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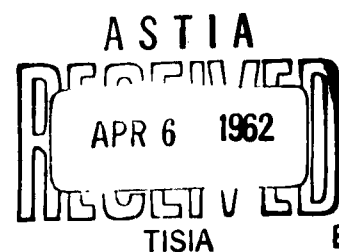


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DESIGN OF A POWER SYSTEM FOR A LUNAR MOBILE SURFACE VEHICLE

61 SPC-3

SPACE SYSTEMS

DEFENSE SYSTEMS DEPARTMENT • SANTA BARBARA, CALIFORNIA



**DESIGN OF A POWER SYSTEM FOR A LUNAR
MOBILE SURFACE VEHICLE**

R. H. Brody

**Report
61SPC-3**

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**DEFENSE SYSTEMS DEPARTMENT
GENERAL ELECTRIC COMPANY
SANTA BARBARA, CALIFORNIA**

SUMMARY

Available power generators are examined to determine which are applicable to a lunar roving vehicle. Two classes of vehicles are of interest: a 300 lb. Surveyor type vehicle; a 3000 lb. Prospector type vehicle. It is concluded that a solar cell-battery combination is best and a design is presented for each vehicle, specifying important power system parameters.

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INTRODUCTION

This report contains three sections. The first section briefly discusses some boundary conditions (lunar environment, vehicle propulsion, mission life, etc.) that a given power system must satisfy. The second section discusses the operating parameters of specific power generators. The third section presents a design for two types of mobile vehicles. One vehicle is in the 300 lb. Surveyor class and the other is in the 3000 lb. Prospector class. Only the power generator, the source itself, receives detailed attention. The power distribution and regulation system is a nasty but straightforward problem; it is considered in detail only in the last section.

Results published in GENERAL DESIGN CRITERIA FOR MOBILE INFORMATION SYSTEM VEHICLE, by M. E. Myton, will be used throughout this report.

SECTION I

SYSTEM BOUNDARY CONDITIONS

Two problems facing the system designer immediately become apparent when one considers the lunar environment. First, the environment is extremely hostile. Secondly, very little, factually, is known of it. Much information has been deduced but not measured. A JPL document, TR 34-159, (Ref. 1) describes the lunar environment which will be assumed for this report. The environmental characteristics most troublesome to the power system designer are temperature, vacuum, and micrometeorites.

Consider the problem of temperature control. The lunar surface will vary from 120°K to 400°K. The power source will generate some heat within itself and will be irradiated by direct sunlight as well as sunlight reflected from the lunar surface. Energy reflected from the moon's surface will be a function of the particular position and orientation of the vehicle at any given time. If the particular power generator chosen should be temperature sensitive, provision must be made to protect it. In the case of solar cells, for example, the designer must provide means for radiating heat to space to keep the solar cells cool. On the other hand, the generator may produce so much excess heat that the vehicle must be protected. This would occur, for example, if a nuclear reactor is utilized for prime power source.

The problem of operating in a vacuum is essentially one of obtaining suitable lubricant for moving parts. Care must be taken to use materials of low vapor pressure or excessive loss with resultant weakening will occur.

The vulnerability of particular power systems to micrometeorite damage is difficult to assess. The frequency of meteorite impact on the moon is not well known although some data is available (Ref. 2). The susceptibility of power sources to micrometeorite damage will be discussed only qualitatively in this report.

Lunar surface features that the vehicle must traverse or circumvent are of interest because they fix the propulsive power requirement. As an example, a computation was made (Ref. 3) for a 3000 lb. vehicle with a tire width of 10 inches traveling over level, loose, sandy soil. The power required for 3 mph was then 910 watts. There is an additional short time peak power requirement. It is intended that the vehicle programmer should detect emergency conditions such as moving into the shadow of a cliff where solar cells would be in-operative, or vehicle sliding slowly forward down a dangerous slope. Then sufficient stored power must be available to provide four times normal full power to the propulsion unit for a period not to exceed 10 minutes.

An interesting breakdown of future space power requirements with areas of optimum applications as a function of mission length is shown in Fig. 1 through Fig. 4, data obtained from Ref. 4. Note that these figures are concerned with space power conversion in general. Fig. 5 shows an estimate of optimum application of power conversion methods specialized for lunar surface vehicles. A strong dependence on mission length is apparent here also. All vehicles beyond Surveyor are extrapolations of course. The power sources noted on Fig. 5 refer to the primary power system only. For example, Prospector may utilize solar cells for primary power but will require batteries for energy storage and could conceivably also utilize an RTG. Several sources are not noted in Fig. 5, such as solar thermionic or thermoelectric. These systems require a degree of orientation and reflector control that is believed to be incompatible with a moving vehicle and the present state of the art. The justification for this will be given in Section II.

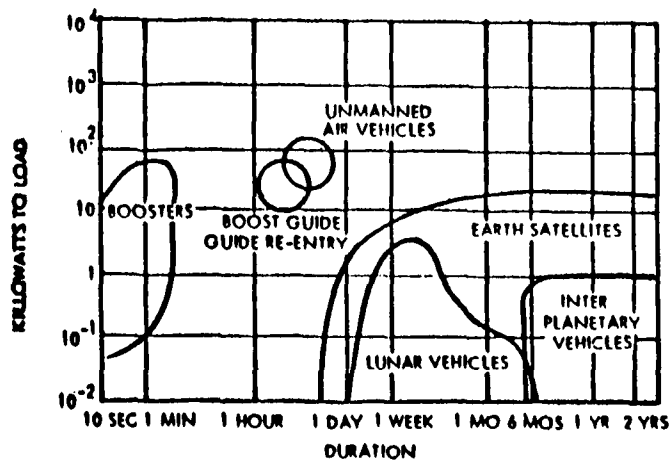


Figure 1. Synthesis of Forecasts of 1962 Requirements

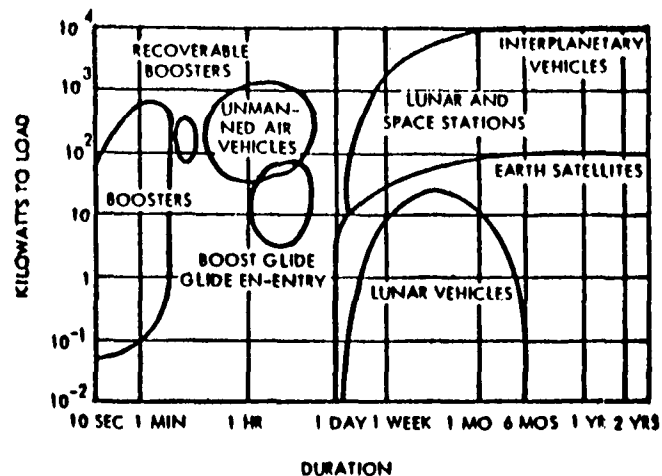


Figure 2. Synthesis of Forecasts of 1966 Requirements

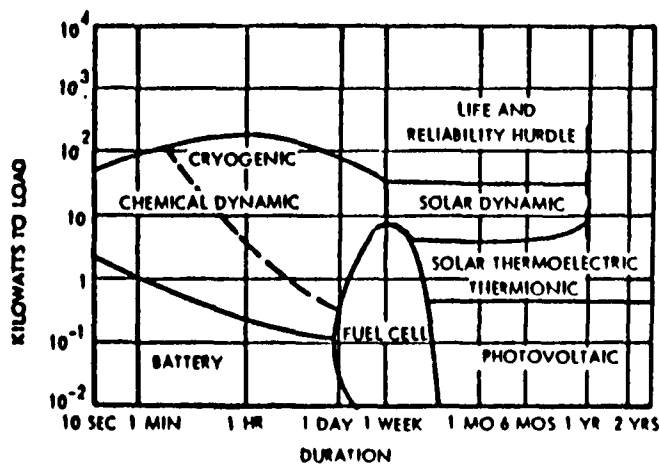


Figure 3. Forecasts of 1962 Areas of Optimum Application of Energy Conversion Methods

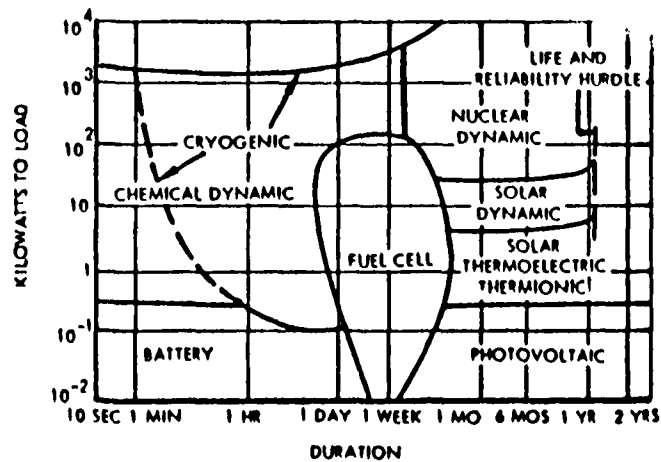


Figure 4. Forecasts of 1966 Areas of Optimum Application of Energy Conversion Methods

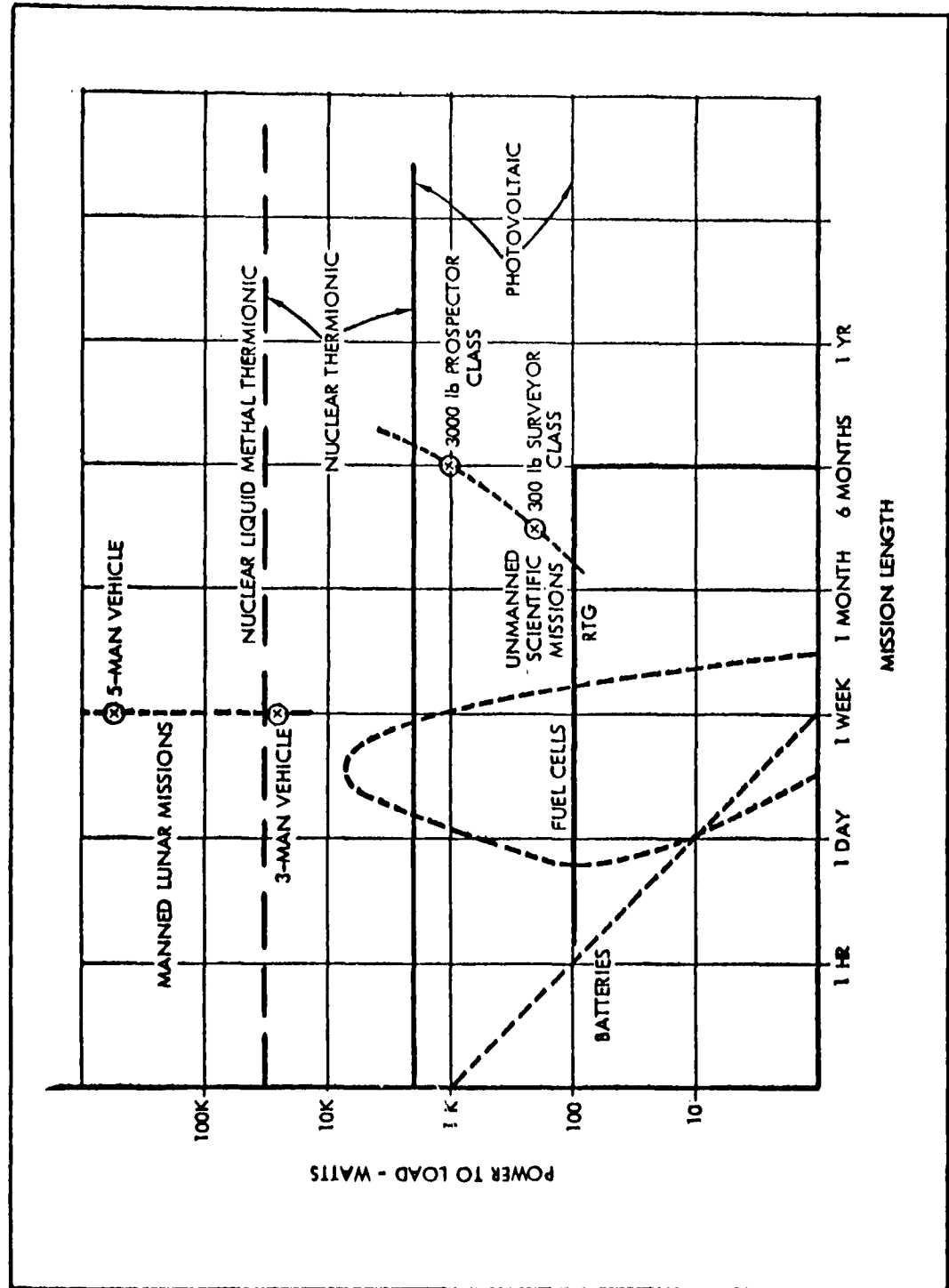


Figure 5. Power Conversion Applicable to Lunar Vehicles

SECTION II

PERFORMANCE OF SPECIFIC SYSTEMS

A. Solar Conversion

Photo Voltaic

Silicon solar cells are now available with efficiencies in the order of 9 to 10 percent at 20°C. The efficiency falls at elevated temperatures and makes environmental control a critical item. Fig. 6 shows the effect of temperature on cell efficiency. On the lunar surface, solar energy is available at the rate of 1400 watts/m² for normal incidence. Assuming an efficiency of 10%, 140 watts/m² of active cell area can be realized. In an actual solar cell power supply design, other considerations will reduce this maximum attainable value. The duty cycle, cell temperature, use of oriented or non-oriented arrays, erosion of micrometeorite cell covers, protection of cells during pre-flight, launch, and landing must all be carefully taken into account. When this has been done, 80 watts/m² is generally available. In the same way, the power-to-weight ratio for bare cells is 20 watts/lb.

A look at Ranger I is instructive. Here two 10 square foot oriented panels are covered with 8680 cells. Power output is 180 watts which gives 97 watts/m². The two panels weigh 19 pounds each for 4.75 watts/lb. Figures above include a UV rejection filter. This performance probably represents the best that can be done at present with solar cell power supplies when utilized in Satellites or space probes.

Photo Emissive Effect

Photo emission has been known for many years but little effort has been made in the past to use it for the conversion of radiant energy into electrical energy of sufficient magnitude for use as a practical source of power. A photo-emission diode is analogous to the thermionic diode. When the photo emissive layer is illuminated, light photons release their energy to the electrons upon

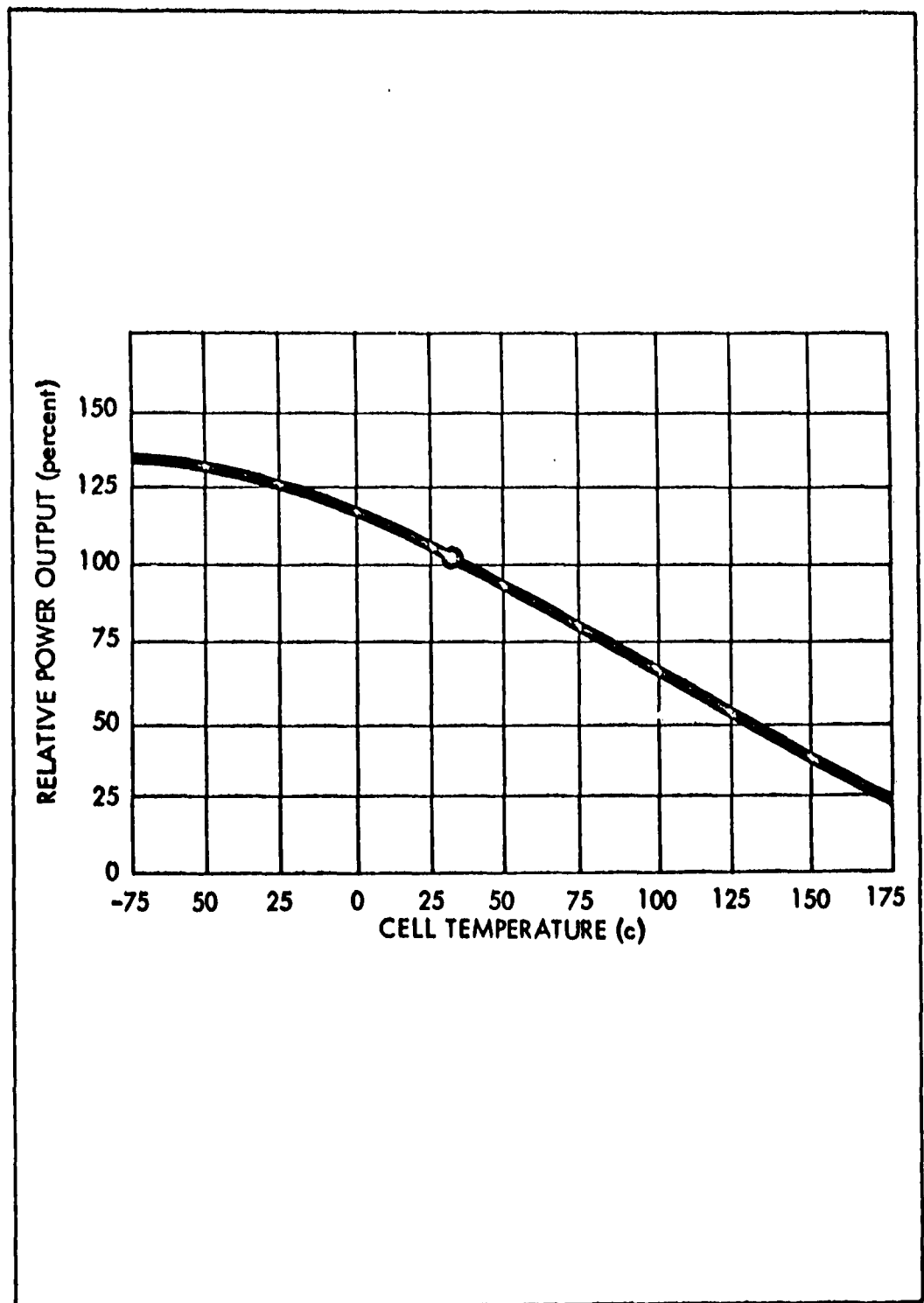


Figure 6. Variation of Silicon Solar Cell Maximum Power Output with Temperature

collision. If the amount of energy is greater than the work function of the material, the electron escapes with a kinetic energy equal to the excess of the quantum energy over the work function and it will reach the anode. In the thermionic diode, the anode is maintained at a lower temperature than the cathode, but in the photo-emission diode, the anode is kept "dark" to prevent emission which would result in an opposing current. It is also subject to space charge limitations. Recently, several organizations have been investigating the possibility of increasing the efficiency of a device utilizing this effect, and 3 percent appears obtainable. The chief advantages for this device are a high power-to-weight ratio and relatively low cost per unit power. Preliminary calculations indicate that approximately 3 kw of power at 28 volts can be obtained if the cells are mounted on a 30-foot diameter balloon. 150 to 50 watts/lb appears feasible. The material costs are fairly low and large area devices may possibly be fabricated by vapor deposition of the photo-emissive materials.

Thermo-electric Effect

Solar-powered thermoelectric generators should be considered as a source of power for space applications. A great deal of effort is presently being expended in the investigation of thermoelectric materials and generator design. A survey of the literature and reports resulting from various programs sponsored by the Department of Defense indicates that a number of thermoelectric materials have been investigated and their efficiencies over a given operating temperature range have been determined. In general, the melting points of these materials are low and their use in a solar powered thermoelectric generator does not appear feasible. Materials capable of operating at higher hot junction temperatures are required in order to obtain a reasonable Carnot efficiency. A recent paper (Ref. 5) discusses the merits and design problems of a solar-powered thermoelectric generator and its possibilities for space use. The data presented was obtained from the performance of an actual device and will be used later in this discussion for comparison to other systems.

Thermionic Emission

General Electric has developed a cesium vapor thermionic convertor with the following characteristics. The convertor operates with a 15 to 17 percent efficiency which provides a power output of 23 w at 1530° C. 10,000 hr. steady operation seem attainable for the device in applications where seal temperatures can be kept below 500°C. The convertor was developed to operate in a cathode

temperature range of 1200° to 1500°C. for applications in space vehicles. Power output drops to 12 w at 1330°C. These performance figures can be considered typical of state-of-the-art diodes.

