NASA
SPACE VEHICLE
DESIGN CRITERIA
(GUIDANCE AND CONTROL)

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SPACECRAFT SOLAR CELL ARRAYS

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MAY 1971

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

GUIDE TO THE USE OF THIS MONOGRAPH

The purpose of this monograph is to organize and present, for effective use in design, the significant experience and knowledge accumulated in operational programs to date. It reviews and assesses current state-of-the-art design practices, and from them establishes firm guidance for achieving greater consistency in design, increased reliability in the end product, and greater efficiency in the design effort for conventional missions. The monograph is organized into three major sections that are preceded by a brief introduction and complemented by a set of references.

The *State of the Art*, Section 2, reviews and discusses the total design problem, and identifies *which* design elements are involved in successful design. It describes succinctly the current technology pertaining to these elements. When detailed information is required, the best available references are cited. This section serves as a survey of the subject that provides background material and prepares a proper technological base for the *Design Criteria* and *Recommended Practices.*

The *Design Criteria*, shown in Section 3, state clearly and briefly what rule, guide, limitation, or standard must be considered for each essential design element to ensure successful design. The *Design Criteria* can serve effectively as a checklist of rules for the project manager to use in guiding a design or in assessing its adequacy.

The *Recommended Practices*, as shown in Section 4, state how to satisfy each of the criteria. Whenever possible, the best procedure is described; when this cannot be done concisely, appropriate references are provided. The *Recommended Practices,* in conjunction with the *Design Criteria,* indicate how successful design may be achieved.

The design criteria monograph is not intended to be a design handbook, a set of specifications, or a design manual. It is a summary and a systematic ordering of the large and loosely organized body of existing successful design techniques and practices presently used in the majority of flight programs. It does not treat the many promising developmental efforts being made throughout the photovoltaic community, such as lithium-doped solar cells, cadmium-sulfide solar cells, roll-out solar arrays, and solar arrays designed for extreme temperature-intensity environments (for example, Mercury, Jupiter, and the Grand Tour). These efforts will be discussed in separate monographs.

FOREWORD

NASA experience has indicated a need for uniform design criteria for space vehicles. Accordingly, criteria are being developed in the following areas of technology:

Environment

Structures

Guidance and Control

Chemical Propulsion

Individual components of this work will be issued as separate monographs as soon as they are completed. This document, "Spacecraft Solar Cell Arrays," is one such monograph. A list of all monographs in this series can be found on the last page of this document.

These monographs serve as guides in NASA design and mission planning. They are used to develop requirements for specific projects and are also cited as the applicable references in mission studies and in contracts for design and development of space vehicle systems.

This monograph was prepared under the cognizance of the NASA Headquarters Office of Advanced Research and Technology and reviewed and published by the Jet Propulsion Laboratory. M. J. Barrett and R. G. Lyle of Exotech Systems, Inc., were principal investigator and program manager, respectively.

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Contributions in the area of design and development practices were also provided by many other engineers of NASA and the aerospace community.

Comments concerning the technical content of this monograph will be welcomed by the National Aeronautics and Space Administration, Office of Advanced Research and Technology (Code RP), Washington, D.C., 20546.

May 1971

CONTENTS

FOREWORD iii

1. INTRODUCTION 1

2. STATE OF THE ART 1

2.1 Historical Background 1

2.2 Flight Experience 2

2.3 Solar Cell Characteristics 3

2.4 Effects of Space Environment 8

2.5 Solar Cell Protective Covers and Coatings 14

2.6 Adhesives 15

2.7 Solar Array Construction 16

2.8 Testing 26

3. CRITERIA 29

3.1 Assessment of Design Objectives 29

3.2 Components of the Solar Cell Array 31

3.3 Testing 34

4. RECOMMENDED PRACTICES 34

4.1 Assessment of Design Objectives 35

4.2 Components of the Solar Cell Array 37

4.3 Testing 42

REFERENCES 44

NASA SPACE VEHICLE DESIGN CRITERIA MONOGRAPHS ISSUED TO DATE 47

SPACECRAFT SOLAR CELL ARRAYS

1. INTRODUCTION

Design for any spacecraft includes its electrical power needs and the system to supply them. The availability of sunlight has encouraged the development of solar cell arrays, Simplicity, relatively modest cost, and high reliability have caused this system to be chosen to supply sustained electrical power for almost all unmanned spacecraft.

The performance of a spacecraft solar cell array depends on many parameters. Foremost, of course, are the intensity of sunlight incident on the array and the conversion efficiency of the solar cells. The conversion efficiency is affected by angle of incidence, temperature, the corpuscular radiation environment, occasional meteoroid collisions, ultraviolet radiation, and the electrical load. Designing an array for high performance throughout the mission must take into account both these parameters and the interaction of the array with the spacecraft. Typical design considerations are storage during launch; motive power for deployment; weight; thermal balance; requirements for batteries and power conditioning; and reconciliation of any conflicting orientation requirements with those of the communication antenna or optical sensors.

This monograph is limited to flight proven choices for solar cells and arrays for conventional missions. Subjects that properly belong outside the scope of this monograph, such as array articulation or deployment, are treated only briefly. Design constraints arising from magnetic cleanliness requirements and structural flexibility are covered in NASA SP-8016 and NASA SP-8037, respectively.

Many additional considerations, practices, and requirements exist for missions involving environmental extremes (near and far Sun missions), and treatment of such missions is beyond the scope of this monograph. Moreover, the objective of describing current, state-of-the-art systems precludes consideration of advancements that may be used in the near future (for example, roll-out arrays, cadmium-sulfide solar cells, and lithium-doped solar cells).

2. STATE OF THE ART

2.1 Historical Background

The photovoltaic effect – creation of an electric potential by the absorption of light – was first described by E. Becquerel in 1839. He observed that when two identical electrodes were immersed in an electrolyte, a potential developed between them if one was illuminated while the other remained dark. Further research before 1900 greatly increased the number of electrode–electrolyte systems known to be photosensitive. In 1877, W. G. Adams and R. E. Day reported the photovoltaic effect in a solid. Before 1940, interest centered around Cu_2O cells. The analytical relationship between illumination and induced potential was developed by A. D. Garrison in 1923 and extended by E. Adler in 1940. During this period several solid cells using Se, PbS, and Cu_2O were studied but the first practical photocell, patented in 1926, was of Cu–Cu_2O. The mechanism and equivalent circuit of this cell were described in 1930 by W. Schottky.

Improvements in semiconductor purity during the 1940s and 1950s led to the discovery of photovoltaically sensitive *p/n* junctions[1] in silicon, germanium, and lead sulfide, with most of the early work being done on germanium. The development of vapor diffusion techniques and "high quality" silicon made possible the solar battery introduced by Bell Laboratories in 1954 (ref. 1). A 6% conversion efficiency was realized. This was raised to an average of 8% by 1955, with some units reaching 11%. An interesting further account of the course of early research is given by Crossley, Noel, and Wolf (ref. 2).

2.2 Flight Experience

The use of solar cells for spacecraft power started with Vanguard 1, launched March 17, 1958. Six panels, each with eighteen 2 × ½-cm ungridded *p/n* solar cells[2], were used as a secondary power source. Protected by 0.16-cm (1/16-in.) quartz windows and with a very low power demand (ref. 3), these solar cells continued to power a radio signal until February, 1965.

Explorer 6, launched August 7, 1959, was the first spacecraft to use solar cell paddles. Of the four 51-cm^2 paddles, one failed to extend fully and lock. The solar cells of the resultant three-paddle array rapidly degraded in the Van Allen belt and all transmission was lost in 2 months (ref. 4).

This failure was followed by a string of successes, and solar cells became a preferred power supply. Then, on July 9, 1962, a high-altitude nuclear explosion, "Starfish," released an estimated 10^{25} fission electrons that became trapped in the lower region of the Van Allen belt. The resultant damage to solar cell arrays, evident in figure 1, rapidly caused a number of spacecraft to cease transmission.

