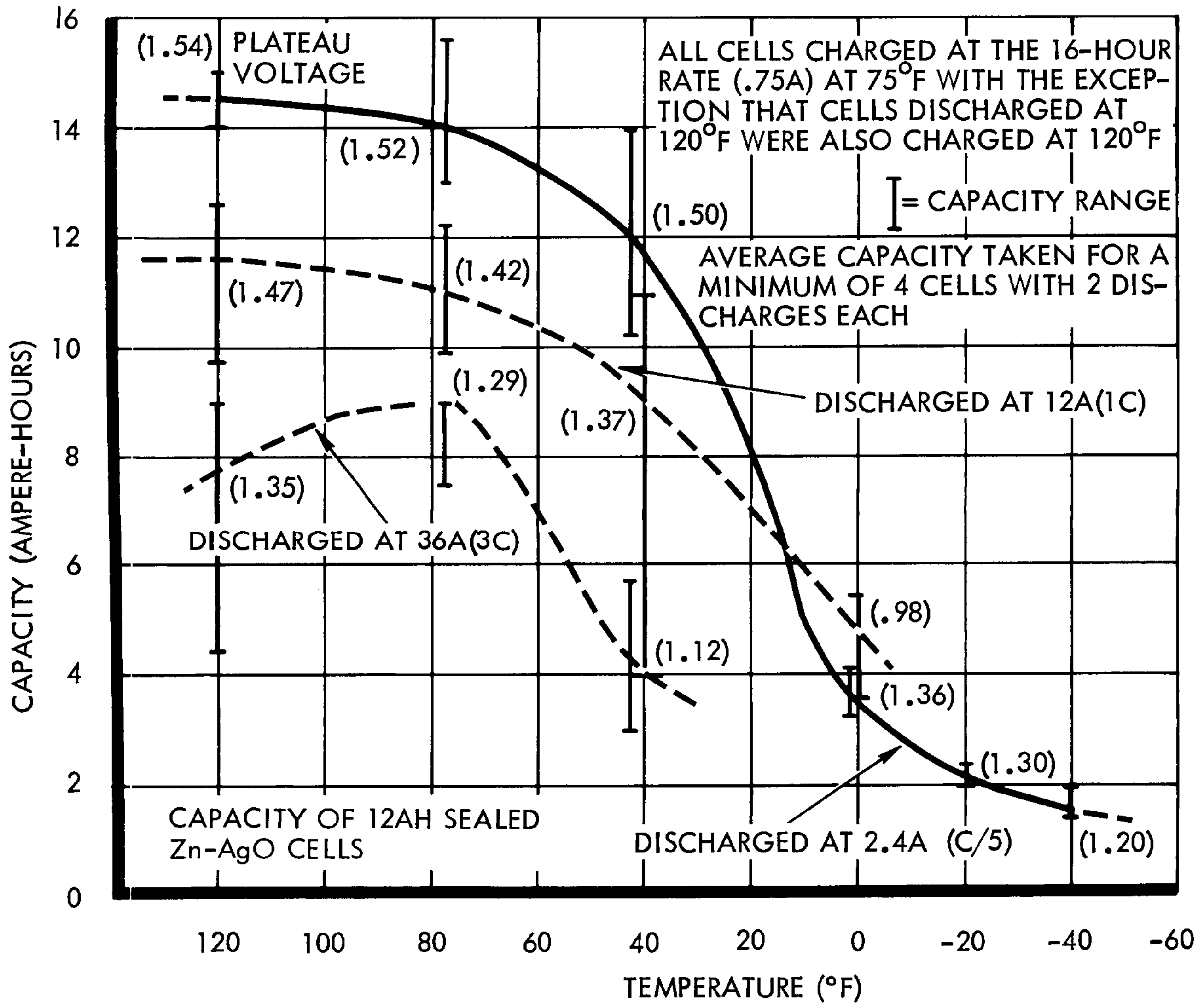


Figure 2-3. 12AH Sealed ZnAgO Cell Capacity as a Function of Discharge Rate and Temperature.

2.1.3.1 Mechanisms of Deterioration

There are several deterioration mechanisms in silver-zinc battery cells.

2.1.3.1.1 Solubility of Zinc Oxide (Reference 3). As the battery is discharged, the negative plate is converted from zinc metal to zinc oxide which in turn dissolves in the potassium-hydroxide electrolyte to form solutions of potassium zincate. These solutions readily become supersaturated, particularly on subsequent cooling, and the zinc oxide or potassium zincate may tend to precipitate out in areas removed from the negative plate. Carried away from the negative electrode, this material, is unavailable on recharge for reconversion to zinc metal and a loss in capacity of the negative plate results. In addition, the large concentration of dissolved potassium-zincate provides raw material for growth of whiskers of zinc which may short circuit the battery. In cells designed for long cycle life, the total quantity of electrolyte is reduced to minimize the removal of active material from the negative plate, which degrades low temperature performance.

2.1.3.1.2 Solubility of Silver Oxide (Reference 4, 5, and 6). Silver oxide from the positive electrode is soluble and tends to be removed from the positive plate. If allowed to reach the negative plate or certain insulators, layers of metallic silver may be deposited causing possible shorting of the electrodes.

Advantage is taken of the oxidizing nature of the dissolved silver ions to remove the silver from the electrolyte by interposing multiple layers of cellophane or other separator materials between positive and negative plates. The cellophane reacts with the dissolved silver ions to form metallic silver which is deposited in the cellophane. The presence of the cellophane does, however, decrease ion mobility; thus, separator materials which physically impede the transport of silver may also be used, usually in combination with reactive separators.

2.1.3.2 Methods of Heating

Silver-zinc batteries of the manually activated type have been heated electrically with strip or blanket heaters, which are powered either externally or with power from the battery itself.

Automatically-activated batteries have been heated during activation by incorporating a heat exchanger in the activation mechanism, through which heat generated in the pyrotechnic gas generator is transferred to the electrolyte immediately prior to dispensing it into the cells (Reference 7).

Pyrotechnic or rapid chemical heating of activated silver-zinc batteries by external application of heat has not been reported.

2.1.4 Silver Oxide-Cadmium Batteries (References 8 and 9)

The silver-cadmium system has a lower voltage and a higher chemical stability than the silver-zinc. This system is normally used only as a long life, secondary battery since the silver-zinc primary system shows considerable weight advantages over the best silver-cadmium battery.

This systems basic construction is similar to that of the silver-zinc system, utilizing plates of various thicknesses depending upon multiple layers of cellophane or other cellulosic materials for removal of dissolved silver from the electrolyte and for prevention of migration of particulate material between electrodes. Because of the similar construction and electrolyte composition, the deterioration rate due to sensitivity to temperature is also similar to that of the silver-zinc cell deterioration rate; i.e. cell performance increases with higher temperature and decreases at lower temperature.

For applications in which the silver-cadmium must be cycled rapidly (i.e., charged and discharged within a relatively short time), the performance of the battery suffers greatly due to reduction in temperature. Because of the batterys inability to accept high-rate charge at low temperatures without developing terminal voltages so high, the danger of excessive gas evolution exists. Figures 2-4 through 2-9 show the performance of silver-cadmium cells over a range of temperatures in which the batteries were charged and discharged at moderate rates at various temperatures. The loss in capacity is primarily due to the inability to charge at safe voltages. Even at room temperature, silver-cadmium cells may show a serious deterioration in charge acceptance due to the use of excessive currents. As a

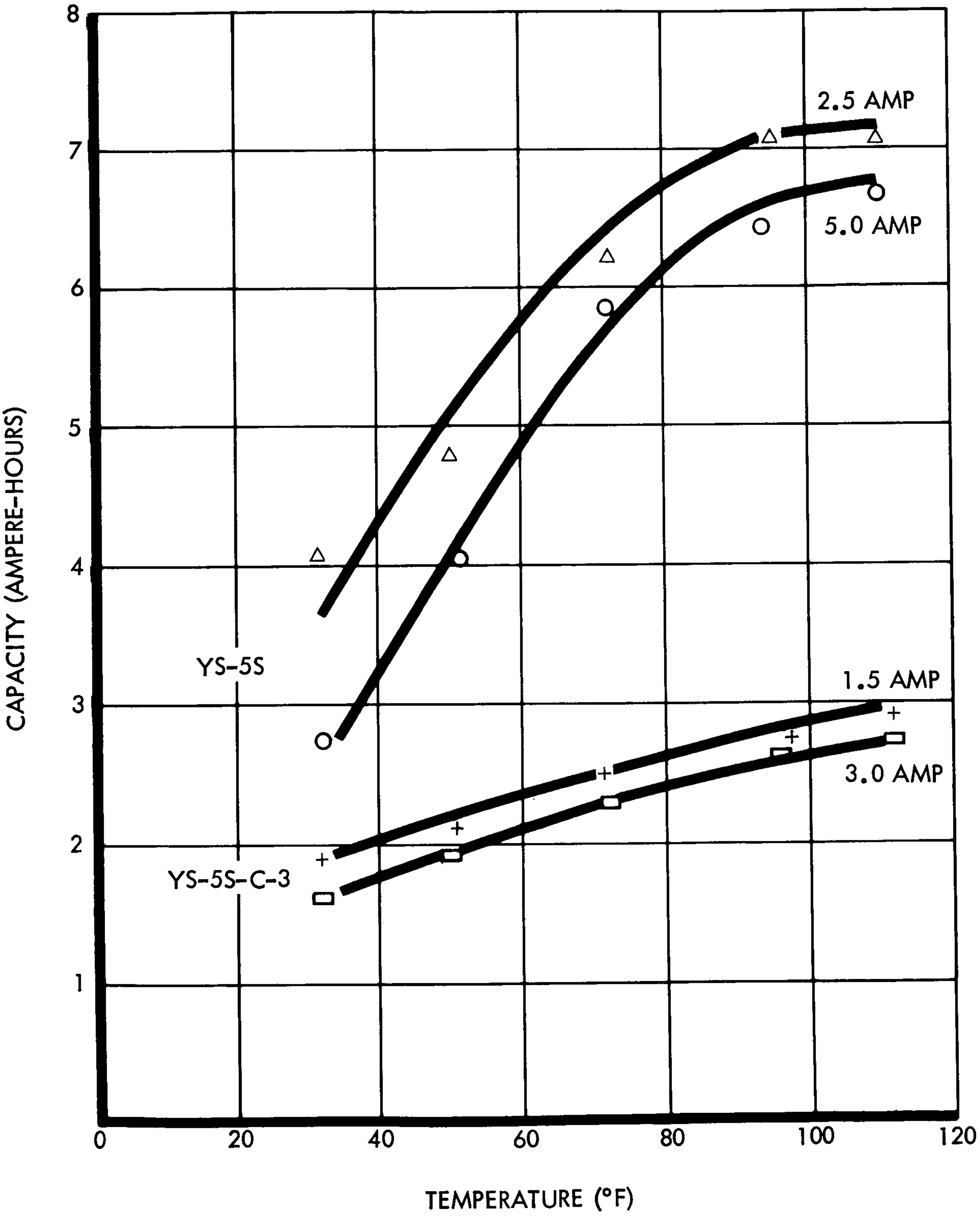


Figure 2-4. Silver-Cadmium Cell Capacity in Ampere-Hours versus Temperature (Discharge Current Is the Parameter)

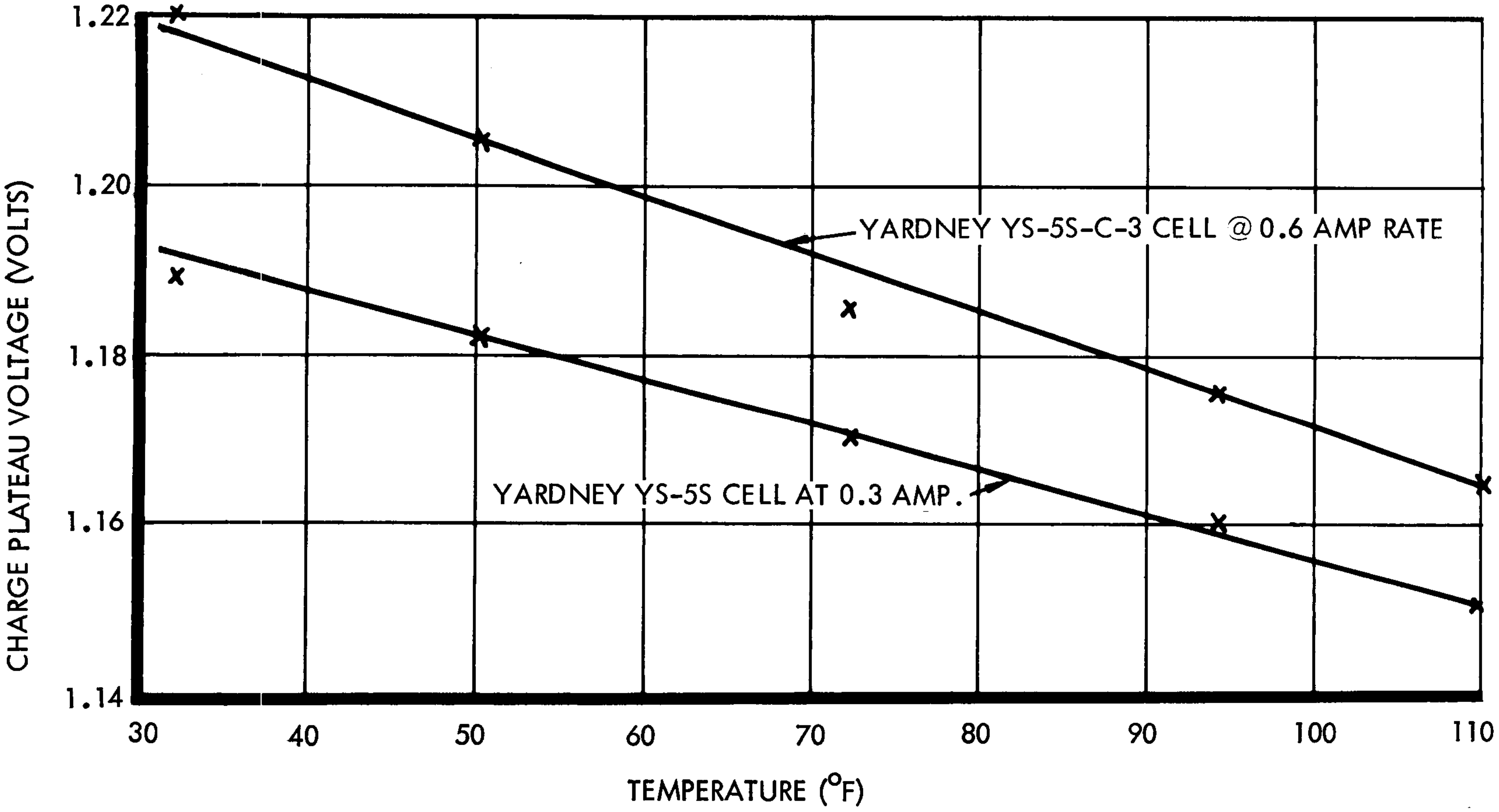


Figure 2-5. Silver-Cadmium Cells, First Charge Plateau Voltage versus Temperature

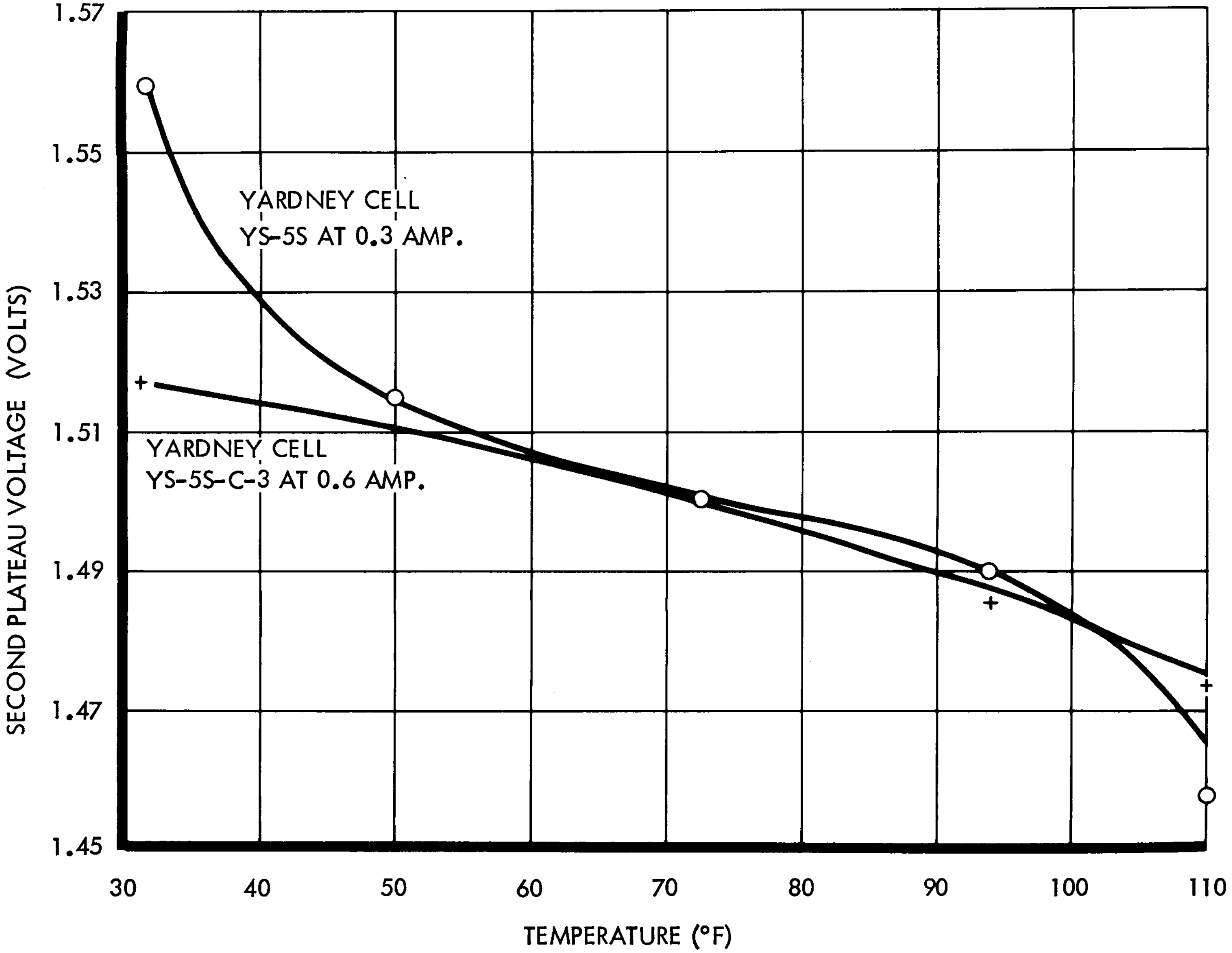


Figure 2-6. Silver-Cadmium Cells, Second Charge Plateau Voltage versus Temperature

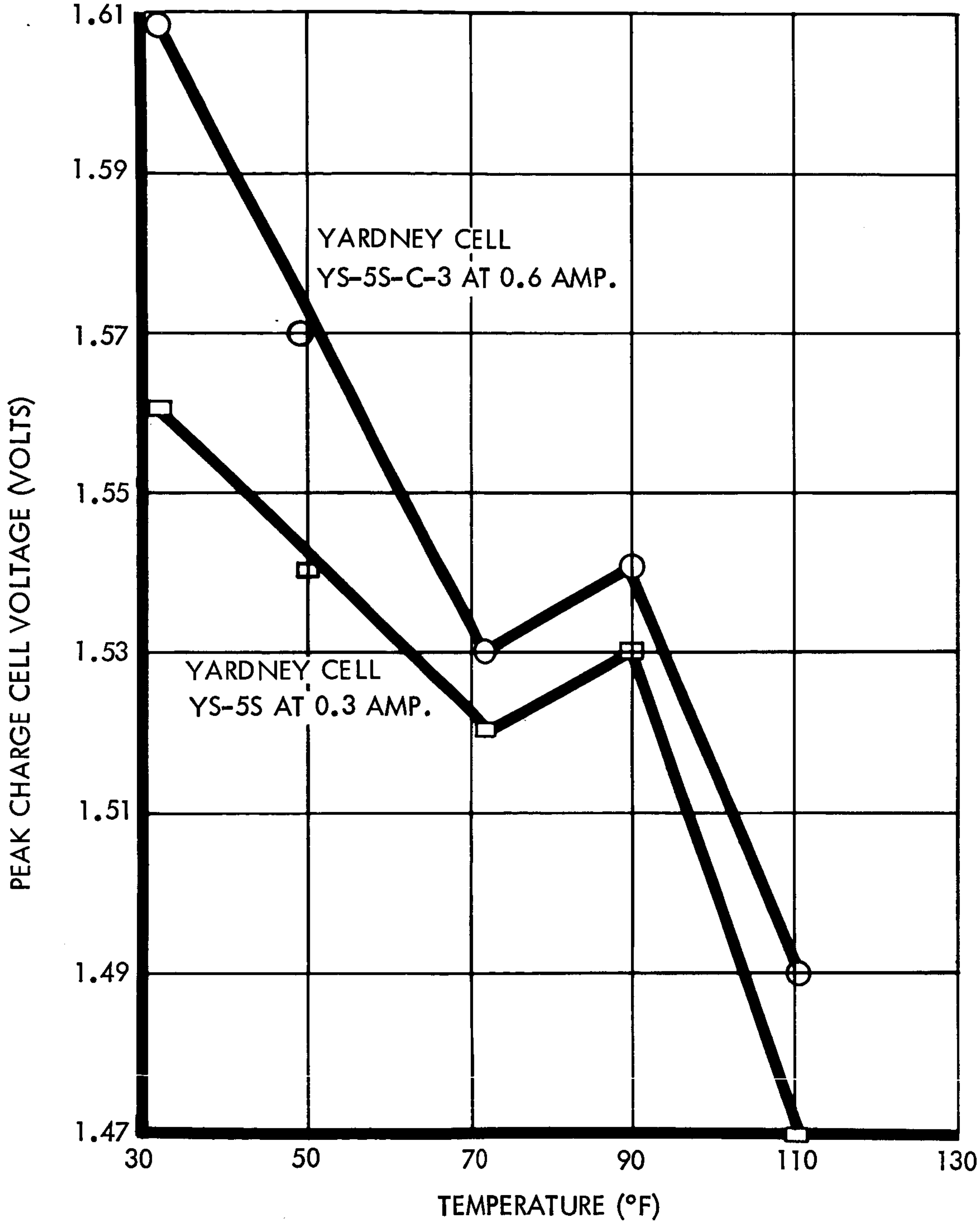


Figure 2-7. Silver-Cadmium Cell, Peak Charge Voltage versus Temperature

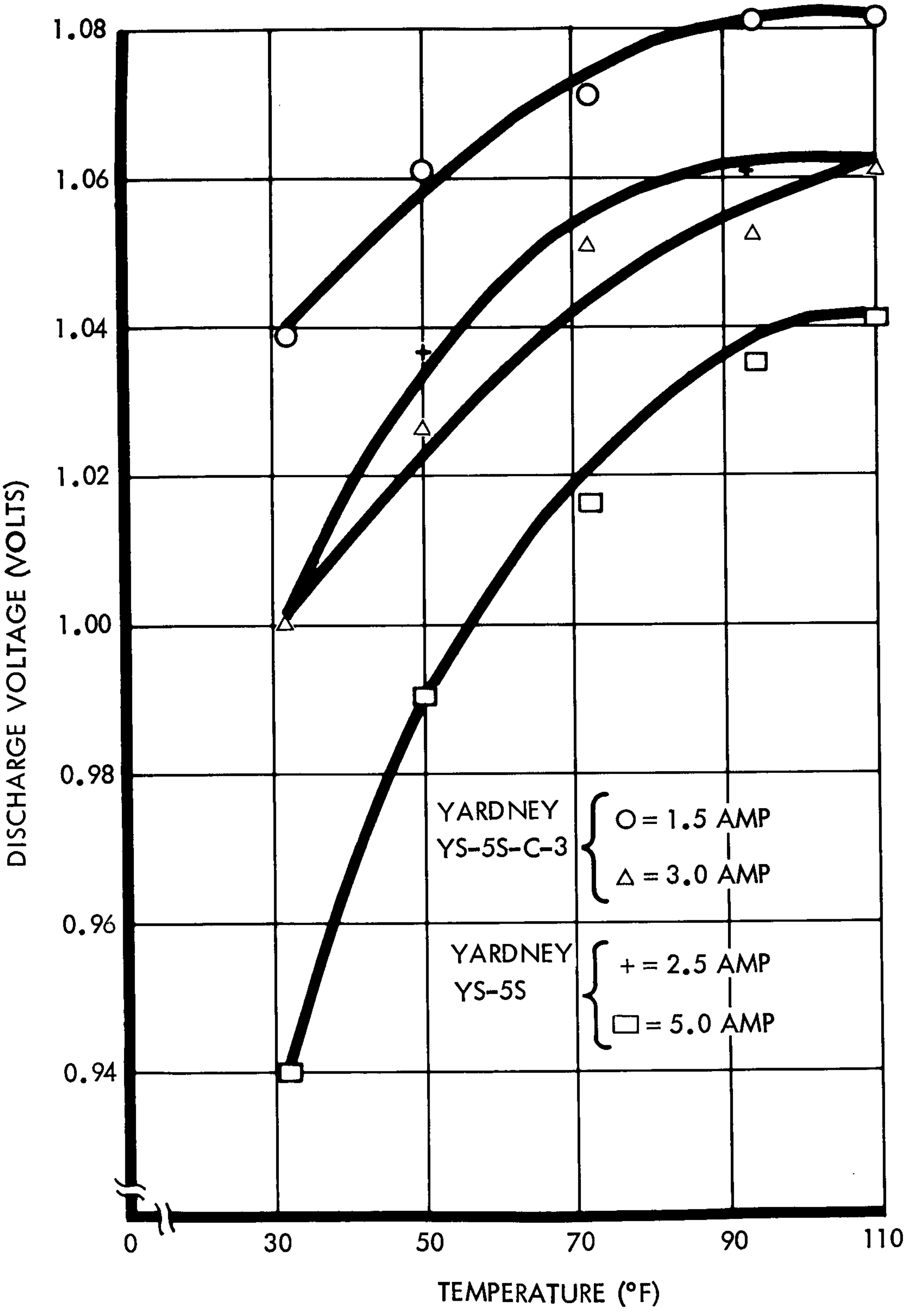


Figure 2-8. Silver-Cadmium Cells, Discharge Voltage Plateau versus Temperature

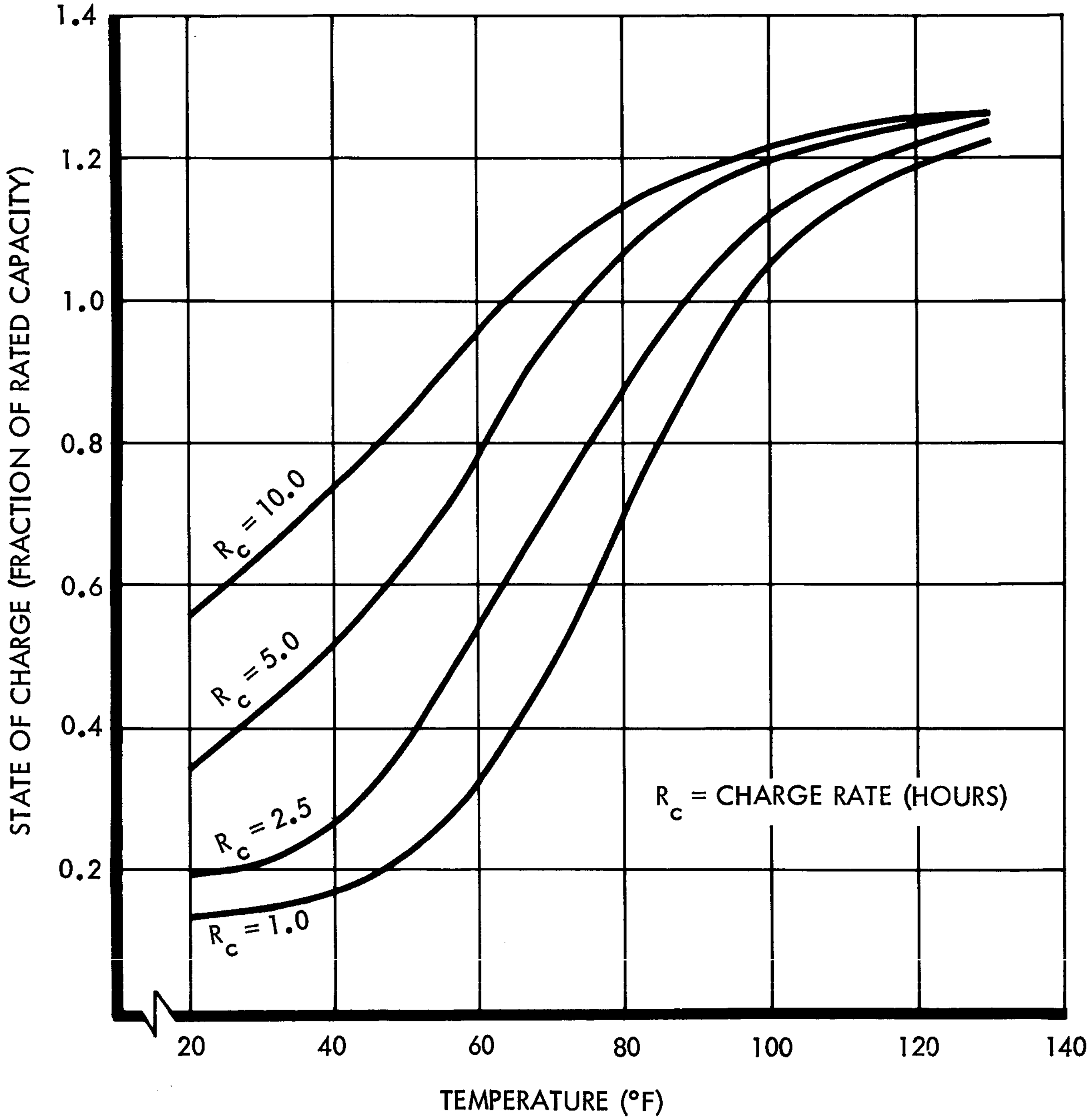


Figure 2-9. Silver-Cadmium Cells. Maximum State of Charge versus Temperature (Charge Rate is the Parameter.)

result, the minimum acceptable environmental temperature is very strongly dependent upon the charge currents required. By charging slowly over long periods, discharge performance at low temperatures may be optimized.

For a preliminary evaluation of silver-cadmium battery applicability at commonly-used charge and discharge rates (C/10 charge and C/2 discharge), the following temperature limits are suggested:

	°F
Minimum Safe Temperature	-65
Minimum Practical Temperature	+20
Minimum Recommended Temperature	+50
Maximum Extended Temperature	+100
Maximum Safe Temperature	+165

Heating methods which are acceptable for use with silver-cadmium cells will depend to a great extent upon the cell case construction, since this type of cell is available in both plastic and metal cases with ceramic seals. In general, the plastic-cased cells are not practical for use with high rate heating devices.

2.1.5 Nickel-Cadmium Batteries (Reference 10)

The nickel-cadmium system has a higher overall plateau voltage than the silver-cadmium lower plateau (1.2 volts versus 1.0 volt) This system, based upon present methods of construction, has a lower practical energy-density.

This systems performance at normal discharge rates (in the C/5 to C range) is best in the range of 15 to 25°C, and falls at lower temperatures due to: 1) increased polarization, 2) heat-induced spontaneous decomposition of the tri-valent nickel-oxide at higher temperatures and 3) loss of charging efficiency at elevated temperatures. At very low discharge currents, the system will give useful capacity at temperatures as low as -40°C. Figures 2-10 through 2-14 show the variation in performance with temperature, based upon a voltage-limited charge at constant current.

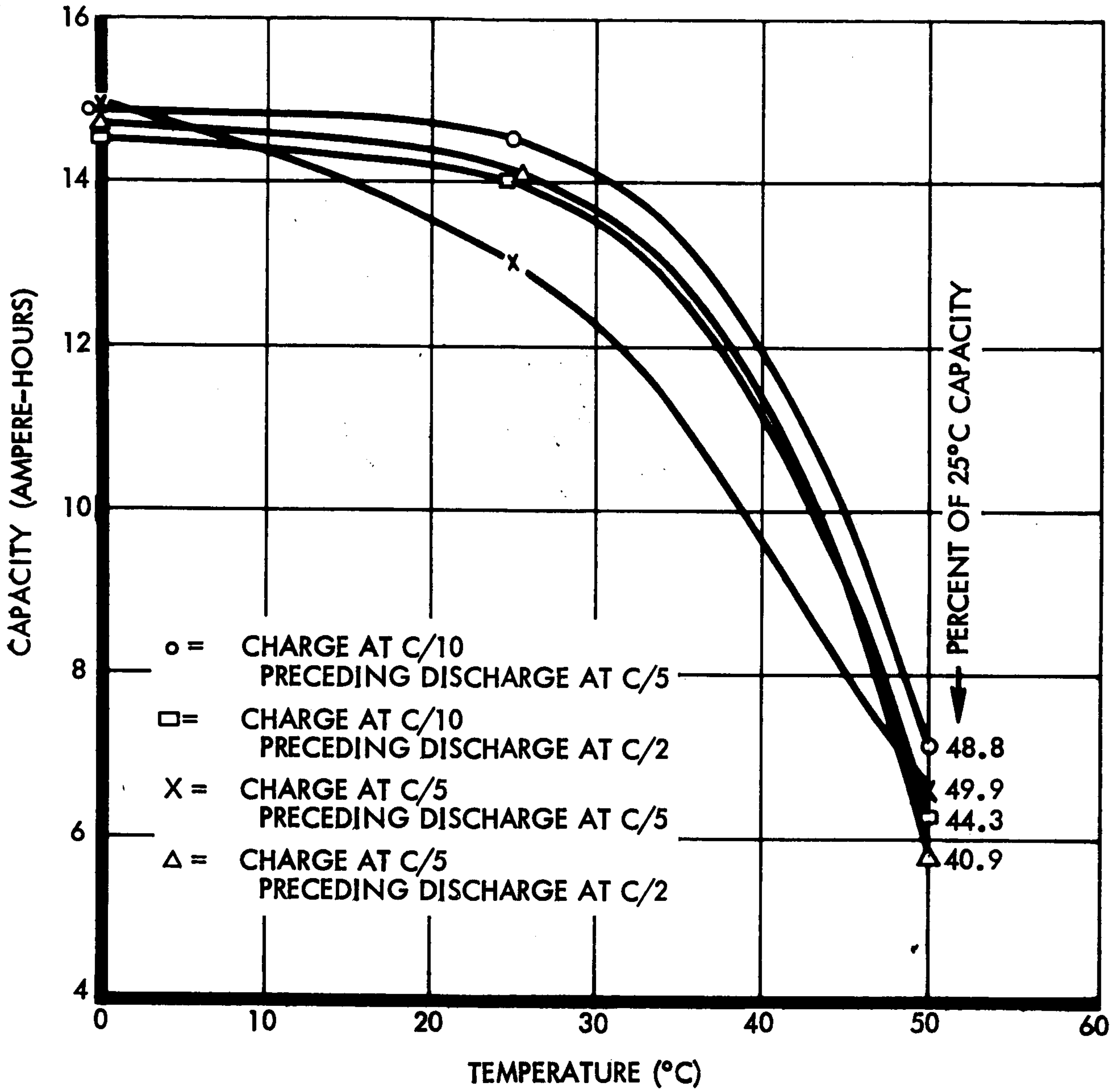


Figure 2-10. 12AH Nickel-Cadmium Cells, Average Cell Output versus Temperature

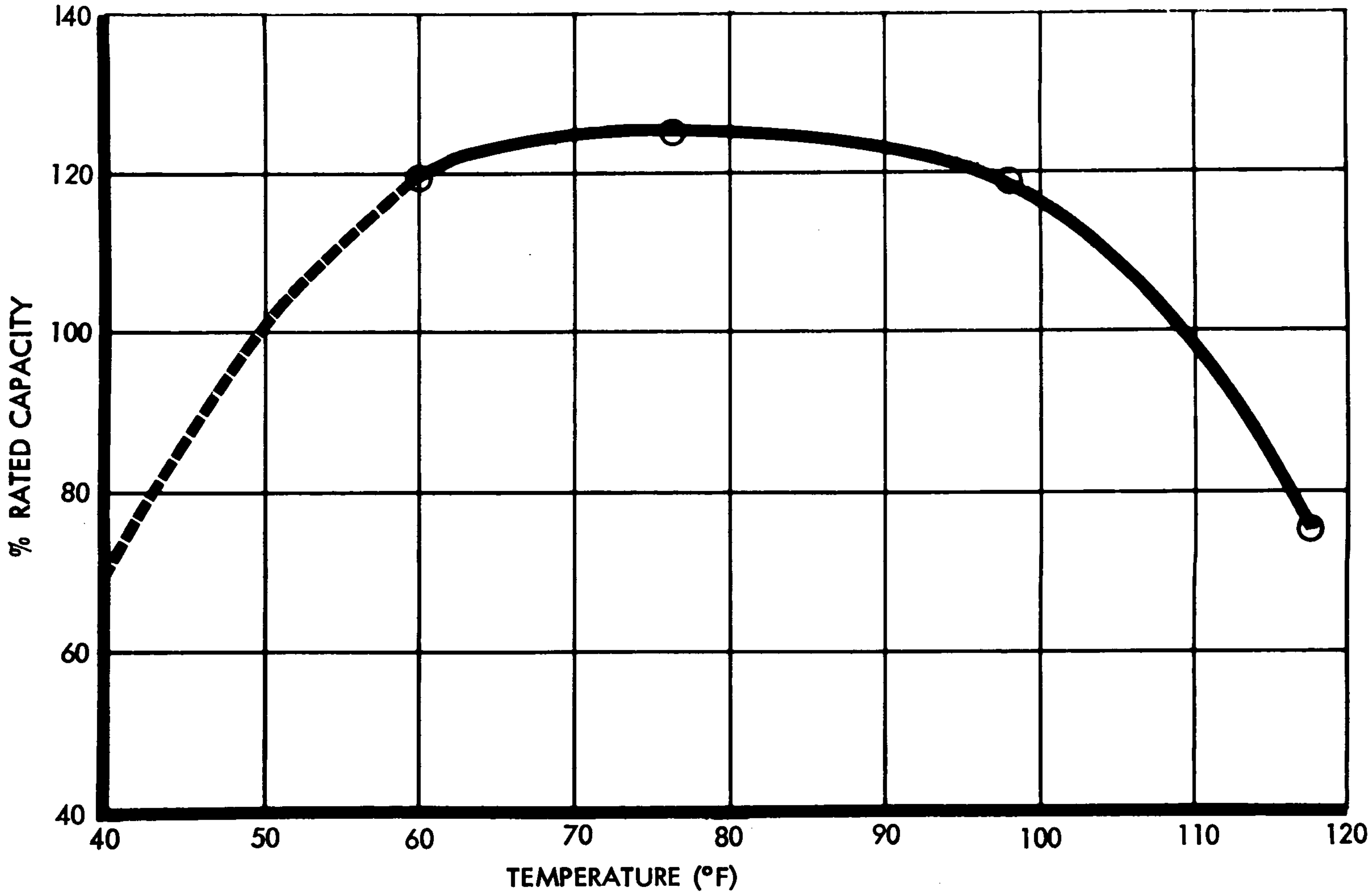


Figure 2-11. 20AH Nickel-Cadmium Cells, Available Capacity versus Temperature at c/3 Discharge Rate

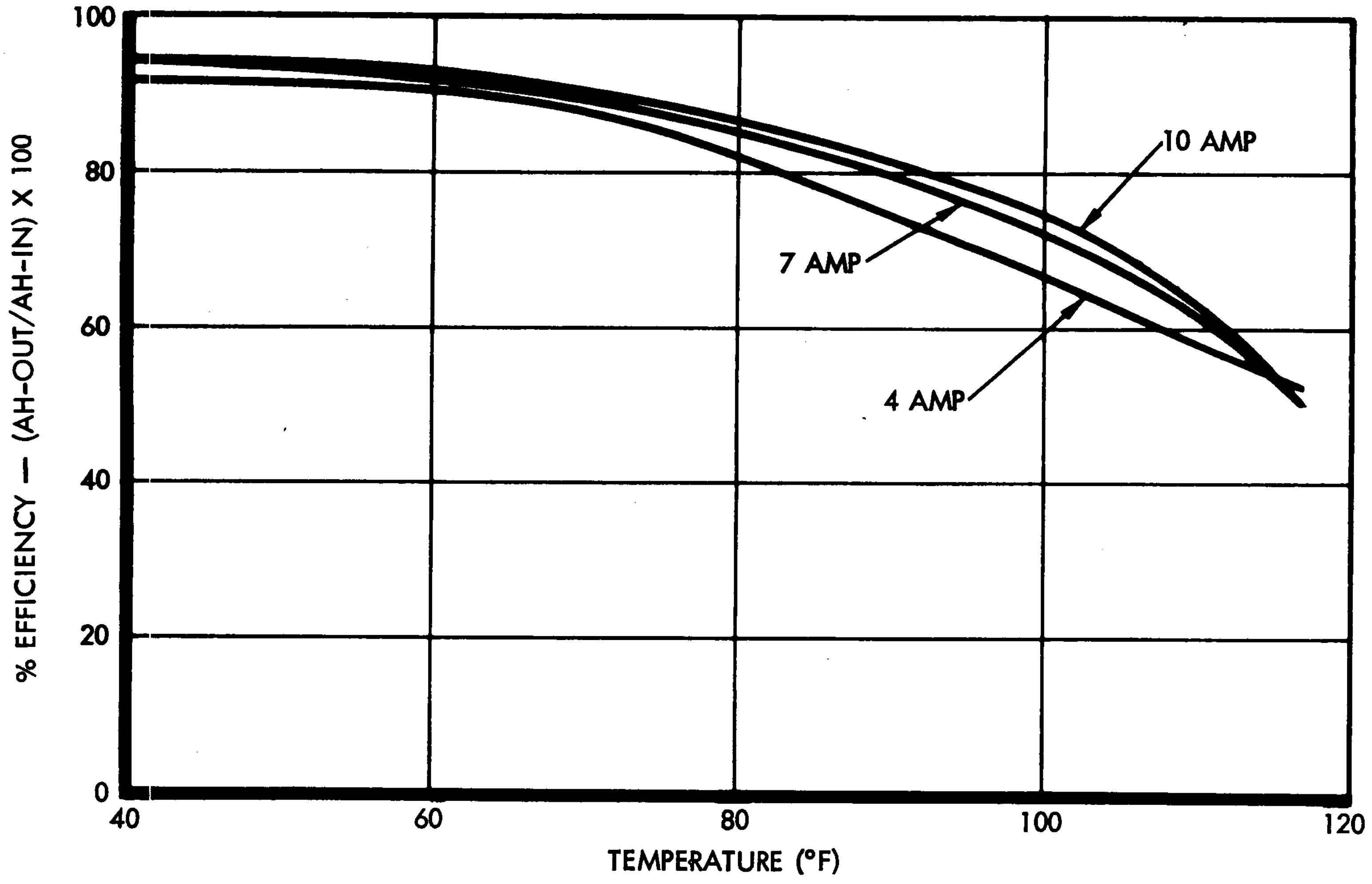


Figure 2-12. 20AH Nickel-Cadmium Cells, Overall Storage Efficiency versus Temperature (Charging Current Is the Reference Parameter, Using a Voltage-Limited Charge.)

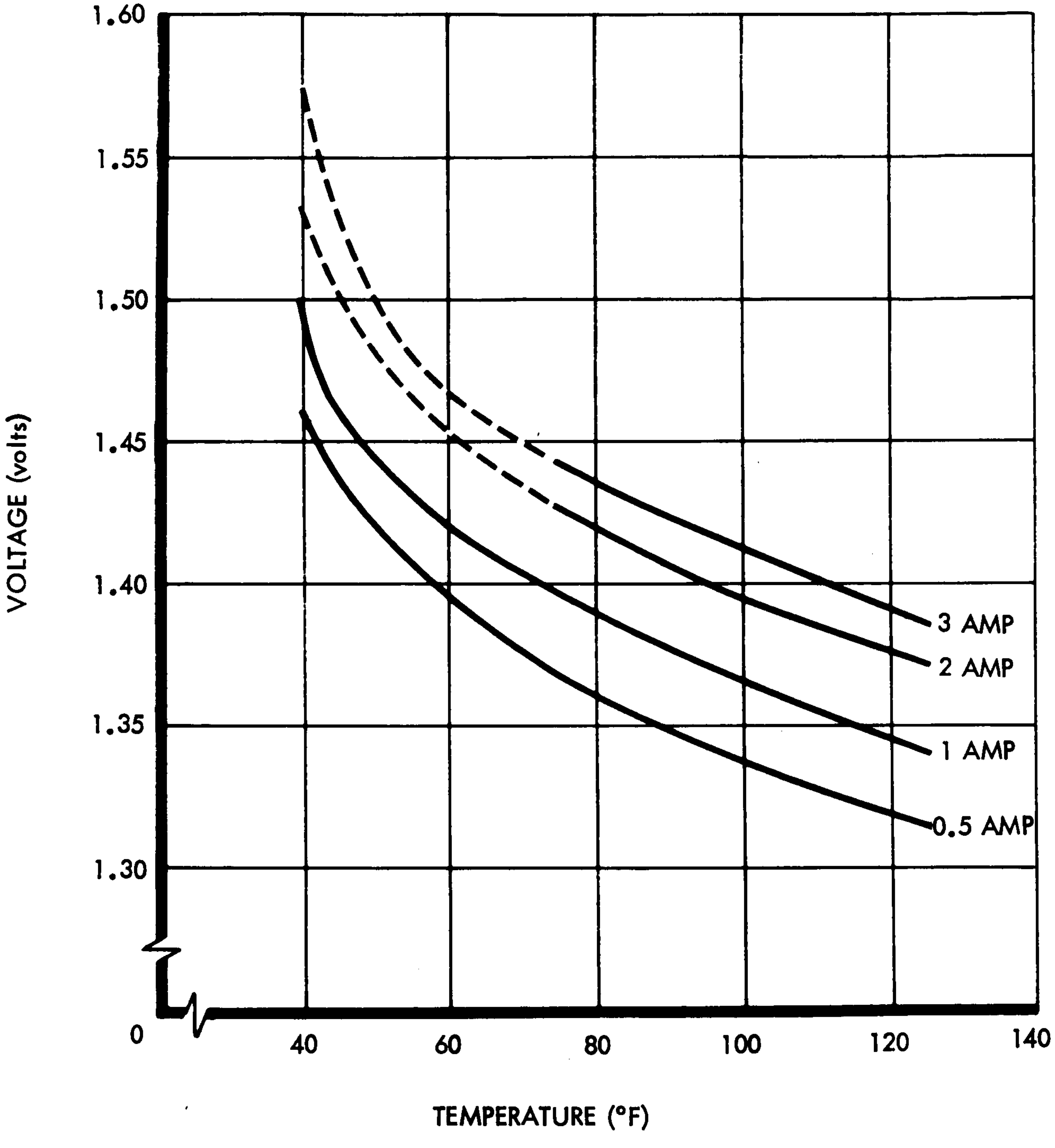


Figure 2-13. 20AH Nickel-Cadium Cells, Voltage versus Temperature at Constant Steady-State Overcharge Current

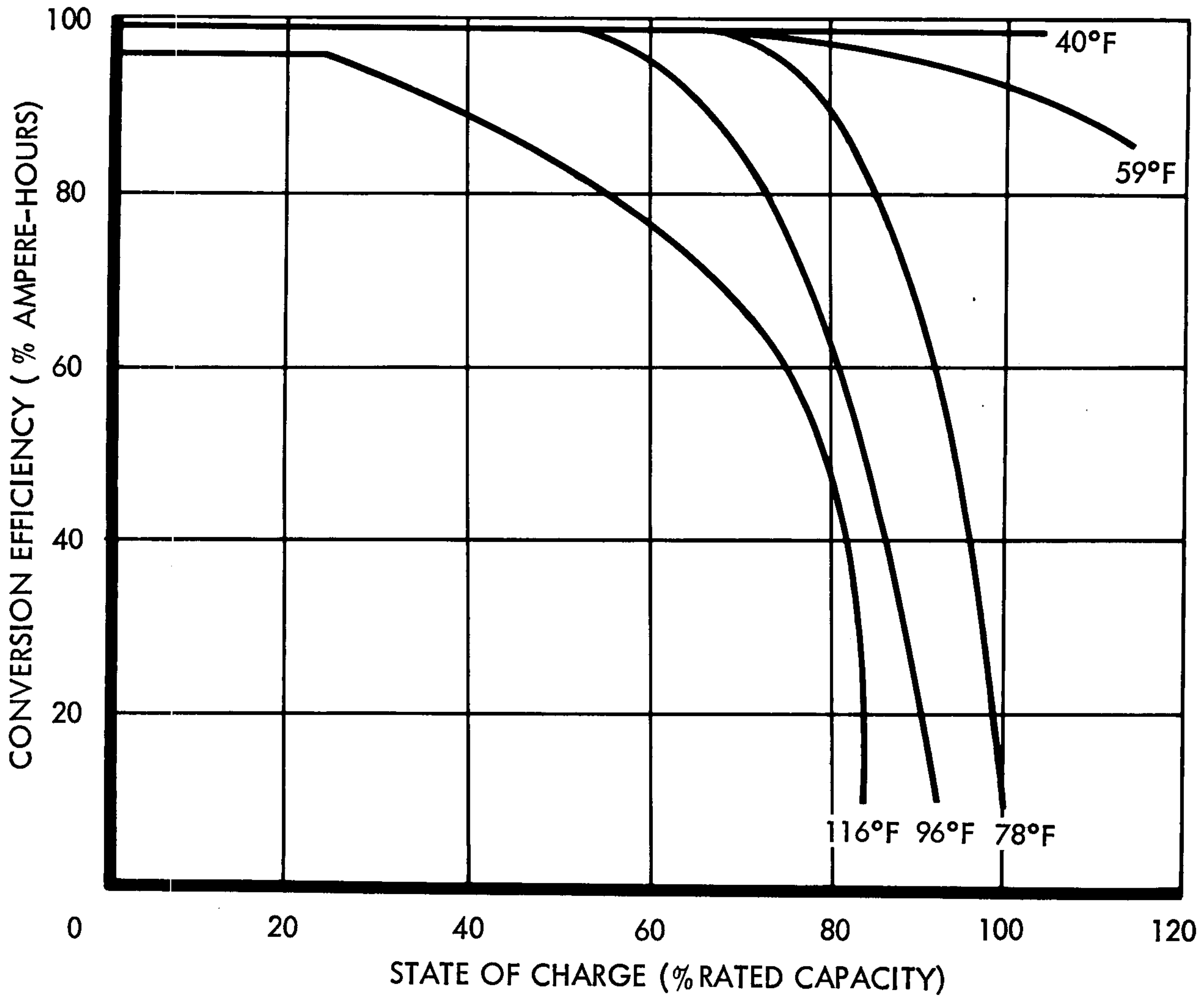


Figure 2-14. 12AH Nickel-Cadmium Cells, Storage Efficiency versus State of Charge

Safety during the charging process in a sealed, electrolyte-starved, nickel-cadmium cell is dependent upon the ability of the cell to accept an overcharge at a potential below that at which hydrogen gas is generated. The current acceptance of the cell at any voltage falls with decreasing temperature as shown in Figure 2-15, and the survival of the cell at low temperatures is dependent upon the suitability of charge control devices and the amount of time available for charging.

2.1.5.1 Heating and Temperature Control

No extensive work has been reported in heating nickel-cadmium batteries although they are readily heated by means of environmental control devices, such as electric heating blankets. Since most sealed nickel-cadmium cells for aerospace use are packaged in ceramic-sealed, welded, stainless steel containers, they are capable of withstanding local external overheating for short periods better than the plastic-cased silver-zinc and silver-cadmium cells.

Batteries have been constructed in which the distribution of heat from cell to cell and throughout the battery structure is controlled by lightweight heat conductors built into the battery structure, such that with a 25-watt heat throughout, a differential of less than -5.6° C can be measured between the hottest and coldest external areas of the battery.

2.1.5.2 Deterioration Mechanisms

Deterioration of capacity in nickel-cadmium cells is poorly defined but appears to be associated with microstructural changes in the active materials of the electrodes.

Catastrophic short-circuit failures have been observed due to: 1) shedding of granular nickel from portions of the electrodes which puncture the separators, 2) a process by which the brazing allow of certain types of cells is attacked and redeposited, short circuiting the ceramic terminal insulators and 3) chemical attack of the separator by the electrolyte in the presence of an oxidizing agent (Reference 11). The temperature at which the latter reaction occurs is raised by the introduction of nylon and polypropylene separators which are more resistant than the original cellulose type.

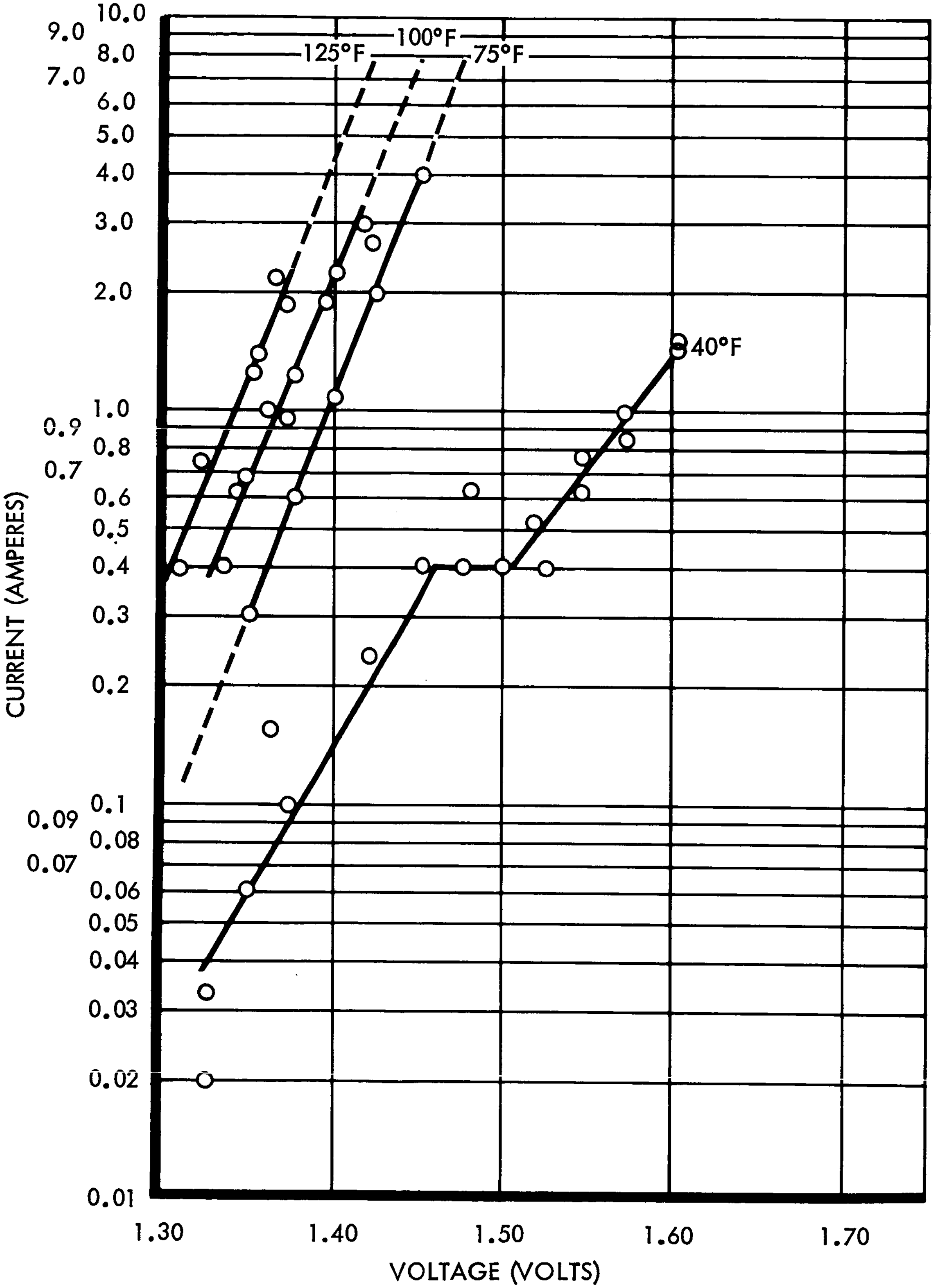


Figure 2-15. 20AH Nickel-Cadmium Cells, Effects of Voltage and Temperature on Steady-State Overcharge Current

For preliminary evaluation of the usefulness of sealed nickel-cadmium batteries, the following data are suggested:

	°F
Minimum Safe Temperature	-70
Minimum Practical Temperature	+20
Minimum Recommended Temperature	+50
Maximum Extended Temperature	+100
Maximum Safe Temperature	+125

2.1.6 Zinc-Mercuric Oxide Batteries

The zinc-mercuric oxide system can be of practical use only as a primary cell system because the product of cell discharge is metallic mercury which undergoes irreversible mechanical changes due to coalescing of the liquid mercury globules although the electrodes are chemically reversible. The zinc electrodes also display similar reversibility problems as in the zinc-silver oxide system.

In the currently used commercial structure, the zinc-mercuric oxide cell is only useful as a low rate cell (from 10 to 100 hour rates), although high rate structures have been designed.

Its performance varies more rapidly with temperature than similar high-rate structure cells due to the larger interelectrode spacing, which magnifies electrolyte-conductivity effects. Therefore, in order to increase shelf life, the electrolyte must be saturated with potassium zincate and the zinc electrode amalgamated with mercury which reduces electrolyte conductivity at low temperatures and increases polarization by requiring diffusion of zinc through a surface layer of mercury.

Figure 2-16 shows a typical relationship of capacity and temperature at various discharge rates (Reference 12).

2.1.6.1 Heating

Zinc-mercuric oxide batteries have been heated electrically from outside sources and by their own energy. Pyrotechnic-type heaters have been used in the form of replaceable cylindrical cartridges and as

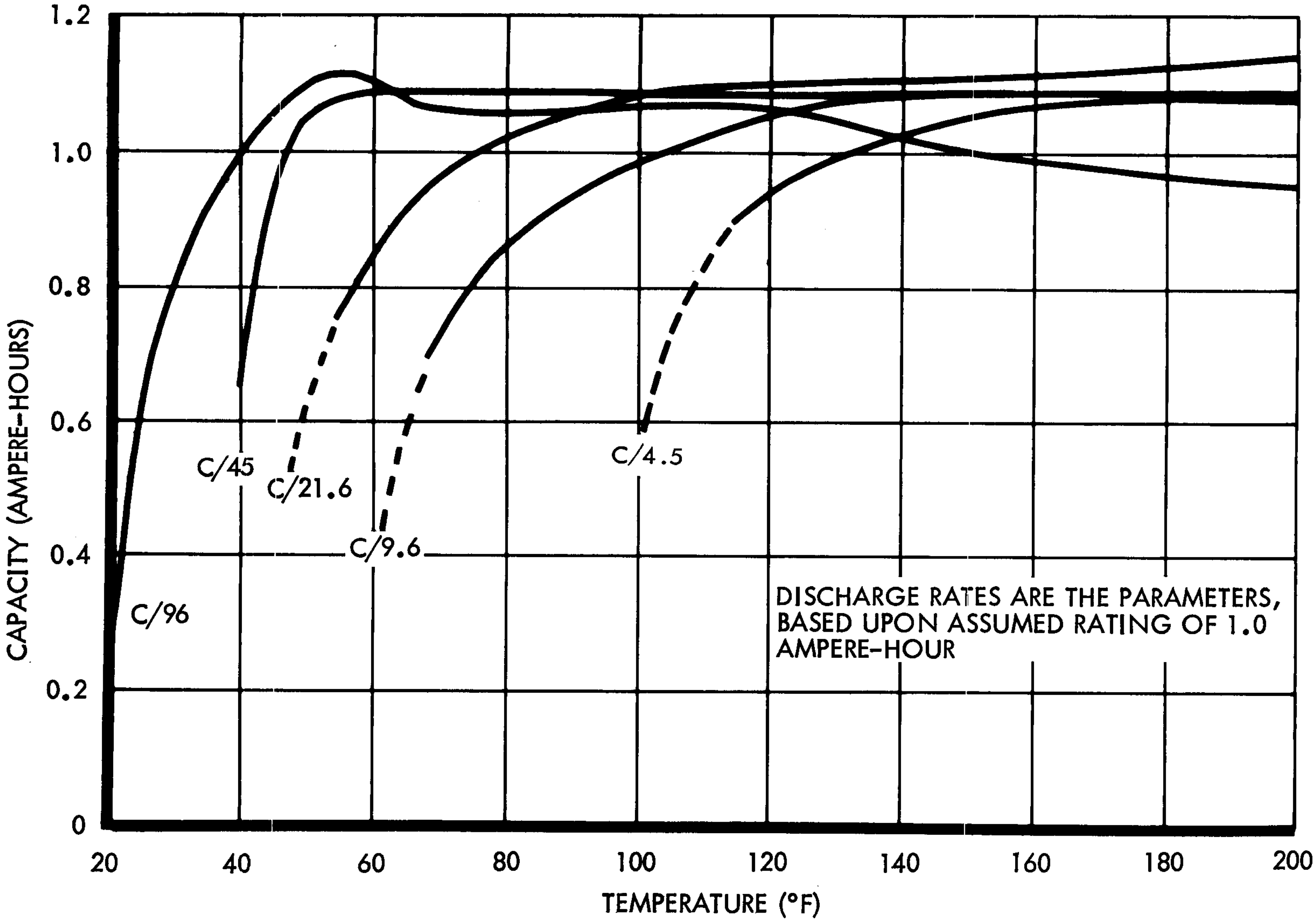


Figure 2-16. Zinc-Mercuric Oxide Cells, Initial Capacity versus Temperature

builtin heaters for heating the batteries from -51°C to operating temperature in a few seconds. However, internal battery temperatures in local areas rose to approximately 204°C and use of these heaters was limited. Common use of these cells in hearing aids worn in an inside pocket has also resulted in their being warmed with body heat.

2.1.6.2 Deterioration Mechanisms

Mercuric oxide reacts with the cellulose of the absorbent separator in the presence of koh solutions to produce mercury and the oxidation products of cellulose. The rate of reaction increases with temperature until, at temperatures from 204 to 232° C, it occurs with explosive violence. In order to improve cell shelf life, barriers of parchment paper, microporous plastics, or polyvinyl alcohol are used to separate the cellulose from the mercuric oxide.

Zinc reacts slowly with koh in the presence of trace impurities or the metallic cell top to evolve hydrogen gas, a reaction which is also accelerated with temperature. The gas generated is vented off by a valving action built into the cell container. Gas generation rates are reduced by amalgamation of the zinc (increasing the hydrogen overpotential at the surface) and by saturation of the electrolyte with potassium zincate (one of the reaction products).

For a preliminary evaluation of the usefulness of the zinc-mercuric oxide cell, the following typical values are suggested:

	°F
Minimum Safe Temperature	-65
Minimum Practical Temperature	+40
Minimum Recommended Temperature	+60
Maximum Extended Temperature	+115
Maximum Safe Temperature	+200

2.1.7 Liquid Ammonia Batteries

Considerable effort has been expended during the last few years to develop batteries that use liquid ammonia, containing dissolved thiocyanate, as the electrolyte. A variety of anode and cathode materials have been

2.1.8 Nonaqueous Electrolyte Batteries (References 13 and 14)

Considerable effort is being expended on batteries using nonaqueous electrolytes in an effort to achieve higher energy densities by taking advantage of the higher output voltages of materials such as lithium and aluminum previously not usable with aqueous electrolytes. These batteries (which use as electrolytes various organic solvents such as butyryl lactone, propylene carbonate, dimenthyl sulfoxide, and others, plus ionizable solutes such as lithium chloride and aluminum chloride, and gaseous solutes) may have some advantage over aqueous electrolyte cells at low operating temperatures (in the case of solvents with low freezing points). However, the conductivity of these electrolytes falls with decreasing temperature (Figures 2-17 through 2-19), and battery voltages fall comparably. We found no published data indicating the relative contribution of ohmic resistivity polarization, activation polarization, or concentration polarization, but it is expected that all three types of polarization will increase with decreasing temperature.

Typical nonaqueous cell discharge data shows a shift in current density of one to two orders of magnitude when decreasing the temperature from +25 to -30°C (Reference 15) and of three orders of magnitude at -55°C, where available current densities are measured in microamperes per square-centimeter rather than milliamperes per square-centimeter.

Heating and other temperature control measures should be no more difficult to apply to cells with organic nonaqueous electrolyte than to ordinary cells, except that additional caution may be required to assure that autoignition of the more flammable species of electrolytes does not occur. The organic base electrolyte cells do not appear to be in a sufficiently advanced state of development for any extensive work to have been done in the area of heating methods.

The following temperature limits are suggested tentatively, but may vary considerably with different electrolyte and electrode combinations.

tested among which are the following (see supplementary Bibliography - Appendix B):

Anode materials

- Lithium
- Magnesium
- Zinc
- Calcium (in solution in circulating electrolyte)

Cathode materials

- Metadinitrobenzene
- Silver Chloride
- Lead Peroxide
- Sulfur (in solution)
- Mercuric Sulfate

Because of the low freezing temperature of ammonia, low temperature performance of the liquid ammonia cells is far superior to that of any other cell type yet developed. However, handling problems, high internal pressures, potentiality for leakage of highly toxic and corrosive gases, and relatively short shelf-life of the system in its present form, make the liquid ammonia battery promising in the near future only as a reserve-type battery.

Based upon currently available data on liquid ammonia batteries, the following limits of operation are suggested:

	°F
Minimum Safe Temperature	No data
Minimum Practical Temperature	-65
Minimum Recommended Temperature	-65
Maximum Extended Temperature	No data
Maximum Safe Temperature	+165

Heating liquid ammonia reserve batteries may present special problems due to the difficulty in leading heat into the battery structure without interfering with the wrap-around ammonia reservoir.

2.1.8 Nonaqueous Electrolyte Batteries (References 13 and 14)

Considerable effort is being expended on batteries using nonaqueous electrolytes in an effort to achieve higher energy densities by taking advantage of the higher output voltages of materials such as lithium and aluminum previously not usable with aqueous electrolytes. These batteries (which use as electrolytes various organic solvents such as butyryl lactone, propylene carbonate, dimenthyl sulfoxide, and others, plus ionizable solutes such as lithium chloride and aluminum chloride, and gaseous solutes) may have some advantage over aqueous electrolyte cells at low operating temperatures (in the case of solvents with low freezing points). However, the conductivity of these electrolytes falls with decreasing temperature (Figures 2-17 through 2-19), and battery voltages fall comparably. We found no published data indicating the relative contribution of ohmic resistivity polarization, activation polarization, or concentration polarization, but it is expected that all three types of polarization will increase with decreasing temperature.

Typical nonaqueous cell discharge data shows a shift in current density of one to two orders of magnitude when decreasing the temperature from +25 to -30°C (Reference 15) and of three orders of magnitude at -55°C, where available current densities are measured in microamperes per square-centimeter rather than milliamperes per square-centimeter.

Heating and other temperature control measures should be no more difficult to apply to cells with organic nonaqueous electrolyte than to ordinary cells, except that additional caution may be required to assure that autoignition of the more flammable species of electrolytes does not occur. The organic base electrolyte cells do not appear to be in a sufficiently advanced state of development for any extensive work to have been done in the area of heating methods.

