ELECTRIC POWER FOR THE SCIENTIFIC
EXPLORATION OF THE SOLAR SYSTEM

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Electric power systems considered to be applicable to the unmanned scientific exploration of the solar system are examined. Typical solar cell–battery systems used on recent missions such as Ranger, OGO and Mariner are described. Representative load requirements and the need for power management and conditioning are reviewed, Endurance, reliability and weight are seen as the more important factors in power source selection. As a consequence, solar cells and batteries can be expected to play a dominant role in the range of interplanetary missions from the planets Venus to Mars, For missions to Jupiter and beyond, isotope thermoelectric (or thermionic) generators are expected to find application, Solar thermionic sources are seen to have potential for missions to Mercury and close–in to the sun.

INTRODUCTION

A comparatively old device, the battery, has been teamed with a real newcomer, the silicon solar cell[1] (1954) to tackle the difficult job of supplying electric power in space,

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In the seven years since Explorer I, the two, used singly or together, have been able to handle all of the space flights undertaken in this country except for four experimental missions which made use of isotope thermo-electric generators. As evidence of the yoeman service being turned in by the solar cell-battery systems, there were 19 spacecraft listed by NASA as "still transmitting" at the end of 1964. The total accumulated operating life for these spacecraft had reached about 25 years with a mean life in excess of 1.3 years. Four had been operating for over two years, while nine had been operating for over one year.

With seven years of flight experience and an impressive record of reliable service to build on, the solar cell-battery electric power system can be expected to play a dominant role in the unmanned exploration of the solar system.

Isotope thermoelectric (or thermionic) generators are expected to see service on missions where the availability of solar energy is limited, or uncertain. A more recent newcomer, the solar thermionic power source may prove to be uniquely useful for missions to Mercury and close-in to the sun.

This paper will examine these power sources together with the equipment required to manage and condition the power supplied to the loads.

SYSTEM ELEMENTS

The elements of a typical space electric power system are shown in Fig. 1. Power sources such as solar cells, isotope thermoelectric and solar thermionic are believed to be the most applicable to the unmanned scientific explo-ration of the solar system. Combinations of these sources may be used for different phases of the same mission, e.g. solar cells may be used for the interplanetary and orbital phases with the landing capsule or vehicle powered from an isotope thermoelectric source.

Energy storage in the form of primary (one-shot) and secondary (rechargeable) batteries is normally required to accomodate peak loads or to power the complete spacecraft when the solar power source is unable to receive solar energy such as during launch and maneuver, or when the spacecraft is in the shadow of a planet, as would be the case for many orbiting missions. Thermal energy storage could theoretically be used in place of rechargeable batteries in conjunction with solar thermionic power sources. However, no practical thermionic converters with integral thermal energy storage have yet been fabricated and any application of solar thermionic power sources in the near future would undoubtedly rely on batteries for energy storage.

Power management elements include operations such as automatic on-board selection between solar cell and battery power, battery recharging, protection of the power source and energy storage devices in the event of a failure in some part of the power conditioning equipment,

The most efficient utilization of the power source and energy storage capabilities is usually achieved when the power conditioning requirements are kept to an essential minimum. Direct energy conversion devices such as photovoltaic (solar cells), thermoelectric and thermionic have an unregulated DC voltage output. Typically, the price for DC to DC voltage conversion with regulation is 10 to 20 percent of the power delivered by the source. The price increases to approximately 40 percent for DC to AC inversion with regulation, i.e. about 40 percent of the unregulated DC power supplied by the source is typically used in the power system to provide the remaining 60 percent as a regulated AC output.

The electric power system used on the Ranger Block 3 Lunar TV series of spacecraft is shown in Fig. 2. Systems essentially identical to this successfully powered the spacecraft bus on Rangers 6 and 7. The Ranger 7 mission was an outstanding success in every way, taking 4300 close-range lunar photos[2] to complete a near perfect mission. The power source consists of 9,792 individual 1 x 2

centimeter boron diffused (p/n), silicon solar cells structurally supported on two panels and electrically interconnected in six diode isolated, redundant, series-parallel sections. The array (both panels) power capability is nominally 210-watts at 28-volt DC with 68 solar cells connected in series and 144 in parallel. During launch and midcourse maneuver when the solar panels cannot be oriented to the sun, power is supplied from two silver–zinc hatteries. Ten electronic modules contain the components and circuits required for power management and conditioning.

USE OF ELECTRIC POWER

The use of electric power on a typical[3] (Mariner 4) planetary spacecraft is shown in Fig. 3. Scientific instruments use 20 percent of the nominal unregulated power source capability. In this example, power is derived from a DG source and supplied as 400 cps and 2400 cps AC to the science loads. Only 12 percent actually reaches the scientific instruments. The remaining 8 percent is the price paid for inversion (DC-AC) and regulation.

Starting with the nominal array capability, which is a computed rather than a measured quantity, a 10% derating is applied as a hedge against possible error in the terrestrial measurements or mathematical extrapolations.

This accounts for the 10 percent listed as uncertainty in Fig. 3. Additional insurance against power shortage resulting from array degradation or component failure is provided in the form of redundancy. The power source design is such that 7/8th of the array should be adequate to satisfy the total encounter load. In addition, a reserve of about 13 percent is available as insurance against flight loads exceeding those measured during ground testing. The

total margin between nominal array power and anticipated loads is about 36 percent. It would be possible for lower priority loads to use this marginal power on an "as available" basis.

In this example, telecommunications requires 28 percent of the source power capabilities or about 90 watts of unregulated power. This is considered adequate for communication over a distance of 140 million miles at a bit rate of 8 1/3 per second. At this telemetry rate, about one low resolution TV picture per 8-hour period can be obtained. For TV missions to the more distant planets it may be necessary to increase the telecommunications power by a factor of 10 or 100. It is not anticipated that the other loads would increase as rapidly, thus the distribution shown in Fig. 3 is more representative of present missions to the near planets and will differ from mission to mission depending on objectives. Part of the power supplied to telecommunications is conditioned and part is unregulated. This resulted from a desire to integrally package the high voltage power supply with the transmitter power amplifier thereby eliminating the need for a high voltage distribution line in the spacecraft harness. In addition to the source power required for conditioning, the power systems uses about 3 percent of its regulated output for generation of phase and frequency synchronization pulses and for power switching.

The guidance and control system uses about 7 percent of the power source capability. The largest guidance and control loads normally occur during launch and maneuver when the batteries are supplying spacecraft power. Engineering telemetry accounts for the remaining 6 percent.

FACTORS IN SPACE POWER SYSTEM SELECTION

Availability was perhaps the most important factor in the selection of mercury batteries for use on Explorer I. Weight restrictions limited power level and mission duration. Solar cells were used successfully on Vanguard I to overcome the restriction on mission duration. On March 17, 1965,

Vanguard I is expected to log its seventh year in orbit with its solar cell powered transmitter continuing to send back temperature data, Weight restrictions again limited power level.

Twelve of the more important factors which are usually considered in the selection of an electric power system for space application are listed in Fig. 4. The selection process involves reaching an acceptable compromise from among these many and frequently conflicting requirements. In the beginning, batteries and solar cells were the only small sources of power available and the selection process was simple.

Today, low power (25-50 watts), isotope thermoelectric generators are available in limited quantities. However, the isotopes (plutonium 238 and curium 244) which appear satisfactory from the standpoint of hazard and compatibility with vehicle and mission restraints are expensive. In addition, it has been difficult for mission planners to assess the true availability of such isotopes.

Solar thermionic systems can be made available in essentially unlimited quantity and may be acceptable from a weight, volume, area and cost standpoint, but flight experience is lacking so that compatibility with attitude control system capabilities and the space environment has not been determined. In addition, ground test experience is not adequate to assure that thermionic converters can satisfy duration and reliability requirements.

A rough comparison of power source weight -vs- distance from the sun is shown in Fig. 5. The range of performance given is believed to be applicable for power sources from a few watts to a few kilowatts in size. Energy storage, power management and conditioning weights are normally independent of sub-probe distance and are not included,

The density (watts/ft^2) of the solar energy available at 4-40 astronomical units (A.U.) from the sun, makes it uneconomical from a weight standpoint to carry the solar

cell array required to provide even a few hundred watts.
On the other hand, mission durations to the planet Saturn,
for example, using all chemical propulsion, may exceed
3-years[4]. Today, the only power source that has demon-
strated such life capability is the solar cell. The AEC
and NASA are engaged in the development of isotope thermo-
electric (and thermionic) generators to improve life and
reliability and to reduce weight. Present performance is
about 1 watt/lb. in modules of 25 to 50 watt. The develop-
ment of larger generators using higher temperature silicon-
germanium thermoelectric couples can conceivably improve
performance to 4 watts/lb. A practical goal for the near
future may be about 2 watts/lb. with one to two years
useful life.

The isotope thermoelectric generator may be useful for
missions where the availability of solar energy is limited
or uncertain, For missions farther from the sun than Mars,
the use of solar energy becomes increasingly less desirable.
The availability of solar energy may be uncertain for such
missions as unmanned planetary, landers. For earth orbiting
missions where the use of gravity gradient or non-mass
explusion attitude control is desirable, sun orientation of
arrays or concentrators may not be practical. Thus there
appears to be a class of missions where the isotope thermo-
electric generator is uniquely useful, The quality of being
"uniquely useful" has been cited[5] as a practical test of new
power sources. Lacking such a quality, new power sources
will seldom be selected over existing and proven methods.