Cesium contained in a reservoir is used to adjust the work functions of the anode and cathode surfaces as well as to create a plasma or interaction space between the anode and cathode. The reservoir is a long, narrow tube, sealed at the tip, extending from the anode side. The tube serves to maintain optimum temperature differential between anode and reservoir. Tip temperature is about 300°C. while anode temperature is in the area of 600°C.

In operation, a rather small space charge limited current is drawn between cathode and anode until the voltage applied across the internal cathode-to-anode region is sufficient to cause breakdown or initiation of a hot cathode arc. Breakdown is followed by an increase in current at practically constant voltage up to the point where saturation emission is drawn from the cathode. Depending on cathode-anode spacing and the vapor pressure conditions, there exist a number of electron mean free paths between the cathode and the anode. In this condition a low-voltage arc is formed in the space between the cathode and anode if their contact potential difference is great enough to initiate and maintain a discharge. Power density of the thermionic convertor is 4.6w per sq. cm., far greater than the 1/2 w per sq. cm. performance limit of vacuum convertors. A further discussion of some principles of thermionic diodes is given in Section II-B.

Comparison of Conversion Systems

In comparing the relative merits of the above systems for the direct conversion of solar energy to electrical energy, a number of factors must be taken into consideration. These are listed below:

- Reliability
- Power per unit weight, area or volume
- Complexity
- Susceptibility to damage by meteorites
- Radiation Damage
- Operating lifetime

Orientation requirement
Conversion of electrical output to
usable form

There is a basic difference between these four systems described above. The photoelectric and photo-emission devices convert the incident radiation directly to electrical output. The thermo-electric and thermionic devices are essentially heat engines and require the concentration of the incident radiation to the device or heat sink to achieve the required rate of heat flux per unit of cathode or heat sink area. The use of solar concentrators introduce problems that require serious consideration. For temperatures in the order of 500° to 750°C., required for thermo-electric generators, small errors in collector geometry and orientation can be tolerated. For temperatures in the order of 1000°C. and above for thermionic diodes, a high degree of accuracy in collector geometry and orientation is essential for reasonable collection efficiencies. A striking difference between the orientation requirement for solar cell arrays and solar collector mirrors exists. For a solar cell array, a deviation of 20° from normal incidence only results in a loss in power of approximately 6 percent. For solar collectors with fairly high concentration ratios, accuracy of orientation of .05° or less is necessary. A good analysis and discussion of this problem is given in Ref. 6 with a suggested solution. But the requirement remains for a complex servo loop with all the attendant equipment.

The effects of erosion or punctures by micrometeorites present problems for all four systems. Erosion of the surfaces of photovoltaic cells can be prevented with transparent, radiation resistant covers. Allowances must be made for the transmission losses in this cover, including the losses that will eventually occur when it becomes frosted because of sandblasting. The effects of punctures resulting in a loss of a cell or cells by larger particles can be compensated for. The cells are usually arranged in series-parallel groups and allowances can be made in the design for a loss of certain number of modules. The possibility of erosion by dust and sputtering caused by the impact of high energy atoms or molecules of the highly reflective surfaces of a solar concentrator must be taken into consideration. A protective cover for solar concentrators does not appear feasible. Punctures would remove only a very small fraction of the total collector surface area. In applications where heat transfer loops are used from the source

to the conversion device and from the device to a heat exchanger for rejection of waste heat, punctures of these components by meteorites would be catastrophic, particularly in those using a liquid heat transfer medium.

Radiation damage to the various components employed in these systems does not appear to be a serious factor. The solar cells on Vanguard I have been operating continuously since March 17, 1958. The Russians claim that the solar cells on Sputnik III have not been affected by radiation. The use of organic materials in construction of these systems introduces a radiation damage problem. This problem is difficult to solve because there is insufficient knowledge of the deleterious effects that may occur in a large portion of the ultra-violet spectrum. A high vacuum environment may also cause a degradation of organic materials. The use of spectrally selective coatings for the heat sink to make it an efficient absorber or as a means to obtain a low ratio of absorptivity to emissivity for solar cells and the structural materials of the array is being investigated by cell manufacturers. The stability of these coatings under radiation and high vacuum environment and under conditions of high humidity prior to launching will have to be determined.

The one system that has been proven in actual use in space applications is that employing the photovoltaic cell and electro-chemical storage batteries. Information for use in comparing the merits of the other systems to that of the photovoltaic cell is in the form of engineering estimates based on present results obtained on experimental models and predicted future capabilities of the conversion device. Some approximate figures shown in Table I indicate the potentialities of these four systems.

At the present time, the photovoltaic system appears to offer the highest watts/square foot, highest system efficiency, and the lowest watts/pound. The thermionic and thermoelectric systems require solar concentrators and the efficiency of these devices depends on the accuracy of the reflector surfaces and orientation. Efficiencies of 60 percent have been indicated for concentrators assuming geometrical perfection and proper alignment. Efficiencies on the order of 30 to 40% appear more likely for large expandable concentrators. The low concentrator efficiency imposes a severe penalty on the thermionic and thermoelectric systems. This is indicated in Table I where the power output in terms of

TABLE I

System	Watts/Pound		Watts/Square Foot of Collector Intercept Area		Possible System Efficiencies ⁴
	Present	Future	Present	Future	
Photovoltaic	1.5	5	7.4	13.5	10-12 percent
Thermionic (1)	7.2 ¹	17	2.2	3.5	2-4 percent
Thermoelectric (17)	20 ¹ , 2, 3	30 ² , 3	0.3	1	2-3 percent
Photoemission	--	50-150 ³	--	3	2-3 percent

1 Experimental

2 The watts/pound are only for the thermoelectric materials.

3 Does not include storage system

4 System efficiency is defined as $\text{Efficiency} = \frac{\text{Electrical output}}{\text{Total intercepted energy}}$

watts/square foot of collector intercept area is shown for the first four systems. This is further illustrated by comparing the required collector intercept area for a thermionic system where 4,680 square feet are required for a 10 kw power supply employing diodes with a 5 percent efficiency. If photovoltaic cells covered an equivalent area and produced 7.4 watts/square foot, the power output would be in the order of 34 kw. To produce 10 kw, only 1350 square feet would be required. To be competitive with the other systems, more efficient, high temperature materials will be required for the thermoelectric system. The photo-emissive system is in its early stages of development and cannot be considered as ready for hardware for several years. The conclusion is that photovoltaic cells would be the best choice for this application.

B. Radio Isotope Fueled Thermionic Generator (RTG)

An RTG promises to be an excellent auxiliary power supply for lunar vehicles where the power requirement is less than approximately 500 watts. Units are rugged, reliable, and because of high operating temperatures, require very little radiator area. Good use can be made of the heat output during the cold lunar night. RTG's in the 5 to 10 watt range are available now, 20 to 80 watts will be available by July 1962, and 350 watts may be available for Prospector in 1964.

Two types of thermionic diodes exist for use in RTG's, close-spaced vacuum diodes and cesium plasma diodes. At present, the close-spaced diode technology is further advanced. It operates with an emitter temperature of 2200°F and an efficiency of 5 - 7%. Power densities of 1 to 2 watts/cm² are possible now with 15% attainable in the future. Cesium diodes presently being utilized in equipment have efficiencies of 9 - 14% and power densities of 4 watts/cm² at 3000°F. The higher temperature clearly makes for a serious materials problem. The latest developmental models, such as that General Electric model described in the section on Thermionic Emission, have considerably better performance.

If one considers only those diodes available at this moment, cesium diodes look best. But present types require at least 2800°F and this leads to two problems; a materials problem in the diode, and fuel problem in the isotope supply. Even more significant, at 3800°F the cesium diode is only slightly better than the vacuum diode at 2200°F. Operation over 3200°F makes the cesium diode clearly superior but

accentuates the problems. To get high temperature from an isotope source, the volume to area ratio must be increased in proportion to the 4th power of the temperature. This means operation at high power levels (kilowatts) or the use of higher power density isotopes than are presently available. The vacuum diode will probably be chosen for most RTG applications.

Two factors must be taken into account when choosing an isotope. The first is that the particular isotope must be capable of supplying required thermal power at the required temperature to drive the diodes. The second factor is availability, which, of course, includes cost. Figure 7 shows some characteristics of possible isotope fuels. The power density shown can be reduced to any desired value by dilution with carbon or nickel. The mechanism by which the heat is produced is straightforward. The isotope is unstable and decays. Emission, either particulate or electro magnetic, occurs and heat is produced when the emitted radiation is absorbed in the fuel or container.

From the point of view of shielding, Alpha emitters are best. The mean free path is quite small. The Alpha particle picks up two electrons and is then a helium atom. In practice, Curcium 242 and Plutonium 238 are perhaps the most suitable.

Curium is available in quantity and approximately \$80/watt now with eventual reduction \$45/watt. Plutonium runs \$1600/watt but the half-life is so much longer that this type of comparison can be misleading. If one considers the dollars per watt-year then plutonium is only \$25.7/watt-year, while curium will reach a low of \$140/watt-year. Other cost figures are: P1-210, \$310/watt; Ce-144, \$9.20/watt, Sr-90, \$770/watt.

The costs of radio isotopes are artificial, being set rather arbitrarily by the Atomic Energy Commission. The more expensive isotopes are produced artificially in nuclear piles. Others are produced by bombardment of nuclei in cyclotrons and other particle accelerators. The cheapest are those separated from the radio active wastes produced by nuclear power generation stations. At present, most of these waste products are stored, buried, or dumped at sea. Abundant though they may be, the high cost of separating, purifying and handling will almost certainly limit RTG applicability to power levels under 1 kw electrical.

Figure 7.
Some Characteristics of Isotope Fuels

Isotope	Power Density (w/cm ³)	Radiation Output	Half Life
Ce-144	12.5	Beta	285 days
Pm-147	1.1	Beta	2.6 years
Sr-90	.54	Alpha	27.7 years
Cm-242	810	Alpha	162.5 days
Pu-238	7.5	Alpha	86.4 years
Po-210	1130	Alpha	138.4 days

C. Nuclear Reactor

This discussion contains three sections. The first is a general review of space nuclear electric power generators. The second is a description of two particular generators designed by General Electric. The third describes some aspects of nuclear power system operation on the moon.

It will be shown how the payload tolerance affects the shield weight. It has been assumed that a conventional payload utilizing transistors can be subjected to 10^7 r of gammas and 10^{12} nvt of fast neutrons. A payload especially designed for a nuclear reactor supply could utilize hard vacuum tubes and radiation resistant transistors in especially designed circuits which are tolerant of component drifts. Such a payload could then be expected to withstand greater than 10^{14} nvt and 10^9 r. Extremely radiation sensitive payloads, i. e., photographic film, should be equipped with individual shields so as to raise their tolerances to the payload design value.

The choice of reactor location is a most important factor. For example, the reactor shield combination could be the heaviest component in the supply system. This shield can vary from a minimum of 250 to 400 pounds for an optimum vehicle arrangement with a radiation resistant payload; 500 to 600 pounds for an optimum vehicle with a conventional transistorized payload; and 1000 to 2000 pounds for an inept vehicle payload arrangement.

Ground handling equipment and launch pad modifications for the nuclear reactor are minor if orbital or lunar surface startup is utilized. Telemetry for the startup must be supplied. Straight-forward "go-no-go" checkout instrumentation will be available. No nuclear checkout at the launch pad would necessarily be anticipated; this could be covered in the acceptance test procedure before delivery.

If a pre-launch nuclear startup is desired, the system would be brought to power using a small 50 kw electrical heater built into the primary cooling loop. The power conversion equipment would be checked and the payload transferred to the reactor supply. The reactor would be checked at very low power and then, just before

launch, taken to full power as electrical heat is removed. If the mission is scrubbed, the reactor or the entire final stage (depending on the separation provisions) must be placed in a shielded storage pit. A cleanup crew should also be available in case of a destructive booster abort.

A comparison of various nuclear electric power systems for space application is presented in Figure 7. Power plant performance of four nuclear systems in the hardware phase are plotted on this Figure with circles. SNAP 10, which is nuclear thermo-electric is shown in the upper left hand corner; SNAP 2, nuclear thermo-electric, near the center of the Figure, and the SNAP 8 and the double SNAP 8 units, are plotted near the center towards the right. The SNAP curve, passing through these points, indicates the relationship of pounds per kilowatts versus power output for higher powers for nuclear turbo-electric systems. Also shown are the regions where solar photovoltaic and solar thermionic are considered to be optimum. The three points labeled STAR are three nuclear thermionic power plants designed by General Electric which are direct radiation cooled. These points are not different power plants but the same sized power plants at three different levels of performance, i. e., at three different times in its development life.

Another type of system is shown in the lower right hand corner of Figure 7. This system is the Liquid Metal Cooled Thermionic System wherein the thermionic converters are integral with the reactor. The shaded region is system specific weight whereas the curves below are reactor specific weight. These power plants extend over the power regions of about 90 KW up to and probably exceeding 1000 KW. The exact region of overlap or advantage between which is direct radiation cooled, and the liquid metal cooled thermionic system, is not known at present. It is felt that the cross-over between one type of power plant and the other will be a function of customer requirements and detail of specifications rather than any definable technical aspects of the power plants known at the present time.

Figure 8 shows a 3 foot long by 1 foot diameter STAR power plant with a shield attached to the three external bus bars. The reactor consists of a uranium carbide type of fuel contained in a central ring, the two end reflectors to which the bus bars are attached, and the thermionic converters which are located on the outer surface

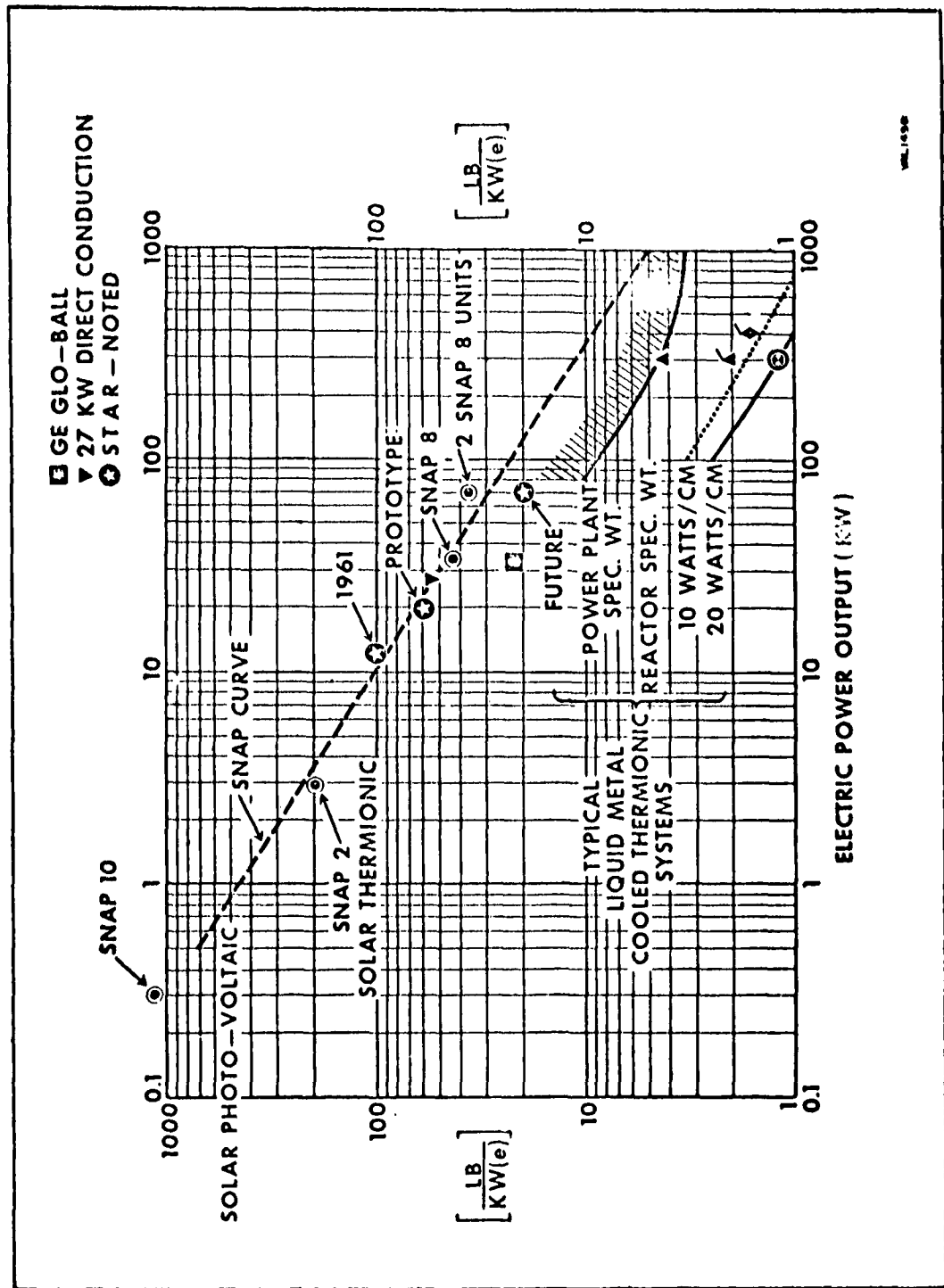


Figure 7. Nuclear Electric Power Systems

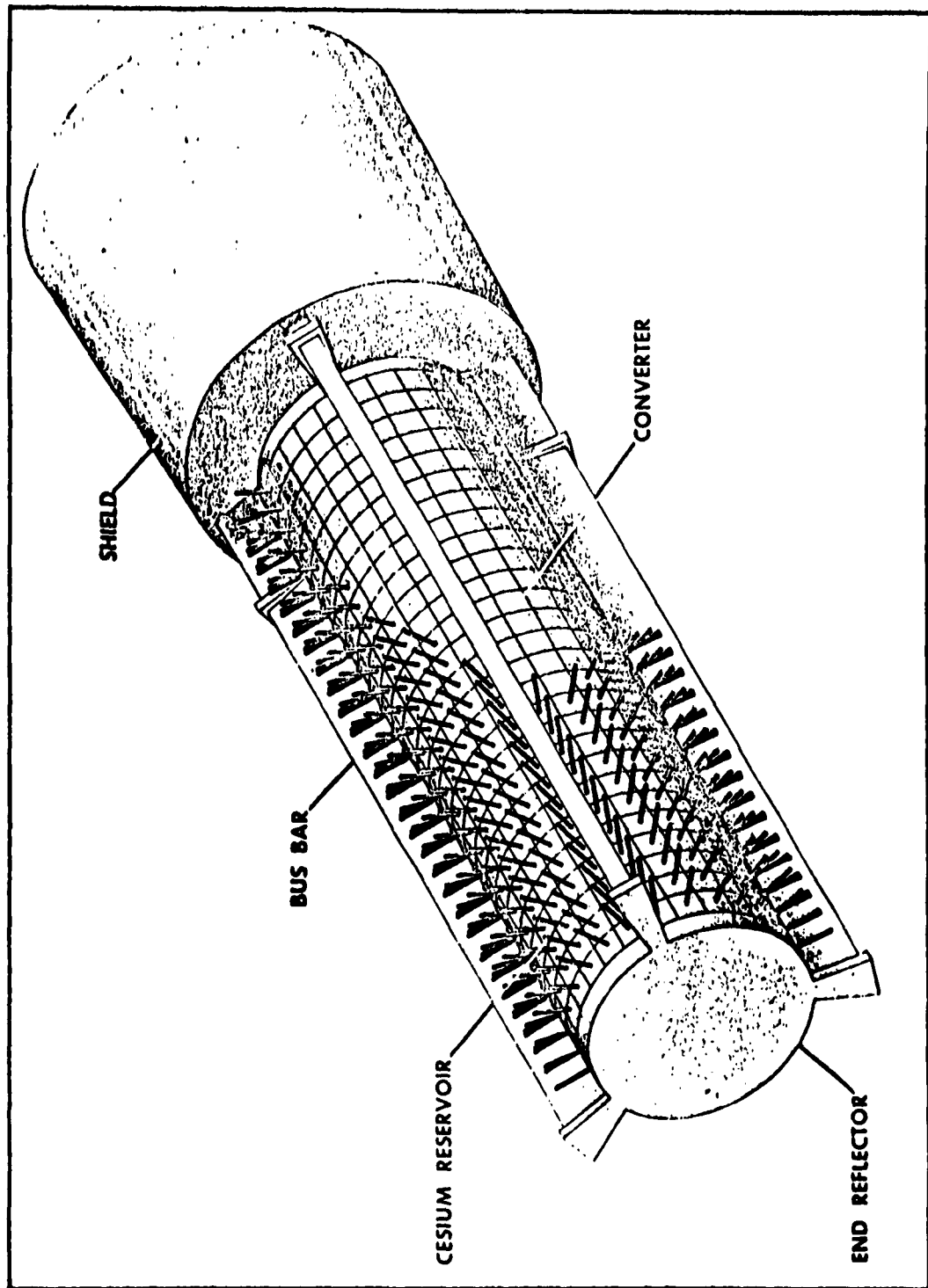


Figure 8. Star Thermionic Reactor

of the reactor. The procupine effect shown in this figure is due to the individual cesium reservoir tubes shown emanating from the thermionic converters.