The following temperature limits are suggested tentatively, but may vary considerably with different electrolyte and electrode combinations.

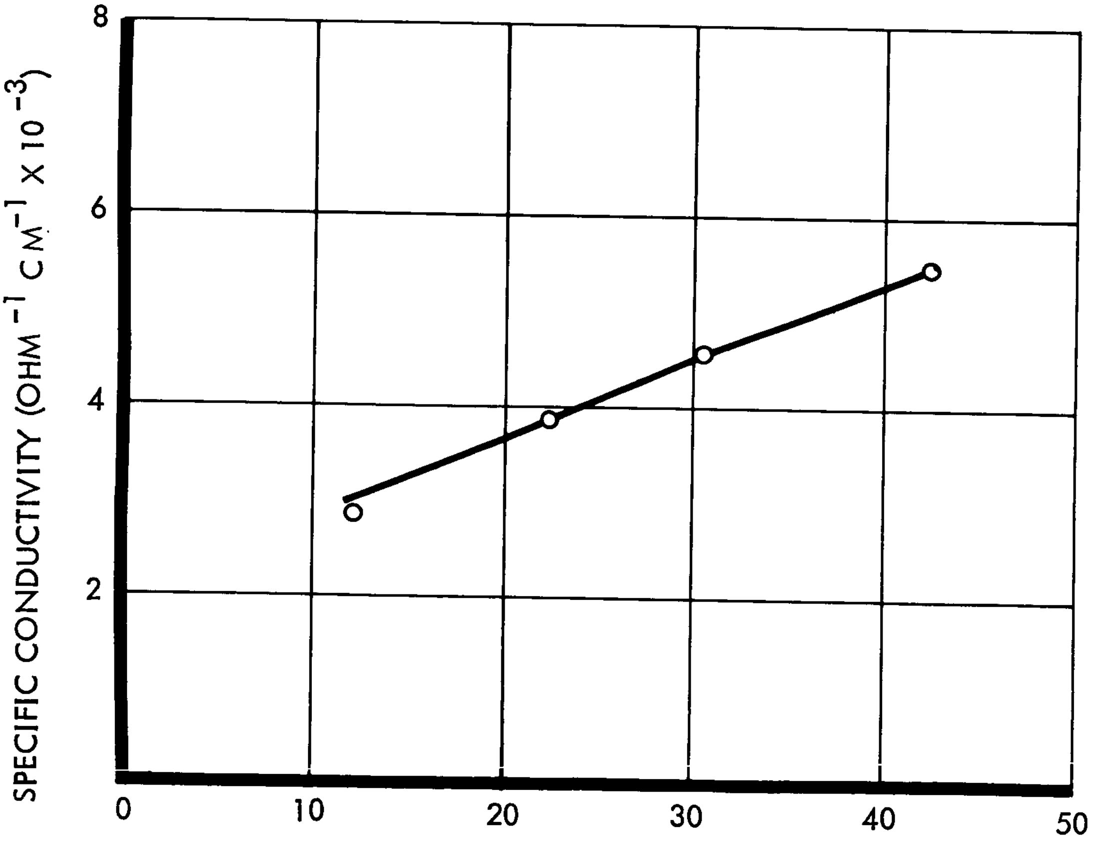


Figure 2-17. Effect of Temperature on Specific Conductivity of Saturated LiF, LiPF, Propylene Carbonate Solution

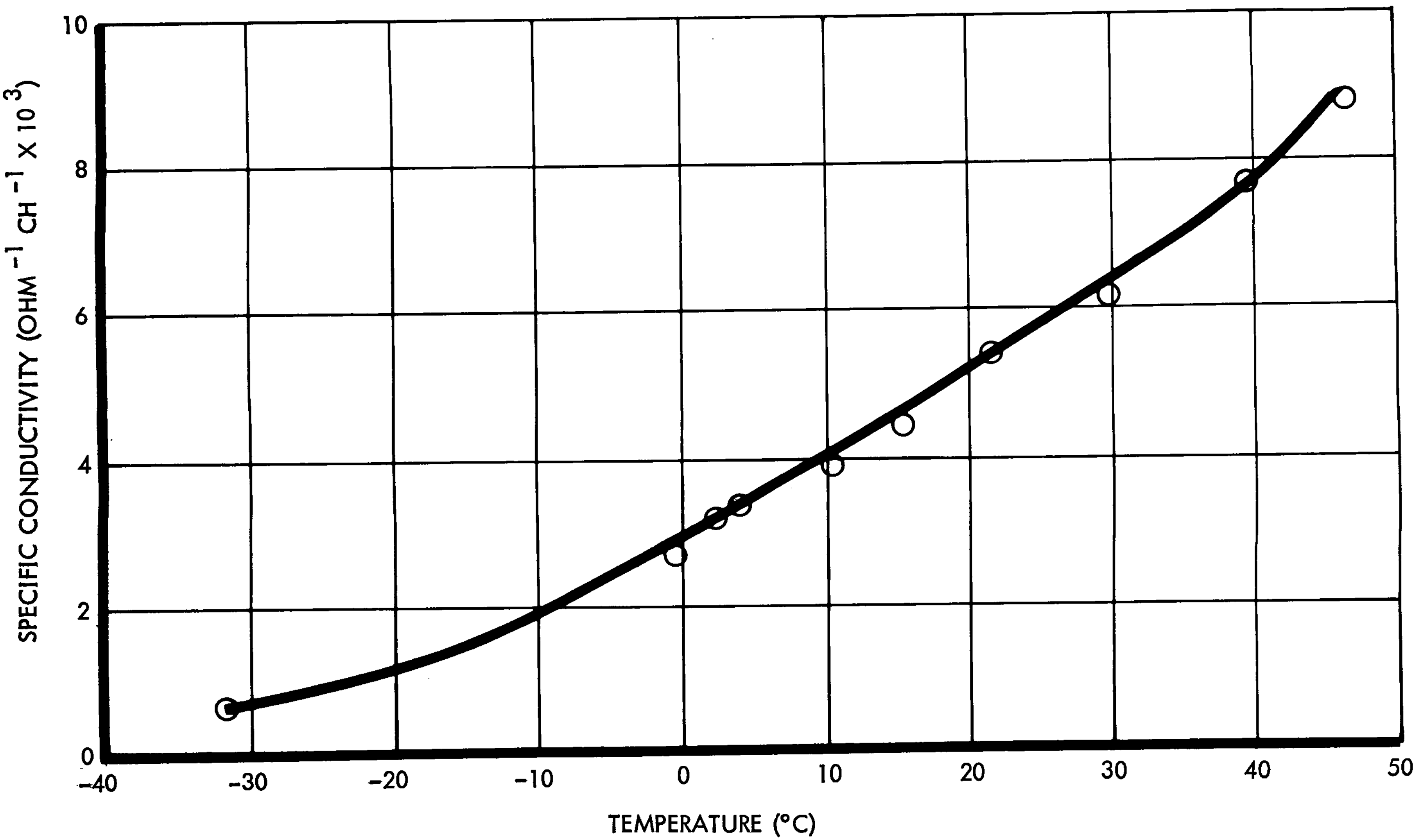


Figure 2-18. Effect of Temperature on Specific Conductivity for the Solution Propylene $AlCl_3$ 10 gm/100 ml — LiCl Saturated

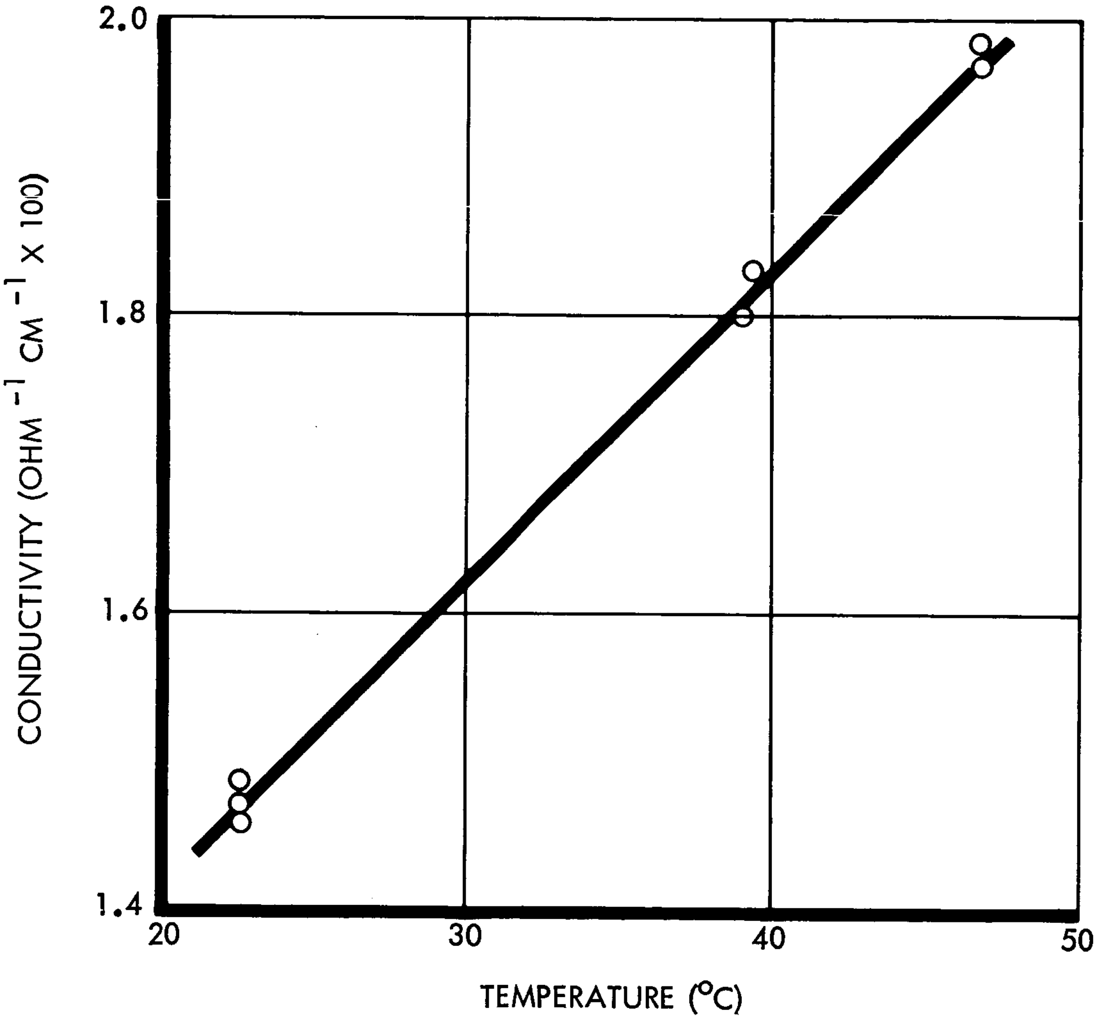


Figure 2-19. Specific Conductivity of Ethyl Pyridinium Bromide-2AlCl$_3$ with 67 Weight Percent Toluene

	°F
Minimum Safe Temperature	*
Minimum Practical Temperature	-20
Minimum Recommended Temperature	+20
Maximum Extended Temperature	*
Maximum Safe Temperature	*

*Data not found; expected to be highly variable between different electrode-electrolyte combinations.

2.1.9 Secondary Fused-Salt Electrolyte Systems (Reference 15)

Fused-salt electrolyte systems offer considerable potential for operation of batteries in low temperature environments, if an adequate heat source, temperature control system and insulation can be carried along. Fused-salt battery technology is divided into two parts:

a) The Thermal Battery: A reserve type heated by pyrotechnics and useful only for very short periods of time (a maximum of 10 to 15 minutes)

b) The Fused-Salt Battery: A secondary type requiring a continuous heat source but useful for an indefinite period as an energy-storage bank for varying loads.

The fused-salt secondary batteries have the advantage of high unit cell voltages available with nonaqueous systems and have higher electrolyte conductivities than any other system, including aqueous electrolyte systems (Figure 2-20).

Disadvantages in a low temperature environment include the need for molten salt containment materials, insulation weight, and large amounts of heat energy to maintain the battery in a molten condition.

One possible concept which could be promising is the combination of a radioisotope thermoelectric power supply and a battery in which the waste heat from the primary power source is used to maintain the battery in the molten condition.

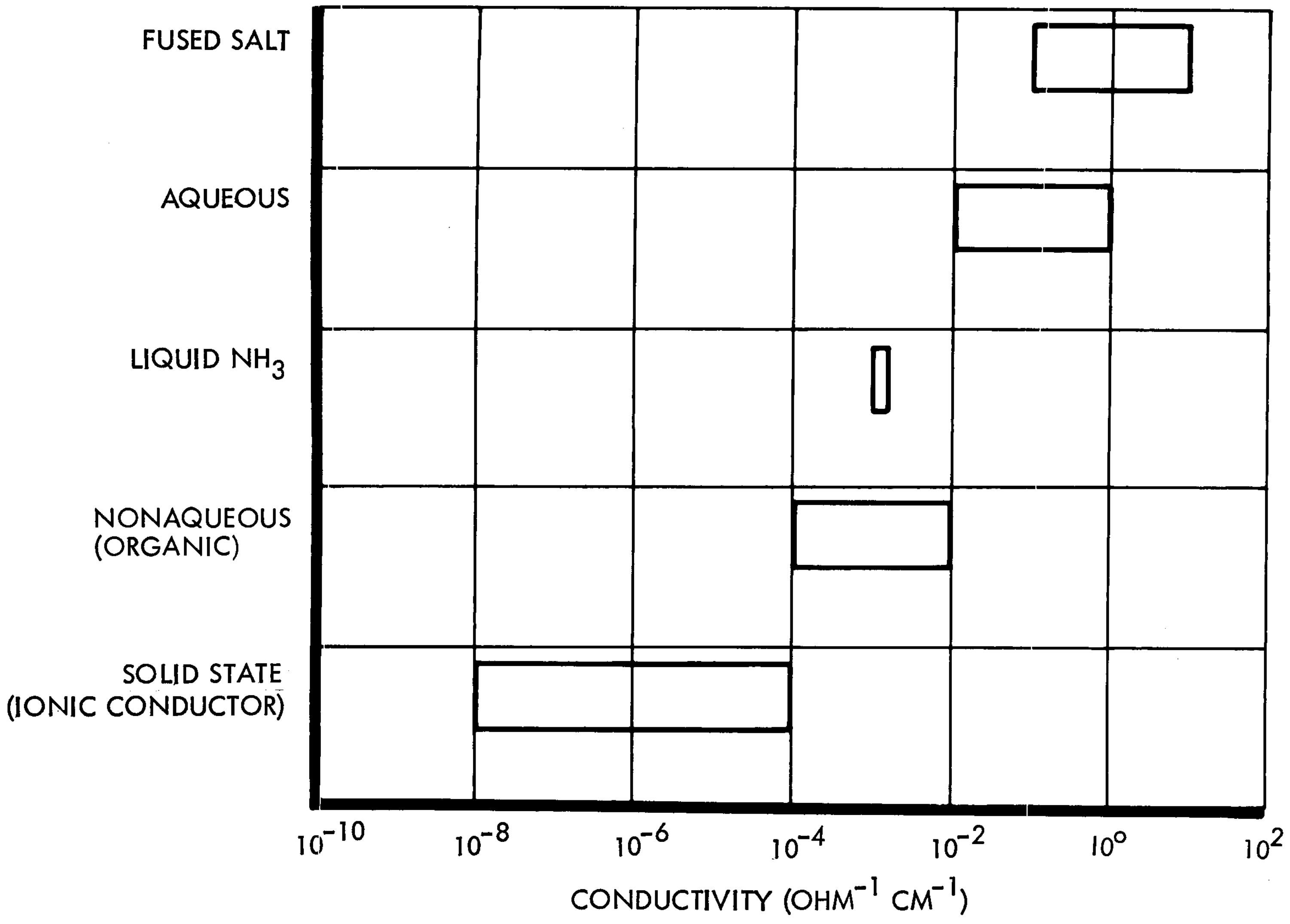


Figure 2-20. Typical Conductivity Ranges for Various Electrolytes

The fused-salt secondary battery is in the laboratory crucible stage; however, sufficient promise is shown to warrant further development since the combination of a relatively high impedance radioisotope power supply and a low impedance battery is attractive for long-term missions in which temperature control of ordinary batteries presents problems.

The choice of electrolytes and electrodes for the fused-salt secondary battery is still broad. As a result, the operating conditions cannot be well defined. However, the broad range of operating conditions would be approximately 400 to 800° C which falls at or near the cold junction temperature range of the more advanced thermoelectric and thermionic generators.

2.1.10 Thermal Batteries

Thermal batteries are reserve primary batteries using fused-salt electrolytes and are intended for use over short times at temperatures from -54° C up.

The limitations on the temperature range over which thermal batteries can be used are unrelated to the battery itself and are a function of the reliability of ignition of the heating component at low temperatures, the danger of self-ignition at high temperatures, and the deterioration of ignition squib and heater material over extended storage at high temperatures.

	°F
Minimum Reliable Ignition Temperature	-65
Maximum Extended Temperature	+185
Maximum No-Fire Temperature	+300

2.1.11 Other Battery Systems

Many other electrochemical battery systems exist but appear to offer no advantage over the above battery systems for low temperature application. Those considered but not investigated in detail in this study. are the following:

Cadmium-Mercuric Oxide
Lead-Acid
Zinc-Manganese Dioxide (both acid and alkaline)
Nickel-Iron
Zinc-Copper (Daniel cell)

2.2 HEAT SOURCES TECHNOLOGY

During Phase I a survey of various methods of providing heat to batteries was conducted. A discussion of the findings of this survey is presented in the following paragraphs.

2.2.1 Heater Technology

Through the investigation of literature on the operation of batteries in Arctic environments, the latest technology of battery heating, by addition or storage of heat, has been established. The methods that have been reported are:

2.2.1.1 Warm Environment Technology

Several methods of heating batteries, by placing them in warm environments, were discussed in the literature reviewed. One method is where portable batteries are worn by personnel in the form of a waistcoat and are kept warm by body heat (Reference 16). Obviously this method is not applicable to unmanned space systems. Another method is where batteries are placed near high dissipation components in a spacecraft, such as the transmitters, converters, and thermal dissipation heat fins for control of primary power. It is noted, however, that this practice is often not feasible because of spacecraft balancing problems and high temperature problems in the near-Earth region of space.

2.2.1.2 Bootstrap Heating Technology (Reference 17)

In bootstrap heating, the battery is short circuited and heat is evolved in the battery cells due to dissipation of the electrical energy stored in the battery. When the battery has been brought to operating temperature, the short circuit is removed and normal battery operation is continued. This method will work only if the battery has, at the minimum storage temperature, sufficient activity to produce heat at a rate higher than the rate of heat loss from the battery. The lowest practical limit for bootstrap heating of most alkaline batteries is estimated at -29°C. Bootstrap heating is more practicable for the nickel-cadmium secondary than the primary batteries but is not commonly used, being likely to damage most types of batteries.

2.2.1.3 Electrical Heating Technology

Where external power is available from the primary power system of a space vehicle, batteries may be heated electrically. If insulation of high quality is used, power required for heating is quite small (on the order of 5 to 50 watts). If heat is applied continuously, electrical heating has the advantage of being readily controllable; the heaters usually take the form of woven fiberglass blankets or strips containing heater wires and thermostatic controls. Batteries using this type of heating are standard hardware on many missile programs requiring battery warmup before launch. No application of electrical battery heating from external sources could be found (for low temperature operations) in published documents on spacecraft design.

2.2.1.4 Heat Evolution Technology (Self-Heating Effect)

There are several sources of heat evolution within a battery which may occur at different times in the normal operational cycle of a secondary battery. These are:

2.2.1.4.1 Reversible Heat. For any chemical reaction at equilibrium (in which ΔH, the molar enthalpy change, is equal to the total heat evolved when the reaction proceeds spontaneously), if the reaction is performed electrochemically, the total available electrical work, ΔG, is given by the equation.

$$\Delta G = \Delta H - T\Delta S.$$

Thus, if the reaction is performed reversibly at constant temperature, pressure, and mass, the total molar enthalpy change (ΔH) will be divided between ΔG (the available electrical work) and $T\Delta S$ (the reversible heat evolved to, or absorbed from, the surroundings).

If the reaction entropy change (ΔS) is positive, heat will be evolved during the reaction. If ΔS is negative, the system will absorb heat, producing more electrical work than the heat content change; this additional electrical work is derived from the heat absorbed from the surroundings.

Upon reversal of the reaction (as in battery charge), the sign of the entropy changes since in any cyclic equilibrium reaction in which the system returns to its original state,

$$\oint \Delta S = 0$$

and an equivalent amount of heat is absorbed or evolved so that the net reversible heat evolved over a complete equilibrium battery discharge-charge cycle is zero.

Since

$$\Delta S = -N_e F \left(\frac{\partial E_o}{\partial T}\right)_P$$

where:

N_e = Electron change in the reaction

F = Faraday's constant

E_o = Open-circuit or reversible potential

ΔS is positive when the temperature coefficient of voltage is negative and the battery cell evolves heat on discharge (reabsorbing it on charge). In systems with positive temperature coefficients of voltage, the reaction is reversed — heat absorption occurring on discharge and heat evolution on charge.

2.2.1.4.2 Heat Evolution Due to Irreversibility. When a battery is operated at appreciable currents, the condition of reversibility no longer applies and an additional quantity of heat is evolved due to the resistive losses in the cell. While these losses occur as ohmic, activation, and concentration gradient potential drops, they may be lumped as a total potential drop equal to the difference between E_o (the reversible potential) and E (the measured voltage at the cell terminals). The power dissipated as heat in the battery is then equal to

$$P = (E_o - E)(I) .$$

On charge, the current reverses direction and, consequently, sign; the result is a net evolution of heat, regardless of current direction due to the departure from reversibility.

2.2.1.4.3 Heat Evolution Due to Side Reactions. Other reactions may occur within the battery simultaneously with or after the normal charge-discharge reactions. Depending upon the nature of the reaction, heat may be absorbed or evolved. One notable special case is a cyclic side-reaction in which no net change occurs in the chemical state of the cell; all of the electrical energy used to drive this reaction appears as heat. An example of the reaction is the oxygen-evolution recombination reaction occurring within a nickel-cadmium sealed cell on overcharge.

2.2.1.4.4 Self-Heat Evolution. The use of battery self-heat evolution for spacecraft applications is practiced in several spacecraft designs. A typical use is in the OGO spacecraft which utilizes the internal dissipation of the battery on overcharge and the dissipation of the battery on discharge to maintain the battery within an operating temperature range of 15 to 35°C. This is accomplished by overcharging the 12 ampere-hour, 22-cell nickel-cadmium battery at the C/40 rate (10 watts/pack) and averaging the overcharge heat with the discharge heat generated for an overall thermal dissipation average of 15 to 20 watts during each orbit. The battery typically shows endothermic heat characteristics during the initial part of the charge, thus balancing out part of the high dissipation rate which occurs during the discharge cycle. The rate of heat generated during the charge cycle is controlled by thermostatic switches which control the charge control system and prevent constant voltage charging after the temperature reaches 35°C, which reduces the dissipation to the preselected 10-watt level. This method of heating the nickel-cadmium battery in a low temperature environment is directly applicable to this study provided that the external power source has the necessary power to accomplish the task and provided that the battery can stand the necessary overcharge current. This method is not practical for silver-zinc or silver-cadmium batteries because they cannot be overcharged at significant rates.

A further review of test data which has been accumulated by several battery test groups has verified that heat generated on charge is not useable except in the overcharge condition. Very cold batteries (below freezing) will not accept high charge rates which reduces the amount of energy which can be returned to the battery or which will result in heating within the battery. A sample calculation of the heat generated on discharge is predicted as follows:

$\Delta S = -N_e F \left(\frac{\partial E_o}{\partial T}\right)_P$ at very low currents

$\Delta P = (E_o - E)I$ for significant discharge currents

E_o = 1.31	nickel-cadmium
1.38/1.15	silver-cadmium
1.82	silver-zinc

E = 1.21	nickel-cadmium at C/2
1.28/1.05	silver-cadmium at C/2
1.45	silver-zinc at C/2

Therefore, as a first order approximation, the heat loss during discharge is as follows:

Nickel-cadmium (140-watt rate; 350 whr battery)
heat loss = 11-watt rate or 8 percent losses

Silver-cadmium upper plateau (150-watt rate; 350 whr battery)
heat loss = 12-watt rate or 8 percent losses

Silver-cadmium lower plateau (140-watt rate; 350 whr battery)
heat loss = 13-watt rate or 9 percent losses

Silver-zinc upper plateau (150-watt rate; 350 whr battery)
heat loss = 27-watt rate or 10 percent losses

Measured data roughly approximates the above theoretical results. Therefore, as a rule of thumb for comparison within the study, the following criteria were established. For normal discharge rates, the internal dissipation of a battery can be assumed to be on the order of 10 percent of the discharge rate of the battery. However, it is noted that this approximation is merely an average and will be much less for low rates and fully charged batteries and may be much higher for high rates and nearly expended batteries or batteries that have been discharging for long periods.

Using the above guidelines, the following analysis shows the amount of heat available for heating a battery using its own dissipation:

$$\text{Heat Available} = 0.1 \frac{\text{Battery Capacity}}{\text{Discharge Time}}$$

For a 600-whr battery at

100-hour rate, heat = 0.6 watts
10-hour rate, heat = 6.0 watts
4-hour rate, heat = 15.0 watts

The estimated heat requirement to maintain an insulated 600-whr nickel-cadmium battery at 4 ^{o} C in a -29 C ambient is 15 watts (Reference Figure 3-7). The estimated feasible operating time, therefore, is 4 hours maximum.

It is concluded from the above analysis that self-heating with battery internal dissipation at low discharge rates (less than C/4 to C/2 discharge rates and at these rates only for short periods of 2 to 4 hours) is not practical. It is noted as part of this conclusion that at temperatures below 4^{o} C, a battery will not provide energy (at the above rates) at an acceptable energy to weight capability.

2.2.1.5 Chemical Heating Technology

Chemical heating has been applied to current batteries in many forms; however, the major applications for operation of batteries at low temperatures have been limited to the heaters discussed below. Published literature on chemical heating of batteries in space is nonexistent although there are several discussions on the application of cryogenically-stored fuels and other fuels in space which may offer some merit for battery heating.

2.2.1.5.1 Fuel Burning Heaters. Fuel burning heaters which use gasoline or propane and air to produce a flame have been used under Arctic conditions. Under emergency military field conditions, fires of wood, mortar shell propellant powder, dung, and other combustibles have also been used. However, none of these methods can be used in space battery applications.

Where short term heating in space (10 to 100 hours) may be required, several cryogenically-stored fuels and oxidants might be considered. For example, hydrogen and oxygen vented from the propellent tanks could be combined catalytically in a flameless reaction, then ducted to the battery case where the water vapor would condense and impart its latent heat of vaporization to the battery. The liquid water would be wicked away and dumped overboard or stored. Figure 2-21 shows the design concept. As part of this study, several fuels were analyzed. Their characteristics are noted below for information purposes.

a) Cryogenically-Stored Fuels

$H_2 + O_2 \rightarrow H_2O$ (g)

ΔH = 57,798 calories/mole = 5,779 Btu/pound
(as the heat of reaction)

b) Storable Fuels and Oxidants

Several storable fuels and oxidants were found which offer some possibility of application for heating. These are broken down into two categories — monopropellants and hypergolic bipropellants.

1) Monopropellants

Hydrazine (40 percent NH_3 decomposed)	1050 Btu/pound
Hydrogen Peroxide	
(to water vapor)	690 Btu/pound
(to liquid water)	1250 Btu/pound

2) Hypergolic Bipropellants

N_2O_4 + Substituted Hydrazines	
(to water vapor)	3000 Btu/pound
(to liquid water)	3400 Btu/pound
ClF_3 + Substituted Hydrazines	3000 to 3500 Btu/pound

Several problems exist when using the above fuels in space. The first is difficulty with the melting point which limits the temperature at which the fuel can be used. These melting points are:

H_2O_2	31° F
N_2H_4	32° F
N_2O_2	11.8° F
Mixed substituted hydrazines	-65 to -76° F
Monomethyl hydrazines	-56° F
ClF_3	-147° F (Boiling point, + 52° F)

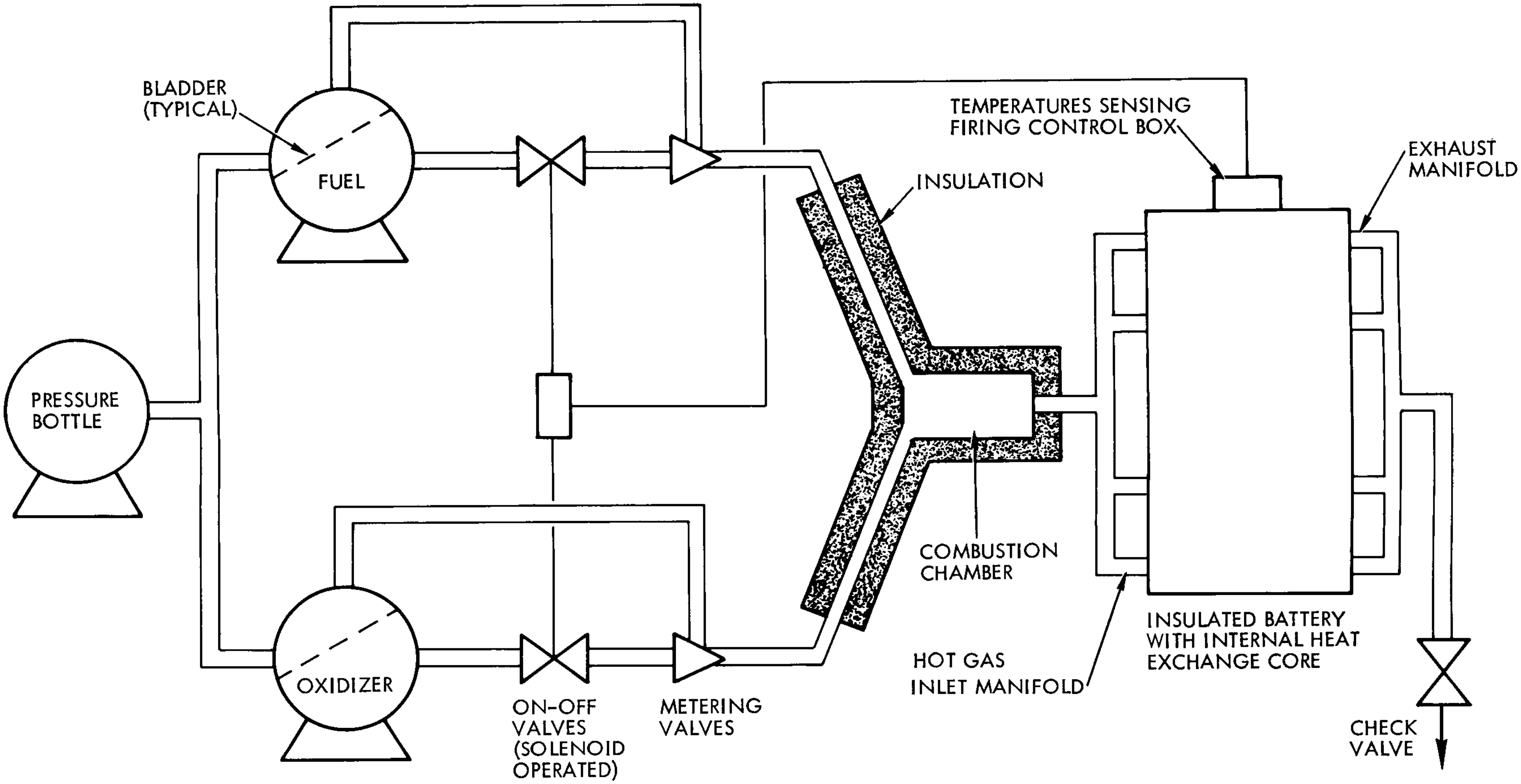


Figure 2-21. Schematic Diagram of Combustion-Heated Battery Installation Using Hypergolic Fuels

In addition to the melting point problem, the following are other problems as noted in the literature:

a) ClF_3 is highly corrosive, giving reaction products containing hydrogen-floride which is also extremely corrosive and difficult to contain.

b) The above fuels having inert reaction products also have high melting points requiring heat to make them available.

c) Metering of the fuels is a serious problem at low rates such as would be required to heat batteries. The maintenance of combustion is also a problem area. Control of stoichiometry of reactants to achieve the value of heat output desired requires accurate metering. Metering orifices in the low energy levels required would be pin-holes, subject to clogging.

d) Those reactants which deliver water as a reaction product may not be allowed to condense to a liquid state. Water must be vented overboard in the vapor state in order to avoid ice formation within the vehicle and around the periphery of the vehicle.

e) The technology of maintaining low-level flames in space has not been developed. Therefore, if this method is to be considered for heating batteries, further refinements in technology will be required before an adequate assessment of the feasibility of using them can be made.

A brief analysis of the theoretical limitations of heating with stored fuels and oxidants shows the application of these heaters to be time limited by weight. This analysis is given in the following paragraphs.

Assume a heat requirement of 20 watts. Then, selecting the heat of reaction of a typical hypergolic bipropellant as discussed above, the fuel weight requirement per day is as follows:

20 watts x 24 hours x 3.413 = 1,640 btu/day required

1640/(3,000 btu/pound) = .55 pound/day at 100% efficiency
(3,000 btu/pound is for $C1F_3$)

Adding tankage, combustion chamber and plumbing as 50 percent of theoretical energy plus 2 pounds for the combustion chamber, the following weight equation was formulated:

$$\text{Heater weight} = \frac{\text{Heating time} \times .55 \text{ lb/day}}{.5} + 2 \text{ pounds}$$

If the heater weight maximum is selected as 25 percent of battery weight for a 600-whr nickel-cadmium battery, the maximum heating time at the 20-watt rate would then be 11.8 days.

Therefore, the use of storable fuels and oxidants is not practical for heating batteries in space because of the technical obstacles discussed above and the basic fuel weight versus heating period limitations demonstrated in the above analysis.

2.2.1.5.2 Chemical Cartridge Heaters. Dry chemical cartridge heaters consist of a cartridge containing a cake or powder mass of dry solid materials which is ignited by a squib or primer device and which reacts exothermically. These cartridges have been used in the past to heat batteries by several methods.

a) One method is to bury the cartridge in the battery and to use the heat evolved to warm the battery rapidly from storage temperature to room temperature. However, this method may result in high local temperatures on the order of 210 to 260° C or higher and is thus limited in its applicability to high energy space batteries because it can damage the plate and separator materials, and cause high-energy battery reactions to go to completion in the form of shorted cells.

b) Another method is to attach cartridges to the mounting platform of the battery and heat the entire battery mass from an external heat source. This method appears to be the only method applicable for low temperature battery applications in space since it is the only way of controlling the rate of heat applied to the battery and controlling the gradients within the battery to keep the battery cell performance balanced.

The exothermic materials used are combinations of materials such as zirconium, magnesium, aluminum, and boron and oxidants such as barium chromate and inert materials with high melting points.

Table 2-1 compares a list of metals versus oxidants and their respective heats of reaction. While some of these heaters are slower than others and may be slowed further by including in the mixture an inert material of high heat capacity, the reaction is usually complete within seconds. It is not practicable within the limits of present published technology to slow the reactions further or to attempt to develop heaters of this type which will achieve gentle heat for hours or days. The application of the heaters is, therefore, limited to short-time warmup or to pulse heating methods which might be extended over periods up to approximately a week, depending upon the weight penalty which can be allowed.

Other types of heating cartridges exist which release heat at slower rates. These are based on an oxidation reduction reaction between organic intervals and an oxidant in the wet or dry state, reactions between air and organic materials, or heat of hydration released by the slow addition of water to calcium-oxide. However, these methods of heating have not been developed to the point where they can be used in a low temperature environment in space.

By the use of battery insulation and several heating cartridges which are fired sequentially by a set of thermostatic controls, it is considered feasible to maintain a battery at operating temperature for hours or days. Figures 1-1 and 1-2 outline the basic designs which could be used separately or integrated for warmup and temperature control. It is believed, with further analysis and technology, that controlled warmup from low temperatures is feasible using this method if some kind of thermal controller can be provided to distribute the heat and control the rate of heat application to the battery. This approach will be discussed further in the Phase II analysis.

2.2.1.5.3 Thermally-Activated Fused-Electrolyte Batteries. Thermal batteries are assembled with the electrolyte, in the form of a solid salt, and are activated by squib ignition of a solid exothermic material. The evolved heat melts the electrolyte and the battery operates for a short time until it has cooled or until the reactants are exhausted. Normal operating times are on the order of 3 to 5 minutes with a maximum practical use time of approximately 15 minutes. Once cooled,

Table 2-1. Metals Versus Oxidizers, Heats of Reaction

Metals	Silver Oxide $-\Delta H^{o}$	Silver Chromate $-\Delta H^{o}$	Barium Peroxide $-\Delta H^{o}$	Lead Chromate $-\Delta H^{o}$	Cuprous Oxide $-\Delta H^{o}$	Barium Chromate $-\Delta H^{o}$	Iron Oxide $-\Delta H^{o}$
Magnesium	139.1(I)	688.5(I)	126.7(I)	560.7(I)	103.6(5)	292.9(5)	239.8
Aluminum	378.0	1869.0	340.8(I)	1485.6	271.5	253.6(N)	200.5
Zirconium	244.1	1206.5	219.7(I)	950.9	173.1	483.5	377.3
Titanium	211.0	1041.0	186.2(I)	785.4(5)	140.0	384.2	278.0
Silicon	187.0	921.0(N)	164.8(I)	665.4	116.0(N)	312.2(12)	206.0(N)
Manganese	89.5	440.5	77.1(4)	312.7	54.0(10)	144.1(23)	91.0(N)
Chromium	252.0(N)	1239.0(N)	214.8(5)	855.6	145.5	63.8	74.5
Zinc	76.5	374.5	63.9	247.7	41.0	104.6	52.0
Tin	124.1	606.5	99.3	350.9	53.1	123.5	17.3
Iron	177.5(8)	866.5(8)	140.3	483.1(11)	71.0(N)	53.1(N)	0
Cadmium	58.2	284.0	45.8	156.2	22.7	50.2	-2.9
Tungsten	174.7	852.5	137.5	469.1	68.2	50.3	
Molybdenum	155.5	759.0	118.3	373.1(17)	49.0	31.1	
Nickel	50.5	250.0	39.0	122.2	15.9	29.8	
Cobalt	51.4	245.5	38.1	117.7	15.0	27.1	
Antimony	145.0	704.0(10)	107.8(6)	320.6	38.5	20.6	
Bismuth	116.1	544.0	78.9	176.1	9.6	-8.3	
Copper	31.5	150.5	16.4	22.7		-29.9	
Nonmetals							
Phosphorus	195.0(I)	192.4(I)	487.6(I)	128.4(2)	213.0	552.8(6)	35.3(15)
Sulfur	142.1	682.5(4)	158.2(4)	101.2(9)	95.9	28.8	-15.1
Selenium	67.9	311.5	86.6	31.3	-6.2		

$-\Delta H^{o}$ Heat of reaction (gram calories per gram)
(N) No reaction observed
(I) Very fast burning rate
(8) Burning time, seconds

it is not practical to reactivate these batteries in their present state of development. It may be feasible to use this type of battery to warmup a low temperature battery by using its electrical energy to heat a battery heater blanket and simultaneously use its exothermic reaction energy to supply heat to the battery mounting platform, thus raising the battery interface temperature as well as the battery heater temperature in a relatively short period of time. Since this combination appears to offer very limited attraction for space application, it was not considered in the Phase II analysis.

It should be remembered that chemical cartridge heaters and thermal batteries will require an electrical ignition pulse or mechanical percussion action to initiate the exothermic reaction. At low external environmental temperatures (-54 to -73° C), ignition can itself become a serious problem which may make it impossible to use these devices for heating low temperature space batteries.

2.2.1.6 Nuclear Heater Technology

The direct conversion of nuclear decay radiation to heat is one of the most fundamental phenomenon of atomic energy. It was not one of the earliest applications because of the initial high cost, limited inventory of radioisotopes, and general reluctance to use radioisotopes if other fuels would serve the purpose. For these reasons, isotope heating was first used in electrical power-producing generators, such as the radioisotope thermoelectric generator (rtg). The need for electricity in remote terrestrial and deep sea applications, and later in space, brought about a general acceptance of isotopes as a source of thermal energy.

In order to meet the anticipated need for isotopes, the U.S. Atomic Energy Commission has sponsored extensive materials and radiation studies so that selective isotopes may be safely and efficiently used for diverse purposes. Since heat production ranked high as a potential application, many hundreds of different radioisotopes were investigated. Increased usage has prompted detailed studies of fuels, production, encapsulation, containment, terrestrial and aerospace safety, ground handling and costs. Although a continuing need exists for the advancement of knowledge in the nuclear field, much has already been accomplished through the substantial expenditure of funds by industry and Government. These funds have enabled novel applications of isotopes (such as space battery heating) to be examined promptly at a moderate cost. Accurate estimates, based solely upon prior art, may be made as to the feasibility, reliability, safety and cost of the concept. Necessary experimental effort may then be expended to resolve specific problems associated with the proposed application.

Continuing interest by the AEC in application of radioisotopes for research, industry and the military is evidenced by the cooperation extended to researchers in the form of technical data, professional assistance and the loan of sealed sources to qualified and licensed users.

Radioisotope production, which started at the Oak Ridge National Laboratory in the late 1940's, has grown from tracer-level quantities to thermal megawatts on an annual basis. The ORNL facilities have been augmented by the vast potential at Hanford which will soon have a full-scale production facility for fission products (including promethium) under the direction of private industry. The Mound National Laboratory artificial radioelement facility will be augmented in a similar manner by the Hanford partitioning to private industry. It is the intention of the AEC to keep pace with increasing demands for isotopes by the most expedient means, thereby permitting potential users to anticipate fuel for new, larger and more numerous applications.

Applications involving the projected use of radioisotopes have a choice of hundreds of radioisotopes, each possessing individual properties. Factors such as the mission purpose, mission duration, power density (specific activity), fuel quantity, decay mode, and attendant radiation influence the selection. A radioisotope heat source for space battery heating must be evaluated for conformance to the following selective specifications:

- Useful life in excess of 1 year
- Fuel availability for from 5 to 50 watts (thermal) per mission
- Acceptably low radiation level to obviate heavy shielding

- Established preparation and encapsulation technology
- Conformance to aerospace safety criteria
- Acceptable cost.

2.2.1.6.1 Projected Availability of Isotope. The U.S. Atomic Energy Commission has made a rigorous study of the properties, abundance, and estimated costs for potential applications of individual ratioisotopes. AEC findings are periodically made available, along with the compound forms most generally suited to present applications. The same general requirements cited above for battery heat sources also apply to many isotopic power devices and advantage can be taken of prior effort. Table 2-2 lists the eight most attractive heat-producers scheduled for, or in, production. It would be prudent to make a selection from this table, and thereby comply with the second requirement on the above list. While it must be understood that Table 2-2 represents a "best estimate" of future production levels, battery heater fuel requirements are so nominal that production at any reasonable level will ensure sufficient material for initial application(s).

2.2.1.6.2 Radioisotope Characteristics. A review of Table 2-3 shows the half-life, decay mode and power density of eight radioisotopes. Five of the eight could provide a suitably long half-life for the proposed mission, as cited in the first item on the specification list. Of these five, Sr-90 and Cs-137 would present a shielding problem due to their energetic bremsstrahlung radiation levels. Cm-244, an alpha emitter, exhibits a substantial neutron flux from a spontaneous fission side reaction. While its high power density will ensure Cm-244 a vital role in future space power, power density alone is not a decisive criterion for battery heating; and Cm-244 would not be attractive for this application. The remaining two radioisotopes, Pm-147 and Pu-238, possess attractive half-lives and adequate power densities.

The fission product, Pm-147, is a radioisotope of an element not found in nature. It decays by emitting a soft beta particle and a very weak gamma ray; therefore, shielding weight would be small. Pu-238 is primarily an alpha emitter; and where slight neutron level is tolerable to instrumentation, it should be an excellent choice. Those missions

Table 2-2. Projected Radioisotope Availability for Power Applications*

ISOTOPE (KWT)	1965	1966	1967	1968	1969
Sr-90	24	24	48	48	48
Cs-137	17	17	48.3	48.3	48.3
Ce-144	30	30	800	800	800
Pm-147	0.36	1.08	2.16	1.8	5.4
Pu-238	5	7	9	11	13
Cm-244	0	0	8	13	13
Cm-242	9.6	120	120	120	120
Po-210	14	14	140	140	140

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*FORECAST BY THE U.S.A.E.C.

Table 2-3. Properties of Radioisotopes for Power Applications*

Isotope	Decay Mode	Half-Life	Chemical Form	Specific Thermal Power			Useful Mission Life
				(watts/kc)	(watts/gm)	(watts/cc)	(years)
Sr-90	Beta	28 years	Titanate	6.5	0.2	0.7	10
Cs-137	Beta Gamma	27 years	Glass	4.8	0.072	0.22	10
Ce-144	Beta Gamma	285 days	Oxide	7.9	2.3	13.8	1
Pm-147	Beta	2.6 years	Oxide	0.37	0.18	1.0	2.6
Pu-238	Alpha	89.6 years	Metal	34.5	0.48	9.3	10
Pu-238	Alpha	89.6 years	Oxide	34.5	0.39	3.9	10
Cm-244	Alpha	18.4 years	Oxide	35.0	2.3	22.4	10
Cm-242	Alpha	163 days	Oxide	36.2	120	1,170	0.5
Po-210	Alpha	138 days	Metal	31.7	140	1,320	0.5

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*Supplied by the Division of Isotope Development, U.S.A.E.C.

which cannot tolerate neutrons may be satisfied by Pm-147. Both fuels, therefore, satisfy the third requirement in the specification list. In addition, Pu-238 satisfies the fourth and fifth requirements since it has already been used for space power. Pm-147 has an established production technology, and encapsulated fuel increments have been fabricated at the Oak Ridge National Laboratory. Aerospace safety evaluation, although incomplete, is under investigation.

2.2.1.6.3 Radioisotope Costs. The individual cost of any component for space use is of extreme importance. Cost is listed last in the table of specifications only because it is a malleable item. Table 2-4 shows projected and present isotope costs. These are subject to reduction based upon increased demands for space electric power systems. Because there are no power programs requiring Pm-147 in quantity, its current cost is relatively high ($485/watt). However, its low radiation level appears attractive for manned missions from both biological and weight aspects. The projected price of $91/watt reflects future use. The AEC has suggested that supply may initially lag the indicated demand and that for near-future requirements, its use as a battery heater be limited to those missions having a low neutron tolerance. The higher cost of Pu-238 is partially offset by its long useful mission-life (10 years) without appreciable reduction in power. This price quoted is for highest power density material necessary for electric power applications. A lower purity fuel would also be applicable for battery heating. The power density of the oxide shown in Table 2-3 is about four times as great as for Pm-147; therefore, even with void space, some latitude exists for the dilution of plutonium while maintaining power density equivalent to that of Pm-147. Fuel cost is not necessarily a governing factor in assessing battery heater cost. Fuels requiring massive gamma or voluminous neutron shields can result in costly heater configurations although the isotopes contained may be modest in price.

2.2.1.6.4 Decay Processes and Their Effect on Isotope Selection. Listed third in the foregoing specifications for ideal battery heater fuel is an acceptably low radiation level. Radiation above a tolerable limit requires shielding. Most missions are weight limited and the requirement for dead weight shielding can seriously reduce the attractiveness of

Table 2-4. Isotopic Power Costs*

Isotope	Material Cost (Dollars)		Projected Yearly Production Rate	Thermal Power ($/watt)	Mission Time (years)
	Present	Projected			
Sr-90	1.00	0.50/Curie	10 Mc	77	10
Cs-137	0.75	0.50/Curie	10 Mc	104	10
Ce-144	0.50	0.04/Curie	100 Mc	5	1
Pm-147	1.00	0.18/Curie 0.01 (Hanford)	15 Mc	485 91 (Hanford)	2.6
Pu-238	1,000.	500/Gram	50 Kg	894	10
Cm-244	---	1,000/Gram	50 Kg	435	10
Cm-242	---	19,800/Gram	80 G	165	0.5
Po-210	88,000.	26,500/Gram	100 G	190	0.5

*Supplied by U.S.A.E.C.

NOTE: Cost estimates are based on projected production levels.

radioisotope use to a mission. It, therefore, is appropriate that the nuclear processes which give rise to radiation be briefly reviewed so that the candidate battery heater fuels may be evaluated.

Radioisotope decay energy is derived from the instability of a radioactive atom and occurs during its transmutation to the ground state of the isotope of a different element. During decay, an energized particle is usually emitted (K-capture being an exception) from the nucleus. In the case of the fission product Pm-147, a beta particle (mass and charge the same as an electron) is ejected whose energy is 0.23 million electron volts (mev). When the particle impinges upon a target or shield, part of that energy is converted to heat while the remainder gives rise to spectrum penetrating rays, known as "bremsstrahlung radiation". A moderate shield of high atomic number (Z) material can attenuate the beta particle, converting it to thermal energy. In the case where the shield is thin or a low Z material is used, the ray may give up some of its initial energy to ionization and escape through the shield.

The second phenomenon cited above which follows transmutation is the reduction of the excited state of the daughter product. In going from an excited state to a ground (stable) state, one or more gamma rays of energies totaling the excited level are emitted. Once ejected from the nucleus, these rays are indistinguishable from an X-ray or a bremsstrahlung ray described above. Therefore, the economical accumulation of heat from radioactive decay depends upon the efficiency of particle attenuation while minimizing bremsstrahlung. Where fission products are employed, external shielding may be required with thickness commensurate with the energy of the beta particle. In most applications the contribution by gamma ray heating is negligible, and additional shielding is used not to collect heat but to reduce the gamma radiation to a level tolerable to humans or electronic components. While all matter will attenuate radiation, the prudent selection can minimize size, weight, and cost, while optimizing integrity.

The second group of heat-producing materials is the useful isotopes of the artificial heavy radioelements. The decay modes of these radioisotopes are more complex and also potentially more useful.

Those presently considered for heat source applications are limited to Po-210, Pu-238, Cm-242, and Cm-244. The naturally occurring alpha emitter, Po-210, is also produced artificially in quantity, but its short half-life (138 days) limits its mission applications. Pu-238, Cm-242, and Cm-244 each have two separate and simultaneous decay modes. The first, alpha particle emission, is a useful source of heat. An alpha particle is ejected from a radioactive nucleus during transmutation. The alpha has the same rest-mass as a helium nucleus (4.0039), being composed of two protons and two neutrons. The alpha particle has a plus-two charge. Since its rest-mass is 8,000 times that of a beta particle, ejection from a nucleus imparts considerably more energy (mev) to it than is possessed by a beta particle, although its velocity is less than the beta. Alpha attenuation mainly occurs within the fuel material itself, and external shielding no greater than several thicknesses of ordinary paper would stop any particles that escaped from the source surface. Unlike betas, alpha particles do not give rise to bremsstrahlung when attenuated in an absorber. For these reasons, alpha emitters usually make excellent heat sources.

The alternate decay modes of Pu-238, Cm-242 and Cm-244 are a spontaneous fission process. Most simply explained, the unstable nucleus ejects particle fissions into a) two fragments or fission products, b) fast neutrons (2 or 3), and c) an energy release in the form of heat and gamma radiation. For use in heat sources, this decay mode is highly undesirable since it complicates the shielding picture by the presence of fast neutrons. Fortunately, the half-life of this decay mode is much longer than for alpha decay (Pu-238 fission occurs once for each 4×10^7 alpha emissions), resulting in a nuisance-value neutron flux from small sources. However, where the Pu-238 quantity approaches hundreds or thousands of thermal watts, the neutron level can be serious enough to require special additional shielding. Body tolerance (and that of sensitive instrumentation) is 10 times lower for fast neutrons and 2 times lower for thermal neutrons than for gamma rays of equivalent energies.

Another important consideration to the evaluation of alpha-emitting heat sources is known as a "tunneling effect". It was stated earlier that an alpha particle may be attenuated by a thin shield, with a total energy conversion to heat. However, it has been found that when a sufficiently energetic alpha particle interacts with certain target nuclei it can, under certain conditions, be absorbed into the nucleus. When this occurs (for example, in O-17 and -18), an alpha-neutron reaction results and neutron radiation occurs. The radiation is most noticeable when the radioisotope is in a compound form, whose anion is susceptible to the alpha-neutron tunneling effect. It follows that careful selection of the compound form can minimize the reaction. Because the susceptible isotopes of such target elements are usually of low abundance and the probability of reaction is also low, the neutron flux from the tunneling effect is quite modest.

A final consideration prerequisite to the use of alpha emitters, although unlikely to be a concern to battery heating, is critical mass. It is well known that when sufficient mass of Pu-239 is brought together, criticality is reached and an uncontrolled nuclear reaction occurs. It was postulated that those artificial radioisotopes whose nuclei contained even-odd neutrons and protons by number would possess relatively low critical masses. However, it was recently found that Pu-238 contains a critical mass that is quite low, despite its even-even distribution of neutrons and protons in the nucleus. This leads to the conjecture that Cm-242 and Cm-244 may also require careful re-evaluation.

The above considerations would not affect battery heating, provided the thermal flux were below several kilowatts. It should not, therefore, be construed as a constraint to this development program. Mention is made of criticality solely to present the reader with a complete image of the decay modes of the candidate fuel materials.

2.2.1.6.5 Summary. The maintenance of optimized battery temperature in the space environment, by means of radioisotope heating, has been reviewed and found to be feasible with present-day technology.

Two radioisotopes which appear particularly attractive for this application are Pm-147, a beta-emitting fission product, and Pu-238, an alpha-emitting artificial radioelement. The radiation aspects of these two isotopes appear to be entirely safe (with the quantities anticipated for use) from a biological and electronic viewpoint, and no problems to ground support personnel or to telemetering equipment are expected with heat sources in the 0 to 50 watt range.

Radiopromethium is attractive for several reasons:

a) the absence of strong gamma radiation,
b) a high melting compound form,
c) no stable isotopes found in nature.

Therefore, its presence on the surface of other planets would be readily explainable by future explorers.

A low cost fuel form of Pu-238 is less expensive than the cheapest fossil-fuel on the basis of dollars per watt-years of output. It also possesses no naturally occurring isotopes or strong gamma radiation and the slight neutron flux may be attenuated.

2.3 THERMAL CONTROL SYSTEMS TECHNOLOGY

Battery heating applications may be classified into two broad categories based upon heating rate.

a) Situations where the ambient environment tends to be cold but does not fluctuate widely, either due to static conditions or an independent thermal control system. In these instances it is only necessary to heat the battery at a fixed rate.

b) Situations where the ambient environment fluctuates widely, either due to dynamic conditions or the inability of a spacecraft temperature control system to cope with all possible conditions.

These classifications are of the most elemental thermal interest and do not take into consideration the total amount of heating required or the nature of the primary source of heat energy.

The first category is the simplest of the two, and the design problem evolves finally into a choice of the rate of heating, the nature of the heat source, and the mechanical and thermal design of a heated battery package. A typical application for such a battery thermal controller appears to be a Mars-landing mission.

The second category involves situations in which the heated battery, under the worst of conditions, must fend for itself completely. Perhaps a more realistic situation is one wherein the ability of the battery to exercise some control over its own temperature is used to augment an overall spacecraft thermal control system. The most obvious application appears to be probes such as missions deep into the asteroid belt and the Jupiter flyby. In addition to the design problems of the simpler heated battery, there is the additional problem of the battery temperature control system.

Selection of a thermal controller divides into two branches indicated by the previous battery classifications:

- a) Constant-Rate Heating
    - 1) Selection of heat rate and source based on the nature of the mission
    - 2) Thermal design of a heated battery package.
- b) Variable-Rate Heating
    - 1) Definition of limits of heat rates
    - 2) Selection of a heat source compatible with heat rates and mission
    - 3) Selection of battery thermal control concepts
    - 4) Thermal design of a complete heated battery assembly.

The tasks associated with the first category are standard and consist mainly of assuring that adequate heat transfer paths exist to supply and distribute heat at the required rates and temperatures and that design margins are adequate.

Under the second category, the primary emphasis is on the selection of battery control concepts which include:

- a) Thermal Switches. The intent is to consider systems whose physical configurations and thus, thermal configurations are substantially altered by temperature gradients or differences so as to make or break contacting heat transfer areas. Devices with which such alterations may be achieved (discussed in the literature) are:
    - 1) Bimetallic elements,

2) Differential expansion structures, and

3) Liquid or gas actuators (bellows and diaphragms).

b) Superinsulation. Superinsulation, basically a radiant heat shield in a vacuum exhibits changes in conductivity due to the absence or presence of either pressure or atmospheric gases. A large amount of published literature exists on this method of thermal control, but details will not be discussed here since an analysis of an application as discussed in the Phase II section of this report lends itself better to understanding the application of superinsulation technology in space.

c) Heat Pipe. This is a device under current development employing liquid-to-vapor phase-change phenomena to effect heat transfer. The device is potentially capable of transferring heat at high rates across small temperature differentials. The device is at the feasibility development level and will be analyzed further in the Phase II discussion.

d) Liquid-to-Solid Phase Change Phenomena. Patent No. 2,984,727, assigned to the GE company, defines a temperature and heater control device capable of maintaining temperature within narrow limits.

e) Heliotropic Devices. Heliotropic devices to collect and funnel solar energy are feasible for probes in the near and central reaches of the asteroid belt. Either absorbers or reflectors may be considered.

f) High Temperature Thermal Radiation Sources. High temperature sources (e.g., radioisotopes) are relatively insensitive to local environment. By suitable shields and controls, such a device could reject most of its heat to space when heat is not needed, or transmit significant amounts of heat to critical (spacecraft) locations when required.

g) Open Loop Systems. These are defined loosely as localized temperature control systems which depend upon externally available energy or matter for their operation. Should the external source be depleted, the system becomes inoperative. Examples of such systems are pyrotechnic, catalytic, or combustion heaters; gas-operated control mechanisms; and control mechanisms operated by external electrical power.

2.4 SPACE MISSION REVIEW

The principal activity during Phase I of the low temperature battery study thermal analysis was the examination of the ranges of battery environments under two general conditions. These conditions were those experienced in a) space probes and b) landing on other planets.

Before examining the thermal aspects of the heated battery in detail, it is important to consider the need for such a device. It was found that spacecraft components generally have two different temperature limits; nonoperating (standby) and operating. Typical temperature-sensitive components of spacecraft have nonoperating limits of -40 to +65° C and operating limits of about -18 to 38° C; while typical batteries, as has been shown, have limits of +4 to 38° C. Thus, independent of any special requirements for the battery, there must be a spacecraft thermal control system which maintains a minimum temperature of at least -40° C.

In reality, it is unlikely a spacecraft would be maintained in a non-operating state in flight for extended periods of time. For a variety of reasons, it is considered necessary to keep the spacecraft "on" continuously or at least turn it "on" frequently. In the case of a deep space probe, for example, even if its primary mission is the collection of data at a distant planet, it is necessary to communicate with the spacecraft frequently during the transit portion of the mission for tracking purposes to check on the state of its health, to switch to redundant components where necessary, and to collect scientific data in mid-course. The spacecraft thermal control system must, therefore, maintain an environment within the operating range for most components (-29 to +38° C).

Since the battery is frequently the most temperature sensitive of the components, it will normally be located within the temperature-controlled region of the spacecraft. The utility of the heated battery is now apparent; it permits the spacecraft thermal control system to maintain a lower limit of -18° C (suitable for most components) while the battery heater, with appropriate insulation, maintains the battery above its minimum required temperature of 4° C. Without the battery heater the entire spacecraft environment would have to be maintained over the narrower battery limits.

When the battery heater is sized to maintain the battery above its 4° C minimum for all conditions of the mission, it is likely to cause the battery to overheat above its 43° C upper limit for some portion of the mission. For a deep space probe, overheating might occur, for example, when the spacecraft is exposed to a higher sun level or when the power duty cycle causes an increased internal temperature. The need, therefore, arises for a controllable heater to avoid exceeding the battery's upper limit. For an electrical heater, this is achieved simply by means of a thermostatic switch. As discussed elsewhere, a radioisotope heater has many outstanding advantages for this application; but since the radioisotope source continously dissipates heat, a mechanical means must be provided to control the heat source. Such active battery heaters are described in detail in Section 3.

In order to quantitatively examine typical battery environments, it was necessary to consider specific spacecraft or landing capsule configurations. Hypothetical configurations are presented in this report which exclude a multitude of other considerations which more specifically define a mission and the means of its accomplishment. The following discussions detail the effort and conclusions derived under the Phase I program.

The battery environment in space is determined by the spacecraft thermal control system which in turn must be designed for a specific mission and spacecraft configuration. We have assumed, however, on the basis of the above discussion that battery temperature control limits are generally more stringent than limits placed on other equipment and may require means of separate control.

The following missions were considered as possible applications where the use of a heated battery would be advantageous and which might typify the requirements for using a heated battery:

Solar Probe
Mars Lander
Mercury Dark Side Lander
Asteroid-Belt Probe
Jupiter Flyby

Of these five missions, examples of all but the last were analyzed at least briefly during the Phase I information search, to determine the feasibility of using heated batteries and to obtain reference data from which a heated battery design could be evolved. The Jupiter flyby, from the thermal viewpoint, is actually a maximum penetration into the asteroid belt and would probably incorporate most features of probes capable of lesser ranges.

Some of the vehicle configurations considered in this preliminary analysis are as follows:

a) Current spin-oriented Pioneer-type spacecraft designs (probes toward the Sun or closer reaches of the asteroid belt)

b) Mars lander with an active thermal control system

c) Mercury dark-side lander with a passive thermal control system.

In order to provide a reference for preliminary analyses, it was assumed that battery heating would be available from a self-contained constant heat source and that regulation of the overall temperature level of a complete spacecraft system, including the batteries, was available using a spacecraft thermal control system. The battery heater eases the task of the thermal control system but does not assume the function of thermal control to any marked extent. Under these assumptions, the most attractive applications were limited to missions typified by the following:

a) Mars lander

b) Mercury dark-side lander of small size and limited lifetime

c) Asteroid-belt probe ranging to about 3.0 AU maximum.

Analysis of solar probe missions did not reveal any readily apparent regimes for using self-contained heated batteries.

Preliminary analyses of asteroid belt probes indicated that Pioneer-type probes beyond 3.0 AU would experience serious low temperature problems, and additional active thermal controls in the battery spacecraft interface may be required. Missions to the further reaches of the asteroid belt, as well as the Jupiter flyby, can take advantage of a heated battery with a self-contained, self-controlled heat source. Also, such a battery could be useful to improve the overall spacecraft thermal control system for missions with lesser thermal extremes.

Specific mission analysis results are discussed below:

a) Solar Probes. Spin-oriented (Pioneer-type) probes and a sun-shielded configuration were considered for distances ranging from 0.2 to 1.0 AU.

From the thermal viewpoint, it is noted that Pioneer is a rotating cylinder covered by solar cells; internal equipment platform temperatures are controlled by a temperature-sensitive louver system. Solar array temperatures, as an index of internal temperatures, are shown in Figure 2-22. The effects of partial silvering of the solar arrays (to obtain low temperatures closer to the Sun) are shown in Figure 2-23. It was concluded that the basic Pioneer design is already capable of missions ranging in distance from 0.4 to 1.0 AU without the necessity for heated batteries since overall temperature levels are mild, and equipment power dissipation can be distributed so as to provide local temperature adjustments where necessary.

The sun-shielded configuration considered the use of a heated battery to maintain reasonable temperatures immediately after launch. However, since the margin of tolerable solar heating as the vehicle approached the sun was quite small, it was extremely doubtful that this particular configuration would be a successful application.

It was thus concluded that solar probes were not a promising field of application for heated batteries, and no further analysis was conducted.

b) Mars Landing Capsule. The general thermal design features and some design data for a Mars landing capsule are shown in Figures 2-24, 2-25 and Table 2-5. Martian surface temperatures are shown in Figure 2-26. The capsule power profile is given in Figure 2-27.

The resulting capsule temperatures, as also shown in Figure 2-27, assume normal cyclic operation. It is evident that the resulting temperatures are too high, and that the louver system is largely ineffectual, working near its upper limit. However, these deficiencies can be corrected by suitable and obvious changes in the thermal design of the capsule, such as lowering external solar absorptivities, decreasing

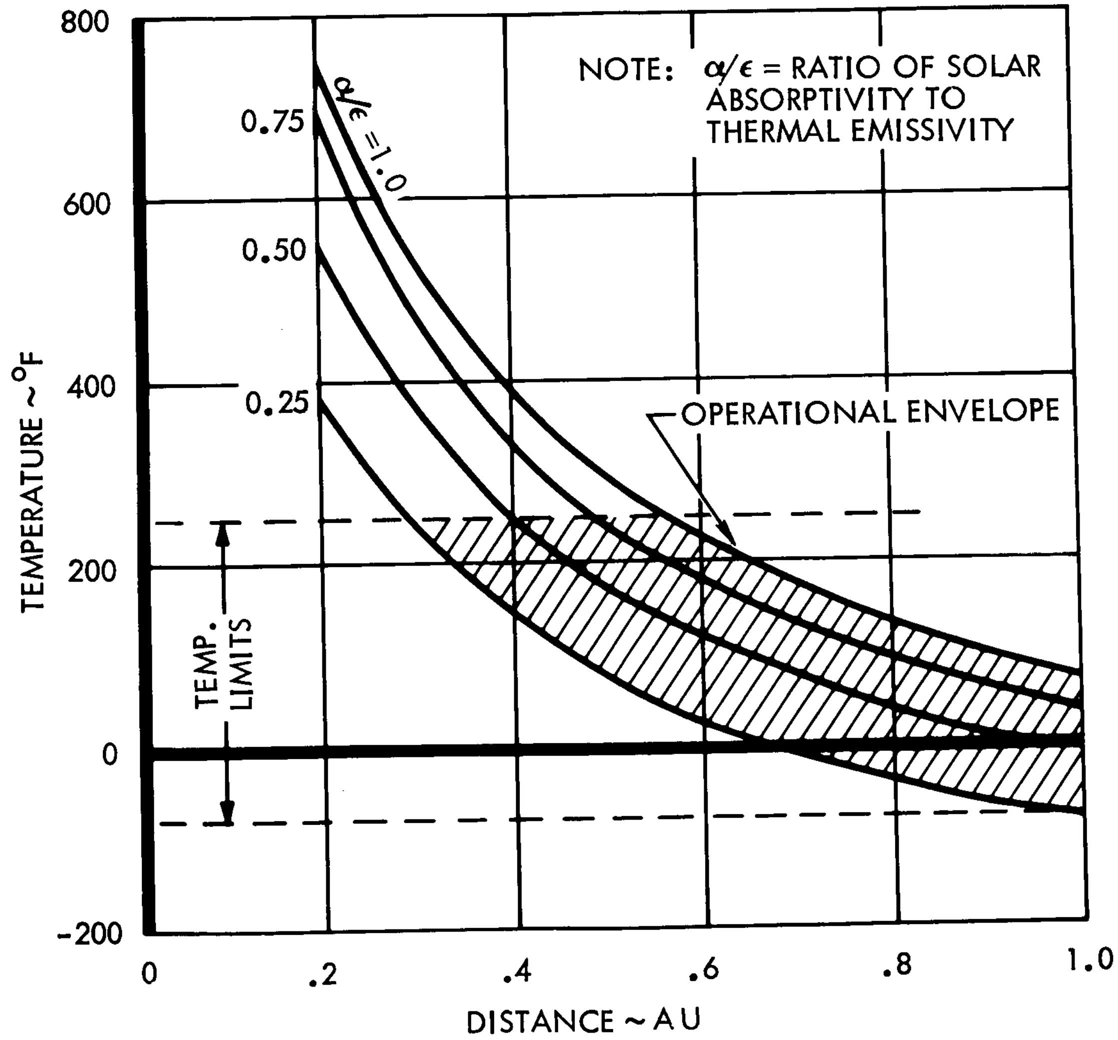


Figure 2-22. Solar Array Temperatures of a Spin-Oriented Solar Probe.