The availability of solar energy is practically unlimited
as the spacecraft approaches close to the sun. This would
be very advantageous if the tanperature of the solar cell
array and spacecraft could be maintained at near Earth values.

Based on current technology, $125^{\circ}C$ is assumed to re-
present an upper temperature limit for the silicon solar cell.
Some degradation may result even at this temperature. An
oriented array will reach an equilibrium temperature of
$125^{\circ}C$ at a distance of about 0.68 A.U. For operation closer
to the sun, some means of active temperature control would
be required.

Since the solar thermionic generator normally operates at heat rejection temperatures around 500-600°C and does not require direct exposure to solar energy, there is reason to believe that such a power source could operate reliably and efficiently on missions close-in to the sun. Except for an uncertainty concerning the durability of the specular reflectance of solar concentrator surfaces in the space environment, there is reason to believe that the solar thermionic source would be unaffected by the greater exposure to energetic particle radiation expected for operation close-in to the sun. For example, thermionic converters have been operated successfully for many hundreds of hours in the core of nuclear reactors in support of thermionic reactor power source development.

SOLAR CELL TEMPERATURE

The equilibrium temperature of a solar cell array similar to that used on Mariner 4 is shown in Fig. 6. The temperature would range from -200°C at about 40 A.U. (near Pluto) to +760°C at 0.1 A.U. As discussed previously, present silicon solar cell technology imposes an upper temperature limit of around 125°C.

One method for controlling the upper temperature of a normally sun oriented solar cell array is shown in Fig. 6b. The array can be maintained at 125°C by a controlled tilting so that the effective solar energy incident on the array is kept at about 300 mw/cm² as the spacecraft travels from 0.68 to 0.2 A.U. The reliability of this method may be questionable for large tilt angles, however. Other, more active, cooling methods employing radiators located at right angles to the plane of the array with liquid or gas heat transfer could conceivably be used with some sacrifice in reliability.

The use of solar cells with higher temperature capabilities than silicon is also a possibility. Gallium arsenide solar cells, for example, may possibly be used to extend operating temperature to 250°C. Unfortunately, the cost of gallium arsenide cells is higher than for silicon and the conversion efficiency that has been achieved thus far is below that for silicon, except at high temperatures where both are poor. A newer material, gallium phosphide, is being investigated for

possible operation up to 500°C. To date, only crude, low efficiency solar cells have been made with gallium phosphide and much work remains to determine feasibility.

SOLAR CELL POWER SYSTEMS

The characteristics of typical solar cell power systems that have been used on lunar, earth orbiting and planetary missions are listed in Fig. 7. All three spacecraft are designed for operation with the solar cell array oriented to the sun. Total power system weight ranges from 124 to 211 pounds. The 77-ft^2 solar cell array on the Orbiting Geophysical Observatory (OGO-I) is the largest that NASA has flown to date.

The planetary system shown in Fig. 7 is that used on the Mariner 4 Mars Mission. Achieving a low spacecraft weight was a necessary and critical design requirement for this mission. The area of the solar array was increased from 25.5-ft^2 used on Mariner 2 to 70-ft^2 for Mariner 4. The array weight on Mariner 2 was 45 lbs. This could only be increased to 70-lbs for Mariner 4. Thus it was necessary to reduce the array specific weight from 1.75 to about 1.0 lb/ft^2. This was accomplished by decreasing the weight of the structure used to support the cells from 1.25 lb/ft^2 to about 0.60 lb/ft^2 and the weight of the electrical components from 0.5 lb/ft^2 to about 0.4 lb/ft^2. The Mariner 4 specific power of about 9.5 watts/lb (1 A.U.) is the best that has been flown to date and the array capability of about 670-watts (1 A.U.) is the largest.

NASA's Space Power Technology Program is working toward further structural and electrical weight reductions which may make it possible to package and deploy large area (500 to 5000 ft^2) arrays producing 20 watts/lb and greater.

The cycle life requirements for lunar and planetary missions normally are such that the higher energy density silver-zinc battery can be used. The number of charge-discharge cycles for earth orbiting missions typically range from 500 to over 20,000 depending on orbital period and

desired mission life, The nickel-cadmium battery is usually selected when life requirements exceed a few thousand cycles.

The energy density in watt-hours per pound that can be obtained from the silver-zinc and nickel-cadmium batteries is dependent on state-of-charge, battery temperature, and discharge rate? An important factor in battery design for the Mariner 4 mission was ability to accept a trickle charge for extended periods of time (about 250-days). An important factor in the design of rechargeable batteries for orbiting applications is the rate that the battery must be recharged, For the typical temperature and discharge rates experienced on the lunar and planetary missions, the silver-zinc battery has an energy density of about 40 wh/lb.

The nominal array power as a function of days from launch is shown in Fig. 8 for the Mariner 2 Venus Mission[7]. Planet encounter occurred 109 days after launch and radio contact was lost 20 days after the Venus flyby was accomplished. The nominal power (310-w) available at encounter was well in excess of the cruise load (152-w) requirements.

The nominal array power as a function of days from launch is shown in Fig. 9 for the Mariner 4 Mars mission. Planet encounter should occur 228 days after launch when the spacecraft is 232.4×10^6 KM from the sun. Communication must be maintained for about 10 days after encounter to return the complete sequence of TV pictures. The nominal array power at launch (684-watt) is more than twice that at Mars encounter (320-watt).

It may appear that the arrays have been oversized for both the Mariner 2 and 4 missions. However, there is an unacceptable risk associated with a design which attempts to use the nominal array power on a "must" Basis. A derating of at least 20 percent is considered necessary to establish acceptable confidence in power source success. Such a derating philosophy would need to be applied to isotope thermoelectric or solar thermionic power sources to assure adequate power margins,

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SOLAR CELL ARRAY

The cross section of a typical solar cell array is shown in Fig. 10. Solar energy is incident on a transparent cover glass which typically ranges in thickness from 6 to possibly 30 mils depending on the amount of shielding from radiation damage considered necessary for the mission. Lunar and planetary spacecraft typically employ 6-mil cover glass while spacecraft that spend appreciable time in the earth's radiation belts may use 30-mil quartz covers8. The cover glass usually has an antireflection coating on the surface toward the sun and a multilayer interference filter for reflecting the sun's ultraviolet energy on the side nearest the solar cell. The glass cover provides protection from energetic particle radiation, from erosion and solar ultraviolet energy which could darken the adhesive.

A transparent adhesive is used to bond the cover glass to the silicon solar cell. This adhesive should have sufficient strength to hold the cover on the cell through the launch vibration, yet allow the easy removal and replacement of damaged cover glass. It should have good transmittance to solar energy, yet provide an effective conduction path for heat transfer from the cell to the cover glass for re-radiation. It should be insensitive to ultraviolet and energetic particles. It should be flexible at low temperatures (-120°C) yet retain satisfactory bond strength at 125°C. Transparent silicone adhesives are typically used.

The design of the "n" and "p" contact bus bars should provide a strong, low resistance electrical interconnection between solar cells with minimum stress on the "n" and "p" region ohmic contacts during assembly or subsequent thermal shock. Bus bars have been formed from thin sheets and small diameter wires of copper, kovar and molybdenum.

Copper may cause unacceptable stress during thermal shock, while kovar has been found satisfactory from a thermal shock standpoint but has a high electrical resistance and is magnetic. The solderability of molybdenum poses a problem unless it is nickel plated.

The "n" and "p" region ohmic contact strength and resistance place restrictions on maximum operating temperature and on the time-temperature control of soldering processes. Best results are usually obtained with short time at soldering temperature such as is obtained in tunnel oven or hot plate processes. The tunnel oven process also provides more positive control of maximum temperature and is considered superior at this time.

An opaque silicon adhesive such as RTV-40 is typically used for bonding solar cells to structural substrate. The substrate is normally covered with a lightweight dielectric coating to provide adequate electrical insulation.

ISOTOPE THERMOELECTRIC GENERATOR

A conceptual diagram of an isotope thermoelectric generator is shown in Fig. 11. Such configurations are typical of those being considered for generators ranging in size from tens of watts to a few hundred watts? This generator[10] was designed to provide about 280-watts and is intended to re-enter intact from orbit in case of a launch abort. Slightly over 5 lbs of curium 244 fuel was considered adequate based on a generator efficiency of 5 percent. The total generator weight was estimated to be about 90 lbs resulting in a specific power of slightly over 3 watts/lb.

The design concept shown in Fig. 11 is based on the use of germanium silicon thermoelements with a hot junction temperature between 800 and 850°C. The cold junction would operate at about 300°C.

SOLAR THERMIONIC SOURCE

Ever since Edison observed the thermionic emission of electrons from an incandescent filament in one of his electric lamps, there has been experimental evidence for the thermionic conversion of heat into electricity. In an 1883 patent, Edison describes having witnessed the flow of an electric current in a vacuum between a heated electrode and a cooler electrode made positive by an external voltage. Since the electron was not a well defined constituent of matter at that time, it is doubtful that Edison viewed the phenomena as it is understood today.

The thermionic effect has been studied extensively over the past eighty years and is responsible for a still thriving industry in electron tubes. Only recently has it been seriously considered for the conversion of heat. into electricity. The first detailed electronic and thermodynamic study on the subject was initiated in the mechanical engineering department at the Massachusetts Institute of Technology in 1953. Hatsopoulos reported the results of this work in his doctor's dissertation[11] in 1956. Several papers[12-17] on the subject were published in 1958-59 by scientists at MIT and in the research laboratories of RCA, G.E., Los Alamos and Thermo Electron Engineering Corp. With the greater emphasis given to space exploration following the successful launch of Sputnik I in October 1957, the possibility of using the newly re-discovered thermionic converter to generate electric power in space from the naturally occurring solar energy was proposed.