During operation there are no moving parts of the power plant. The negative temperature coefficient of the fuel causes the reactor power plant to maintain a constant fuel temperature. Startup of this power plant is accomplished by first moving the two halves of the power plant together and then moving one end reflector into the final position to achieve criticality. Heat is conducted from the nuclear fuel directly to the cathode of each thermionic convertor on the surface of the reactor. The heat which is not directly converted to electricity is then rejected from the anode of each convertor by radiation to free space. The anode also acts as a neutron reflector for the reactor.

The essential simplicity of this power plant implies a high degree of reliability from a design point of view. The small vulnerable area, i. e., the cesium tubes, and the boundary around each thermionic convertor on the surface, assures minimum susceptibility to meteorite penetration. Penetration of any convertor will only cause the loss of that convertor, and not the loss of the entire power plant, because the converters in any one ring are connected in parallel and adjacent rings are connected in series. Thus, with this series-parallel arrangement, the loss of any one convertor only diminishes the power output of the reactor power plant by the output of one convertor. This type of power plant does not depend on any gravitational field for its operation and hence, its feasibility can be provided in ground test. It would not require any boost vehicles to demonstrate feasibility.

The performance of the STAR power plant is presented in Figure 9. Tabulated on the left side are the various performance parameters and temperatures. These parameters are tabulated for the three levels of performance corresponding to three different points in the power plants development cycle. No shield or power conditioning equipment weights are included. The parameters listed in the 1961 Column are those which are felt to be attainable at the thermionic convertor performance already or about to be demonstrated by the General Electric Company. It is felt that the performance level tabulated under the Prototype Column could be attainable in a power plant on prototype tests in 1964. The parameters tabulated under the Future Column are those felt attainable

	1961	PROTOTYPE	FUTURE
WEIGHT (LBS)	1300	1200	1400
POWER (ELECTRICAL)	13 KW	20 KW	70 KW
TEMPERATURES (°K)			
FUEL	2460	2470	2700
CATHODE	2300	2300	2500
RADIATOR	1290	1560	1825
SPECIFIC WEIGHT (LB/KW(e))	100	60	20

Figure 9. Star Performance Summary

near the end of the development cycle of the power plant and represent the final development level of performance attainable for this type of a system. These points are all for a three foot long power plant.

STAR power plant length can be varied over a wide range from, say, one foot long, limited by critical mass, up to as long as is desirable from a weight point of view. The convertors for any sized power plant would be identical. The fuel thickness would vary slightly for different length power plants. This would allow the power plant to be readily tailored to specific power requirements.

A one foot long power plant would be in the 500-700 pound range. The power output would be roughly one third of those tabulated. Thus, a power plant 1 foot long at the 1961 level of performance would produce 4.3 KW of power at a specific weight of $600/4.3 = 140$ lbs/KW, for example. This defines a curve for different sized STAR power plants that is much flatter than the SNAP curve. This points out the attractive specific weight of the power plant even for the 1961 level of technology.

The second type of power plant system is the Liquid Metal Cooled Thermionic System. Such a system is shown in Fig. 10, which is a schematic diagram of 1000 KW electric power plant (LCT-2). It consists of the reactor, the liquid metal headers which connect the reactor to the heat exchanger, and the primary liquid metal coolant pump. Between the reactor and the heat exchanger is a neutron shield. The radiator shown is a conical radiator roughly 33 feet long, a little less than 10 feet in diameter at the large end, and a little less than 4 feet at the small end. The radiator in this particular design is segmented into 6 sections, each section being connected to a one sixth of the secondary coolant loop of the heat exchanger. The radiator is sized so that any 5 sections provide adequate cooling for rated power of the power plant. This provides a redundancy in the radiator such that a meteorite penetration of any section of the radiator would only drain that section of its fluid and hence, not cripple the power plant. This will minimize the amount of micrometeorite shielding on each of the radiator tubes and thus save weight.

This arrangement also provides a very localized region for the activated liquid metal that passes through the reactor and thereby

allows for a localized shield to protect any vehicle component or person from the effects of the radiation from the heat exchanger, if the mission or vehicle requires dose limits. This system has been examined also in the 90 KW electric range. The dimensions for such a system would be roughly one third of those shown in Figure 10.

The performance level for the LCT-2 is shown in Figure 11. As indicated the electric power is 1000 KW, the thermal power is 5500 KW, and the weight less shield and any power conditioning equipment is 5000 pounds, thus yielding a specific weight of about 5 lbs/KW. This does include about 300 pounds of shielding to provide for adequate radiation tolerance for the liquid metal pump units, but does not include shielding to meet unknown or unspecified customer shield requirements. The corresponding figure for the 90 KW electric plant examined was about 11 lbs/KW. However, it appears reasonable that for the ranges of voltage required for propulsion engines, i. e., ion engines, the conversion equipment would weigh on the order of 3 to 7 lbs/KW in the 1000 KW sizes complete with its cooling system. This means that 8 to -2 lbs/KW for 1000 KW electric power systems for primary propulsion use appears feasible.

There are three aspects of lunar operation that the designer must be aware of. The first of these is the effect of reflected heat from the lunar surface. Fig. 12 shows a STAR type power plant located above the lunar surface. Since the lunar surface has a very low conductivity, the radiation from the reactor will heat up the lunar surface to some temperature, which depends upon the length of time of exposure. The radiation level of the reactor is about 12 watts/cm² and one can compare this with the sun's radiation in the region of the earth of about .14 watts/cm². One can see from a thermal energy point of view that the effect of the reactors thermal radiation must be accounted for on the lunar surface. If the lunar vehicle system is such that this temperature variation causes excessive temperature variation on the surface of the reactor, a heat shield will have to be placed between the reactor and the lunar surface. The non-varying effect of this heat shield would be taken into account in the reactor design. This will probably derate the reactor somewhat, but will give a constant power output and environment regardless of whether the reactor is moving or stationary on the lunar surface.

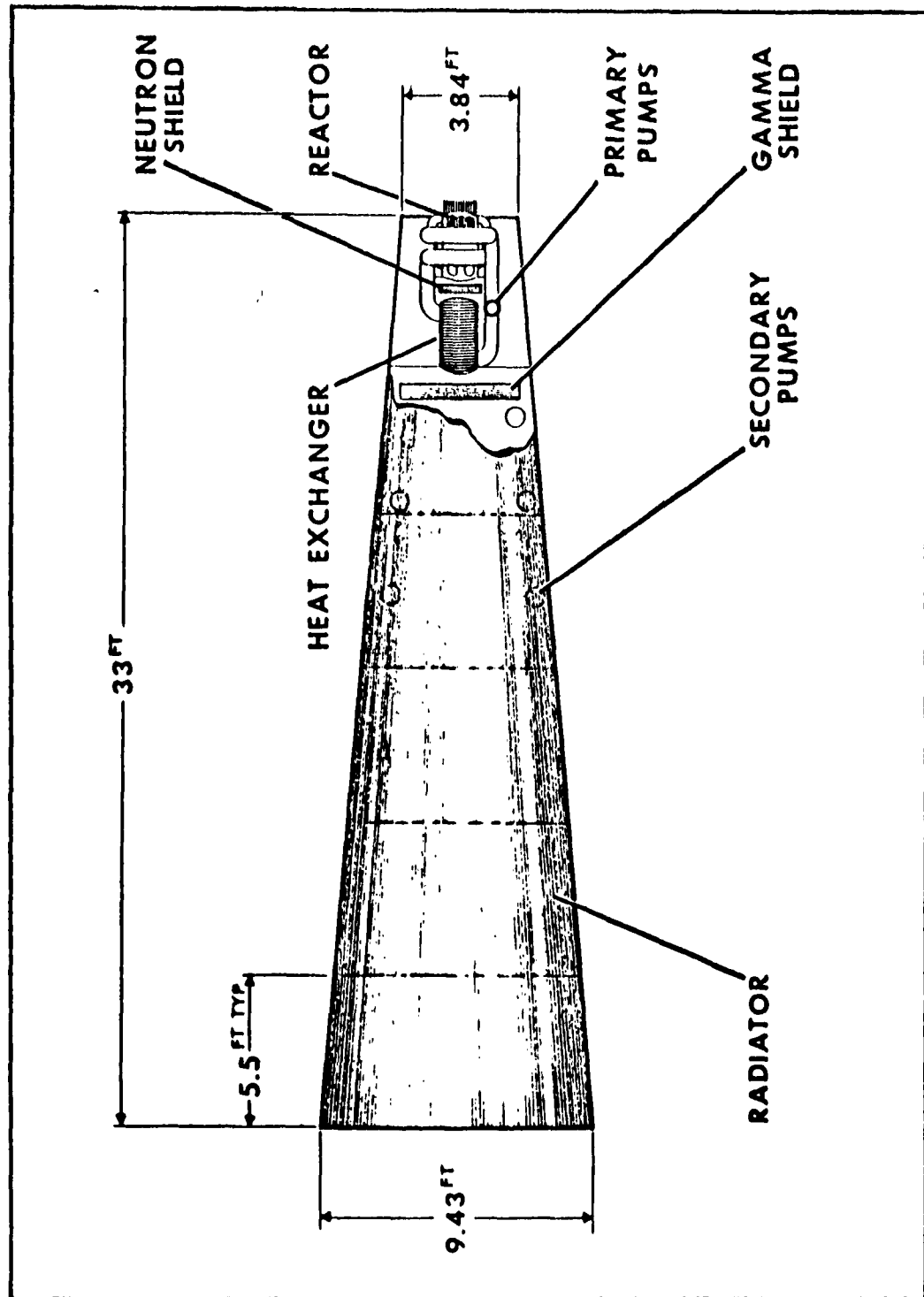


Figure 10. 1000 kw (e) Liquid Metal Cooled Thermionic System

POWER (ELECTRIC)	1000 KW
POWER (THERMAL)	5500 KW
WEIGHT (LESS SHIELD, AND POWER CONVERSION EQUIPT.)	5000 LBS.
SPECIFIC WEIGHT	5.0 LBS/KW
SPECIFIC POWER	.44 KW/Kg

Figure 11. Liquid Metal Cooled Thermionic Reactor LCT = 2

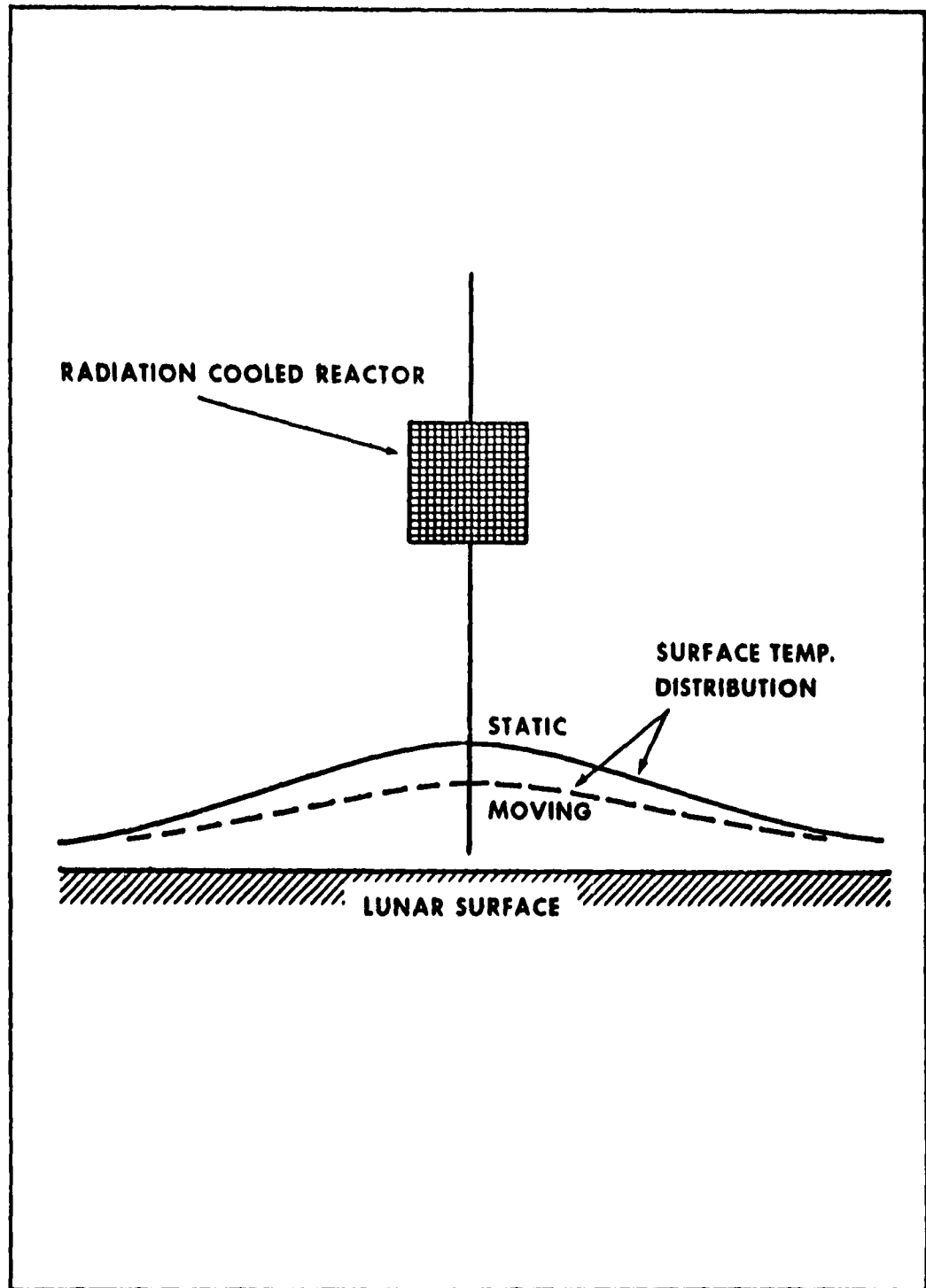


Figure 12. Effect of Reflected Heat

The next consideration is that of materials stability as a function of radiation dosage. Fig. 13 shows the region of critical radiation tolerance for various kinds of materials plotted versus dosage in Rep on a log scale. As can be seen, metals and ceramics have the highest tolerance and semi-conductors have a very wide range and go down as low as 2×10^2 Rep. Elastomers have a fairly narrow range and plastics have a fairly broad range. Note that there is a wide range of materials available to a designer that give various radiation tolerances. Depending on system considerations, radiation damage of materials can be averted in vehicle design. To put some specific numbers into this situation, Fig 14 was prepared which is the reactor shielding required versus neutron dose for a 50 KW thermal reactor and separation distance of 20 feet between the dose receptor and the reactor. This reactor power corresponds to about 10 KW of electric power with an efficiency of about 20 percent which is roughly in the range of power of interest on the lunar surface for later missions. The fast neutron dose in one year plotted as nvt is shown on a log scale as the ordinate and the lbs/ft² of beryllium is shown as the abscissa on a linear scale. 10^{12} nvt corresponds to about the tolerance dose of the most sensitive germanium transistors. The radiation tolerance for hard vacuum tubes is around 10^{16} , thus one can see by the selection of various electronic components that shielding may be varied from about 340 lbs/ft² down to zero. Thus, the designer has several alternatives available to him. It is obviously necessary to study the whole vehicle and not just a collection of existing components before any specific conclusions can be reached. (10^{12} nvt corresponds to about 2×10^2 Rep which will give a correspondence between Figures 13 and 14.)

The third area of lunar surface operation of nuclear electric power plants is that of lunar surface activation. Figure 15 is a plot of contact dose (milliroentgens per hour) from activated lunar surface versus the lbs/ft² of beryllium required for a reactor thermal power of 50 KW with a separation distance of 20 ft. and 7.8 hours of exposure (7.8 hours is three half-lives of silicon 31). This exposure will yield nearly saturated activity for silicon 31 by neutron absorption in silicon 30. This silicon is present in the basaltic type rocks of the moon. For the purposes of this calculation, calcium magnesium silicate was assumed. As can be seen, a dose rate of 1 rem/hr would result from 7.8 hours of exposure of the reactor over the lunar surface. One should remem-