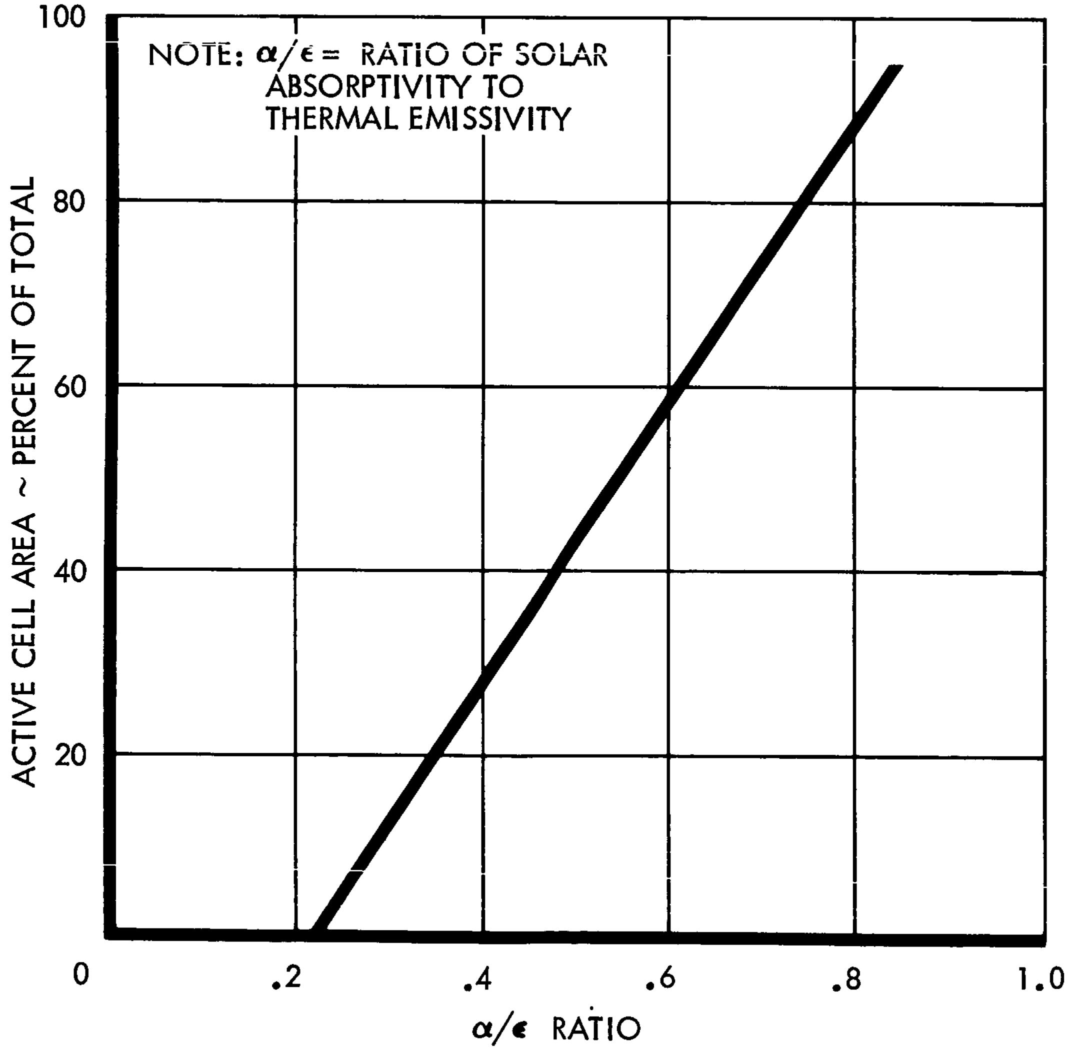
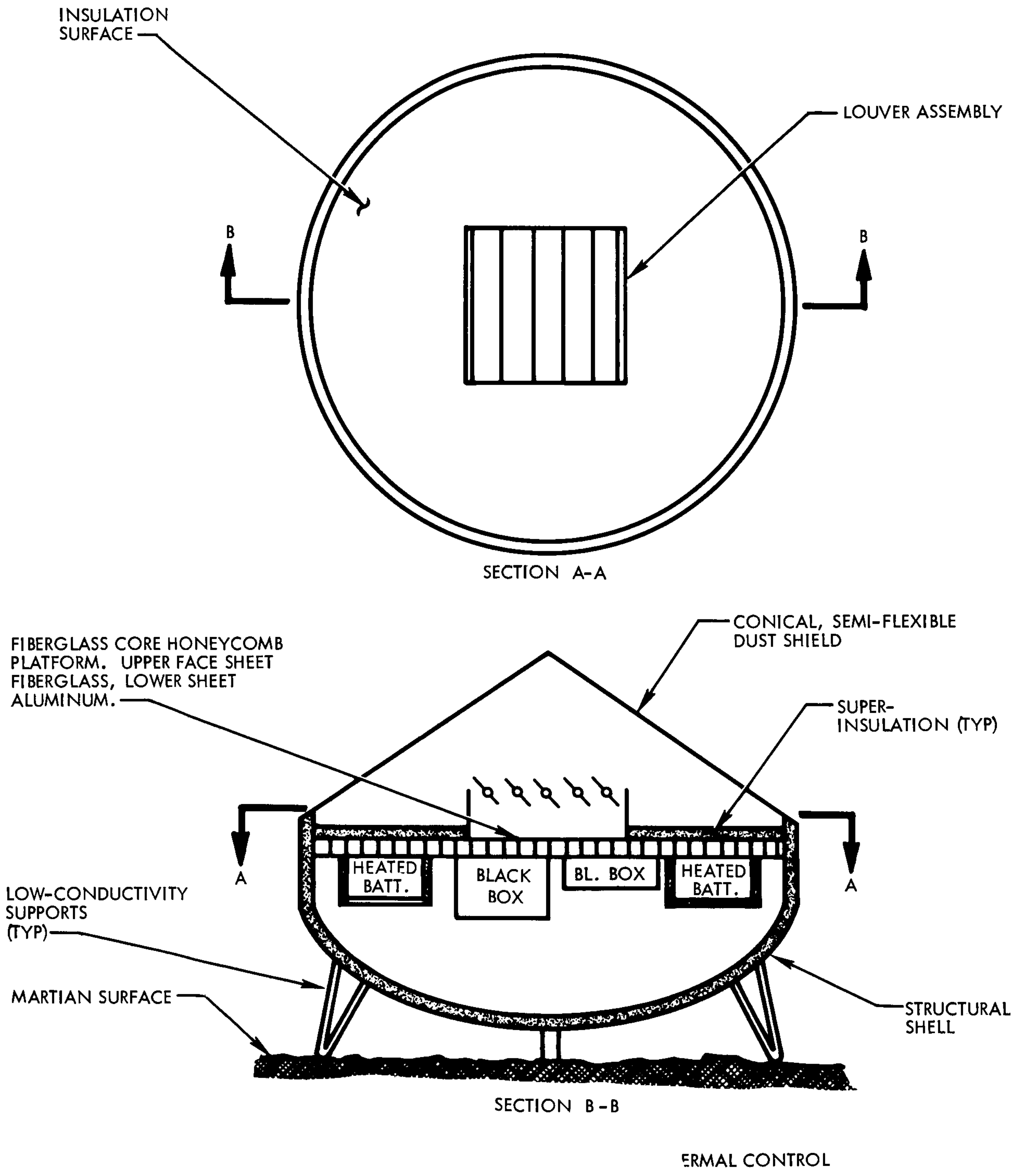


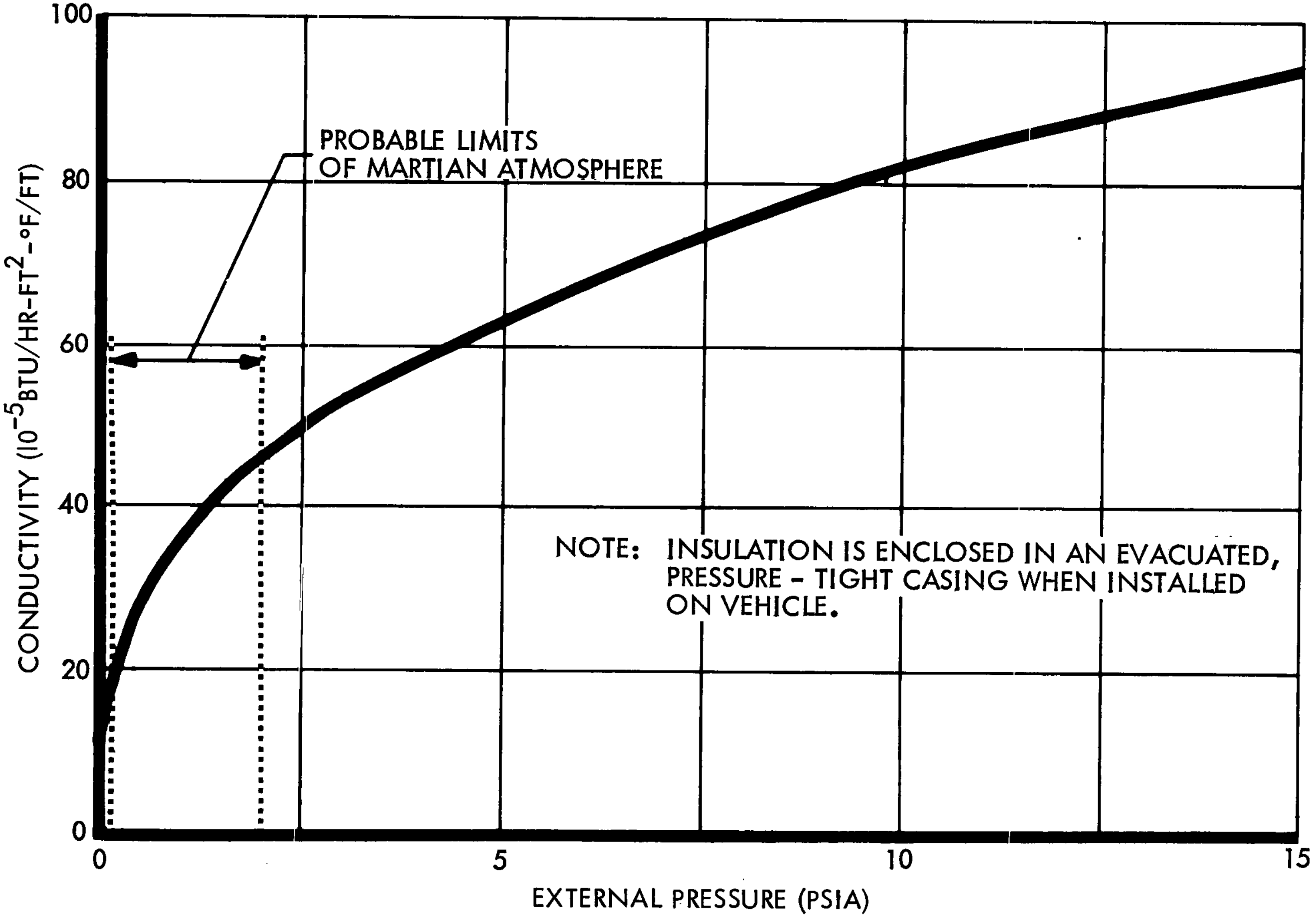
Figure 2-23. Effect on Active Cell Area by Partial Silvering of Pioneer-Type Solar Cells to Obtain Lower α/ε Ratio



SECTION A-A

SECTION B-B

Figure 2-24. Proposed Thermal Control System for a Mars Lander Study Model



NOTE: INSULATION IS ENCLOSED IN AN EVACUATED, PRESSURE - TIGHT CASING WHEN INSTALLED ON VEHICLE.

Figure 2-25. Thermal Conductivity of Encapsulated Superinsulation

Table 2-5. Assume Design Parameters for Mars Landing Capsule Study Model

DIMENSIONS	
Radius	20 inches
Height of Conical Shield	16 inches
Length of Cylindrical Section	10 inches
Depth of Spherical Bottom Segment	10.4 inches
Louver Area	16 x 16 inches
WEIGHTS	
Batteries	Two at 11 pounds each
Platform and Electronics Equipment	100 pounds total
THERMAL PROPERTIES	
Conical Shield	External $\alpha/\epsilon = 0.32/0.8$ Internal $\epsilon = 0.8$
Louvers	$0.20 \leq \epsilon \leq 0.74$ Actuation Range 48°F (closed) to 92°F (full open)
Cylindrical Section	External $\alpha/\epsilon = 0.32/0.1$
Bottom	External $\epsilon = 0.1$
Platform	Core $k/L = 1.0$ Btu/hr-ft^2-°F Upper face sheet (fiberglass) $k/L \longrightarrow 0$ Lower face sheet (aluminum) $k = 90$ Btu/hr-ft^2-°F/ft
Multi-Layer Insulation	1 layer of 1 mil aluminized mylar 9 layers of 3 mil Dexiglas material 8 layers of 0.25 mil aluminum foil Total Blanket Thickness 3/16 inch $k/L = 0.03$ Btu/hr-ft^2-°F

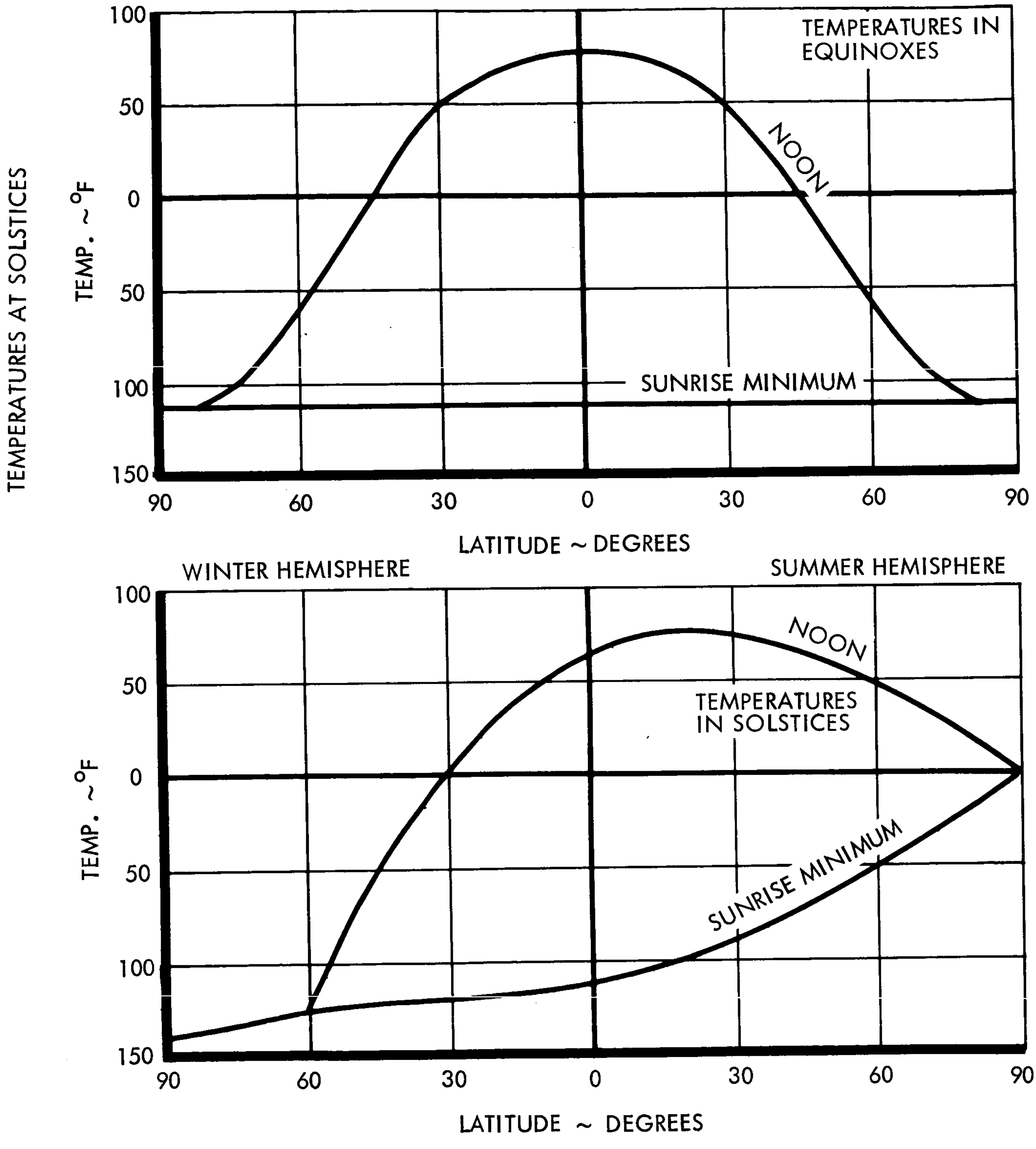


Figure 2-26. Martian Surface Temperatures for Preliminary Studies (Reference 18)

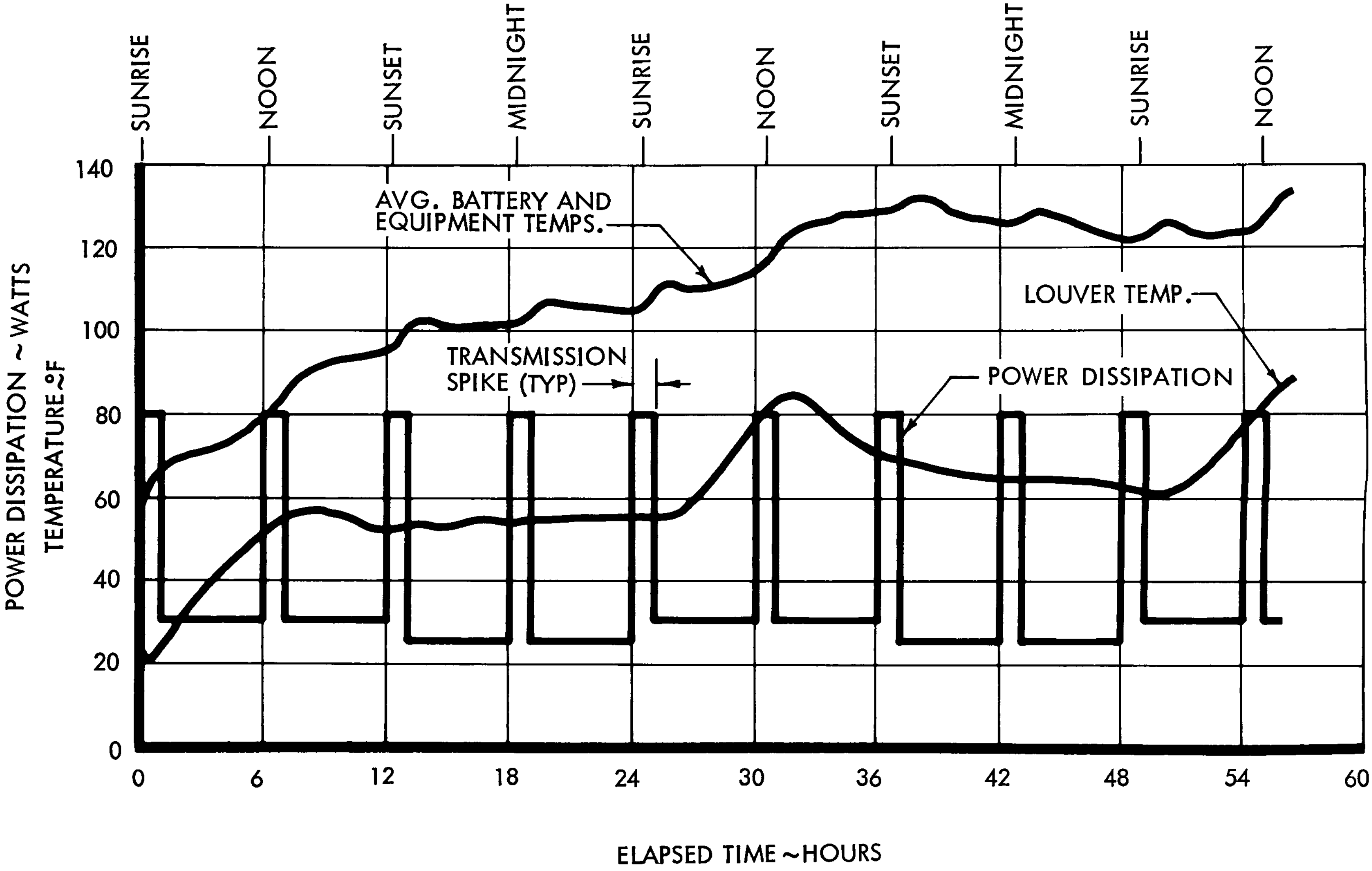


Figure 2-27. Mars Capsule Study Model Temperatures and Power Cycles During Summer

insulation thicknesses, increasing external emissivities, or increasing louver areas. However, these measures must be balanced against the probability of operation under less favorable conditions than the Martian summer.

Figure 2-28 shows the ability of the battery heaters to maintain reasonable temperatures in the event that transmission spikes are omitted from the power cycles for extended periods. It is seen that the battery heater dissipation results in acceptable temperatures, ensuring the continued life of the capsule during periods of idleness.

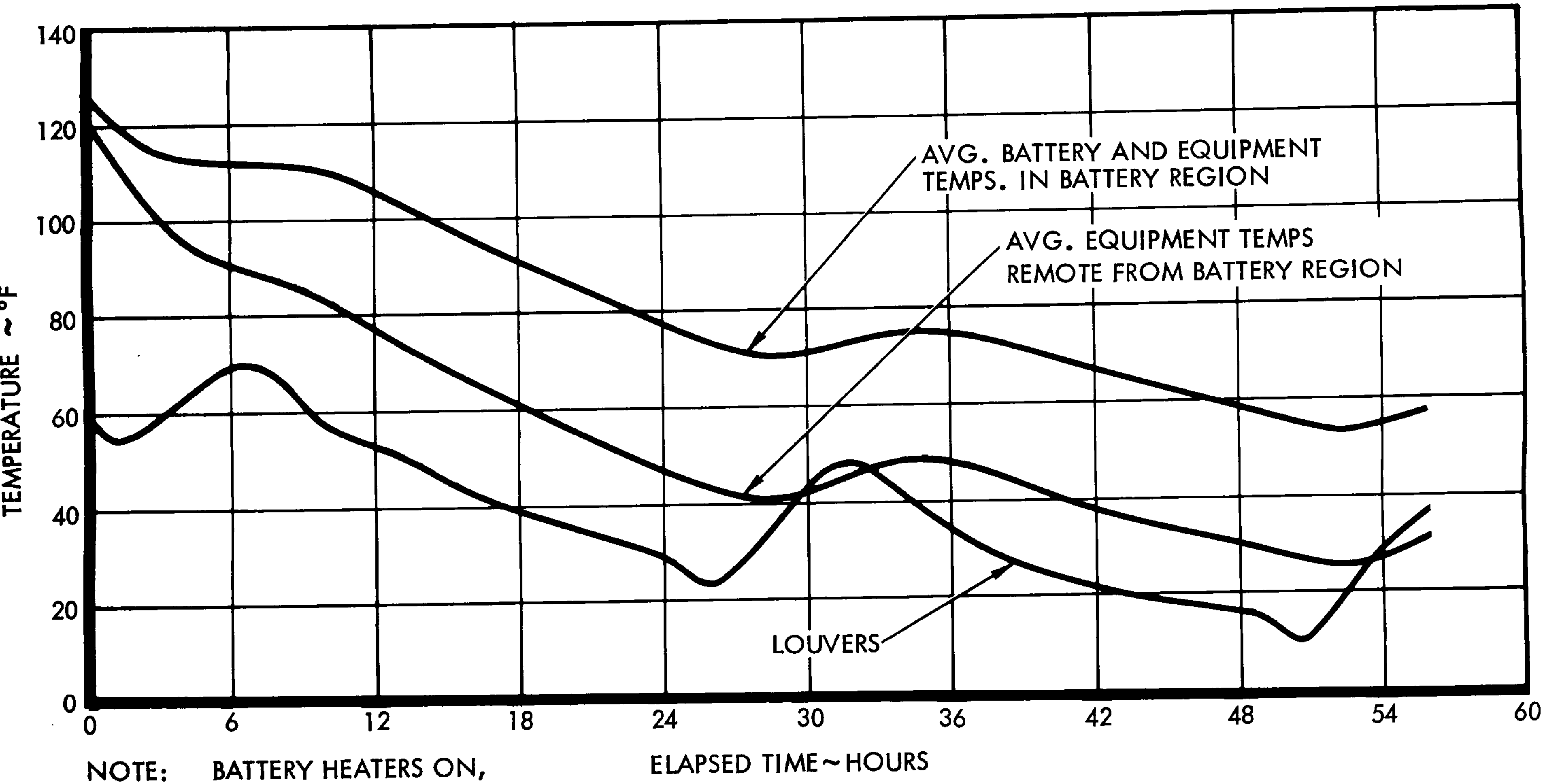
c) Mercury Dark-Side Lander. The protection of the Mercury landing capsule during the transit period, as opposed to operations on the dark side of Mercury, may be extremely difficult due to possible drastic changes in the local environment. During transit, the capsule approaches the Sun to a distance of 0.31 to 0.47 AU, so that the local intensity of the solar constant has increased to a value of 5 to 10 times over that in the vicinity of the Earth. On the dark side of Mercury, however, there is no sunshine at any time, and the ambient temperature is extremely low, perhaps -240° C. There is little surface atmosphere.

A simple but highly arbitrary scheme has been postulated to maintain the capsule in suitable condition for transit and Mercurian surface conditions. By shielding the capsule from the Sun during transit, but otherwise exposing it to the space environment, the conditions it encounters on Mercury will not be greatly changed.

In the simplest case, the capsule may be a sphere containing a heated battery and other electronic components. If the average total heat dissipation is 30 watts, the required insulation conductivities and capsule size may be estimated as shown in Figure 2-29

d) Asteroid-Belt Probes. Figure 2-30 shows a section of a spin-oriented, Pioneer-type vehicle somewhat modified for penetration to the inner regions of the asteroid belt. Figure 2-31 shows the variation in the effective values in the solar constant experienced by such a vehicle.

The external temperatures experienced by such a vehicle are shown in Figure 2-32 . By suitable internal modifications, it is estimated that the current thermal design would enable penetration to 2.0 AU. The addition of a heated battery with a radioisotope heating element, would probably increase this



NOTE: BATTERY HEATERS ON,
AT A CONSTANT RATE
OF 20 WATTS (TOTAL)

SUNRISE = TIME ZERO, AT
WHICH TIME EQUIPMENT
IS TURNED OFF

Figure 2-28. Mars Capsule Study Model Temperatures During Summer with all Equipment Turned Off

NOTES:
1) SPHERE HELD ABOVE SURFACE BY LOW CONDUCTIVITY SUPPORTS.
2) TOTAL INTERNAL POWER DISSIPATION = 30 WATTS CONSTANT.
3) K/ℓ = SUPERINSULATION CONDUCTIVITY BTU/HR-FT^2-°F, IN SEALED, EVACUATED BATTS.
4) δ = APPROXIMATE INSULATION THICKNESS, INCHES.
5) EXTERIOR EMISSIVITY = 0.8.

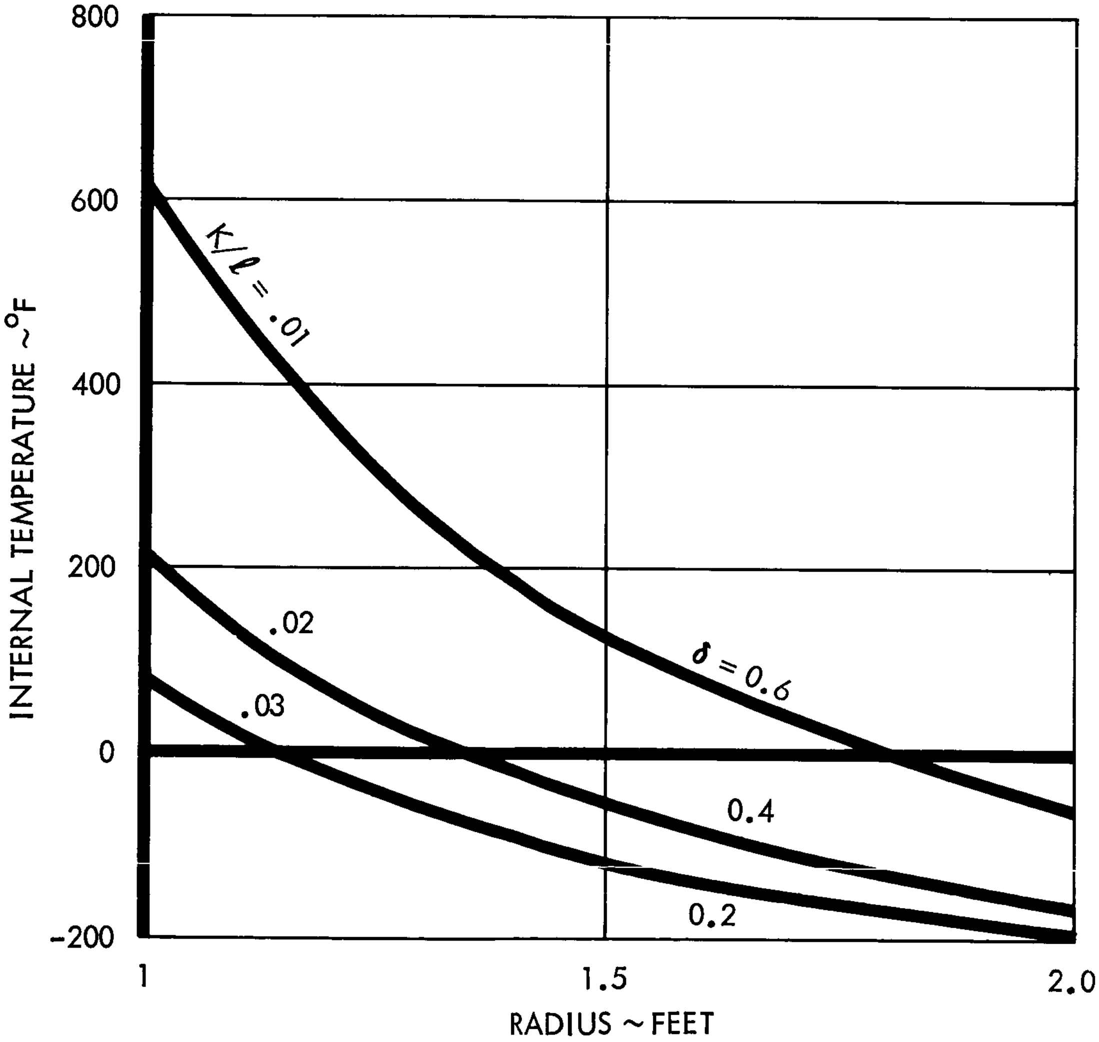


Figure 2-29. Temperatures and Radii of Insulated Spheres at the Dark Side of the Planet Mercury

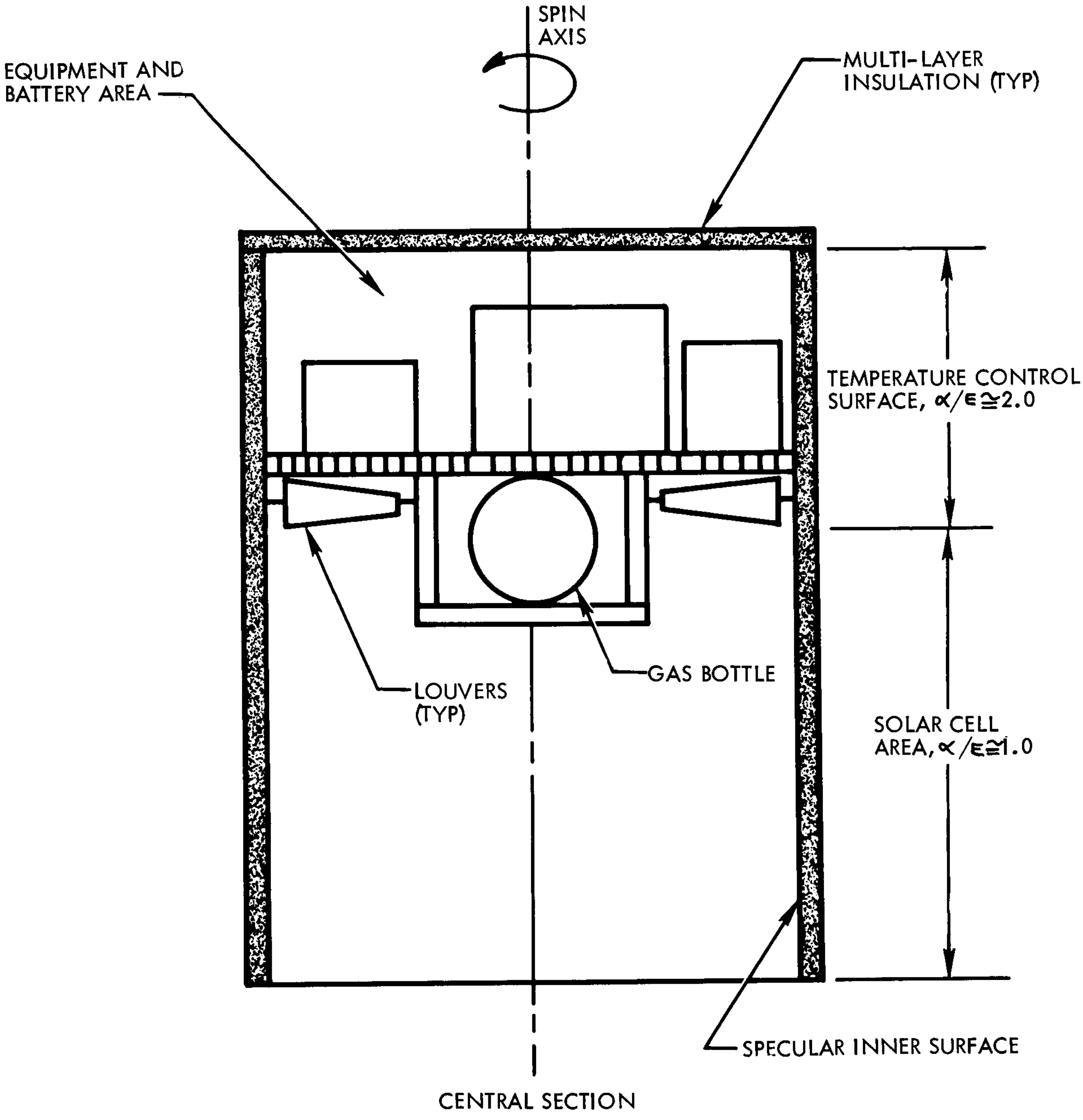


Figure 2-30. Pioneer-Type Vehicle (Modified for use as Asteroid-Belt Probe)

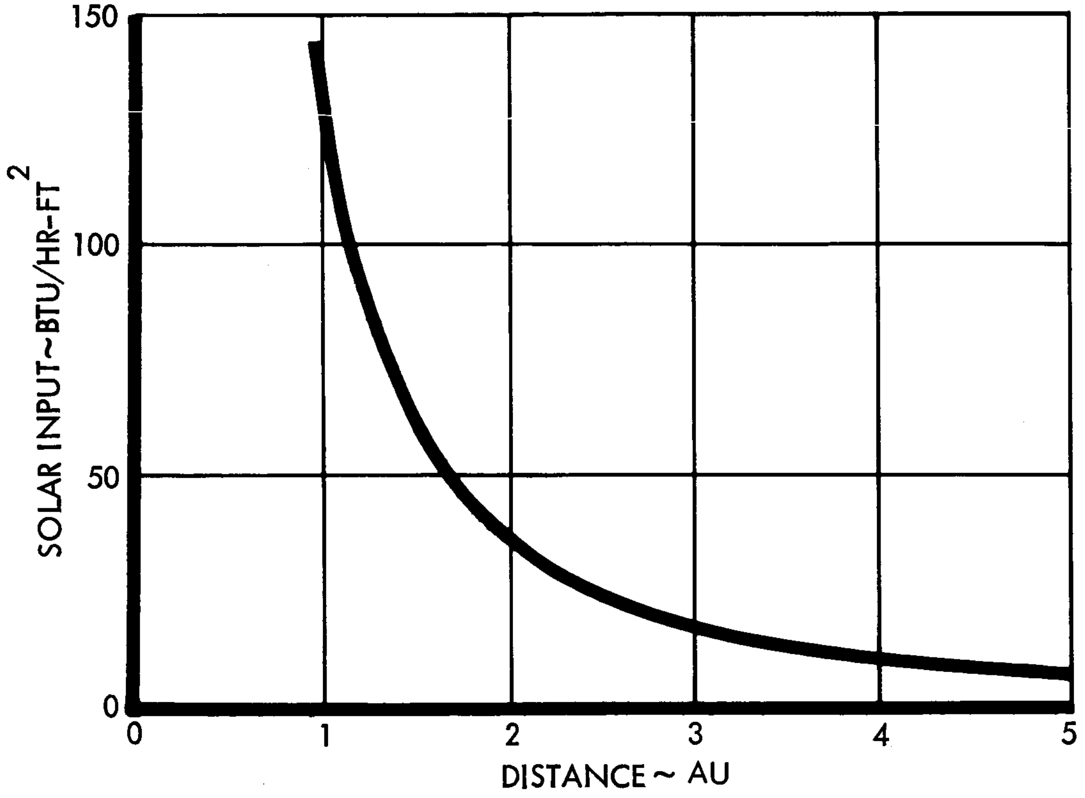


Figure 2-31. Value of Effective Solar Input for Spin-Oriented Vehicles, Spin Axis Perpendicular to Ecliptic

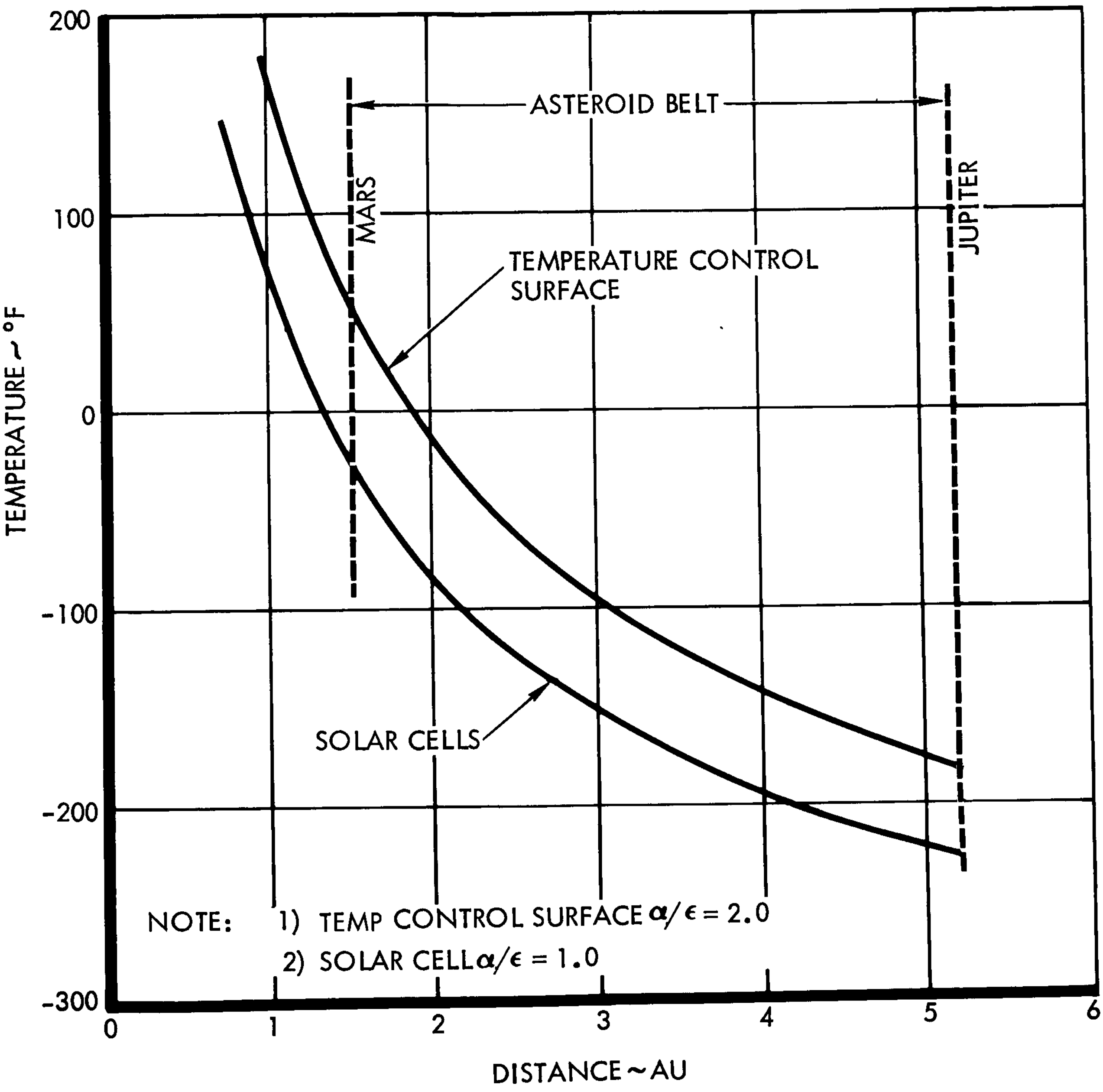


Figure 2-32. External Temperatures of Pioneer-Type Asteroid Belt Probe

penetration to about 3.0 AU since equipment heating would be less dependent on the availability of power from the solar cells.

To summarize the potential applications for a heated low temperature battery in space, the following listing of potential missions is included to provide the reader with knowledge of the breadth of the requirement for this type of battery.

- Deep Space Probes
    1) Pioneer missions beyond 2.0 AU
    2) Jupiter missions
    3) Asteroid-belt probes
- Planetary and Lunar Landers
    1) Mars hard-lander experiment packages
    2) Mars soft-lander vehicles
    3) Lunar landers experiencing lunar days and nights
    4) Jupiter atmospheric penetration probes or hard lander experiment packages
    5) Jupiter soft lander vehicles
    6) Any other lander experiencing extreme temperature cycles or low temperature environments
- Planetary and Lunar Battery Standby and Storage in Uncontrolled or Semicontrolled Environments
    1) Beacons
    2) Emergency batteries for survival
    3) External energy storage for crowded space laboratories
    4) Other applications paralleling Earth Arctic conditions
    5) Earth orbiters with batteries located in remote areas.

The above list is by no means all-inclusive but does include the missions being planned or considered for the future. The requirement for a low temperature battery in these missions is premised not merely on the need to maintain a battery at a normal operating temperature of 4 to 38^{o} C but also on the high probability that this battery will simplify the thermal control system design for these missions and may, in fact, make it possible to use a high energy battery where it would otherwise not be practical. More detailed analyses of some of these missions are contained in the Phase II analysis and will show the effect achieved with a small amount of heat on the probable performance of a battery in these types of missions. The major considerations which will be included in this study are

a) Mission requirements

b) Thermal systems and tradeoffs

c) Power system tradeoffs

d) Heating methods tradeoffs

e) Heated battery system design.

2.5 LITERATURE SEARCH AND INDUSTRY SURVEY CONCLUSIONS

On the basis of the technology found pertaining to the use of batteries in low temperature space applications, the following conclusions were drawn before entering the Phase II analysis:

a) Numerous potential applications exist for low temperature battery designs in space missions. The applications fall into three general areas:

   1) Deep space probes

   2) Planetary and lunar landers

   3) Planetary and lunar battery standby and storage in uncontrolled and semicontrolled environments

   These categories require a low temperature battery for either extended periods of operation in low temperature environments or periods of operation in low-temperature/high temperature cyclic environments in which the battery cannot be controlled totally with the spacecraft thermal control system.

b) Present battery technology limits acceptable operating temperatures to a temperature range of 4 to 38° C with minimum battery survival temperatures to -54° C. This range for operation can be extended but the penalties for extension are severe in terms of life and energy/pound. Extension of the -54° C limit cannot be justified on the basis of technology reported to date.

c) Heat sources evaluated in this survey which are applicable to low temperature battery applications include the following:

   1) Electrical heating using surplus power system energy

   2) Radioisotope heating using an active thermal control regulator

   3) Chemical cartridge heating for limited time periods and under limited environmental conditions.

   The remaining methods of heating analyzed offer very little merit for application to a low temperature space battery. Of the most promising sources listed above, the constant heat sources appear most attractive for major emphasis in the Phase II analysis. However, a limited amount of additional analysis effort was expended to evaluate the other heating methods to the point where the limits of technology could be defined, but implementation of these methods into designs was not attempted.

d) The literature reviewed on thermal control systems was not conclusive. There were no promising methods of controlling heat delivered from a constant source to a black box in cycling environments. Therefore, it was decided to place additional emphasis, in the Phase II study, on analyzing potential thermal controllers and developing the necessary theoretical models to verify their feasibility.

This concludes the Phase I literature search, industry survey, and preliminary conclusions which premised the Phase II analysis.

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3. ANALYSIS

This section of the report deals with the analyses which were conducted to determine the system requirements for:

a) Low temperature environments in space

b) Heat sources for batteries in space environments

c) Control of the heat transmitted to and from a battery during its operation in space, and

d) Development of models for types of space missions which appear most in need of heated low temperature battery designs.

The analyses, reported herein, were necessarily brief because of the limitation in the scope of the study, but an attempt was made to cover as many of the battery design problems and interfaces as possible and to develop the technology which was available into a feasible design for a low temperature battery for space applications.

3.1 SYSTEM REQUIREMENTS FOR LOW TEMPERATURE ENVIRONMENTS

3.1.1 Battery Requirements

A study was undertaken to determine the requirements for batteries in various types of space missions involving low temperature environments. In order to estimate the size of batteries, several mission power requirements analyses were made. These are typified by the analyses reported herein for a 20-watt Mars lander, a 100-watt Mars lander, a 100-watt Jupiter flyby and a 1000-watt roving vehicle. Table 3-1 contains the information required to generate a power profile and power loading schedule for the 20-watt lander; it is estimated that a 3 ampere-hour (approximately 80 whr) battery would be required to support the mission. Table 3-2 contains similar information for the 100-watt lander; it is estimated that a 20 ampere-hour (approximately 500 whr) battery would be required. Table 3-3 represents the 100-watt flyby, in which the battery would be used primarily to deliver transient loads and intermittent pulses; it is estimated that 6 ampere-hour (approximately 160 whr) battery would be required. Table 3-4 represents the 1000-watt roving vehicle in which the battery size could vary drastically; it is estimated that a

Table 3-1. Mars Lander, 20-Watt Power Source

ASSUMPTIONS:

20-Watt Solar Array Power Generator

Data Storage Unit	0.3 watt
Digital Telemetry Unit	1.0 watt
Timer	0.4 watt
Command Receiver	1.5 watts
	3.2 watts Continuous Loads

3.2/0.75 = 4.3 watts (0.75 = converter efficiency)

Experiments = 5 watts

Transmitter = 60-watt input (15 to 20 watts radiated)

23-hour, 56-minute day ≅ 24 hours

Daylight period = 11 hours, 58 minutes ≅ 12 hours

Lander capsule relays data to orbiter in a 6-hour orbit. Interrogates once per orbit.

12.0 hours x 20.0 watts = 240 whr input per day.

(4.3 watts) (24 hours) = 100.4 whr steady state loads per day

139.6 watt-hours available for battery charge and non-continuous loads

139.6 x 0.65 = 90.5 whr stored (0.65 = battery efficiency)

Transmitter = 60 watts

Experiments = 5 watts

60 + 5 = 65 watts 65/0.85 = 71 watts non-continuous loads

90.5 watt-hours/71 watts = 1.28 hour = 76.5 minutes/day of non-continuous load operation

4 interrogations per day of 19 minutes each.

Battery Sizing
52 whr night loads (continuous)
44.5 whr night transmitter loads
96.5 whr

96.5 whr/28 volts = 3.4 ampere-hours

Use 5 ampere-hour NiCd battery at approximately 70% depth of discharge

Battery Weight = (22 cells) x (0.55 lbs/cell) x (1.2 packaging factor) = 14.6 pounds

Battery Heat Evolution (estimated)

During 4.3-watt discharge = 0.4 watt
During 69.3-watt discharge = 7 watts

Table 3-2. Mars Lander, 100-Watt RTG Power Source

ASSUMPTIONS:

100-Watt RTG Power Generator

Desire TV picture, 1.5 x 10^6 bits per picture

Non-Directional Antenna

Lander broadcasts to 6-hour orbiting satellite

Transmitter power = 100 watts (1,500 bits/second estimated)

LOADS (Cont)

Digital Telemetry Unit	2.0 watts
Data Storage Unit	2.0 watts
Command Receiver	2.5 watts
Timer	0.4 watt
	6.9 watts

6.9/0.75 = 9.2 watts (0.75 = converter efficiency)

9.2 watts x 24 hours = 221 whr steady state loads

Energy accumulated/day = 100 watts x 24 hours = 2,400 whr

Less Standby loads 221

2,179 whr

Times 0.65 (battery efficiency) x0.65

Available for TV transmission 1,416 whr

100-watt transmitter requires 350-watt input.

350/0.85 = 412 watts (0.85 = converter efficiency)

1,416 whr/412 watts = 3.4 hours

(3.6 pictures/hour) (3.4 hours) ≅ 12 pictures/day

or ≅ 3 pictures/orbit

Battery storage requirements are:

1,416 whr/4 = 354 whr

354 whr/28 volts = 12.7 ampere-hours

Battery Weight = (22 cells)(1.9 lbs/cell)(1.2 packaging factor) = 51 lbs

Battery Heat Evolution

During 9.2-watt discharge ≅ 0.9 watt

During 421.2-watt discharge ≅ 42 watts

Table 3-3. Jupiter Flyby, 100-Watt RTG Power Source

ASSUMPTIONS:

100-watt RTG Transmitter
10-watt Transmitter
16-foot Dish-Antenna
at 10 watts -- 850 bits per second
at 100 watts -- 8,500 bits per second
Basic mission is photographic plus data measurement.

LOAD ASSUMPTIONS:

Camera Operation	1 picture	60 watts	5 minutes
Develop and Dry	1 picture	70 watts	30 minutes
Photo Readout	1 picture	30 watts	3 minutes
Transmit	1 picture	350 watts	

$$\frac{1.5 \times 10^6}{8.5 \times 10^3} = 1.77 \times 10^2 = 2.94 \cong 3 \text{ minutes}$$

Alternatively, if a 10-watt transmitter is used

Photo Readout	30 watts	30 minutes
Transmit	35 watts	

Steady-State Loads

Attitude Sensors	1.0 watt
Command Distribution Unit	0.5 watt
Command Receivers	2.5 watts
Decoders	1.0 watt
Digital Telemetry Unit	1.0 watt
Data Storage Unit	0.5 watt
Experiments	5.0 watts
	11.5 watts

Average conversion efficiency = 0.75

Total steady-state power = 11.5/0.75 = 15.3 watts

Table 3-3. Jupiter Flyby, 100-Watt RTG Power Source (Continued)

CONTINUOUS DATA TRANSMISSION -- 10-WATT TRANSMITTER

35 WATTS/0.85 = 41.1 WATTS

TOTAL POWER WHILE NOT PHOTOGRAPHING = 56.4 WATTS

TOTAL REMAINING POWER FOR BATTERY RECHARGE =

100.0 - 56.4 = 43.6 WATTS

POWER REQUIRED DURING PHOTOGRAPHIC CYCLE

$\frac{\text{6.5 WATTS}}{0.7}$ (EXPERIMENTS OFF)	= 9.3 WATTS
10-WATT TRANSMITTER (COMMAND RECEIPT, HOUSEKEEPING, ETC.)	= 41.1 WATTS
	50.4 WATTS
CAMERA = 60 WATTS/0.85	= 70.6 WATTS
	121.0 WATTS

NET POWER FROM BATTERY = 21.0 WATTS

DEVELOP -- STEADY-STATE LOADS, 10-WATT TRANSMITTER	50.4 WATTS
DEVELOPER 70 WATTS/0.85	82.5 WATTS
	132.9 WATTS

NET POWER FROM BATTERY = 32.9 WATTS

PHOTO READOUT AND TRANSMIT

STEADY-STATE LOADS	9.3 WATTS
PHOTO READOUT, 30 WATTS/0.8	37.5 WATTS
TRANSMIT 350 WATTS/0.9	390.0 WATTS
	436.8 WATTS

NET POWER FROM BATTERY = 336.8 WATTS

TOTAL BATTERY ENERGY REQUIRED

21.0 WATTS	X	5 MINUTES	=	105 WATT-MINUTES
32.9 WATTS	X	30 MINUTES	=	990 WATT-MINUTES
336.8 WATTS	X	3 MINUTES	=	1000 WATT-MINUTES
TOTAL				2096 WATT-MINUTES
			OR	35 WATT-HOURS

Table 3-3. Jupiter Flyby, 100-Watt RTG Power Source (Continued)

At 65% recharge efficiency, Total Input Requirement to battery = 53.8 watt-hours

At 43.6-watt input, Recharge time = 1.27 hours

Total Cycle Time

Camera	5 minutes
Develop	30 minutes
Readout and Transmit	3 minutes
Recharge	76 minutes
	114 minutes

or 1 picture plus 3.9×10^6 experiment data bits every 2 hours.

Battery size (assuming 50% depth of discharge) =

35/0.5 = 70 whr

70 whr/28 volts = 2.5 ampere-hours ≅ 3 ampere-hour battery

Weight = 7.9 pounds

However, this size is inadequate to deliver 452 watts (16.2 amperes); therefore, the battery must be sized for current delivery. A convenient short time discharge rate with adequate voltage regulation is 3C to 4C.

16.2/3 = 5.4 ampere-hours (nearest size is 6 ampere-hours)

Weight = 15.9 pounds

Heat Evolution (estimated)

	Power from Battery	Estimated Heat (watts)
Photographic	21	2.3
Processing	33	3.5
Transmit	337	72
Recharge	-54	-1 (absorbed)
Overcharge	- 5	5

Table 3-4. Mars Lander, 1 KW Power Source

Assumptions:

1 KW RTG Power Source Mounted Outboard of the Main Vehicle.

Battery Energy Storage used for peak loads

Vehicle roves on command, with low quality TV

100 x 100 lines, 3 gray levels, 1 frame/10 seconds

Vehicle halts, transmits high-quality picture, (1.5 x 10^6 bits/picture)

Oriented sunflower antenna mounted on stabilized platform over vehicle. Transmits directly to Earth.

Other Experiments -- 300 watts, used only when vehicle is halted.

Two transmitters --

1 low bit-rate -- 20 watts radiated power

1 high bit-rate -- 200 watts radiated power

Both operate at 35% efficiency.

Drive power -- 1.5 KW -- speed 2 ft/sec on level.

Antenna orientation and stabilization

Gyro reference platform	35 watts
Antenna stabilization drive (150 W avg 250-watt peak)	150 watts
TV camera -- low resolution	10 watts
high resolution	15 watts
DTU	2.5 **watts**
DSU - Record	15 watts
- Playback	40 watts

Table 3-4. Mars Lander, 1 KW Power Source (Continued)

Command Receivers	0.5 watt
Steering -- 20 watts average 250 watts peak	20 watts
Miscellaneous steady-state loads	10 watts

Power During Vehicle Travel

	Conditioned	Raw
Locomotion	1,500 watts	1,500 watts
Steady-state loads - Misc.	10	12.5
- DTU	2.5	5.0
- Command Rec.	0.5	1.5
Steering	20.0	20.0
Gyro Reference	35.0	43.8
Antenna Stabilization Drive	150.0	150.0
TV Camera	10.0	12.5
Transmitter (20 watts)	57.1	71.5
Spotlights	200.0	200.0
Total Raw Power		2,016.8 watts
Less RTG Output		1,000 watts
Stored Energy Requirement		1,016.8 watts

Base Power when Vehicle is halted

Steady-state loads - Misc.	10 watts	12.5 watts
- DTU	2.5	5.0
- Command Rec.	0.5	1.5
Gyro Reference	35.0	43.8
Transmitter (20 watts)	57.1	71.5
Total Raw Power		134.3 watts

Table 3-4. Mars Lander, 1 KW Power Source (Continued)

POWER DURING EXPERIMENT OPERATION				
BASE POWER			134.3	WATTS
DATA STORAGE UNIT	15.0	(80%) TIME	15.0	
DATA PLAYBACK	40.0	(20%) SHARING	10.0	
EXPERIMENTS	300.0		375.0	
			534.3	WATTS
POWER DURING HIGH RESOLUTION TV				
BASE POWER			134.3	WATTS
DATA STORAGE	15	(80%) TIME	15.0	
DATA PLAYBACK	40	(20%) SHARING	10.0	
TV CAMERA	15		18.7	
TRANSMITTER (200-WATTS)	570		715.0	
SPOTLIGHTS	200		200.0	
TOTAL RAW POWER			1,093.0	
LESS RTG OUTPUT			1,000.0	
POWER FROM BATTERY			93.0	WATTS

ANALYSIS OF TRANSMITTER CAPABILITY

TRANSMITTER CAPABILITY -- WITH 16-POINT DISH

20 WATTS -- ESTIMATED 3,500 BITS/SECOND FROM SURFACE OF MARS TO EARTH.

LOW QUALITY TV -- 100 x 100 x 3 -- 30,000 BITS

AT 1 FRAME/10 SECONDS, TRANSMITTER REQUIREMENTS ARE 3,000 BITS/SEC.

HIGH-QUALITY TV -- 200 WATTS $\cong$ 35,000 BITS/SECOND

REQUIRED $(1.5 \times 10^6)/(3.5 \times 10^4) = 0.43 \times 10^2 = 43$ SEC/FRAME

PICTURE TRANSMISSION $\cong < 1$ MINUTE

Table 3-4. Mars Lander, 1 KW Power Source (Continued)

PROPOSED CYCLE

VEHICLE TRAVELS PRE-SET DISTANCE OF 0 TO 50 FEET AT 2 FT/SECOND.

BATTERY DRAIN = 1.016 KW FOR 25 SECONDS ≅ 0.28 AMP-HOURS.

VEHICLE HALTS, TRANSMITS 2 HIGH RESOLUTION TV PICTURES TO SHOW PATH AHEAD IS CLEAR.

BATTERY DRAIN = (93 WATTS x 2 MIN. TRANSMISSION TIME)/
(60 MIN/HR x 28 VOLTS) ≅ 0.11 AMP-HOURS

TOTAL BATTERY DRAIN = 0.28 + 0.11 = 0.39 AMP-HOURS; HOWEVER, BATTERY MUST BE SIZED FOR PEAK CURRENT.

MAXIMUM CURRENT ≅ 1.016 KW/25 VOLTS = 40 AMPERES.

IF WE ASSUME A MINIMUM RATE OF 2C, BATTERY CAPACITY BECOMES 20 AMPERE-HOURS (MINIMUM).

LIMITING DISCHARGE TO 50% BEFORE RECHARGE EXCEPT IN EMERGENCIES, AVAILABLE CAPACITY = 10 AMPERE-HOURS.

VEHICLE WILL TRAVEL SLIGHTLY LESS THAN 1/4 MILE BETWEEN BATTERY CHARGES, USING 20 AMPERE-HOUR BATTERY.

(10 AMPERE-HOUR TOTAL)/(0.39 AMP-HOUR/50 FT) ≅ 1,270 FT.

VEHICLE HALTS -- RECHARGES.

IF VEHICLE IS AT MINIMUM POWER = 134.3 WATTS,

THEN, POWER AVAILABLE TO CHARGE BATTERY = 1,000 - 134.3
865.7 WATTS.

865.7 WATTS/32 VOLTS (22 CELL NICD) = 27 AMPERES.

CAPACITY REQUIRED (EFF. = 0.8) = 10/0.8 = 12.5 AMP-HOURS INPUT.

12.5/27.0 ≅ 0.41 HOUR ≅ 25 MINUTES (RECHARGE TIME PRIOR TO NEXT 1/4-MILE TRAVEL).

POWER REQUIREMENT CHANGES WHEN VEHICLE LOSES SIGHT OF EARTH (I.E., ANTENNA ORIENTATION SIGNAL). COMPUTER WILL BE REQUIRED TO PREDICT EARTH POSITION RELATIVE TO VEHICLE WHEN EARTH COMES INTO VIEW.

Table 3-4. Mars Lander, 1 KW Power Source (Continued)

Quiescent Power Level

Miscellaneous	10 watts
DSU	0.5
DTU	1.0
Command Receiver	2.5
Gyro Reference	35.0
Computer	150.0
Antenna Slew	10 watts (average) 250-watts peak

Still below 1 kw, battery will charge overnight. If vehicle uses experiments during recharge period, recharge period is lengthened to

Power available to charge battery = 1,000-534.3=465.7 watts

465.7 watts/32 volts = 14.6 amperes

12.5 ampere-hours/14.6 amperes = 0.86 hours ≅ 51 minutes

Battery Heat Output Estimates

Vehicle Travel Phase (25 seconds)	300 watts
High Resolution TV (2 minutes)	10 watts
Recharge (high rate -- 25 minutes)	5 watts (absorbed)
Recharge (experiments on -- 56 minutes)	2.5 watts (absorbed)

battery on the order of 20 ampere-hours (600 whr or more) would be required.

Figures 3-1 and 3-2 show the power profiles for two of the above missions. From the profiles the power requirement can be visualized. Analyzing the battery requirement further, we see that the typical battery size is somewhere in the range of 3 to 30 ampere-hours for most applications. This requirement when compared with a mission duration fixes not only the size but the type, because battery life is one of the main limiting factors in selection of a battery for space applications. Typically, any mission requiring a battery life of less than a year could use a silver-cadmium, nickel-cadmium, silver-zinc or zinc-mercury battery; if the period is extended to two years, it is possible that a silver-cadmium battery could be used and the nickel-cadmium battery could definitely be used. If the mission duration is more than two years, the test data published throughout the literature selects the nickel-cadmium battery. The only exception to these criteria would be the requirement that a battery be transported unactivated to the end of the mission and then activated. Present technology has demonstrated that this type of operation using a silver-zinc battery may have a life of 3 to 5 years.

Based upon the above analysis, the next logical step is to compare the sizes of batteries with their watt-hour and operating temperature requirements. Under the assumption that the data presented in the Phase I analysis and the conclusions drawn are valid, the lower temperature limit selected has been restricted to -18° C for the purposes of this comparison. Figure 1-3 presents a general summary of the available sizes for the various types of batteries. The nomograph was used to size batteries in accordance with the watt-hour requirements determined above in order to determine heat required to sustain battery temperatures in the operating range in a low temperature environment.