In 1957, Hernqvist[18] working at the RCA laboratories in Princeton, New Jersey, investigated the use of cesium in the thermionic converter in an attempt to overcome the limitations of space charge on current density. It was found that the introduction of cesium gas produced very marked improvement in converter power density and efficiency. In 1958, Hernqvist[18] and others at RCA, tested a cesium filled thermionic converter using concentrated solar energy. This is believed to be the first generation of electric power from solar energy via thermionic conversion. More recently, Rouklove[19] has reported obtaining an output of over 100 watts from a five converter thermionic generator heated with solar energy from a 9.5-ft diameter parabolic concentrator.

Solar thermionic modules of the type shown in Fig. 12 have been under investigation for about four years, The solar concentrator is a 5-ft diameter, electroformed nickel paraboloid capable of efficient operation at concentration ratios of 14,000. The thermionic generator is similar to that tested by Rouklove[19]. Solar energy enters the $\frac{1}{2}$-inch cavity aperture and heats the cathodes to about $2000^\circ K$. Electrons emitted from the cathode are collected by a cooler anode placed about 0.002-inch away. The electrons flow through the series connected converters and through

the load. The anode operates at about 1000°K. Cesium gas is introduced into the interelectrode space to reduce the effect of the electron space charge and serves to modify the work function of the cathode and anode. Power densities greater than 20 watts/cm2 have been obtained.

SUMMARY

The solar cell-battery elecrric power system is expected to play a dominant role in the unmanned scientific exploration of the solar system. Isotope thermoelectric (or thermionic) generators should see service for missions where the availability of solar energy is limited or uncertain. Solar thermionic power sources may prove to be "uniquely useful" for missions to the planet Mercury and close-in to the sun.

The most advanced solar cell array flown to date is that on Mariner 4 with a nominal power (1 A.U.) of 670-watts, giving about 9.5 watts/ft^2 and 9.5 watts/lb.
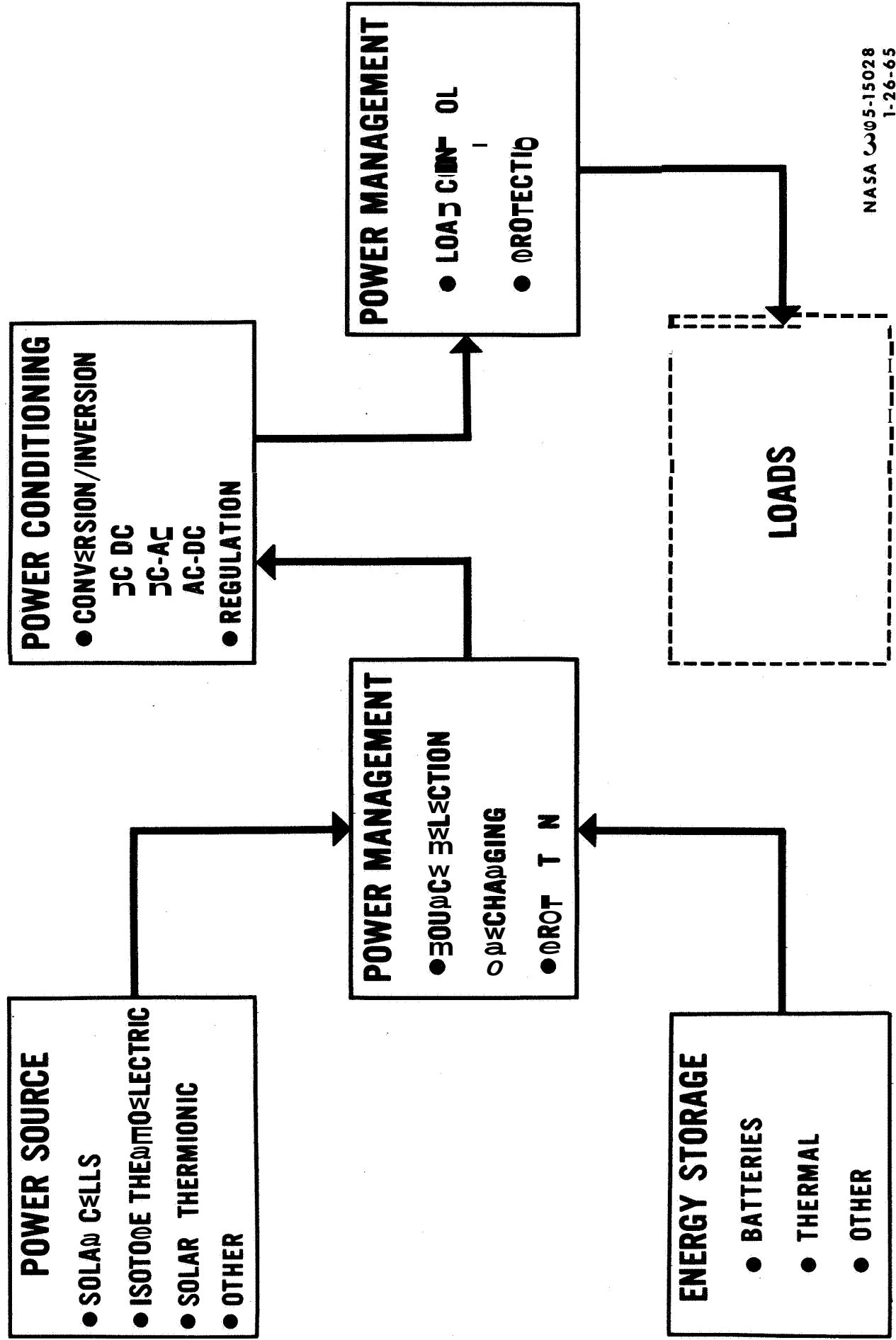
Improvements in life and reliability are required before isotope thermoelectric generators can be used with confidence. The solar thermionic power source will require substantial further development to reach an operational status.

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FIG. 1: ELEMENTS OF SPACE ELECTRIC POWER SYSTEM

POWER SOURCE
- SOLAR CELLS
- ISOTOPE THERMOELECTRIC
- SOLAR THERMIONIC
- OTHER

POWER CONDITIONING
- CONVERSION/INVERSION
  - DC-DC
  - DC-AC
  - AC-DC
- REGULATION

POWER MANAGEMENT
- LOAD CONTROL
- PROTECTION

POWER MANAGEMENT
- SOURCE SELECTION
- CHARGING
- PROTECTION

ENERGY STORAGE
- BATTERIES
- THERMAL
- OTHER

LOADS

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FIG. 2: RANGER SPACECRAFT ELECTRIC ELECTRIC POWER SYSTEM

SOLAR PANEL

SOLAR PANEL

BATTERIES

POWER CONDITIONING MODULES

NASA RN-3988-11.64 REV 1-26-65

FIG. 3: USE OF ELECTRIC POWER ON PLANETARY SPACECRAFT

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1-26-65

Pie chart segments:
- REDUNDANCY - 13%
- RESERVE - 13%
- CONDITIONING - 4%
- REGULATED (AC) - 6%
- UNREGULATED (DC) - 18%
- TELECOMMUNICATIONS - 28%
- POWER - 3%
- GUIDANCE AND CONTROL - 7%
- ENCODER - 6%
- REGULATED - 12%
- CONDITIONING - 8%
- SCIENCE - 20%
- UNCERTAINTY - 10%
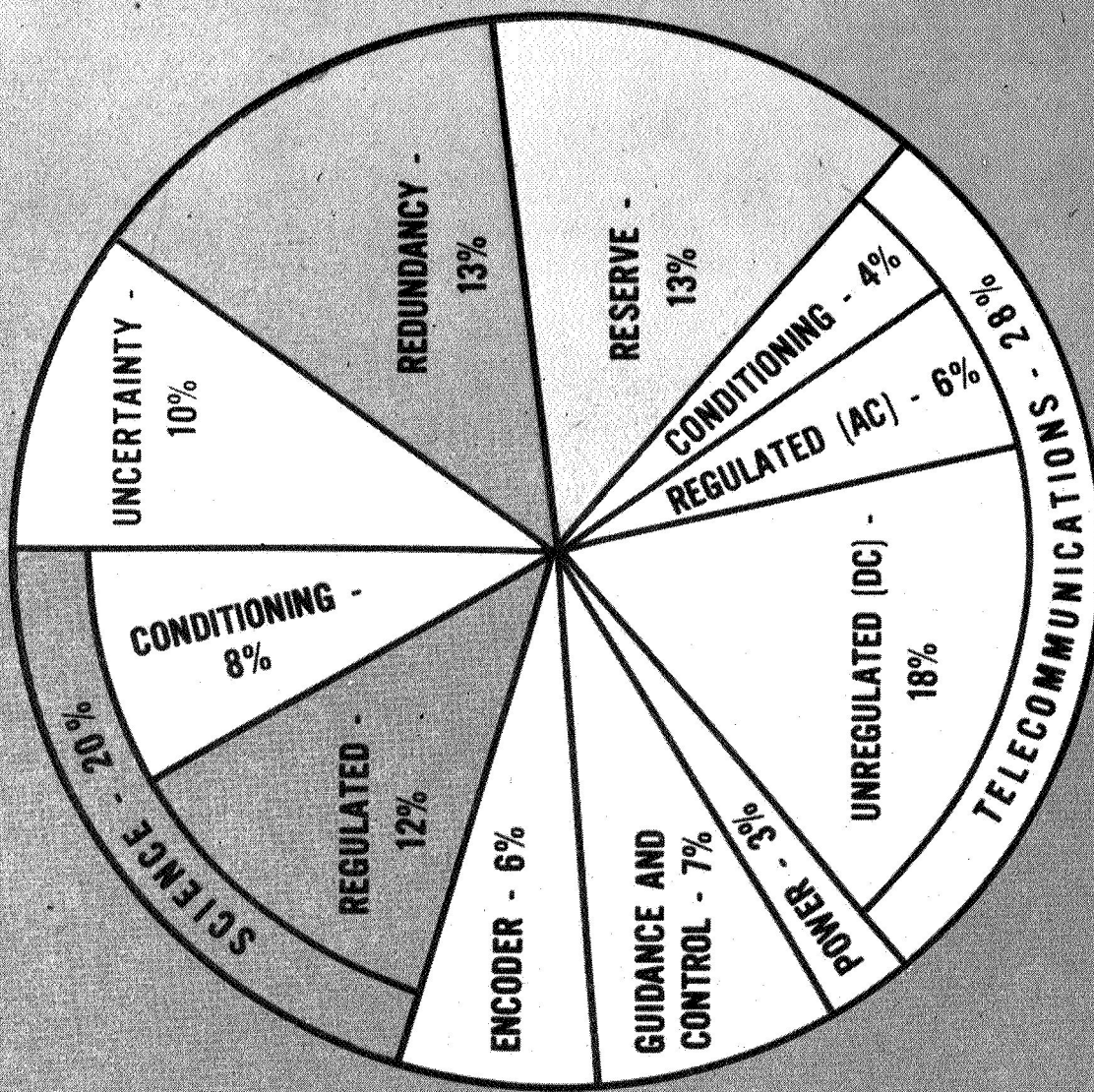
FIG. 4: FACTORS IN SPACE POWER SYSTEM SELECTION