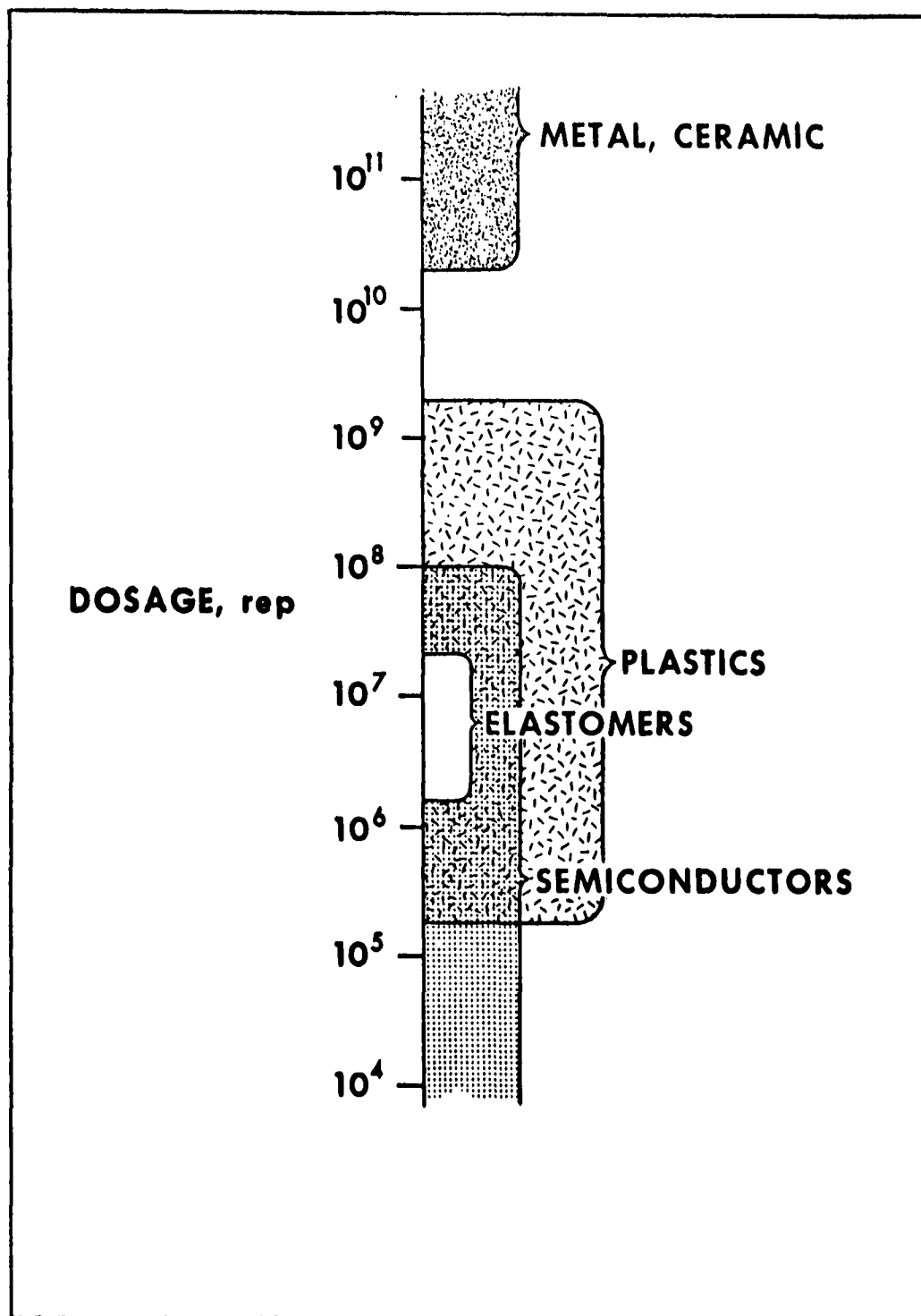


Figure 13. Material Stability Versus Radiation Dose

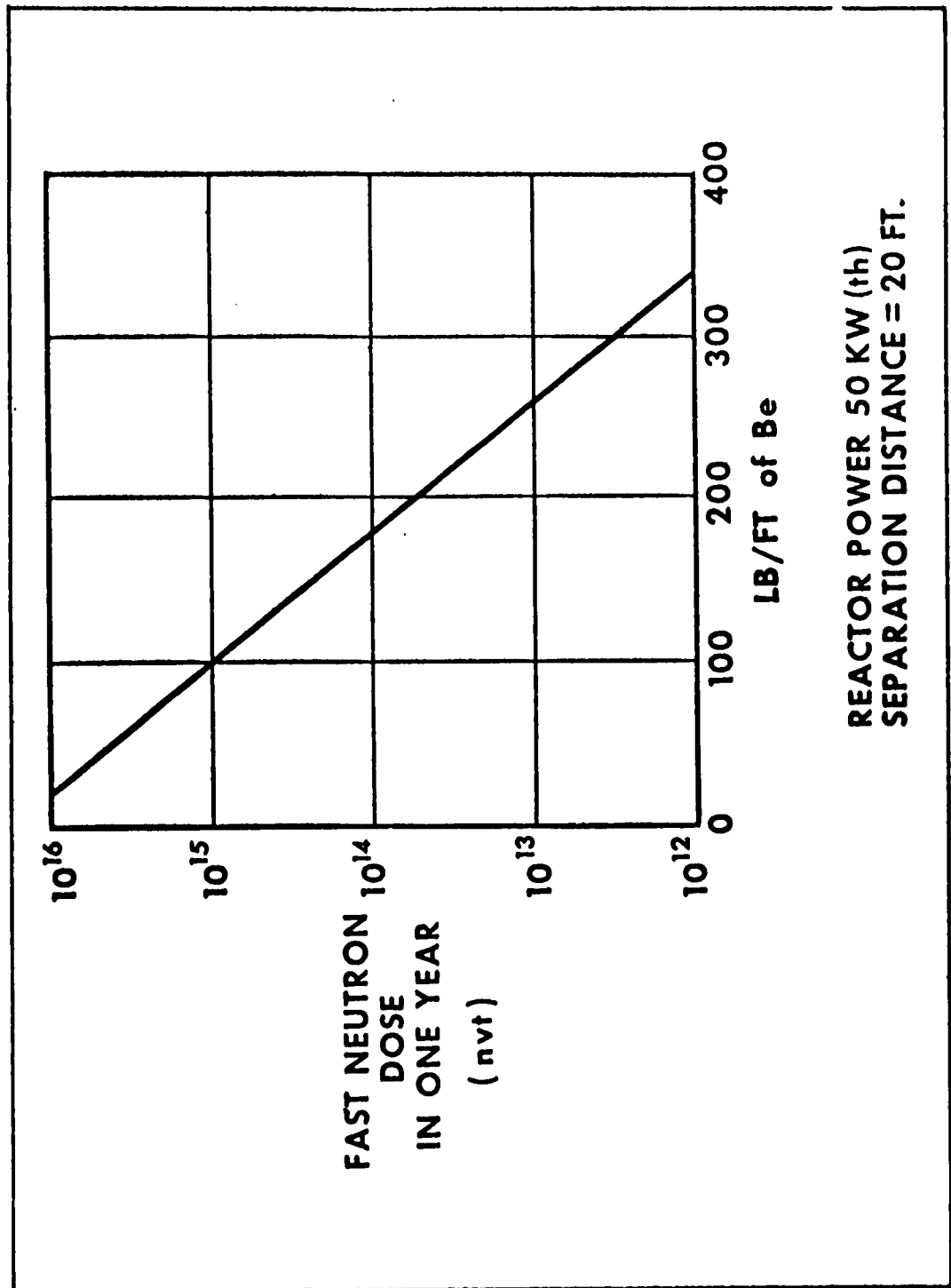
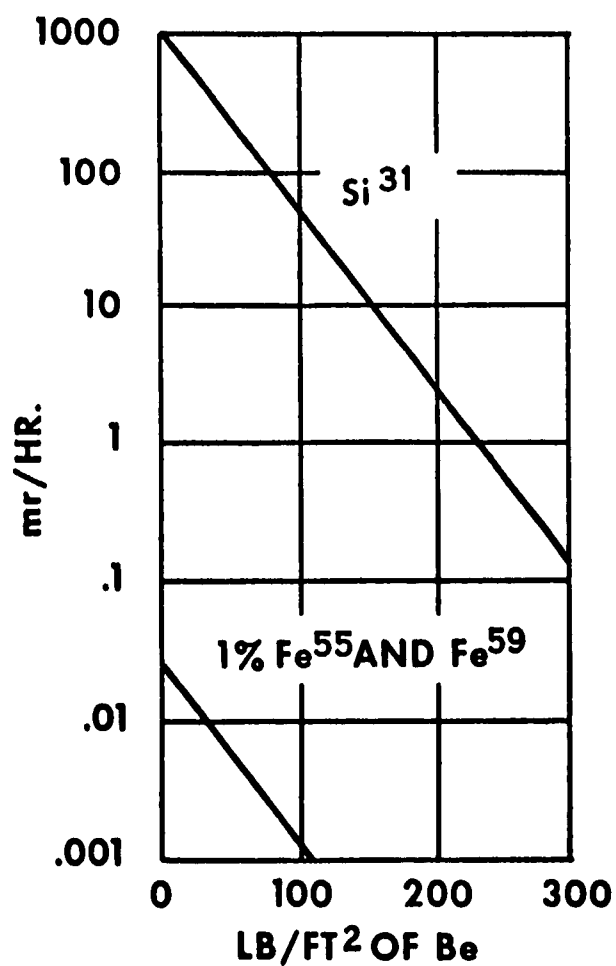


Figure 14. Reactor Shielding Versus Fast Neutron Dose



REACTOR POWER = 50 KW (th)

SEPARATION DISTANCE = 20 FT.

7.8 HOURS EXPOSURE

Figure 15. Lunar Surface Activation Versus Shield Weight

ber that this activity has a half-life of 2.6 hours; therefore, two days after exposure, the dose rate would be essentially negligible for human occupation of the area. This does not mean that it would constitute an acceptable background for low background counting of lunar samples. It might be desirable to do the low background counting on the moon prior to reactor operation in that specific area.

Since the constitution of the rock on the surface of the moon is not known, another calculation was made for an assumed iron concentration in the rock of one percent. The predominant activities here being iron 55 and 59 which is due to neutron absorption in iron 54 and 58. The level of activation at the end of 7.8 hours exposure is very low indeed. The primary reason for this is the relatively long half-life of iron 55 and iron 59, which is 2.9 years and 45 days respectively. Thus for long exposure times, the dose rate would be proportional to the exposure time up to, say roughly, .6 of the isotope half-life. Therefore, long exposures could lead to high levels of long lived activity if iron is present in the lunar surface. Other materials in the lunar surface, such as manganese, cobalt, and the like, could also have similar effects. Calcium and magnesium do not appear to be significant compared to the silicon. Thus, one can see that the specific requirements of the mission determine to a great extent (1) the separation distance of the reactor from the lunar surface, (2) the time of exposure, and (3) whether there is any shielding required to meet the mission requirement. It can be seen, if human habitation is a consideration, no shielding at all would be required for long times after shutdown, whereas on the order of 300 lbs/ft² of shield required of beryllium could be necessary if dose rates less than 1 milliroentgen/hr. immediately after exposure were required.

D. Batteries

Batteries, electrochemical devices which convert stored chemical energy directly into electrical energy, are the most efficient energy converters known today. Electrochemical cells are generally classified into two groups, primary and secondary, based on the nature of the chemical reactions. Primary cells are discarded when output drops below a usable level. Secondary cells convert chemical energy by reactions that are essentially reversible. A battery, of either primary or secondary type, consists of two or more cells connected in either a series, parallel, or

series-parallel arrangement to provide the needed power. The five basic components of a cell are:

Anode - the negative electrode from which electrons flow into the external circuit. Anodes are reducing agents which give up electrons and go into solution, forming positive ions.

Cathode - the positive electrode into which electrons flow from the external circuit. Chemically, cathode materials are oxidizing agents which can accept electrons with ease.

Electrolyte - a solution that permits ionic conduction between anode and cathode.

Separator - an inert, porous insulating substance to physically separate the anode and cathode; ions in solution can flow between the electrodes.

Seal - a composition to prevent loss of electrolyte and water while permitting gas to escape.

The theoretical energy that can be withdrawn from the primary or secondary cell depends on the chemical reactions which occur at the anode and cathode. In general, most cells operate below their theoretical limits because of polarization effects, irreversibility of the electrode reactions, or ohmic losses.

The four major classes of primary cells are dry, wet, reserve, and fuel or continuous feed. Most of the new developments of the last 10 years have been of the dry, reserve, and fuel-cell types.

Secondary or storage cells are those which can be discharged and then recharged by reversing the current. These cells are usually used as energy reservoirs or energy storage devices in electrical systems rather than a prime power source. The following criteria are used to evaluate a secondary cell:

Cost

Service Life - usually given in terms of duty cycle

Energy-storage capacity - expressed as watt-hours per pound or unit volume

Rate at which energy can be withdrawn from the cell - the maximum rate of discharge - expressed in watts per pound or unit volume, or as the time in which all of the available stored energy can be taken out.

Rate at which energy can be stored in the cell - the maximum rate at which the cell can be charged - expressed in watts per pound or unit volume, or as the time in which the cell can be recharged.

Charge retention; or better, rate of loss of charge - the tendency of a battery to dissipate energy stored in it by internal reactions - expressed as the percent of stored energy lost per unit of time.

Operating temperature range and the effect of temperature on the other properties.

In some special cases other factors are also important, such as whether it can be operated in any orientation.

Three battery types which have definite applicability to a lunar vehicle are compared in Table II. These three are nickel cadmium, silver zinc and silver cadmium.

The sintered-plate, sealed nickel cadmium battery is used extensively in satellites today. Some typical characteristics of a developmental battery are taken from Gulton data. The battery consists of separate steel cased, hermetically sealed cells. Capacity is 12 watt hours/lb and 1.2 watt hours/cubic inch. Voltage with no load is 1.37 volts/cell and falls to 1.2 with nominal load. Charging voltage may range from 1.1 to 1.58 volts/cell. NiCd batteries in general have the following characteristics: operational temperature range is -60° to $+200^{\circ}$ F. Cycle life is very good. In some cases manufacturers will state "There is no known limit". Maximum discharge current will range up to 25 time rated ampere-hour capacity. Internal resistance will be approximately 1 milliohm for 10 amp-hour cells.

Silver zinc cells, available only since World War II, are characterized by extremely high watt/lb ratio and fast discharge rates. Typical data for several high rate batteries discharged to 80% would be: 1 hour rate, 20 to 25 cycles; 30 minute rate, 10 to 15 cycles; 10 minute rate, 5 to 10 cycles. Low rate silver zinc batteries show better charge-discharge characteristics. Up to 200

TABLE II
CHARACTERISTICS OF THREE APPLICABLE BATTERY TYPES

Type	Application	Advantages	Disadvantages	Service Life		Operating Voltage for light drain
				Float or Shallow Discharge	Regular or Deep Discharge	
Silver-Zinc	1. Specialized military needs where weight and volume are at a premium	1. High capacity 2. Excellent performance at high discharge rates	1. High cost 2. Short cycle life	-	10-300 cycles	1. 40 to 1.50
Silver-Cadmium	1. Low rate; long life applications	1. Cycling life and charge retention better than silver-zinc cell	1. High rate performance not as good as silver-zinc cell	-	300 - 1,000 cycles	1. 05 to 1.10
Nickel-Cadmium	1. Sealed units used in space vehicles	1. Ruggedness 2. Long life 3. Can operate from 180F to -65F 4. Can be in various charge or discharge states without harm 5. Can be hermetically sealed	1. High cost 2. Lower cell voltage than lead-acid	Up to 25 years	Up to 1,500 cycles	1. 10 to 1.30

complete cycles or 1000 partial cycles over a 3 year wet life are obtainable. Storage temperature range is: wet, +100°F to -55°F; dry, +165°F to -85°F. Internal resistance is typically 0.3 milliohms. Open circuit voltage is 1.86 volts and 1.50 volts under reasonable loads. Under optimum conditions 80 watt hours/lb and 5.6 watt hours/cubic inch can be realized.

Silver cadmium cells are almost identical to the silver-zinc cells except for the use of cadmium in place of zinc. Capacities are not as high as that of the silver-zinc cell but cycling life and charge and discharge cycles can be obtained. Where only shallow cycles are required, 2000 to 3000 cycles can be obtained. Open circuit voltage is 1.4 volts and nominal voltage under load is 1.1 volts. Operational temperature range is +165°F to -20°F. Internal resistance ranges from 0.1 ohm for a one amp-hour cell to 1.5 milliohms for a 300 amp-hour cell. Present silver cadmium batteries will deliver 22 to 34 watt hours/lb and 1.5 to 2.7 watt hours/cubic inch.

SECTION III

DESIGN CONSIDERATIONS FOR POWER SYSTEM
FOR 300lb/3000lb VEHICLE

A. System Specifications

This section presents a power system design for both the 300 and 3000 pound vehicles. The specifications for these systems are given below.

I. 3000 lb Mobile Vehicle

Primary Power System

A. Generator	Power Output	Average	1 kw at 28 volts $\pm 10\%$
		Maximum	Surge requirement to be handled by secondary power supply
	Weight		Minimum weight commensurate with high reliability
B. Distribution System			
	A. C. regulated		400 cps $\pm 5\%$ 110 volts rms $\pm 1\%$ full (load to no load) 20 watts maximum load (short circuit protected)
	D. C. regulated		18 to 22 volts 115 watts voltage regulation $\pm 1\%$ full load (to no load) short circuit protected

D. C. unregulated

28 volts
short circuit protected

C. Environmental Control

Primary power system must survive the cold of lunar night and operate during the succeeding lunar day.

Secondary Power System

Purpose

To provide energy for peak power requirements and emergency motor operation

A. Total Stored Energy

1000 watt hours

Weight

Minimum weight commensurate with high reliability

Voltage

28 volts

Peak Continuous Current Drain

36 amps

B. Environmental Control Sub-System

Batteries will require protection from heat of lunar day and cold of lunar night.

II. 300 lb Mobile Vehicle

Primary Power System

A. Generator
Power Output

100 watts $\pm 10\%$
28 volts

B. Distribution System
A. C. regulated

400 cps
110 volts
10 watts

D. C. regulated	18 to 22 volts 10 watts
D. C. unregulated	28 volts
C. Environmental Control	Power system must survive lunar night and operate again succeeding lunar day.

Secondary Power System

The specification here is analagous to that of the 3000 lb. vehicle above. A battery storage system is desired with the total energy storage of 500 watt hours.

III. The following data applies to both vehicles

Average vehicle speed	1.5 MPH
Maximum vehicle speed	3 MPH
Average grade to be traversed	5° to 15°
Maximum grade to be traversed	30°
Vehicle travel direction	any
Initial vehicle position on Moon	-30° to +30° latitude, 20° to 60° E. Longitude
Maximum G load	20 Earth G's, non-operating 4 Earth G's, operating

B. Selection of Power Source

On the basis of sections I, II, and the specifications listed above, selection of the power source can now be made. Of all the solar conversion systems the solar cell array is most applicable. An RTG looks very promising for the 300 lb vehicle where the power requirement is only 100 watts. But the thermal problem involved in rejecting approximately 2KW of heat is formidable. This assumes an overall efficiency for the RTG of 5%. Note that this thermal problem starts at launch assembly when the RTG is installed and continues throughout all phases of the mission. For the larger vehicle which requires 1000 watts, the RTG is impractical. Development of such a power source does not appear attainable by 1964 and the cost would be prohibitive, several millions

of dollars per unit. A nuclear reactor as a power source could be utilized for the large vehicle. It seems clear that for larger vehicles, on the order of 500 to 10,000 lbs requiring 3KW or more, the reactor will be a logical choice. In this case the reactor is marginal for the 3000 lb vehicle and completely unsuitable for the 300 lb vehicle. If the payload of the 3000 lb vehicle did not consist of equipment for precision scientific measurements - all highly sensitive to β , γ , and n radiation - the reactor would be a serious contender. But preliminary estimates of shielding required obviate the use of the reactor here. Mission analysis has shown that self-contained vehicle power units are mandatory for these vehicles. If a separate power station with some form of power transmission to the vehicle - cable, RF transmission, etc. - would be compatible with future mission requirements, the reactor looks good. For the time period and mission length contemplated, fuel cells appear to have no utility. The conclusion then is that solar cell arrays will be utilized for both vehicles for primary power source and batteries will comprise the secondary or back-up power source.

C. Oriented/Non-oriented Array Considerations*

Either oriented or non-oriented arrays can be used. The more complex oriented arrays will be investigated first. The stowage problem is perhaps the first to consider. The relatively small volume available in the nose cone makes folding of the array to some compact package necessary. The problems of launch shock and vibration can generally be solved more easily with a folded rather than extended array.

A number of folded panel configurations are possible. The final design would depend on the volume and shape of space allocated to this equipment in the vehicle. The stowed configuration will depend strongly on the final panel selection.

In Figure 16 "sunflower" opening system is illustrated. This system consists of eight rigid triangular sections supporting the solar cells. The sections would be opened in a manner of an umbrella. Upon opening the figure becomes octagon shaped. It would be rigidized by some suitable hinge interlocking methods. When in folded position, it has a conical shape. While this shape does not lend itself to good stowage, it does have an advantage of opening with a simple, reliable, single movement.

* Much of the following report has been taken from Ref. 14, supplied to G. E. by Hoffman Electronics on an informal working basis.

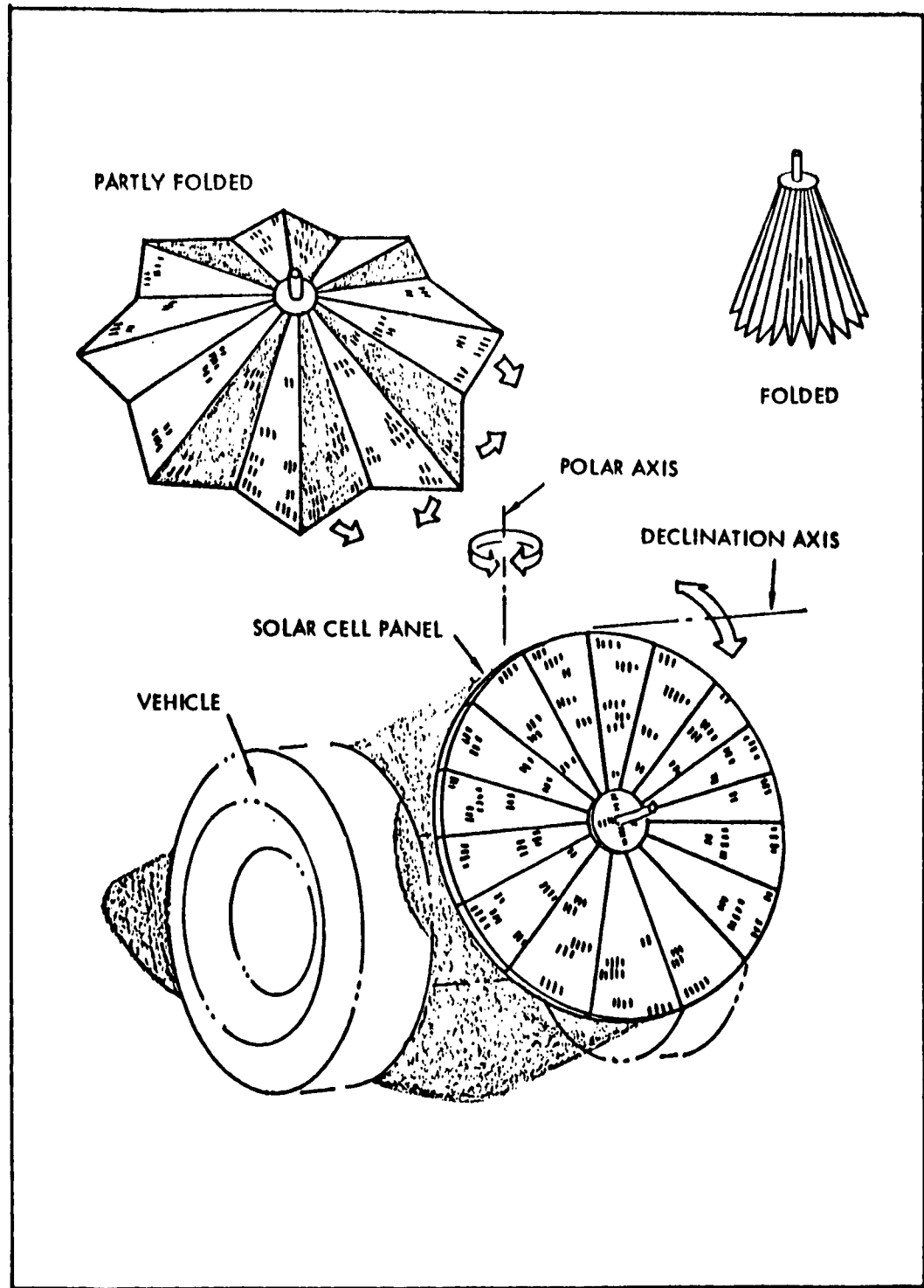


Figure 16. Solar Cell Panel

Figure 17 illustrates a square panel, accordin folded into a rectangular configuration. This is a desirable shape from the standpoint of compact stowage. In unfolded position, the unit exhibits a significant advantage of giving extreme rigidity per unit weight. This structure could be built from thin wall tubing of approximately 1/2" diameter of 4130 steel or perhaps even special full molybdenum tubing. In open position, it would take more mechanism to fold and unfold than the previously mentioned method. Evaluation of this approach would again be based on vehicle space allocation. Its final weight would be something less than the sunflower.