Radiation damage became a subject of intense interest, and changes were implemented to improve radiation resistance of arrays. These included the *n/p* solar cell, increased base resistivity, and more careful shielding by coverslides. As power requirements increased, detailed criteria for the array were evolved. Ranger Block II, Mariner 2, Mariner 4, Nimbus 1 (1964), and Pegasus (1965) marked a return to flat-mounted solar cells to accommodate thermal expansion better than the popular rigid-shingling method.[3] In flat-mounting, however, only the coverslide and its adhesive shielded each cell. On several spacecraft launched in 1967 and 1968, coverslides slightly smaller than the solar cells were used for ease in construction and because of tolerances necessary in cell and coverslide size. Any adhesive that extruded around the coverslide was carefully cleaned away. As a result, on Intelsat 2-F4, Applications Technology Satellite (ATS)-1, and Gravity Gradient Test Satellite (GGTS) about 5% of the solar cell front area was bare or covered only by a thin contact bar. The rapid degradation of those solar cell arrays has been attributed to low energy protons of the outer region of the Van Allen belt entering the bare surface and damaging the diode junction (ref. 6).

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[1]Adjacent regions in a crystal having *p*-type and *n*-type conductivity, respectively; i.e., the majority carriers of charge are unoccupied electron energy levels in the former and extra electrons in the latter.

[2]The *p*-type conductivity region was at the illuminated surface, overlying the *n*-type conductivity region.

[3]See Section 2.7.1 and figure 16.

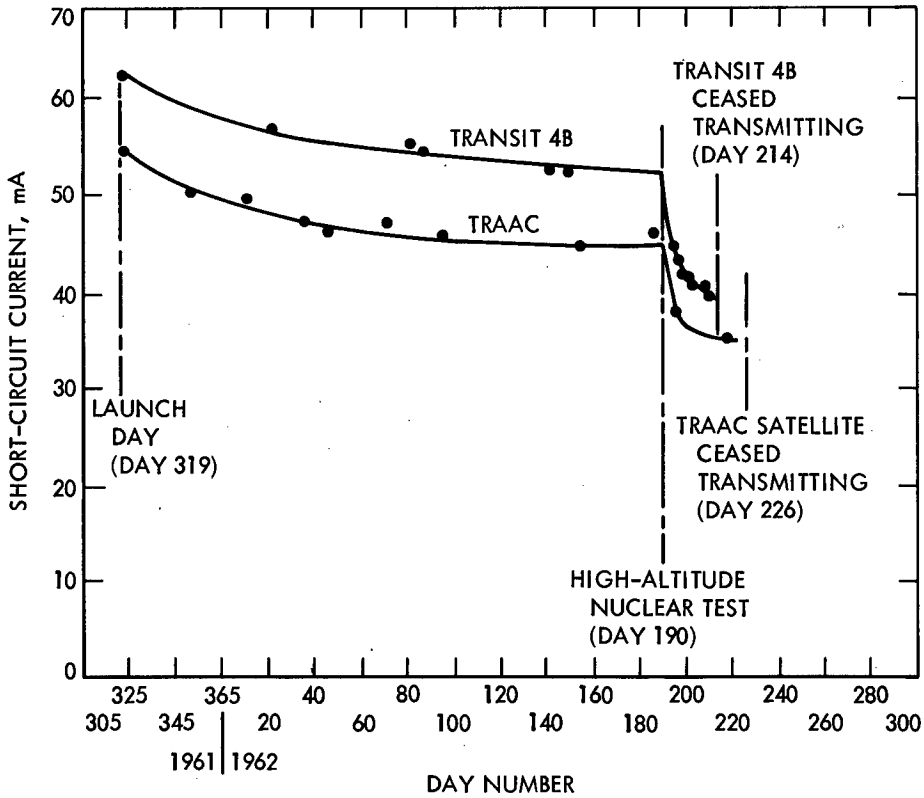

Figure 1.–Solar cell output as a function of time for Transit 4B and TRAAC (ref. 5).

The present direction of design effort is to higher efficiencies, lower cost and less weight per watt, larger arrays, longer lifetimes, and better reliability. Innovations such as larger solar cells, lithium doping, contact improvements, and light substrates hold promise of making possible better solar cell arrays for the future. For the present, silicon solar cell arrays are established as the most reliable and economical generator of sustained power in space, yet they are a limiting factor to the useful life of spacecraft.

2.3 Solar Cell Characteristics

2.3.1 Physical Description

The solar cell used for almost all spacecraft arrays is a rectangular wafer cut from a silicon crystal and prepared so that a diode junction lies just below its front surface. At present, a commonly used size is a 4-cm square, about 0.20 to 0.40 mm (8 to 16 mils) thick. Cells as small as 1 × 1 cm and as large as 2 × 6 or 3 × 3 cm have been flown. As shown in figure 2, the front surface has a contact bar along one edge, with 5 or 6 grid lines from it across the surface. The rear contact completely covers that surface of the cell.

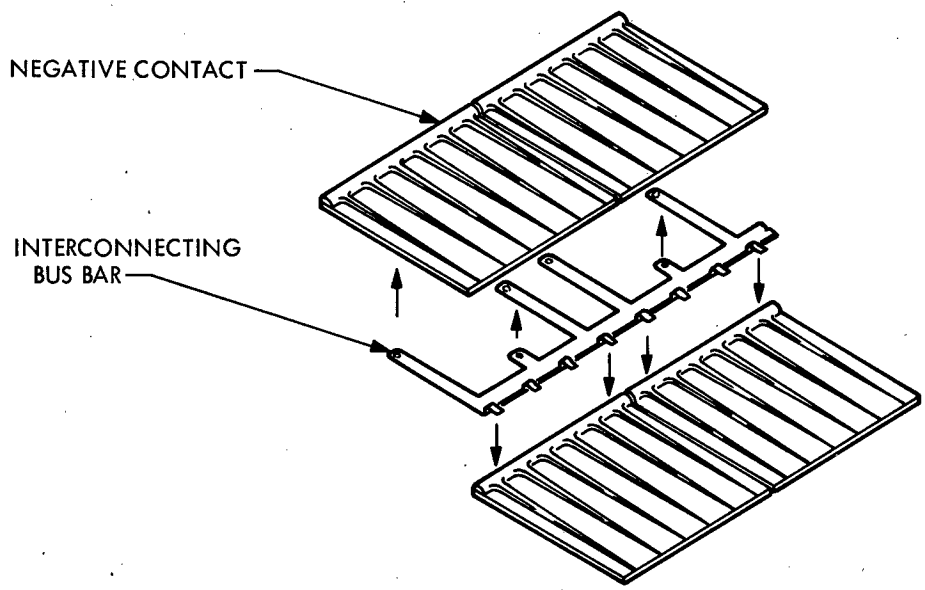

Figure 2.–Four typical solar cells (2 × 2 cm) and interconnector. The thickness of the front (negative) contact is exaggerated.

The contact is frequently an evaporated layer of titanium overlaid with silver and sintered; a coating of solder provides protection from corrosion during prelaunch storage but can reduce thermal shock capability. Recent studies have shown that corrosion resistance may be achieved without solder by the addition of palladium to the contact (ref. 7).

The front active surface of the solar cell is coated with a thin layer of silicon monoxide to act as a quarter-wave antireflective surface. This enhances the absorption of light in the silicon, particularly in the red part of the spectrum. Only a small amount of blue and infrared light is reflected, giving solar cells a characteristic blue appearance instead of the lustrous gray appearance of bare silicon.