- DURATION
- MISSION
- AVAILABILITY
- RELIABILITY
- WEIGHT
- COMPATIBILITY
- o ENVIRONMENT
- o POWER LEVEL
- AREA
- COST
- VOLUME
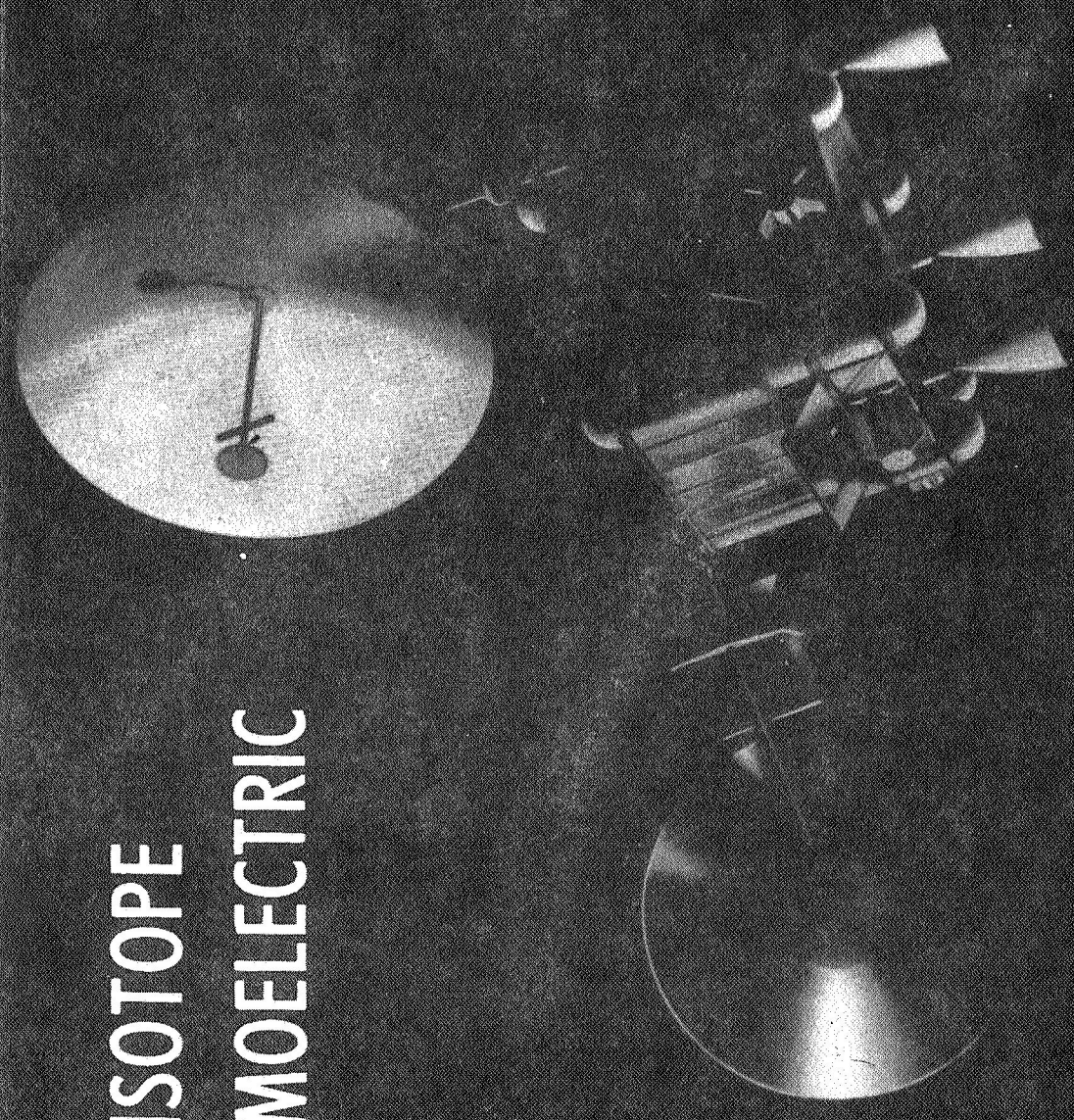- HAZARD
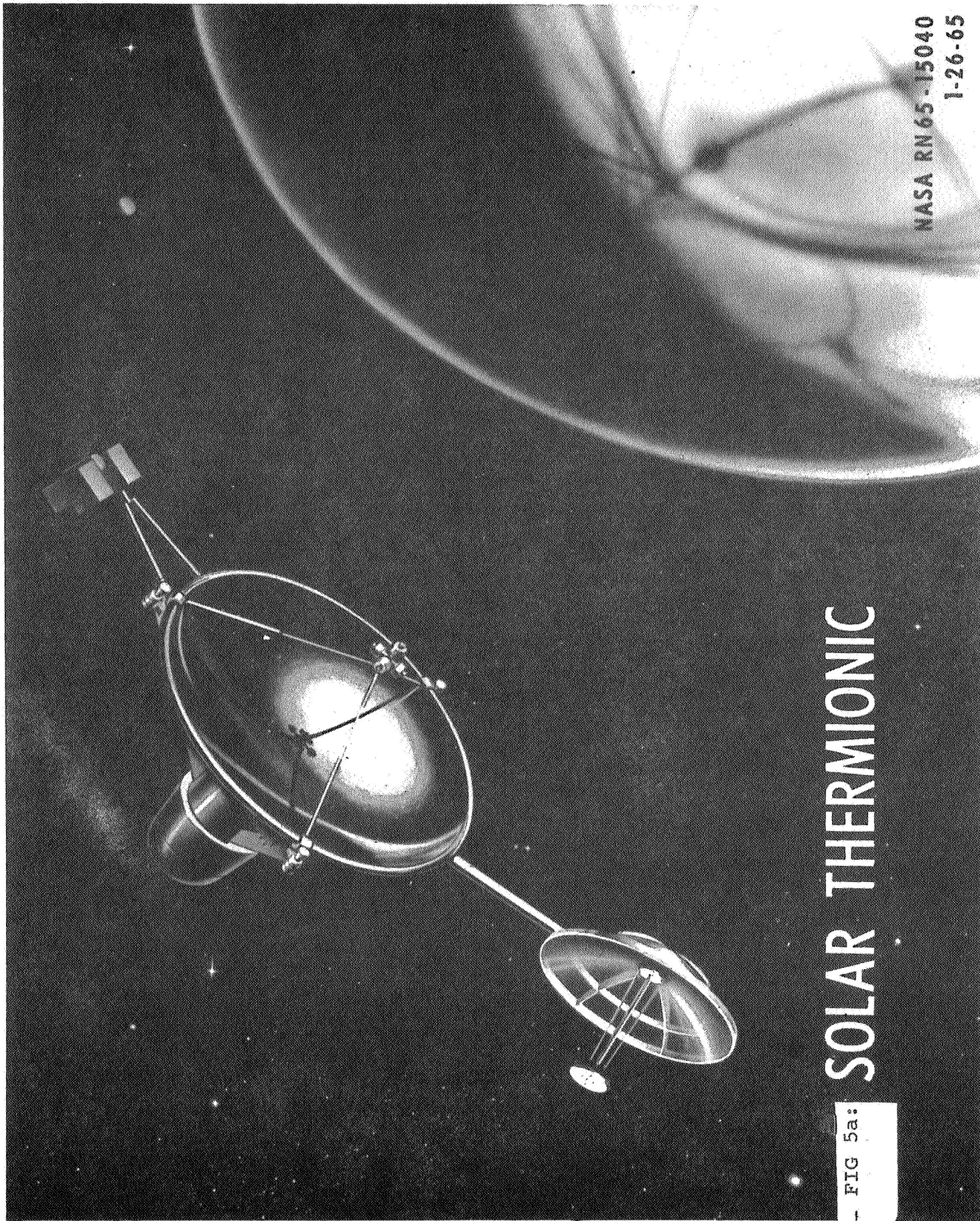
FIG 5a: SOLAR THERMIONIC