An additional step in the analysis was determination of the amount of heat which the battery could be expected to supply during charge or discharge. No heat would be available from a primary battery during the charge cycle; the thermal dissipation during the discharge cycle would be similar for both primary and secondary batteries. Figure 3-3 shows the estimated available internal heat generated versus overcharge and

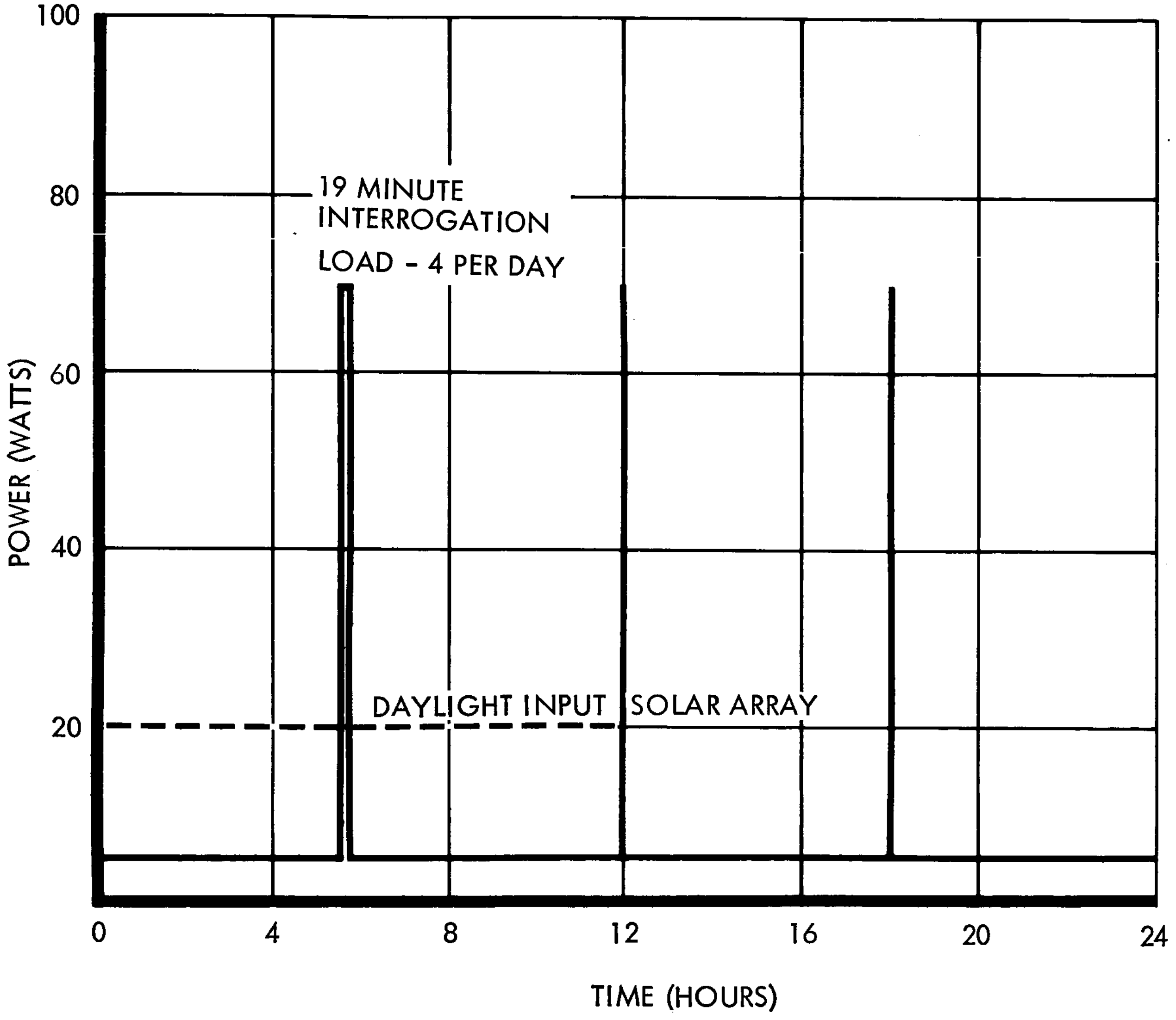


Figure 3-1. Mars Lander, 20-Watt Power Source, 3 Ampere-Hour Battery, Real-Time Data Readout

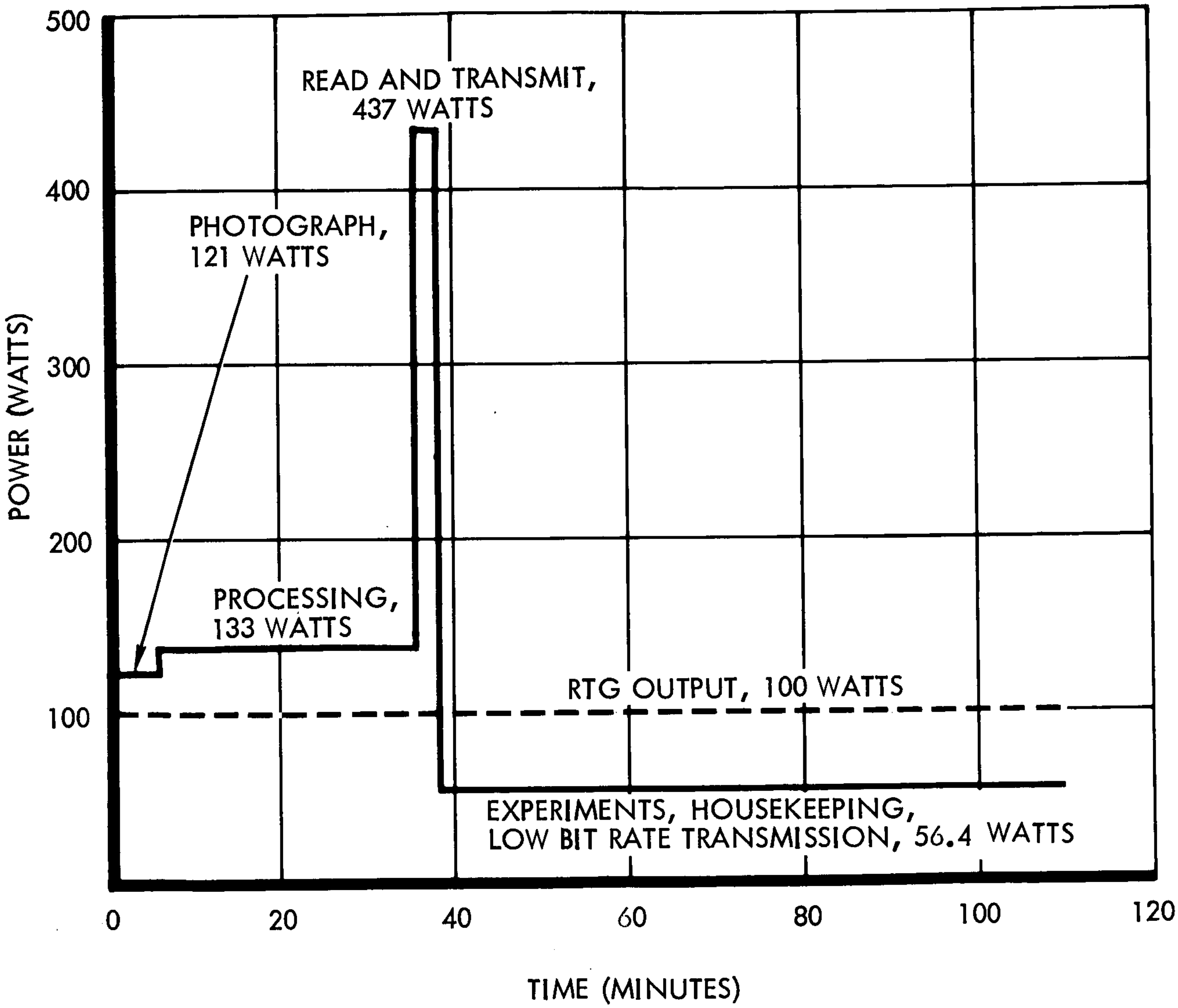
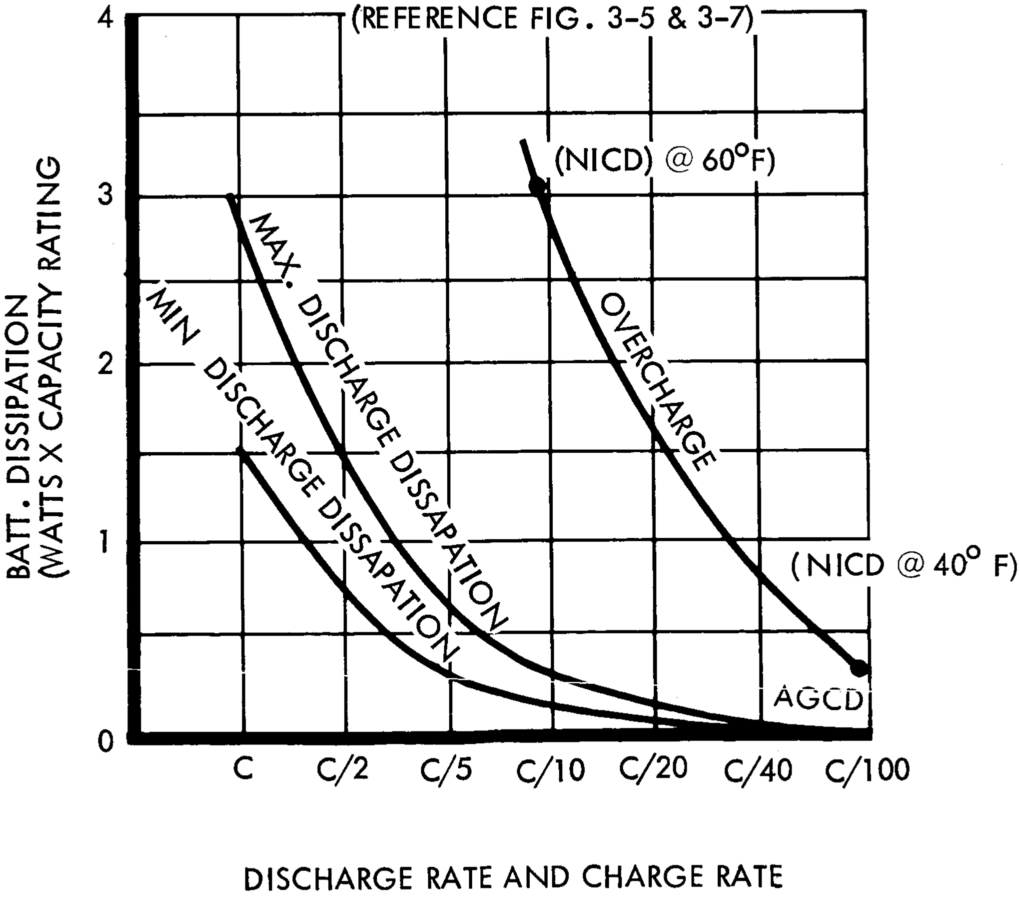


Figure 3-2. Jupiter Flyby 100-Watt Power Source, Load Profile



EX. 6 AH BATTERY ESTIMATED SELF HEAT

DISCHG AVG DISS. AT C/10 RATE = 1-2 WATTS
OVERCHG AVG DISS AT C/40 RATE = 5-6 WATTS
OVERCHG AVG DISS AT C/100 RATE = 1-2 WATTS

THE NET REQUIREMENT WOULD BE AN ADDITIONAL 8-10 WATTS NOMINAL TO MAINTAIN THE BATTERY TEMP. AT 40°F IN A-20°F ENVIRONMENT AND 20-30 WATTS IN A-300°F ENVIRONMENT

Figure 3-3. Estimated Average Internal Heat Generated Versus Overcharge and Discharge Rate, 28 Volt Secondary Space Battery discharge, for a 30-volt secondary space battery. The curves shown are approximations and indicate the order of magnitude of heat expected from the battery to support temperature control. The accuracy of the curves is poor for any specific design because the rates, condition of the battery, depth of discharge and other pertinent factors would have to be considered. Figure 3-3 includes an example of a 6 ampere-hour battery to permit the reader to compare the order of magnitude of the heat available from the battery during normal operations.

To provide balanced cell operation and to avoid local hot spots when the battery is heated, it is desirable to maintain the cell differential temperature from top to bottom and across the cell stack within 5.6° C. Temperatures within the differential must be maintained during charging to prevent cell damage and are desirable for discharge as well to avoid degradation of separators due to hot spots.

Once the foregoing information was accumulated, we were in a position to proceed into the thermal analysis to establish system requirements for a low temperature battery. The analysis, in the next paragraph, is based upon the conclusions, stated in the discussion of the search of the literature, that the battery operating temperature should be limited to 4 to 38° C, and that under no circumstances should the battery be allowed to drop below -54° C. However, additional analyses were performed in an attempt to examine the effect of an extremely low temperature environment on the battery design in the event that new technology can be developed to support the hypothesis that batteries can indeed be frozen and maintained at extremely cold temperatures, warmed up, and used without serious degradation.

3.1.2 Thermal Requirements

This paragraph presents the analysis which was undertaken to determine the heat required to warm batteries from low to useful operating temperatures and to sustain batteries within a useful operating temperature range when the surrounding environment is extremely cold. As previously noted, it is not expected that a battery placed in a normal operating spacecraft would drop below -40° C. Figures 3-4 and 3-5 show the approximate heat required to warm a battery from the -29° C ambient to +4° C in 4 hours, and to maintain the battery at a minimum of 4° C in

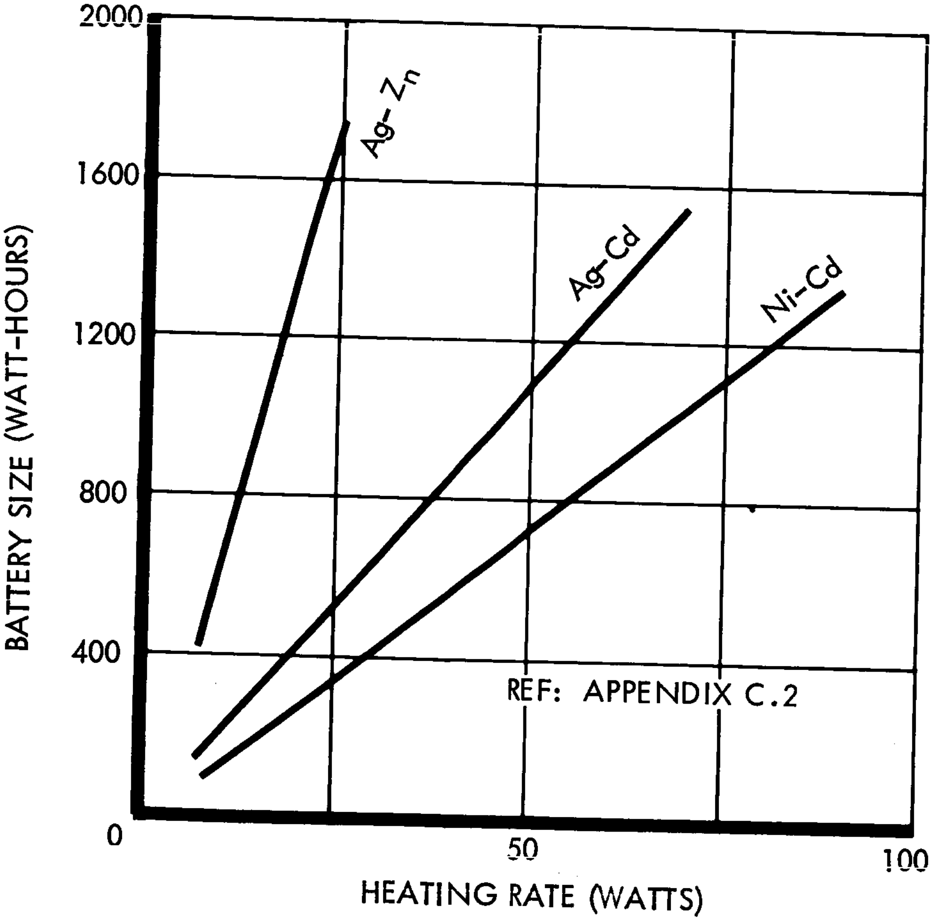


Figure 3-4. Heat Rate Required to Warm a Battery to 40°F in 4 Hours Starting at -20°F

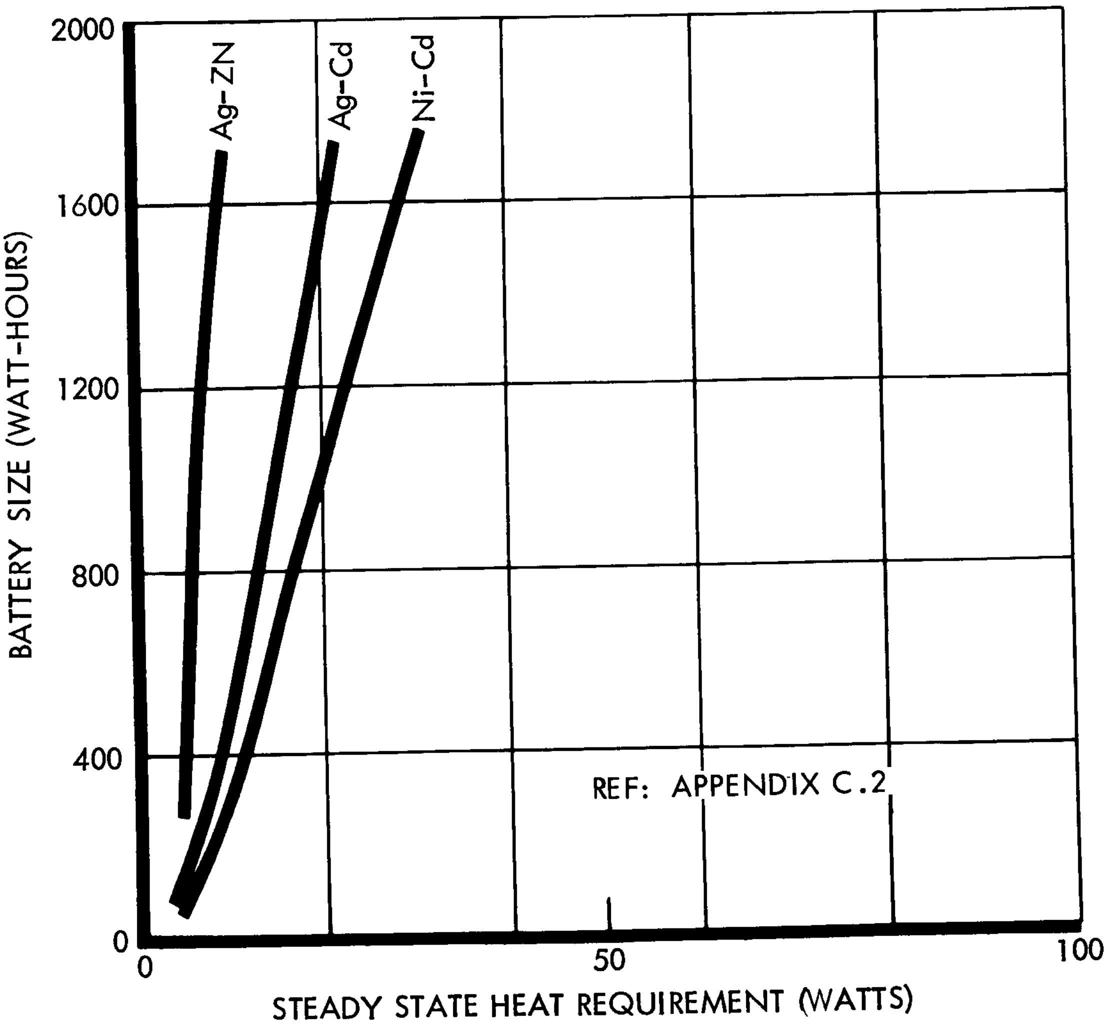


Figure 3-5. Steady-State Heat Requirement to Maintain a Battery at 40°F in a -20°F Environment

the ambient. The curves in the figures may be shifted in either direction along the horizontal axis by a factor of approximately two depending upon specific battery designs and spacecraft environmental control system designs. However, the curves indicate that the battery sizes which are anticipated for deep space probes and other space missions (from 50 to 1600 whr) would normally require from 5 to 30 watts of heat to maintain the minimum operating temperature in a -29° C environment. The amount of heat which would be required to warm a battery in a period of 4 hours is significantly higher, even when continuously applied over the 4-hour period. Sample calculations used to derive the curves are contained in Appendix C.

Extending this analysis further to a more extreme environment (-184° C was chosen based on the Work Statement goals for potential low temperature battery operation), a theoretical computation was made to derive the amount of heat required to warm a battery from a -184° C ambient to +4° C in 4 hours and to maintain the battery at a minimum of +4° C in the ambient (Figures 3-6 and 3-7). Again the curves are indicative of the order of magnitude of heat required but can vary by a factor of two along the horizontal axis depending upon specific designs.

From this analysis it is seen that the heat required to keep the smaller batteries (0 to 600 whr) warm in the given environment would be on the order of 20 to 50 watts. The amount of heat required to warm a battery from -184° C in 4 hours would be higher for small batteries and increases rapidly with size to the point where attempting to heat a battery of more than 5 ampere-hours is not particularly feasible. However, for the battery sizes anticipated for most space probes, it does appear feasible to use special mounting configurations (isolating the battery from its environment) to maintain a battery in extremely low temperature environments, within a useful operating range, with less than 50 watts.

Based on these analyses, the following thermal requirements are presented as criteria for designing a heated low temperature battery.

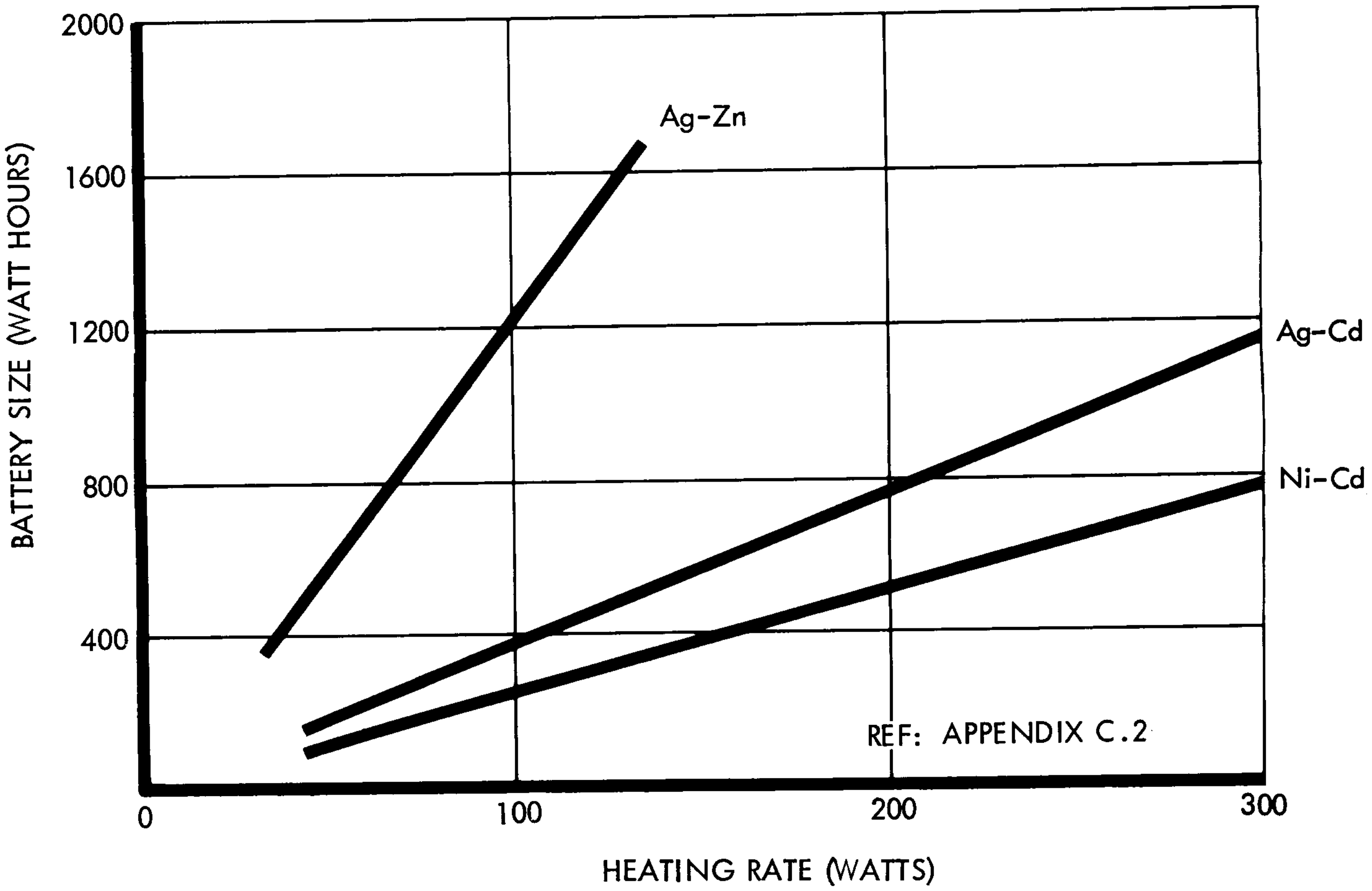


Figure 3-6. Heat Rate Required to Warm a Battery to 40°F in 4 Hours Starting at -300°F

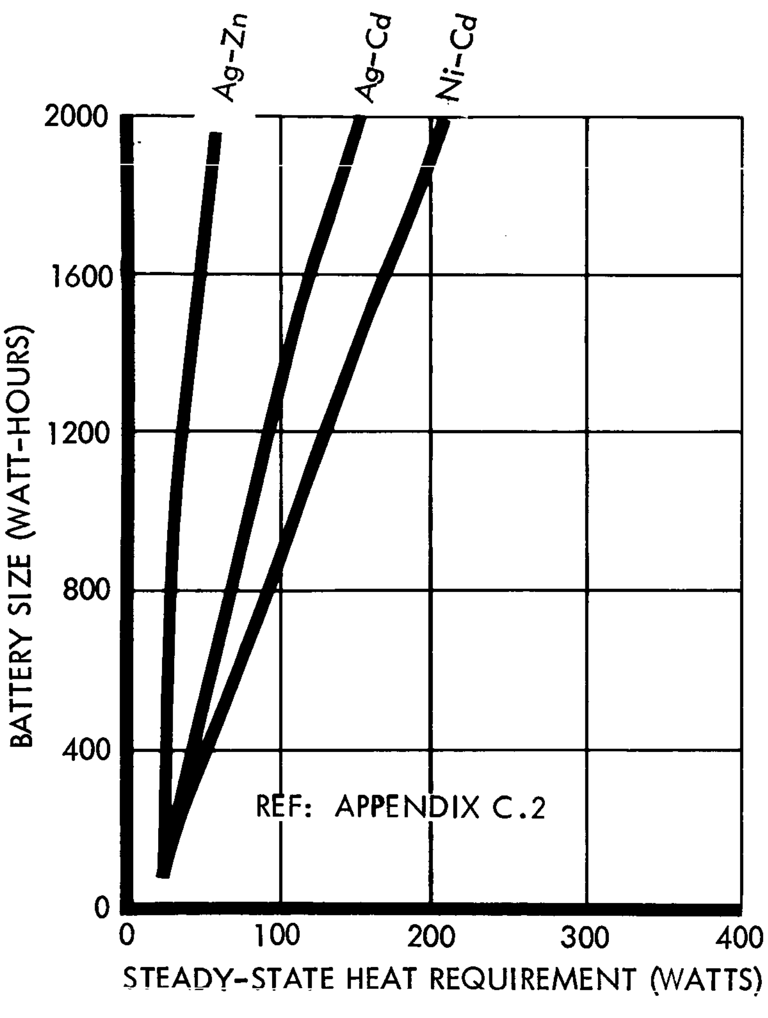


Figure 3.7 Steady-State Heat Requirement to Maintain a Battery at 40° F in a -300° F Environment

The battery heater should be capable of supplying 5 to 50 watts of heat and of removing the heat from direct application to the battery in the near-Earth space environment or in thermally cyclical environments. The heat should, preferably, be provided by continuous heat sources or electrical power dissipation sources. Only in very limited circumstances would short-term warmup heating be desirable because once the batteries are heated to the operating temperature they must be sustained within the operating temperature range. From the thermal dissipation characteristics of the battery, shown in Figure 3-3, it is apparent that the battery cannot accomplish this function by itself.

The study of thermal requirements has shown that a battery being heated in a space environment is an entirely different system from that usually encountered in cold terrestrial operations because the means of heat transfer is primarily by conduction through the mounting base or mounting offsets and not by convection. Radiation, when insulation is used, causes minimal heat loss compared to the conduction path. The thought occurs that it would be wise to totally insulate the battery and keep it completely isolated from its environment. In fact, such a procedure could not succeed because the battery could not sustain dynamic environments during launch and re-entry; it must be securely fastened to the space vehicle to avoid serious damage to the battery and the vehicle. Therefore, the heat losses cannot be eliminated but merely minimized.

In summary, the discussion of Section 2, concerning the expected thermal environment of the battery in the space vehicle, is noted. The conclusions presented in this report are, essentially, that in normal use, the battery in a space vehicle will not encounter temperatures below -40° C, and that in applications in which a battery might encounter temperatures below -40° C, extremely complicated thermal control systems will be required to maintain electronics as well as battery temperatures in a useful operating range. The intent, then, of placing a heater in a battery is to aid the thermal control system and to provide the means to isolate the battery from nominally cold surrounding temperatures.

3.1.3 Environmental Requirements

The environmental qualification requirements for space probes specified below are derived from typical NASA specifications for space probe missions. It is again noted that the levels specified are qualification levels which are simulated conditions more severe than service environments in order to provide assurance of detecting design deficiencies. However, the specified conditions are not expected to exceed normal design safety requirements.

3.1.3.1 Vibration

The battery would normally be vibrated in each of three mutually perpendicular axes with one axis aligned in such a manner that the acceleration vector is parallel with the thrust axis of the assembly. The battery would be subjected to the vibration excitation defined below, applied and measured at the mounting point of the battery:

a) Complex Motion Vibration

	Sinusoidal		Gaussian Random	
Axis	Frequency (cps)	Acceleration (g, 0-peak)	Frequency (cps)	PSD Level (g^2/cps)
All	10 to 500	5.0	20 to 500	0.03
	500 to 2000	10.0	500 to 2000	0.07

b) Spacecraft Mode Response Vibration

Axis	Frequency (cps)	Acceleration (g, 0-peak)
All	50 to 250	20

3.1.3.2 Linear Acceleration

While in a launch operative mode, the battery would normally be subjected to a sustained acceleration environment shown in the table below:

Unit Orientation	Acceleration (g)	Duration (minutes)
Mounted in accordance with Figure 3-8	50.0	3.0

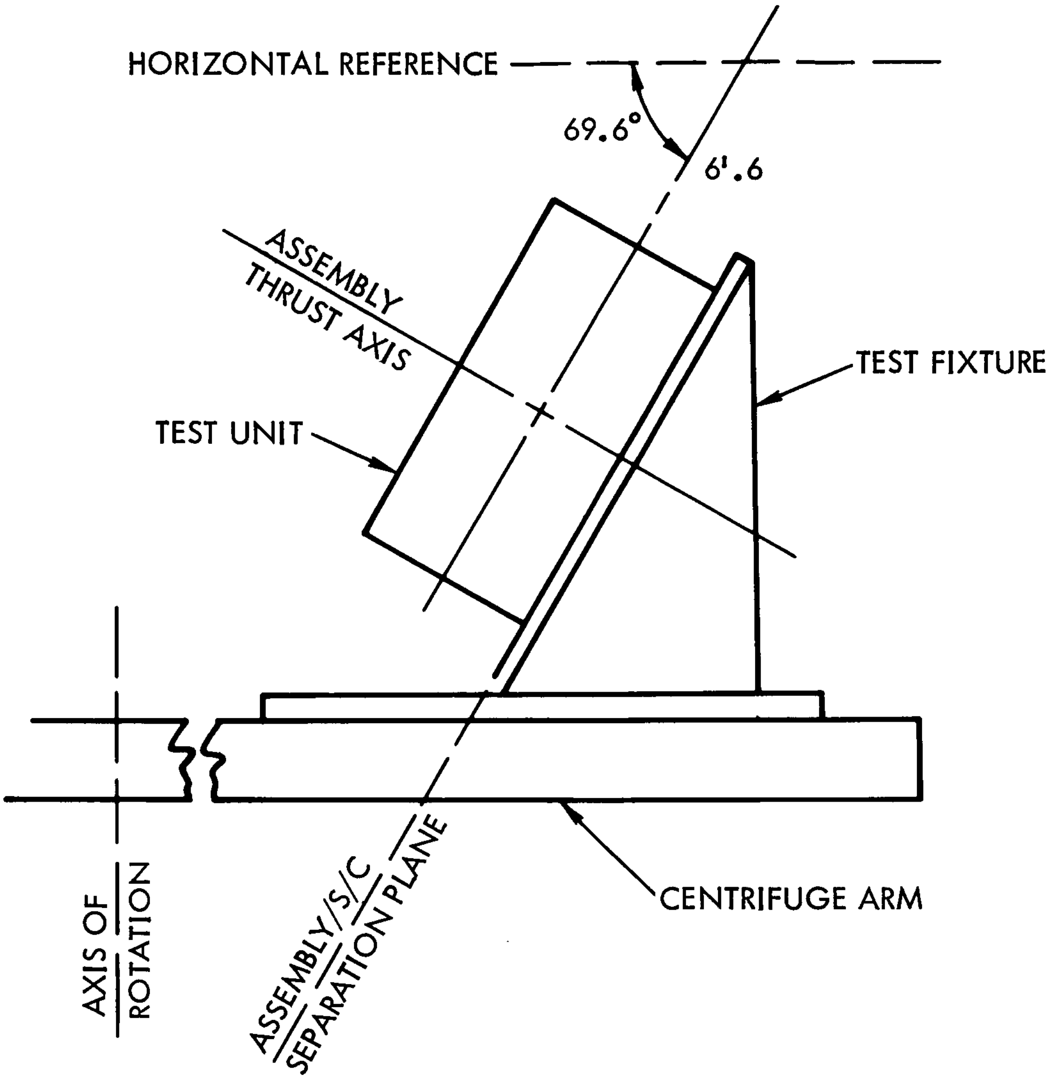


Figure 3-8. Unit Orientation, Linear Acceleration Test

The applied acceleration field would be measured at the midpoint of the assembly. The battery, attached to the centrifuge through its normal spacecraft mounting hardware at the separation frame between the spacecraft and the assembly, would be mounted in such a manner that the applied acceleration vector would equal the resultant acceleration vector of the thrust acceleration and the spacecraft spin-induced acceleration. The assembly would be mounted on an acceleration test fixture in such a manner that the inboard edge of the assembly, as mounted in the spacecraft, would face the inboard edge of the centrifuge test facility.

3.1.3.3 Shock

While nonoperating, the battery would be subjected to a total of nine shocks — three shocks along each of three coordinate axes. The shock machine would develop a terminal peak sawtooth acceleration pulse of 6 plus 1 minus 0 microseconds duration with a peak acceleration of 50 ± 5 g. Shocks would be applied through the normal mounting provisions of the battery.

The above specifications are supplemented by the following requirements of an orbiting vehicle mission typified by the NASA Orbiting Geophysical Observatory. The qualification requirements for this mission are as follows:

a) Vibration

1) Sinusoidal Vibration

Sinusoidal vibration is applied individually along each of three coordinate axes at a sweep rate of one-half octave per minute through all frequencies shown in the following tabulation:

Sinusoidal, One Sweep per Axis

5* to 250 cps	250 to 400 cps	400 to 3000 cps
3.5 g rms	6.5 g rms	13 g rms

2) Random Vibration

Random vibration would be applied along each of three coordinate axis individually in accordance with the conditions tabulated below:

---

*Vibration displacement limited to 0.4-inch double amplitude.

Random

20 to 2000 cps**	0.1 g^2/cps	12 minutes per axis

b) Acceleration

The OGO battery was subjected to an acceleration test while operating under simulated boost phase conditions of load and acceleration. The battery was centrifuged for three minutes in each direction along each of three coordinate axes as follows:

Thrust axis Y-Y	13 g
Transverse axis X-X	3 g
Transverse axis Z-Z	3 g

c) Shock

The OGO battery shock levels in the standby nonoperating condition were as follows:

Thrust axis	45-g terminal peak sawtooth, 2.2 msec pulse period
Transverse axes (X-X and Z-Z)	22-g terminal peak sawtooth, 4.4 msec pulse period

One shock was applied along each axis through the normal battery operating base.

It is concluded from the above specifications that the expected environmental requirements for a battery for a space probe would be as follows:

a) Complex Motion Vibration

	Sinusoidal		Gaussian Random	
Axis	Frequency (cps)	Acceleration (g, 0-peak)	Frequency (cps)	PSD Level (g^2/cps)
All	5 - 250	3 - 5	20 - 2000	0.1
	250 - 400	5 - 6.5		
	400 - 3000	10 - 13		

**Rolloff above 1000 cps at 13db per octave on an acceleration basis. The overall level was greater than 11.3 g rms.

Notes:

1) Vibration displacement shall be limited to 0.4 inch double amplitude.

2) The sinusoid shall be swept at a rate of 0.5 octave per minute.

3) The random vibration excitation shall be rolled off at 12 decibels per octave (on an acceleration basis) above 1500 cps.

4) The true g rms level for random excitation, excluding sinusoidal energy, over the frequency range of 20 to 2000 cps shall be maintained within ±10 percent.

b) Spacecraft Mode Response Vibration

Same as above

Axis	Frequency (cps)	Acceleration (g, 0-peak)
All	50 - 250	20

c) Linear Acceleration

The typical acceleration requirement which should be considered in designing a battery would be an acceleration requirement of 50 g with the assembly mounted as shown in Figure 3-8 for a duration of 3 minutes.

d) Shock

The design of a battery for space probe applications should consider the shock requirement discussed below:

Thrust Axis	50 ± 5 g 2 to 6 msec pulse period
Transverse Axes	25 g 4 msec pulse period.

Going beyond these requirements, the study also investigated the dynamic environment of soft landers and, briefly, of hard landers. The requirements for soft landers are typified by information obtained on the LEM vehicle and Surveyor as follows:

a) LEM Vibration

	Sinusoidal		Gaussian Random	
Axis	Frequency (cps)	Acceleration (g, 0-peak)	Frequency (cps)	PSD Level (g^2/cps)
All	10 - 500	5	10 - 500	0.03
			500 - 2000	0.07

It is noted that these requirements fall within the space probe requirements established above.

b) LEM Shock

15 g peak sawtooth

11 ±1 msec rise time

0 - 2 msec decay time

Total of 18 shocks

c) Surveyor

1) Transit and Lunar Approach

Vibration is within the requirements established in the preceding sections of this discussion. Acceleration of 7 to 10 g along the mounting axis would be experienced.

2) Lunar Impact and Operation

Shock of up to 30 g is anticipated.

The specifications above for soft landers are within the requirements for batteries for normal space probes; other than verification of the negative "g" performance characteristic, no special investigations are required.

An investigation of the requirements for a hard lander did not provide any real working data since no published data could be found. However, discussion with spacecraft dynamic design groups led to the conclusion that it would be reasonable to expect high level shocks on the order of 100 to 200 g for hard landers. The environments for launch and flight operations of space probes apply to hard landers as well; therefore, the only additional requirement for the design would be to consider the high impact shocks and methods of damping the shocks before they reach the cells.

This concludes the dynamic environment studies conducted under this contract.

3.1.4 Power System Requirements for Batteries

This paragraph includes a brief analysis and discussion of the requirements for the use of batteries as parts of power systems in space missions. Preceding sections of this report discuss the types of missions expected for batteries. Under closer examination, the question arises, why is the battery needed in power systems. Typically, the answers were found to be as follows:

a) Batteries are usually required to supply transient loads which are abnormally high compared to the average load of the system and can thus be used to save power system weight and complexity.

b) Batteries are required to supply spacecraft equipment loads during periods when there is no power available from the primary power source. For example, a spacecraft with a solar array cannot supply primary power in eclipses; it must rely on the battery.

c) If power supply degradation might occur due to micrometeoroid bombardment or radiation damage, batteries are used to support failure analysis and to enable transfer of certain experiments and loads to other operating modes in order to sustain the mission. Batteries are also used to support the last or critical part of the mission to obtain the maximum experimental data which can be derived from the mission.

d) Batteries which are used to support the end of the mission might be remotely activated systems or other reserve batteries. Such batteries would require activation energy from a primary power system which would itself contain a battery for the reasons mentioned above. Therefore, the conclusion that the only requirement for a low temperature battery for probes is at the end of the mission is considered to be erroneous. In a situation in which a tow-along vehicle might be put into space in a nonoperative condition, it is conceivable that a reserve battery system might be used to supply the end of mission power. However, it is anticipated that the checkout period once the vehicle is activated would require from 5 to 7 days in order to verify that the system is operating or to make necessary adjustments to the system to obtain the desired performance, or both. Therefore, the minimum heating period anticipated for this type of vehicle would be 5 to 7 days plus a warmup period.

In summary, the normally expected application of a battery in space is that of on-line standby plus bus regulation. In this mode the battery is not providing internal self-heat and if the environment drops below 4° C, some type of heating would be required to keep the battery operational; charging the battery below 4° C is extremely hazardous and the capacity rating of the battery begins to drop severely below this point. The data submitted in Section 2 supports this argument.

3.1.5 Activation and Startup in Space

The methods which might be used to activate a battery and start a heater in space were analyzed. The first consideration for battery activation in space is provision of a circuit to accept ground commands to preprogrammed functions into the spacecraft command and control system. Once the commands are received, the activation or startup sequence must be automatic and highly reliable, there being no means of surveillance from the ground during these sequences.

To activate a battery in space requires different technology from that presently used to activate batteries on the launch stand in missiles, because of zero gravity and the fact that the system must be completely sealed over a long life time. Therefore, new technology would have to be developed for activation and provision made to warm the electrolyte prior to activation, if the battery is below operating temperatures. Cartridge heaters have been used to warm batteries on the ground and could be applied in space provided the temperature of the system was above approximately -54° C. To initiate the above sequence, an electrical signal would be required which must be provided either by a primary power system or by a battery.

The typical method of operating batteries in present space systems is to activate the batteries and seal them before launch, designing the batteries to operate over the required prelaunch and mission life. Using this approach, the difficult problems of activation in space are avoided and no startup specification would be required for the battery itself.

Heater activation in space can be accomplished easily, or may be very difficult depending upon the type of heater being considered.

To activate an electrical dissipation heater, a simple thermostatic switching element is required and existing technology is available. To use constant heat sources in space in environments of other than steadily-decreasing temperature or steady-state temperature, a thermal switch control element is required to provide active control of the heat delivered to the battery. Such a device is discussed in later sections of this report and, after a more thorough analysis, it is apparent that the technology to accomplish this function remains to be developed.

The final heating method considered, that of pyrotechnic heating cartridges, presents much more complex problems. The cartridges must be maintained within a useful operating temperature range and an external electrical signal must be provided to heat an "electric match" to the temperature required to initiate the chemical reaction. The chemical activity of the cartridge goes to completion in a matter of seconds; therefore, it is necessary to fire a series of devices sequentially. Present technology has been developed to accomplish sequential firing in ground applications, but the literature analyzed in this study could not be extrapolated to a space system because the thermal problems and environments were entirely different.

An analysis was conducted to determine the means necessary in space to activate and start up power and spacecraft systems. From the analysis made, it is apparent that many of the spacecraft systems, as well as certain transfer functions within the power system, cannot be started unless there is a battery in the system to provide the transient loads, which leads to the basic premise that it is necessary to keep the battery on the line in its useful operating temperature range at all times in order to provide for spacecraft system startup requirements. This basic premise is used as an assumption for designing a battery for a low temperature space application, thus narrowing the choice of methods to that of either constant heat sources such as radioisotopes or to electrical heaters which require additional power in the spacecraft power system.

To conclude this brief analysis, the studies made and the literature reviewed indicate that startup in space becomes increasingly complex when periodic heating is used, and that the technology required to activate batteries in low temperature space environments has not been developed. The analysis showed that startup in a space environment below the freezing point of batteries is not reliable for either the chemical heaters or the battery. The analysis made it apparent that any startup functions in space are to be avoided if at all possible, and that the use of continuous heat sources or electrical heating to maintain batteries continuously within a useful operating temperature range is highly desirable.

3.2 HEAT SOURCE ANALYSIS

A detailed analysis of heat sources, based on the requirements outlined in paragraph 3.1.1 for battery performance and in paragraph 3.1.2 for thermal requirements, is presented in this paragraph. The analysis is limited to the heat source characteristics which are applicable to low temperature battery systems in space.

3.2.1 Bootstrap Heating

As discussed in the literature survey section of this report, bootstrap heating is severely limited in application. Its use is not feasible below -29° F, and it results in a large loss of capacity (approximately 30 to 50 percent) when used even under the most ideal conditions. It is, therefore, ruled out as a method of heating batteries in space and was not considered further in the study.

3.2.2 Electrical Heating - Battery Energy

As discussed briefly in the Phase I section of this report, the use of battery energy for electrical heating, other than bootstrap heating, can be applied in three ways:

a) Overcharge dissipation controlled by overcharging secondary batteries from the primary power system

b) Heating the battery with its internal dissipation during discharge

c) Heating the battery through a thermostatically controlled heater blanket using its own energy, which is more efficient than bootstrapping if adequate thermal control is provided

Looking into the methods further, the following facts could be accumulated in deciding which types of heating methods were feasible and which type to use.

3.2.2.1 Overcharge Self-Heating

Overcharge dissipation control is feasible for one type of battery only, the nickel-cadmium. It is limited to thermal dissipation corresponding to the C/40 rate maximum above 4° C (C/10 rate above 16° C). Figure 3-3 shows a curve of heat dissipation versus overcharge rates which can then be specified for battery size. For example, a 12 ampere-hour, 22-cell nickel-cadmium battery, overcharged at the C/40 rate at 4° C, will dissipate approximately 10 watts, which is available internally to maintain battery temperature. It should be noted, however, that the operation of the battery in the charge mode before entering overcharge is endothermic and the battery is quite capable of absorbing heat from its environment and thus can cool rapidly to lower temperatures where charging difficulty will be encountered. This happens when it can least be afforded because in typical power systems, excess energy is not available to heat the battery during charge; the battery cannot heat itself and still absorb and convert electrical energy to stored energy.

To conclude, heating a battery with its overcharge characteristic is feasible for limited applications. It is not feasible for any system other than the nickel-cadmium battery because no other system satisfactory for space can be overcharged at significant charge rates. A nickel-cadmium battery may be maintained at 4° C in environments as low as -29 to -40° C if the battery size is larger than approximately 400 whr (comparison of C/40 dissipation rates for nickel-cadmium batteries with Figure 3-5 heat requirements). This method is not feasible for lower temperatures.

3.2.2.2 Discharge Self-Heating

Heating a battery with its internal dissipation during discharge is feasible in certain modes of operation. As already discussed under bootstrap heating, battery warmup using battery discharge self-heat is not feasible. However, maintaining battery temperature during discharge is feasible provided the battery is already in the useful operation temperature range and the discharge rate is approximately 10 to 20 times higher than the heat rate required to keep the battery within its useful operating temperature range. This statement is based on the theoretical calculations presented in the Phase I study (i. e., internal dissipation rate is approximately 10 percent of battery discharge rate) and on calorimetry test results on the nickel-cadmium and silver-cadmium battery reported in the literature. As a general rule, it is concluded that heating the battery during discharge is feasible but not practical for space applications in a low temperature environment because it cannot be used to warm the battery and it requires very large battery capacities to sustain the discharge rates necessary to maintain battery temperature over extended periods of time. In other words, the losses from the large battery increase, increasing the rate of heat dissipation required within the battery, thus increasing the capacity required of the battery and load required to keep the battery warm.

3.2.2.3 Electrical Heaters

The practice of using the battery energy to heat the battery with a heater blanket is much more efficient than attempting to heat by bootstrapping or internal losses because all of the energy removed from the battery in the form of electrical energy and the heat dissipation within the battery during discharge can be accumulated as heat for the battery, ignoring losses to the environment. It is, therefore, considered feasible for short periods of time to maintain battery temperature using this method, and also to warm up a battery in a low temperature environment provided that the battery temperature is not lower than -29° C nominal and -40° C in the worst case. The method can be accomplished by insulating the battery and using thermostatic controls to regulate battery temperature.

The limitations of this method were analyzed as follows:

Battery Warmup -- Nickel-Cadmium Battery

Battery Size (whr)	Heat Rate Required (Fig 3-4) (watts)	Internal Dissipation (watts)	Net Heat Required (whr)	Capacity Required (% of Rating)
200	15	1	56	28
600	42	4	152	25.3
1,600	108	10	392	24.5

Conclusion: The capacity increment required for warming a nickel-cadmium battery in space is 25 to 30 percent of its rating. Warmup from temperatures of less than -29 to -40° C is not feasible with this method.

Sustained Heating -- Nickel-Cadmium Battery

Battery Size (whr)	Heat Rate Required (Watts)		Net Heat Required (Watts)		% Capacity Required Per Hour	
	-29° C (Fig 3-5)	-184° C (Fig 3-7)	-29° C	-184° C	-29° C	-184° C
200	8	21	7	19	3.5	10
600	14	70	13	63	2.2	11
1,600	28	--	25	--	1.6	--

Conclusion: The use of battery energy to keep a battery at 40° F in an environment of -29° C is feasible for periods of 10 to 30 hours (assuming a 50 percent depth of discharge load requirement) -- 4 to 5 hours in a -184° C environment.

3.2.3 Electrical Heating - Power System Energy

Electrically heating low temperature batteries using power system energy is an ideal solution if it is possible. The main restriction on this method is availability of the power; one of the typical limiting features in spacecraft design is the power-versus-weight-versus-cost tradeoff. For example, Figures 3-9 and 3-10 show sample weight tradeoffs using power system heating versus radioisotope heating, to maintain battery temperature. Figure 3-11 shows a cost tradeoff using solar array power versus rtg power versus radioisotopic heating to provide from 0 to 100 watts of power to heat a battery. It has been our experience, and we believe the experience of NASA and the Air Force, that surplus power to heat a battery in a low power spacecraft is not practical because of

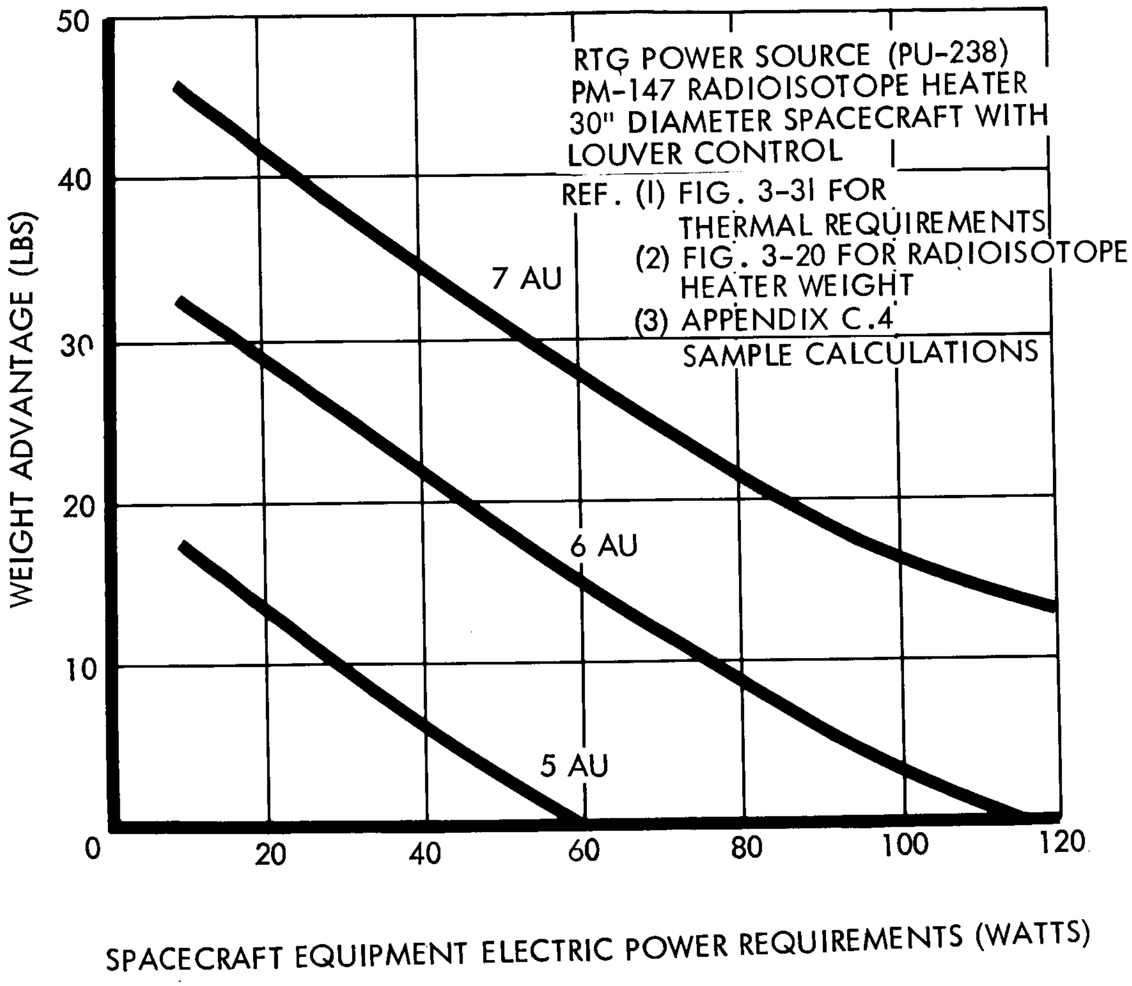


Figure 3-9. Weight Advantage of a Direct Radioisotope Heater Over Electric Heating Versus Required Spacecraft Electric Equipment Power

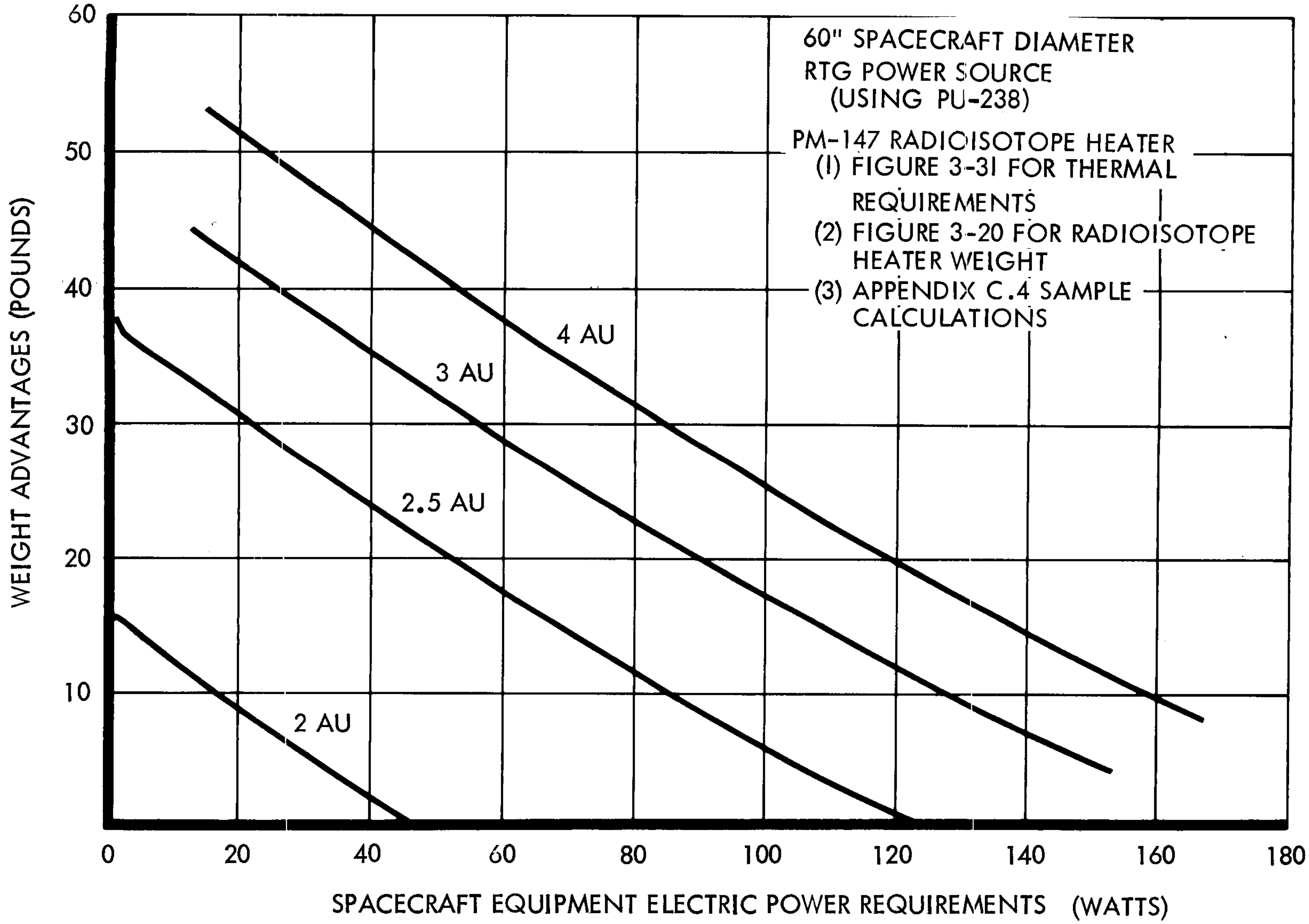


Figure 3-10. Weight Advantage of a Direct Radioisotope Heater Over Electrical Heating Versus Required Spacecraft Electric Equipment Power Load

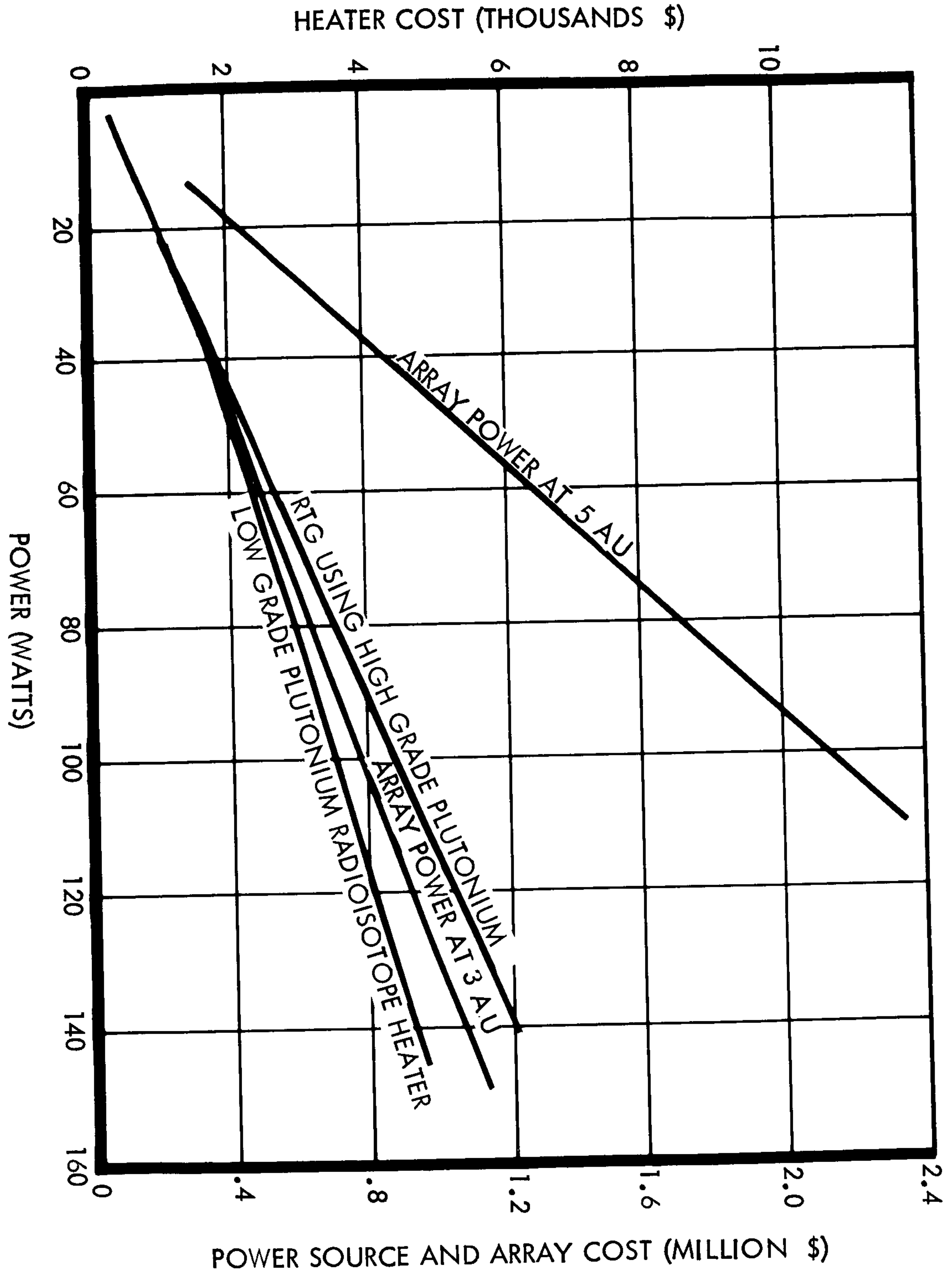


Figure 3-11. Cost Tradeoff Solar Array Power, RTG Power, Radioisotope Heating

weight and cost limitations, and power system degradation redundancy requirements. Going further into the method, heating the battery is easily accomplished by insulating the battery and providing a dissipative insulated heater package in the battery package with thermostatic controls regulating the delivery of electrical energy to the dissipation unit. This method of heating can be improved by adding an active thermal controller between the battery and its mounting interface (environment) to reduce the power required. The benefits from the method are covered more thoroughly in the Thermal Control section of this report.

It is concluded that heating batteries electrically with power system energy is indeed the optimum way provided the electrical energy is available from the power system when it is needed, and the weight and cost penalties can be tolerated. It must be remarked at this time that power system energy is usually not available in Earth orbital satellites during eclipse, nor is it available in a vehicle which does not have its power system activated in storage or in transit with the mother ship. Therefore, caution should be used in assuming that this method is "always available to heat the battery."

3.2.4 Radioisotope Heating

As discussed in Section 2 of this report, the feasibility of radioisotope heating in space vehicles has been demonstrated. It appears that two radioisotope sources are available and applicable to heat batteries in low temperature space environments; this paragraph of the report investigates the two heat sources further, including an analysis of radiation, heat production, physical characteristics, radiation interfaces with space applications, biological interfaces, electronics interfaces, instrumentation interfaces, and source encapsulation. A discussion of a radioisotope battery heater conceptual design is included to evaluate the best method of using the isotopes. A discussion of the feasibility of handling and of safety criteria involved in such use of isotopes is included.

3.2.4.1 Evaluation of Pm-147 and Pu-238 Heat Sources

Tabular isotopic power data present the outstanding characteristics of various isotopes such as principle decay mode, power density of pure materials, and decay rays and particles associated with the atom in question. Before a particular radioisotope can be used, its characteristics as a heat source, such as chemical purity, nuclear purity, source power and capsule power densities, must be studied because such characteristics may present specifications wholly different from those cited for the atomic nuclei in question.

3.2.4.1.1 Radiation Aspects of Pm-147. Microcurie specimens of Pm-147 display the true characteristic decay processes of this radioisotope. A relatively low maximum-energy (0.22 mev) beta particle is ejected from the nucleus and the atom transmutes mainly to the ground state of stable Sm-147. Beta particles are emitted in a continuous spectrum from zero to maximum energy (0.22 mev). In extremely thin samples of Pm-147, the beta range exceeds the sample thickness so that virtually no secondary radiation is observed and no measurable heat is generated within the sample.

Several changes occur in energy and flux when a multicurie quantity of the radioisotope is prepared in a denser form. The formation of heat within the source displays a conversion of energy which is the result of a series of interim nuclear processes. A second change is the inadvertent introduction of contaminating radioisotopes with individual decay modes into the source.

Heat Production in Pm-147 Oxide. Less than half of the Pm-147 beta decay energy is available for heating. Part of the beta energy is converted to heat while the complementary fraction is emitted simultaneously as a neutrino. As the beta passes through matter, it gives up energy by elastic collision creating secondary ionization, until the beta particle is captured by a positive ion.

Isotope heat results from the recombination of ions formed by the passage of the beta through matter. Heat, in the form of phonon emission, also occurs when a beta particle passes close to an atom or molecule, giving up part of its energy to increase the amplitude of vibration of the lattice structure. As the excited state decays by phonon emission, the lattice returns to its natural frequency. Another source of heat arises from the inelastic collision of a beta particle with a planetary electron. Instead of creating an ion-pair, the beta replaces an orbital electron, causing it to absorb energy and go to a higher orbit. The displaced electron subsequently falls to a lower energy state, giving rise to a photon. This, in turn, creates ion-pairs which also recombine to yield heat.

Shielding for Pm-147. The process of Pm-147 beta decay is shown in Figure 3-12. The short maximum range (0.07 mm) of the beta in the source suggests an absence of shielding requirements for this radioisotope. However, a process associated with beta attenuation that is thermally nonproductive complicates the picture. When an electron is accelerated in the electric field of the nucleus of an atom, it gives rise to a ray. Possessing a continuous spectrum from zero to the maximum energy of the beta, the ray (bremsstrahlung) has considerably greater range than the beta and no finite attenuation thickness in the absorber. Therefore, although no shielding is required for the 0.22 mev beta particle, shielding will be required to reduce the bremsstrahlung radiation.

The energy and flux of bremsstrahlung associated with a 1-watt heat source of Pm-147 oxide were measured and plotted and are shown in

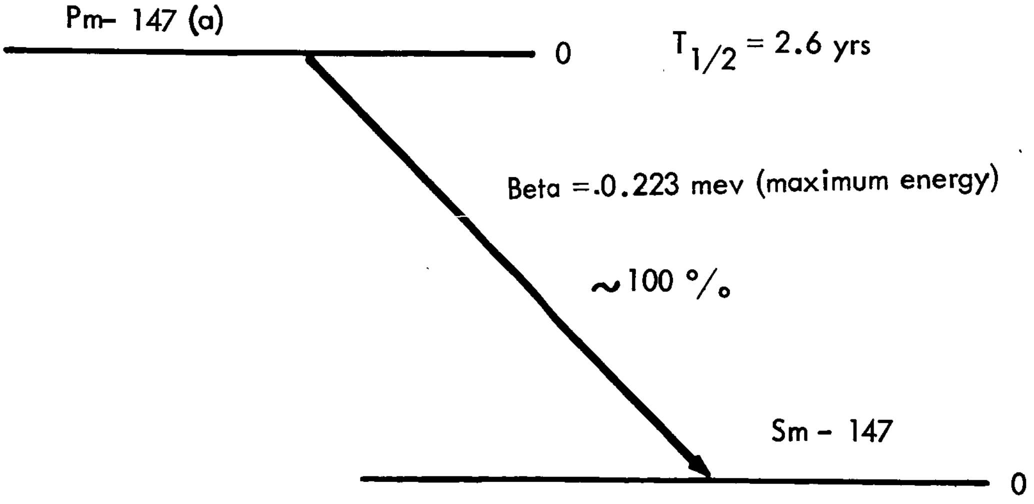


Figure 3-12. Pm-147 Decay Scheme

Figure 3-13. The spectral distribution ranges from 40 to 220 kev, a maximum value for the Pm-147 beta particle. An energy peak at 120 kev, is in agreement with similar bremsstrahlung energy values reported in the literature (Reference 2), but erroneously reported (Reference 3) to be a gamma ray from an excited state of the daughter, Samarium-147. (The absence of a correspondingly lower beta energy of approximately 100 kev fails to substantiate the assumption.)

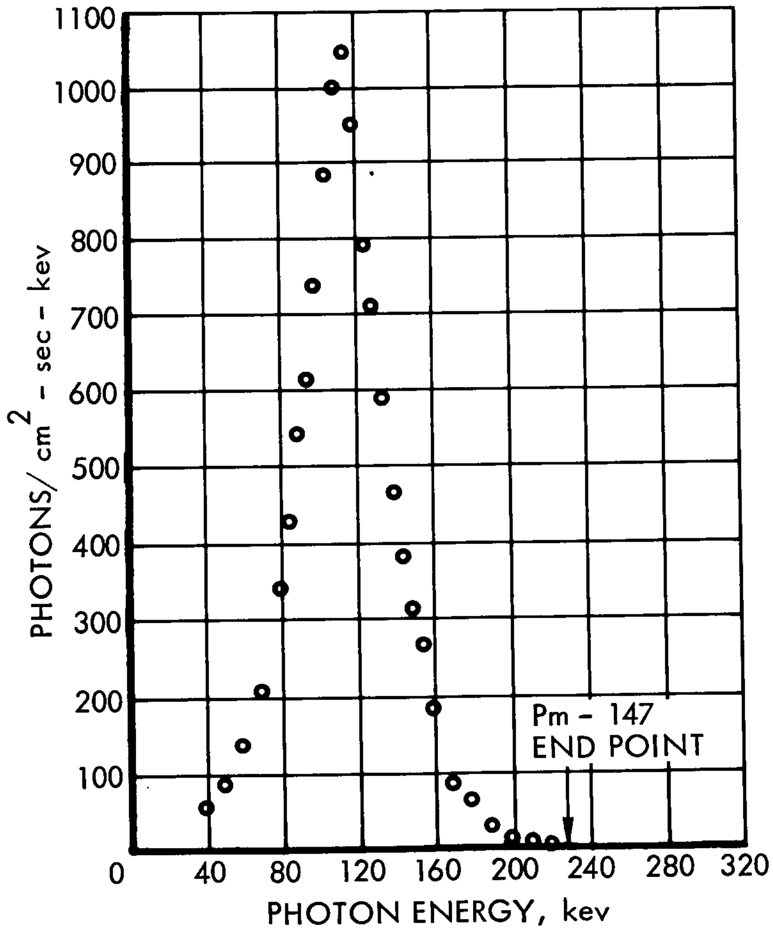


Figure 3-13. Differential Number-Energy Spectrum of a 1-Watt Encapsulated Pm - 147 Source at a Distance of 1 m (Ref. 1)

The numerical integration of bremsstrahlung spectra shows (Reference 1) that the flux of photons with energies between 0 and 220 kev is 6.2×10^4 photons/cm^2-sec at a distance of 1 meter from a 1-watt encapsulated source. A shield of 2 centimeters of lead would reduce this flux by a factor of 7×10^9. Therefore, a 20-watt (thermal) source (with

2 centimeters of shielding) would have a photon flux of 0.02 photons/cm^2-sec at 1 meter. Because the heat plate anticipated for a battery heater would provide partial shielding and because double encapsulation capable of withstanding terminal velocity impact would provide additional shielding, the shadow shielding provided by the battery combines to exceed the 2 centimeters of lead shielding described above. Therefore, if a Pm-147 heat source could be obtained in an isotopically and elementally pure state, shielding would present no problem.

3.2.4.1.2 Radiation From Pm-146 and Pm-148 Contaminants

Promethium is unique among fission products in that there are no stable isotopes of the element. Although the Pm-147 fission yield from U-235 is low (2.16 percent) (Reference 2) as compared to other isotopes such as Sr-90 (5.77 percent), Cs-137 (6.17 percent), or Ce-144 (5.30 percent), the fission production of stable isotopes to these materials dilutes their power density. All promethium is radioactive; unfortunately, several other radioisotopes of promethium are formed simultaneously with Pm-147. See Figure 3-14.