2.3.2 Energy Conversion by Solar Cells

The silicon solar cell is a diode, having a thin "surface" region, and a thicker "base" region. Solar cells are termed *n/p* if the surface region is of *n*-type silicon, with electrons as majority carriers, and the base region is of *p*-type silicon, with holes as majority carriers. The reverse construction is the *p/n* solar cell. When cells are identified by their resistivity, it is usually understood that the electrical resistivity value is that of the base region. Typical modern arrays are built with 1–10 ohm-cm, *n/p* silicon solar cells, although cells of higher base resistivities and *p/n* construction are available for special applications.

Sunlight, in the wavelength range between roughly 0.4 and 1.0 μm (which represents the major portion of the solar spectrum), is absorbed by the silicon, causing the creation of free charge carriers. Some of the free charge carriers generated near the *p–n* junction diffuse to, and cross

over, the junction. This provides current which can flow to an external load through the electrodes attached to the front and rear of the cell. The wavelength beyond which the cell ceases to respond with an appreciable current will vary slightly with the junction depth and the transport properties of the material (ref. 8). Light energy that is absorbed but not converted to electricity will appear as heat. Figure 3 shows the typical spectral response for modern cells.

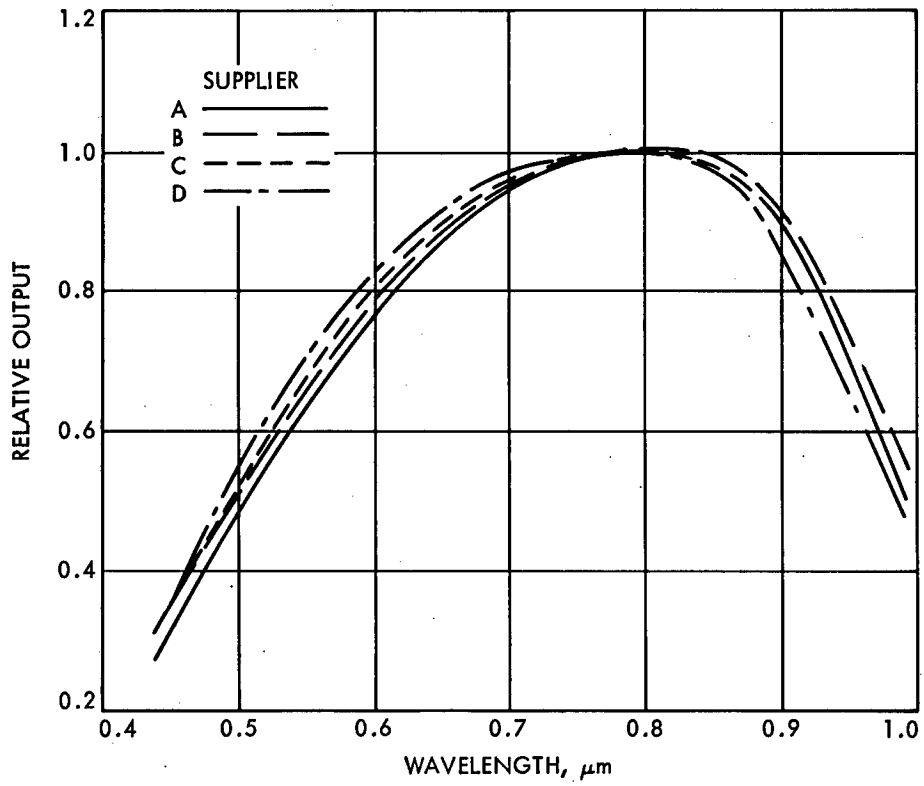

Figure 3.–Measured relative energy conversion of typical cells versus the wavelength of light (ref. 9).

The mean absorption depth of photons in silicon is a function of their wavelength. The shorter wavelengths are absorbed close to the surface, while red or infrared radiation may penetrate several hundred microns deeper. Because of recombination in the surface region and at the front surface, a large portion of the power generated by the shorter wavelengths where the solar spectrum peaks is lost if the junction is too deeply placed (ref. 9). It is therefore common to have the junction between 0.3 and 0.5 μm below the surface. More short-wavelength photons are then absorbed near the junction, where an efficient collection of carriers results.

Of prime concern to the power supply designer is the cell efficiency η. Cell efficiency is generally given as the ratio of maximum power output per unit area of cell to sunlight power incident on the unit area. This is required for determining the total amount of cell area needed to produce the required power. Theoretically, for silicon at 20°C, the efficiency should be about 22% (ref. 10); in practice, however, individual cells seldom exceed 12%, and oriented arrays in flight

can yield about 8–10% (efficiencies of nonoriented arrays may be as low as 1 or 2%). Many factors cause this disparity; some of the most important are

(1) Optical reflection and coverslide transmission losses (about 2%, with a blue filter) reduce the light available.

(2) Electrodes cover part of the surface (up to 10%), masking it to the incident light.

(3) A large fraction of the carriers generated far from the junction will not reach the junction because of recombination.

(4) Surface recombination of carriers reduces the number reaching the junction.

(5) The series resistance of the solar cell absorbs some of the power generated.

(6) The diode action of the solar cell reduces the current output.

More detail may be found in reference 8.

When the cells are mounted, further losses in efficiency often occur because of manufacturing tolerances, fabrication processes, cell mismatching, coverslide losses, connection resistances, shadowing by other parts of the array or spacecraft, and the angles introduced when not all of the cells are positioned to face the Sun at one time.

2.3.3 Electrical Output

The electrical output of a given solar cell is dependent on the intensity of sunlight and its angle of incidence, the temperature of the cell, and the resistance in the circuit. Illumination and temperature are largely determined by the mission environment and the design of the spacecraft for attitude and thermal control; these factors are treated in Section 2.4. The load or resistance completing the circuit is very important in determining the power output. This may be seen from figure 4, a sample of solar cell "*I–V* curves" necessary for array design. The figure shows typical curves for current outputs of 2 × 2 cm *n/p* solar cells versus the voltage across the cells. The power provided by a solar cell is determined by the *I–V* curve and the resistance of the external load, using conventional circuit theory.

2.3.4 Effects of Cell Thickness and Area

While the use of the thinnest possible solar cell is desirable on spacecraft because of the weight savings, the efficiency and current due to carrier generation in each cell decrease with decreasing thickness, so that a greater number of cells are required for a given amount of power. Each added element contains not only the weight of the silicon, but also the weight of adhesive and coverslide, and possibly substrate and structural material if the panel area must be increased. In determining the desired cell thickness, therefore, the designer must determine whether there will be a net reduction of weight by using a thinner cell.