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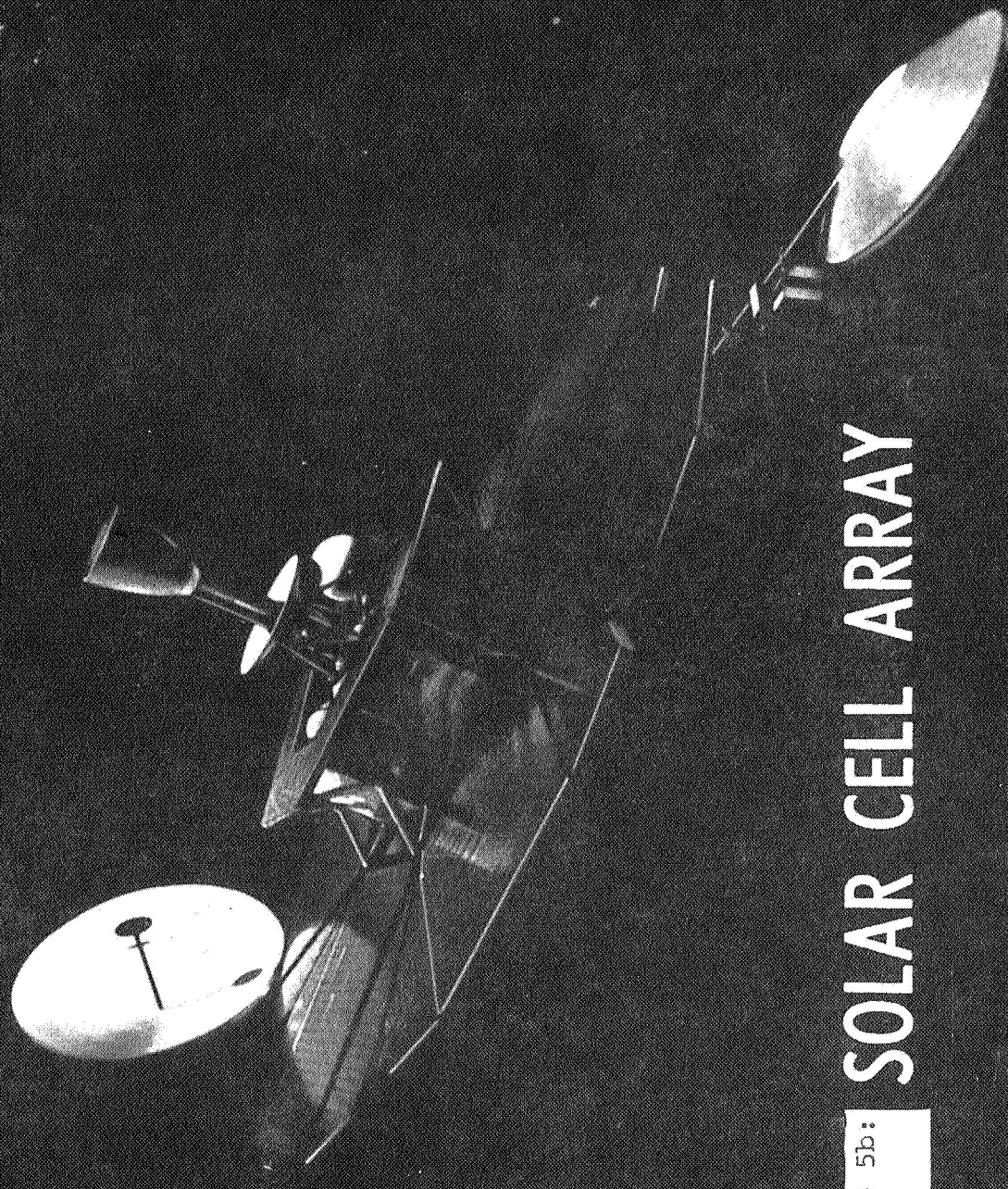
FIG 5b: SOLAR CELL ARRAY

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FIG 5c: ISOTOPE THERMOELECTRIC

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FIG. 5a: POWER SOURCE SPECIFIC WEIGHT VS DISTANCE FROM SUN (TYPICAL)
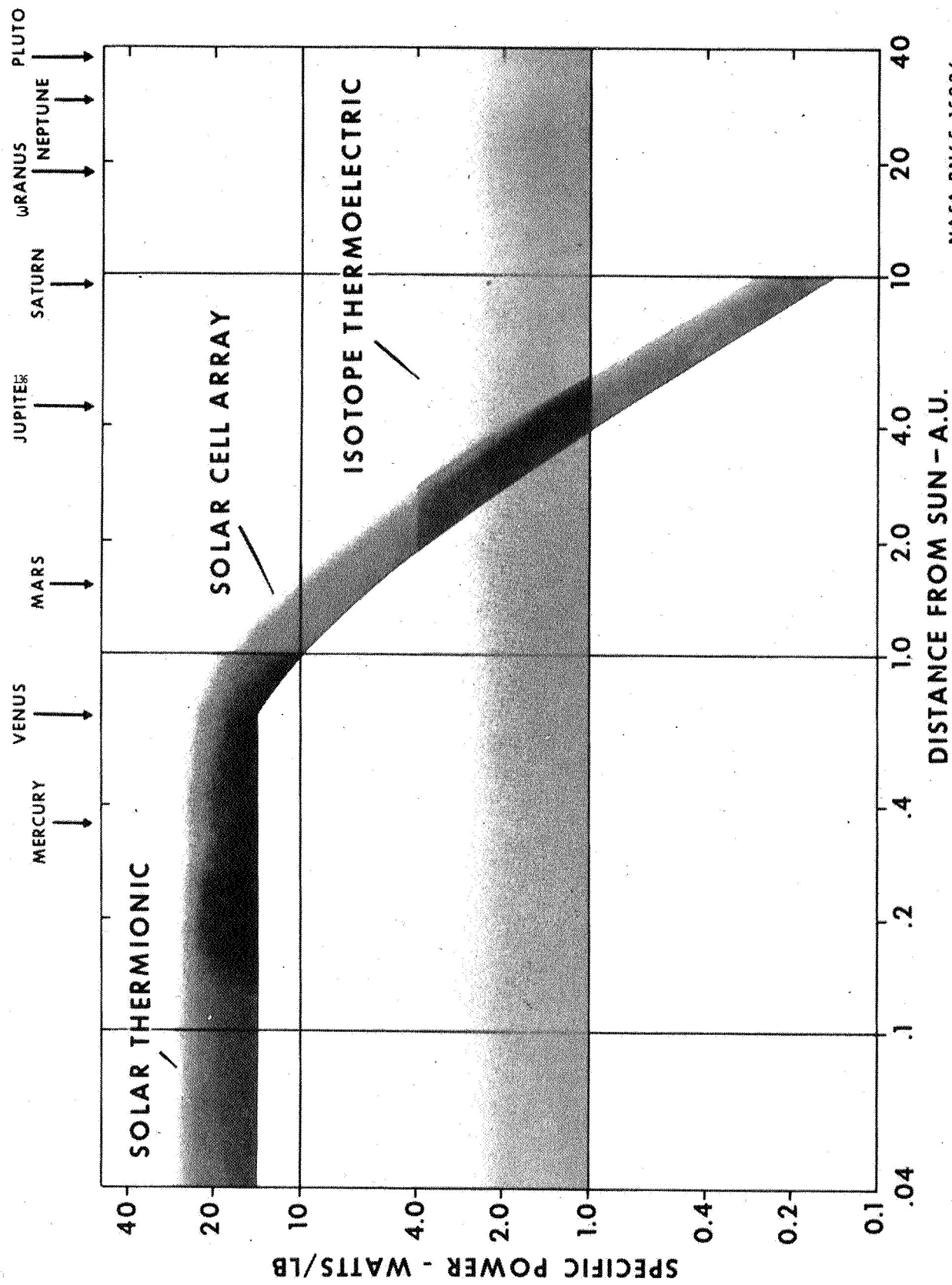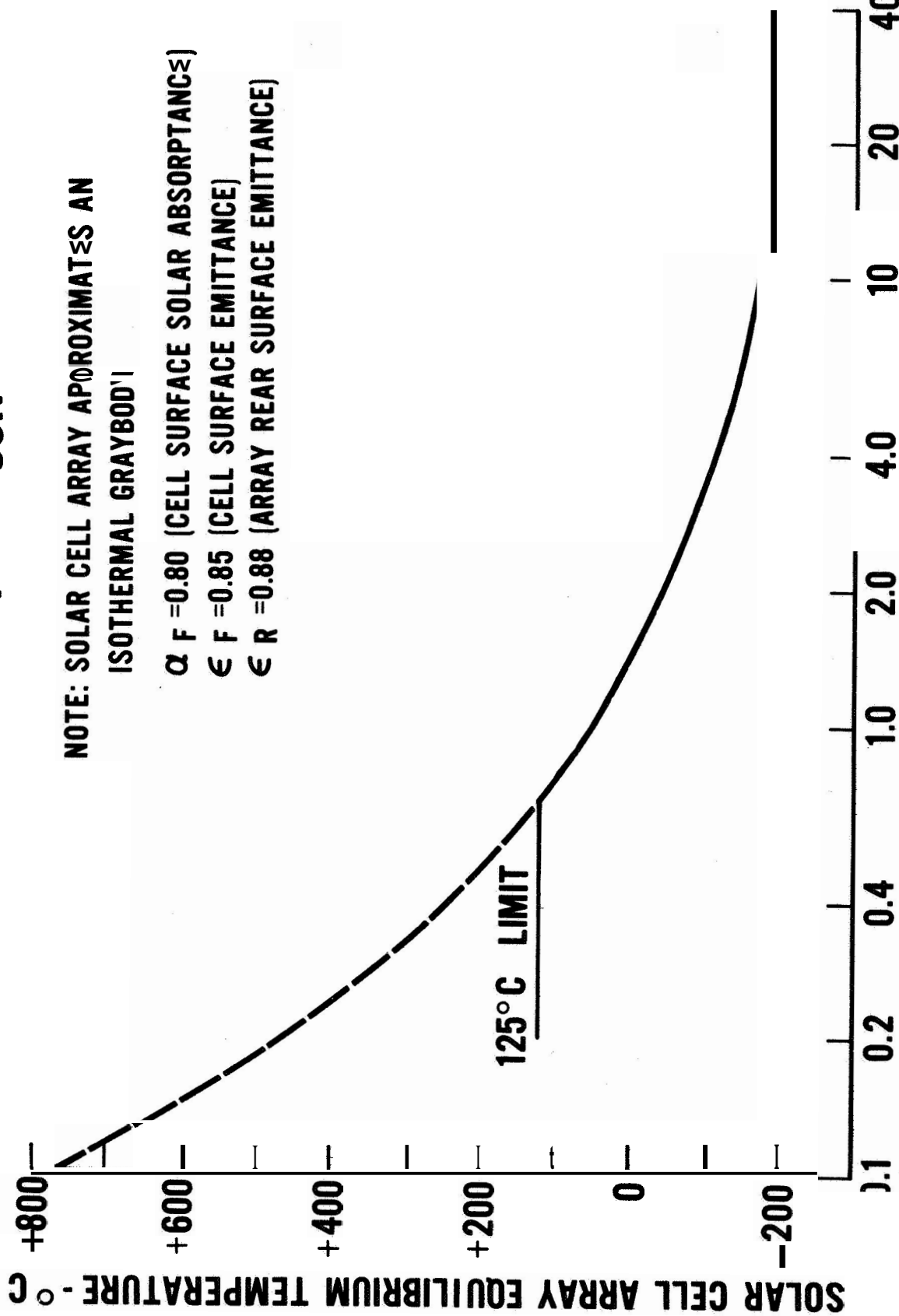
NASA RN65-15036
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FIG. a: SOLAR CELL ARRAY EQUILIBRIUM TEMPERATURE VS. DISTANCE FROM SUN