Figure 18 illustrates another simple square configuration in the open position. This design would rotate into and out of closed position and maintain its square configuration. The rotation could be done around a central support post which would be telescoping or otherwise erected and mounted firmly into the vehicle. The four segments of this unit could also be circular. In closed position, this configuration would be a drum shaped unit, and would nicely lend itself to storage in the vehicle if storage area was of circular configuration. In open position, this unit would be a four leaf clover configuration. This circular shape is shown in Fig. 19

Figure 20 illustrates a single gimbal supported panel, the advantage here is that both the polar and declination axis are easily provided for. This approach would provide for simple and reliable orientation as well as unfurling. The disadvantage would be in the bulkiness of the unfolded configuration. This, in general, represents a trade-off of reliability and simplicity for volume.

The sun sensor system consists of sensing modules and control circuits. Several possible forms the sun sensor might take are shown in Figures 21, 22, and 23, and several compatible control circuits are shown in Fig 24. The sensing units on the side of the panel are normally in shade. When sunlight strikes on one side (or combination of sides) a relay is closed energizing a D. C. motor which in turn will drive the panel normal to the sun's rays. Upon reaching this position, the relay opens, stopping the D. C. motor. A sensor unit on the bottom side of the panel will be used for initial orientation. Orderly orientation will take place regardless of the landing attitude. The operating threshold for acquisition and tracking will be adjustable to values of incident radiation from 5 to 100 watts per square foot.

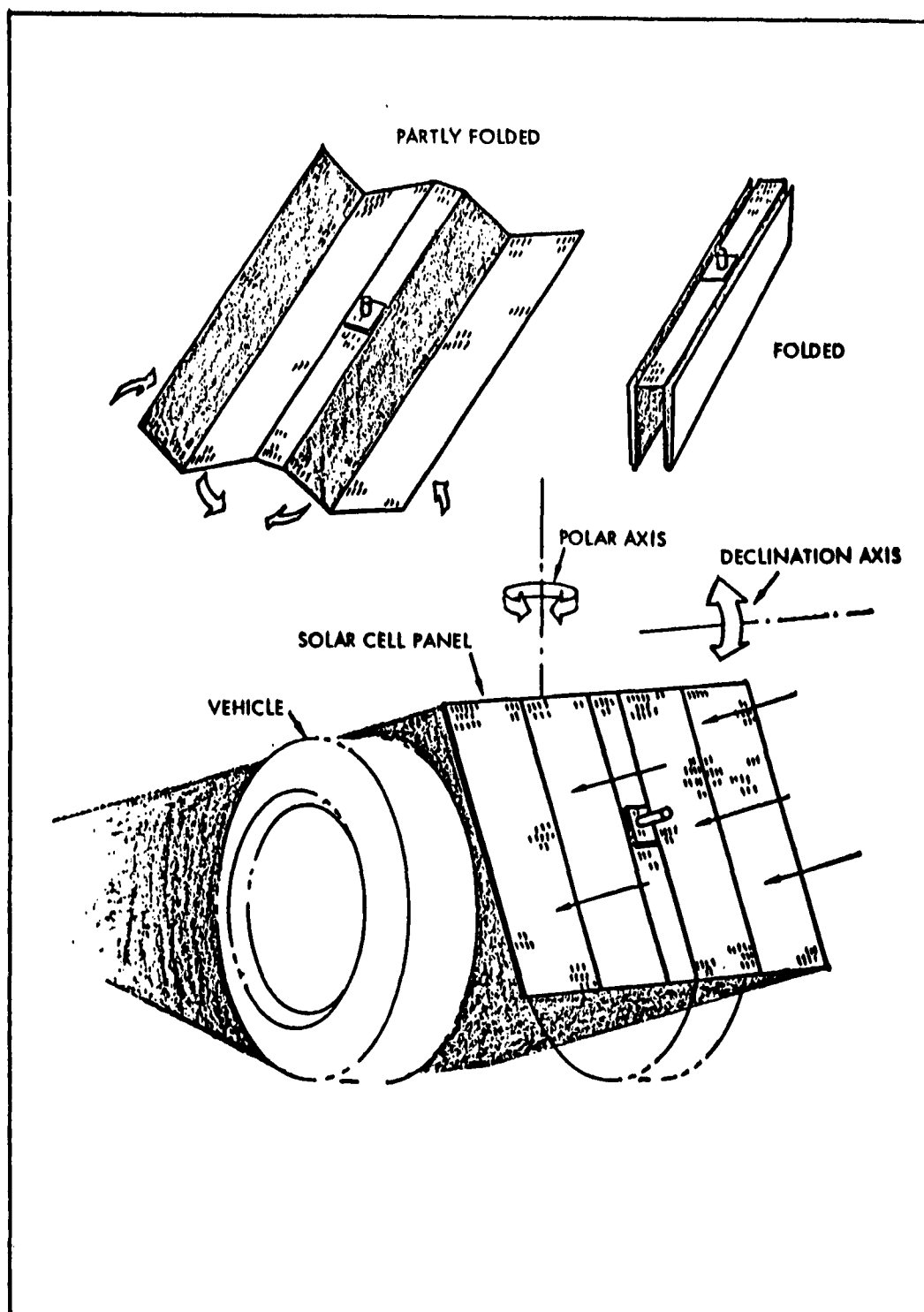


Figure 17. Solar Cell Panel

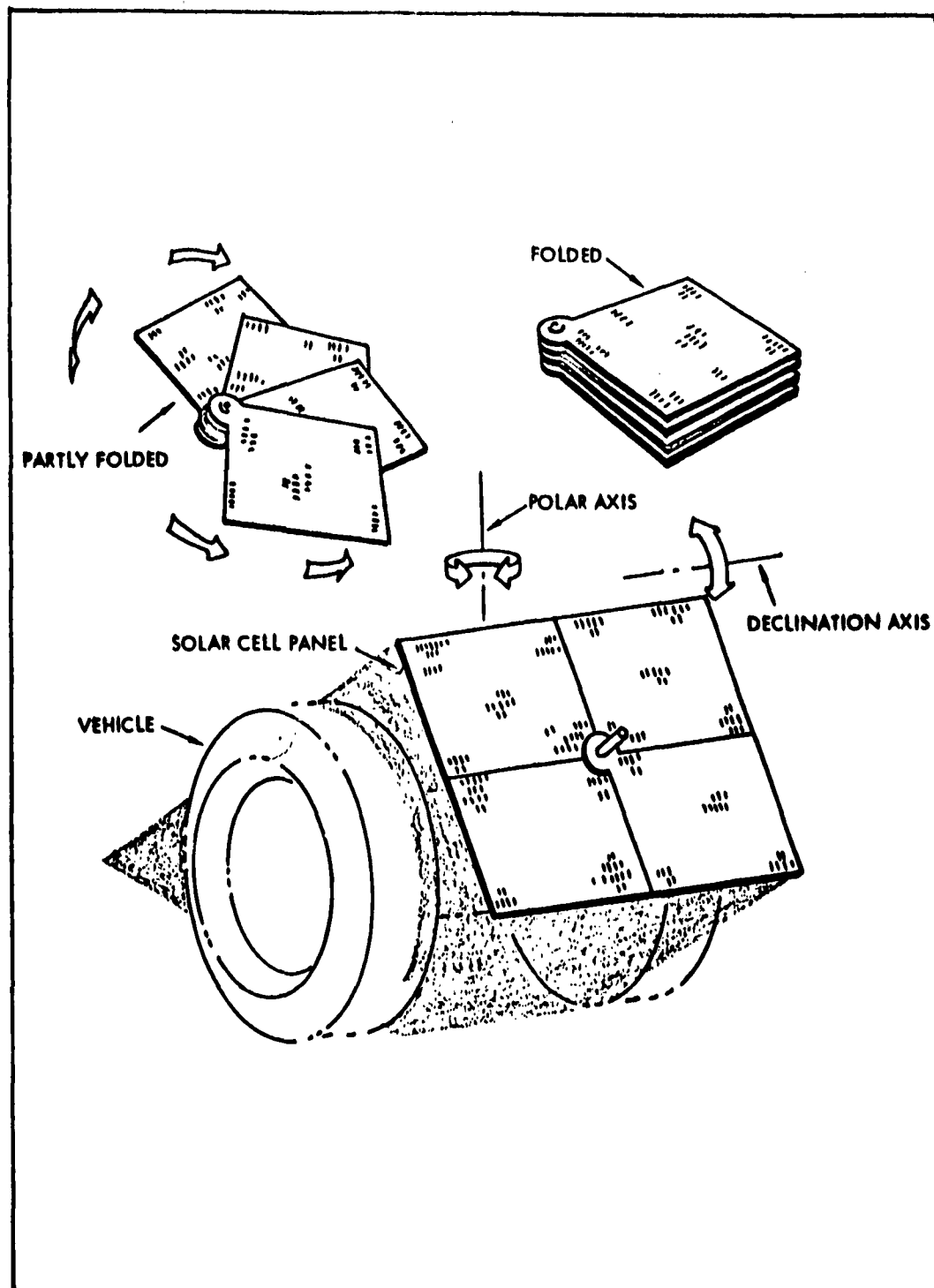


Figure 18. Solar Cell Panel

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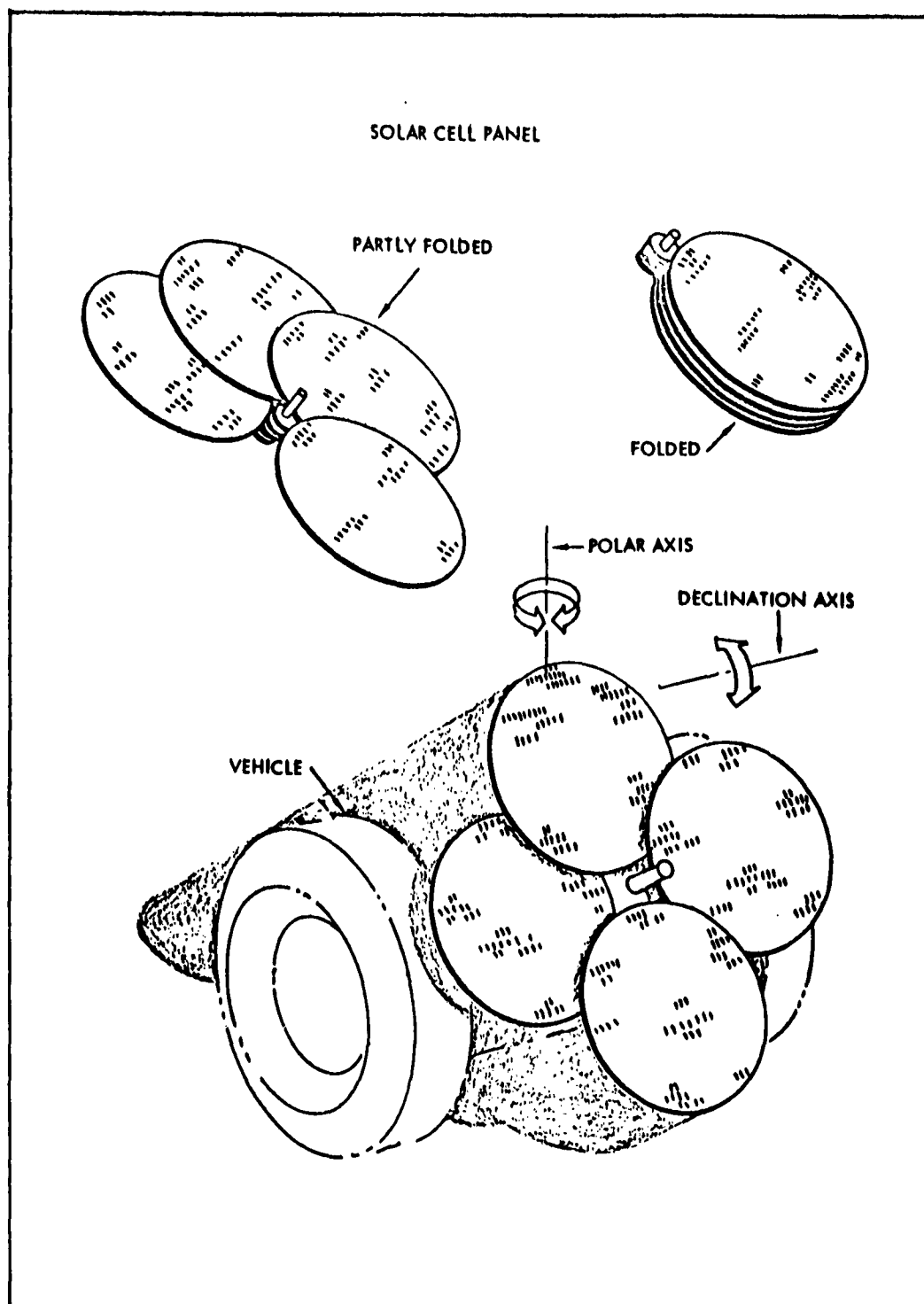


Figure 19. Solar Cell Panel

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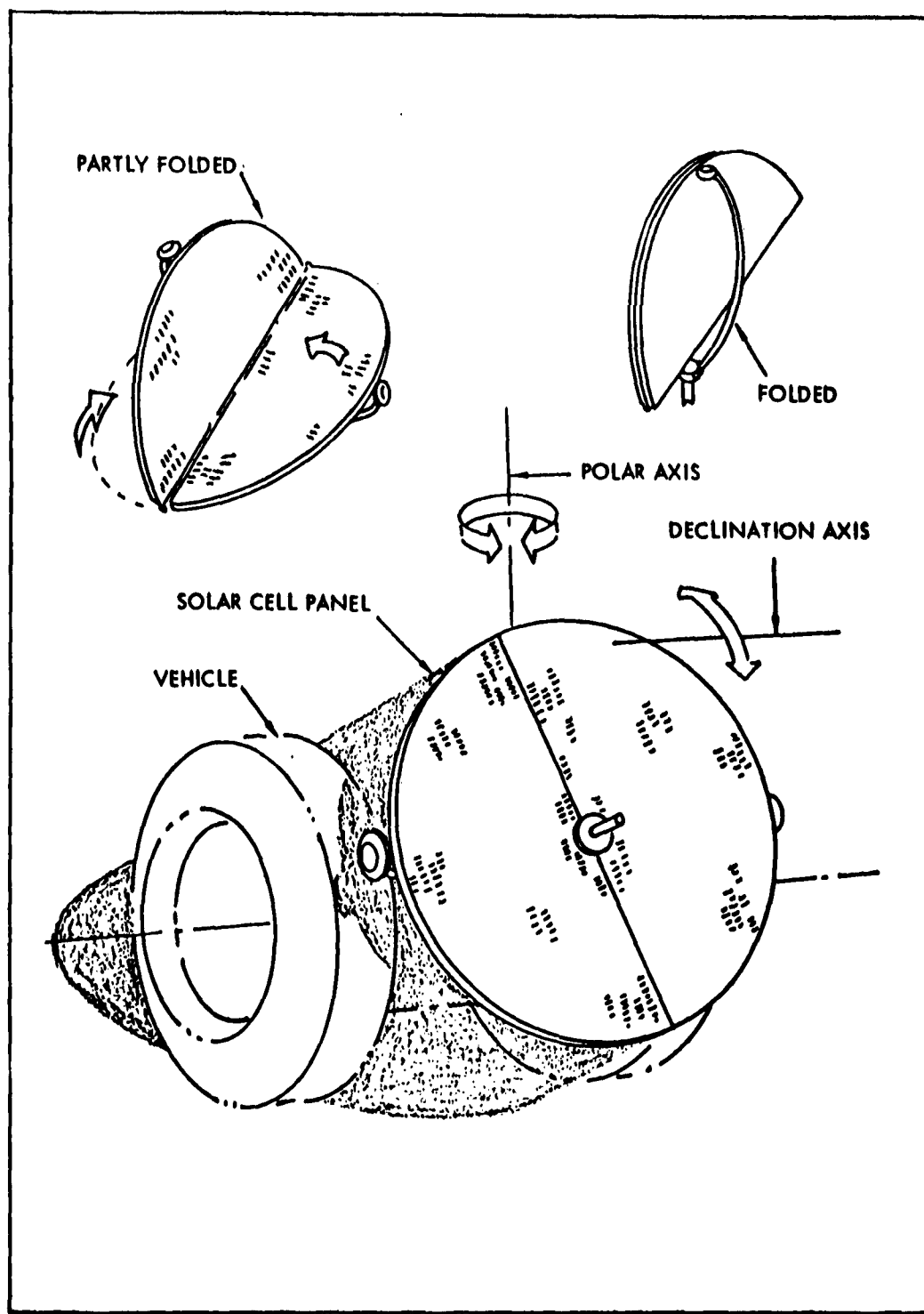