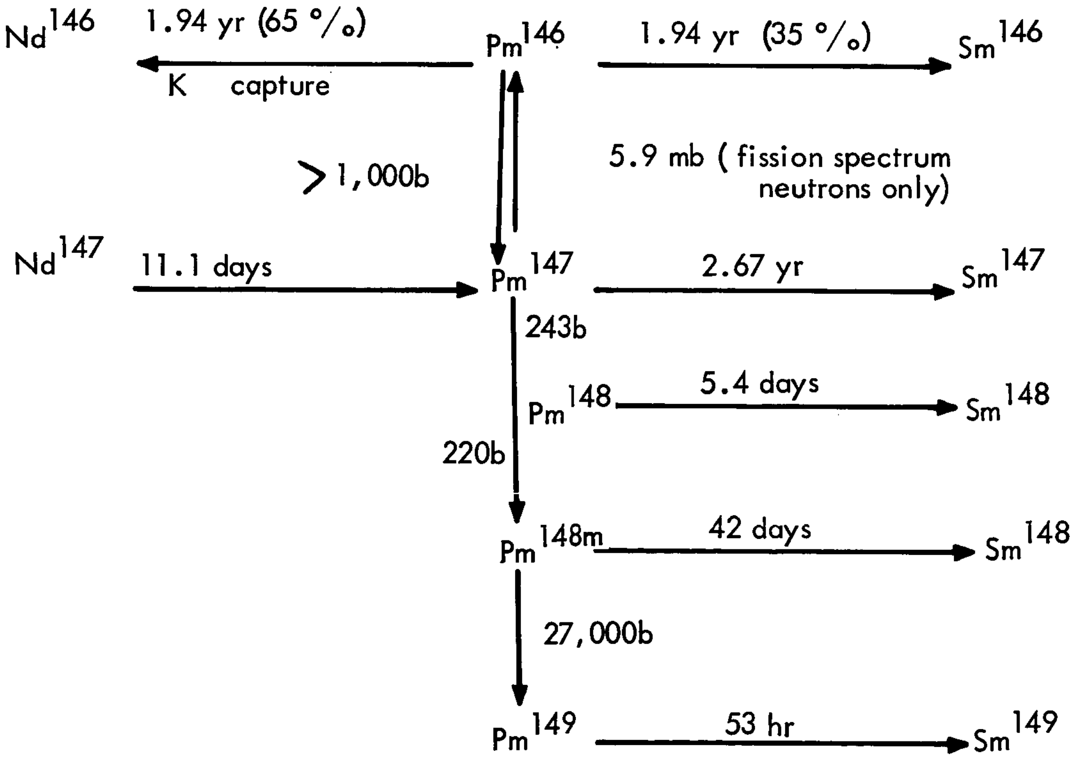


Figure 3-14. Promethium Generation and Decay Mechanisms (Reference 2)

Pm-147 is obtained from the radioactive decay of Nd-147 as well as by direct fission. Once formed, Pm-147 becomes subject to neutron absorption in the reactor and transposes to other isotopes of promethium.

One such reaction, the absorption of a neutron followed by the ejection of two neutrons from the nucleus, yields Pm-146 whose half-life approximates that of Pm-147. A second reaction involves the Pm-147 absorption of a thermal neutron followed by the emission of a gamma ray to form either Pm-148 or Pm-148m. The short half-life of Pm-148 facilitates its removal by decay. The buildup of the isotope in the pile is restricted by its subsequent (neutron, gamma) transposition to Pm-149 which has a short half-life and decays to stable Sm-149.

The presence of Pm-146 or Pm-148 in quantity would increase the the shielding required for Pm-147 and decrease its usefulness. All three isotopes are chemically identical and cannot be separated by chemical means. Electromagnetic separation is feasible but impractical.

a) Pm-146 Contamination Effect and Removal

The half-life of Pm-146 (1.9 years) closely resembles Pm-147 (2.67 years) so that the former cannot be removed by the process of "aging". The Pm-146 effect may be seen in Table 3-5 (Reference 4). The radiation from 8.37 x 10^{-2} percent of Pm-146 through 2 centimeters of lead would raise the background to 1.2 rad/hour at 1 meter, as compared to the less-than 1 millirad/hour from a pure Pm-147 source. The increased radiation is incompatible with nuclear detection equipment aboard a spacecraft and presents a biological hazard to personnel.

Means may exist for reducing the quantity of Pm-146 by "burning it out" in the reactor. This might be accomplished by subjecting the source to in-pile thermal neutrons. Referring to Figure 3-14, Pm-146 can absorb a neutron and convert to Pm-147. The cross-section for the reaction, though not known, is believed (Reference 5) to be in excess of 1,000 barns for thermal neutrons, as compared to a 0.0059 barn fission-spectrum neutron cross-section for the (n, 2n) conversion of Pm-147 to Pm-146.

The lesser value for Pm-146 shown in Table 3-5 was measured for the Yankee Reactor, whereas the larger

Table 3-5. Radiation Levels From A 20-Watt Promethium-147 Heat Source Containing Various Radiocontaminants (Reference 5)

Contaminants	Half-Life	Max. Beta Energy (mev)	Gamma Energy (mev)	Un-Shielded (rads)	2-Cm Iron (rads)	2-Cm Lead (rads)	2-Cm Uranium (rads)
Pm-146($5x10^{-5}$%)	1.94y	0.78	0.749 0.745 0.453	3.3	0.25	0.0009	0.0001
Pm-146($8.37x10^{-2}$%)				20	.10	1.4	0.15
Pm-148m(0.01%)	43d	2.5 2.0 1.2 1.1 0.76 0.64 0.57 0.47	0.10 0.19 0.29 0.41 0.55 0.63 0.73 0.90 1.01 1.46	4.6	2.1	0.55	0.10
Eu-152(0.25%)	12.7y	1.46 1.05 0.68 0.36 0.22	0.12 0.24 0.34 EC conv	240	170	63	20
(Plus)							
Eu-154(0.25%)	16y	1.84 1.60 0.83 0.55 0.25 0.15	0.12 0.25 0.59 0.69 0.71 0.73 others				
(Plus)							
Pm-146($8.37x10^{-2}$%)							

value was calculated. Future power reactor Pm-147 sources may contain 4 x 10^{-4} percent of Pm-146 (Reference 5).

An increase in photon energy attributable to Pm-146 is shown in Figure 3-15. The data plotted were limited to the 100 to 800 kev energy region (Reference 1) and show only Pm-146 gamma ray peaks. As expected from the decay scheme (Reference 6) shown in Figure 3-16, the peaks are for the 453 and 745 kev photopeaks from the internal transition decay mode (65 percent) to Nd-146 and the 749 kev gamma (35 percent) from the excited state of Sm-146. Although the abundance of Pm-146 gamma rays is markedly less than from Pm-147 bremsstrahlung, 2 centimeters of lead shielding (which would reduce the bremsstrahlung by a factor of 7 x 10^9) would reduce the higher energy Pm-146 gammas by a factor of only 10 to 20. This leaves a flux of 3 x 10^3 photons/cm^2-sec at 1 meter for a 20-watt source.

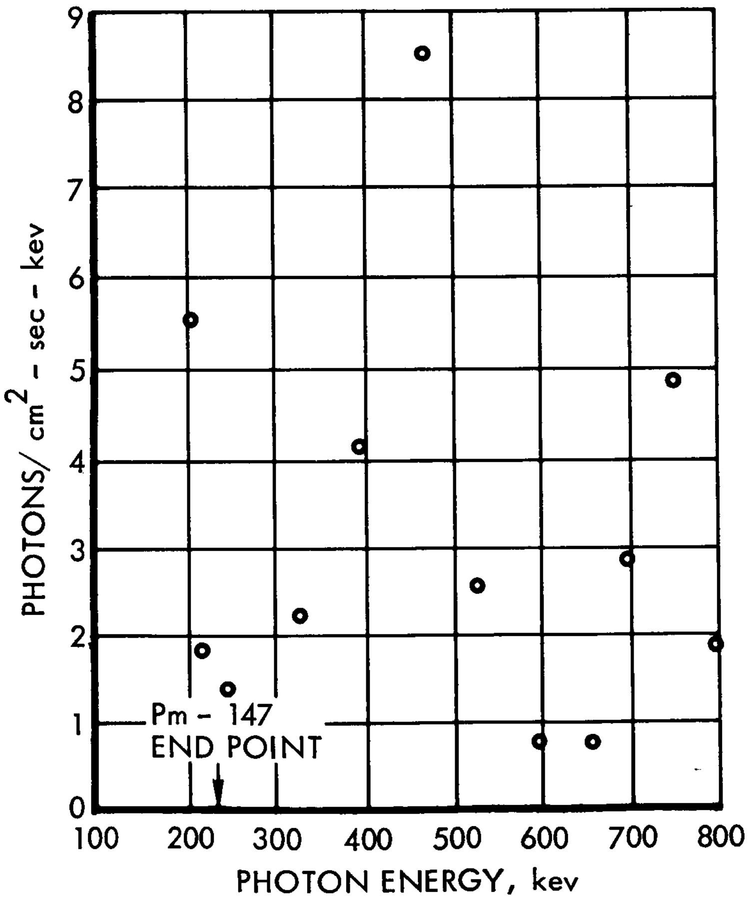


Figure 3-15. Differential Number-Energy Spectrum of the Observed Contaminants in a 1-Watt Pm-147 Source at a Distance of 1 m (Ref. 1)

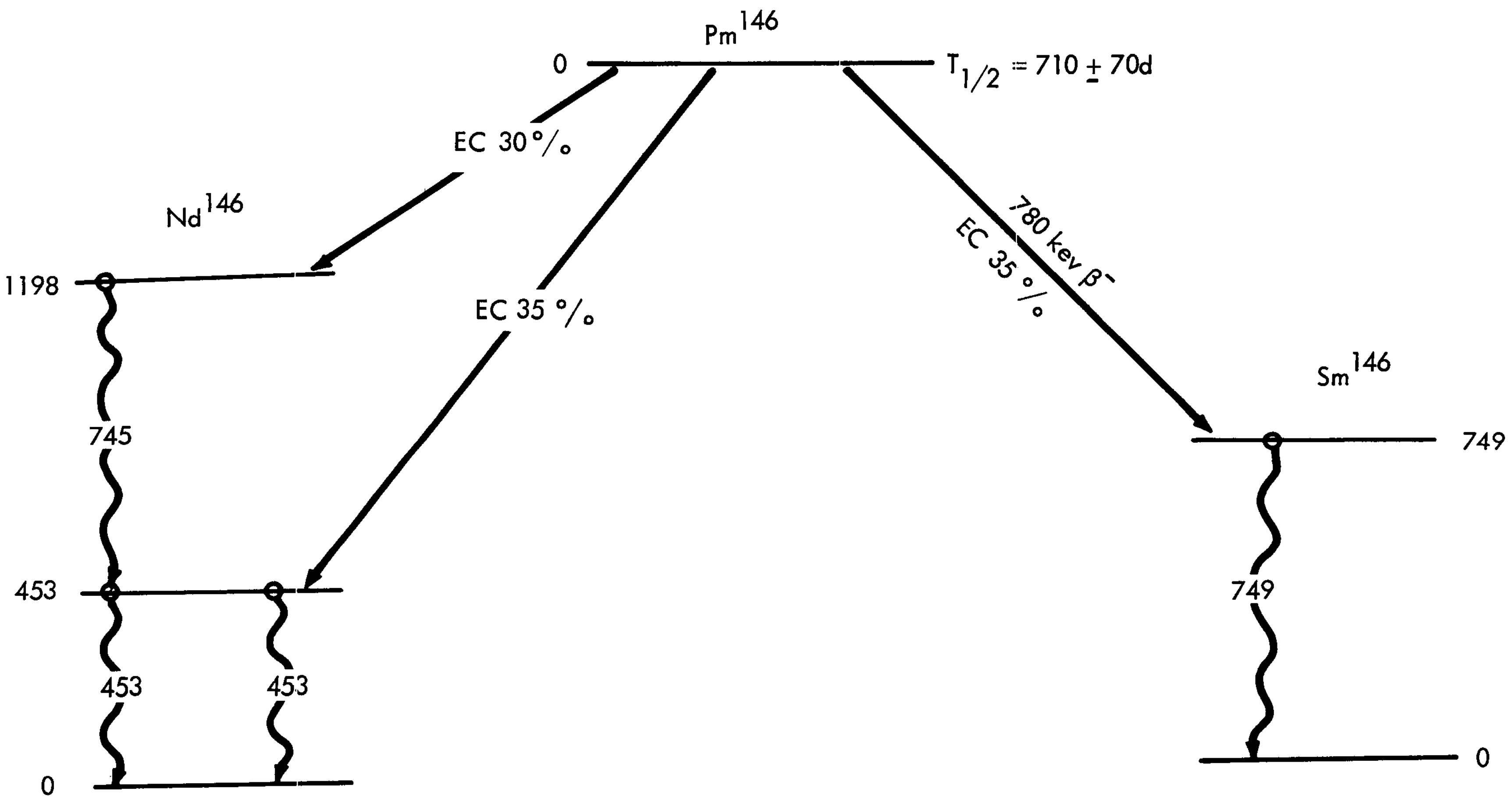


Figure 3-16. Pm-146 Decay Scheme (Reference 6)

b) Pm-148 Contamination Effect and Removal

Referring to Figure 3-14, Pm-147 may convert to either Pm-148 ($T_{1/2}$ = 5.4 days) or Pm-148 m ($T_{1/2}$ = 42 days). The thermal neutron capture cross-section for either reaction is about the same -220 to 243 barns, respectively. Pm-148 is not formed by any other in-pile reaction or by the decay of a radioactive parent.

To assess the radiation due to the presence of this isotope in a Pm-147 heat source, one must examine both the beta and gamma energies (Table 3-5). The former vary from 0.47 mev (40 percent) to 2.5 mev (6 percent) with a total of 15 percent of over 1 mev. From such highly energetic betas, one can expect correspondingly high bremsstrahlung radiation. In addition, gamma rays from the excited states of daughter Sm-148 vary from 0.55 mev (92 percent) to 1.46 mev (2 percent). Since bremsstrahlung of higher energy are anticipated, the primary gamma rays pose a lesser shielding problem.

The presence of small quantities of Pm-148 in a Pm-147 heat source could be catastrophic to shield. It has been estimated (Reference 2) that a Pm-147 source aged less than 2 years would require shielding comparable to Sr-90 or Cs-137 sources of equivalent thermal output.

The effect from 0.01 percent of Pm-148 in a 20-watt Pm-147 heat source is shown in Table 3-5. The dose rate at 100 cm is 550 millirads through 2 centimeters of lead. This radiation level would be incompatible with nuclear detection equipment and biologically hazardous to ground-handling personnel.

A Pm-147 heat source must be essentially free of Pm-148 for battery heating. One means of removal is by decay. The half-life of Pm-148 is 5.4 days, and of Pm-148 m is 42 days; if the Pm-147 source is aged for 3 years, essentially all of the 5.4-day material will have decayed away and the longer-lived 148 m isomer will be reduced to 1.7 x 10^{-6} percent. Once formed (Figure 3-14), Pm-148 m undergoes a reaction (n, gamma) converting to Pm-149 (53 hours). The thermal neutron cross-section for this reaction (27,000 barns) is large when compared to the Pm-148 m cross-section which serves to limit the amount of Pm-148 m present in the reactor at any time; therefore, it is reasonable to assume that the Pm-148 m will not constitute a shielding problem although isolated high-energy bremsstrahlung rays may occur. Fortunately, the flux will be extremely low.

3.2.4.1.3 Radiocontaminants in Pm-147. It was stated earlier that Pm-147 (being a fission product) must be separated from all other isotopes formed. It is impossible to prepare pure promethium, but it is possible to prepare Pm-147 free of measurable quantities of contaminating radioelements. Many elements separate readily while others (radioeuropium, in particular) present greater difficulty; however, multiple separations can remove all detectable europium at the cost of some Pm-147. The necessity for its removal is dramatically shown in Table 3-5. A 20-watt heat source of Pm-147 with low fractional percentages of Eu-152 and Eu-154 would constitute a near lethal dose if unshielded. Even with 2 centimeters of lead shielding, the dose rate would be 20 rad/hour which is unacceptable.

Both europium isotopes possess long half-lives; neither can be removed by source aging. Both emit betas with greater than 1 mev energy and energetic bremsstrahlung radiation can be expected; therefore shielding would be impractical and removal necessary.

3.2.4.1.4 Conclusions and Recommendations for Pm-147. Sufficient work has already been performed at the Hanford and Oak Ridge National Laboratories to establish state-of-the-art production for Pm-147. The material produced to date has been prepared for applications other than space heating. The ability in the future, to produce thermal kilowatts exists; interim production should satisfy immediate requirements for limited numbers of 10 to 30 watt (thermal) sources.

The effect of radiocontaminants on shielding of Pm-147 is known and has been measured. Techniques have been developed to selectively remove the undesirable radioisotopes from Pm-147 by aging, burnout or chemical separation.

The compound promethium oxide would provide an excellent fuel form for space use, provided complete containment is required. Other compounds, capable of burnup and dispersion are being investigated by the AEC (Reference 7).

The level and flux of radiation anticipated from 10 to 30 watt heat sources of Pm-147 (without including added shielding from heat block or batteries) appear acceptable, provided certain specifications of purity are met. These are:

Radioeuropium contamination	nil
Pm-146	less than 4×10^{-4} percent
Pm-148	nil

3.2.4.1.5 Physical Aspects of Pu-238 Fuel. The preponderance of information relating to Pu-238 fuel is classified and has been discussed in a classified addendum to this report. However, the physical characteristics of the fuel may be presented because they influence re-entry and hence aerospace safety.

The nuclear, physical and chemical characteristics of plutonium and promethium define the materials of construction, as well as size, weight, and geometry. Therefore, a comprehensive analysis of each fuel is prerequisite to the preparation of conceptual designs.

Plutonium Dioxide Properties. Intended space use of plutonium fuel requires the knowledge of many physical and chemical properties if one is to satisfactorily maintain the isotope within a capsule, despite all credible accidents or conditions. The following data are excerpts from a recent Mound Laboratory data sheet:

- Physical Properties of Pu-O_2

Melting Point:	No available data for Pu-238-O_2 2280 ± 50° C Pu-239-O_2 2300° C Pu-239-O_2 (Mound Laboratory)	 (Ref 8)
Boiling Point:	4413° C Pu-238-O_2 (Vapor Pressure Data)	
Vapor Pressure:	$\log p\,(\text{atm}) = (-29{,}240 \pm 530)/T + (8.072 \pm 0.239)$	

Density: Theoretical = 11.46 gm/cc

Sintered particles (1 hour)

800° C	5.4 gm/cc
900	5.8
1000	6.4
1100	7.0
1200	7.5
1300	8.2
1400	9.0
1500	9.8
2000	11.4

Note: Best plasma torch spheres obtained to date:

$Pu\text{-}238\text{-}O_2$ = 11.0 gm/cc

$Pu\text{-}239\text{-}O_2$ = 9.7 gm/cc

- Thermal Properties of $Pu\text{-}O_2$

Heat of Vaporization: $Pu\text{-}239\text{-}O_2$;
$\Delta H_{298} = 147.1 \pm 4.3$ kcal/mole (1 atm He) (Ref 8)

$Pu\text{-}239\text{-}O_2$;
$\Delta H_{298} = 153.8$ kcal/mole (1 atm O_2) (Ref 9)

Thermal Expansion Coefficient:

$Pu\text{-}239\text{-}O_2$ (Ref 10)

Temperature (° C)	$\Delta L/L \times 10^6$
25	110
100	695
200	1380
300	2345
400	3355

Heat Capacity: C_p at 325°K = 17.4 cal/°K per mole

- Mechanical Properties

Hardness: $Pu\text{-}239\text{-}O_2$ = 1163 kilograms/mm^2 (Mound Laboratory)
$Pu\text{-}239\text{-}O_2$ = 105 kilograms/mm^2 (Ref 11)

Note: No variation from center to edge of sample observed by Mound.

Crush Strength: $Pu\text{-}238\text{-}O_2$ (74-149μ dia) => 1 kilogram

- Solubility

  1×10^{-9} gm/ml after 53 days in Maine sea water (Mound Laboratory) (Ref 11)

  2.76×10^{-8} gm/ml in sea water

- Particle Properties

  The values relate to the Mound method of fabrication. Microspheres of plutonium dioxide are pressed and sintered to form pellet sources. Therefore, theoretical density cannot be reached (see above).

  Packing Fraction: 37-250μ dia Pu-239-O_2 spheres = 75% packing fraction

  Particle Size: 35-250μ dia spheres have been produced.

- Radiation Characteristics of Pu-238-O_2

  Specific Activity: 17.4 curies/gm Pu-238

  Neutrons: approx. 4×10^4 ${}_0n^1$/sec/gm

Plutonium-238 Decay Scheme. Plutonium-238 has two distinct decay modes. The principal decay is by alpha-particle emission, which provides the energy for heat. The second decay is by spontaneous fission wherein the atom disintegrates into two radioisotope fragments, accompanied by neutrons and the release of energy.

- 238 Alpha Decay of Plutonium

  Half-Life: 86.4 years

  Daughter: Uranium-234 (half-life: 2.5×10^5 yr)

  Energy Distribution (Alpha):

Energy (mev)	% Abundance
5.491	69
5.450	31
5.352	0.13
5.208	0.005

Energy Distribution (Gamma):

Energy (mev)	% Abundance
0.0438	0.038
0.099	8×10^{-3}
0.150	1×10^{-3}
0.203	4×10^{-6}
0.760	5×10^{-5}
0.810 (0.875)	2×10^{-5}

- Spontaneous Fission of Plutonium-238

Half-Life: a. 3.8×10^{10} years
b. 4.9×10^{10} years

Products: Two fission products per fission are formed with an isotopic abundance distribution similar to that reported for U-235. In addition, 2 to 3 fast neutrons are emitted per fission, plus energy in the form of heat and gamma radiation.

3.2.4.2 Radiation Interfacing With Space Applications

It has been shown that, although Pm-147 and Pu-238 represent isotopic fuels with perhaps the least radiation problems, radiation must be seriously considered as an undesirable by-product of all isotope applications. Interface problems from radioisotopes for use in space fall into three categories:

a) Biological-level radiation

b) Electronics-level radiation

c) Instrument-level radiation

If high purity standards are maintained during the separation of either fuel, the problems attendant upon shielding may be minimized. However, it is not possible to produce either radioisotope with the complete absence of contaminating adulterants, and it becomes necessary to determine the level that must be endured or shielded against for hypothetical missions requiring heated batteries.

3.2.4.2.1 Biological-Level Radiation Interfaces. For heat of less than 100-watts, there will be no problem for ground-handling personnel from either Pm-147 or Pu-238 sources, provided proper purity levels are adhered to. Beta-gamma shielding is intrinsic to the construction of nuclear heat blocks. First, the fuel pellets themselves serve as shields for beta and (to a lesser degree) gamma rays; fast neutrons will not be thermalized by the fuel. Next, the double encapsulation required for containment of isotopes serves as final attenuator for the beta particles and as additional absorber for gamma rays. In space applications, where terminal velocity impact containment is required, and where complete containment through re-entry may be required, capsule wall thickness and alloy optimize shielding of the fuel within. Finally, a heat block, required for the uniform distribution of heat to the battery, can be made to provide excellent neutron thermalization and absorption and at least some gamma attenuation. Because it is anticipated that the encapsulated fuel cylinders would be sealed within the heat block at the fabricator's facility, the block would become the basic unit at the launch site. With either Pm-147 or Pu-238 fuel inside, the heat block may be handled by qualified personnel with no injurious effect. Once mounted aboard the spacecraft, because of further shielding by spacecraft components, the block would present an even lower radiation level to personnel working in the vicinity.

3.2.4.2.2 Electronic-Level Radiation Interfaces. Considerable investigation was performed on the ANP (aircraft nuclear propulsion) program to relate radiation damage to electronic components as a function of type and quantity of dosage. To forecast the integrated dosages which electronic equipment aboard a spacecraft would receive is not possible until a clearer definition of mission, equipment, configuration, and related unknowns is provided; however, with the levels of beta-gamma or neutron-gamma radiation that may be expected from 20-watt heaters of either Pm-147 or Pu-238, it is unlikely that a serious problem would develop.

Shielding electronic equipment from radiation may be enhanced by considering the battery itself as a shield. Additional attenuation may be provided by positioning components of known insensitivity to radiation between the heater and more sensitive devices. Similarly, the battery will not be likely to suffer damage from radiation, although this assumption can and should be verified by experimental procedure. Semiconductors are the electronic components most prone to radiation damage, but would be subject to space radiation as well as solar radiation which, in certain cases, could be more detrimental than that which is anticipated from the isotope battery heater. The radiation which reaches equipment by the indirect beaming from scatter radiation is dependent upon the configuration and materials of construction of the spacecraft and cannot be estimated. Clearly, the effects of radiation on electronic equipment should be determined by test procedure as a logical outgrowth of this program.

3.2.4.2.3 Instrument-Level Radiation From a 1-Watt Pm-147 Source.

If the spacecraft which will employ heated batteries does not carry sensitive nuclear detection devices aboard, the consideration of instrument-level radiation will not apply. However, it is probable that in the following missions detection and measurement of radiation and particles will be among the prime requirements: (a) enroute to a distant planet and (b) in the immediate vicinity of the objective.

Advantage may be taken of an investigation performed at the Jet Propulsion Laboratory, Pasadena, California. An encapsulated Pm-147 source was studied to determine compatibility between the source and various sensitive nuclear detection instruments for Mariner II-type missions. The fuel form employed was promethium oxide (unsuitable for re-entry burnup), the inner double encapsulation (Reference 12) was applicable only (see Figure 3-16) for terrestrial application, and the outer flanged capsule arbitrarily (References 7, 8, and 13) assigned a wall thickness. The source was "well aged" and the photon energy levels were those to be expected from the Pm-146 contamination. The investigation being preliminary, no spacecraft component shielding was considered. The conclusions drawn were as follows:

a) Mariner II-type detectors could be operated at a distance of one meter from a 1-watt source with 0.64 cm of lead shielding the source. Such detectors would operate with 0.32 cm of lead shielding, but the source effect would have to be subtracted from the data obtained.

b) Additional shielding would be required for scintillators. Two centimeters of lead would reduce bremsstrahlung by 7 x 10^9, so that the flux at 1 meter would be less than 10^{-3} photons/cm^2sec. However, the 450 and 750 kev gammas would be reduced by only 10 to 20, leaving a residual flux of 100 photons/cm^2sec, a value too high to allow measurements of natural gamma rays (Reference 1).

The responses of Mariner II-type radiation detectors to the Pm-147 source are summarized in Table 3-6. The source used for the tests was not designed for space application and was not provided with sufficient thickness or type of capsule material to satisfy aerospace safety requirements; therefore, the values in air are of only academic interest, and those through lead more closely related to actual space requirements although greater thickness may actually be required. However, it is interesting to note that in every instance the count-rate for the Pm-147 source was from 1/4 to 1/20 the average count-rate sensitivity of the detector.

Table 3-6. Response of Mariner II-Type Radiation Detectors to a 1-Watt Pm-147 Source

Detector	Absorber	Count-Rate At 1 m (counts/sec)	Avg. Count-Rate Mariner II-Type (counts/sec)
Anton 213	Air	8.9	
Anton 213	Air Plus 0.125 in Lead	0.15	0.6
RCL-10311	Air	216	
RCL-10311	Air Plus 0.125 in Lead	5.7	20
Ion Chamber	Air	0.0238	
Ion Chamber	Air Plus 0.125 in Lead	0.000125	0.002

While the source is 1/20 that anticipated for use in space by the study (20 watts), no attenuation by a heat block, battery or intervening equipment is considered, although these components would be present in the proposed application.

3.2.4.2.4 Estimated Radiation-Levels from a 20-Watt Pu-238 Source. It is important to have an estimate of the anticipated "conservative" radiation levels for a 20-watt heat source of either Pm-147 or Pu-238; however, any values obtained from extrapolation of other projects or interpreted from classical data should be recognized as such. It is necessary that an experimental investigation be conducted to determine actual numbers that take into consideration additional attenuation from components, scatter from the spacecraft, etc. It is also necessary to obtain radioactive material from recent purification methods so that it may illustrate the degree of contamination-induced radiation for near future requirements.

Table 3-7 shows estimated radiation levels as a function of distance for a 20-watt Pu-238 heat source. The values were extrapolated from SNAP 9A data (Reference 14) and relate only to Pu-238. If contained reentry is required, a high melting compound of plutonium must be used that may effect the neutron contribution by alpha-neutron tunneling.

Table 3-7. Estimated Radiation Levels from a 20-Watt Pu-238 Heater* (neglecting shielding from battery and components)

Position	Gamma Radiation (mr/hr)	Neutron Radiation (mrem/hr)
Surface	3.45	10.0
50 cm from center	0.08	0.4
100 cm from center	0.02	0.1

*Extrapolated from SNAP-9A data supplied by the Martin Company, Baltimore, Maryland.

Additional radiation data may be obtained from Appendix D of this report.

3.2.4.3 Source Encapsulation

Safety requires that large sources be encapsulated to prevent the spread of particulate contamination, regardless of intended environment. All such sources must be protected against rupture due to all foreseeable incidents which might occur on land or in the ocean. If the application is for space, these criteria must be met in addition to any restrictions that would comply with aerospace safety.

If the source is designed to burnup on re-entry, the device enclosing it must be made to disperse and permit the capsule to be exposed to heat above 200,000 feet. Above this altitude, homogeneous dispersion of micron-sized fuel particles will not pose a hazard to life. However, if the capsule cannot be exposed to burning above 200,000 feet, it must be constructed so that it will remain intact upon impact with the Earth. If complete containment is required, an ablative shield must be provided external to the capsule (may be external to the device employing it) such that protection against burnup is provided. Here again, the capsule must be able to impact the Earth intact.

In order to provide protection to the fuel within, the capsule dimensions, wall thicknesses, and materials must be carefully determined. Because high temperature materials are not always inert to the fuel material (at elevated temperatures), a double encapsulation array would be used. Chemically inert inner material must be selected, while the outer shell must possess high temperature structural strength. Because capsule design data cannot be estimated but must be determined by an exacting procedure, shielding cannot be estimated based upon capsule wall thickness.

Encapsulation of source material presents an excellent opportunity to reduce the radiation level without adding shield weight to the system. Encapsulation confines beta and alpha surface emission to the capsule and thereby decreases the volume in which the conversion of energy to heat occurs.

Fuel capsule configurations have been received from the Mound and Oak Ridge National Laboratories (Reference 7 and 15). The designs reflect present capability of pellet fabrication and encapsulation. Terminal velocity impact protection has not necessarily been included because mission requirements were not considered; however, the capsule designs do include materials and wall thicknesses capable of safely containing the sources in a terrestrial environment.

The plutonium heat source designs are included in Appendix D.

3.2.4.3.1 Pm-147 Fuel Encapsulation. The design in Figure 3-17 shows a doubly encapsulated promethium oxide source, predicated upon an outside diameter of 5/8 inch and overall length of 1 inch, a size which permits the fueling of 4 thermal watts at the indicated compound power density. A capsule of this size would be desirable for applications requiring up to 50 thermal watts in a heat block configuration. The outer envelope may be provided with additional wall thickness to withstand impact; or a group of capsules may be inserted into a tube of proper thickness and this, in turn, introduced into the heat block.

The assembly drawing indicates a concentration of spacers at one end of the capsule to protect the fuel while heliarc-welding the lid to the cup. Similarly, thick lids are required to ensure complete containment. Advantage may be taken of capsule-end thicknesses to reduce lateral gamma-ray intensity at both ends of the block. Dimensioning of the inner cavity was predicated upon the use of the oxide. If other compounds of promethium are used, the power level from a source of this size may be less or greater, depending upon the promethium content per cc of pellet.

The indicated use of 316 low carbon stainless steel would permit re-entry burnup of the cladding and exposure of the fuel. It has excellent structural strength and may be welded with a rod of the same material and resists corrosion in the event the capsule were to fall into water. However, its compatibility to new compounds of promethium would require verification before final selection could be made. Various alloys of the Hastelloy series have been used successfully for terrestrial isotope containment.

A second promethium oxide capsule design, shown in Figure 3-18, a 1-watt source originally designed for JPL (References 12 and 16), would be appropriate for low-level thermal requirements. Similar small sources could be distributed uniformly throughout a heat block if less than 10 watts were needed.

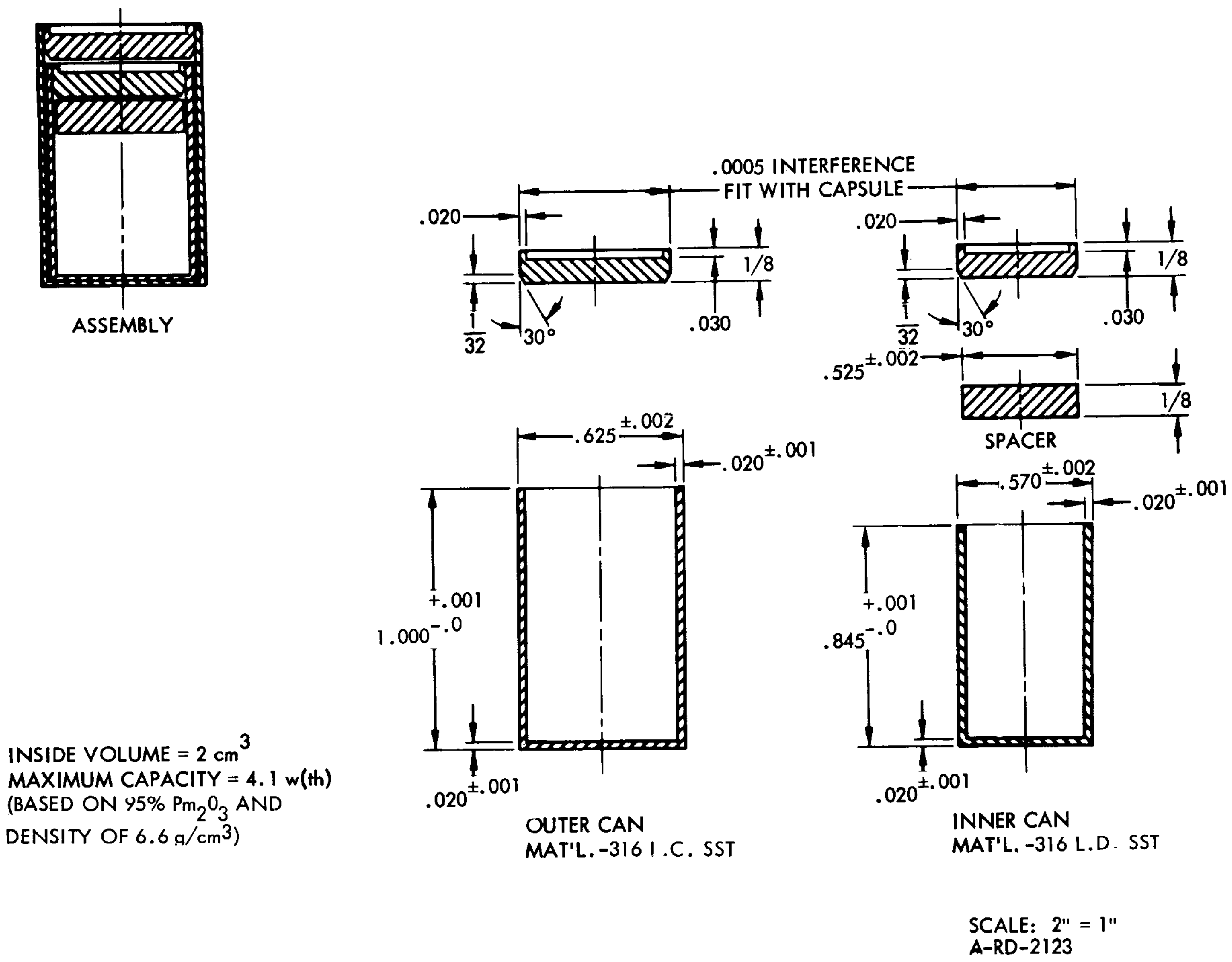


Figure 3-17. 4-Watt Pm_2O_3 Source Capsule—Oak Ridge National Laboratory

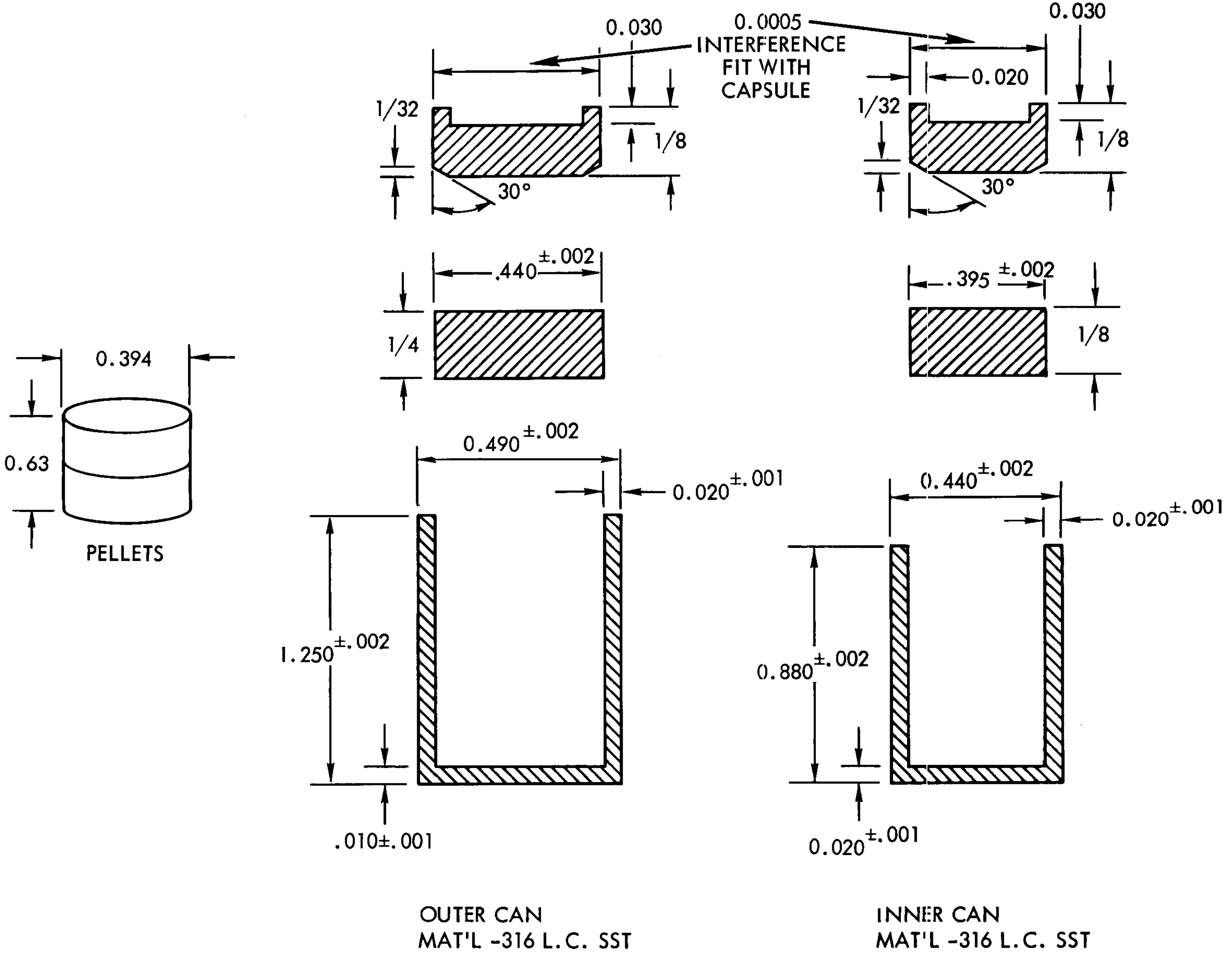


Figure 3-18. 1-Watt Pm_2O_3 Source Capsule—Oak Ridge National Laboratory

The double encapsulation shape is approximately the same as above, as are the wall thicknesses of each container. A second spacer, added before sealing the outer can, may be eliminated and the capsule foreshortened appropriately. As in the design of Figure 3-17, dimensions do not provide against terminal velocity impact. Cavity dimensions and power level are based upon use of promethium oxide, rather than a more readily burnable form of the fuel. As before, wall thickness may be increased either in the outer encapsulant or by introduction of a third tube capable of accommodating a series of 1-watt sources. One or more such tubes may be sealed into the heat block.

3.2.4.3.2 Pu-238 Fuel Encapsulation. Being an alpha-particle emitter, encapsulated Pu-238 tends to accumulate helium gas as a function of time, by the mechanism of velocity loss of the alpha through interaction with matter. As the velocity approaches zero, the alpha, with a positive charge of 2, becomes a helium atom by attracting two planetary electrons. Unlike gas evolution from valence change (upon decay), helium buildup cannot be eliminated or even reduced by compound selection. Containment becomes a function of capsule strength; however, selection of the compound can have direct bearing upon compliance with the present philosophy of intact and contained re-entry. In this case, a compound having a high melting point is requisite because melting and volatilization add to pressure buildup within the capsule. Many compounds which are chemically inert when solid display corrosiveness when molten, attacking the capsule liner and penetrating it. Molten metal fuels can also penetrate metal liners by dissolving them and forming eutectic alloys.

Pu-238 metal is already a space electric power fuel and has been thoroughly investigated. The metal as well as several compound forms are available and others are being evaluated, including the oxide, carbide, and nitride. In each a high melting compound is sought, together with a possible increase in specific power without an increase in neutron evolution through the "tunneling effect."

Because of the relatively small quantity of Pu-238 which would be required to provide from 5 to 50 thermal watts, the oxide might be acceptable and provide material commensurate with present production methods and prices.

A discussion of Pu-238 encapsulation may be found in the classified appendix of this report. However, the presence of helium gas buildup poses some interesting problems in capsule design. It is not entirely known whether the gas evolved remains interstitially entrained in the fuel matrix, or whether it accumulates within the capsule proper. Probably a combination of the two phenomena occurs, due to a time lag in diffusion through the fuel. Whatever the mechanism, the internal pressure of a heat capsule of Pu-238 increases as a function of time; it becomes necessary to provide a void space within the capsule for the gas to accumulate, and increased wall thickness to prevent creep of the metal under pressure and at elevated temperature. While this is a serious problem for high temperature thermoelectric generators, it is lessened for our application by the very moderate operational temperature of the battery heater.

The need for capsule void space decreases the power density of the encapsulated plutonium fuel. As shown in Table 3-8, the specific activity (watts/cc) of the metal form of a low concentration plutonium fuel is 1.71--nearly equal to promethium oxide. However, when both fuel materials are encapsulated and a void space is provided for the plutonium, the resulting capsule specific activity drops to 1.28. The longer half-life of Pm-238 compensates for the drop as shown in Table 3-9. Because Pm-147 has a half-life of 2.6 years, its initial specific activity degrades as a function of mission time, while that of the plutonium remains fairly flat. After 2 years the plutonium metal capsule has a greater specific activity than the promethium capsule. If a 3-year mission were contemplated, even plutonium oxide capsules would have a greater specific activity than the promethium, a compensatory factor which tends to maintain the equivalence of the two fuel forms for certain missions, while indicating the superiority of each for selective mission objectives.

Table 3-8. Physical Characteristics of Pm-147 and Pu-238 Heat Sources

Isotope	Half-Life (yr)	Compound Form	Density (g/cc)	Power Density (w/g)	Specific Activity Fuel (w/cc)	Specific Activity + Void (w/cc)
Pm-147	2.7	Oxide	6.6	0.27	1.78	1.78
Pu-238	86	Oxide	11.4	0.10	1.14	0.86
		Metal	15.5	0.11	1.71	1.28

Table 3-9. Comparative Fuel Power Densities Based on Mission Duration

Isotope	Capsule Power Density		
	T_o (w/cc)	After 1 Year (w/cc)	After 2 Years (w/cc)
Pm-147 Oxide	1.78	1.37	1.07
Pu-238 Oxide	0.89	0.89	0.89
Pu-238 Metal	1.29	1.29	1.29

3.2.4.4 Radioisotope Battery Heater Design

The design concept of an isotope heated battery is predicated upon the following parameters:

a) Quantity of heat required

b) Isotope used as heat source

c) Re-entry philosophy

d) Mission conditions

The source of heat for the battery should consist of a heat block containing sealed fuel capsules within its cross-section thickness. If the fuel must burn and disperse upon re-entry, it may be a segmented block held together by readily combustible straps, with the block made from the same material: magnesium or a thorium-magnesium alloy might be employed. If the fuel must remain intact, the heater plate would be of an ablative material. Neutrons might also be absorbed (if necessary) in the heater block if the block were of a high neutron cross-section capture material. A third consideration is contained re-entry of a Pu-238 source in which the sensitivity of the mission requires absorption of neutrons. Careful selection of high temperature neutron shielding, such as beryllium metal or beryllia, would provide for both requirements. Where uniform heat transfer at high temperature with a low delta T is required, ablator selection is not easy; however, because of the relatively low battery temperature required (4 to 42° C), even a poor conductivity ablator should not increase fuel temperature above 204° C. Capsules for early flights of SNAP-9A (Pu-238) were designed to burn during re-entry; the change to survival through re-entry has initiated rigorous tests to ensure the containment of fuel. Battery heater technology can utilize the results of such tests, thereby saving time and reducing the cost of preflight acceptance testing.

Ideally, a nuclear heater should, with minimum radiation, deliver maximum heat to the battery. In the space environment, heat must be delivered by conduction or radiation. Tradeoffs are usually encountered between the most economical means of heating and the most economical means of positioning a nuclear source aboard a spacecraft. Four-pi envelopment of the source by battery cells can ensure the former but not necessarily the latter economy, a seeming contradiction which stems from the cross-section density of secondary cells, as well as the apparent density of battery arrangements. Three configurations are presented:

a) A central source surrounded by a circle of cylindrical cells.

b) Small heat sources sandwiched between individual prismatic cells.

c) Strings of individual sources sealed into a heat block common to an array of prismatic cells.

Descriptions of the configurations, shown in Figure 3-19, follows:

a) Central Heat Source. Heat transfer from the source to cells may be by conduction through a light-weight heat block. When Pm-147 is used the block must be able to disintegrate in order to expose the source for burnup. With Pu-238, the block may serve as an ablative shield and neutron absorber. With either fuel, a gamma-ray "shine" would exist between cells, providing only a marginal solid-angle for the positioning of nuclear detectors. This configuration is further limited by the number of cells which may be grouped around a single heat source. It would, however, permit end-on-end stacking of similar arrays and does suggest a means of heating cylindrical cells if they are required. Figure 3-19-A is a sketch of this configuration.

b) Sandwiched-Source. Maximum transfer of heat from source to cell results from individual sources between each cell of a stacked prismatic configuration of cells. However, encapsulation of either Pm-147 or Pu-238 to resist terminal velocity impact would require the fuel to be mounted in metal plates of from 5/8 to 1 inch in thickness. The resulting geometry would elongate the battery stack and add to bracket requirements. If Pm-147 were employed, the entire configuration must be able to disintegrate and expose the interior sources to burnup. With Pu-238, provision must be made for ablative protection and neutron absorption which would further enlarge the overall dimensions. This configuration, shown in Figure 3-19-B, would provide maximum gamma-ray shielding along the major axis of the array.

c) Heat Block. The third heater configuration resembles a hotplate, in that the sources are encased within a plate which is coupled to the underside of

FIG. A

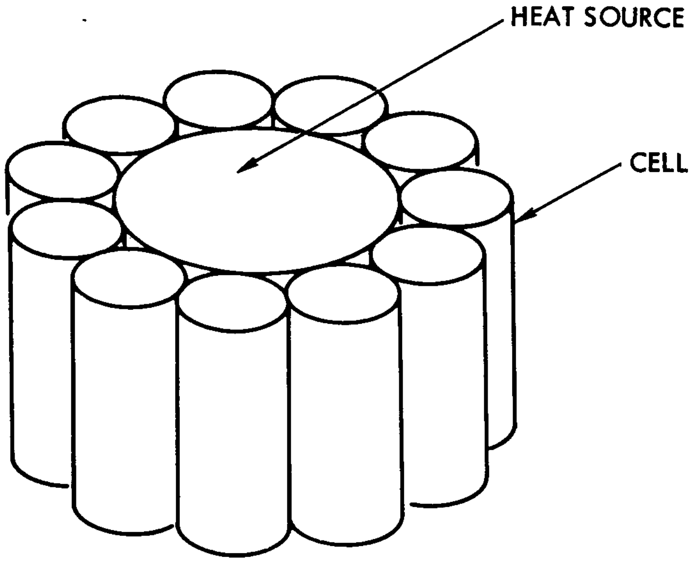


FIG. B

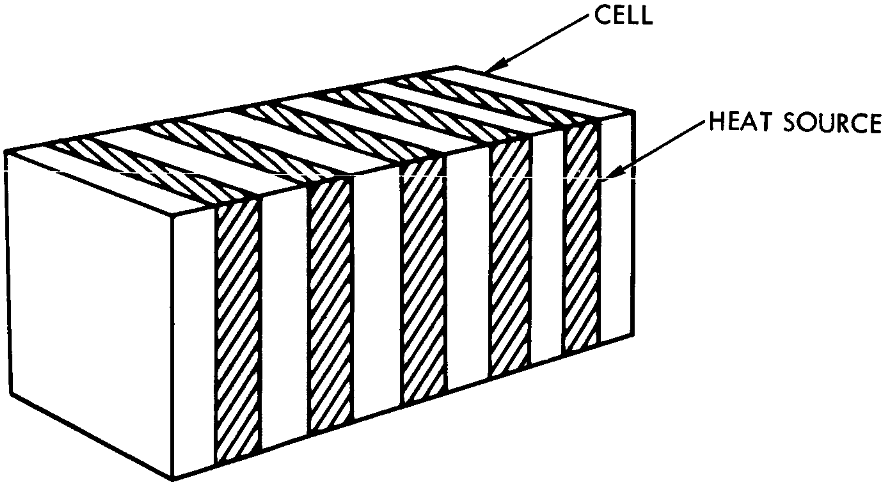


FIG. C

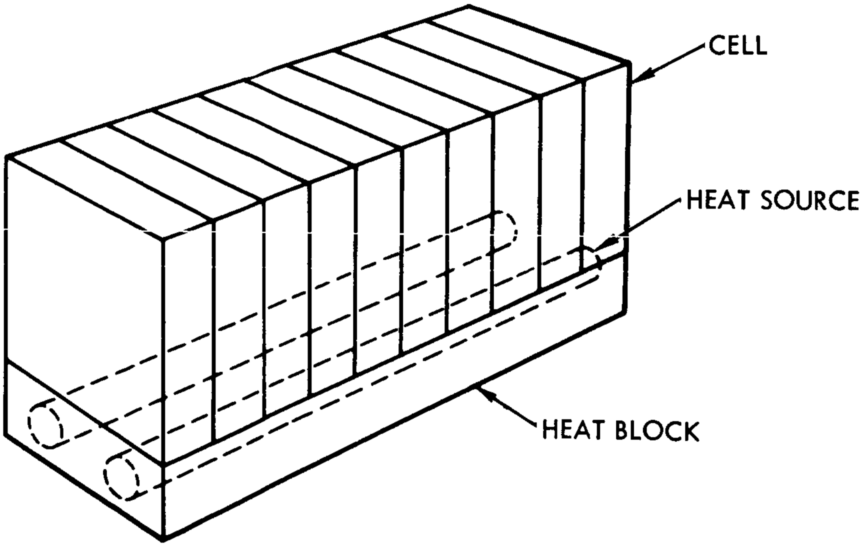


Figure 3-19. Battery Heater Configurations

stacked prismatic cells. Although potentially less economical thermally, better thermal control can be exercised by this design. Overheating can become a serious consideration where thermal control of a battery is required. If insulation were placed on all exposed surfaces of the heat block and active thermal control introduced on one face, control may be exercised. Figure 3-20 shows the estimated weight for the heat block configuration.

Distributing individual encapsulated sources throughout the cross-section of the block would provide additional gamma-ray attenuation when using Pm-147 fuel. If the block were made of magnesium or other readily combustible metal and then segmented, the fuel could be readily exposed to burnup upon reentry. The battery array above the block would also serve as a gamma-ray shield, subtending a large solid angle and providing a major portion of the spacecraft volume for placement of nuclear detection experiments. For Pu-238, the block could be made of an ablative neutron absorber such as beryllium. The liquid within the cells would further reduce neutrons through the solid angle which the battery subtends, permitting certain particle detectors to be carried aboard.

Employment of a separate block permits the last minute mating of nuclear source and battery, thereby easing the problem of source storage during all prelaunch operations.

Finally, the heat block design permits the use of cylindrical-shaped fuel pellets which is consistent with present fabrication methods. Figure 3-19-C illustrates this approach.

3.2.4.5 Aerospace Safety and Ground Handling Criteria

The safe utilization of radioisotopes is usually predicated upon a) containment of the source material and b) attenuation of the radiation from a source to an acceptable level. For space use, re-entry burnup and dispersion above 200,000 feet of small quantities of radioactive material will not constitute a hazard. If either requirement is met, then the heat producing qualities of the source may be applied with the same degree of confidence as other heat producing media. State and federal laws provide rigid specifications which must be met before using radioisotopes for any purpose. In addition, the use of

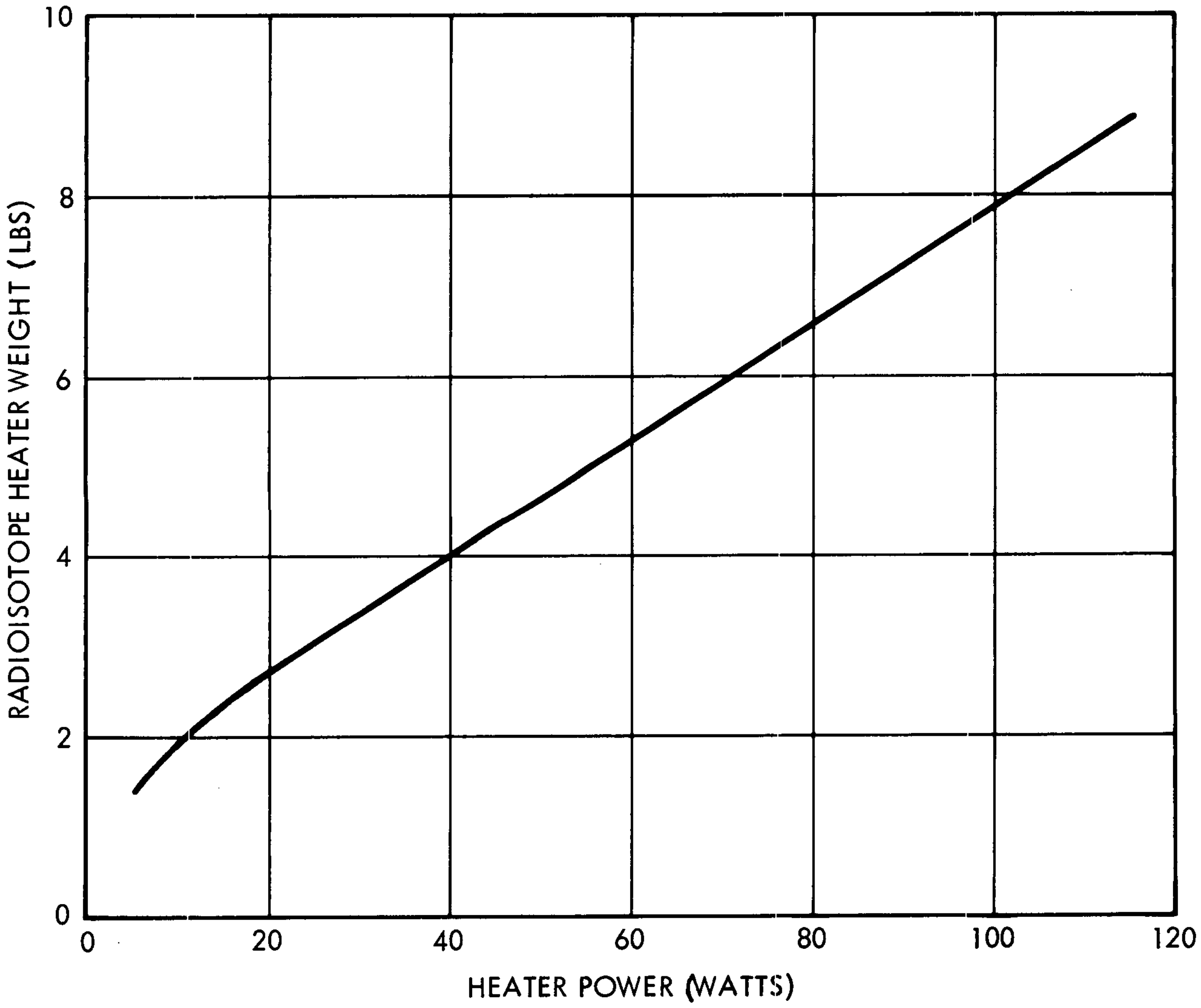


Figure 3-20. Estimated Heater Weight for a Promethium-147 Battery Heater (Burnup Design for Re-Entry)

radioisotopes to heat space batteries depends upon successful compliance with specifications for the following before orbital flight:

a) Source shipment from ORNL/MNL to launch area

b) Residence at launch area

c) Transfer to vehicle compartment on pad

d) Liftoff

e) Preorbital flight

In each instance, the isotope must remain encapsulated despite fire, explosion, impact, or other possible accidents. The encapsulant and shield must also provide sufficient reduction of radiation to prevent danger to life. One-watt encapsulated sources of either Pm-147 or Pu-238 do not pose a serious problem because the anticipated (small) capsule size permits positive testing before actual use by impact machines, fire simulators, etc. Large sources, such as those required for thermoelectric generators are naturally more difficult to test. The last item on the foregoing list requires explanation; a "grey area" exists between launch and orbit where environmental conditions prevent the clear delineation between preorbital and orbital position. Consider first, the return to Earth from an altitude of 75 miles (first-stage premature burnout or engine failure). Air friction would not provide a sufficiently high temperature to melt the battery heater and fuel capsule; therefore, a capsule designed to withstand terminal velocity impact would remain intact. If, on the other hand, a successful orbit were achieved and the spacecraft ultimately decayed by predetermined criteria, the heater and ablator would either provide ablative protection for the capsule, permitting it to survive impact on the Earth, or the heater would immediately (and by design) disintegrate and expose the capsule to upper atmosphere heating to disperse micro-sized particles above 100,000 feet. The "grey area" exists where the flight has proceeded beyond first-stage burnout and "coasting altitude" but does not orbit; here a ballistic return to Earth cannot guarantee burnup because insufficient heat will be experienced by the capsule. Partial burnup can deposit (a) a damaged capsule or

(b) chunks of fuel on a relatively small area. Past policy has been to launch the vehicle over water so that it will either orbit or fall back into water.

3.2.4.5.1 Ground Handling Requirements. From the time of final closure of the outermost encapsulation, the radiosotope may be considered a sealed source. Its transfer to other facilities for incorporation into a battery heat block may be required to conform to ICC shipping regulations governing radioactive materials. Once received by the contractor (either at his laboratory or at the launch site) surveillance and control must be exercised. A fireproof, explosionproof exclusion area must be available for storage. Health-physics control of the source must be guaranteed. Periodic inspection of the source container or cask must be made, noting radiation level and the absence of radioactive material on the surface.

Cask temperature must be controlled to prevent thermal excursion of the source(s) within. Appropriate radiation detection and counting equipment are required to determine the source condition. Procedures for the stepwise loading and unloading of the fueled heater aboard a spacecraft must be drawn up and all personnel indoctrinated as to their individual responsibilities. It is estimated that a team of two men (minimum) could satisfactorily perform all tasks associated with loading a 20 watt (thermal) isotope battery heater aboard a spacecraft.

Promethium-147 Ground Handling Considerations. Before the radiation level of a 1-watt source of Pm-147 can be determined, the concentrations of Pm-146, Pm-148, and europium must be known. Since the second may be present in a quantity which is dependent upon the age of the fuel, an infinite number of potential radiation levels is possible. Once a specific fuel batch has been earmarked, analytical and experimental analyses will be required to determine unit radiation level, which in turn, will define the additional shielding required for shipping containers. From a review of the literature (Reference 4), it seems reasonable to predict that a 20-watt array of sealed Pm-147 sources could be held in the hand for a short time without danger, the sole constraint being that the Pm-147 be well aged and relatively free (less than 0.01 percent) from europium contamination.

The capsules may be loaded into the battery heater block at the source fabricating laboratory or at the contractor's facility. The relative desirability of loading at either location depends upon how the battery is connected to the block and the subsequent testing; in either instance, a permit will be required of the contractor for retention of sealed sources because he will ultimately require possession of them.

Plutonium-238 Ground Handling Considerations. Through prior experience gained on the SNAP-3 and SNAP-9A thermoelectric generators, the procedures for ground handling of plutonium sources (in quantities 30 times that anticipated for battery heaters) has been established. SNAP-3, launched from Cape Kennedy, and all SNAP-9A launchings from Vandenberg provide documented experience at both the AMR or PMR.

The presence of neutrons from plutonium will require special detectors. Wipe testing of the storage container will be similar to that required for Pm-147. Pu-238 sources permit a more definitive forecast of radiation level, and an unshielded array of 1-watt capsules totaling 20 watts of Pu-238 will have an approximate combined radiation level of:

gamma: 2.5 mrad/hr @ 50 cm

neutron: 5 mrad/hr @ 50 cm

These values assume an encapsulation equivalent to 1-cm lead. Once sealed into a heat block, the 20 watt source may safely be held in the hand.

Ideally, the heat block could serve as ablative shield and neutron thermalizer and absorber; however, the clever assembly of more than one material may be required to achieve this goal.

3.2.4.5.2 Aerospace Safety Requirements. It has been demonstrated, at least in theory, that it is more difficult to design for complete burnup than for absolute containment. As shown in the Pm-147 aerospace safety considerations (3.2.4.5), complete dispersion cannot be guaranteed for all phases of flight due to the unique situation that exists in the "grey area"; therefore, evaluation of the design needed to contain plutonium completely would be easier at this point.

Four hypothetical mission objectives have been assigned to the battery heater program, representing (in theory) space and planetary environments of extreme cold:

Mars Landing

Mercury Dark Side Landing

Asteroid Belt Probe

Jupiter Flyby

a) Terrestrial Return. In reviewing the appropriateness of carrying a radioisotope source aboard a spacecraft on any of the above missions, aerospace safety is the prime consideration. This evaluation is greatly simplified since all missions have the same prelaunch, pre-orbit flight and parking Earth-orbit characteristics. Therefore, no matter what ultimate objective the missions might have, common protective requirements would cover either fuel. The heat required for battery heating on any of the missions is within an estimated 5 to 50 watts, sufficiently small to permit the specifications for land transportation, launch site residency, and launch area handling, to be common to all. The chief difference in specifications would arise from the use of Pm-147 or Pu-238 which have different radiological characteristics and different radiation properties, and would, therefore, require selective methods of control. However, ground handling procedures of both Pm-147 and Pu-238 have been established and immediate benefits may be taken.

Capsules for either isotope must be designed to withstand the terminal velocity impact that would be experienced by failure to orbit; in addition, Pu-238 capsules must return from orbit intact. Development will require analytical dimensioning and experimental verification of adequacy. Similar effort was performed for the SNAP-3 and SNAP-9A rtg preflight acceptance tests and is a prerequisite to the use of all radioisotope sources in space.

b) Extra-Terrestrial Flight. When the aspects of selective containment (Pu-238) or burnup (Pm-147) have been satisfactorily demonstrated, followon aerospace safety constraints would apply to the beyond-Earth-orbit flight. Serious consideration must be given to the safeguards against the vehicle being fired from other than a tangential attitude. Protective measures (Reference 17) can be taken to prevent firing the vehicle toward Earth; the technique has been employed on past exploratory missions. Therefore, once the vehicle has been put into an Earth orbit, it will either fire off, never to return, or will misfire and gradually decay as in past satellite flights. Re-entry data exist for a SNAP-9A which was designed to burn and did so at the desired altitude with the desired isotope dispersion. No case is known to the writer of an isotope having been returned by ablative protection and complete containment; however, exhaustive testing in simulated conditions is in progress. The success of Project Mercury capsules indicates that it is feasible.

All hypothetical missions considered for the study involve exploration beyond the Earth. Two of the capsules, the Jupiter flyby and the asteroid belt probe, will remain in space during the mission and eventually enter a solar orbit, obviating further safety considerations. For such missions, tangential Earth orbit escape would constitute the extent of concern for safety.

c) Distant Planet Soft Landing. Those missions which will soft-land on Mars or Mercury may be subject to extraterrestrial constraints. The transfer of Earth-type organisms to other planets is undesirable; much study has gone into the possible methods (Reference 18) of preventing this occurrence. The problem becomes one not only of decontamination but of knowing the relative efficiency of decontamination. To the biologist, a body is either sterile or it is not. Sterilization of an entire spacecraft cannot be verified; however, several decontamination methods appear promising. One method involves the autoclaving of components for 24 hours at approximately 135° C at 1 atmosphere pressure of ethylene oxide. The sealed fuel capsules and heat block would not experience degradation from this treatment provided a heat path were maintained from block to autoclave wall for the removal of isotope heat. The presence of ethylene oxide would not affect the capsules or the heat block. Care would be required to avoid the removal or chemical damage to a silicone-type interface film between the block and battery. Resistance of a sealed battery to sterilizing conditions is not known and, batteries being heat sensitive, would have to be investigated. The nuclear heating components would probably be less sensitive to thermal-chemical sterilization than the battery.

d) Distant Planet Impact. A final aerospace safety consideration is of the possible impact upon a planet by an exploratory spacecraft carrying radioisotopes aboard. Where the atmosphere density is either negligible or unknown, the braking quality of the gas envelope cannot be relied upon. therefore, hard impact will result in (1) the burial of the intact or ruptured fuel capsule of (2) a surface dispersion of the isotope. The former presents a near-ideal situation, as the source would be localized and essentially "contained" by the strata of planetary crust surrounding it; the latter could present problems by distributing radioactivity which would mislead future explorers.

The selection of two radioisotopes (Pm-147 and Pu-238) not found in nature on Earth affords high probability that they would not be found as natural radioisotopes on other planets of our solar system. Therefore, future analyses of a planet's soil for radioactivity could deduct the contribution from promethium or plutonium. Further, the small quantity of radioisotope required for battery heating would, if distributed over an impact area, be extremely difficult to locate. Natural radioactive decay during a period of 10 half-lives would burn up all but one-tenth of one percent of the original quantity. Pm-147 (2.6 year = $T_{1/2}$) would decay to the above level in only 26 years; Pu-238 would require 860 years. If this consideration should be critical to the evaluation of a mission, then the selection of Pm-147 would be appropriate.

3.2.5 Cryogenically-Stored Fuels

The analysis of cryogenically-stored fuels was confined to that discussed in Section 2 after preliminary investigation showed that the type of heat-source using such fuel would be extremely limited in application in space. The first reason for the limitation is the difficulty of keeping cryogenically-stored fuels from leaking overboard. (Approximately 0.1 to 0.2 lbs per hour for 10 to 100-hour vented tank systems — 10 to 30 lb tank weight). The second reason is that the technology required to sustain low level burning rates and flames in space has not been developed and a more thorough investigation, therefore, is not practical.