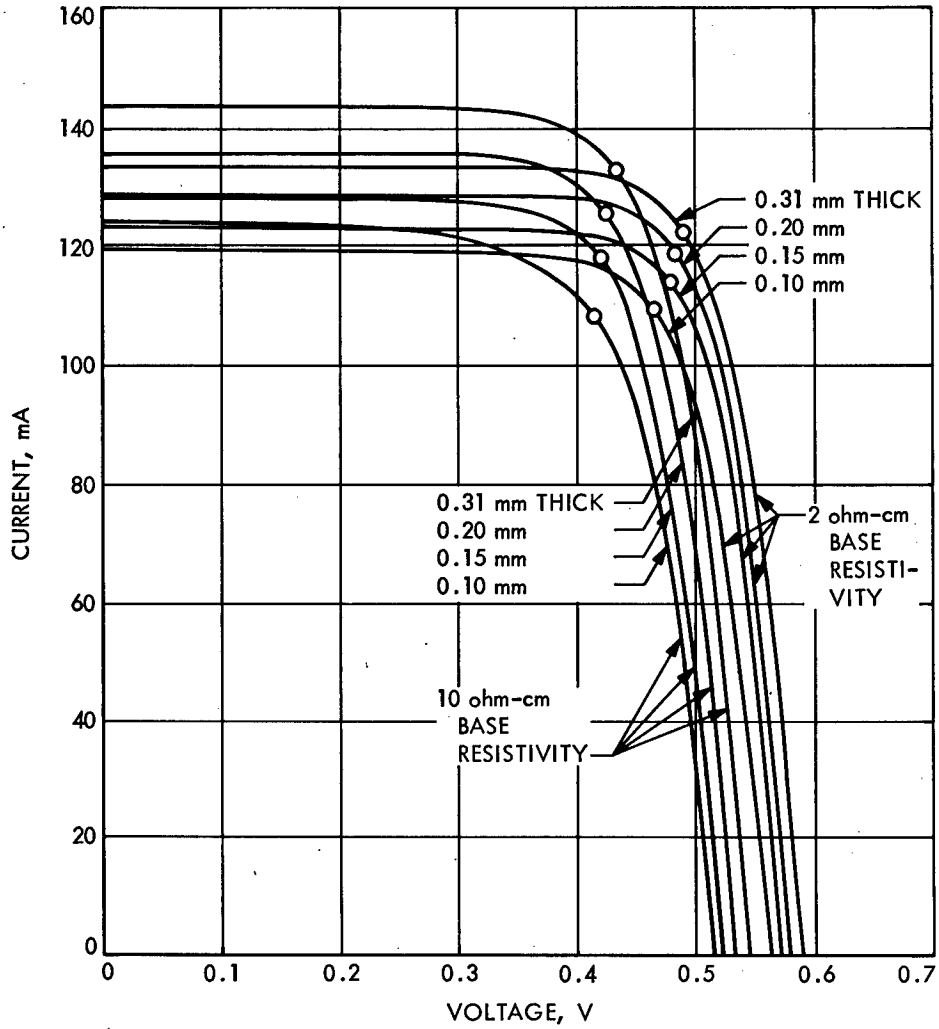

Figure 4.–Typical current output versus voltage across silicon solar cells, for several cell thicknesses and base region resistivities. Measured in 140 mW/cm² simulated sunlight, for 2 × 2 cm *n/p* cells at 28°C (ref. 11).

Figure 5 (ref. 11) shows the relationships between cell efficiency, resistivity, and thickness. A leveling off is seen to take place above 0.38 mm (15 mils). For thin cells, the rear surface condition influences the cell efficiency. It has been suggested (ref. 12) that with a good rear surface, thin cell efficiency would not decrease as much as indicated in figure 5.

There are practical limitations on the minimum cell thickness. Because of brittleness, manufacturing yields drop significantly as thickness is reduced below 0.20 mm (8 mils). Because the cost of an array rises in inverse proportion to the yield of individual cells, cells of less than 0.10 mm are impractical in the present state of technology.

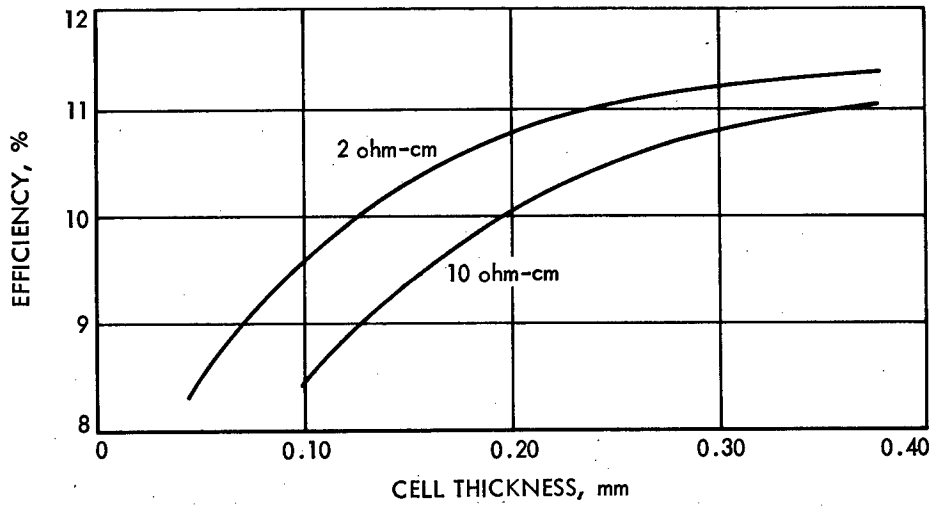

Figure 5.–Typical n/p solar cell efficiency vs silicon thickness for nominal 2 and 10 ohm-cm base resistivities. Sunlight simulator (X-25 L spectrosun) 140-mW/cm^2 balloon calibration; active area 3.9 cm^2; cell temperature 28°C (ref. 11).

The photovoltaic current developed in a cell is directly proportional to the illuminated surface area, i.e., the overall area of the cell, less the area covered by the contact material (collection strip and grid lines). As shown in figure 2, metal grid lines extend from the front surface contact to reduce series resistance. The design of the grid lines varies somewhat among manufacturers, but typically a 1 × 2 cm cell will have about a 1-mm-wide contact along one side. If the contact is on the 1-cm side, there will be three grid lines; if it is on the 2-cm side, there will be five or six grid lines. A 2 × 2-cm cell typically has six grid lines. Titanium overlaid with silver is generally used.

2.4 Effects of Space Environment

2.4.1 Intensity of Illumination

A cell directly facing the Sun receives the maximum possible sunlight. If cell orientation deviates from this direction by an angle θ, the incident energy varies nearly according to the cosine of θ and the output is reduced by approximately the same factor. As the angle increases beyond about 40 deg, edge effects, especially when a coverslide is used, cause the reduction to deviate slightly from this law. Measurement must be relied on for each specific geometry when accurate assessment is required for the output of solar cells illuminated at large angles from the surface normal. Both theory and experiment show that the short-circuit current varies directly with illumination intensity. Figure 6 shows the I_{sc} variation with illumination intensity determined experimentally for 0.2 mm, 10 ohm-cm n/p cells at room temperature.

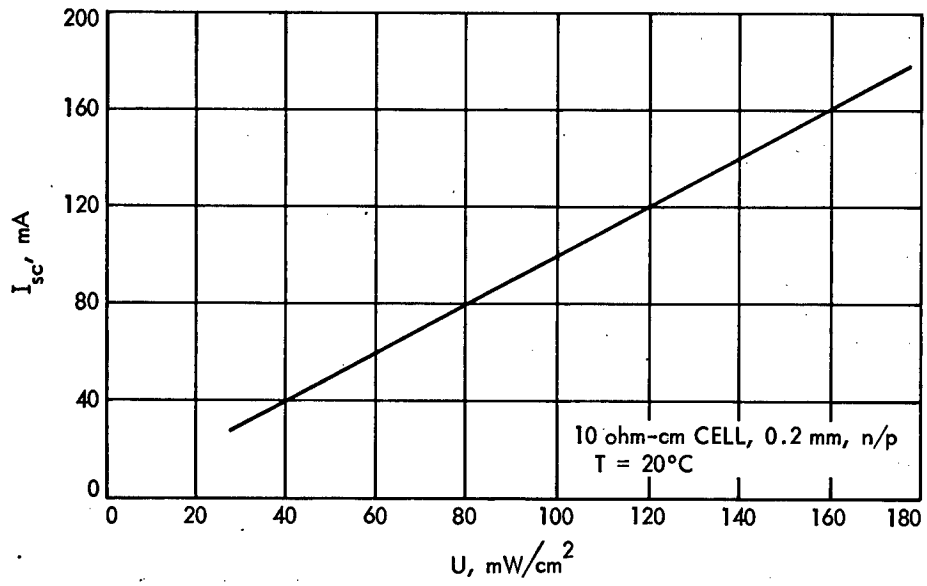

Figure 6.–Variation of short-circuit current I_{sc} with illumination intensity U (ref. 13).

The open-circuit voltage is affected to a lesser degree by light intensity. Measurements, as shown in figure 7, indicate a nearly linear relationship, with a slope of 0.2 mV mW^{-1} cm^{-2} above approximately 60 mW cm^{-2} (ref. 13).