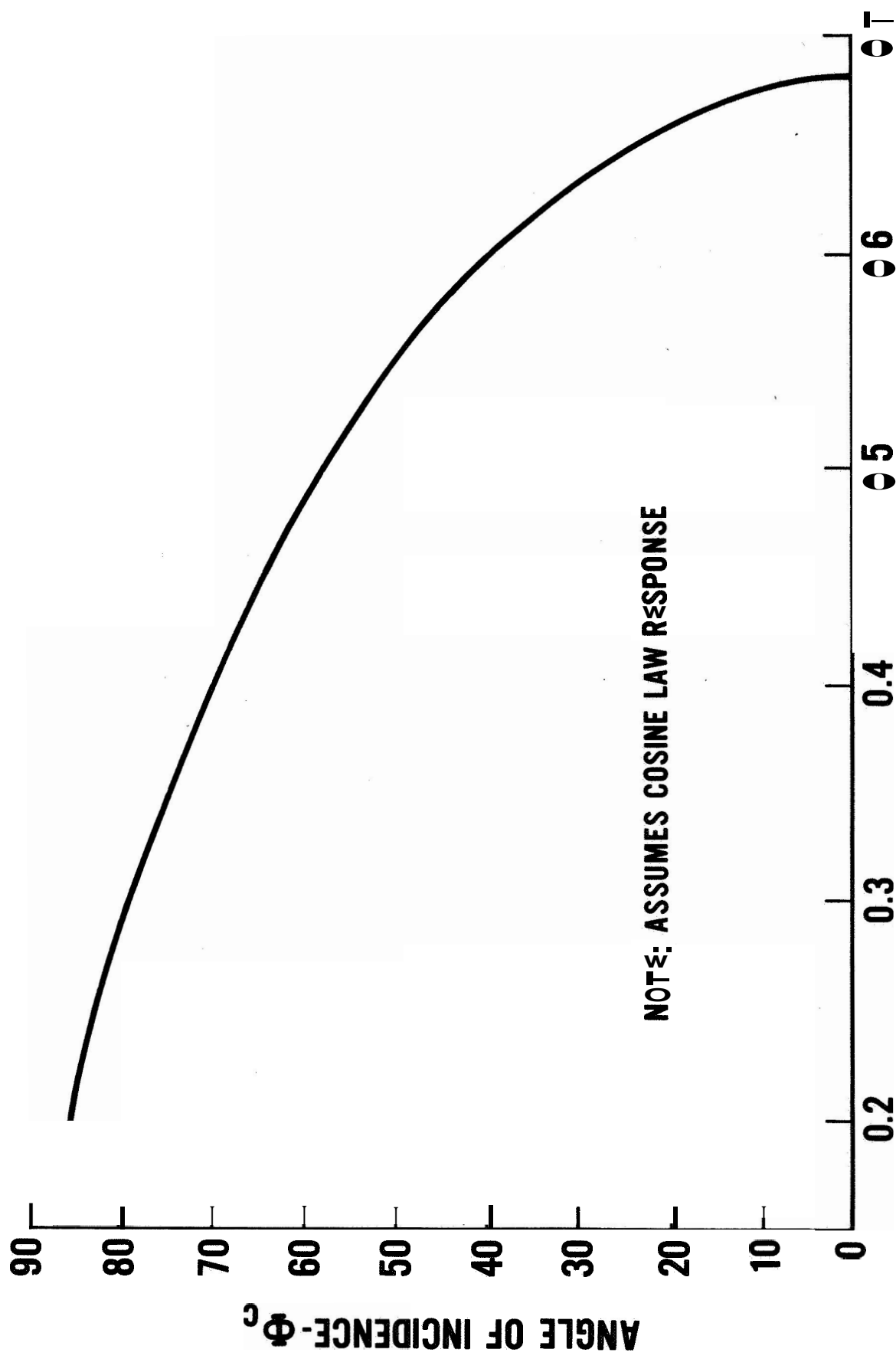NOTE: SOLAR CELL ARRAY APPROXIMATES AN ISOTHERMAL GRAYBODY

$\alpha_F = 0.80$ (CELL SURFACE SOLAR ABSORPTANCE)
$\epsilon_F = 0.85$ (CELL SURFACE EMITTANCE)
$\epsilon_R = 0.88$ (ARRAY REAR SURFACE EMITTANCE)

125°C LIMIT

DISTANCE FROM SUN A.U.

SOLAR CELL ARRAY EQUILIBRIUM TEMPERATURE - °C

NASA RN65-15032
1-26-65

FIG. 6b: **TILTING METHOD FOR COOLING SOLAR CELL ARRAY**

ANGLE OF INCIDENCE-Φ_c

DISTANCE FROM SUN – A.U.

NOTE: ASSUMES COSINE LAW RESPONSE

FIG. 7: SOLAR CELL POWER SYSTEMS

CHARACTERISTIC	LUNAR[1]	EARTH SATELLITE[2]	PLANETARY[3]
SPACECRAFT, LBS	806	1,073	570
POWER SYSTEM, LBS	124	211	153
SOLAR ARRAY			
AREA, FT^2	24.4	77.0	70
NUMBER OF CELLS	9,792	32,256	28,224
WEIGHT, LBS	44	109	70
POWER, WATTS	210	600	670 (EARTH) 320 (MARS)
BATTERY			
TYPE	SILVER-ZINC	NICKEL-CADMIUM	SILVER-ZINC
WEIGHT, LBS	53	65	33
CAPACITY, AMP-HOURS	50	28	50
NUMBER CELLS	28	44	18
MANAGEMENT AND CONDITIONING			
WEIGHT, LBS	27	37	50