To conclude, cryogenically-stored fuels do not appear to offer any advantages for use with low temperature batteries nor does the development of new technology appear to offer any further inducement.

3.2.6 Storable Fuels and Propellants

From the data given in paragraph 2.2 of this report, it is quite obvious that storable fuels and propellants are not practical for use as low-rate heaters, being extremely difficult to control. The technology required to use them in space in this manner has not been developed and the materials involved do not lend themselves readily to operation in the environments under consideration. Most of the materials with reasonable boiling points also have high freezing points and materials with low freezing points have very low boiling points. Materials that would operate over the desired temperature range are so corrosive and difficult to handle that the proposed application is considered infeasible. Therefore, this type of heater is eliminated from further consideration for battery heating.

3.2.7 Chemical Cartridge Heating

Chemical cartridge heating was investigated during the search of the literature reported in Section 2 and is further analyzed in this paragraph for warmup and short-term heating. Pyrotechnic heaters are available and have been used in past battery designs to heat the electrolyte before or during activation. However, no discussions exist in the literature on the use of pyrotechnics to directly heat batteries in very low temperature environments. The discussions of battery heating requirements set forth in Figures 3-4 and 3-6 in paragraph 3.1.1 show the average amount of delivered heat which would be required from a pyrotechnic device to heat a battery from -29° C to an operating temperature of +4° C and from -184° C to an operating temperature of +4° C. To provide the required heat in a controlled manner using a pyrotechnic device would be extremely difficult; for example, to heat a 600 whr nickel-cadmium battery from -29 to +4° C in 4 hours may require approximately 12 cartridges, with sequential firing mechanisms weighing approximately 7 to 8 pounds. To heat the same battery from -184° C would require approximately 68 controlled sequenced firings and would weigh at least 40 to 45 pounds, which is obviously unrealistic. A sample calculation, Table 3-10, provides the reader with the means to determine the parameters.

After analyzing the literature, it was determined that it is possible to operate cartridge-type pyrotechnic heaters in environments having temperatures at least as low as -54° C; however, there are several problems which limit their application. One limitation is that a typical heater can only survive exposure to high temperature (85°C) for approximately 12 weeks, or to 204° C for one week, without degradation.

Table 3-10. Sample Calculation, Estimate of Chemical Heater Firings and Weight

ASSUMPTIONS

A. Maximum allowable battery cell hot-spot temperature = 160° F

B. Cartridges will be mounted to a good thermal conductor heat sink to distribute heat from a 15-second heat input.

C. 600 watt-hour battery, nickel-cadmium system.

CALCULATIONS

Battery weight at 10 watt-hours/pound = 60 pounds.

From Figures 3-4 and 3-6

Heat requirement to warm battery
-20 to +40° F = 160 watt-hours
-300 to +40° F = 960 watt-hours

Assume the heat capacity of the battery is sufficient to enable warmup increments of approximately 5° F without hot spots and excessive heat loss. The typical specific heat for a space battery is estimated to be 0.16 to 0.18 Btu/lb-° F. This value is based on TRW experience with similar heterogeneous bodies and the data published by the A. D. Little Company (Reference 19).

To warm from -20 to +40° F in 4 hours, 12 heat increments are required, sized as follows:

$(160 \times 3.41)/(12 \times 0.9)$ = 51 Btu/cartridge where heat delivery efficiency to the cells is assumed at 90% of theoretical heat of combustion

To warm from -300 to +40° F in 4 hours, 68 heat increments with dispersed cartridge locations are required, sized as follows:

$(960 \times 3.41)/(68 \times 0.9)$ = 53 Btu/cartridge

To determine heater weight for the above conditions, the following analysis is presented:

From Table 2-1: A typical heat of reaction for slow-burning (15 second) cartridges equals 373 gram-calories per gram (molybdenum and lead chromate) was selected.

Computing cartridge weight:

(373)(0.00116 watt-hours/g cal)(3.41) = 1.48 Btu/gram

Exothermic material weight for 53 Btu = 36 grams

Assume cartridge packaging weight is 100% of material weight.

Then, total either 51 Btu or 53 Btu cartridge weight equals

(2 x 36 grams)(1 lb/454 grams) ≅ 0.1616 pound

Table 3-10. Sample Calculation, Estimate of Chemical Heater Firings and Weight (cont.)

To heat from -20 to +40° F heater weight = 12 cartridges plus sequencer igniters plus baseplate mount. Assume -- sequencing weight equals cartridge weight and baseplate mount equates to the total cartridge and sequencer weight for simplicity of approximation =

4 x total cartridge weight.

Then, heater weight = 4 x 12 x 0.1616 = 7.6 pounds

To heat from -300 to +40° F heater weight =

68 x 0.1616 x 4 = 43.5 pounds

From Figures 3-5 and 3-7

Heat requirement to maintain battery at 40° F

-20° F environment = 14 watts

-300° F environment = 70 watts

To maintain the battery at 40° F minimum in a -20° F environment

(14 whr/hr) (3.41/0.9) = 53.2 Btu/hr

where heat delivery efficiency to the cells is assumed at 90% of theoretical heat of combustion.

To determine heater weight for the above condition, the following analysis is presented:

(373)(0.00116 whr/g cal)(3.41) = 1.48 Btu/gram

Exothermic material weight for 53.2 BTU/hour = 35 grams/hour.

Assume cartridge packaging weight is 100% of material weight, then

53.2 Btu/hr cartridge weight = 2 x 35 x (1 lb/454 grams)

$\cong$ 0.15 pound/hour

Using the same weight factor assumptions stated previously for sequencing and baseplate mounts, heater weight $\cong$ 0.6 pounds/hour

To maintain the battery for extended time period, estimated weights are as follows:

Time	Weight Estimate
1 hour	0.6 pound + baseplate factor
10 hours	6.0 pounds
100 hours	60.0 pounds

Estimated heater weight to maintain a battery at 40° F in a -300° F environment

(70 whr/hour)(3.41/0.9) = 265 Btu/hour

$$\text{Exothermic weight} = \frac{265 \text{ Btu/hour}}{1.48 \text{ Btu/gram}} = 179 \text{ grams/hour}$$

Heater weight = 4(179/454) $\cong$ 1.59 pounds/hour

To extrapolate these results downward in temperature was impossible from the data available so no conclusion can be reached as to the ultimate life of the device at lower temperatures.

Chemical cartridge heaters are ignited by using an "electric match"; i.e., the application of an electrical firing signal which essentially heats the active materials of the heater to their ignition point. Once ignited, the materials burn to completion. The process, shown in some data to be sensitive to thermal shock, has successfully performed despite thermal shocks from -62 to +100° C at rates which were not specified.

To conclude this discussion, it is the opinion of the writers that the application of pyrotechnic heating devices to warm batteries in low temperature environments is not generally feasible; however, it may be feasible to use them under the following circumstances:

a) Sustaining battery temperature in low temperature environments for up to one week

b) Battery warmup from nominally cold environments on the order of -29° to -54° C to operating temperatures, with limited potential for warmup in colder environments.

These potential methods of using pyrotechnic heaters are further limited by the assumption that an element will be available to control the rate of the application of heat to battery cells, and are premised on the assumption that cells can be stored in environments from -54 to -184° C and possibly colder, without damage. There are no published data available to verify such assumptions; proof would require an accumulation of new technology.

3.2.8 Summary of Heat Source Analysis and Methods for Space Applications

From the above analysis, the following summary is presented to compile the data into a comparison of heating methods for various time durations and to show tradeoffs required in the selection of the above heat sources. The following table outlines potential battery heating methods and conclusions derived on each:

Table 3-11. Potential Battery Heating Methods

	Conclusions
BIOLOGICAL	
Body Heat	Not applicable except in space suits
Exothermic Bacteriological Phenomena	Not applicable
CHEMICAL	
Chemical Cartridge Heaters	Limited applicability for short term heating and battery warmup; limited to one week of gentle heating or rapid warmup from environments as low as -54° C with potential at lower temperatures
Heats of Fusion	
-- Hydration	Not applicable; cannot initiate reaction at low temperature
-- Chemical Reaction	Not applicable; cannot be controlled
Flame Heaters	Very limited applicability; technology has not been developed for low-level flame heaters in space, and development of technology is questionable because of difficulty in metering and limited storability of fuels. The method is also in doubt because of incompatible freezing points and boiling points of fuels with low temperature battery applications.

Table 3-11. Potential Battery Heating Methods (Cont.)

	Conclusions
MECHANICAL	
Friction	Not applicable; too heavy and complex
Heat Pumps and Heat Engines	Not applicable; requires mechanical energy which is typically not available in a spacecraft.
Heat Pipes	Potentially feasible; new technology is being developed but is not advanced to the point where it can be fully evaluated.
ELECTRICAL	
Bootstrap Heating	Very limited potential (not applicable); exhausts battery rapidly and does not heat a battery below -29° C; very inefficient method.
Battery Dissipation (Discharge)	Limited application; battery internal dissipation on discharge can be used to sustain battery temperature if the discharge rates are at least in the C/4 to C/2 range. Method cannot be used for warmup, and is, therefore, limited to 2 to 4 hours of battery operation.
Battery Dissipation (Charge and Overcharge)	Limited potential if power system energy is available. Battery dissipation during charge will not sustain battery temperature until the overcharge region is approached. If the battery is too cold during this period, it cannot be charged. However, once charged, a nickel-cadmium system can use the internal overcharge dissipation for heating if the battery is capable of accepting a high enough charge rate. Method cannot be used for silver-cadmium or silver-zinc systems.

Table 3-11. Potential Battery Heating Methods (Cont.)

	Conclusions
Resistive Heaters	
-- Battery Energy	Limited applicability for short periods. More efficient than bootstrap heating, may be used for warmup if the battery temperature is above -29° C. Disadvantage is the loss of battery capacity; if required continuously over a mission, the weight becomes exorbitant (in excess of 7 tons for 20 watts on a Jupiter mission).
-- Power System Energy	Feasible and recommended if power system surplus energy exists. However, there are significant tradeoffs between providing electrical system energy and heating by other methods from cost and weight standpoints. It must also be considered that certain types of power systems cannot continuously provide energy over the whole mission.
NUCLEAR	
Radioisotope	Excellent potential; the method offers low-weight continuous-heat and can maintain battery temperature in a useful operating range over a wide range of temperature excursions, if used in conjunction with an active thermal controller. Disadvantage in handling and possibly with radiation for some very sensitive radiation detection experiments. An additional problem is safe re-entry and containment.

A comparison of the crossover points of different types of heating methods is presented in Table 3-12 to show the relative performance of each type of heater versus time. From this analysis, it was concluded that the following methods of heating are feasible for use with low temperature batteries in extremely low temperature environments.

a) Electrical Heating Using Power System Energy with Thermostatic Controls. The technology for this method exists provided surplus energy is always available from the power system and neglecting the implications of cost and weight penalties.

b) Radioisotope Heating. This method of heating offers an excellent potential with a minimum development of new technology for providing a continuous heat source which can be controlled to regulate the battery temperature independent of orbits, eclipses, and losses of power system surplus energy. The method is the most efficient studied for this application.

c) Chemical Cartridge Heating. This method of heating offers limited potential in that it is time-limited and is not easily controlled. The method is based on the assumption that a battery can be frozen, warmed and used. The study could not support the assumption with existing technology.

All other heating methods studied have very narrow applications for low temperature space batteries and were, therefore, ruled out as not worth further consideration.

3.2.9 Summary and Discussion of Limits of Technology

The primary limitations in the technology encountered during the study of heat sources were as follows:

a) Warmup heating systems applied to low temperature batteries are dependent on the assumption that a battery may be frozen and then warmed for later use. The technology to support this assumption does not exist in the literature surveyed.

Table 3-12. Comparison of Heating Methods by Weight and Energy Density Applied to Battery Warmup and Battery Temperature Control

Case 1 -- Warmup a 600 Watt-Hour Nickel-cadmium Battery to 40° F in 4 Hours

	-20° F Environment (42 watts required per Figure 3-4)		-300° F Environment (240 watts required per Figure 3-6)		
Method of Heating	Weight (lb)	Effective Battery Energy Density (watt-hr/lb)	Weight (lb)	Effective Battery Energy Density (watt-hr/lb)	Reference Analysis Summary and Comments
Bootstrap	20-30	5-6.7	NA	NA	Approximately 30-50% of capacity required per unpublished TRW experimental data. Not feasible below -20° F.
Battery Self-Heat - Discharge	NA	NA	NA	NA	42 watts x 4 hours = 168 watt-hours required. Normal dissipation equals 10% of delivered energy; would require 1680 watt-hours which is greater than 600 watt-hours battery capacity.
Overcharge	NA	NA	NA	NA	Not feasible to overcharge a $T \leq 40^\circ$ F.
Electrical Heater	15-16	7.5	NA	NA	Reference paragraph 3.2.2.3 for calculations; heat required $\cong$ 25.3% of battery capacity; battery will not deliver energy below -65° F.
Power System Energy - Electrical Heater	13-14	8.2	73	4.5	RTG Power increment weight is approximately 0.3 lb/watt + 1 lb for the heater elements. Weight penalty: 42 (.3) +1 = 13.6 lb 240 (.3) +1 = 73 lb Energy density: $\frac{600 \text{ watt-hour}}{13.6 + 60} = 8.2 \frac{\text{watt-hour}}{\text{lb}}$ @ -20° F $\frac{600 \text{ watt-hour}}{73 + 60} = 4.5 \frac{\text{watt-hour}}{\text{lb}}$ @ -300° F
Radioisotope Heater Pm-147 Reentry Burnup Design	NA	NA	NA	NA	If a radioisotope heater is used, no warmup is required.
Stored Fuel Heater (e.g., (H_2 and O_2 at 1700 whr/lb)	NA	NA	NA	NA	Not compatible with mission environments.
Chemical Cartridge Heater (Fast Rate)	7.6	8.9	43.5	5.8	Reference paragraph 3.2.7 (Table 3-10) for calculations: no technology developed below -65° F.

Table 3-12. Comparison of Heating Methods by Weight and Energy Density Applied to Battery Warmup and Battery Temperature Control (continued)

TABLE 3-12. Comparison of Heating Methods by Weight and Energy Density Applied to Battery Warmup and Battery Temperature Control (cont.)

Case 2 -- Sustaining Heat Requirement of 20 Watts (Theoretical)

	Heat Requirement Duration								
	1 Hour		10 Hours		100 Hours		1,000 Hours		
Method of Heating	Weight (lb)	Effective Energy Density* (whr/lb)	Weight (lb)	Effective Energy Density* (whr/lb)	Weight (lb)	Effective Energy Density* (whr/lb)	Weight (lb)	Effective Energy Density* (whr/lb)	Reference Analysis Summary and Comments
Bootstrap	NA		NA		NA		NA		Short term heating only.
Battery Self-Heat - Discharge	0	10	NA		NA		NA		Reference paragraph 2.2.1.4.4.
Overcharge	See Power System Energy Heating								Reference paragraph 3.2.2.1; source of heat energy is power system.
Electrical Heater	2	9.7	20	7.5	200**	NA		NA	Reference paragraph 3.2.2.3; example analysis: 10 hours x 20 watts = 200 whr 200 whr@ 10 $\frac{\text{whr}}{\text{lb}}$ = 20 lb.
Power System Energy - Electrical Heater	7	8.9	7	8.9	7	8.9	7	8.9	Present technology for rtg power is approximately 1 lb/watt plus heater; predicted capability is 0.3 lbs/watt plus heater; example analysis: (20 watts x 0.3 lb /watt) plus 1 lb heater = 7 lb $\frac{600 \text{ watt-hour}}{60 + 7}$ = 8.9 whr/lb
Stored Fuel Heater (e.g., (H_2 and O_2 at 1700 whr/lb)	14.2	8.1	15.3	8.0	32.4	6.5	175	2.5	Assume fuel weight@ 50% of theoretical energy of reaction = $\frac{(10 \text{ hours x } 20 \text{ watt}}{850}$ ≅ .24 lbs Leak rate ≅ 0.1 lb/hr ≅ 1 lb for 10 hr Catalytic combuster ≅ 2 lb O_2 Tank ≅ 8 lb (Reference 20) H_2 Tank ≅ 4 lb (Reference 20) Total weight for 10 hours ≅ 15.3 lb
Chemical Cartridge Heaters (Fast Rate)	1.7	9.7	8.6	8.7	86	4.1	NA	NA	Reference paragraph 3.2.7 (Table 3-10) for example analysis; baseplate function assumed for 1 hour ≅ 2 times computed cartridge and mount weight; Table 3-10 values scaled from 14-watt to 20-watt requirement.
Radioisotope Heater Pm-147 Reentry Burnup Design	4.8	9.3	4.8	9.3	4.8	9.3	4.8	9.3	Heat source weight (Reference Figure 3-20) plus active thermal controller @ approximately 2 lb.

*Effective Energy Density Referenced to a 600 Watt-Hour Requirement at 10 Watt-Hours/lb

**First order approximation neglecting increased losses

b) The technology to use radioisotope heaters for this application is generally available but several questions were encountered which cannot be answered within the confines of the study. The questions are related primarily to the radiation fields which would be encountered when the heaters are used in conjunction with battery packages in spacecraft systems.

c) The technology to use a pyrotechnic heater in a low temperature space application for any extended period of time has not been developed. To expand present technology into a more usable method would require additional development of the heater cartridges, sequencing methods, and a thermal control regulator which could be used to prevent overheating the cell materials.

d) The technology required to use the other systems eliminated from the study could not be determined.

Of the three prospective methods of heating batteries in low temperature environments which have been isolated by the study, it is apparent that greater development of new technology will be required to use chemical heating methods than that required to use radioisotope heaters. To use the radioisotope heaters, new technology will be required to determine radiation profiles versus battery design configurations and to verify a safe launch and re-entry. The use of electrical heating from existing power system energy should not require further new technology for the heater or heat source; the technology is available and fairly straightforward (resistive heaters and thermostatiscally controlled heater blankets).

3.3 THERMAL CONTROL ANALYSIS

During Phase II a detailed evaluation of various means of effecting battery temperature control was conducted. Several concepts examined during Phase I were eliminated from further consideration. Heliotropic devices were not compatible with vehicles going away from the Sun because the Sun's energy is not available when most needed. High temperature radiation sources offered no advantage over hot plates of moderate temperature, thus eliminating the use of radioisotopes or other high temperature devices within the battery assembly. Open-loop systems were considered but difficulty was encountered in devising appropriate methods to be compatible with the entire vehicle system and the possibility of freezing batteries and then reheating them. Therefore, consideration of open-loop systems was limited to an analysis of heat capacity under the Heat Source discussion above and some hypotheses as to methods of application; the limits of technology were reached very rapidly and it is therefore considered one of the areas to be studied in greater detail for development of new technology in thermal control systems and methods of operating standardized batteries at extremely low temperatures.

Of the remaining concepts, it appeared that insulation would always be necessary, but hardly sufficient. The heat pipe, a thoroughly adequate device for temperature control, appeared most suited for situations requiring cooling at high heat transfer rates rather than the usual battery situation requiring heating at low rates. From these evaluations the conclusion that a thermal switch utilizing the volumetric change of a substance passing from the solid to the liquid state (or vice versa) was the most promising method, particularly when used with a radioisotope-fueled hot plate.

3.3.1 Battery Temperature Control by Thermal Switches

Thermal switches, as considered in this study, are systems whose physical configurations, and thus thermal configurations, are substantially altered by temperature gradients or differences so as to make or break contacting heat transfer areas. The definition has been further expanded, in the present instance, to include in one complete and self-contained system the following major elements:

a) Battery
b) Battery heater
c) Battery temperature controller

Attention has been focused upon the temperature controller in particular.

The basic physical phenomenon utilized for the actuation of these thermal switches is the incremental change in volume of solids, liquids, or gases due to change in temperature. Three specific manifestations of this phenomenon are considered:

a) Distortion of bimetallic elements
b) Change in dimensions of heated structures
c) Change in volume of filled bellows or diaphragms

The present conclusions indicate that the degree of temperature control obtainable by the first two methods is markedly inferior to systems employing bellows or diaphragms.

Two bellows-actuated systems have been devised. The function of these systems is based upon the volumetric differences between the solid and liquid phases of certain materials. The essential features of the devices are described in Section 5, entitled New Technology.

In the overall context of the heated battery study, an application for temperature-controlled batteries on missions directed away from the Sun is highly desired. In particular, if a first-generation space probe is specified, it may be assumed that the spacecraft operates at a relatively low power level, that the prime power source is either the Sun or a small radioisotope thermoelectric generator, and that spacecraft temperature control is dependent to a large degree upon intercepted solar energy. Due to the decay in the intensity of the local solar constant, it would probably be necessary to design the spacecraft to operate with the highest (approximately 37° C) allowable internal environment temperatures near the Earth. At the first destination of interest (Mars, 1.52 AU) temperatures may still be acceptable (approximately 0° C) although much lower. Further into the asteroid belt, somewhere between 3 AU and the vicinity of Jupiter (5.2 AU), local and perhaps extensive regions of extreme cold (less than -20° C, possibly as low as -75° C) may exist within the spacecraft. Thus, as far as the battery package is concerned, its local environment changes from an initially warm condition to an increasingly cold condition. It is noted that the rather limited design of the first-generation probe would have to be extended considerably, to include internal power and heat sources of higher levels, in order to perform satisfactorily at extremely deep penetrations.

The duration of the mission also affects the local battery environment of the spacecraft. The journey to Mars takes 8 or 9 months; going across the asteroid belt to Jupiter would take about 2.7 years. Deep space missions have several influences on the design of the battery package. Some of the design factors are:

a) A separate battery heater must be located at or very near the battery.

b) The heater element is very probably a radioisotope, in order to perform for long periods without depriving other equipment of energy.

c) The heater must operate continuously.

d) The heater tends to promote high battery temperatures early in the mission.

e) Early in the mission, the battery and heater must be thermally separated or a sink must be provided to absorb excess heater output.

f) If the heat sink is the spacecraft structure, maximum heater temperature is on the order of 150° F in order to prevent overheating of other equipment.

The distortion of bimetallic strips or coils is frequently used to effect temperature control. The amount of distortion is roughly proportional to deviation of temperature from some reference value, the structural flimsiness of the sensitive element, and its major dimensions.

The distortion obtained in small size (most practical) strips or elements is well suited to control or actuate thermal systems. Two examples are the operation of electrical switches and positioning of the louvers of the Pioneer spacecraft thermal control system. It is noted that the development of strong forces or large deflections are not required in such applications.

In the present case, a strong force would be required to push two surfaces together under high pressure to obtain reasonably good contact heat transfer coefficients in vacuum or thin atmospheres. Sealing the elements in a pressurized container would improve the contact heat transfer, but would in turn provide a thermal short circuit by heat conduction across the pressurizing gas. Large elements would tend to be insensitive to temperature changes, if this improvement is considered.

It is considered that the disadvantages noted make the use of bimetallic elements undesirable. A possible heater control design employing bimetallic discs is shown in Figure 3-21. The performance of this heater control is not satisfactory.

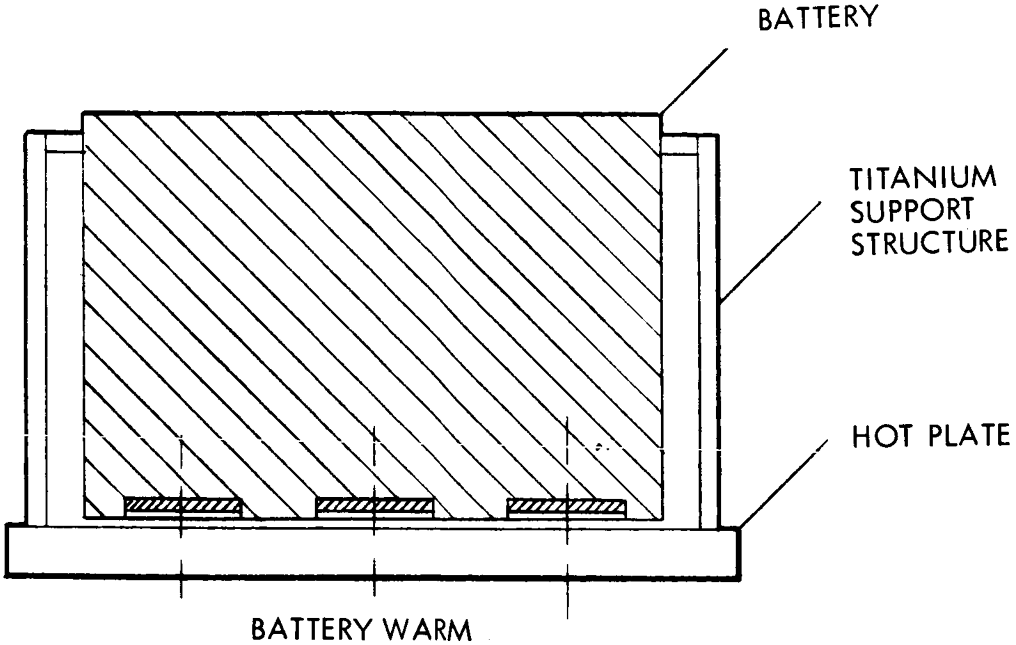
Differential expansion structures are systems utilizing the linear coefficient of thermal expansion (of metals) to press surfaces together at low system temperatures and to hold them apart at high temperatures. In concept, these devices would resemble a clamp whose jaws are the battery and heater. Two or more temperature sensitive rods connecting the "jaws" would cause the heat transfer surfaces to either contact or separate, depending upon the rod temperatures as defined by the battery and heater. In contact position, heat would be transferred at a relatively high rate across the contact interface; when separated, a small leakage would exist by radiation across the gap and by conduction through the control rods themselves.

In practice, however, several shortcomings are evident in the design concept outlined. These are listed below:

a) The ideal control rod should have low thermal conductivity, and a high coefficient of thermal expansion. Data examined during this study indicate that metals having low conductivities also have low expansion coefficients.

b) Since rod expansion in any case is relatively small, dimensional tolerances in design are extremely small if the rods are of reasonable length (6 to 12 inches).

c) Temperature control of the battery is sloppy, since the control elements (rods) are actually sensitive to a



BATTERY WARM

BATTERY COOL

Figure 3-21. Cross Section of Battery Showing Action of Bimetallic Temperature Control Buttons temperature which is the average of the battery and heater temperatures.

d) The contacting surfaces must be smooth, even, and parallel. Cold-welding could be a problem and temperature differences between rods would destroy parallelism.

It is concluded, on the basis of these shortcomings, that extreme difficulty would be encountered in development of a workable design and therefore the concept is considered to be infeasible.

Bellows actuated by the volumetric response of a contained liquid to temperature changes are considered to be undesirable because of the small magnitude of the response. Gas-filled bellows, although more sensitive, require extremely high pressure to develop the forces (in response to temperature) required in the current applications.

An excellent response is available when certain materials pass from the solid state to the liquid as the result of applied heat. Benzene, which melts at 5.5° C, increases volume by 13.3 percent. It can be seen that by placing solid benzene in a cylinder of 1-inch radius and 3 inches long, a theoretical increase in length of 0.4 inch is attainable if the cylinder is allowed to expand only in the axial direction. Also, if the ends of the cylinder are partly constrained, reasonably high pressures and heat transfer coefficients can be developed.

The design features of a feasible battery temperature control device is described in Section 5.0. If the heater has a relatively high heat dissipation rate, it can be mounted between the battery and spacecraft structure to serve as a spacecraft heater as well. If the heat dissipation rate is low, the battery may be mounted in the spacecraft on thermal standoffs, and the heater(s) attached to one (or more) of the battery surfaces.

Although the concept is relatively simple, it is recognized that problems will arise in refinement of mechanical details, such as perfecting sliding or moving members, and in designing a suitable bellows. The design is believed to be within the limits of current materials technology.

Three applications of the thermal expansion of materials have been considered for use in the development of a thermal switch suitable for battery temperature control. Only one of these applications which employs a bellows and a material which changes from solid to liquid at a suitable temperature, is considered to be acceptable. It is recommended that this design is worthy of further study, and perhaps eventual exploitation.

3.3.2 Use of the Heat Pipe for Battery Temperature Control

Heat transport is accomplished by vaporization of the contained fluid at a hot location and its condensation at a cold location. Vapor transport is accomplished by the small pressure differences within the system and liquid return by capillary action within the porous annular column within the pipe. In physical appearance the heat pipe resembles a thick-walled pipe of small bore, with the external wall and ends sealed from its environment. The heat pipe will transfer heat at high rate until the cutoff temperature (varies with the working fluid being used) is reached. When the temperature is reached that solidifies the fluid, the device ceases to function as a heat pipe and any remaining heat transfer is by conduction through the materials alone. By choice of appropriate working fluids any desired cutoff temperature may be obtained; for instance, if water is used, this temperature is 0° C. Although performance of the heat pipe at temperatures slightly above the freezing point has not yet been proven, there is no firm evidence to indicate that operation would be unsatisfactory.

TRW Systems has recently been studying the potentialities of the heat pipe to determine whether further development should be pursued. In principle, the device is a physically small, closed-loop system, which utilizes phase change phenomena to transport heat at relatively high rates across the small temperature differentials it requires for operation. For the present purposes, it has been assumed that successful development is possible. Application of the heat pipe to battery temperature control has been investigated. If successful development is achieved, the heat pipe shows promise of being able to maintain satisfactory battery temperatures over a wide range of ambient temperatures. The ambient may range from hot to moderately cold.

An application of the heat pipe as the temperature control device within a heated battery assembly was analyzed on a simplified quantitative basis. A configuration based on these components was devised and within the limited bounds of the present analysis appeared capable of performing satisfactorily.

Figure 3-22 shows a battery mounted inside a spacecraft structure. A constant-rate heater has been placed on the upper side of the battery, and the lower side of the battery is mounted on a plate attached to the spacecraft structure. This plate is connected to a heat pipe which transmits heat to a radiating plate. During the analysis, the spacecraft structural temperatures were assumed to be higher than is desired for the battery. The objective is to maintain moderate battery temperatures even though structural temperatures are high. The solution is to locate the battery at a point on the mounting plate which is more responsive to the operating temperature of the heat pipe than to the spacecraft structure.

Figure 3-23 shows the temperatures resulting from the configuration described in Figure 3-22. As indicated, the heat pipe operates well over a wide range of high structural temperatures. However, its operation at low structural temperatures is only fair over a relatively small range.

The limited analysis performed demonstrates the feasibility of the heat pipe for battery temperature control under high ambient temperatures. Several other considerations become evident, such as, the consideration of optimum battery support location relative to the mounting plate, the choice of pipe material dimensions, and the choice of radiator size and parameters. It is evident that for even the simplest of circumstances there are many parameters which must be considered before an optimum design can be specified because the heater is useful only over a small portion of the environment range.

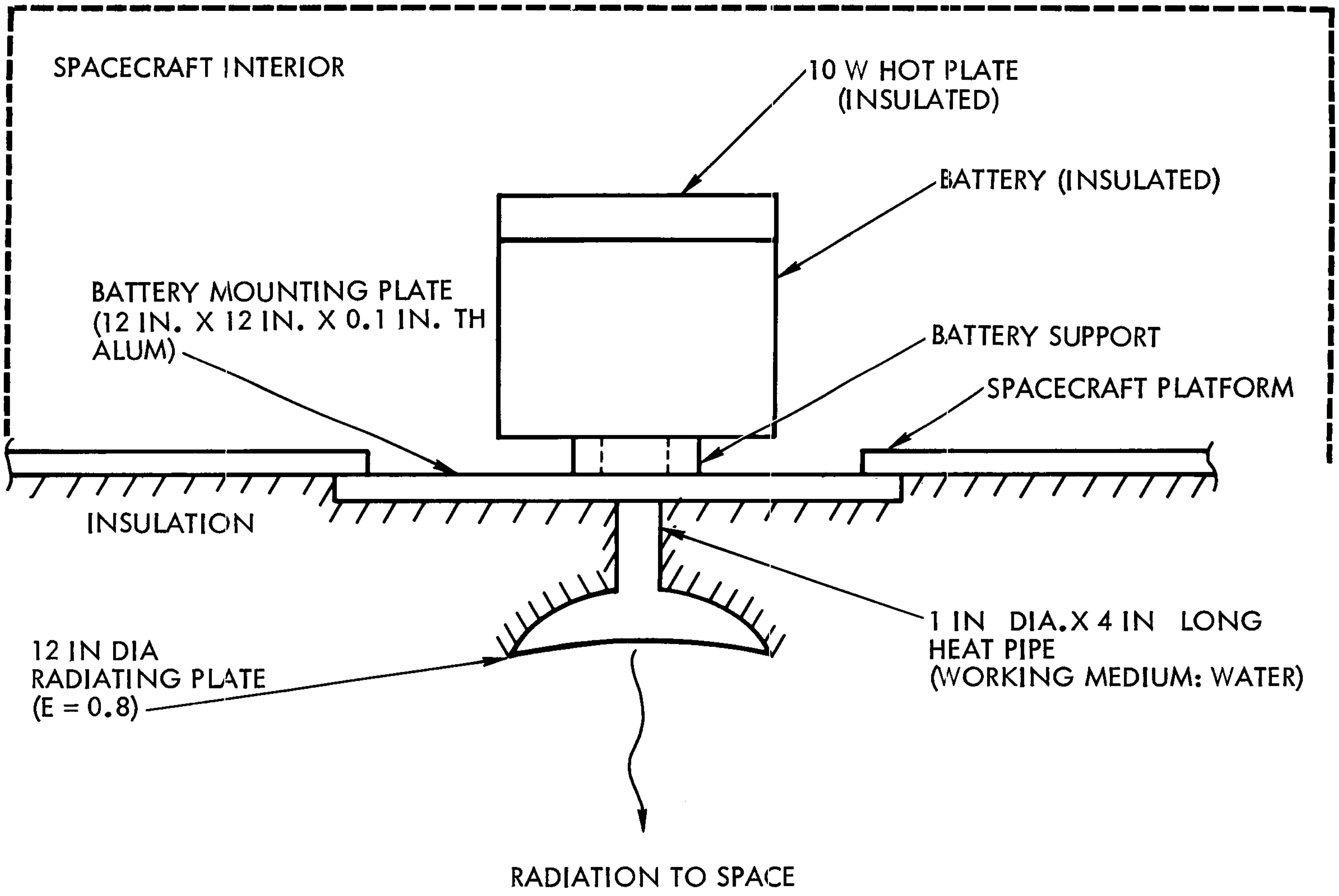
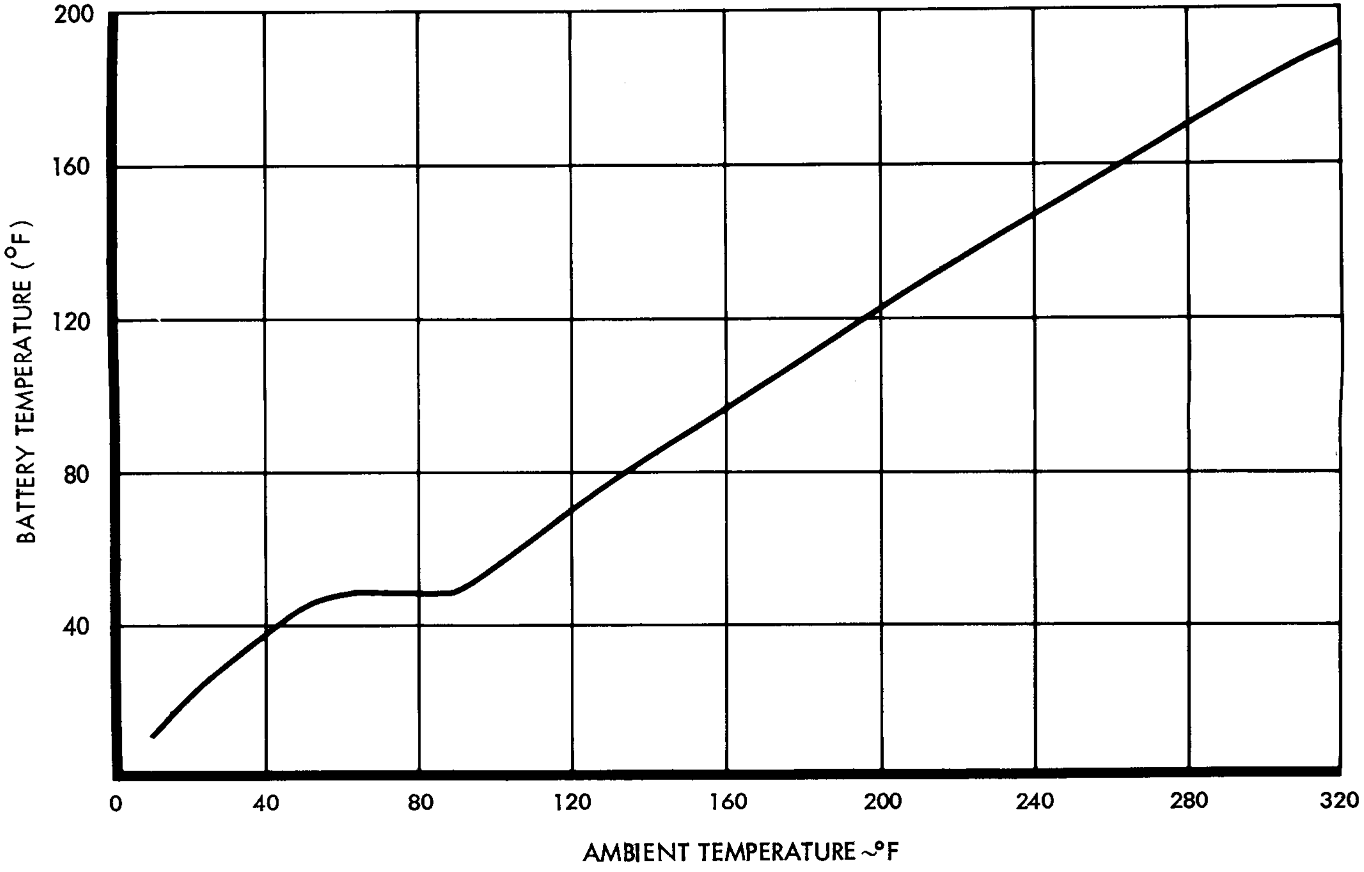


Figure 3-22. Spacecraft Battery with Heat Pipe and Radiation Fin



NOTES:

1) BATTERY SUPPORT IS A 4" DIA. NARROW CIRCULAR BAND LOCATED CENTRALLY ON A MOUNTING PLATE
2) HEAT PIPE KA/L (INOPERATIVE) = 8 BTU/HR - °F
3) TEMPERATURE GRADIENTS BETWEEN HEATER, BATTERY, AND SUPPORT ASSUMED NEGLIGIBLE

Figure 3-23. Battery Temperature as a Function of Ambient Structural Temperatures Using a Heat Pipe for Temperature Control

It is concluded that, in principle, the heat pipe is an excellent means for depressing the temperatures of the battery environment if the ambient temperature is high. However, since performance at low ambient temperatures is limited, it is considered if the units are coupled with a temperature controller, control could be extended to lower ambients and thus wider ranges would be possible than with a simple heater alone. Unless such a controller is justified, it seems doubtful that both a heater and a heat pipe would be required over smaller ranges; either a heat pipe or a heater being adequate by itself for most situations.

3.3.3 Uses of Insulation

An extensive analysis was conducted to illustrate the uses of multilayer reflective insulation. Results indicated that insulation alone provided essentially one-point thermal control unable to tolerate significant deviations from a design point. It was further shown that with the addition of louvers for temperature control the tolerable range of environment was widened considerably. Finally, it was demonstrated that even further improvement was possible if a heater was added, as well as the louvers. The following comments related to figures illustrate these points.

Figure 3-24: Maximum Rate of Insolation as a Function of Distance from the Sun — The availability of solar energy decays rapidly with distance from the Sun. At greater distances, very little solar energy is available for any purpose.

Figure 3-25: Approximate Transit Times versus Distance from the Sun — At greater distances, long mission durations accompany the lack of solar energy. A preferred heat source is one which is independent of distance or time, i.e., a radioisotope or a primary power system heating increment.

Figure 3-26: Variation of Conductivity of Installed Multiple Layer Reflective Insulation as a Function of Number of Reflective Layers — The efficiency of insulation is limited by practical difficulties encountered in installation. If some insulation is good, more insulation is not necessarily better, particularly in view of the joint heat leaks which become significant when the insulation k/ℓ becomes small. Compensation for these undesirable effects can be provided by a heater.

Figure 3-27: Temperature Effects on Multiple Layer Insulation - The temperature level, as well as the temperature

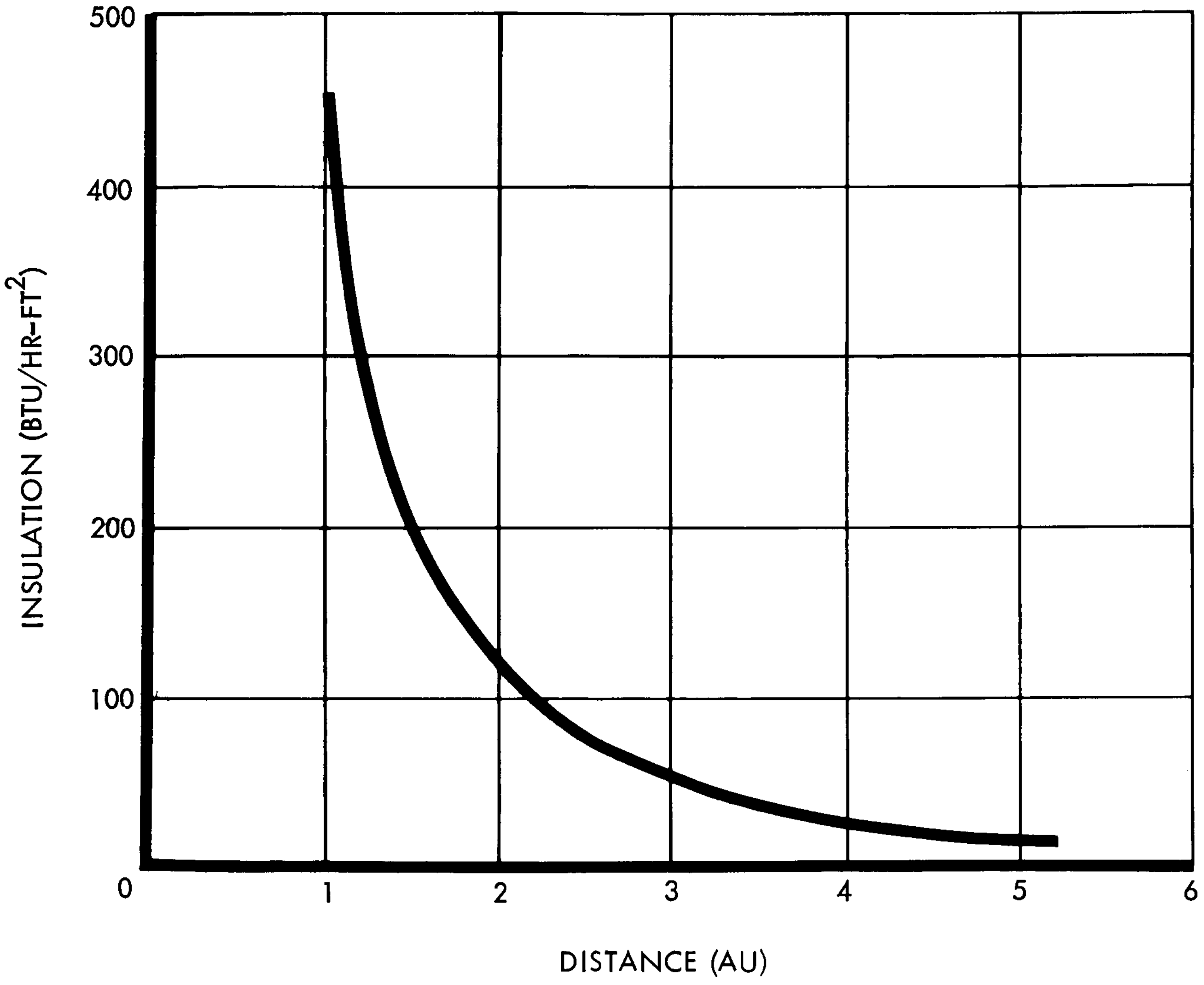


Figure 3-24. Maximum Rate of Insolation as a Function of Distance from the Sun

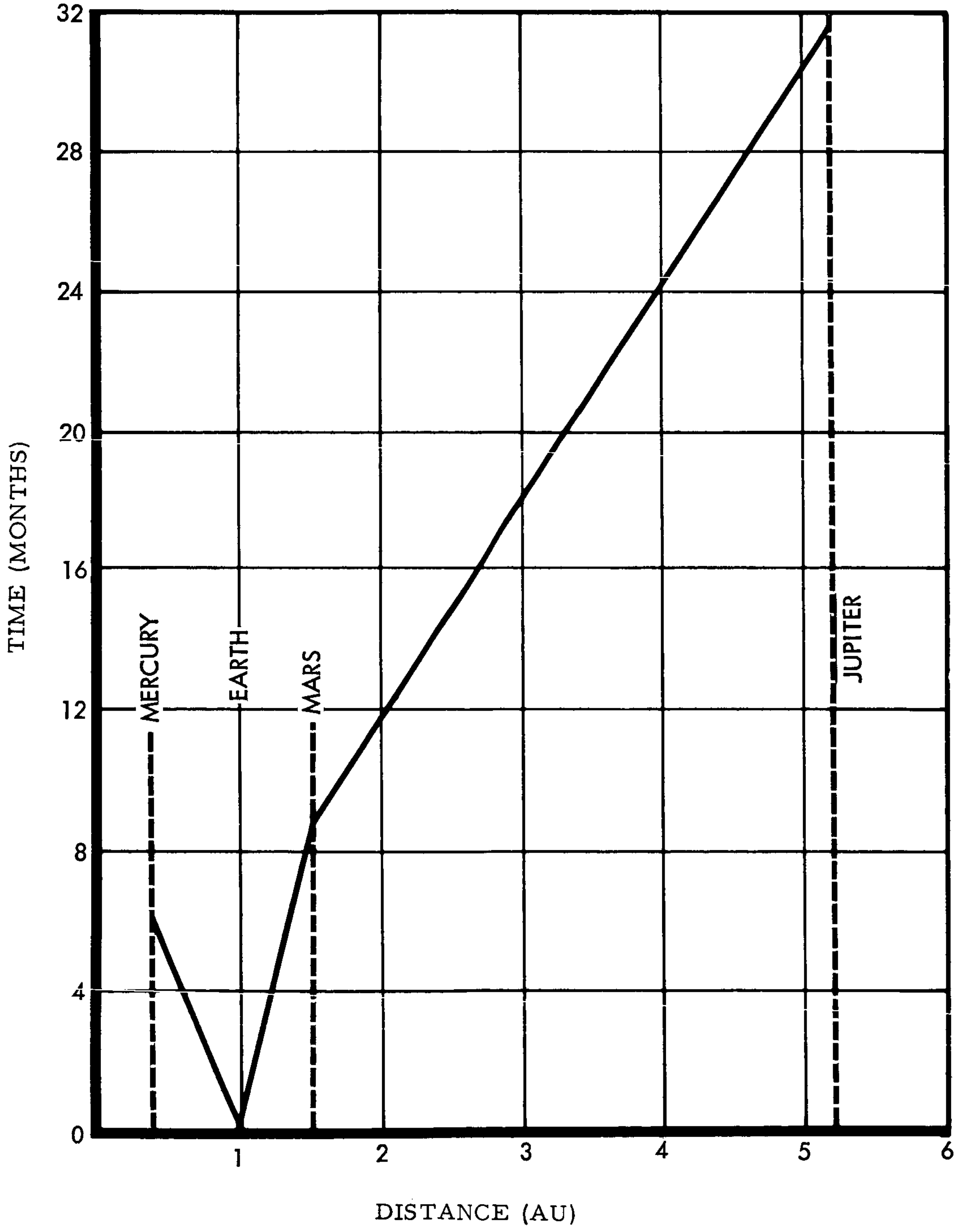


Figure 3-25. Approximate Transit Times Versus Transit Distance

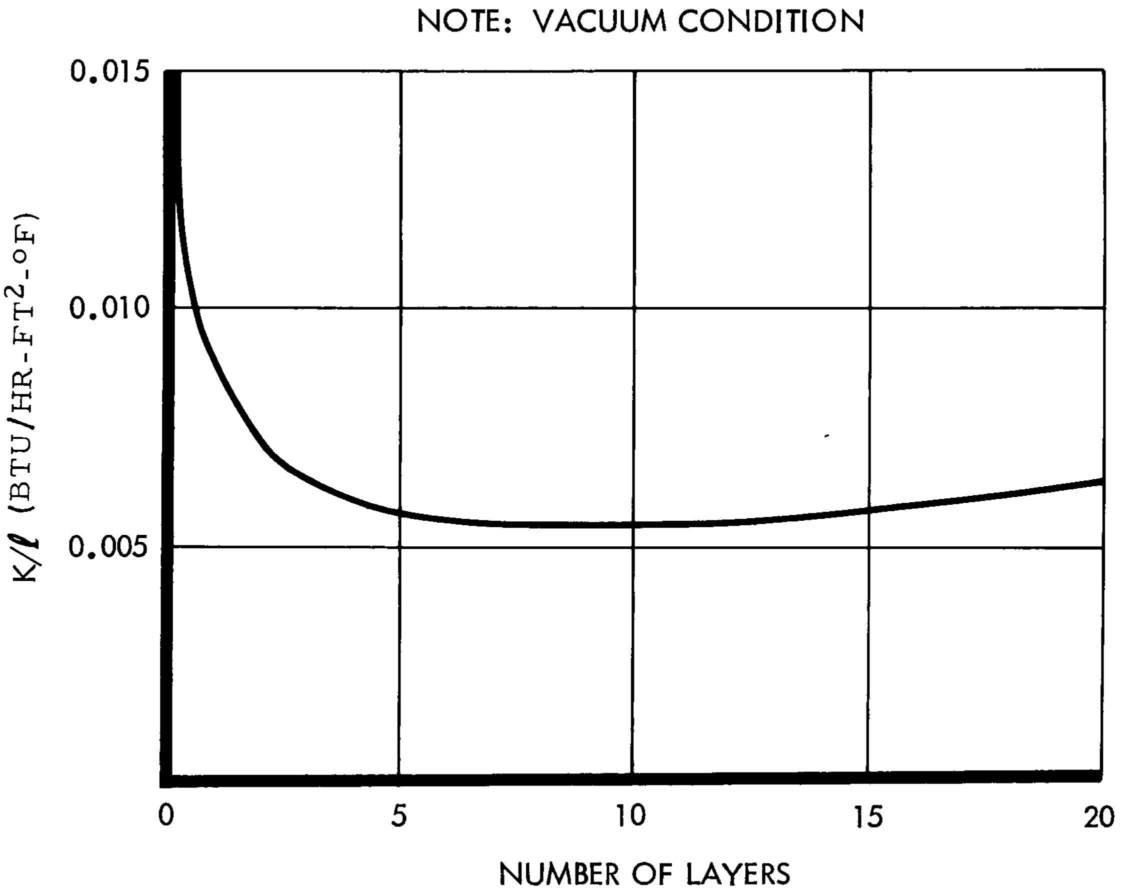


Figure 3-26. Variation of Conductivity of Installed Multiple Layer Reflective Insulation as a Function of Number of Reflective Layers

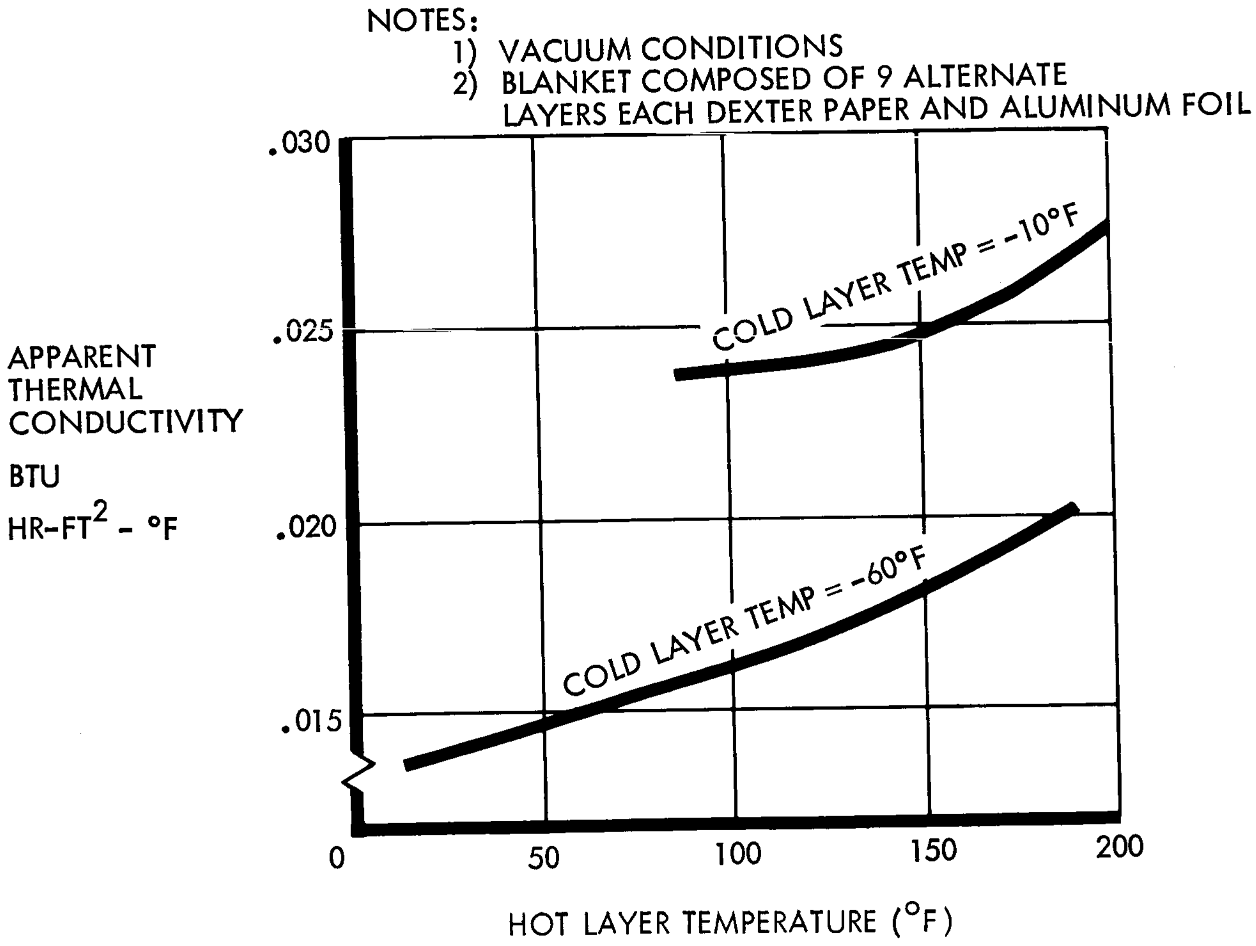


Figure 3-27. Temperature Effects on Multiple Layer Insulation Blanket

difference, affects the performance of multilayer insulation. In a vehicle operating at various distances from the sun, this effect may be of significant and undesirable magnitude.

Figure 3-28: Net Heat Loss from a Spacecraft Caused by an Uninsulated Area - The spacecraft must communicate with or through its external environment, as by antennae, sensors, or experiment apertures. This necessity combined with attachment of the insulation to complex surfaces and joints usually results in localized areas that are insulated poorly, if at all. Inspection of Figure 3-28 shows that these areas could easily lead to intolerable heat losses in a low powered spacecraft. A radioisotope heater could make up for these losses in a direct manner, where most required.

Figure 3-29: Control of Radiant Heat Flux by Louvers - Approximate performance data is presented based on the current Pioneer vehicle design. For a temperature increase from 40° F to 100° F, heat rejection increases by a ratio of 5:1.

Figure 3-30: Size and Power Effect on Internal Temperatures of Insulated Spherical Spacecraft - Temperature control of low powered spacecraft by insulation alone, although adjustable to provide satisfactory internal temperatures at a given distance from the Sun, will result in unacceptable temperatures at different distances. The addition of a simple louver system can afford substantial improvement in control if the spacecraft is properly oriented. The further addition of a heater may be necessary to obtain the degree of control required, depending on vehicle size, insulation efficiency, and distances from the Sun. The temperatures of large spacecraft are more difficult to control than those of smaller ones. A small vehicle with a louver/heater system can cope with greater distance than the same system in a larger vehicle. However, consideration of practical aspects of insulation reveal a greater extreme in heater requirements as shown in Figure 3-30, particularly for smaller spacecraft.

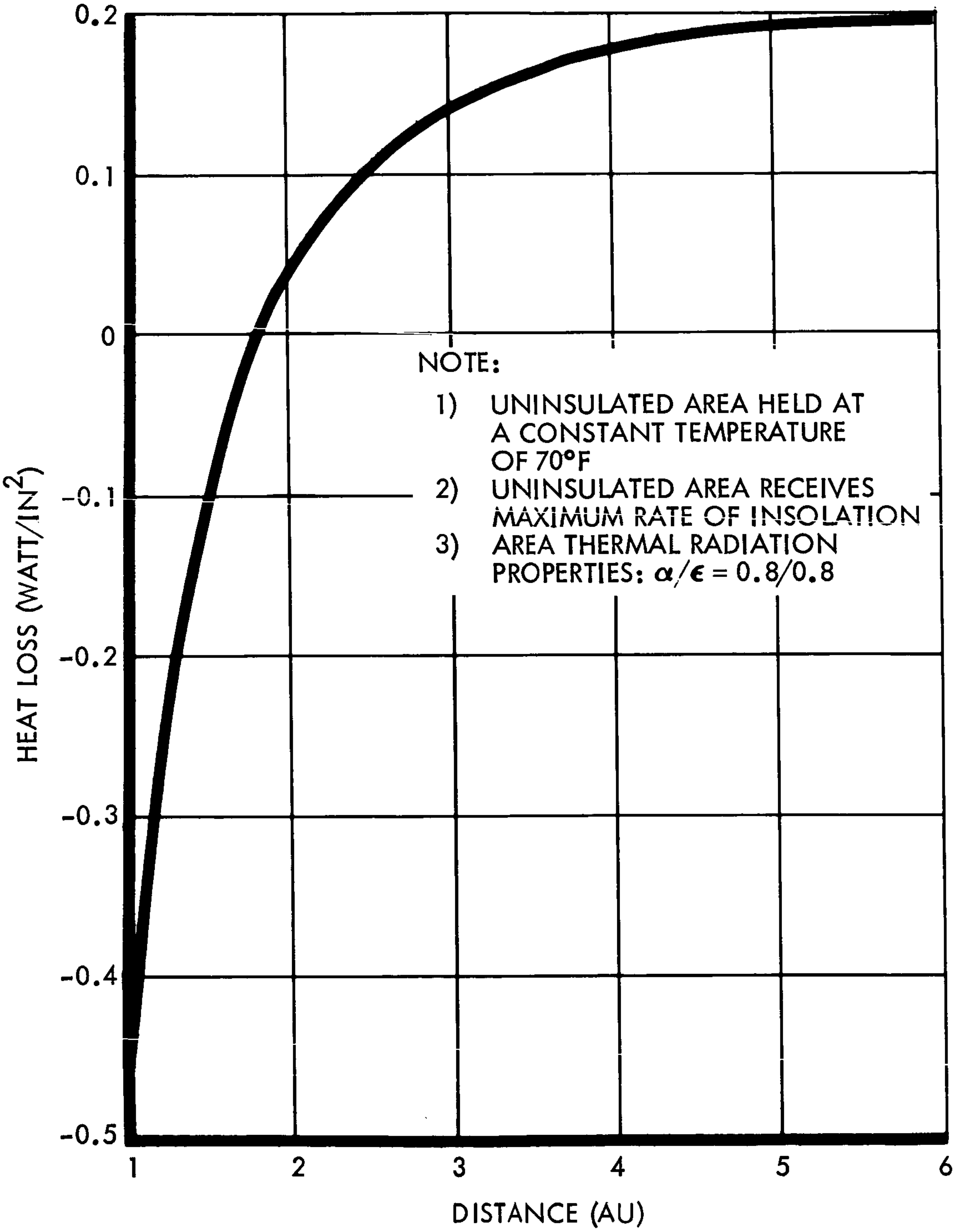


Figure 3-28. Net Heat Loss Caused by an Uninsulated Area in a Spacecraft

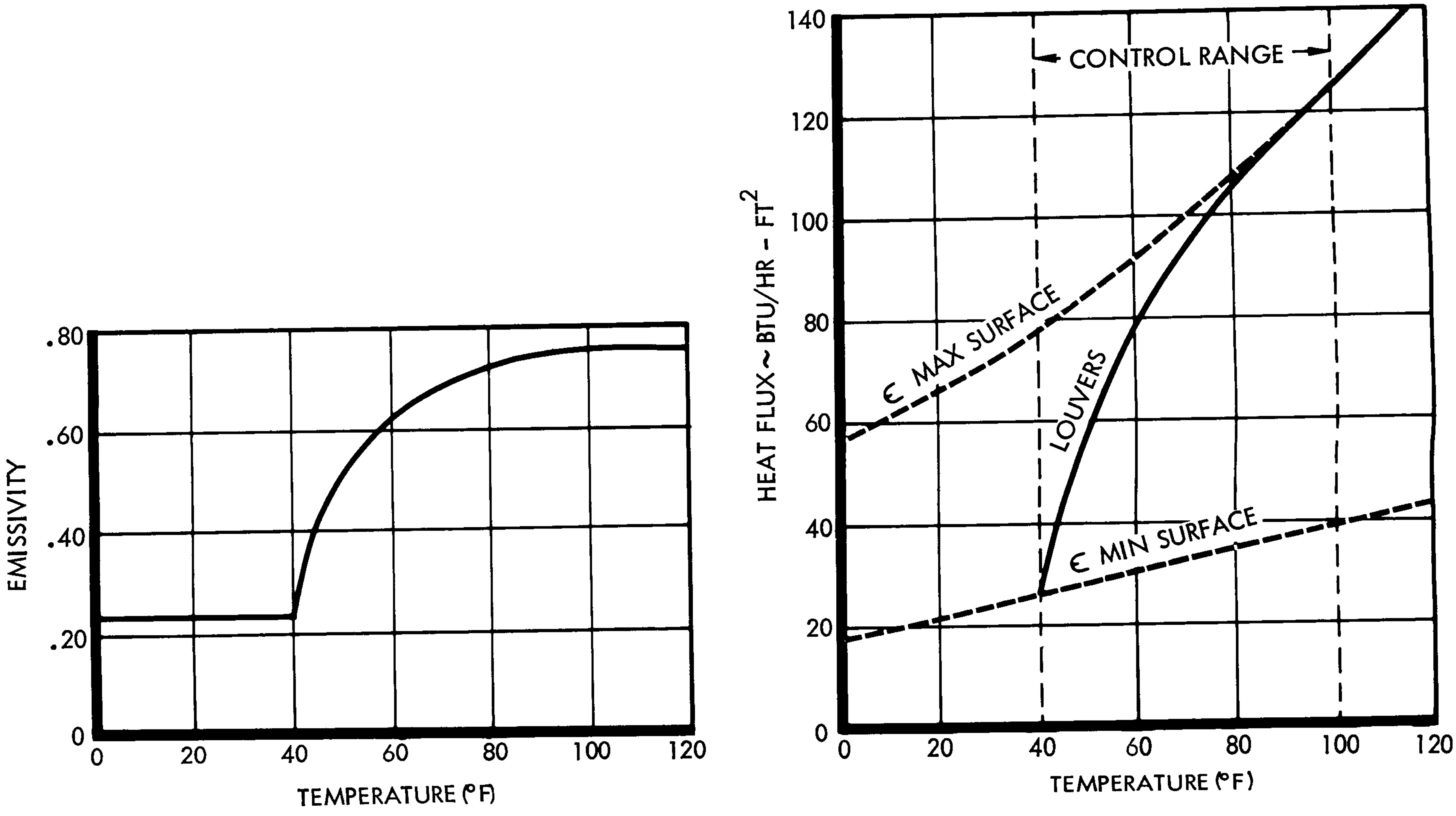


Figure 3-29. Control of Radiant Heat Flux by Louvers

NOTES: 1) SPACECRAFT INSULATION CONSISTS OF 6 ALTERNATE LAYERS EACH OF ALUMINUM FOIL AND DEXTER PAPER.
2) EXTERNAL = 0.15/0.05.
3) AVERAGE CONTINUOUS LOAD IS 10 WATTS.
4) REFERENCE APPENDIX C.3 FOR MODEL DESCRIPTION

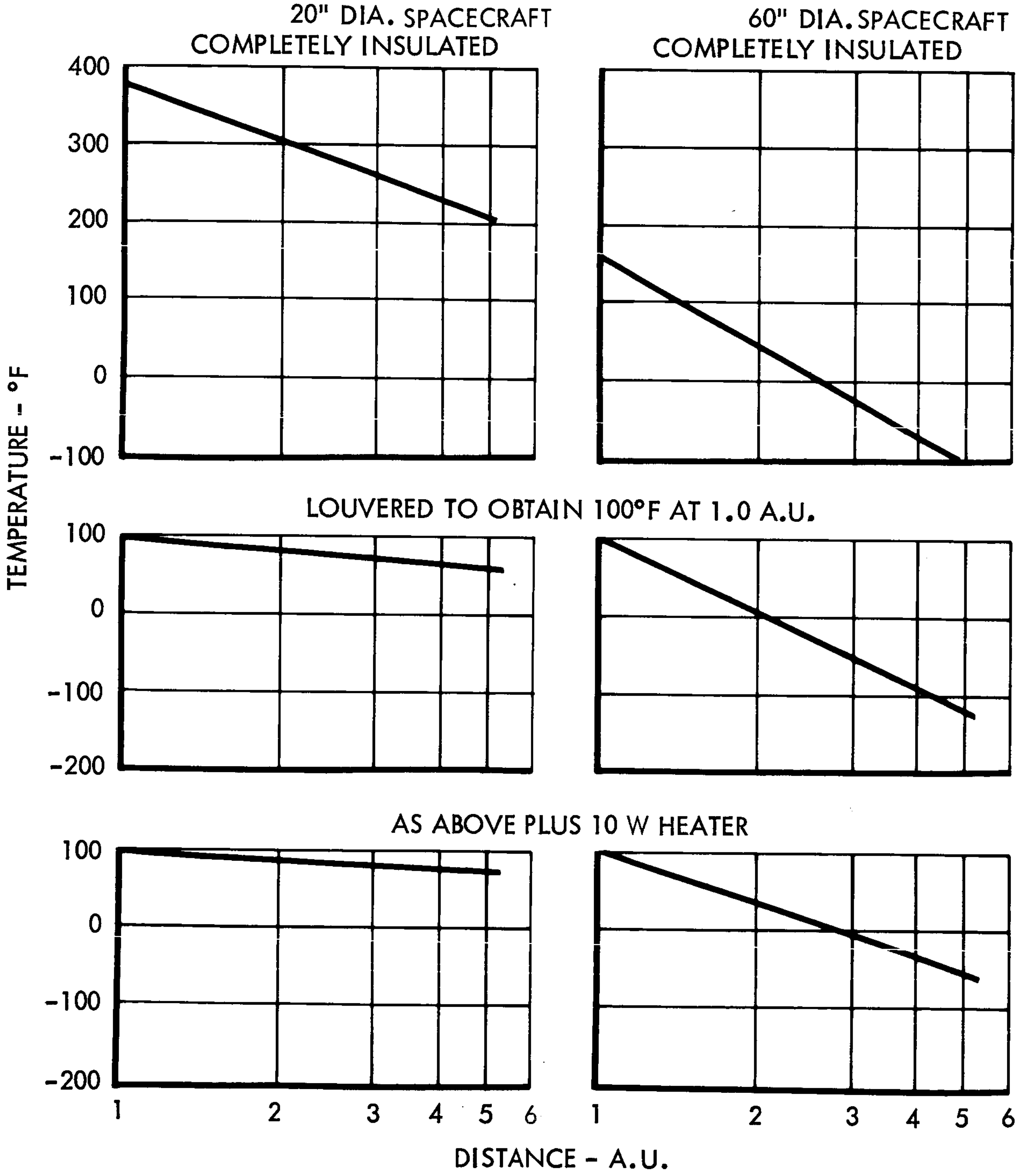


Figure 3-30. Size and Power Effects on Internal Temperatures of Insulated Spherical Spacecraft

General Comment - Batteries are generally less tolerant of low temperatures than other electronic equipment, therefore, in applications where spacecraft heating is required, it will usually be advantageous to locate the heater at the battery, since less heat is needed, better control of battery temperatures is obtained, and useless heating of other equipment is minimized. It has been demonstrated, that a 10-watt heater can readily maintain a typical battery at ~~4° C~~ 22°C above its immediate environment.