Figure 20. Solar Cell Panel

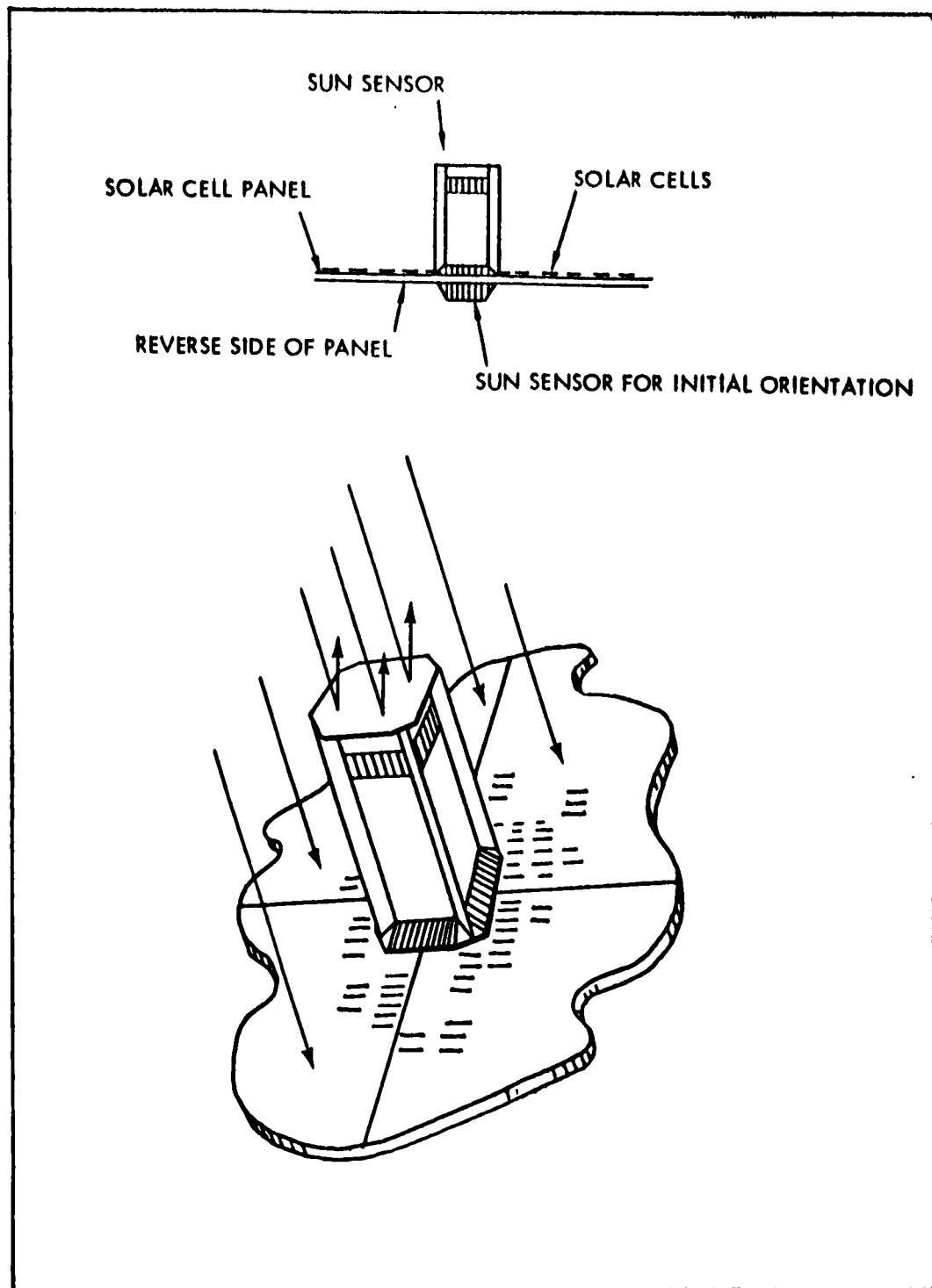


Figure 21. Solar Cell Panel Sun Sensor

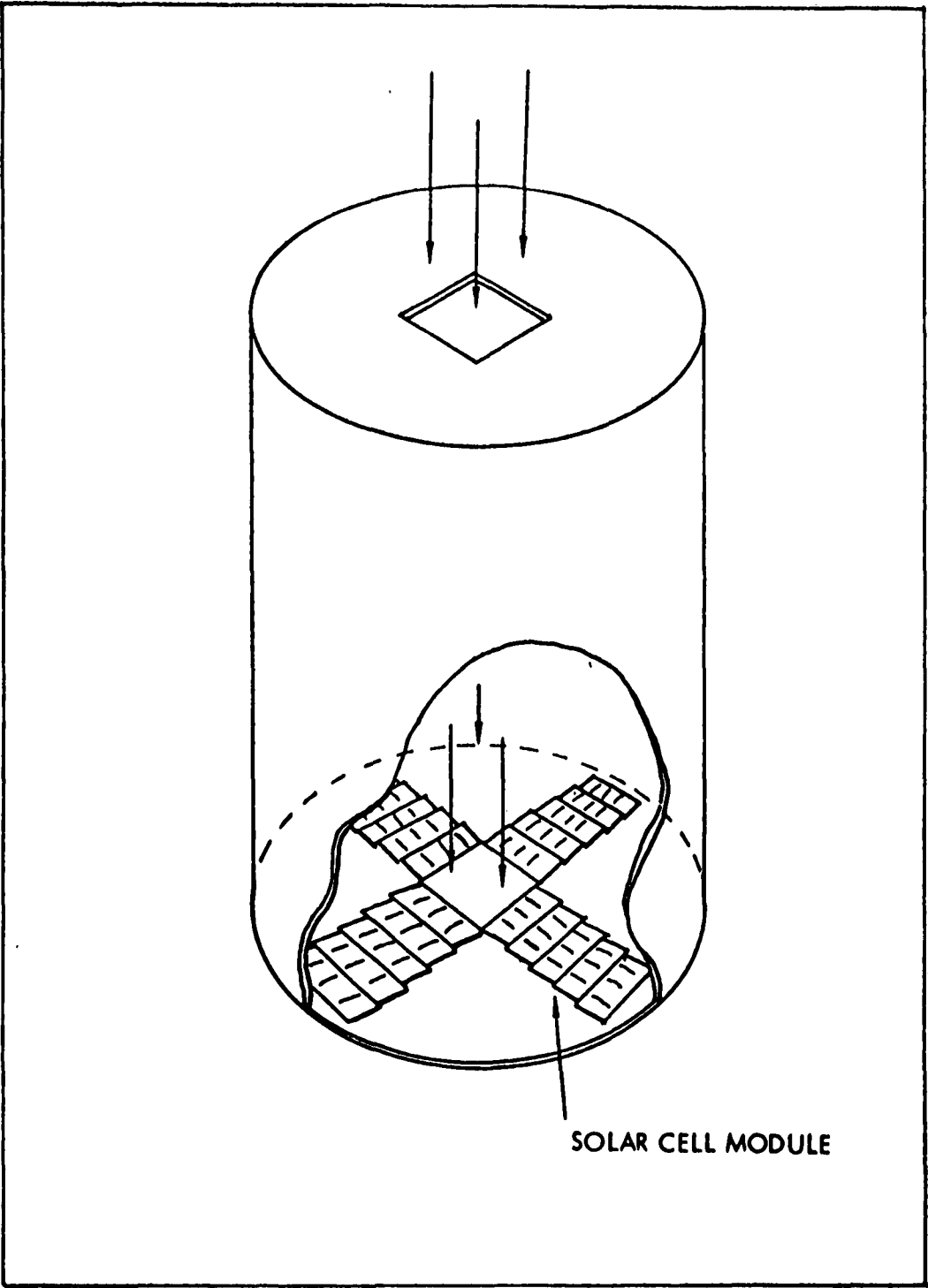


Figure 22. Solar Cell Panel Sun Sensor

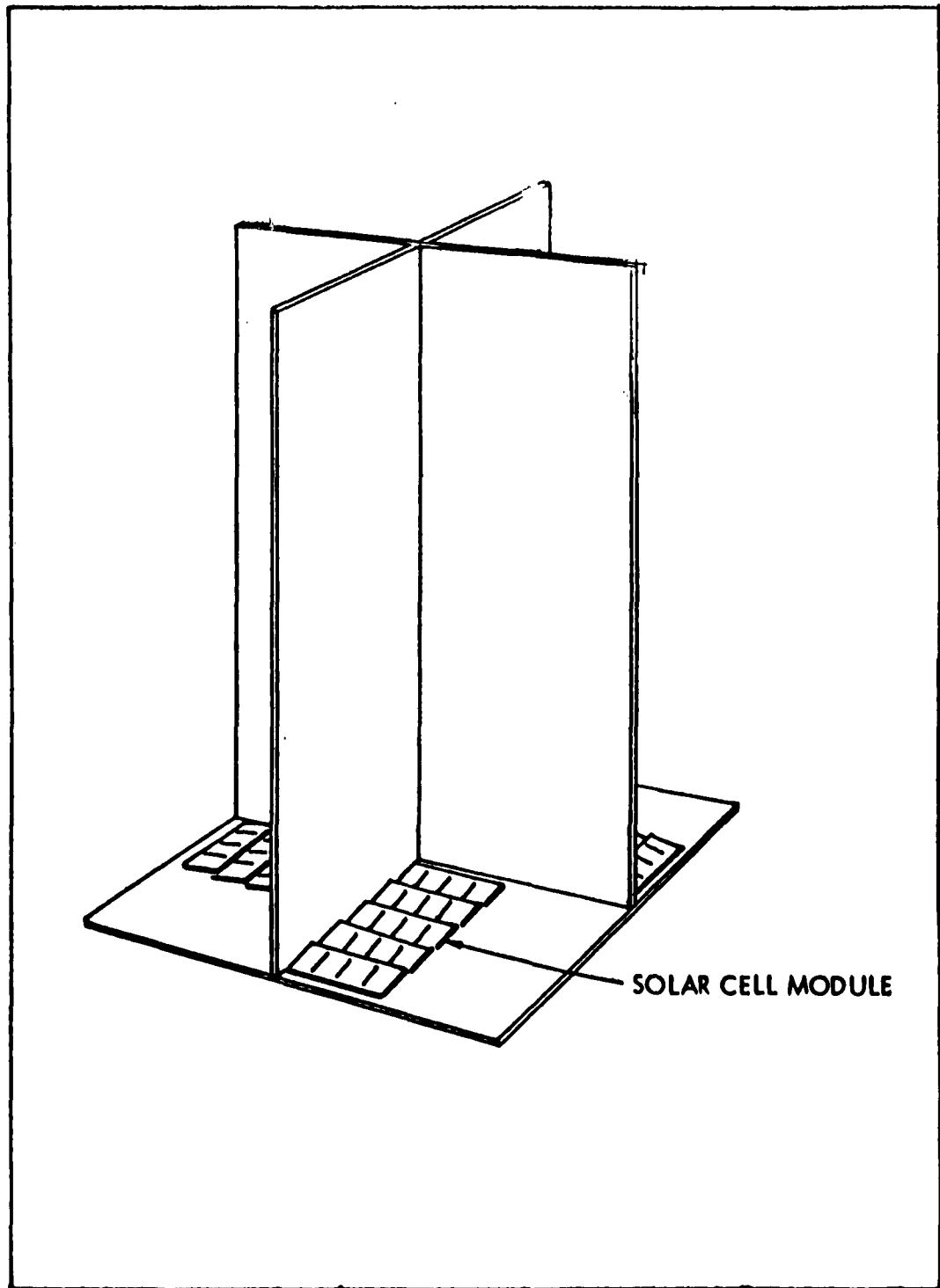


Figure 23. Solar Cell Panel Sun Sensor

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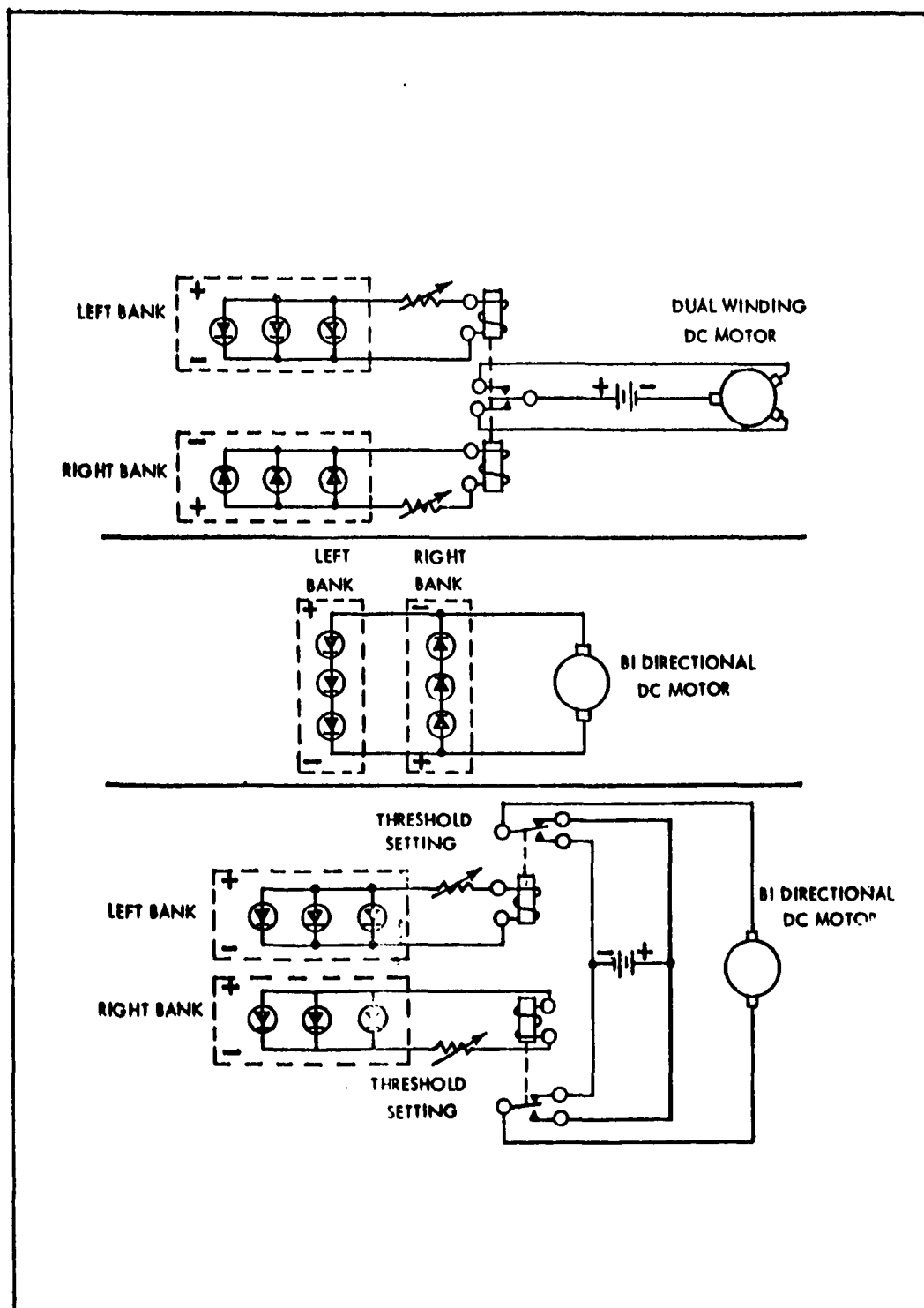


Figure 24. Solar Cell Control Circuits

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The array in the stowed configuration will be shock mounted in the vehicle to enable it to withstand the required 20 earth G's. One approach to this is to completely surround the package with pneumatic bags of suitable dimension. The bags would be inflated during launch, and deflated upon lunar landing. This could be squib or otherwise actuated. The actuation would also trigger off an orderly unfurling process.

The actual panel unfurling could be actuated by springs or a positive drive D. C. motor. A reliable latch mechanism must be devised which will insure restraint of the panel during high launch phase vibration.

Various panel erection approaches are available at this time. The objectives sought in erection are rigidity, reliability, simplicity of erection and light weight. The scissors boom shown in Fig 25 is one feasible approach. One of the significant advantages in this approach is that one of the boom arms extend the panel into a position between the vehicle and the sun, thus giving the vehicle partial or total protection from the sun rays.

To give the panel balance, a counter weight would be installed at the opposite end of the boom. This counter-balance would, of course, not consist of dead weight. The batteries, for example, could be mounted here. This approach gives the solar panel a distinct dual duty; one of furnishing power and another of aiding in thermal control.

If the vehicle is designed so that the scissors arm is not feasible, the same principle could be incorporated into a telescoping arm as shown in Fig. 26. The telescoping arm could be cable and pulley activated.

The solar cell assembly design can be handled in one of three ways. The first is the conventional flat pattern approach; the second is a reflective channel concentrator and the third is the waffle concentrator approach. Both channel and waffle concentrator reduce the number of cell with consequent reduction in cell cost but both increase panel area. With channel concentrators, the panel area is increased about 20% while with waffle concentrators, an area increase in excess of 100% is to be expected. Since such a drastic area increase would grossly complicate folding and erection problems, the waffle approach cannot be considered here.

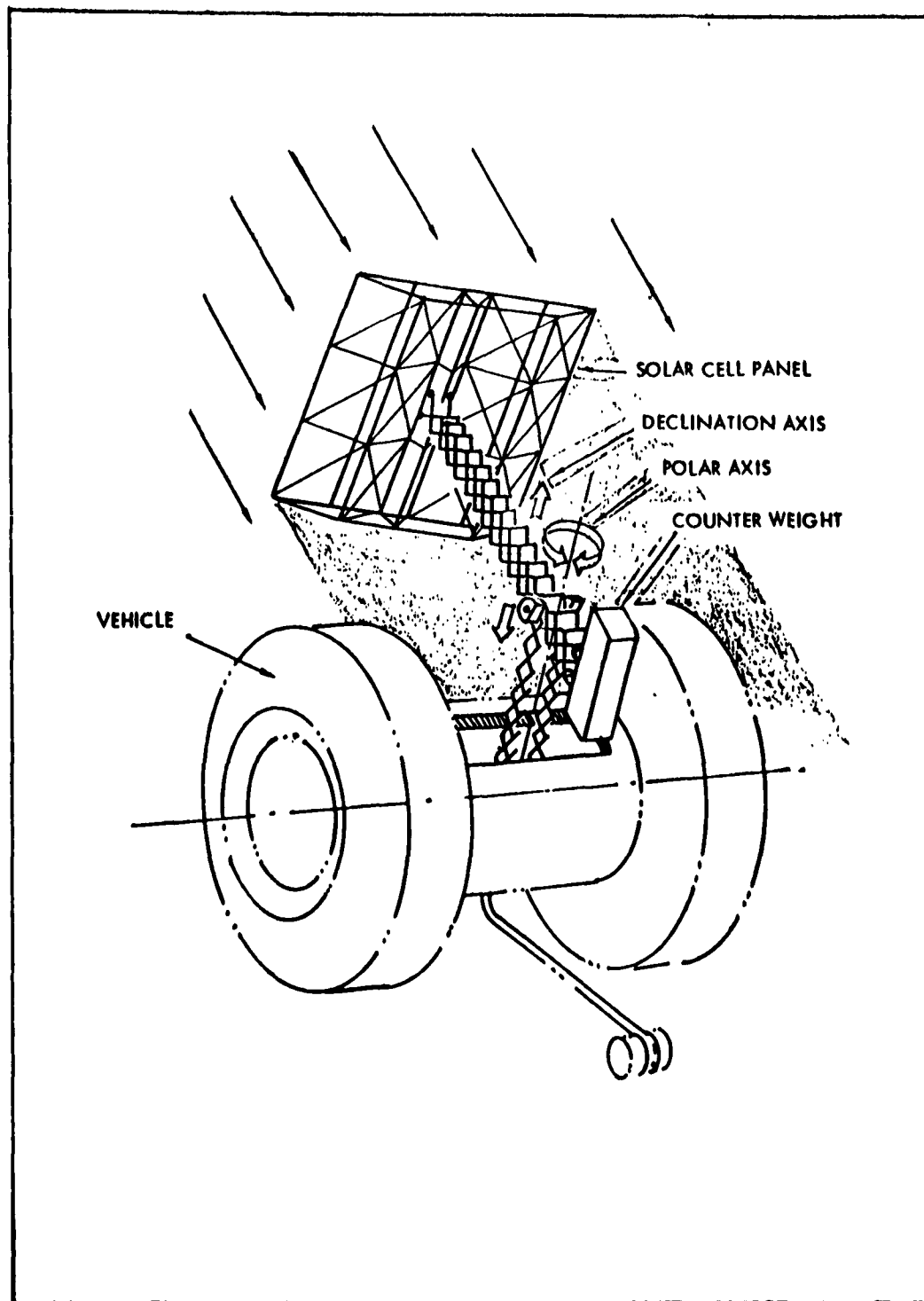


Figure 25.