The maximum available power at a fixed temperature is also proportional to illumination, as indicated by the measurements of figure 8. However, for near-Sun missions, where the illumination exceeds about 280 mW cm^{-2}, there is a tendency for the available electric power to saturate (ref. 14), unless care is taken to insure very low series resistances. Calculations of the effect of illumination under space conditions should include any change in temperature of the cells due to change in illumination. (In figures 6, 7 and 8, the temperature was held constant.)

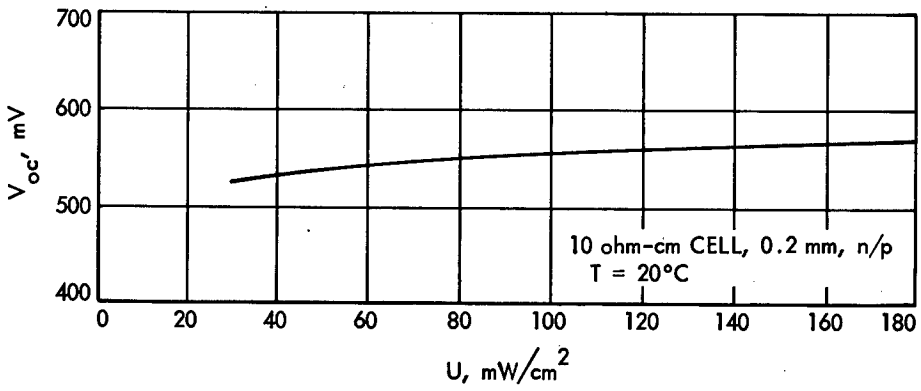

Figure 7.–Variation of open-circuit voltage V_{oc} with illumination intensity U.

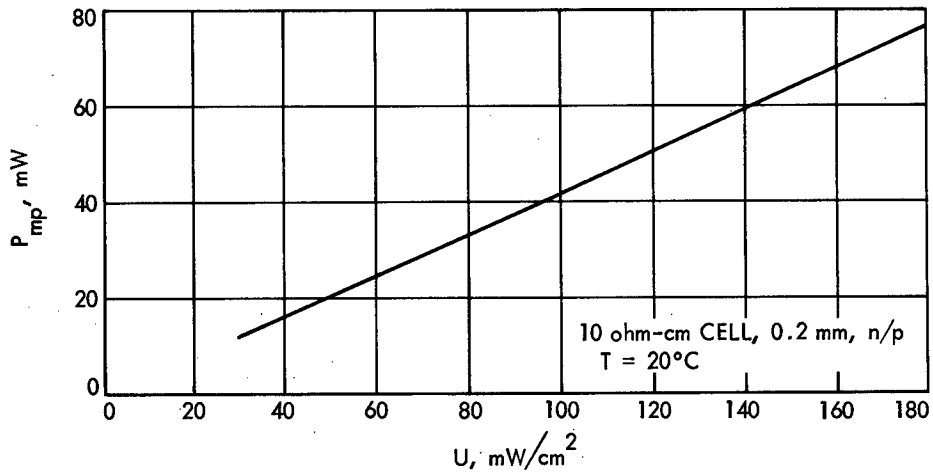

Figure 8.–Variation of maximum power P_{mp} with illumination intensity U.

2.4.2 Effect of Temperature

In general, the short-circuit current I_{sc} increases slowly with temperature. Values varying from 10.0 μA/cm^2-°C to 56.4 μA/cm^2-°C (refs. 11, 15, 16) are given by different experimenters. The specific temperature coefficient for any given lot of cells is a function of the manufacturing process and is measured if an accurate knowledge of this parameter is required. The temperature coefficient of I_{sc} increases with prolonged exposure to radiation (ref. 17).

The temperature coefficient of the open circuit voltage V_{oc} was found by the authors of references 11, 15, and 16 to be −0.5 mV/°C to −3.5 mV/°C. The change in V_{oc} is due primarily to the change in the diode current. While the temperature dependence of the diode current in the dark is well known, it has not yet been fully analyzed under conditions of illumination. Figures 9–11 show the temperature variation of I_{sc}, V_{oc}, and P_{max} for a typical cell under several illuminations corresponding to solar intensities found between the orbits of Venus and Jupiter.

A mechanical effect of temperature on solar cell arrays is caused by the different coefficients of thermal expansion of the various components. Because silicon has a low coefficient of thermal expansion as compared with most materials, it requires flexible connectors to substrate and circuitry.

2.4.3 Effects of Energetic Particles

The trapping region of the magnetosphere contains a population of protons and electrons known as the Van Allen Belt. The density of particles is greatest at altitudes of 3700 to 18,500 km above the equator, with the belt extending with decreasing densities and varying energy spectra beyond the altitude of synchronous satellites at 42,600 km. Beyond synchronous altitudes and in deep

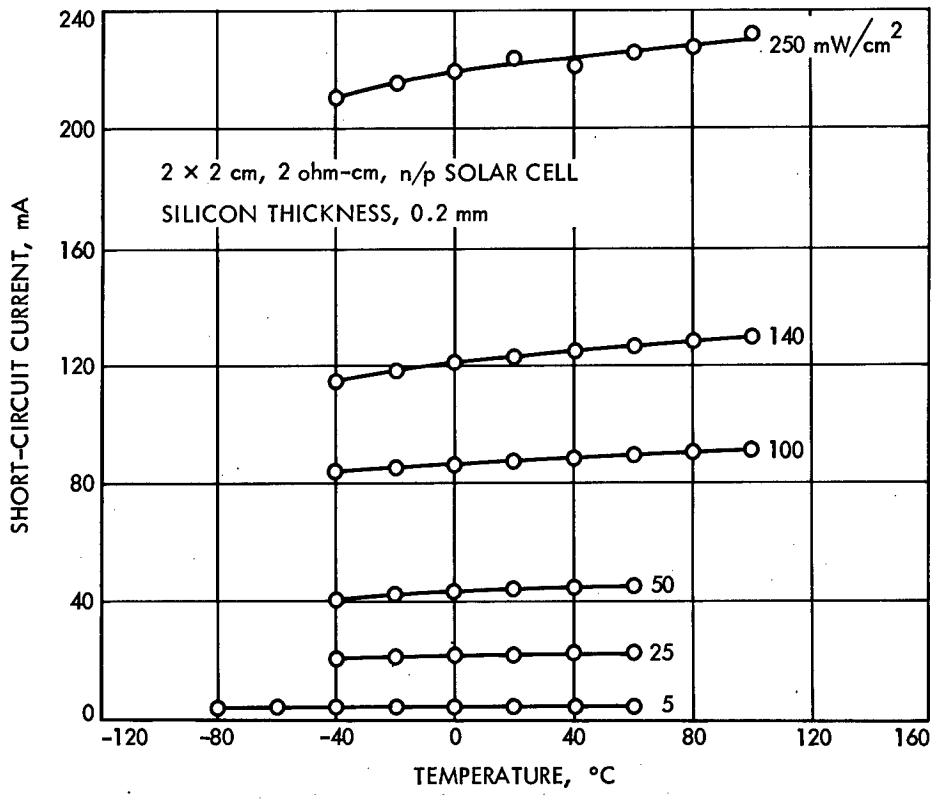

Figure 9.–Measured variation of short-circuit current with temperature (ref. 17).

space, the radiation environment consists of the very-low-energy "solar wind" of protons, occasionally accompanied by severe bursts of high-energy protons from solar flares. Aside from the earth, only the planet Jupiter shows evidence of having a trapped belt of protons and electrons.