From the background of the preceding information, it is possible to construct a general power-size relationship for a class of spherical, relatively well insulated spacecraft of the type considered. Figure 3-31 depicts the relation between vehicle size, average power level, and mission distance. Given any two of the three variables, the third can be determined within the ranges considered.

It is shown that the temperatures of relatively small spacecraft can be controlled over long distances with a modest quantity of power. As the spacecraft size increases, temperature control generally requires a substantial increase in power. The addition of a battery heater to make up for any power deficiency would have three beneficial effects:

a) Directly increase nominal spacecraft range as shown in Figure 3-31.

b) Permit the battery to be operated at temperatures appreciably above the local levels.

c) Assuming that other spacecraft equipment can operate satisfactorily at temperatures of less than 4° C, an increase in distance or reduction in power can be obtained.

3.3.4 Temperature-Controlled Battery Installations and Performance in Space

Analyses were made on four applications of heated battery installations. The analyses considered two applications of the device described in Section 5, one application of a closely related device, and an application of a heater and louver system. Brief and simplified analyses represent typical performance data on the operation of a radioisotope battery heater.

NOTE:

1) S/C INSULATION - - 6 ALTERNATE LAYERS EACH, ALUM FOIL AND DEXTER PAPER, $K/\ell = .021$ BTU/FT^2 - HR - °F EXTERNAL $\alpha/\epsilon = 0.15/0.05$
2) TEMPERATURE AT 1.0 AU = 100°F
3) TEMPERATURE CONTROL BY LOUVERS. LOUVER AREA ≅ 0.03 SQ. FT. / WATT, LOUVER WT≅2.8 LB/SQ. FT.

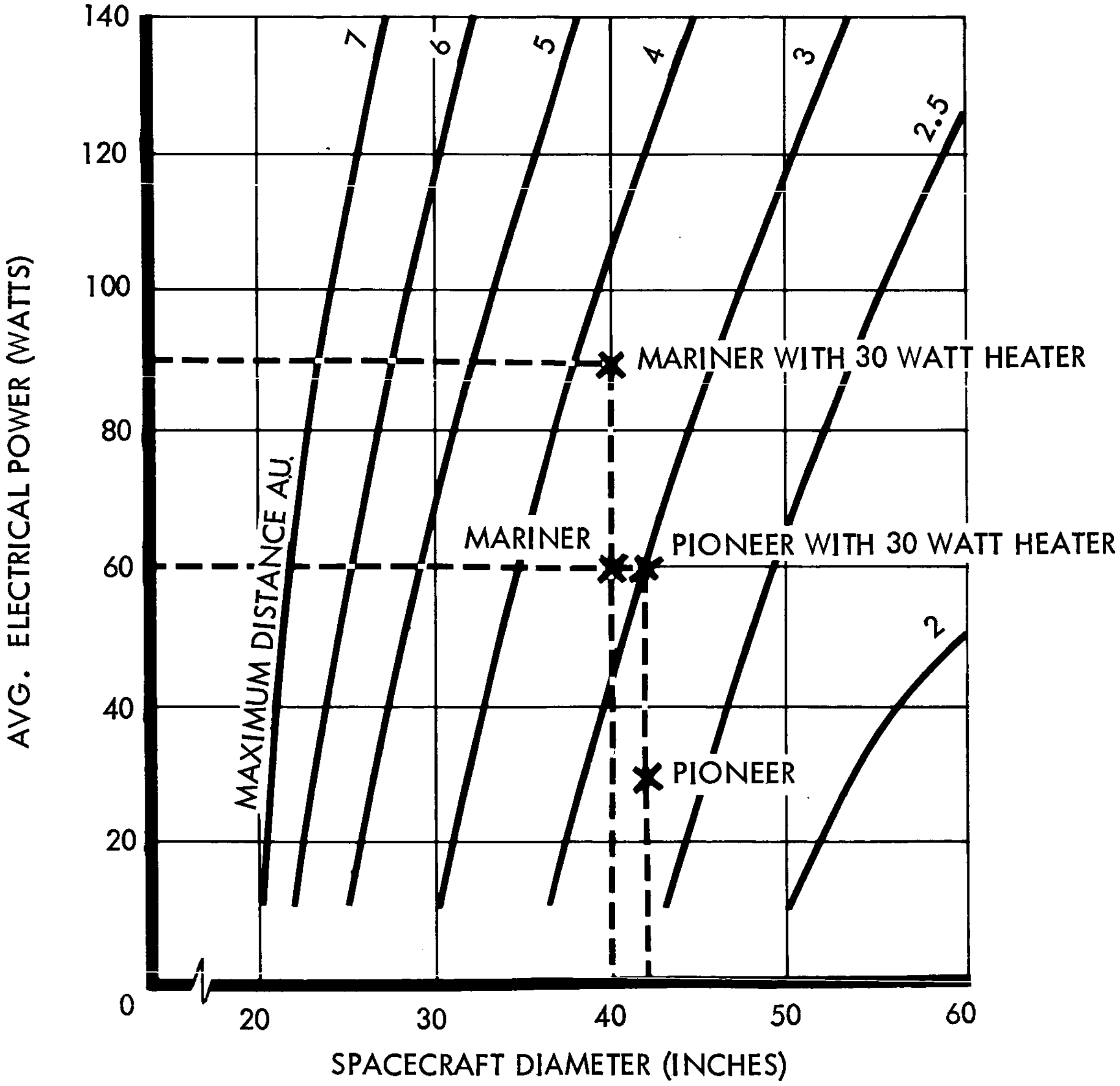


Figure 3-31. Power to Size Requirements for Spherical Spacecraft Above 40°F

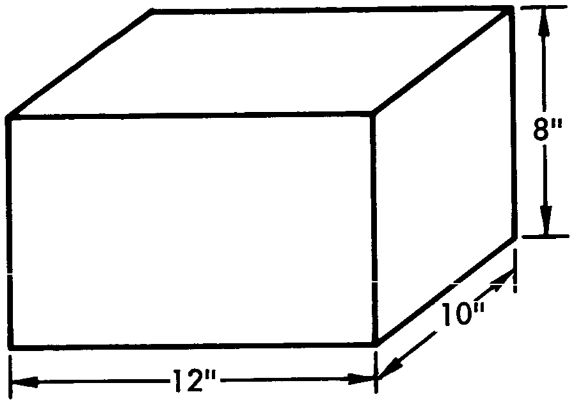
Battery temperatures were maintained within the approximate range of 4° C to 38° C in a simulated spacecraft environment that varied from -21° C to +24° C. The battery heater ratings were assumed to range from 10 to 30 watts. The heaters were further assumed to be powered by radioisotopes, and thus could not be turned off. In addition to maintaining suitable battery temperatures in the midst of low ambient temperatures, the heaters also delayed the formation of a low temperature environment. The extent of this effect, however, could not be calculated since specific spacecraft designs and missions were not available.

Unless a direct path to the external environment is available to the battery heater, the interior of the spacecraft is also heated and must be utilized for ultimate rejection of all or much of the battery heater output. Numerous thermal interactions, most of which are unrelated to battery operation, determine the overall battery environment; no general definition of the interaction of the battery heater and spacecraft equipment could be firmly specified. It was assumed that the maximum environment temperature was about 100° C, but the circumstances determining this level has not been defined. It was further assumed that it is desirable to operate the battery above 4° C when the environment is below that temperature.

The battery configuration considered is shown in Figure 3-32. Of the four battery/heater configurations studied, No. 1 is identical to that of Figure 5-1 in Section 5. The remaining configurations are shown in Figures 3-33 to 3-35.

Table 3-13 provides the results of the study. Minor heat transfer paths, such as by conduction through insulation caps or titanium rods of small diameter were not considered, since they account for the transfer of less than 1 watt under the worst condition, and somewhat less under more favorable conditions.

Considering the configurations in turn, the following comments are made. Configuration No. 1, (Figure 5-1) is best suited for situations requiring extensive heating of the mounting platform under all conditions, thus requiring a relatively high heater power rating. A tolerable lower



NOTE: LOWER 10" X 12" AREA CONSIDERED AS MOUNTING SURFACE

Figure 3-32. Basic Battery Configurations

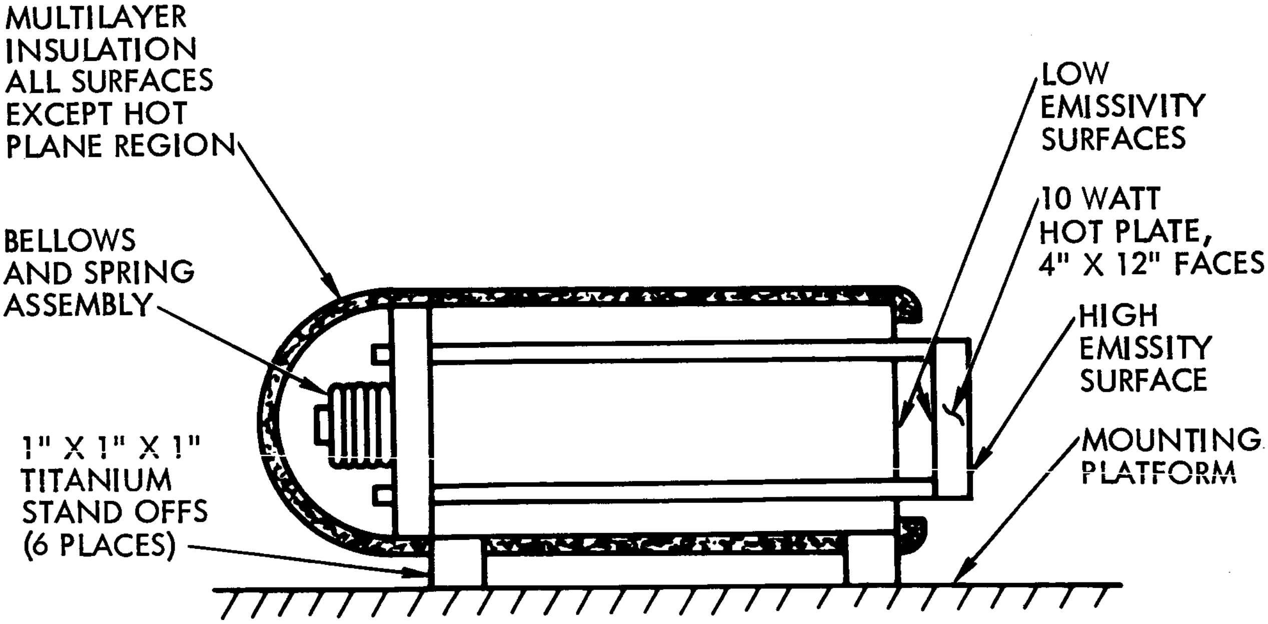


Figure 3-33. Schematic Diagram of Configuration No. 2 Battery Installation

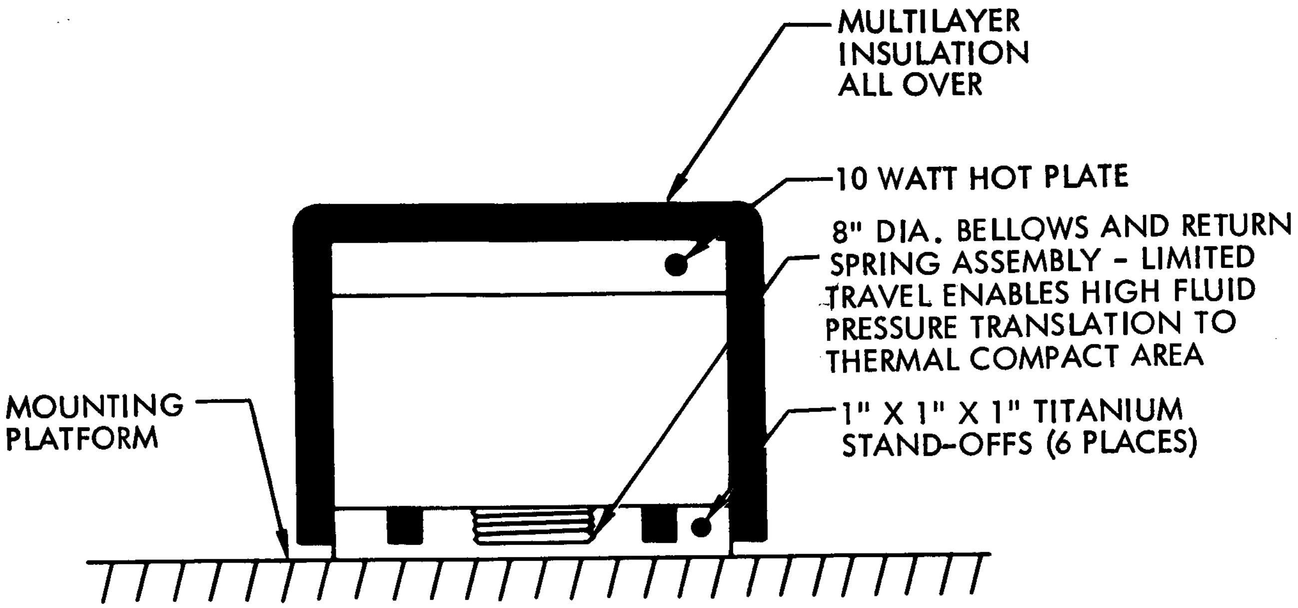


Figure 3-34. Schematic Diagram of Configuration No. 3 Battery Installation

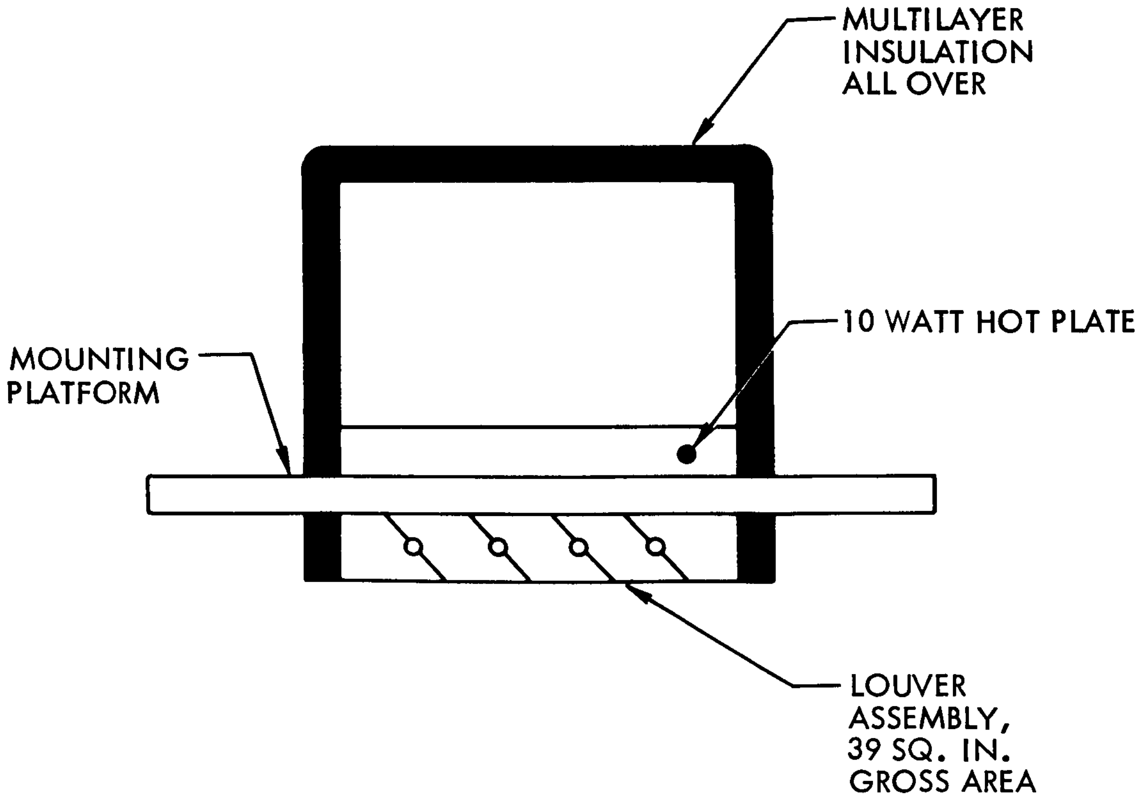


Figure 3-35. Schematic Diagram of Configuration No. 4 Battery Installation

Table 3-13. Battery Heater Performance For Various Types of Heater Configurations

Configuration No.	Heater Rating (Watts)	Maximum Temperature (°F)		Minimum Temperature (°F)		Comments
		Environment (Specified)	Battery	Environment	Battery (Specified)	
1	30	75	100	15	40	Most effective utilization of configuration is in situations where substantial spacecraft heating, as well as battery heating, is required.
2	10	92	100	- 6	40	Hot-plate temperature is about 200°F when environment temperature is maximum.
3	10	86	100	0	40	Control action is equivalent to Configuration #2 but construction is simpler. Small differences in temperatures between #2 and #3 have little significance, depending only on minor differences in design assumptions. Temperature levels in either case should be approximately equal.
4	10	100	100	See Comment	40	At battery and hot-plate temperatures of 40°F, only 2 watts are radiated by louvers. 8 watts are available for heating local structure to some temperature between remote structural and battery temperatures.

environment temperature would be considerably higher with a heater of lower rating but identical in size.

Configuration No. 2 (Figure 3-33) is a variation of Configuration No. 1 in which the mounting attachment is made to the battery pack rather than to the hot plate. Although a heater with less power is used, its performance is somewhat superior to the higher-powered heater of Configuration No. 1. As indicated previously, however, apparent superiority may not be true on a complete systems basis. It is also noted that Configuration No. 2 tends to distribute heat to more remote areas than Configuration No. 1.

Configuration No. 3 (Figure 3-34) is thermally equivalent to Configuration No. 2, with respect to the battery. Thermal contact is established by fluid pressure, rather than by spring force, resulting in a simple construction. These features are believed to be advantageous and a Disclosure of Invention covering Configuration No. 3 has been submitted.

Configuration No. 4 (Figure 3-35) is not related to the other three, except for the presence of the heater. It is, however, presented as an established method of rejecting the heater output directly to the space heat sink without intermediate heating of the spacecraft when temperature levels are high and the louvers fully opened. Conversely, when temperature levels are low, the louvers are closed and most of the heater output is directed to the spacecraft structure adjacent to the battery. As long as this adjacent structure is maintained above 4° C, the battery temperature will tend to be about equal or slightly higher.

In typical spacecraft structures, regions remote from the battery could be considerably colder than 4° C under the circumstances, possibly less than -18° C. In the absence of specific detail, however, these remote temperatures cannot be further defined.

Batteries can readily be maintained at temperatures of 4° C to 38° C in a spacecraft environment that ranges from -21° C to +33° C, by the use of radioisotope heaters ranging in power output from 10 to 30 thermal-watts. Either bellows-type or louver temperature controls may be used.

It should be noted that the environmental temperature limits quoted are derived from conservative preliminary assumptions concerning controller design features. A thorough design study would extend the tolerable range of environment, particularly at the lower limit.

The results of a detailed analysis of battery temperatures are shown in Figure 3-36. The lower temperatures show the system working just below the limits of its control capacity. The higher temperatures are at a point within the control range, and indicate the cyclic action of the controller. It is estimated that the cyclic action indicated is exaggerated in amplitude as well as frequency, because of certain simplifications which were employed in the analysis to avoid computation costs not warranted for purposes of demonstration.

The recommended placement of the hot plate is between the battery and the mounting platform. It is also feasible to place the battery between the hot plate and mounting platform, but investigation has shown that this arrangement leads to undesirable battery temperature gradients and is less responsive to the ambient environment.

3.3.5 Heated Battery Configuration for Terrestrial Applications

Batteries heated by radioisotopes is a suitable method for terrestrial applications if satisfactory solutions to non-thermal problems are available. The applications consist basically of either operation or standby in frigid climates or other adverse temperature conditions. Specific applications which might be considered include military aircraft or automotive equipment, oceanographic and special-purpose buoys, and meteorological data stations.

A thermal analysis has been made of a heated battery system employing a 30-watt heater and the temperature control system described in Section 5. The basic battery pack is shown in Figure 3-37. The battery is assumed to be insulated and mounted as shown in Figure 3-38. The temperature control details have been omitted from the illustration.

At an ambient temperature of 24° C, the maximum battery temperature will be approximately 38° C. At any given low ambient temperature, the battery temperature is influenced strongly by the detail design of the

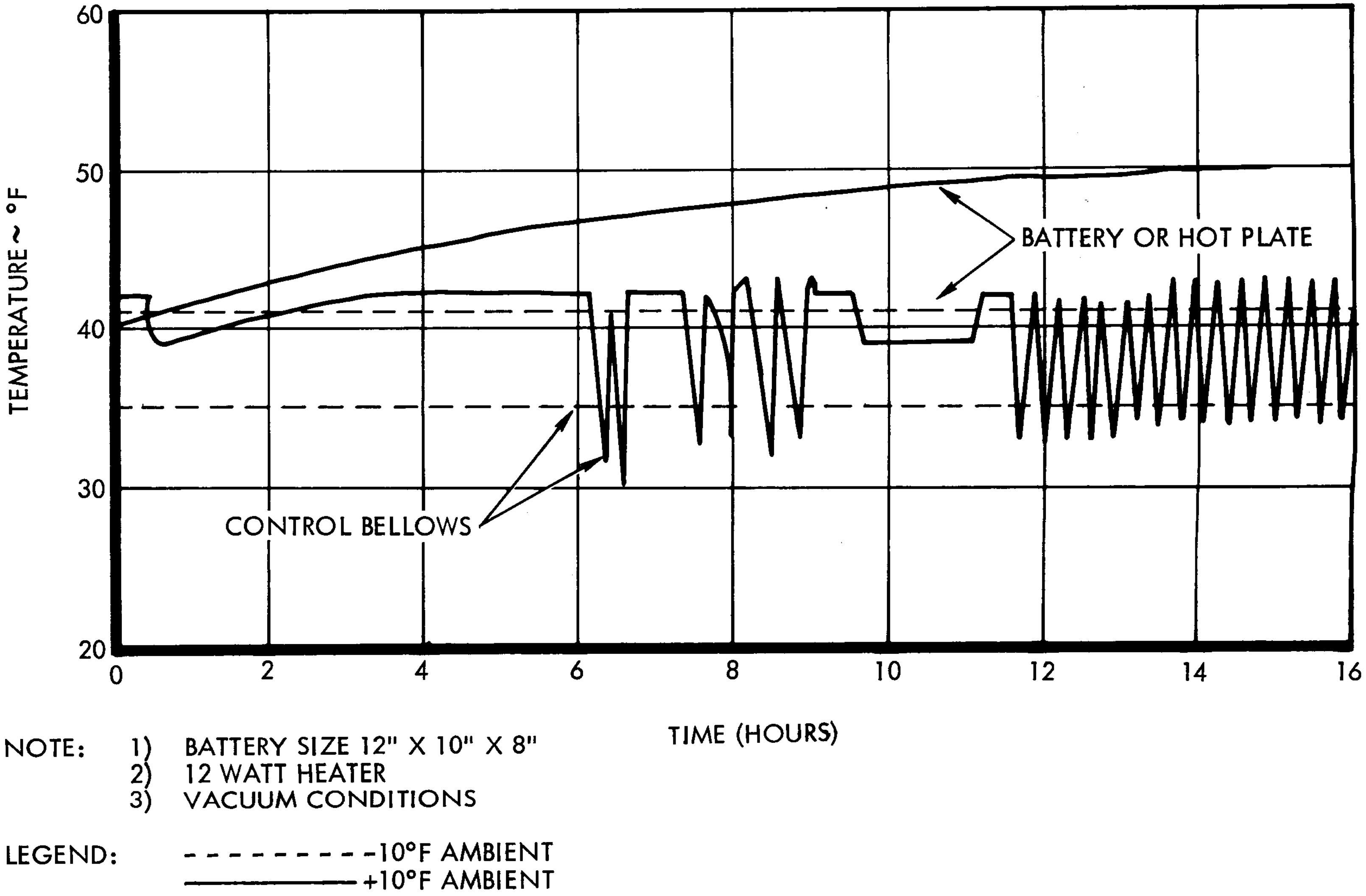
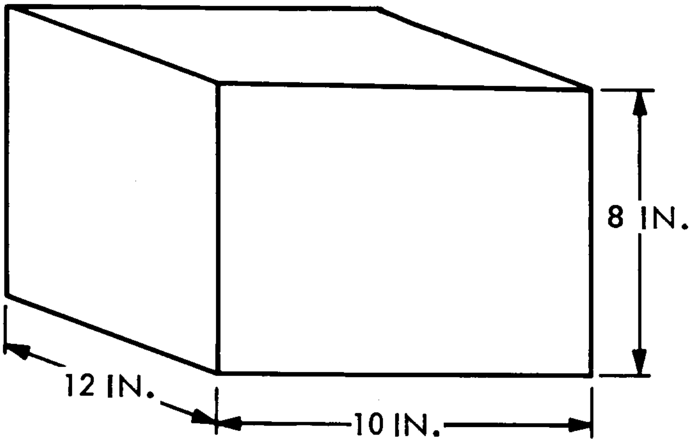


Figure 3-36. Typical Battery Temperatures Using Hot Plate and Bellows Controller



NOTE:

BASE IS ONE OF
10 IN. X 12 IN. SURFACES

Figure 3-37. Basic Battery Size

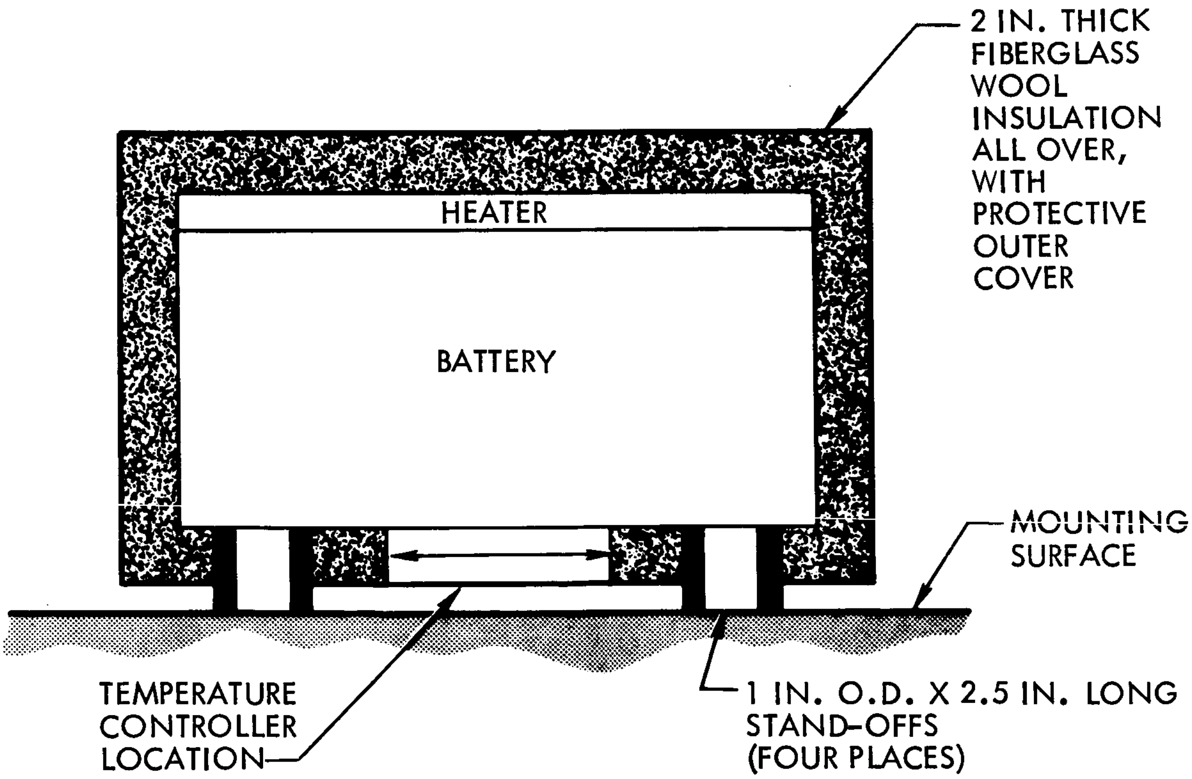


Figure 3-38. Battery Installation

standoffs shown in Figure 3-38. (At high ambients the standoffs have a smaller effect.) Table 3-14 lists battery temperatures for an ambient of -50° C, considering several standoff designs. The temperature controller will, of course, maintain a relatively constant intermediate battery temperature over a range of ambients, as determined by the controller set point. This is illustrated by Figure 3-39.

If a radioisotope-heated battery can be adequately shielded, or if the level of radioactivity is low, many terrestrial applications are possible. The life of isotope fuel in the battery heater may exceed that of several batteries, and when necessary, the fuel can be replaced readily.

Table 3-14. Battery Temperatures in an Ambient of -50° C

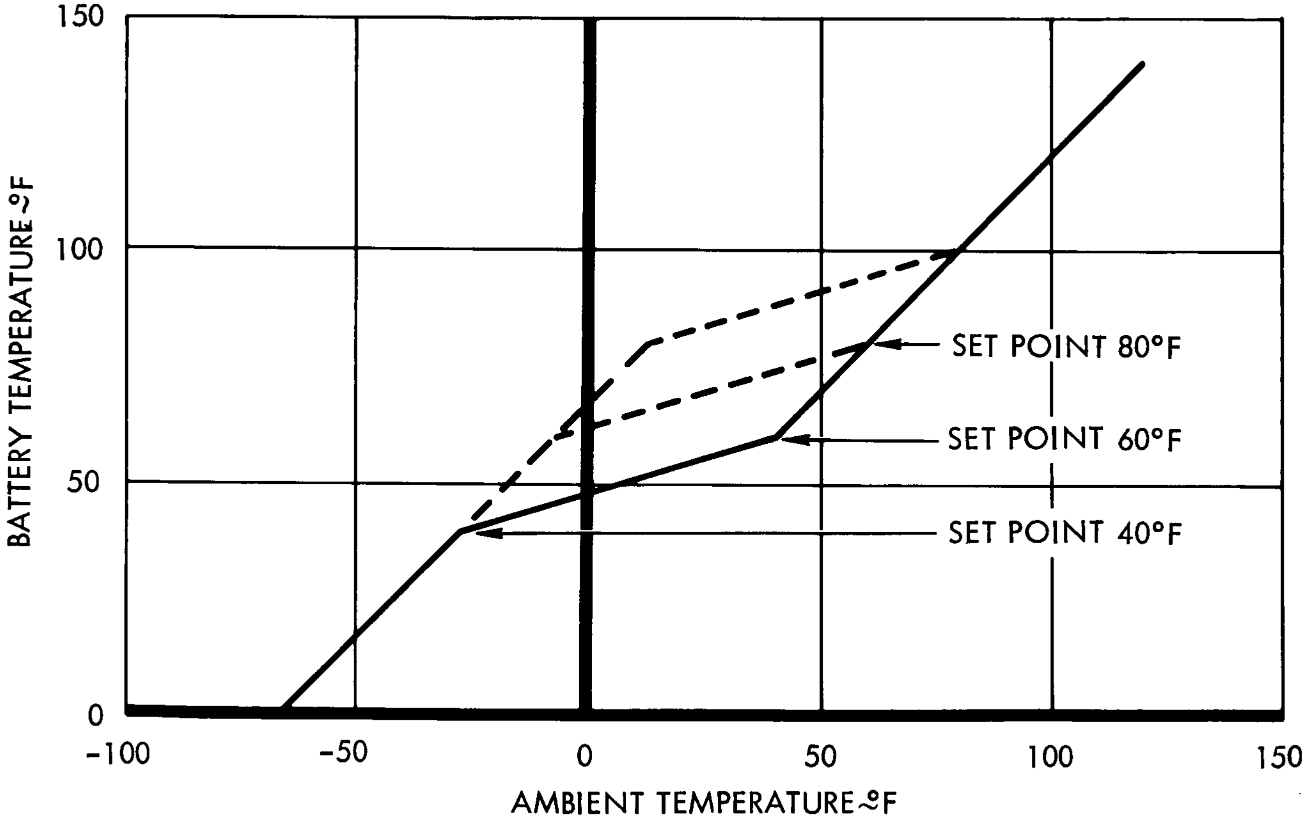
STANDOFF MATERIAL	STANDOFF DESIGN	BATTERY TEMPERATURE (°C)
STAINLESS STEEL	SOLID ROD	-24
TITANIUM	SOLID ROD	-19
PLASTIC OR FIBERGLASS	SOLID ROD	- 5
STAINLESS STEEL	HOLLOW TUBE, 1/8" WALL	-13
TITANIUM	HOLLOW TUBE, 1/8" WALL	- 6

NOTE:

A. BATTERY HEATER RATED AT 30 WATTS.

B. BASIC STANDOFF DIMENSIONS ARE 2-1/2" LONG BY 1" O.D.

C. BATTERY TEMPERATURES MAY BE ADJUSTED FOR OTHER AMBIENTS BY ADDING THE DIFFERENCE BETWEEN A GIVEN AMBIENT AND -50° C.



NOTE: AMBIENT TEMPERATURE DEFINED AS THAT OF SURROUNDING AIR AND MOUNTING SURFACES

Figure 3-39. Battery Temperatures Using an Active Thermal Controller and a Radioisotope Heater

3.4 SELECTION OF HEATED BATTERY SYSTEMS FOR MODEL ANALYSIS

This paragraph presents a brief summary of the analyses conducted to select the battery system and battery system components to be implemented into a model for further study.

3.4.1 Mission Considerations and Tradeoffs

Table 1-2 and Section 2. presented summaries of potential missions which might be considered for low temperature battery applications. Of these missions it was decided to consider the following for analysis:

a) Asteroid belt probe (deep space probe)
b) Jupiter flyby
c) Mars lander
d) Mars hard lander
e) Mercury probe and lander
f) Solar probe

After investigating these missions further in the literature search and industry survey, it was found that the solar probe was not applicable.

The Mercury dark-side lander, a preliminary analysis of which is reported in Section 2., was re-examined in depth as a possible application for heated batteries. It was concluded that the problems associated with defining a functionally sound vehicle were so complex that relatively detailed assumptions and analyses concerning a heated battery could not be justified.

In Section 2. it was stated that, for a simple Mercury lander which approached the planet behind a sun shield, transit and planetary conditions would be ideally equivalent, and an insulated sphere was selected as a lander configuration. It was further noted that this scheme was highly arbitrary. These points, and others, will now be considered further.

Figure 3-40 shows the hot-side temperatures (approximate) of two simple Sun shields in the vicinity of Mercury. Temperatures at the

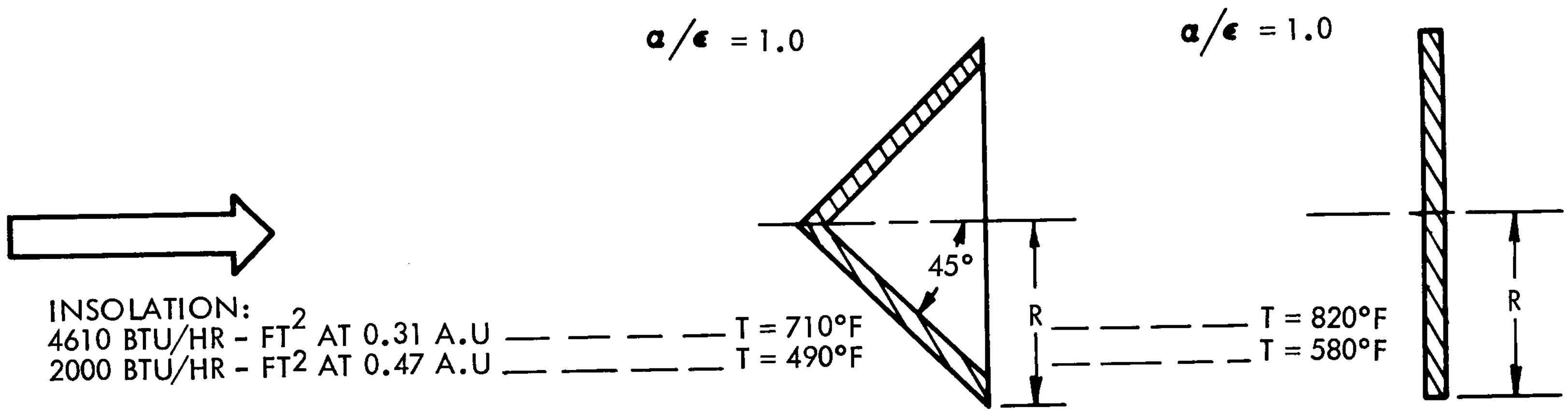


Figure 3-40. Temperatures of Hot Sides of Conical and Flat Sun Shields in the Vicinity of Mercury

perihelion (0.31 AU) and aphelion (0.47 AU) of Mercury are presented. Specifying surface properties that are reasonably stable ($\alpha = \epsilon = 0.8$), the shield temperatures will range from 260° C to 425° C, depending on configuration and distance from the Sun. Corresponding spacecraft surface temperatures are lower, but there will be temperature potential of several hundred degrees Fahrenheit forcing heat into the payload; the resultant heat flux can be 20 to 50 watts. Internal dissipation forms the balance of the spacecraft heat load, and is probably of the same magnitude as the solar load.

In considering the mechanics of landing, it seems advisable to approach the planet in its shadow for a considerable distance to avoid exposing spacecraft surfaces to sunlight during landing maneuvers (hot spots of 650° C could develop). Whatever precautions must be taken to protect against impact damage add to the complexity of the landing problem.

A thermal control method (other than Sun shields) which may be employed is the application of massive thicknesses of high-temperature multi-layer insulation. An insulation material consisting of alternate 1 layers of nickel foil and quartz fiber paper has been tested by the Linde Company at temperatures up to 1200° C. The approximate weight required to reduce solar heat loads is 5 lb/sq ft. If the insulation is to be effective, on a planet as well as in space, additional thickness and weight of insulation may be required because of thin atmosphere. The temperature control system would have to adjust from an external temperature in space of approximately 260° C, to one on the planet of perhaps -240° C. Thermal control employing these methods is considered unsatisfactory at this time.

A reasonably complete outline of a Mercury landing mission, and the spacecraft or capsules required to accomplish it, is considered to be beyond the scope of the present study. Consideration of a heated battery for this mission, beyond what has already been done, seems unjustified at this point.

In reviewing the remaining missions, it was apparent that the Jupiter flyby mission and asteroid-belt probe missions were essentially the same. It was therefore decided to perform a model analysis of the three following missions:

a) Jupiter flyby (deep space probe)
b) Mars soft lander
c) Mars hard lander

These models are discussed in detail in paragraph 3.5.

In selecting the models it was necessary to consider various tradeoffs in power system design in order to select the designs which would meet the requirements for heated-low temperature batteries in general, since the scope of the study could not possibly consider all designs. As part of these tradeoffs, a comparison was made between heating costs versus heater power requirements versus various methods of obtaining the heater power. Figure 3-11 is a chart indicating this cost tradeoff. A second tradeoff was made to compare the weight advantage of the different types of heaters. This tradeoff is shown in Table 3-12.

When it became apparent that the major competitive heaters were electrically powered type using excess power from the power system and radioisotope heaters, a continuation of the tradeoff study produced a comparison of the weight advantage of using direct radioisotope heaters versus electric heaters for several different power requirements. This study was based on the information presented in Figure 3-32 in the form of a generalized power-to-size requirement for a spherical spacecraft. Again the tradeoff was general in nature, because many innumerable tradeoffs are possible by varying designs. The last tradeoff made was to consider the availability of heat as the situation demands. In the case of deep space missions where the temperature is continuously decreasing, heat could be obtained from a steady-state heat source because the power system would experience power source degradation and an increase in the amount of power required to heat the entire spacecraft over time. Therefore, there is a high probability that the power available from a primary power system will be very limited at the end of deep space missions.

In evaluating lander missions, availability of power from the primary power system to heat the battery is usually not possible during periods when the battery should be heated because the spacecraft or lander vehicle is cold and not illuminated. If an rtg is used to supply the power, there may be enough power available; however, it may be more desirable to add a separate heat source available during these periods to reduce the size of the primary power system and avoid having to carry up to one hundred times the radioisotope, in the primary power system, that would be required as a separate source to heat the battery. For hard landers, the difficulty in using the power system as a heat source is even more apparent because the power source should be small and might even be the battery itself in order to stand the shock of landing. On the basis of the above arguments and the above studies, it was concluded that the models to be developed in the study would be restricted to those utilizing constant heat sources and that, until advanced technology is available on batteries to withstand freezing and on chemical heaters to operate in extremely cold environments, no further consideration should be given to short period heating studies because such heaters are generally not adaptable to most space missions.

3.4.2 Battery Specifications

From the technical analyses conducted in the above paragraphs the battery selected for analysis in the model was defined as follows:

a) The battery would be a secondary battery, probably of the nickel-cadmium or silver-cadmium type.

b) The thermal conductivity of the battery should be on the order of 25 Btu/hr-ft^2-$^\circ$F to minimize the temperature differentials within the battery and to provide for balanced cell operation.

c) Battery construction should be such that the battery would be mounted to a baseplate. The battery would be an integrated thermal design structurally capable of withstanding the dynamic environments specified in paragraph 3.1.3, and capable of restraining each individual cell to intercell pressure differential levels of 250 psi minimum.

d) The battery selected should conform with the whr/in^3 nomograph specifications shown in Figure 1-3.

e) The battery selected should not require remote activation in space and should be capable of continuously operating on the bus to supply load transients and bus regulation if necessary.

3.4.3 Heater Specifications

The heater design selected for detailed model analysis would be a radioisotope heater utilizing either Pm-147 or Pu-238 radioisotopes. The sources would be mounted in a heat sink capable of being applied to the bottom of the battery. The heat source should provide for varying the amount of heat installed in the heat source from 5 to 50 watts. If an electrical heater is used, it must supply heat in accordance with the same specifications as the radioisotope heater.

3.4.4 Thermal Controller Specifications

Thermal control specifications to support the above heaters are of two types:

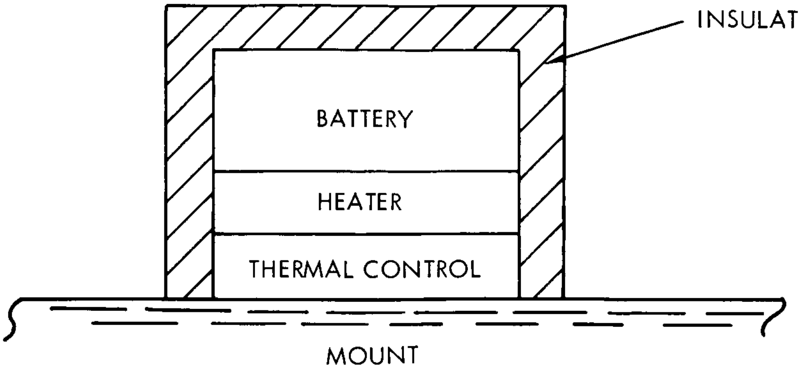
a) Thermostatic control switch which controls an electrically heated thermal blanket would be required for electrical heating. This type of switch is a standard item used on present spacecraft designs.

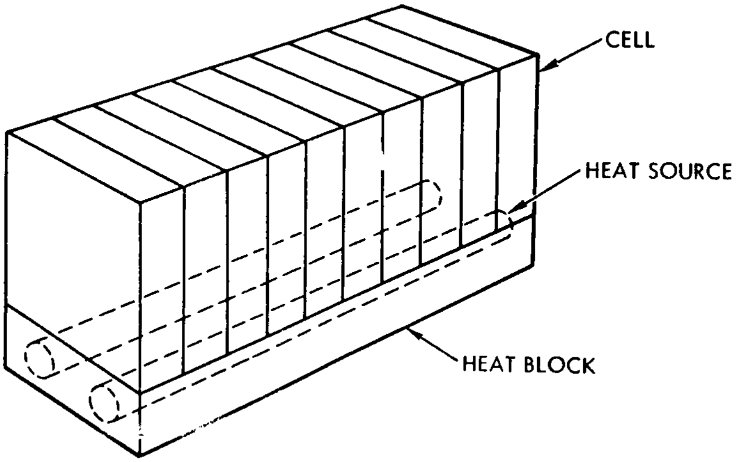
b) To use a constant heat source such as a radioisotope, a thermal switch would be required to regulate the transfer of heat from the battery to its spacecraft mounting surface. This controller should be capable of varying the thermal conductivity across the mounting interface from near zero to about 2 $Btu/hr\text{-}ft^2\text{-}^\circ F$. This thermal switch should connect the battery to its mounting surface when the battery becomes too hot (above 30° C) and should be capable of isolating the battery from its mounting surface at any temperature below 5° C.

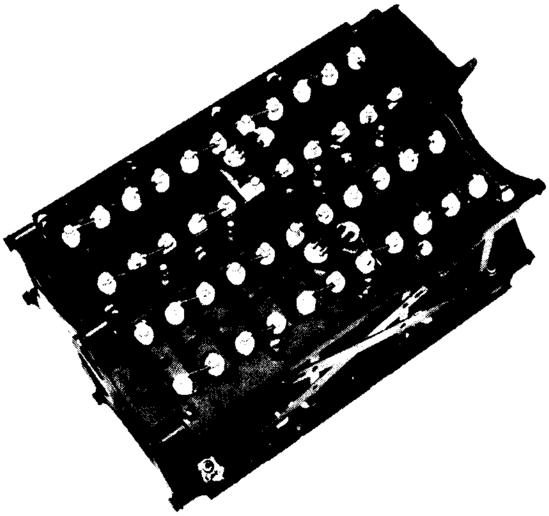
3.4.5 Design Configurations

The design configurations selected for analysis is shown in Figure 3-41. The figure shows the overall battery design, the heat block design concept, a typical standard spacecraft battery design, and a thermal control element. The models discussed in the next paragraph in general make use of this basic design concept. Several analyses were conducted to determine the proper arrangement of these elements such as placing the heater on top of the battery and placing the heater below the thermal controller. However, none of the arrangements performed as well as the one selected, as defined in the technical discussion in previous paragraphs.

LOW TEMPERATURE BATTERY DESIGN







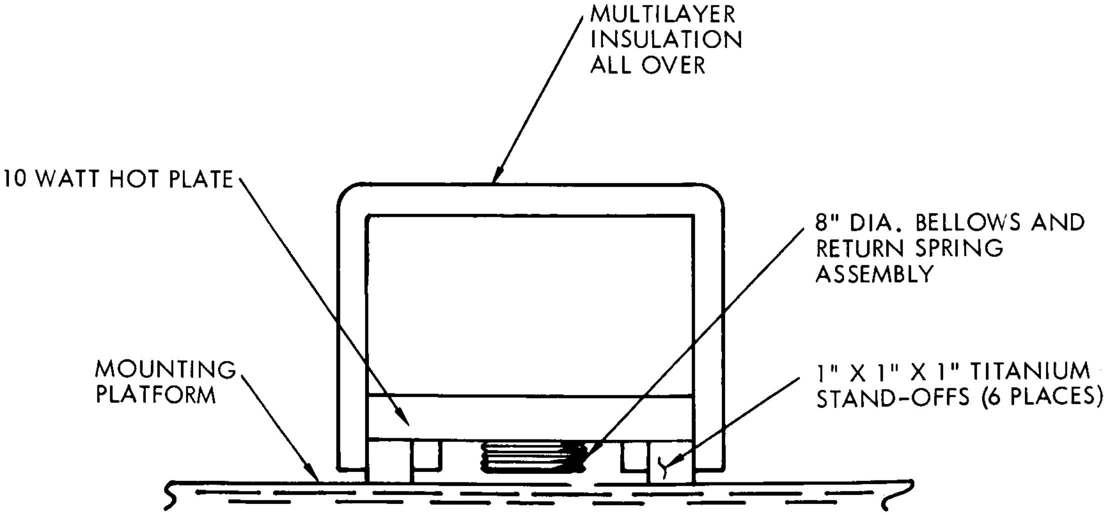


Figure 3-41. Low Temperature Battery Design

3.5 MODEL ANALYSES

The following discussion summarizes the model analyses which were conducted to verify feasiblity of the design concepts selected. In general, the models demonstrated feasibility and established design parameters for a low temperature battery.

3.5.1 Deep Space Probes (Analyzed in terms of a Jupiter flyby configuration

The thermal capabilities of a particular spacecraft configuration, described in Figure 3-42 were examined to determine if deep space probes were for heated batteries. A temperature control system, devised for the purposes of this study, is shown in Figure 3-43. In this control system, the power versus size relationship to maintain specific moderate internal temperatures is shown in Figure 3-44. When the 100-watt power system design outlined in paragraph 3.1 was superimposed on this model, it was found that no heater would be required for the battery unless the battery was located in a local cold spot area. However, for larger spacecraft or lower power missions, Figure 3-44 establishes the expected crossover points where heaters would be required. Note: Figure 3-32 previously discussed is an extension of this concept for deep space probes using spherical spacecraft.

From these analyses and models, it was concluded that for deep space missions battery heating may not only be highly desirable but it may also be absolutely necessary for certain types of vehicles. The heat may be provided directly and with the least dependence on other spacecraft systems by using radioisotope heating sources sized either to heat the whole spacecraft system to a minimum battery operating temperature level or to provide a differential temperature between the spacecraft platform and the battery to maintain the battery within its useful operating temperature range within the spacecraft.

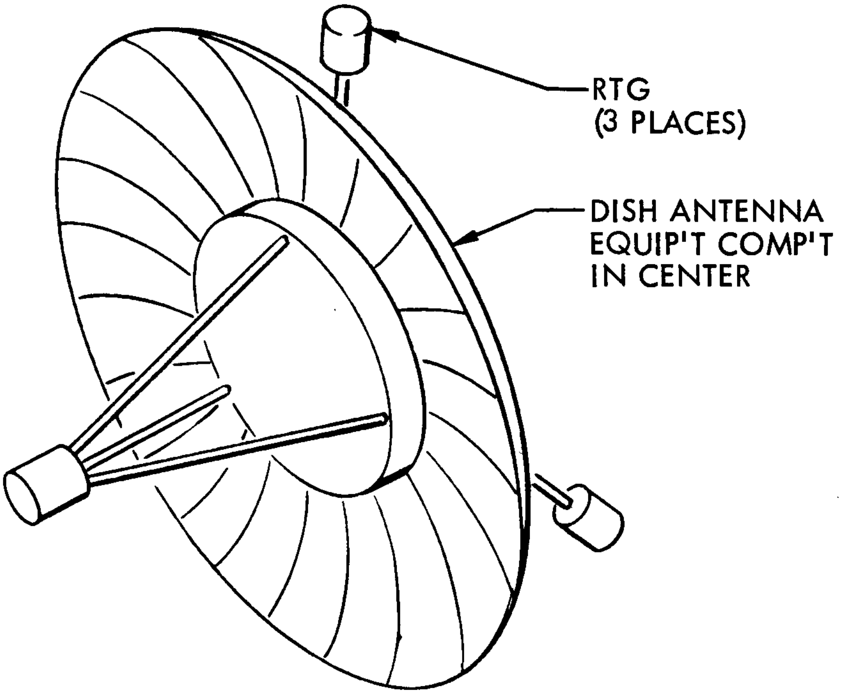


Figure 3-42. Jupiter Flyby Configuration Selected for Model Analysis

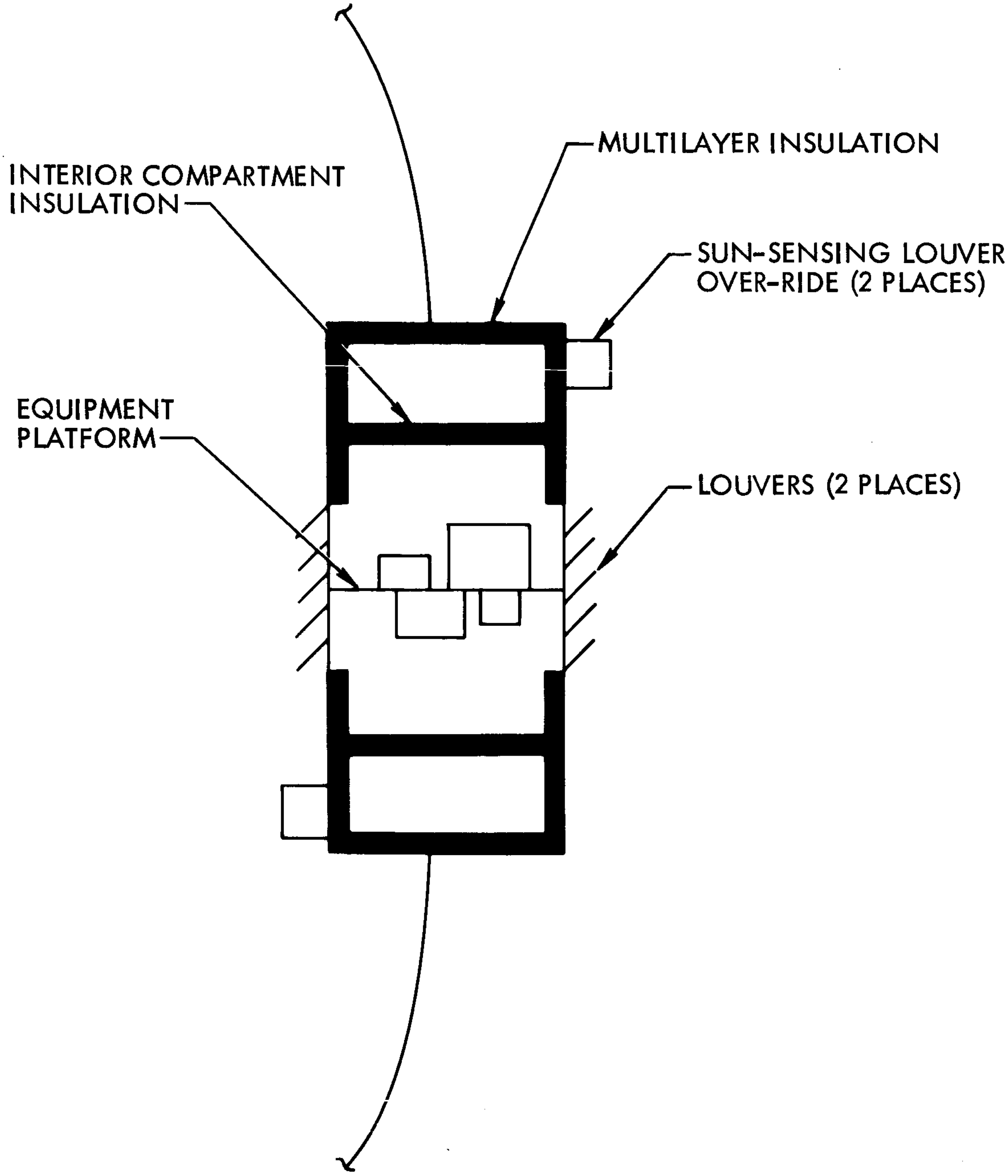
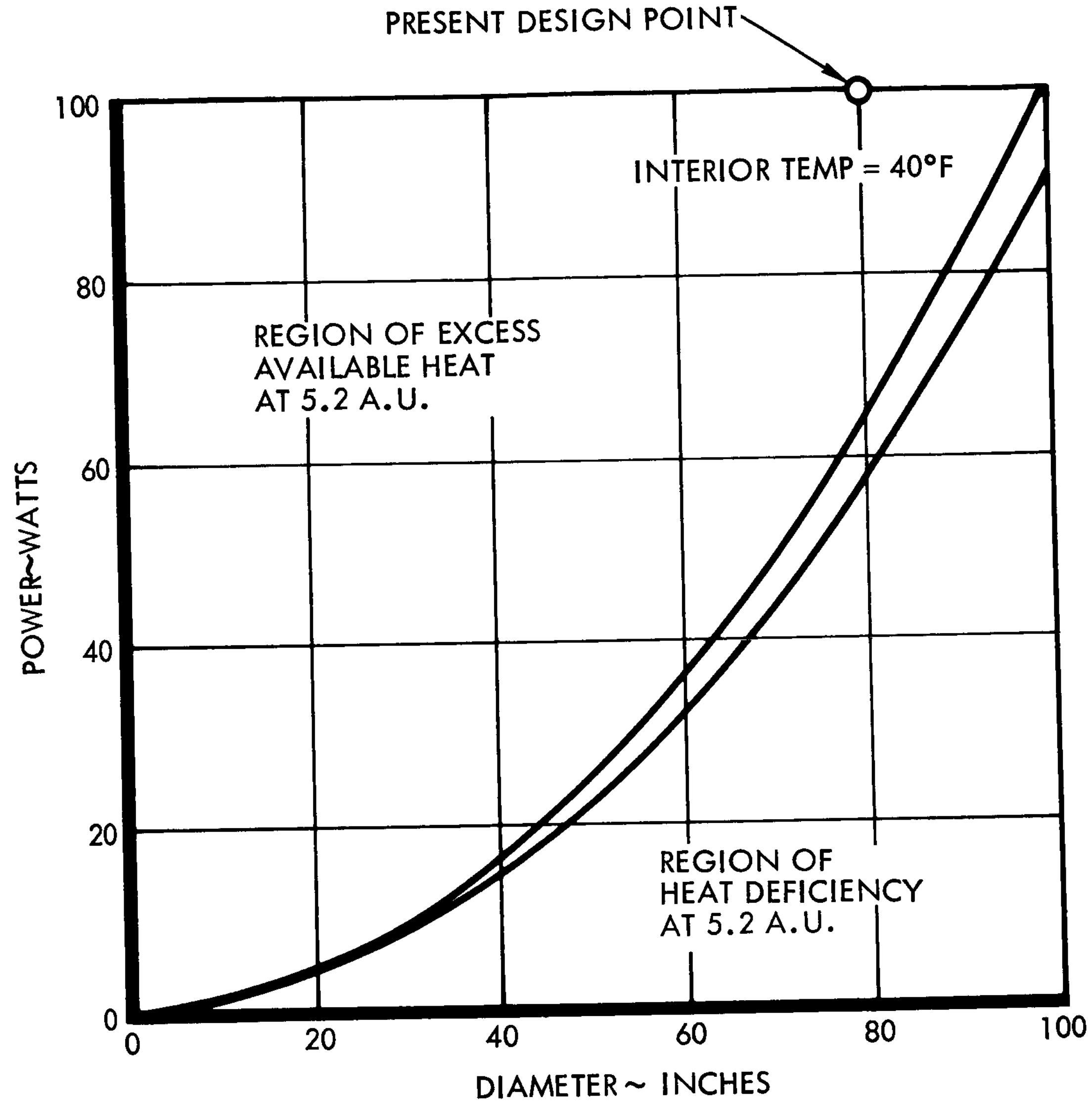


Figure 3-43. Section View of a Jupiter Flyby Vehicle, Showing Temperature Control Components



NOTE: REFERENCE DIMENSION IS EQUIP'T COMP'T DIA.

Figure 3-44. Estimated Thermal Power Requirements for Jupiter Flyby Vehicles

3.5.2 Soft Landers (Spacecraft Concept)

The temperatures of a battery in a Mars surface capsule similar to that outlined in Section 2. of this report were analyzed in greater detail using a computer model in conjunction with the TRW thermal analyzer program, which is processed on the IBM 7094 digital computer. The equations defining a thermal dynamic system are converted to an analogous rc network which reflects the lumped parameter, finite difference approximations that must be made in order to arrive at a practical method of solution. Although the basic program is fixed in form and general in nature, the specification of given networks was unique for this case. An example of the thermal network employed in the analysis of the Mars soft lander is shown in Appendix C.

Figure 3-45 shows that even in a vehicle with low average power, a battery heater of 7.5-watt capacity is capable of maintaining satisfactory battery temperatures under cyclic ambient conditions such as would be encountered on Mars. Of particular interest is the relationship of the battery and platform temperatures for a 7.5-watt heater. It can be seen that the average of the times at these two temperatures would be nearly constant. It is also interesting to note the performance of the battery with no heater and with excessive heat. With the addition of a controller in conjunction with the heater to act as a thermal switch, tying the battery to the platform when the battery temperature exceeds a preset minimum temperature (in this case, 6° C), the battery temperature can be maintained above that of the platform and a heated battery without a controller by a significant amount. It was concluded from this analysis that heating a battery in a soft lander spacecraft was definitely feasible and may be absolutely necessary. This model analysis was performed on a low power vehicle using the battery specifications indicated in Table 3-1. If the battery size or the spacecraft size was increased without a power level increase, heating would become imperative.

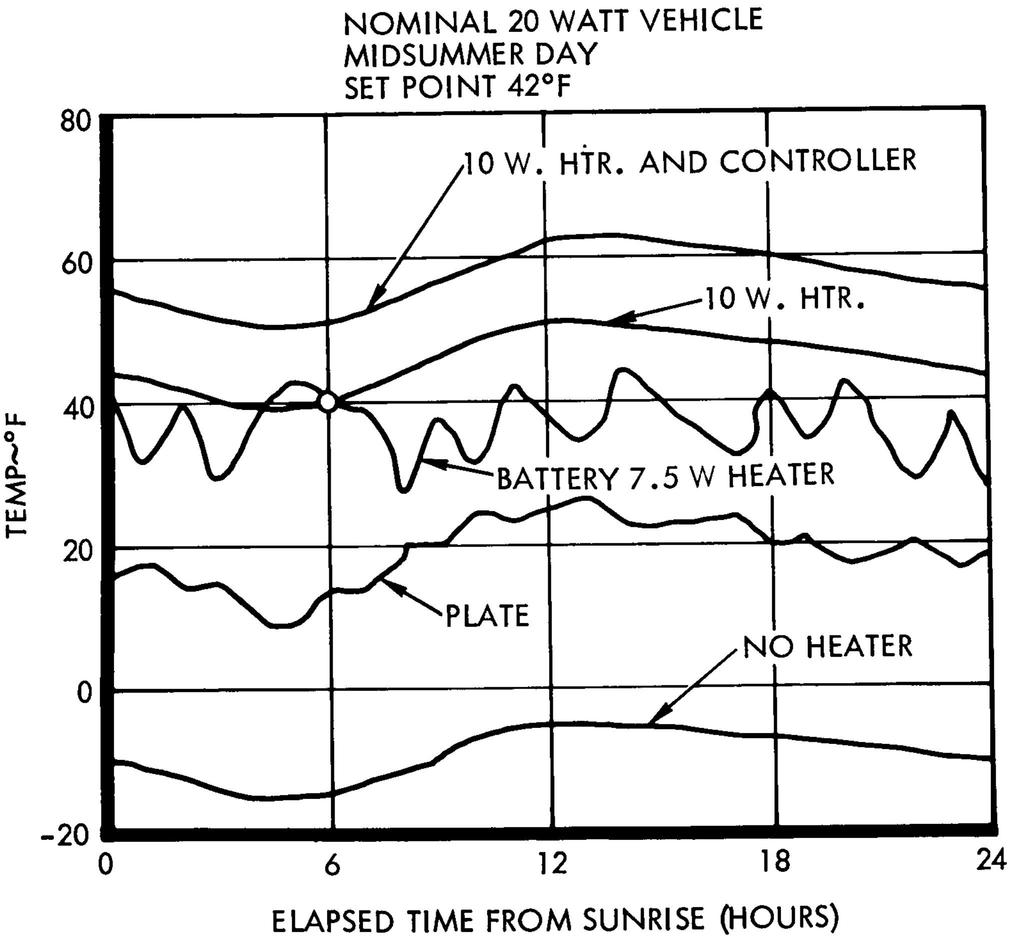


Figure 3-45. Mars Lander Model Battery Temperatures

3.5.3 Hard Landers (Battery Package Concept)

The analysis performed to study a single battery package concept was that of a Mars hard lander (Figure 3-46). This system was devised as a structurally stronger but thermally poorer configuration than the Mars soft lander previously considered. In fact, this configuration might be one which would be similar to a standby battery configuration, a surveillance experiment or guidance beacon configuration. It was assumed that fragile louvers had to be replaced by a bellows system for thermal control and that the interior of the capsule could not be as thoroughly insulated from the external environment as the previous model due to structural requirements. As a further restriction, it was assumed that the battery heater must be capable of maintaining thermal survival of the battery without depending on heat from any other source. The design, shown in Figure 3-41, provides two bellows—one to control battery temperature and the other to control overheating the hot plate. In addition, a heat accumulator between the battery and heater provides a further damping action to restrict battery temperature fluctuation. Both the bellows and the accumulator employ substances undergoing phase changes within the temperature ranges encountered. The thermal network employed as the model for this analysis is shown in Appendix C. The temperatures which this system maintained using a 30-watt radioisotope heater are shown in Figure 3-47. Battery temperatures are held between 34 and 52° F while the ambient temperature fluctuates between -100 and +77° F. Heater temperatures were kept at moderate levels. From this analysis it was demonstrated that using a constant heat source in a hard lander, which is essentially uninsulated in a cyclical environment, is feasible. It should also be noted that no other method of providing for the maintenance of operating and surviving temperatures could be found for this application. Without proceeding into more detailed analysis, but by comparing these results with those specified in paragraph 3.1, wherein an analysis was made of the amount of heat required to keep the battery warm in extreme low temperatures, it was concluded that this system could be extended with some refinements to environments as low as -300° F.