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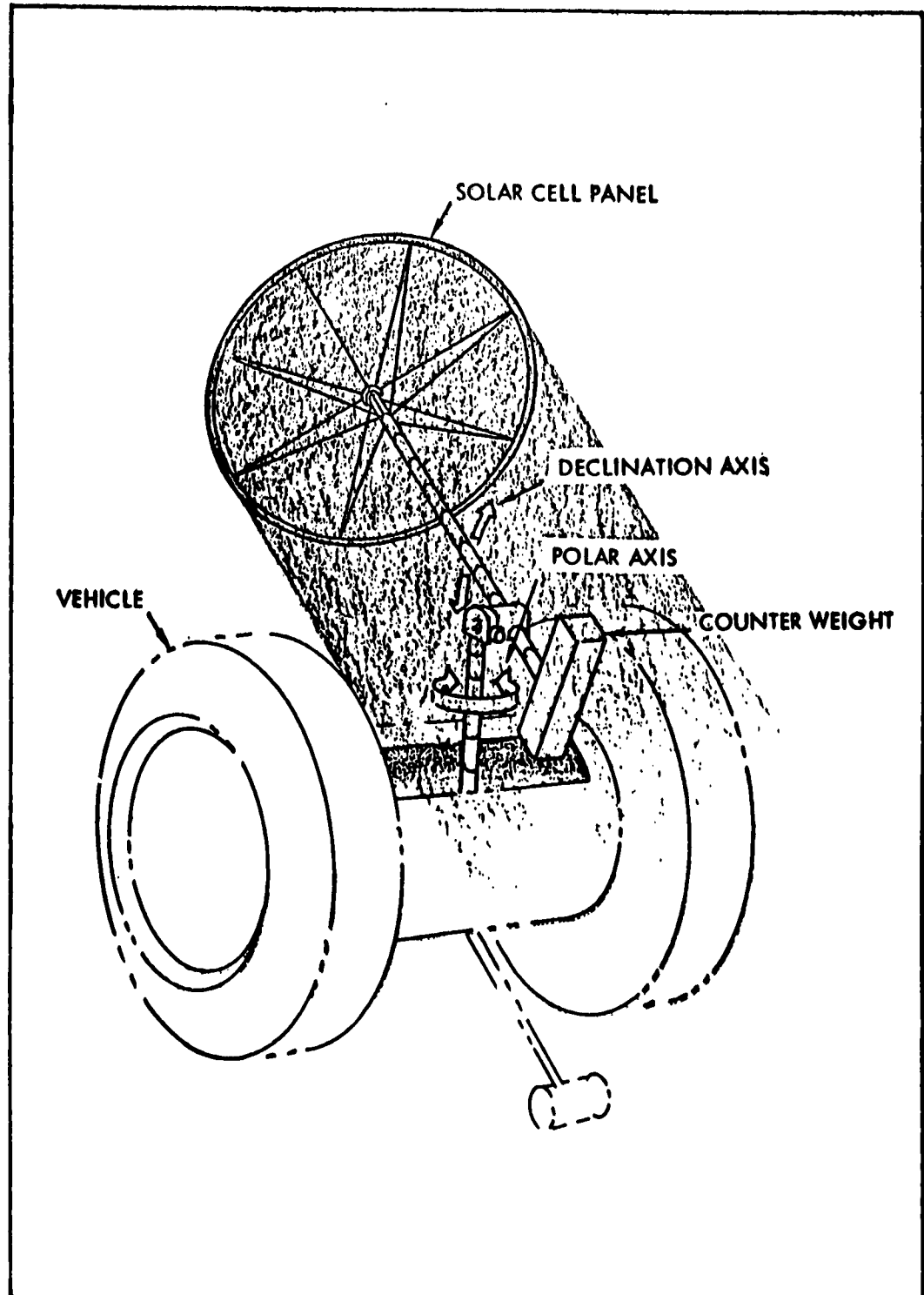


Figure 26.

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A dual servo system has been investigated as a solution to the orientation problem. This system would consist of a conventional solar cell actuated servo system to correct both the polar axis and the declination axis for the relative sun movement, while a separate gravity actuated servo system would correct for lunar terrain irregularities. The desirable objectives are to conserve the solar power and limit its use for sun position correction only. The gravity servo will be used to correct for terrain changes. An important point emerges here. If the terrain were highly irregular, array correction could require more power than that produced by the cells. If, however, the terrain does not exceed grades of more than 15 degrees and if proturbances larger than 10 centimeters are not encountered as specified in NASA technical release no. 34-159, then the gravity servo may not be needed. In this case, a simple pendulum system would suffice.

The gravity servo is basically a gimbal system with two degrees of freedom. The solar panels are counter-blanced by the battery pack or other heavy components. The polar and declination correction servos are located above this gimbal. Solar power is used to correct for sun position only. Two basic approaches are illustrated in Figures 27 and 28.

As an alternate approach to the oriented solar panel a fixed array may be utilized to give the necessary power. This fixed array has a number of advantages over an oriented panel.

1. Higher reliability due to the elimination of sensors, servos, linkages, etc., is obtained.
2. No extra power is needed for orienting mechanism. This applies to solar cells as well as reserve battery power.
3. Elimination of orienting structure is possible, thereby allowing utilization of space to other vehicle functions.
4. A fixed solar array allows for easier design of other vehicle functions as its position is predetermined.
5. A more compact vehicle design may be used as no allowance for panel movements and clearances are necessary.
6. It provides predetermined shading to vehicle.

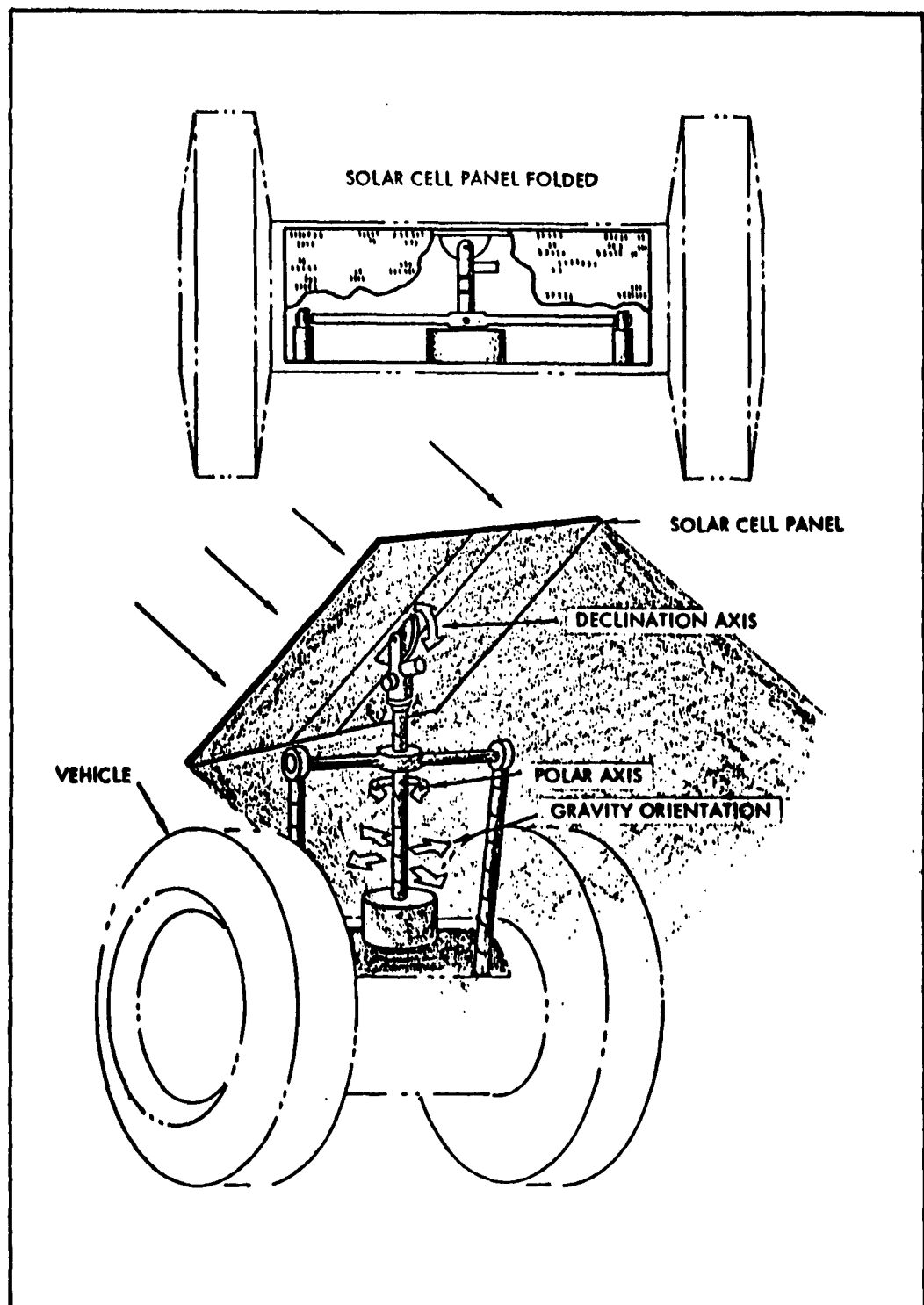


Figure 27.

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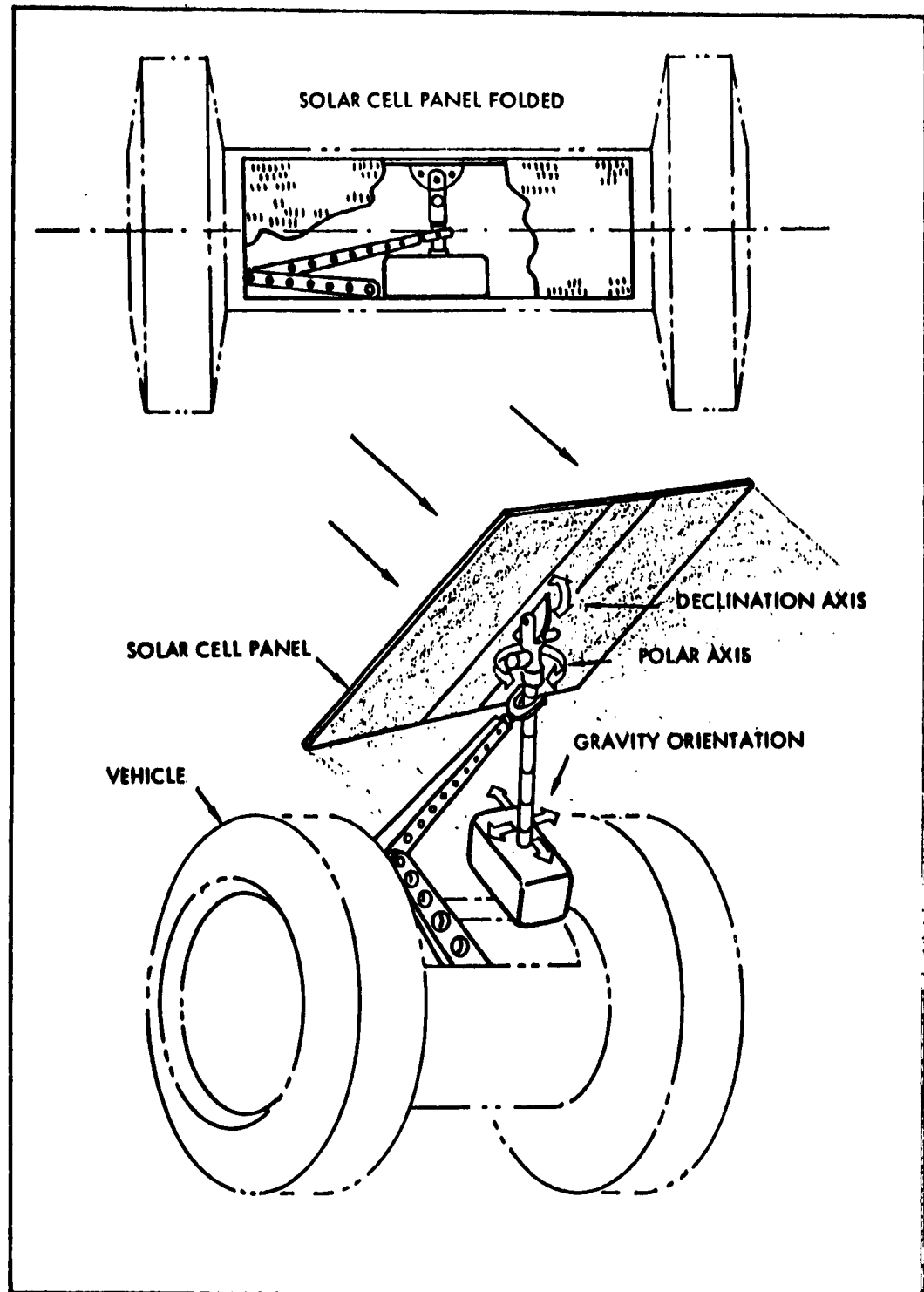


Figure 28.

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7. It provides better cooling to solar cell array due to larger surface exposed to space.
8. Increased system reliability is possible with redundancy of solar cells.

In the section following, on the 300 lb vehicle, two power system designs are presented. One is for an oriented and one for a non-oriented array. The comparison is quite interesting.

D. Power System Design Parameters - 3000 lb Vehicle

In order to approximate the power required for mechanical actuation of the solar panel, certain assumptions were made:

1. From the specifications the vehicle will cover up to 534 miles in a period of one lunar day.
2. It is assumed that the vehicle will, even under good terrain conditions, encounter a proturbance which will necessitate maneuvering 2 times in every mile.
3. It may be assumed that it will take 180° maneuver to circumvent a proturbance, etc., and the average time period will be 1 minute for the maneuver.
4. The vehicle then will be provided with servo power for 1068 min. (534 x 2).
5. It is approximated that about 200 watts will be consumed by the servo system during maneuvers. Therefore, 3600 watt hours of power will be consumed for maneuvering in one lunar day.
6. The duty cycle would be 5% based on the following figures.

Two maneuvers per mile gives two minutes per mile of maneuvering time. At 1.5 miles per hour the vehicle covers one mile in 40 minutes. The fraction of time spent maneuvering is then 2/40 or 5%.

7. Since 3600 watt hours are to be charged in 336 hours, 10.7 watt hours (3600/336) of additional power must be provided

from the solar panel. The battery loss factor of 20% necessitates additional 12.85 watt hours capacity.

8. Additional cells needed to compensate for these losses are twelve, five cell, shingles in series, connected 8 in parallel, or a total of 96 shingles.
9. Power requirements for azimuth or declination servo will be less than 5% of the polar servo and, therefore, will be discounted.
10. Any other power used once or a few times only is discounted as it is covered by safety factor in polar servo.

The additional power required due to electrical power distribution and regulation losses is given below. According to the specification 135 watts of power is to be regulated.

1. The loss due to regulation is 40% or 54 watts loss.
2. Continuous loss is 1.93 amps (54/28).
3. This will be compensated for by a series of twelve 5 cell shingles wired 37 times in parallel (twelve 5 cell shingles = 28 volt .052 amp) 37 in parallel = 28 volt, 1.92 amp. 54 watt. This is a total of 96 shingles or 480 cells.

The final design parameters for the 3000 lb vehicle power system using oriented arrays is shown in Figures 29 and 30. Figure 29 shows what can be done with existing state-of-the-art 1961 equipment and is presented to give the data in Figure 31 more meaning. Figure 30 shows what can reasonably be expected in 1965. The data in these two figures is conservative on the whole.

NiCd batteries have been chosen for the large vehicle. The large number of charge discharge cycles anticipated clearly favor NiCd. Improvement in AgZn batteries extrapolated to 1965 is indicated in Figure 30. The substantial reduction in weight with increased high discharge capabilities will be very welcome if this improvement can be realized.

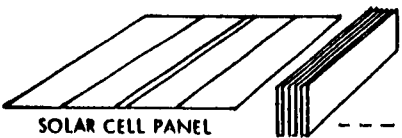
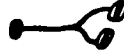
		I-VEHICLE 3000 lbs		II-VEHICLE 300 lbs	
		WT lbs	VOL IN ³	WT lbs	VOL IN ³
1.	 SOLAR CELL PANEL	300	35000	28	1700
2.	 SENSOR	4	45	4	45
3.	 ORIENTING MECHANISM	25	30	20	20
4.	 ORIENTING STRUCTURE	75	2700	20	350
5.	 POWER REGULATOR	16	300	6	225
6.	 STOWAGE AND UNFURLING MECH.	45	700	15	375
7.	 BATTERIES	I-VEHICLE 1000 W-HRS	II - VEHICLE 500 W-HRS	110	181
		5000 W-HRS	1000 W-HRS	550	905
		10000 W-HRS	2500 W-HRS	1100	1810
8.	 BATTERY CASE	10% OF BATTERY WEIGHT			
9.	 BATTERY LOUVER THERMOSTAT	50	1000	700	200
10.	 CIRCUIT CONNECTIONS	12	700	5	300
11.	 N ₂ - Cd CELLS 72.5% DISCHARGE DISCHARGE RATE 1 HR				
TOTAL WEIGHTS					
I-VEHICLE		II - VEHICLE			
1000 W-HRS		500 W-HRS		648	40656
5000 W-HRS		1000 W-HRS		1132	41380
10000 W-HRS		2500 W-HRS		1737	42285
				169	3296
				229	3396
				413	3668

Figure 29. Design State of the Art Best Data 1961

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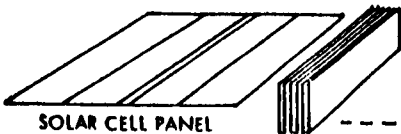
		I-VEHICLE 3000 lbs		II-VEHICLE 300 lbs	
		WT lbs	VOL IN ³	WT lbs	VOL IN ³
1.	 SOLAR CELL PANEL	250	35000	13	1700
2.	 SENSOR	2	45	2	30
3.	 ORIENTING MECHANISM	15	25	7	15
4.	 ORIENTING STRUCTURE	50	2500	9	300
5.	 POWER REGULATOR	11	250	6	150
6.	 STOWAGE AND UNFURLING MECH.	30	600	7	250
7.	 BATTERIES	I-VEHICLE 1000 W-HRS	II - VEHICLE 500 W-HRS	38	570
		5000 W-HRS	1000 W-HRS	190	3000
		10000 W-HRS	2500 W-HRS	380	5700
8.	 BATTERY CASE	10% OF BATTERY WEIGHT			
9.	 BATTERY LOUVER THERMOSTAT	6	120	10	40
10.	 CIRCUIT CONNECTIONS	5	500	2	150
11.	 Ag-Zn CELLS 58.5% DISCHARGE DISCHARGE RATE 1 HR				
TOTAL WEIGHTS					
I-VEHICLE		II - VEHICLE			
1000 W-HRS	500 W-HRS	411	39610	92	2969
5000 W-HRS	1000 W-HRS	578	42040	108	3205
10000 W-HRS	2500 W-HRS	787	44740	171	4160

Figure 30. Design Objectives Target - 1965 Power System -
Packaging Weight Volume

E. Power System Design - 300 lb Vehicle - Oriented Array

The overall power systems will be similar to the 1 KW system but of course scaled down. The performance characteristics will be similar in all respects with exception to power output. Comparison tables of weight and dimension are shown in Figure 30 and Figure 31 and have the same significance as discussed above.

While five stowage configurations are considered for the 1 KW system, only two appear propitious for the 100 watt power generator. The straightforward square or rectangular panel, folded 4 times, and the round configuration folded once look best. The construction approaches would be essentially the same as those used in the larger vehicle. The shock mounting to withstand 20 earth G's would be along lines discussed for the 1 KW system.

The approaches used on the 1 KW unit for panel support and erection on a scaled down basis. Either the telescope or scissor boom approach would be used, depending on the configuration of the finalized vehicle. Counterbalancing of the panel would still likely be used. If the vehicle is in proportion to the panel, the sunshading feature will also be incorporated.

The sun sensor system will be identical to that in the 1 Kw generator.

The operational period for the large vehicle will be after 1965 regardless of accelerated booster programs. The small vehicle, however, may be operational as early as 1963. It is, therefore, appropriate to say a few additional words about the choice of batteries for this small vehicle.

Considering the reliability and weight requirements of the storage system, the choice of battery types is limited to a silver-zinc system. Figure 30 shows parameters for a NiCd system which was included for purposes of comparison.

The AgZn battery with potassium hydroxide electrolyte will survive through the lunar night and operate the following day. Tests have been conducted on batteries stored for four days at -100°F. There was no deterioration of performance. Further tests at lower temperatures and under simulated space environments are being performed for this battery type.

At the present state of the art, the AgZn cell gives 30 to 35 watt hours/lb which for a 1000 watt hour system would be 28 to 33 lbs. The 28 volt system would have a discharge rate of 35.7 amps for 1 hour.

The difficulties with these type cells is the limitations of charge-discharge cycles and the protection necessary to prevent overcharging. The overcharge protection can be easily provided to limit the recharge to 2.02 volts maximum. The present cell design is limited to an average of 30 charge-discharge cycles. The discharge would be made to 70% capacity and approximately 16 hours necessary to recharge the batteries. Assuming one complete charge-discharge cycle per 24 hour day, the two moon day operation would require about 28 charging cycles which could be met by this system.

In the development and testing stage is the Ag Cd battery system which would give a very good compromise between the high weight problem of the NiCd battery and the low recharge cycle-rate of the AgZn system. This battery is very promising and further investigation may prove it more feasible for this program.