Exposure to protons of energies above a threshold about 100 eV and electrons above about 200 keV results in radiation damage to solar cells. These energies represent the approximate threshold for the particle to knock a silicon atom from its position in the crystal; this leads to damage that is evidenced principally by a decrease in the photovoltaic current and a deterioration of the diode junction properties. Conventional *n/p* silicon solar cells lose about 10% of their maximum power when exposed either to 10^{11} protons cm^{-2} of 5-MeV energy, or to 10^{14} electrons cm^{-2} of 1-MeV energy. Thin solar cells lose efficiency at a slower rate: measured power outputs of 0.10- and 0.40-mm solar cells, after each was exposed to 10^{15} electrons cm^{-2} of 1 MeV, were nearly the same (ref. 12). These figures are only estimates; in any actual situation the amount of deterioration is affected by the solar cell design and the coverslide, and there will be a spectrum of energies of electrons and protons to be considered.

Shielding by coverslides is especially effective when protons are the dominant cause of damage. A 0.15-mm fused silica coverslide will prevent all protons below about 4.3 MeV from penetrating the solar cell. This is frequently the major fraction of the protons encountered. Because of their

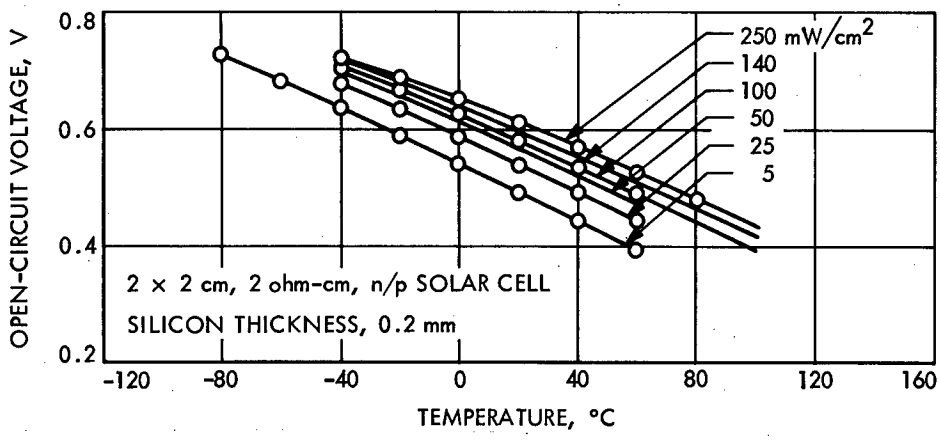

Figure 10.–Measured variation of open-circuit voltage with temperature (ref. 17).

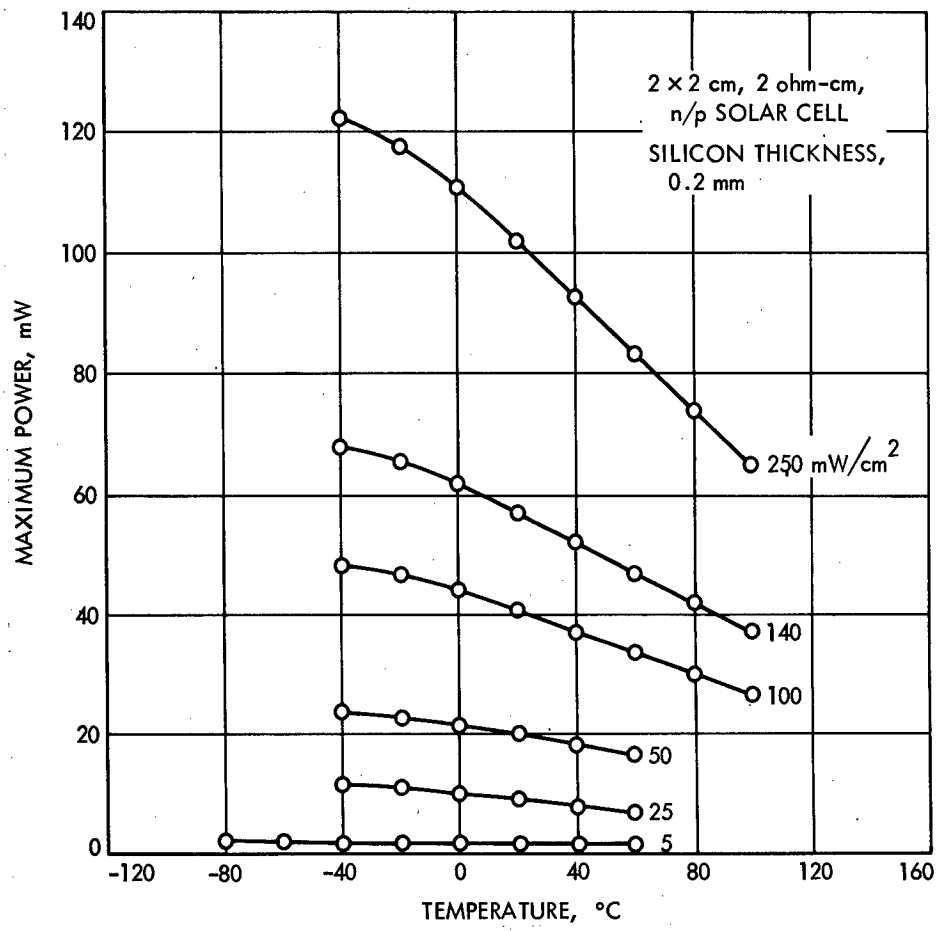

Figure 11.–Measured variation of maximum power with temperature (ref. 17).

longer range of penetration, electrons are less easily shielded, but the slowing down they undergo in penetrating a coverslide can provide a considerable effect due to the strong dependence of damage on the electron energy.

Damage by radiation is the principal cumulative effect that degrades solar cell array output in space. Extensive research into this effect has accumulated a wealth of information that permits fairly accurate calculations (ref. 18). The variability of the radiation environment, especially the solar flare protons, is perhaps the principal source of uncertainty in such calculations.

Some estimate of the results of combined electron and proton damage in space for a proposed solar cell array can be obtained from results of flight experiments. One such experiment, aboard the ATS-1 spacecraft in synchronous orbit, produced the results shown in figure 12.

Degradation also occurs in some optical filters and adhesives used with coverslides (see Sections 2.5 and 2.6). Broadband transmission losses of 3 to 4% were reported due to degradation of blue filters exposed to 10^{16} electrons cm^{-2} of 1.5 MeV (ref. 20). It was found that silicone-type adhesives are less affected by electrons than the epoxy type used on early arrays (ref. 21).

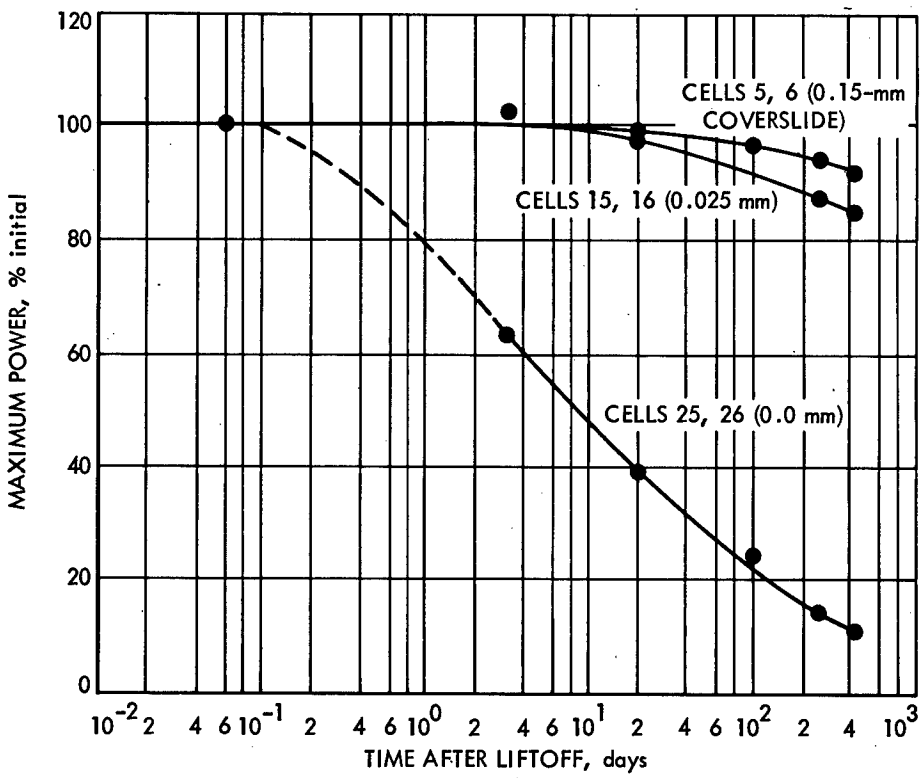

Figure 12.–Maximum power vs time for 0.30-mm-thick solar cells with various coverslide thicknesses (ATS-1, ref. 19).