(1) RANGER 7 BUS (DOES NOT INCLUDE TV POWER SUPPLY) (2) OGO-1 (REF. 18) (3) MARINER 4 (REF. 19)
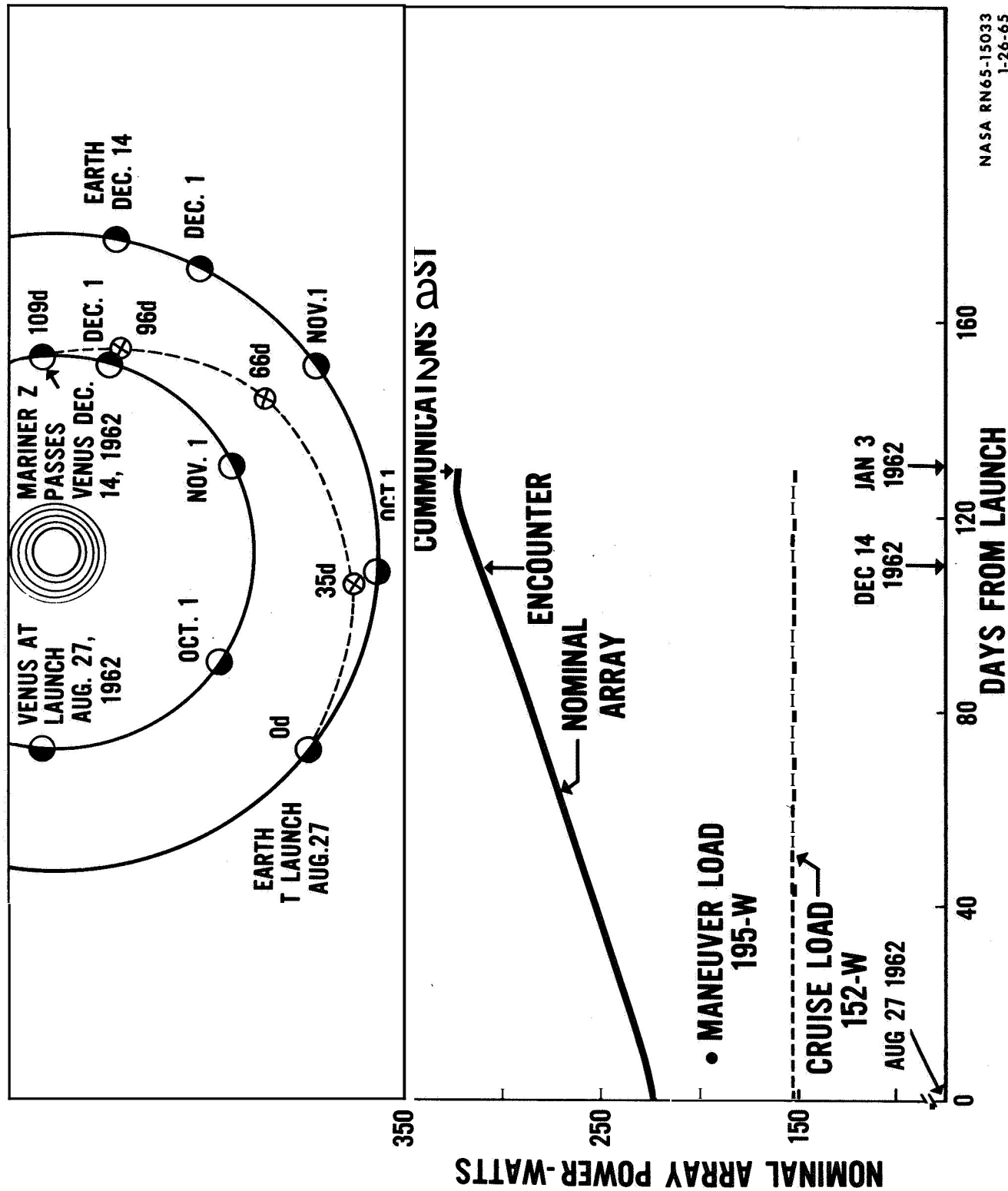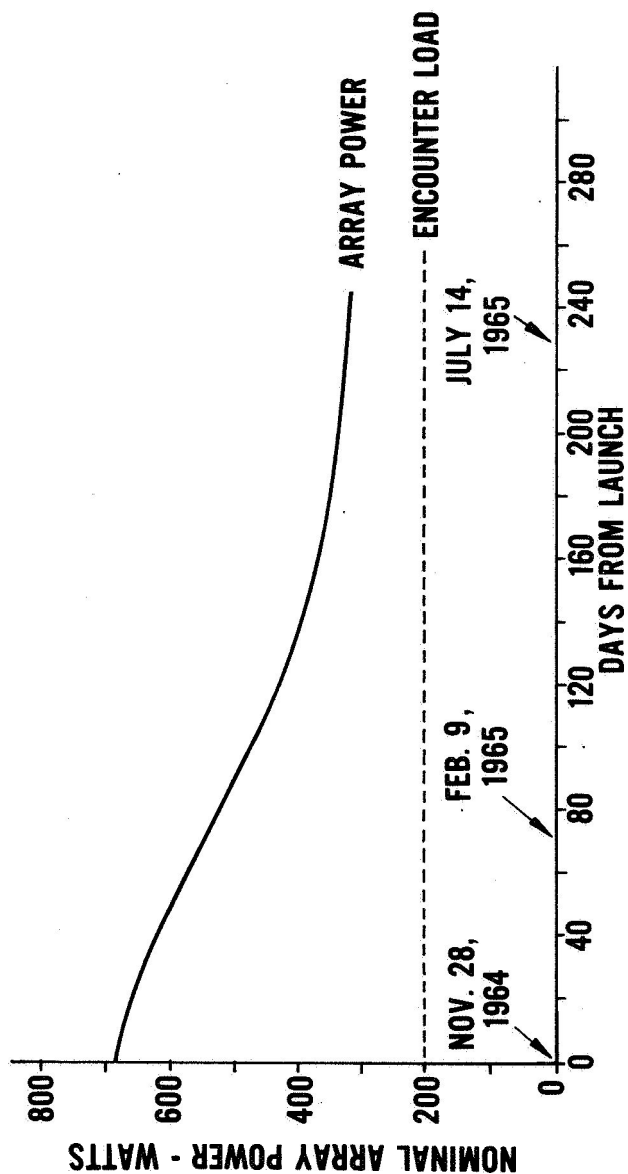
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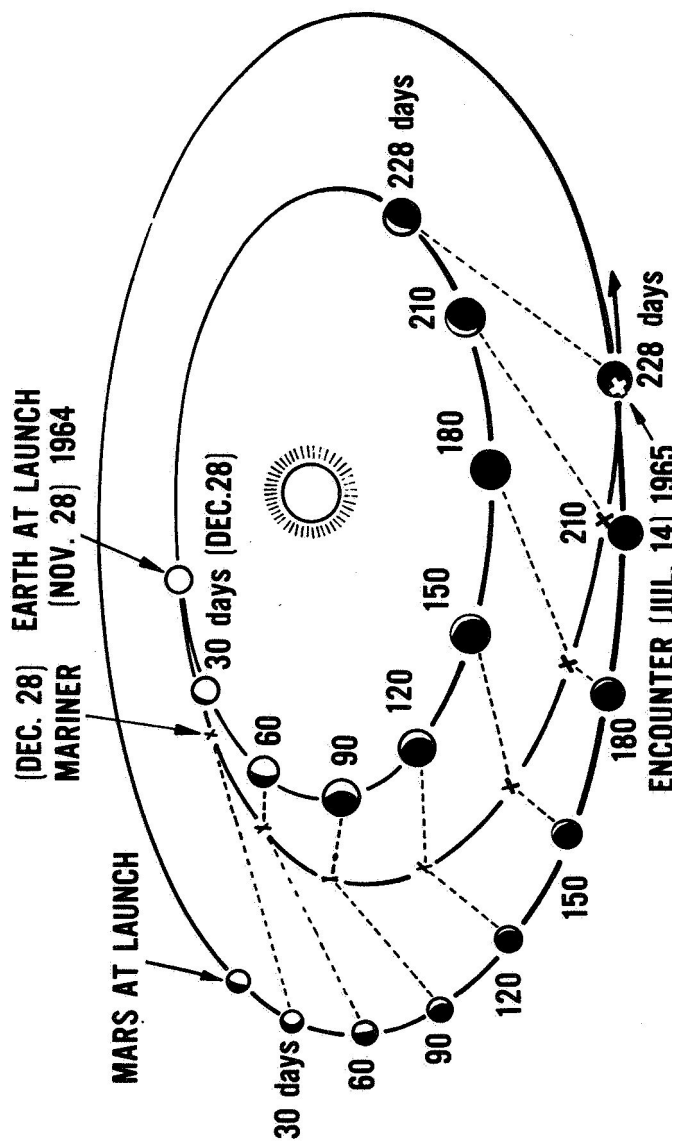
FIG. 8: **MARINER 2 VENUS MISSION**

NASA RN65-15033
1-26-65

FIG. 9: MARINER 4 MARS MISSION

(SOLAR ENERGY)