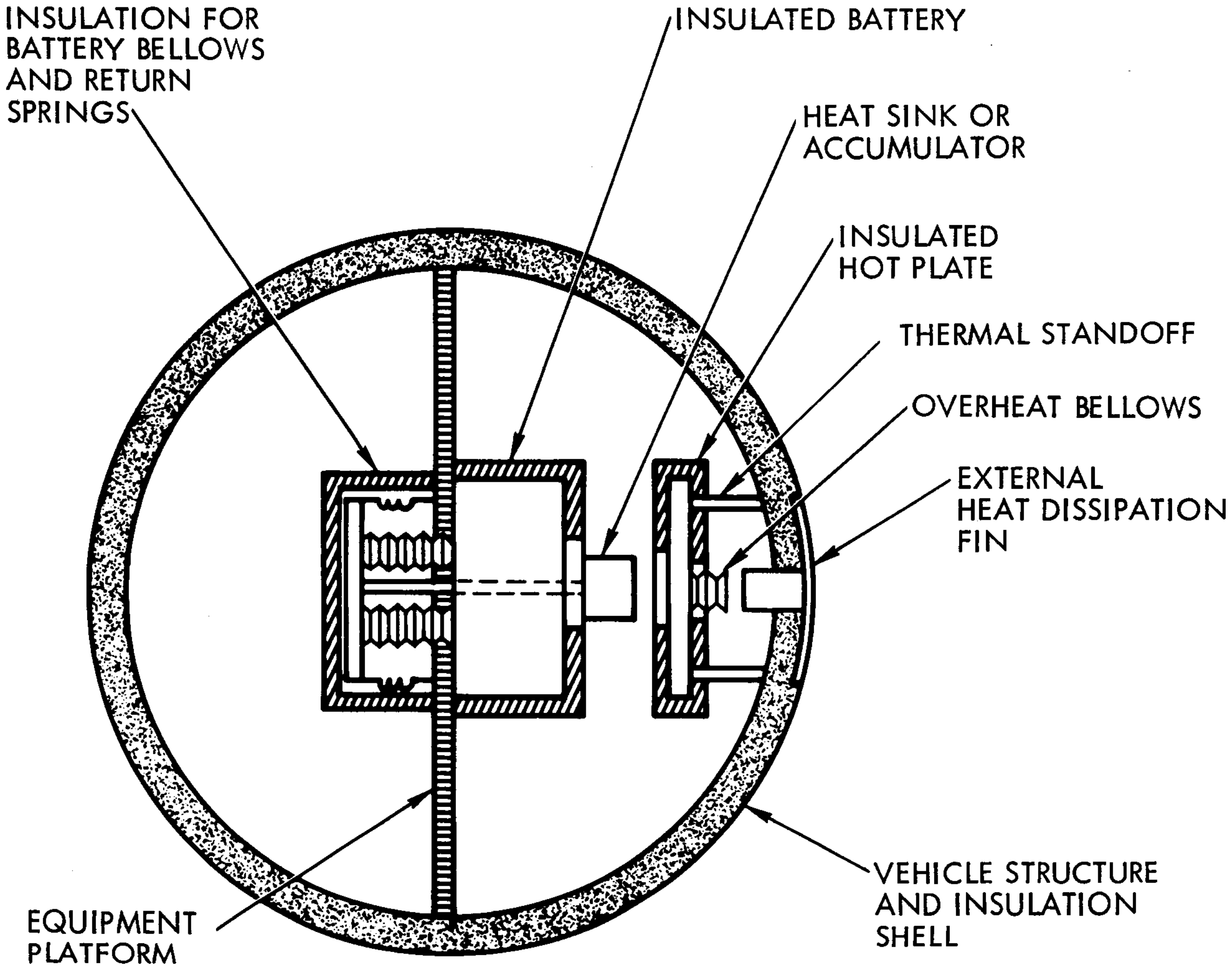


Figure 3-46. Temperature Control Features Components of a Mars Hard Lander

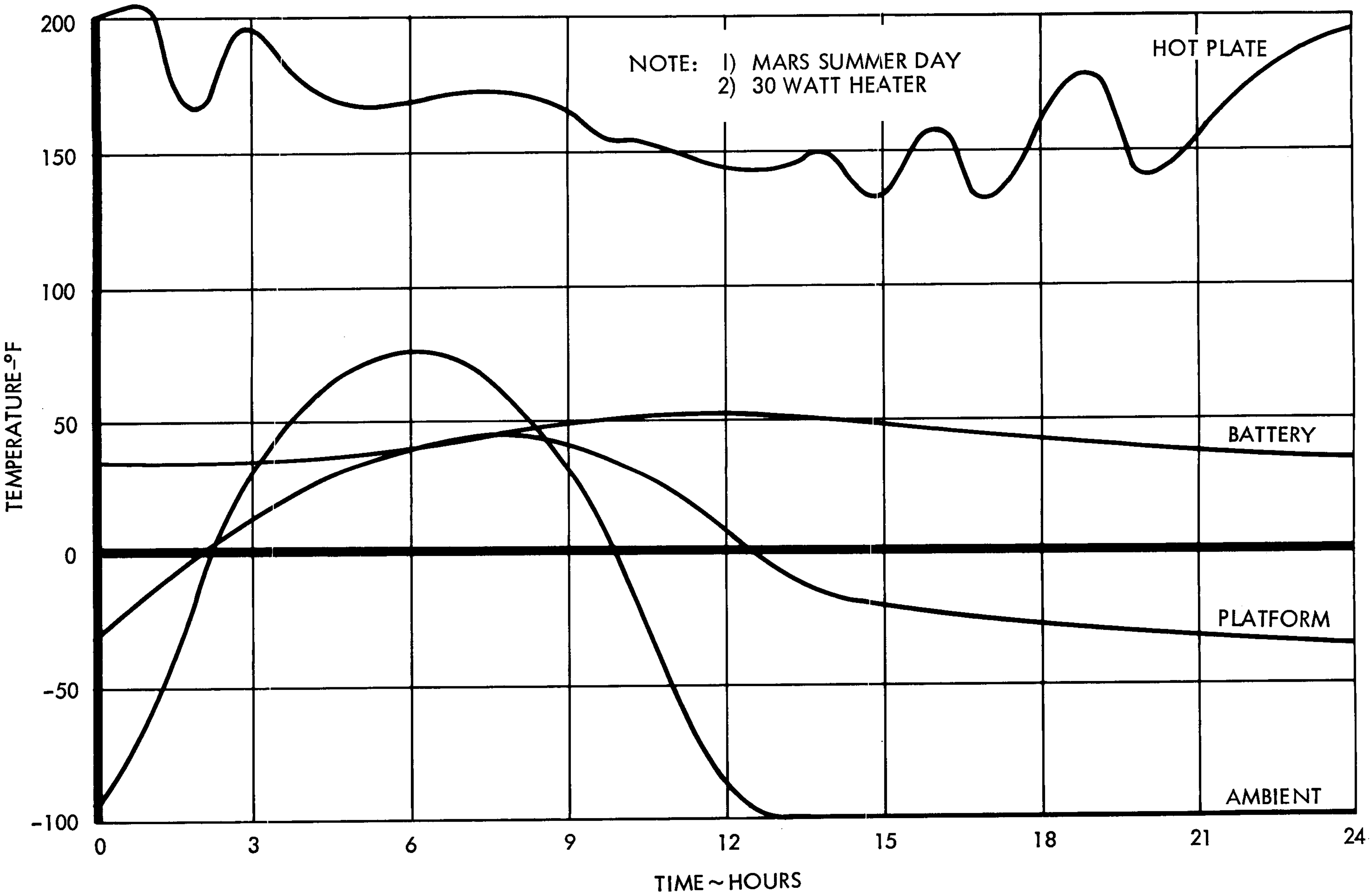


Figure 3-47. Component Temperatures of Mars Hard Lander as a Function of Elapsed Time from Sunrise

3.6 SUMMARY OF PHASE II WORK

During the Phase II analysis, the system requirements for a low temperature battery were reviewed, defined, and condensed to the point where they could be used for the development of a low temperature battery system. Included in the system requirements analysis were analyses of battery requirements per se, system thermal requirements, dynamic environmental requirements, power requirements for various types of systems and missions, availability of power for heat, and activation and startup. Once the system requirements were established, detailed analyses were then conducted on heat sources. The heat sources which appeared to be good for low temperature battery missions were isolated and defined: these were radioisotopes; electrical heaters using primary power system energy; and, possibly, chemical cartridge heaters for short interval warmup periods. Limits of technology were also defined.

Thermal control analyses were conducted in some detail because the technology has not been extensively developed, other than insulation, for this particular type of application. Several new systems were defined and disclosed as new technology.

Once the above analyses were completed, heated battery systems were selected for detailed analyses in empirical and computer models. Selection considered mission tradeoffs and mission requirements; battery specifications; heater specifications; thermal control specifications; and, finally, preliminary design configurations. The selection of a heated battery system concept was successfully accomplished.

Once the system concept had been selected, model analyses were conducted using the basic concept in a simulated deep space probe model, a simulated hard lander (battery package concept) model, and a soft lander (spacecraft concept) model. The results of the analyses with the models in advanced reiterative computer processes demonstrated not only the performance of the models within the limits defined but also the feasibility of operating a heated battery system at very low temperatures, in configurations which are also compatible with the normal anticipated warmer environments which would be encountered in the near-earth part of the mission or during warm portions of cyclical temperature environments.

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4. DEVELOPMENT PROGRAM DEFINITION

The development program required to provide the technology necessary to design a battery for very low temperature space environments (-73 to -143° C) is outlined below. The outline was developed based on the findings of the literature search and technical analyses detailed in this report. The development program is premised on:

a) Preliminary design concepts developed in the study.

b) Anticipated pertinent space milestones derived from official NASA reports through 1966 and estimated through 1975 by members of the aerospace industry including TRW, articles published in journals and periodicals, testimony by NASA officials to congressional officials, and conclusions based on NASA published documents. (The information presented is considered the best available estimate as of March 1965.)

c) Projected development milestones required to meet the above space milestones.

d) Areas of missing technology revealed by the study.

4.1 PRELIMINARY DESIGN CONCEPTS

The preliminary design concepts established by the study, which were developed to meet future requirements for heated batteries operating in low temperature space environments, are as follows:

a) A low temperature battery system design will be required which consists of a high thermal conductivity cell assembly to optimize cell balance, a heater assembly which can be readily assembled to a battery and loaded with variable amounts of heat energy sources, a thermal control switch which can control the rate of heat delivered to the battery and the rate of heat loss from the battery in cold and hot environments, respectively, and an insulated battery system package which minimizes undesired heat losses to the battery surroundings and through the battery mounting interface. Figure 3-44 shows the system concept pictorially.

b) Two concepts of battery operation are envisioned:

   1) One method is to store standard activated or unactivated batteries in an extremely low temperature environment in the nonoperating condition, reheat them to operating temperature, and use them for short term, end-of-mission operation.

   2) The second method is to maintain the battery continuously on the line in a normal operating condition (the most typical power system method of utilization).

c) Use of radioisotope heaters in low temperature battery systems for long duration missions and continuous in-service requirements.

d) Control of the rate of heat delivery from an external heat source to the battery and the heat loss from the battery through its mounting interface using an active thermal controller which is compatible with any of the following heat sources:

   — Radioisotopes heaters
   — Chemical cartridge heaters
   — Electrical dissipative heaters

e) The battery environmental design should be such that the environmental interface would be at the system level. Therefore, the battery system would consist of all of its components within one package isolated from its environment through the thermal controller and standoffs.

f) The battery system should be capable of operation with minimum change in steady-state or decreasing temperature environments to -185° C; the same system design should be capable of operating in fluctuating temperature environments from +38 to -185° C or lower without damaging the battery and with provision for maintaining the battery in its normal operating temperature range of 4 to 38° C at all times.

g) The battery cell assembly should consist of standard, sealed, secondary space batteries (in rare circumstances reserve batteries may be considered) which are assembled into an integrated cell assembly which has a uniform thermal conductivity of approximately 25 Btu/hr-ft^2-$^\circ$ F/ft as a minimum. It is desirable to keep the temperature gradient across the battery cells and from the top to the bottom within -6° C to avoid damaging the battery during charging operations.

The above design concepts establish the general system design which has been developed by this study. Several of the above concepts require development of new technology before they can be implemented into a low temperature battery design.

4.2 ESTIMATED PERTINENT SPACE PROGRAM MILESTONES AND PROJECTED DEVELOPMENT PROGRAM MILESTONES SCHEDULE

This section establishes an estimated schedule of pertinent space mission program milestones toward which a low temperature battery design program should be directed. It is based on data from numerous information sources including members of the aerospace industry, articles published in journals and periodicals, testimony of NASA officials to congressional committees, and NASA published documents. Table 4-1 is a list of the missions which may require availability of low temperature battery technology. The table shows the estimated launch dates and an estimated technology need-date in order to proceed with design.

From the above space mission schedule, the following projected low temperature battery development program milestone schedule has been projected. The milestones are presented in terms of a work statement summary for each milestone and the sequencing of the milestones is shown in Table 4-2.

a) Low Temperature Battery Storage. Determine, by testing, the effects of low temperatures to -185° C on standard battery designs. Activated secondary, reserve, and the unactivated reserve batteries should be investigated. Objectives of the investigation should be life degradation, catastrophic failure modes caused by freezing, warmup problems, optimum methods of storage, and improvements required in cell design or packaging.

b) Low Temperature Storable Battery Development. Based on results of the above study, develop batteries that can be stored for extended periods at low temperatures to -185° C. Part of the development should include optimization of warmup heating methods for the various battery designs.

Table 4-1. Mission Milestones for Potential Low Temperature Battery Applications

	Technology Need Date	Mission Date
Planetary Missions		
Lunar Lander	1966- 1968	1967- 1970
Lunar Beacon and Lander Experiments	1967	1969- 1971
Mars Orbiter	1966	1969
Mars Lander	1971+	1975+
Small Space Laboratory	1967	1970
Large Space Station	1970	1975
Probes		
Pioneer, 1.2 au	—	1965
Galactic Probe	1968	1971
Advanced Pioneer Mission	1967	1970
Near-Asteroid Probe	1969	1972
Deep-Space Planetary Probe	1970	1973
Far-Asteroid Probe	1971	1974
Jupiter Probe	1972	1975
Jupiter Flyby	1975	1979
Saturn Flyby	1979	1983
Comets	1971	1974
Mercury	1969	1972

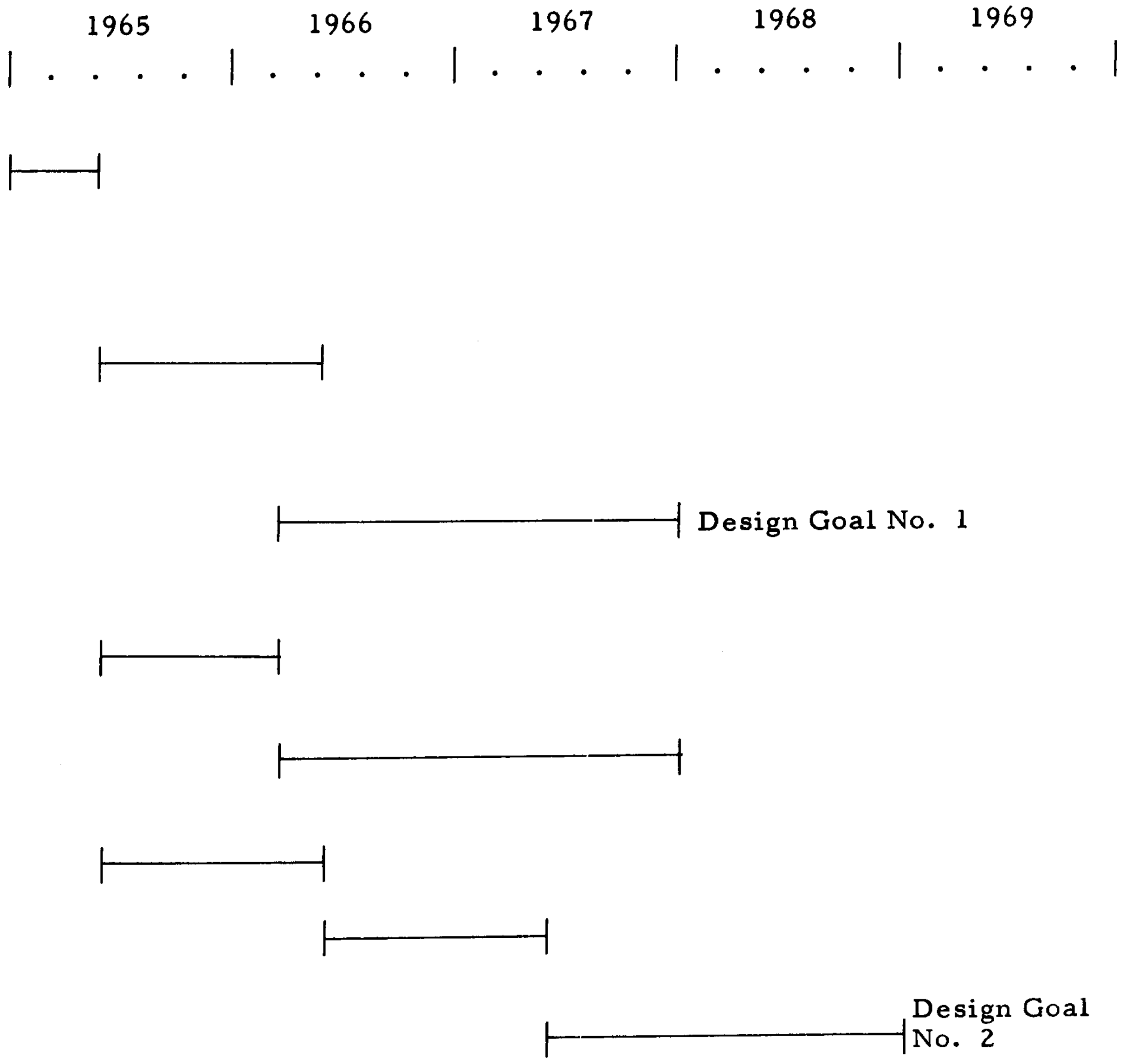
Target Date for Low Temperature Battery Prototype:

(1) Storage battery with warmup heater - 1967

(2) Continuous operation design with constant heat source - 1968

Table 4-2. Development Program Milestones, Low Temperature Battery for Space Applications

Reference Target Date for Prototype



(1) Study to develop a low temperature battery for space probes

(2) Laboratory study of technology required to store batteries at low temperatures and operate them after warmup

(3) Low temperature storage battery development; heater development for warmup; fabricate and test

(4) Constant heat source evaluation studies (radioisotope heater experiments)

(5) Low temperature battery constant heat source development

(6) Thermal controller development for high rate heat transfer changes

(7) Material compatibility study for battery packaging

(8) Build and test prototype battery for continuous operation

c) Low Temperature Constant Heat Source Studies. Develop the technology required to establish the radiation attenuation characteristics of various battery configurations and types of batteries:

   1) Test the radiation characteristics of Pm-147 and Pu-238 fuel capsules containing sources from 5 to 100 thermal watts.

   2) Evaluate radiation fields surrounding a typical radioisotope heated battery configuration for compatibility with scientific experiments carried aboard the spacecraft.

   3) Evaluate compatibility of various heated battery configurations in spacecraft mockups and determine radiation levels and patterns.

   4) Optimize geometry for battery design.

d) Low Temperature Constant Heat Source Development. Develop a constant heat source battery heater and test the unit for safety and heated battery system compatibility.

e) Thermal Controller Technology Development. Develop a thermal controller capable of isolating a heated battery system from the spacecraft structure at low temperatures and capable of conducting the heat from a heat source and from battery internal dissipation up to rates of approximately 0.4 watt/in^2 battery baseplate area during warm periods of operation.

   1) Study experimental controller configurations based on the analyses specified in the study.

   2) Study controller working fluids for compatibility with the desired working medium performance.

   3) Investigate various bellows design configurations.

   4) Build and test optimized controller configurations for compatibility with battery operation in steady state and cycling environments as low as -185°C Determine heat requirements experimentally to maintain acceptable battery operation.

f) Material Compatibility Study. Build several low temperature battery packages and study the structural compatibility problems of material interfaces, battery mounts, temperature cycling heat losses, etc., in environments to -185° C. Optimize material selections to provide necessary design information for actual flight hardware development.

g) Prototype Design and Development. Design, fabricate and test at least two low temperature battery systems. One system should contain a constant heat source for extended periods of operation and one system should be developed for low temperature storage and end-of-mission warmup. The units should be tested for electrical, thermal, and dynamic performance and life.

4.3 NEW TECHNOLOGY AREAS RECOMMENDED FOR INVESTIGATION

This study has shown the need for development of new technology required to support low temperature battery design for future missions as follows:

a) To store standard batteries at extremely low temperatures and reheat them to operating temperatures without damage from freezing. Determine the present limits of technology and establish the necessary information required to demonstrate failure modes and degradation versus low temperature and duration of storage at low temperature.

b) To install radioisotope heaters in low temperature battery systems for long duration missions. The major new technology requirements include good radiation field measurements for batteries used as shielding devices and in conjunction with spacecraft systems and components, effects of radiation on sensitive instrumentation, and a design for safe launch and re-entry which can be used in the battery system concept described above.

c) To design a thermal controller switch device which can actively control the heat distribution from an active heat source to an isolated black box mounted in or on a thermally conductive structure. The technology for accomplishing this function in space has not been successfully developed to the level required for the application.

d) To develop the materials technology required for mounting the battery to a spacecraft structural system which is flexing under the stress of fluctuating temperature environments from +38 to -185° C. The technology required to fully evaluate this problem was not available for this study and would be required for either a heated battery or an electrochemical low temperature battery.

e) To evaluate the feasibility of warming batteries from cold to normal operating temperatures with cartridge heaters. This study could find very little existing technology below -52° C and startup technology at temperatures as low as -185° C is nonexistent.

f) To design a fully-integrated battery system as discussed earlier in this report. The technology for accomplishing higher thermal conductivity within the cell pack should be extended; once the technology required to design heater assemblies has been established, the technology for integrating the heater into the system must be evaluated further; once a thermal control switch has been developed which can control rate of heat to the battery and rate of heat lost from the battery, it should be integrated with a heater and battery to determine the extent of its capabilities in low and high temperature environments anticipated in space probe missions; and the technology for minimizing heat loss through battery standoff mounts which must have structural capability should be investigated further to develop adequate mounting methods in the interface between the battery system and the spacecraft.

This concludes the statement of new technology recommended for investigation. If stated problems can be investigated within the development milestone schedule outlined, it is reasonable to expect that a low temperature battery is feasible and can be used for future space missions.

5. NEW TECHNOLOGY

New technology developed under this contract was limited to theoretical disclosures and conceptual ideas, there being no testing authorized for reduction to practice.

5.1 BATTERY TEMPERATURE CONTROLLER

The battery temperature controller is shown pictorially in Figure 5-1. It is designed to maintain battery temperatures at specific moderate levels in ambient environments that range in temperature from close to the battery upper limits on the high end to extremely cold levels on the low end. In particular, the device is intended for use under the hard vacuum, zero gravity conditions experienced by spacecraft, although it can also function in the presence of atmosphere or gravity.

By suitable arrangement of the heater and temperature control mechanism, the unit (of which the controller is an element) may be used not only to heat the battery but also to eliminate localized cold regions in the surrounding structure.

The phenomenon employed to operate the controller for the expansion of a material undergoing change from the solid to the liquid phase. The advantages of the device are as follows:

a) The temperature controller is completely self-contained; no remote actuator or external source of power required for operation.

b) The controller can function either in a simple on-off mode or as a proportional controller.

c) The unit (battery, heater, and temperature controller) can be located in unattended, remote, or relatively inaccessible locations for long periods of time.

d) The controller is simple and rugged in design, employing well-known materials and fabrication techniques.

e) The device can be used on other equipment that possess relatively large, flat surfaces.

With reference to Figure 5-1, the controller operates much like a common vise, one jaw of which is the battery, the other is the heater. The bellows are filled with a material such as benzene, which is a solid at temperatures below 6°C. At higher temperatures, benzene melts to form a liquid occupying 13.3 percent greater volume than in the solid state. Using either suitable bellows design, or spring restraints, the bellows are thus contracted when cold, and the battery is pressed against the hot

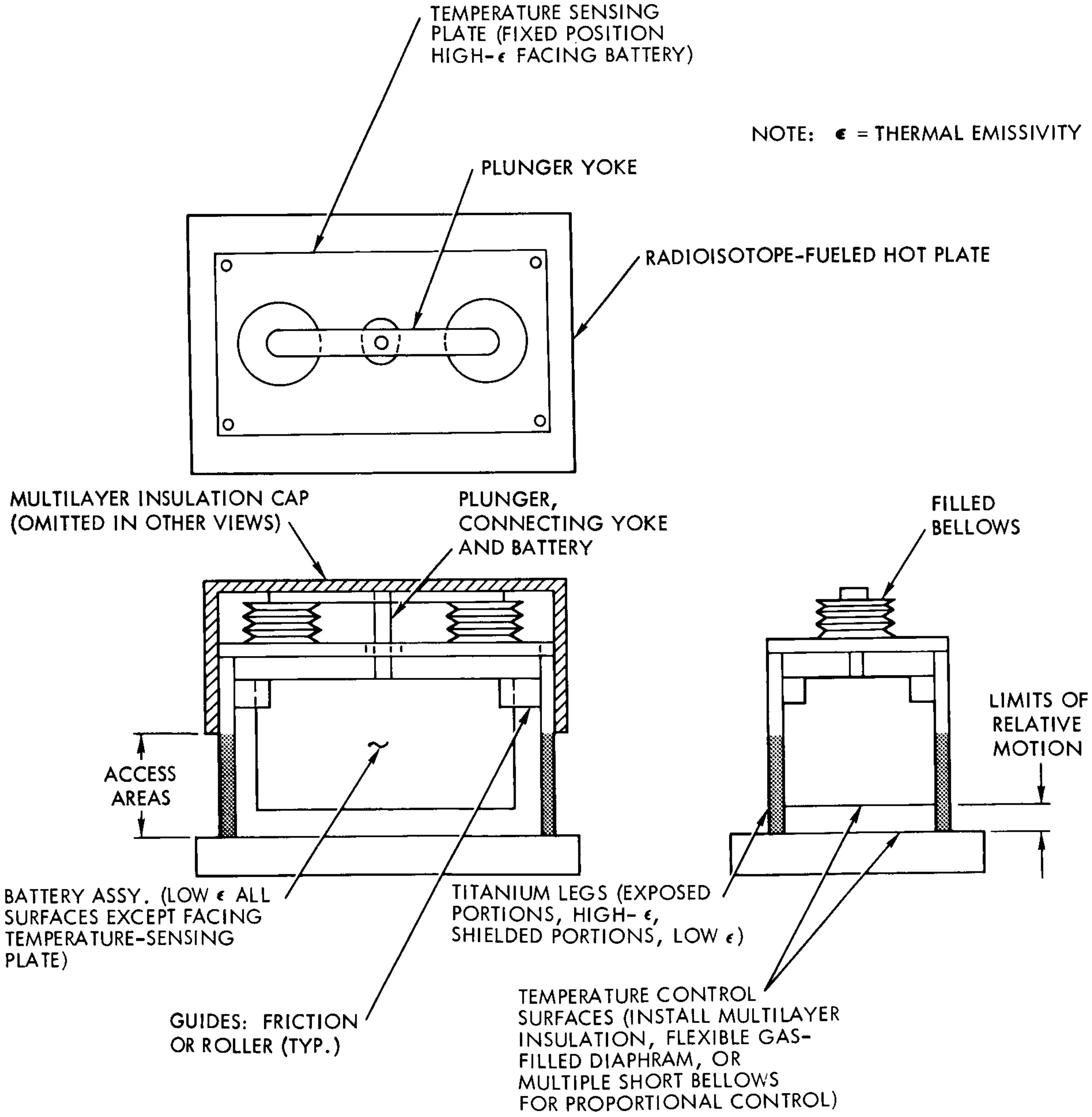


Figure 5-1. Bellows-Actuated Battery Temperature Controller

plate by means of the plunger and yoke. When heated, the bellows expand and the battery moves away from the heater.

By placing either a flexible, gas-filled diaphragm, a batt of multilayer insulation, or several small bellows of various extended lengths between the battery and hot plate, a proportioning effect may be obtained; thus, the rate of battery heating is in direct proportion to demand. If no filler is used, highly damped on-off operation is obtained.

Undesirable heat transfer paths are minimized by extensive use of insulation, low emissivity surfaces, and materials of low thermal conductivity.

The configuration of Figure 5-1 shows a relatively large hot plate. In this instance, the hot plate would be mounted directly to the spacecraft structure, thus heating the structure as well as the battery.

The novel features of this device are as follows:

a) The bellows not only actuate, but power the temperature control system.

b) The heater need not be turned off at high battery temperatures.

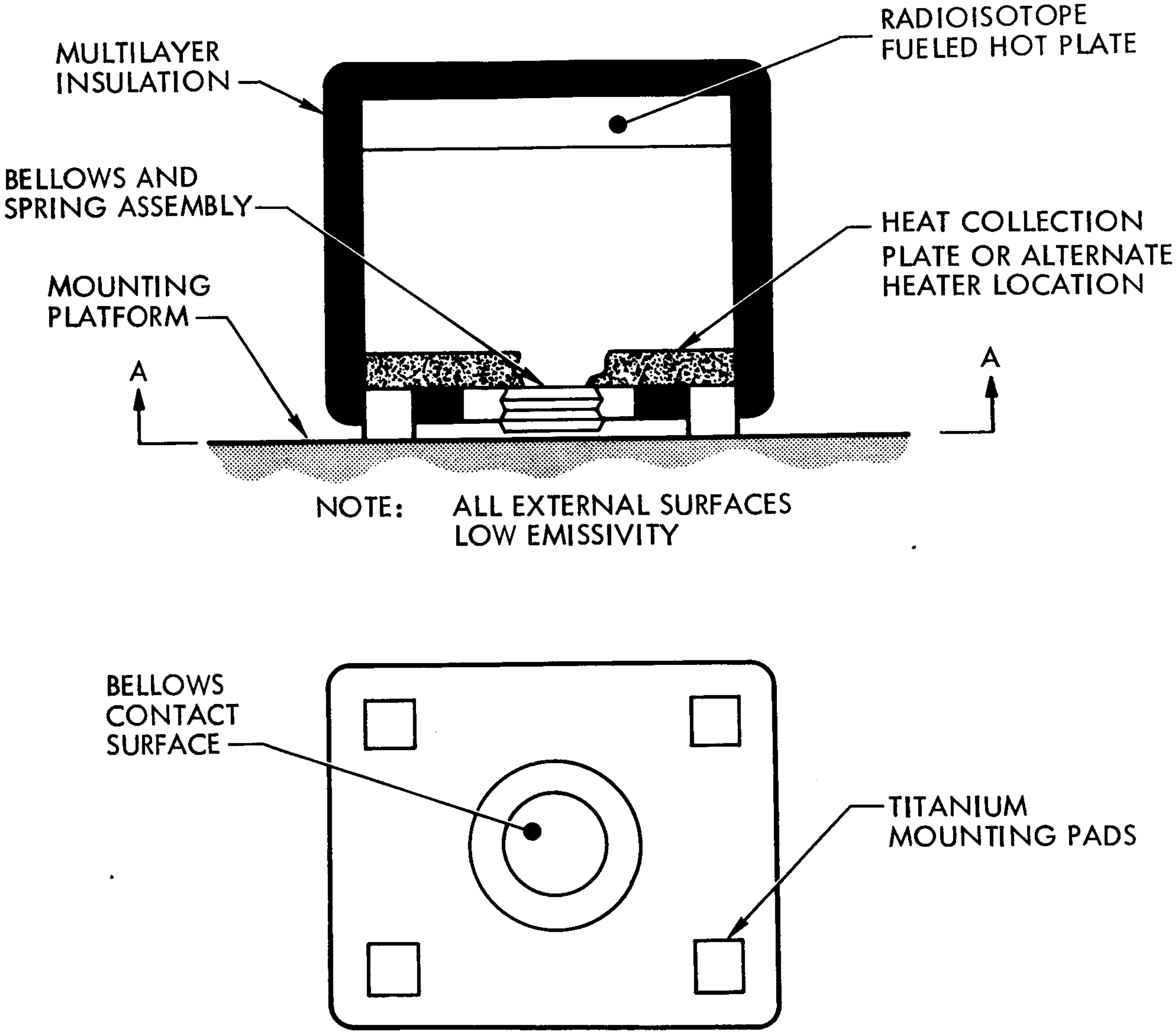
c) The inherent mode of heating is in proportion to demand.

d) Proper configuration allows heating of several batteries, or other equipment, at differing temperatures by a single hot plate at a specific design temperature.

e) The control system will operate either in space or on Earth.

5.2 TEMPERATURE CONTROL DEVICE

Figure 5-2 shows the conceptual model of a device designed to maintain batteries or other equipment temperatures at specific moderate levels in ambient environments that range in temperature from close to the equipment upper limits on the high end to extremely cold limits on the low end. In particular, this device is intended for use in environments experienced by a spacecraft, although it can also function in planetary or terrestrial environments. In general, it is intended for use with heaters that operate at a constant but otherwise uncontrollable power level, such as a radioisotope-fueled hot plate.



NOTE: ALL EXTERNAL SURFACES LOW EMISSIVITY

VIEW A-A

Figure 5-2. Bellows-Actuated Temperature Control System

By arrangement of the heater equipment, thermal resistance device, and mounting surface in series, the temperature difference between battery and mounting surface is controllable if the thermal resistance is controlled. In the present instance, when the environment is cold, the resistance is high. When the environmental temperature has increased to some specified level, the movement of a filled bellows decreases the thermal resistance, thereby reducing the thermal potential required to transfer the fixed heat flux from the battery.

The general problem solved is the same as in the first disclosure. The same physical phenomenon is employed, which is the expansion of a material undergoing change from the solid to the liquid phase.

The following advantages are cited, with particular reference to the preceding Battery Temperature Controller section:

a) Relative movement between battery and hot plate is eliminated.

b) Pressure on heat transfer surfaces is obtained by expansion of material inside bellows, rather than by spring action. Higher pressures can be obtained.

c) Unwanted heat losses from the hot plate are easier to contain.

With reference to Figure 5-2, the only moving part is the assembly consisting of the bellows and return springs. The bellows are filled with a material such as benzene, which is a solid at temperatures below 6° C. At higher temperatures benzene is a liquid occupying 13.3 percent greater volume than in the solid state. The bellows assembly is thus pressed against both the battery and mounting platform when the battery temperature is above 6° C, promoting flow of heat to the platform. When the battery (and presumably the surroundings) is relatively cool, the bellows are contracted by the return springs, and the flow of heat is retarded, being constrained to pass through the titanium pads which have a low thermal conductivity.

A proportioning effect may be obtained by using several bellows, filled with materials liquifying at different temperatures. With only one bellows, simple on-off operation is obtained, although highly damped.

Undesirable heat transfer paths are minimized by extensive use of insulation, low emissivity surfaces, and materials of low thermal conductivity. A heat collection plate can be used to eliminate internal hot-spots, and to direct the flow of heat to the bellows.

Both this disclosure and the preceding disclosure were analyzed as part of the computer model studies conducted under this contract. The final results using this controller in a model are shown in Section 3.5.

APPENDIX A

TRIP REPORT SUMMARY

TRIP REPORT SUMMARY

Two industry survey trips were made under this contract for the purposes of establishing the state of the art of battery operation at low temperature and obtaining information on nuclear heater technology.

The institutions visited and a summary of the results of each visit are as follows.

NASA/GODDARD SPACE FLIGHT CENTER (2 September 1964)

NASA/GSFC was visited to investigate the latest technology in the measurement of thermal characteristics of battery cells and to determine the accuracies being achieved in this field. Thermal data accuracy is one of the restrictions in the design of a low temperature battery system.

Major discussions concerned measurement methods of the rates of heat generation in batteries and the relationship of such measurements to the thermodynamics and kinetics of the cells. After a review of the TRW Systems approach (separation into reversible heat, irreversible heat, and heat generated by side reactions), it was suggested that this approach was fundamentally sound, but that some difficulty might be encountered in predicting the state of charge of the cell to relate the instantaneous rate of heat generation to the stage of charge.

A review was made of the heat-generation-rate measurement apparatus used by GSFC at the American University. This is a constant temperature flow calorimeter designed to measure heat removed or supplied to the working cell at constant temperature by using thermopiles to measure temperature gradients in a moving stream of fluid of known mass flow rate and heat capacity. Sensitivities of 0.01 to 0.05 watt are possible, depending upon the working fluid which must be varied with the temperature range of operation.

Rough draft copies of two papers were obtained:

"Heat Effects in Nickel-Cadmium Batteries"

"Thermal Effects of Nickel-Cadmium Batteries,"
Electrochemical Calorimetry III

CATALYST RESEARCH CORPORATION (3 September 1964)

The Catalyst Research Corporation was visited to discuss their programs on pyrotechnic heating and catalytic combustion heaters.

Pyrotechnic Heat Powders Heating. These powders, usually consisting of a metal powder, an oxidant, and inert materials, are fabricated in the form of paper. Metals such as Zr, Al, B, etc., and oxidants such as $BaCrO_4$ are used to produce exothermic reaction heat. Inert materials are used to hold the molten reaction products in place for greater uniformity of heat distribution. Specific material compositions are proprietary.

By adding inert diluents of high heat capacity, or by adjusting shape, etc., it is possible to slow the rate of propagation of the exothermic reaction; however, this also results in reduced reliability of initiation and maintenance of the heat-producing reaction at low temperatures. Burning rates of pyrotechnics are variable from 0.1 to 24 inches per second. For practical purposes, pyrotechnic heaters for batteries are considered to be instantaneous sources of heat and therefore must be stored.

Pyrotechnic heating is mainly employed in fused-electrolyte batteries, although CRC has used it in at least one other major application. Efforts to heat liquid-electrolyte cells directly have not been extensive, particularly where rapid heating is required, since heater surface temperatures of 1,500° C are normal with pyrotechnic heating. CRC has devised an automatically activated battery for low temperature operation, in which the electrolyte is propelled by a gas generator through a heat exchanger into the cells. Heat is provided to the exchanger by thermal compounds and the heating rate and electrolyte flow-rate must be coordinated for adequate heating of the electrolyte without boiling. CRC measured heat capacities and heat flow rates of significant cell components as a part of this program and supplied a copy of the final report including these data.

Catalytic Combustion Heaters. CRC has studied catalytic combustion heaters and has found that the catalytic reaction is difficult to initiate and maintain (at very low environmental temperatures). A catalyst consisting of $CuO-MnO_2$ (precipitation formed) is being used in the development of a heater to maintain 10° C in a -54° C environment for one week.

No weight or volume objectives of this program were stated. Liquid or gaseous fuels and air would be used.

U.S. ARMY ELECTRONICS LABORATORIES (8 September 1964)

The USAEL was visited to discuss Army research in low temperature batteries and heating methods used in Arctic environments. This technology could offer much potential for space applications as well.

For some time, USAEL has been involved in methods of battery heating, principally for maintenance of battery performance in the Arctic. They did investigate briefly the use of radioisotope heaters for batteries (with Martin Company) but did not continue the study beyond the early conceptual stages due to apparent hazards involved in handling radioisotopes in the field. No quantitative data not already published were available. (See 13th Annual Power Sources Conference Proceedings.)

USAEL is in the process of investigating liquid-ammonia electrolyte batteries but considers them useful only as reserve batteries at the present time or in the near future. Storage in the unactivated condition is a problem at elevated temperatures due to the high pressure of ammonia. Activated storage also presents deterioration problems.

USAEL is unaware of any research on present day plate-structure batteries, in which batteries are frozen to the point at which the electrolyte becomes fully solidified. USAEL has a battery heating program at W. D. Little Corporation (reported later). Most work on heated batteries has been reduced since the development of the portable waistcoat battery. This battery is maintained at operating temperature by body heat.

ARTHUR D. LITTLE CORPORATION (9 September 1964)

The Arthur D. Little Corporation was contacted to discuss their current contract with USAEL on low temperature batteries. This survey was restricted to operation of primary batteries in the Arctic and covered the entire range of heating methods from human metabolism through fermentation, phase change, flame heating, etc., including an evaluation of radioisotope heating. This evaluation, based on Sr-90 heat sources, resulted in a weight of 700 pounds of shielding to render the battery package safe. Pm-147 was also suggested (weight 2.6 pounds). Two of the most promising techniques suggested by ADL are combustion heating (with some difficulty in maintaining flame at low temperature) and electric heating (using battery power to maintain itself at operating temperatures with resistive heaters).

ADL has made an extensive study of battery insulation and the effect of insulation and battery properties on rate of cooldown. Spent batteries were used in these investigations. An equation for calculation of battery temperatures (based upon the assumption that the rate determining process is based upon insulation effectiveness) was tested, and indicated good agreement with individual cells, but poor agreement with measured battery cooling rates.

One of the chemical heaters evaluated by ADL, which appears to be useful in some circumstances, is one in which a cartridge containing wet chemical reactants is actuated by allowing the reactants to mix. Two types of these reactions are:

1) Oxidation-Reduction Reactions--aluminum and perchlorate--organic compounds and permanganate

2) Heat-of-Hydration Reactions--$CaO + H_2O \longrightarrow Ca(OH)_2$

ADL did not know of any instance in which the modern plate structure alkaline battery had been frozen solid, but did know of an instance in which Leclanche's cells had been frozen and restored to service.

AIAA CONFERENCE (Philadelphia - 4 September 1964)

This conference was attended to obtain the latest research information on thermal characteristics of batteries and to discuss the applicability of this information.

No data were available on survival of present day plate structure batteries after freezing solid, although it was generally considered that the freezing of electrolyte would result in some deterioration of the structure of the plate electrodes.

Discussions were held on the feasibility of battery temperature prediction using theoretical calculations of heat generation rates, plus thermal conductivity calculation methods set forth in a paper presented by Drs. Shair and Preusse. It was considered that data presently available would not be accurate enough and that a combination of calorimetric and temperature-coefficient measurements would be necessary to provide an adequate estimate of the overall entropy of reaction for calculation of reversible heats. These data are not available in the physical-chemistry literature for all types of cells.

NASA LEWIS RESEARCH CENTER (10 September 1964)

NASA Lewis Research Center was visited to discuss the low temperature battery program with its sponsors. In addition, up-to-date information applicable to this contract was obtained from the NASA Lewis Center.

Specific information was made available on the low temperature ammonia battery being developed by Livingston Electronics, as well as more general information on other NASA Lewis Research Center programs.

During this visit, Mr. Unger of NASA Lewis suggested that TRW Systems consider the following missions as part of the low temperature battery study:

Mars Lander
Asteroid-Belt Probe
Jupiter Flyby
Mercury Dark-Side Lander

AEC HEADQUARTERS (Germantown, Maryland) (4 November 1965)
AIAA SECRET SESSION ON SPACE NUCLEAR POWER SYSTEMS (Philadelphia, Pennsylvania)
MOUND NATIONAL LABORATORY (Miamisburg, Ohio)

Information was obtained on Pu-238 and Pm-147 fuel forms, encapsulation technology and availability. Capsule designs were also discussed. A general program objective orientation was given at both AEC installations.

Classified information received is contained in classified Appendix D.

APPENDIX B

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APPENDIX C

ANALYTICAL METHODS AND SAMPLE CALCULATIONS

This appendix is included to present certain analytical methods and sample calculations used to derive technical data within the report. The methods and calculations provide the reader with the means to verify the work accomplished.

C.1 ANALYTICAL METHODS

The complex analyses herein required the use of the TRW Systems Thermal Analyzer Program which is processed on the high speed digital computer IBM 7094. The equations defining a thermodynamic system are converted to an analogous rc network which reflects the lumped-parameter, finite-difference approximations that must be made in order to arrive at a practical method of solution. Although the basic program is fixed in form and general in nature, the specification of any given network may be unique, as has been the case in the study.

An example of the thermal network reflected in the analysis of the Mars Hard Lander is shown in Figure C-1. The network constraints are further defined and established in Table C-1. Similar models were designed for the Mars Soft Lander and Deep Space Probe missions. The thermal response of the sample systems to a cyclical varying environment or a trend change in environment was calculated using the IBM 7094 analyzer with numerous mathematical iterations. The results of these mathematical iterations are temperature histories as displayed in Section 3 of the report text.

C.2 DERIVATION OF THERMAL HEATING REQUIREMENTS

The following sample calculations including equivalent thermal network (Figure C-2) were used to derive Figures 3-4, 3-5, 3-6, and 3-7. The calculation of radiation losses were shown to be negligible; calculation of conduction losses which were significant to this problem are reported. Two environments were considered, -20° F and -300° F, because the mission studies performed concluded that these two temperatures were applicable for a battery mounted within an operating spacecraft

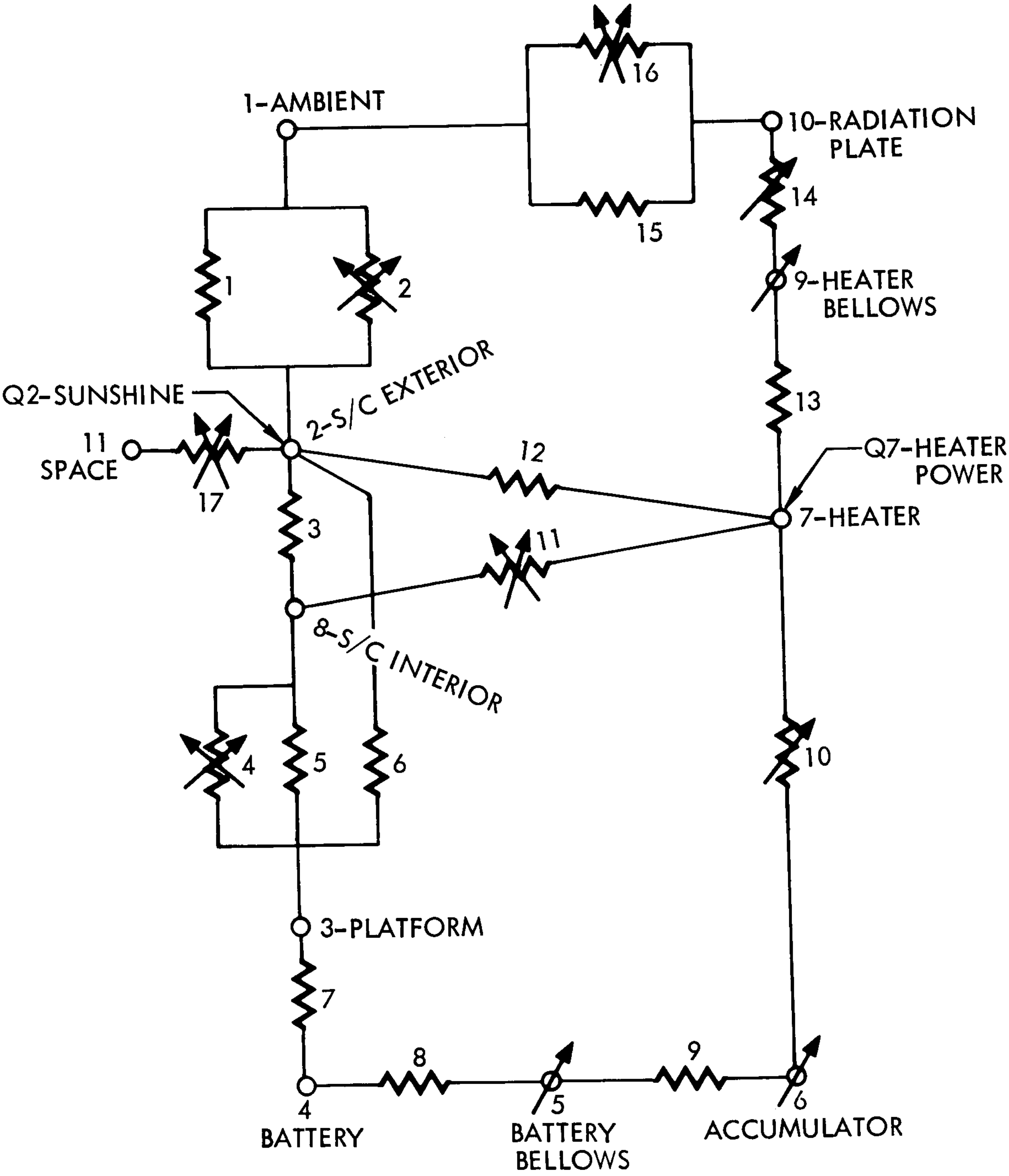
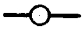
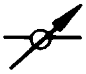


Figure C-1. Thermal Network for Mars Hard Lander

Table C-1. Network Nomenclature and Explanation

Legend	
	Thermal resistance, fixed value
	Variable thermal resistance, single variable
	Thermal radiation resistance, two variables
	Fixed thermal capacitance, finite value
	Variable thermal capacitance
	Thermal capacitance, infinite value

Resistance No.	Description
1	Convection spacecraft exterior to ambient: $R = 1/h_c A$
2	Radiation, spacecraft exterior to ambient: $$R = \frac{1}{\sigma \epsilon_i \epsilon_j A F_{ij} \left(T_i^2 + T_j^2\right)\left(T_i + T_j\right)}$$
3	Conduction, spacecraft insulation from interior to exterior: $R = \ell/KA$
4	Radiation, platform to spacecraft interior
5	Convection,. platform to spacecraft interior
6	Conduction and contact heat transfer, platform to spacecraft exterior: $R = \ell/KA + 1/CA$
7	Conduction and contact heat transfer, battery to platform
8	Conduction, battery bellows
9	Conduction, rod between battery bellows and accumulator
10	ON-OFF thermal switch controlled by battery bellows

Table C-1. Network Nomenclature and Explanation (Continued)

Resistance No.	Description
11	Radiation, spacecraft interior to heater
12	Conduction, heater mounting
13	Conduction, heater bellows
14	ON-OFF thermal switch controlled by heater bellows
15	Convection, radiation plate to ambient
16	Radiation, plate to ambient
17	Radiation, spacecraft exterior to space
	Specific Heat Fluxes
Q2	Solar energy absorbed by spacecraft exterior
Q7	Heater power

and a battery stored in a nonoperative condition outside of a vehicle respectively. Figure C-3 shows the battery configuration analyzed.

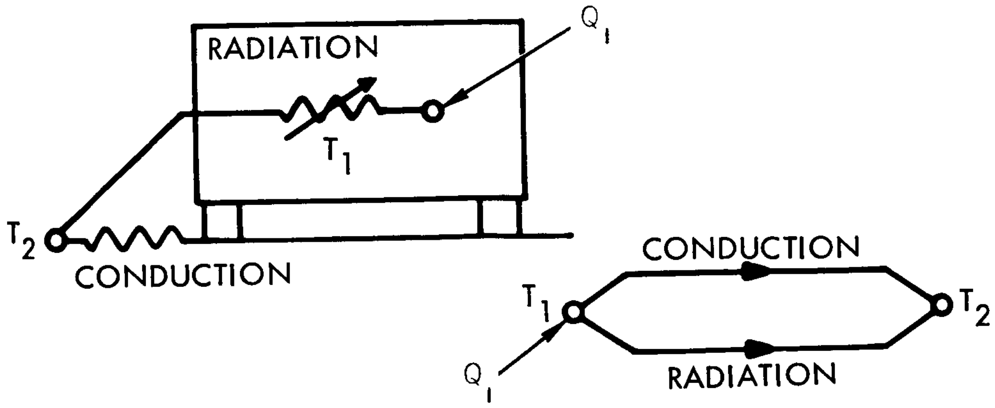


Figure C-2. Equivalent Thermal Network - No Controller

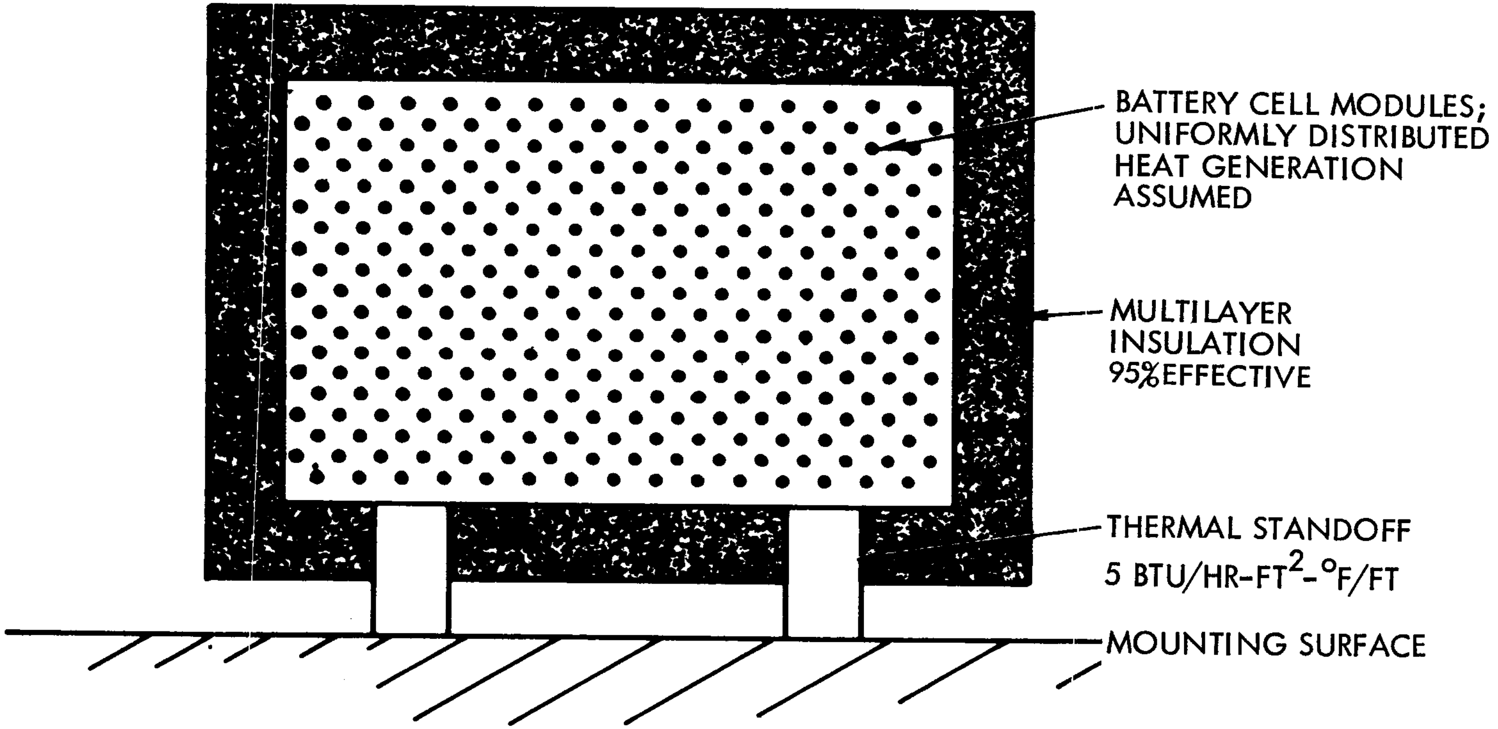


Figure C-3. Idealized Insulated Battery Configuration for Study Analysis

Definitions

Characteristic Dimension of a Cubic Battery	L
Battery Volume	$V = L^3$
Battery Base Area	$A = L^2$
Battery Total Surface Area	$A = 6L^2$
Standoff Contact and Cross Section Area	$A = 0.04L^2$ (for $L^2 \geq 25$ in ^{2})
	$= 1$ in^2 (for $L^2 \leq 25$ in ^{2})
Radiation Loss Area (5%)	$a = 0.05\ (6L^2) = 0.3L^2$
Standoff Height	$\ell = 0.75$ inch
Standoff Conductivity	$K = 5$ Btu/hr-ft^2-$^\circ$F/ft (titanium)
Standoff Contact Coefficient	$C = 50$ Btu/hr-ft^2-$^\circ$F

Steady State Heating Analysis

$$Q_1 = \sigma \epsilon\ a \left(T_1^{\ 4} - T_2^{\ 4}\right) + \frac{T_1 - T_2}{(1/4CA) + (\ell/4KA)}$$

where

L = Length of one battery side, ft

Q_1 = Heater input, Btu/hr or watts

T_1 = Battery temperature, $^\circ$F (40° F from Section 2)

T_2 = Ambient temperature, $^\circ$F (temperature of battery mounting platform)

A = Cross section or contact area of one standoff, ft^2

C = Contact coefficient, Btu/hr-ft^2-$^\circ$F

K = Thermal conductivity, Btu/hr-ft^2-$^\circ$F/ft

$\sigma = 0.1714 \times 10^{-8}$ Btu/hr-ft^2-$^\circ$R^4 (Stefan-Boltzmann Constant)

$\epsilon = 0.85$ (emissivity)

ℓ = Standoff height, ft

a = Radiation loss area (insulation efficiency), ft^2

R = Thermal resistance

Changing the equation to a form which will operate with dimensions in inches,

$$\frac{1}{4CA} = 72/L^2 \text{ for } L^2 \geq 25$$

$$= 2.88 \quad \text{for } L^2 \leq 25$$

$$\frac{\ell}{4KA} = 45/L^2 \text{ for } L^2 \geq 25$$

$$= 1.80 \quad \text{for } L^2 \leq 25$$

$$\frac{1}{4CA} + \frac{\ell}{4KA} = R = 117/L^2 \text{ for } L^2 \geq 25$$

$$= 4.68 \quad \text{for } L^2 \leq 25$$

<u>Assume -300°F ambient</u>

$$q_{rad} = \sigma\epsilon\left(\frac{0.3L^2}{144}\right)\left(T_1{}^4 - T_2{}^4\right) = \text{radiation loss}$$

$$= 0.188L^2 \text{ Btu/hr}$$

$$q_{cond} = (T_1 - T_2)/R = \frac{340}{R} = \text{conduction loss}$$

$$= 2.9\ L^2 \text{ Btu/hr for } L^2 \geq 25$$

$$= 73 \text{ Btu/hr} \quad \text{for } L^2 \leq 25$$

$$\left.\begin{aligned} Q_1 &= (0.188L^2 + 73) \text{ Btu/hr} \\ &= (0.0553L^2 + 21.4) \text{ watts} \end{aligned}\right\} L^2 \leq 25$$

$$\left.\begin{aligned} &= (0.188L^2 + 2.9L^2) \text{ Btu/hr} \\ &= .9L^2 \text{ watts} \end{aligned}\right\} L^2 \geq 25$$

<u>Assume -20° F ambient</u>

$$q_{rad} = 0.076L^2 \text{ Btu/hr}$$

$$= 0.0223L^2 \text{ watts}$$

$$q_{cond} = 0.512L^2 \text{ Btu/hr for } L^2 \geq 25$$

$$= 0.151L^2 \text{ watts for } L^2 \geq 25$$

$$= 3.76 \text{ watts for } L^2 \leq 25$$

$$\therefore Q_1 = (0.0223L^2 + 3.76) \text{ watts for } L^2 \leq 25$$

$$= (0.0223L^2 + 0.151L^2) \text{ watts for } L^2 \geq 25$$

-300° F Ambient Data Tabulation

L	$0.0553L^2$	$0.9L^2$	Q_1 (watts)
4	0.9		22.3
5	1.4		22.8
7.5		51	51
10		90	90
15		201	201

-20° F Ambient Data Tabulation

L	$0.0223L^2$	$0.1733L^2$	Q_1 (watts)
4	0.4		4.2
5	0.6		4.4
7.5		9.7	9.7
10		17.3	17.3
15		39.0	39.0

Watt-Hours

(Reference Figure 1-3)

L	AgZn	AgCd	NiCd
4	256	76.5	51
5	500	150	100
7.5	1,680	504	336
10	4,000	1,200	800
15		4,040	2,690

Warmup Calculations

Assume radiation losses negligible.

Let q = Heat Flux = Q_1

C = Thermal capacitance

T = Temperature

t = Time

R = Thermal resistance

$$q = C \frac{dT}{dt} + \frac{T - T_o}{R}$$

$$\frac{dT}{dt} = \frac{qR - T + To}{RC}$$

$$e^{-t/RC} = 1 - \frac{T - T_o}{qR}$$

Spec weight $\simeq$ 0.0625 lb/in^3 for a AgCd battery (Reference Figure 1-3, 1-hour rate at 0° F)

Capacitance $\cong$ 0.0106L^3 Btu of assuming specific heat equals 0.17 Btu/lb-° F, the specific heat of a heterogeneous body. Measured value of 0.22 for cells is reduced by battery structure and stainless steel cell cases.

Assume a 5 x 5 x 5 inch battery at -300° F.

$C = 0.0106\ (125) = 1.33$

$R = 4.68$

t = 4 hours

T = 40° F, final temperature

$$e^{-\frac{4}{4.68(1.33)}} = 1 - \frac{40 - (-300)}{q(4.68)}$$

q = 154 Btu/hr

= 45.2 watts for a 150 watt-hour AgCd battery

Results of Calculations for a 4-Hour Warmup Period, Final Temperature = 40° F

	-20° F Ambient		-300° F Ambient	
L	Q_1 (Btu/hr)	Q_1 (watts)	Q_1 (Btu/hr)	Q_1 (watts)
5	27.0	7.9	154	45.2
7.5	82	24	465	137
10	187	55	1,060	312

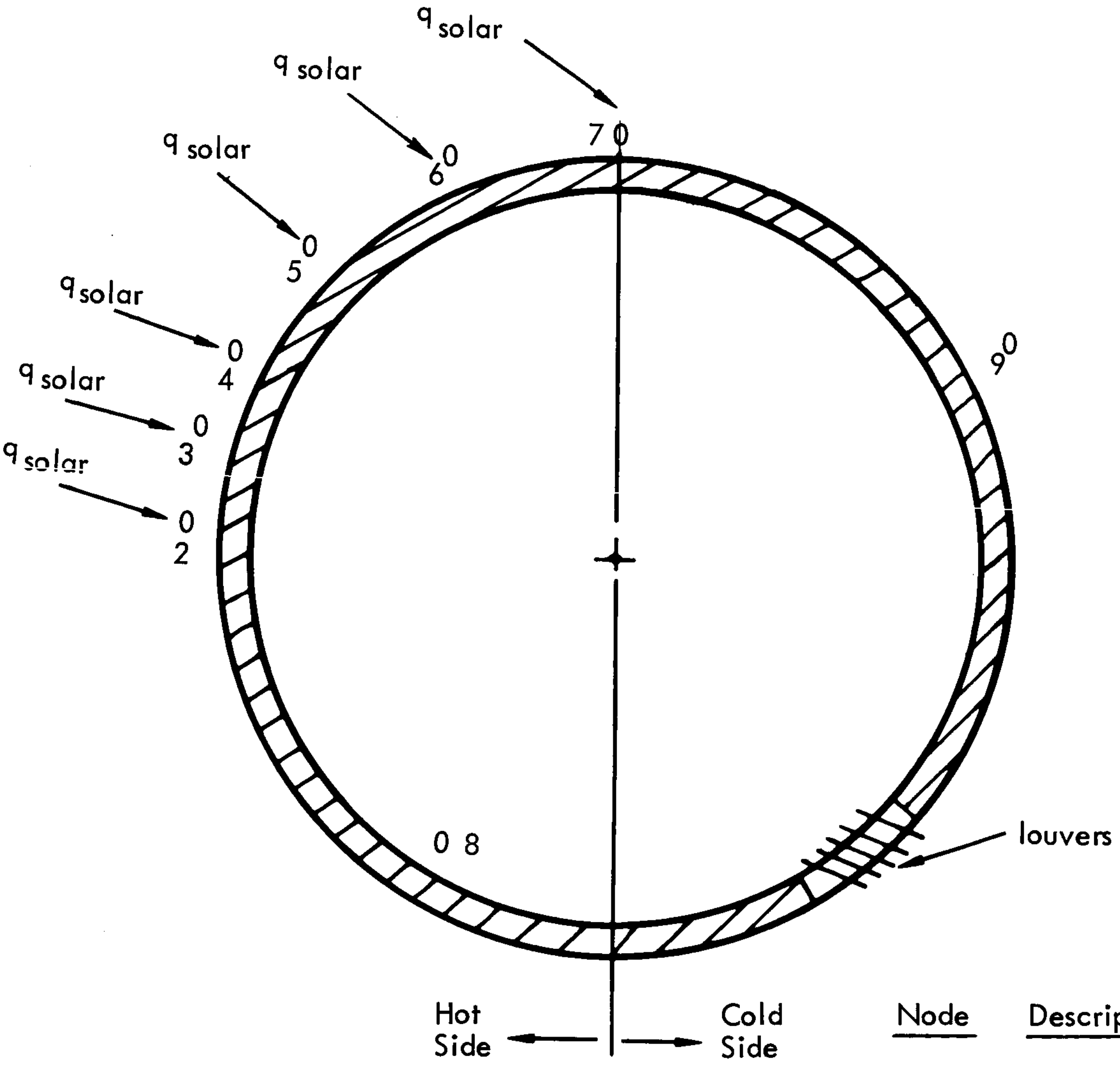
C.3 SIZE AND POWER EFFECTS ON INTERNAL TEMPERATURES OF INSULATED SPHERICAL SPACECRAFT

The following analysis generates the graphical comparison presented in Figure 3-30.

Figure C-4 shows the analytical model used to develop a computer model (submitted separately to NASA Lewis Research Center) to demonstrate the variations of thermal control with insulation, insulation and louvers, and insulation with louver control assisted by a heater. The model considered solar insulation versus AU distance, a 10-watt steady-state equipment load, various spacecraft diameters, and temperature responsive louver control. The analysis consisted of establishing an operating condition for a given model setup and then varying the conditions to obtain the necessary points for the graphs shown in Figure 3-30.

C.4 WEIGHT TRADEOFF ANALYSIS

The following analysis was used to generate a comparison of the weight advantage of a direct radioisotope heater over an electrical heater (rtg power + conversion + heater weight) versus required spacecraft electric equipment power loads. Figure 3-31 provides the power requirements to keep a spherical spacecraft above 40° F as follows:



Node	Description
1	Space Sink
2	Subsolar Point of Insulation Exterior
3 ↓ 7	Other External Insulation Points in Sunlight
8	Vehicle Internal Shell & Equipment
9	Cold Side of Insulation

Figure C-4. Spherical Spacecraft Model for Analysis of Insulation, Louvers and Heater Combinations.

Power Requirement versus Spacecraft Diameter

Distance (AU)	30 in.	Spacecraft Diameter 40 in.	50 in.	60 in.
2			8w	50w
2.5			67	126
3		47w	119	170
4	8w	105	168	200
5	68	150	195	
6	119	186		
7	167			

Using data predicted for a Pu-238, 5-year life, intact re-entry RTG (unpublished TRW data), the equivalent RTG weights for the above power requirements are as follows:

RTG Weight Corresponding to Above Spacecraft Diameter Power Level Requirements

Distance (AU)	30 in.	Spacecraft Diameter 40 in.	50 in.	60 in.
2			14 lb	36 lb
3		35 lb	42	63
4	14 lb	56	60	77
5	42	71	76	88
6	60	83	86	
7	76			

Using the above data, the RTG weight versus electric power required is as follows:

Power (watts)	RTG Weight (lb)
20	24
40	32
60	40
80	47
100	54

Using a 60-inch spherical spacecraft, the power differential requirement for each power level was computed as given:

Electric Heater Power Differential Required versus Load

Distance (AU)	Electric Load 20 w	40 w	60 w	80 w	100 w
2	30 w	10 w	--	--	--
2.5	106	86	66 w	46 w	26 w
3	150	130	110	90	70
4	180	160	140	120	100

The heater weight for the same power requirement and distance was then computed (Reference Figure 3-20) as follows:

Heater Weight Corresponding to Above Heater Power Differential

Distance (AU)	Electric Load 20 w	40 w	60 w	80 w	100 w
2	3.4 lb	2.0 lb	--	--	--
2.5	8.2	7.0	5.7 lb	4.5 lb	3.1 lb
3	11.1	9.8	8.5	7.2	5.9
4	13.0	11.7	10.4	9.1	8.9

Using the above tabulated results, the weight advantage of the radioisotope heater over the electrical heater at various spacecraft power loads (60-inch diameter spacecraft) was computed as follows:

Weight Advantage - 60-Inch Spacecraft

Distance (AU)	Electric Load 20 w	40 w	60 w	80 w	100 w
2	8.6 lb	2.0 lb	--	--	--
2.5	30.8	24.0	17.3 lb	11.5 lb	5.9 lb
3	41.9	35.2	28.5	22.8	17.1
4	51.0	44.3	37.6	31.9	25.1

These results are shown in Figure 3-10. An identical analysis process was used to derive Figure 3-9.

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