2.5 Solar Cell Protective Covers and Coatings

In order to reduce optical reflection and the damage caused by radiation in space and to enhance the thermal emittance of the surface, solar cells are provided with coverslides. In the current technology, two materials are being used almost exclusively: Corning 7940 fused silica and Corning 0211 microsheet glass. The fused silica has a greater resistance to darkening by ultraviolet and electron radiation (ref. 22); in regions where thick coverslides are required, this may be a significant factor. On the other hand, microsheet is less expensive and quite adequate for use in regions where protection may be achieved with a relatively thin shield. Sapphire has been used on some satellites, notably Telstar, and is comparable to fused silica in radiation resistance, but it is generally considered to be too expensive for most projects.

The thickness of the coverslide is dictated by the amount and type of radiation to be encountered as well as by considerations of darkening and weight. Since coverslide thickness may be reduced by adding extra cells to allow for degradation, a tradeoff can be made between shielding and solar panel area. In general, satellites in synchronous orbit require only 0.15 to 0.25 mm (6 to 10 mils) of coverslide, while those spending appreciable time deeper in the trapped radiation region may require 0.5 mm (20 mils) or more. Because of the difficulties inherent in handling very thin, brittle materials, 0.15 mm is a practical lower limit for coverslide thickness.

Close tolerance must be held between the coverslide and the cell. If the coverslide overhangs the cell by more than a few millimeters, there will tend to be an excessive amount of breakage; if the coverslide is too small, the exposed cell area will degrade rapidly when exposed to low-energy protons. This occurred in late 1967 on Intelsat 2F4, ATS-1 and GGTS, where about 5% of the cell area was exposed. The coverslides, therefore, should be no smaller than the solar cells, and a careful match should be made. Several manufacturers have been investigating the use of integral coverslides (i.e., the glass is deposited directly on the silicon by vacuum deposition or sputtering) to eliminate the need for matching the parts. Cells with 0.025- or 0.05-mm integral glass covers have been made but have not yet been thoroughly flight tested.

To reduce reflection losses to about 1.5%, the outer surface is coated with an antireflective coating of magnesium fluoride. To reduce cell heating and adhesive degradation, the opposite side of the coverslide is coated with a filter that eliminates the ultraviolet end of the spectrum ("blue filter") or one that eliminates ultraviolet and infrared simultaneously ("blue-red filter") in wavelengths outside the range utilized by the solar cell (ref. 23). Filtering out the ultraviolet light reduces degradation of the bonding adhesive and is more important than filtering out the infrared energy input, which only causes an increase in the array temperature. For this reason, the cheaper blue filter is frequently specified.

The electrical efficiency of solar cells is enhanced by a thermal design of the array which provides the coolest possible operating temperature for the cells. Materials and combinations of materials are used which minimize the ratio of solar absorptivity to blackbody emissivity (α/ϵ). Both front and rear surfaces of the array should be considered for temperature optimization.

2.6 Adhesives

2.6.1 General Use and Characteristics

Adhesives are used to fasten the solar cell to the substrate and the coverslide to the cell. Silicones and epoxies are the types of adhesives normally used, and several commercially available forms with established performance records in solar cell array applications are often specified. The adhesives are uniformly applied in thicknesses of 0.025 to 0.050 mm (1 to 2 mils). The epoxy-based adhesives are often modified by adding a plasticizer to provide flexibility at low temperatures; silicones are flexible over a wide temperature range. Most adhesives have a limited storage life, so that the time from production of an adhesive to its application must be limited in accordance with the manufacturer's instructions (e.g., 6 months to a year). Even more limited is the "pot life," or the length of time that the adhesive retains acceptable properties after final preparation for its use (e.g., mixing with additives, etc.). Pot life is commonly of the order of one-half to several hours.

The bond to the substrate may in some cases need to be strengthened by coating the mating surfaces with a primer before applying the adhesive. Substrates of Kapton[4] are a notable exception, in which a primer does not enhance the bond. When a primer is not necessary to develop the required bond strength, it is omitted to allow for partial disassembly and repair of a panel after the acceptance tests and inspection. Primers are not used on the coverslide because of possible darkening by ultraviolet radiation.

The large range of temperature to which solar arrays have been subjected precludes the use of some adhesives. The most common limitations are the tendencies toward volatilization and a marked reduction of adhesive properties at the higher temperatures to which arrays may be exposed. Silicone adhesives are relatively stable at temperatures up to about 315°C, but above this, they are subject to degradation; the adhesive may depolymerize, soften, or begin to outgas. The limiting factor at the low end of the temperature range is known as the "glass transition temperature." Below this temperature, the adhesive loses its plasticity and the thermal expansion coefficient becomes extremely small. The glass transition temperature of adhesives depends on the composition of the specific material, (e.g., this temperature is about −60°C for the RTV-600 series and near −101°C for the RTV-500 series of adhesives).

Interactions between different formulations of adhesives have degraded their performance. Compatibility of adhesive materials, particularly during curing, is therefore an important consideration for successful fabrication of solar cell arrays. An example of this type of problem involved the use of an adhesive tape for temporarily securing electrical wiring and replacement coverslides during the rework of an array. Tape adhesives commonly contain sulfur, which is an inhibitor of cure in the adhesives used for coverslides. In this case, bonding of coverslides was not satisfactory until the problem was identified and non-sulfur-bearing adhesive tape was used in temporarily securing parts of the array during adhesive cure. In general, adhesives are sensitive to contamination; control of dust, vapors, and humidity is important.

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[4]A du Pont polyimide film.

2.6.2 Coverslide to Solar Cell Bonding

Coverslides are bonded to solar cells by means of a transparent adhesive. This adhesive must be optically clear, flexible over a wide range of temperatures, easily applied, readily cured, and relatively resistant to degradation by the ultraviolet, electron, and proton radiations that will be encountered by the solar cell array. It must provide a bond between cover and cell that will withstand the shock, vibration, and thermal cycling environments encountered during assembly, tests, transportation, launch and flight. At the same time, the bonds should be weak enough to permit removal of coverslides for repair of an array.

Clear silicone adhesives, when used with coverslides having ultraviolet interference filters, meet these requirements (refs. 21 and 24). They include Dow Corning's Sylgard 182 and its purified version DOW R63-489, Sylgard 184, and General Electric's RTV-602. Comparisons of these formulations show that Sylgard 182 gives the strongest adhesion although it requires a 60°C curing temperature; Sylgard 184 cures at room temperature, permitting replacement of individual coverslides without repeated baking of the array; and RTV-602 also cures at room temperature and is sometimes favored because its weaker adhesion more readily accommodates rework of an array to correct faults discovered during acceptance tests.

Recent evaluations of coverslide adhesives at the NASA Goddard Space Flight Center indicate that Dow Corning's 93-500 clear adhesive might be a preferred alternative when low outgassing characteristics are required to avoid condensation on optical surfaces of the spacecraft.