F. Non-oriented Array

The proposed fixed solar array is shown in Figure 31 as a spherical panel of 6 foot base diameter with a 3 foot opening in the top for other vehicle functions. This panel has a total surface area of 49 square feet. By utilizing a 50% packing factor of cells, this gives a 24.5 square foot solar cell array, which would be uniformly distributed across the surface of the spherical panel. Using the above configuration, the panel will put out approximately 60 watts at sunrise and at average sun elevation of 45% will deliver approximately 100-110 watts using cells of 10% space efficiency. A much greater percentage of the total area of the spherical shell will be "looking" at cold space than at the sun. An ideal situation then exists for temperature control by radiative cooling. Preliminary calculations indicate a panel temperature of approximately 25°C; this corresponds to the rating temperature of the solar cells. Stowage of the spherical panel may be accomplished in many different ways, two of which are illustrated in Figures 32 & 33. High reliability is again indicated by the simplicity of single motion unfurling from stowage to operating configuration.

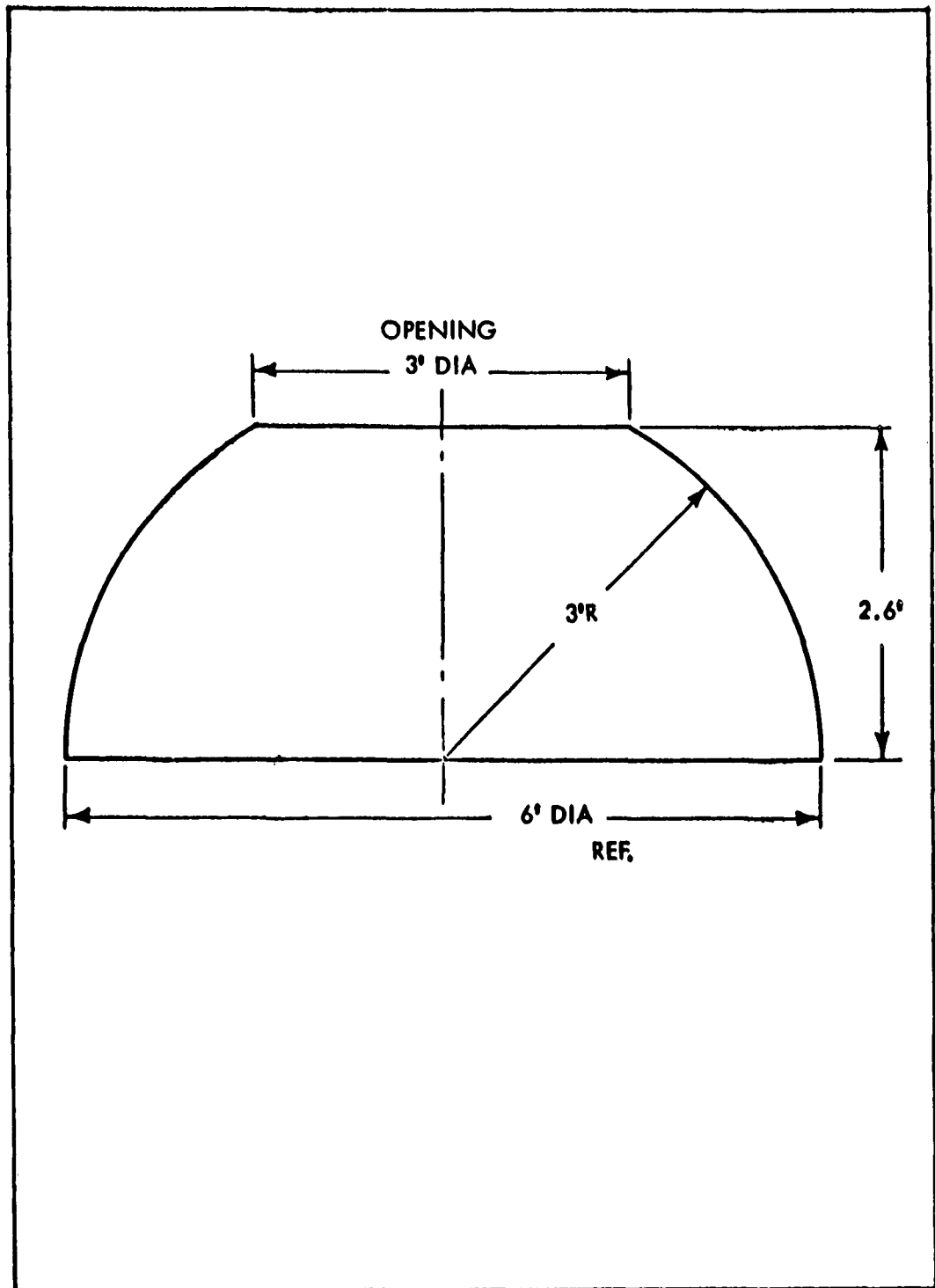


Figure 31.

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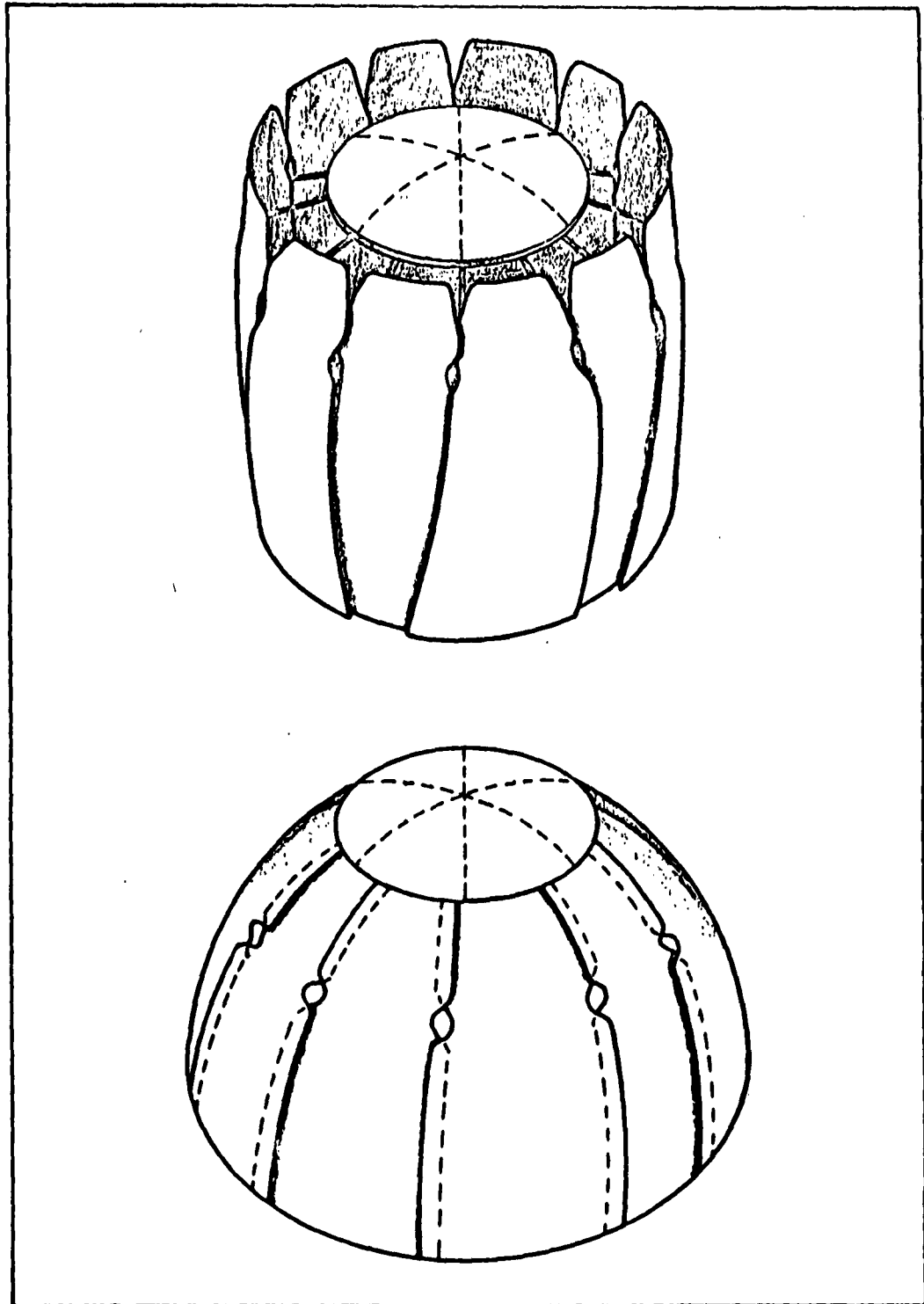


Figure 32.

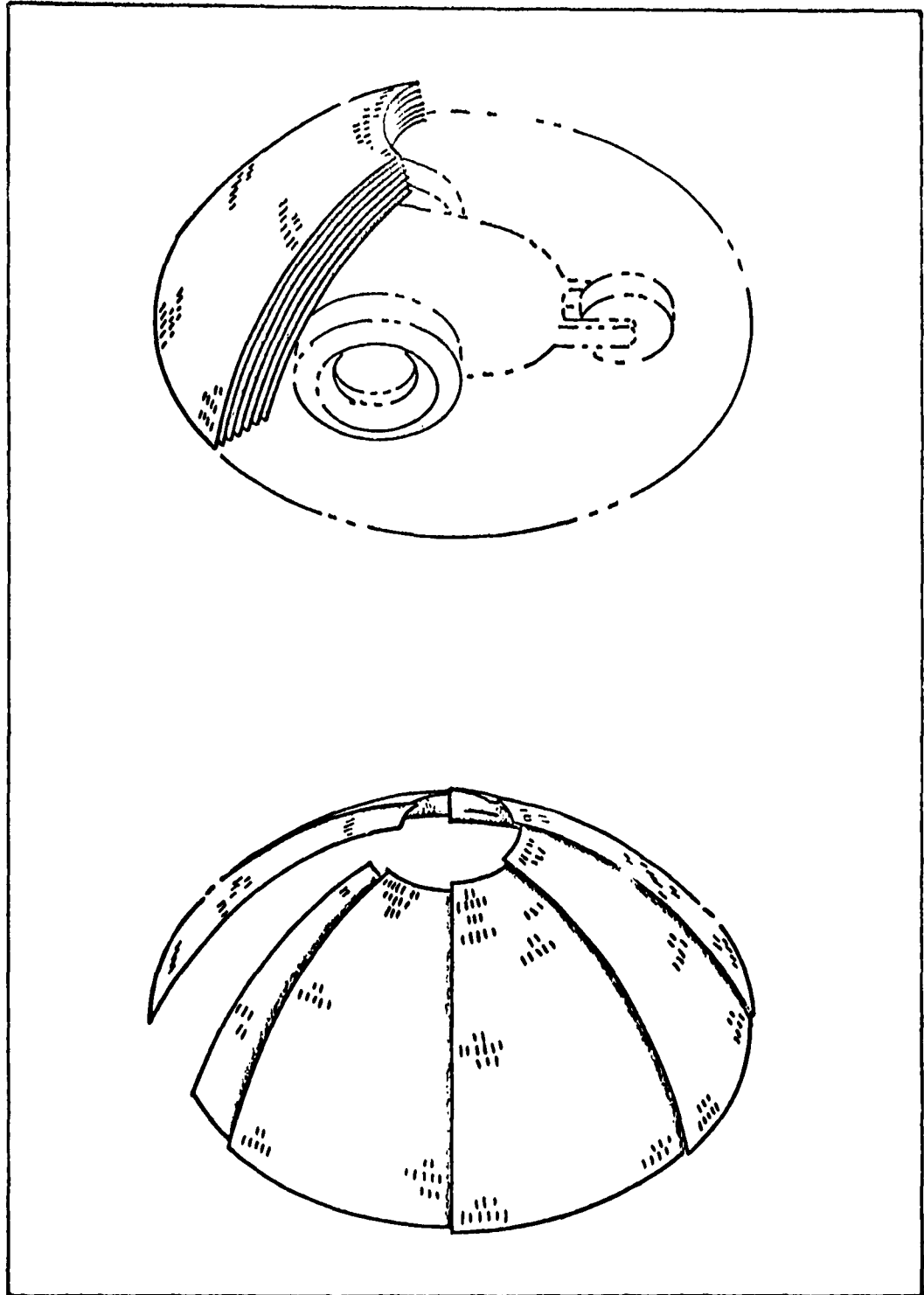


Figure 33.

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The comparison between the two systems is given in Tables III and IV. The surprising result is that both approaches yield essentially the same required equipment weight. On the basis of the extra cells provided, higher anticipated reliability and simplified environmental control, the non-oriented approach is clearly advantageous.

One additional point should be made. The fundamental concern in this report has been with the power system. When inputs from other component systems of the vehicle are available they will have a profound effect on the power system. A good example of this is seen in the Hughes design for the Static Surveyor. The power system in this case consists of solar cells, batteries, and an RTG. The RTG offers a significant advantage when the environmental control of the entire Surveyor is considered. The RTG generates a nominal electrical output, 15 watts. But the thermal output, approximately 225 watts, is of inestimable value in keeping sensitive instruments warm during the lunar night. Such a consideration could easily arise for either of the two mobile vehicles. It must be kept in mind then, that a power system integrated into a completed vehicle might take a different form from that described above.

TABLE III

Estimated Weights of Components in
Oriented Solar Panel Power Supply

	<u>Weight lbs.</u>
1. Solar Cell Panel	13
2. Sensors	2
3. Opening Mechanism	7
4. Orienting Structure	9
5. Power Regulator	6
6. Stowage and Unfurling	7
7. Battery Temperature Control	10
8. Circuit Connections	2
9. Battery Case	3
10. 1 kw/hr Batteries	<u>33</u>
Design Weight Total	92

TABLE IV

Estimated Weights of Components In
Non-Oriented Solar Panel Array

	<u>Weight lbs.</u>
1. Solar Cell Panel	45
2. Stowage & Unfurling	8
3. Power Regulators	6
4. Battery Temp. Control	10
5. Circuit Connectors	1
6. Battery Case	3
7. 1 KW/HR Batteries	<u>33</u>
	106

APPENDIX I
SOME SAFETY CONSIDERATIONS FOR
SNAP 2*

In considering the use of a nuclear auxiliary power system in space, the potential radiological hazards associated with its use must be evaluated. When anticipated by the appropriate design criteria, handling procedures, and operational limitations, it can be shown that these potential radiological hazards do not prevent the use of nuclear power in space. Throughout the design and development of SNAP 2, safety has provided the basis for many design decisions. In order to satisfy the objective of maximum possible safety of the SNAP space reactor systems, a set of safety design criteria for SNAP reactors was formulated. Compromises on the system design are necessary in order to obtain a suitable balance between the safety of the system and the operational characteristics of reliability, simplicity, and weight. The safety design criteria for the SNAP space reactor systems are outlined below.

Safety and Ease of Handling - The reactor system will be designed so that personnel can handle, install, and repair the system before launch with safety and ease.

Prevention of Accidental Criticality - The reactor systems will be designed to prevent criticality of the reactor under any condition except controlled operation.

Inherent Shutdown - The reactor system will have inherent shutdown characteristics (i. e., negative temperature coefficient and fail-safe shutdown mechanisms to prevent reactor operation before or after mission time periods.

Orbital Startup - Reactor system full power operation need not begin until after a suitably safe orbit has been established.

* Taken from "Ballistic Missile and Space Technology", LeGalley, Volume II, Propulsion and Auxiliary Power Systems.

Orbital Shutdown - After the mission has been completed and prior to re-entry, the reactor may be shut down.

Re-entry Burnup - Design of the reactor system and components will ensure the probability of high altitude re-entry burn-up and dispersal of SNAP reactor components.

The four major periods of the operational sequence, the particular safety problems of each, and their evaluation and resolution are discussed below.

Shipment and Integration Period

During the shipment and integration of the reactor into its payload and launch system, the possibility of accidental criticality and an uncontrolled power excursion must be prevented. The SNAP 2 reactor is specifically designed to allow the removal of the reactor's beryllium reflector and thus greatly increase the safety margin that must be overcome for accidental criticality. During shipment and integration the beryllium will be replaced with a thick solid aluminum jacket such that accidental immersion in water, liquid hydrogen, or kerosene cannot cause criticality. During the shipment and integration period the radioactivity remaining in the core from the factory checkout operations will have decayed to a sufficiently low level that personnel working on or around the reactor will be subjected to radiation levels below the AEC established occupational dose rate of 7.5 mr/hr.

By supplementing these physical constraints with carefully planned procedures and trained personnel, the potential of accidental criticality and personnel injury during the shipment and integration period can be even further reduced.

Launch Pad Operations Period

It is not expected to be necessary to operate the reactor at full power on the launch pad. The SNAP 2 reactor is designed such that system operation and performance can be checked out with electrical power supplying the heat in place of the reactor. If future requirements necessitate complete nuclear operation on the launch pad, it can be accomplished.

If the mission is "held" after 30 minutes of reactor power operation on the launch pad, the dose as a function of distance and decay time can be computed. If after a few hours of decay, short time access to a payload section is necessary, a gantry mounted maintenance shield is required. A possible configuration utilizing a 4 inch thick lead maintenance would reduce the dose at the payload region to about 100 mr/hr which allows several hours of payload access without excessive exposure.

If the mission is totally "scrubbed", the reactor can be removed to a shielded storage well by means of a remotely operated manipulator and gantry.

In the case of a chemical accident after reactor operation or accompanied by an accidental power excursion, preliminary analysis indicates only minor hazards outside the normal exclusion radius could lead to temporary evacuation but the combination of decay time and emergency decontamination procedures can restore the launch pad area to usefulness.

Again, even the worst case of launch pad abort during reactor power operation can be handled if appropriate equipment and procedures are made available.

Launch to Orbit Period

The significant problem during the launch to orbit period is the possible chemical explosion accompanied by an uncontrolled reactor power excursion. Only during the early stages of launch does the missile path pass over land. For this period, the hazards analysis performed for the launch pad period is applicable which indicated only minor hazards outside the normal exclusion radius. After lift off, the dispersal and dilution factors for the altitudes associated with the missile path over land will further decrease these minor hazards. The remainder of the abort conditions for the launch phase will exist over an ocean region in non-populated areas and far from islands or major cities. Thus, the potential hazards to the general populace from a personnel as well as contamination standpoint is negligible over a complete range of possible abort conditions.

Re-entry Period

In the first three periods considered, the hazards are at all times subject to control through site selection, meteorological limitations, emergency procedures, range safety, etc. The unique problem associated with reactor re-entry results from the undetectable location of re-entry and the fact that radiation is undetectable by an unaware populace.

The objective of the SNAP development program is to design for fuel element high altitude burnup and dispersal to result from re-entry heating. Preliminary calculations supplemented by arc-jet experiments indicate that this objective can be achieved. In order to evaluate the significance of contributing fission products to the earth's atmosphere through re-entry burnup and dispersal of SNAP systems, the resultant buildup of Sr^{90} has been calculated. It can be shown that the re-entry of one SNAP 2 system each year after one year of operation will, after 60 years, result in an equilibrium Sr^{90} concentration in the earth's atmosphere that is about 1/240 of the level then existing from bomb testing prior to 1960. Or, in other words, SNAP 2 systems could be employed at the rate of 240 per year for the next 60 years and only contribute an amount equal to the Sr^{90} level remaining then from the bomb testing prior to 1960.

Until complete re-entry burnup and high altitude dispersal have been demonstrated, there exists an immediately available solution to the hazards associated with the intact re-entry of a SNAP system. The problem can be solved by allowing sufficient time for radioactive decay such that intact re-entry does not constitute a radiological hazard. This decay time is achieved by limiting the use of SNAP systems to orbital altitudes which have the requisite orbital lifetime for decay. This approach must be supplemented by orbital startup of the system. This capability, which is a SNAP 2 development objective, allows a complete safety appraisal of the orbit prior to system startup and fission product generation. It can be shown that orbital lifetimes beyond 300 years or about 600 miles for a typical large vehicle, lead to negligible dose rates. Therefore, use of SNAP 2 in orbits of greater than 300 years duration coupled with orbital startup results in no re-entry radiological hazard.

In conclusion, it has been shown that radiological hazards do not significantly limit the use of SNAP 2 in space. The use of high altitude orbits and orbital startup eliminate the re-entry hazard by allowing long decay times prior to re-entry.

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