ANTIREFLECTION COATING

GLASS COVER

MULTILAYER INTERFERENCE FILTER

TRANSPARENT ADHESIVE

"N" CONTACT BUS BAR

"N" REGION OHMIC CONTACT

S i O x COATING TRANSPARENT

"N" DOPED SILICON

DIODE JUNCTION

"P" DOPED SILICON

"P" REGION OHMIC CONTACT

"P" CONTACT BUS BAR

PRIMER FOR ADHESIVE

ADHESIVE

PRIMER FOR ADHESIVE

ELECTRICAL INSULATOR

SURFACE OF SUBSTRATE SURFACE

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1-26-65
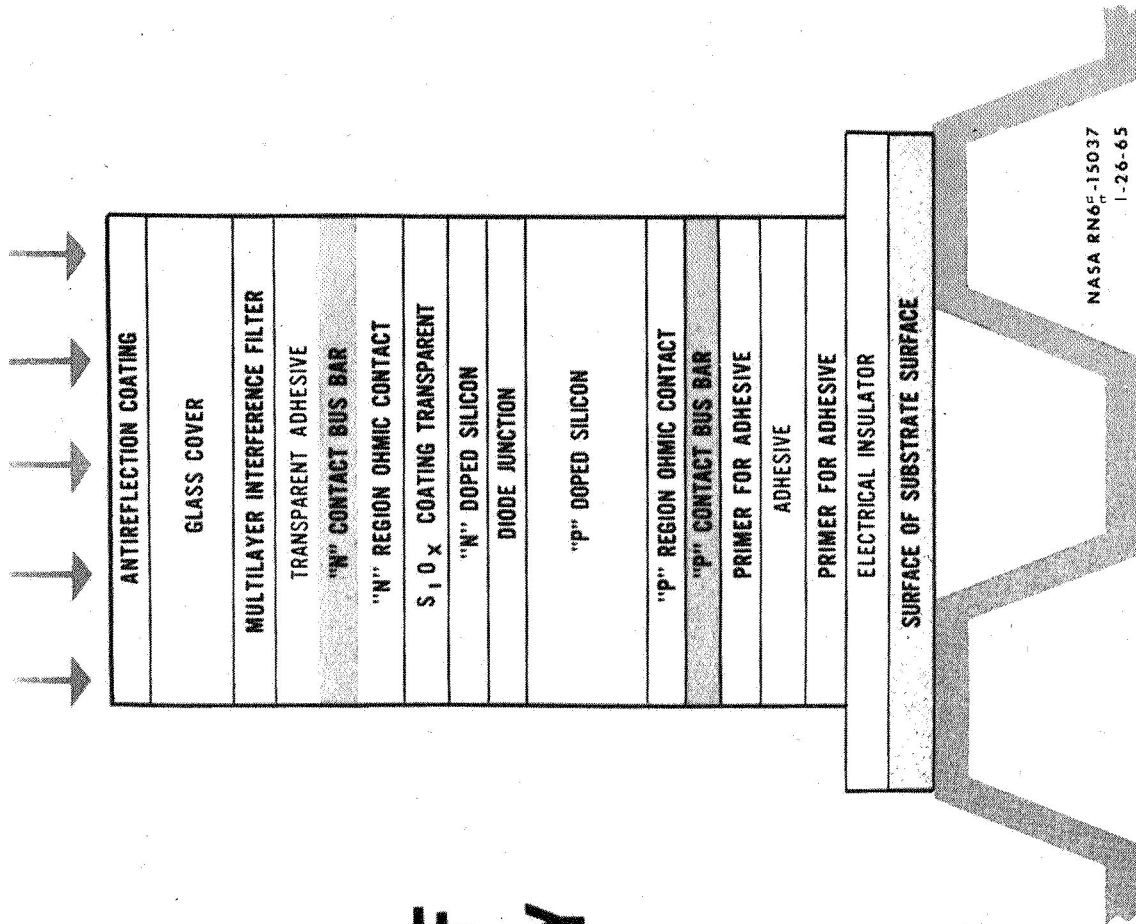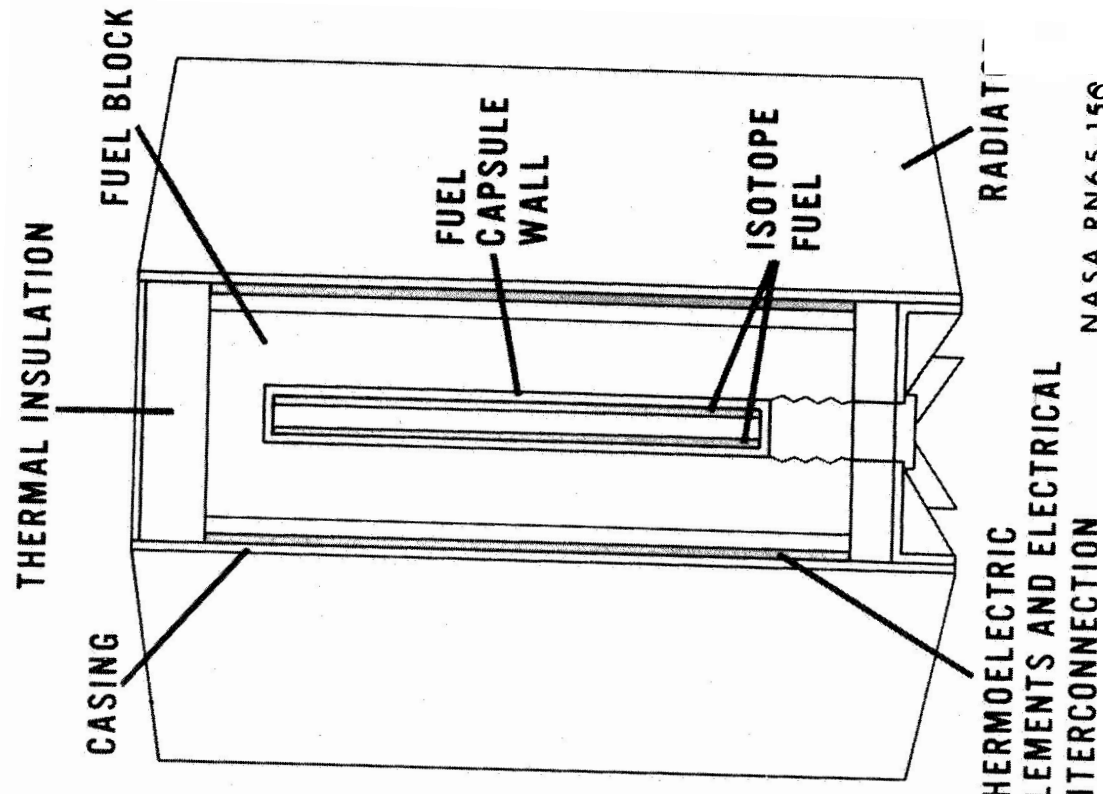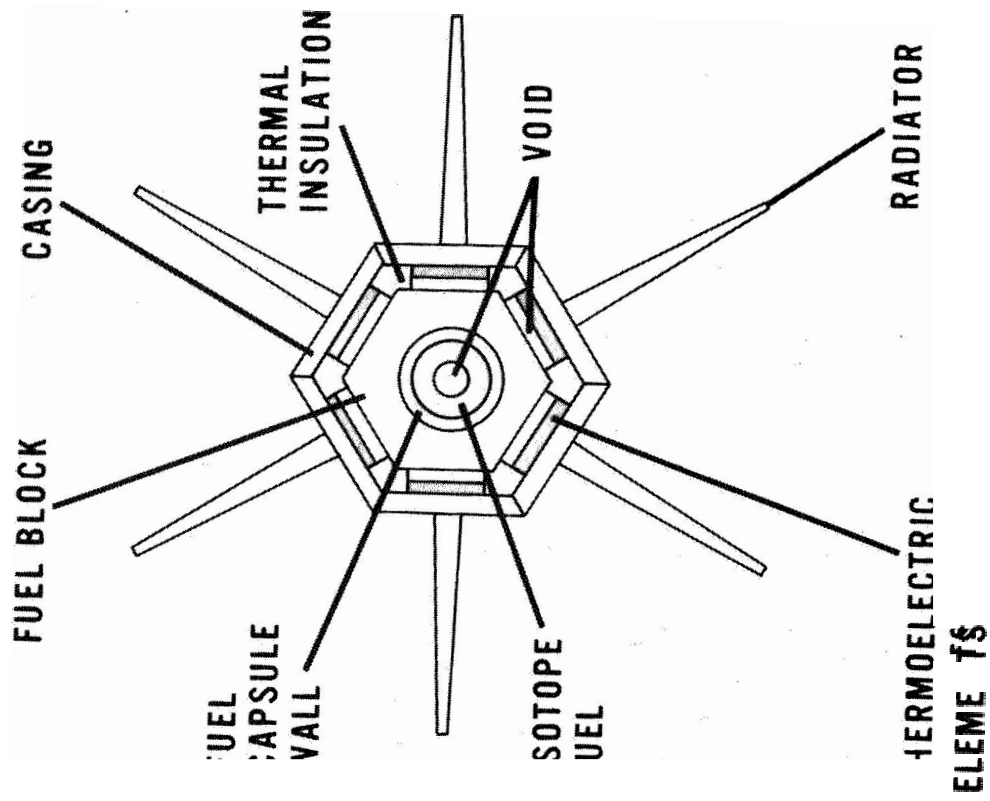
FIG. 10: CROSS SECTION OF SOLAR CELL ARRAY

FIG 11: ISOTOPE THERMOELECTRIC GENERATOR (CONCEPT)

FUEL BLOCK

THERMAL INSULATION

FUEL CAPSULE WALL

ISOTOPE FUEL

RADIATI

CASING

THERMOELECTRIC ELEMENTS AND ELECTRICAL INTERCONNECTION

NASA PN65 156

FUEL BLOCK

CASING

THERMAL INSULATION

VOID

RADIATOR

FUEL CAPSULE WALL

ISOTOPE FUEL

THERMOELECTRIC ELEME f$

— FIG. 12: SOLAR THERMIONIC SYSTEMS

MIRROR GENERATOR ASSEMBLY

CATHODE

ANODE

METAL CERAMIC SEAL

ANODE COOLING FIN

LOAD

GENERATOR SCHEMATIC

NASA RN-4208-11.64