2.6.3 Bonding of Solar Cell to Substrate

For insuring that the solar cell adheres to either flexible or rigid substrates, a stronger adhesive than that used for coverslides is desirable, but transparency is normally not required. As well as the other requisite properties mentioned above for coverslide applications, the adhesive should have high thermal conductivity, provide electrical insulation between solar cells (although for metal substrates, electrical insulation is usually obtained by means of a dielectric material), and provide damping of mechanically and thermally induced disturbances in the solar cell assembly. Candidate adhesives often selected are RTV-118, RTV-40, RTV-41, RTV-511, RTV-560, RTV-3145 (all available from General Electric), and Silastic 140 (Dow Corning).

2.7 Solar Array Construction

2.7.1 Mounting of Solar Cells

Conventionally, solar cell arrays are assemblies of modules comprising groups of electrically interconnected cells fitted with coverslides and mounted on a substrate. A cross section of the elements associated with each cell in a module is shown in figure 13.

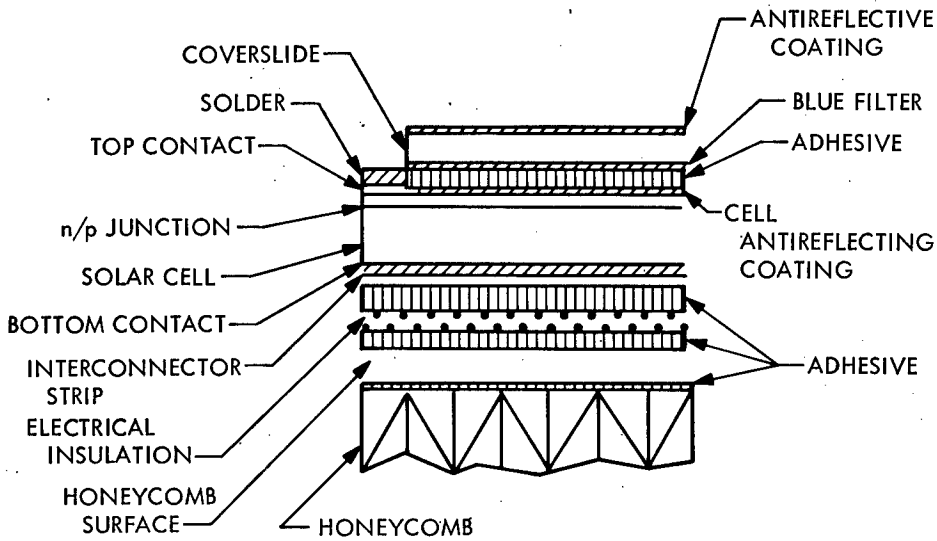

Figure 13.–Schematic of a conventional *n/p* solar cell element (ref. 25).

The principal module construction techniques are shingling and flat-mounting, shown schematically in figure 14. Shingling permits the top contact of each cell to be placed under the bottom of the next; enough cells are strung together in this manner to make up the design voltage for the module. Rigid shingling is being gradually abandoned, principally because defective solar cells cannot be replaced easily after assembly and test and because of poor thermal shock characteristics. Flat-mounted cells do not overlap. This module construction, permitting easier rework, requires interconnectors. Generally, the cells in adjacent rows are connected in parallel, and the rows of the module are connected in series. Flat mounting is becoming increasingly popular since its use on the Nimbus and Pegasus spacecraft.

Recently, a flexible shingling technique has been devised to permit close packing of solar cells in combination with the flexibility offered by interconnectors. This patented method of module construction, used on Intelsat satellites, is compared with the earlier techniques in figure 14.

The underside of the module generally contains the return electrical connection. This may be an expanded silver mesh (ref. 27) or a plastic foil with a printed circuit that completes the current loop as closely as possible to the cells to minimize magnetic field generation (ref. 28).

The three design choices for mounting solar cell arrays are body mounting, paddle mounting, and mounting on oriented panels. In all of these cases, a number of basic cell elements (fig. 13) are fabricated into modules, which are in turn assembled into arrays.

Body-mounted cells, i.e., cells that are cemented directly to the satellite skin or to light metallic sheets that are then attached to the satellite, have frequently been used on navigational and communications spacecraft. This method of mounting, which is the most inexpensive, is generally very reliable, since it minimizes the number of mechanical parts required. However, the efficiency of such an array is limited since only a fraction of the cells face the Sun at any time. As a first

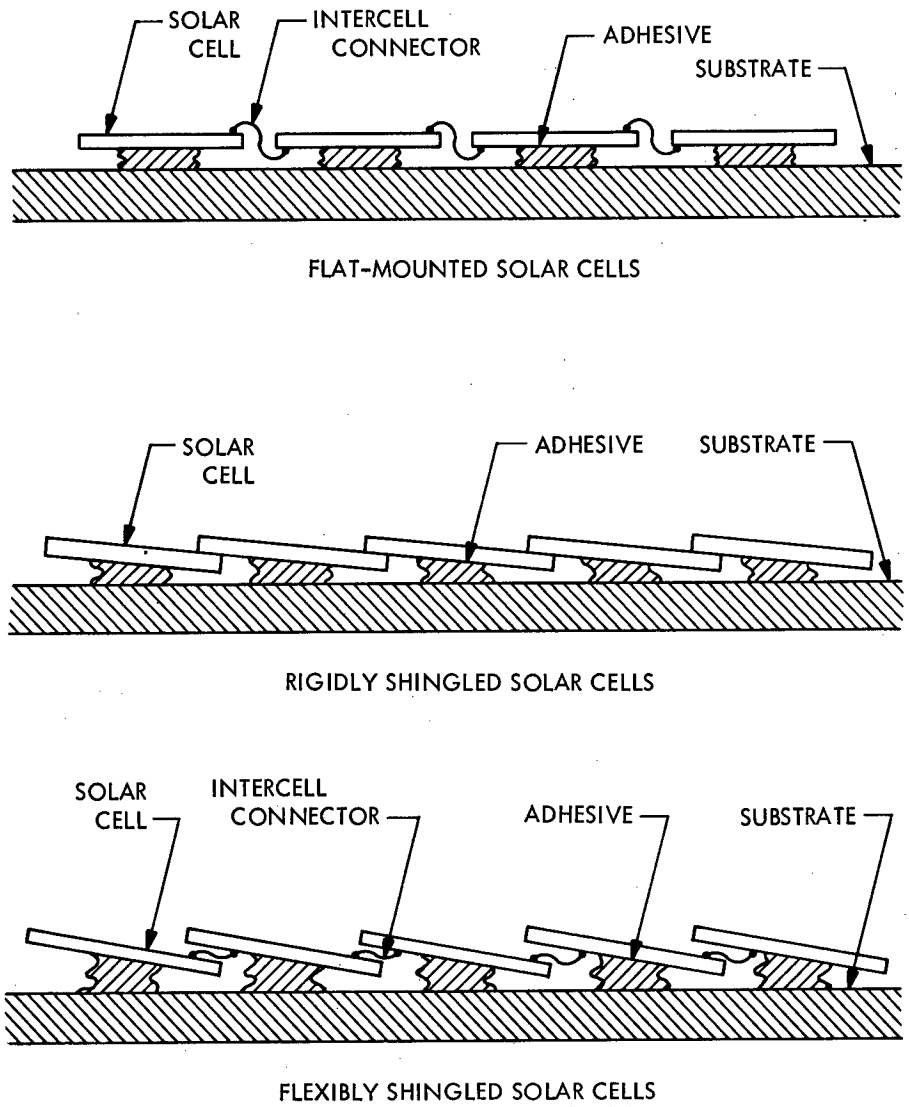

Figure 14.–Schematic portrayal of three panel construction techniques (ref. 26).

approximation, the amount of power available may be estimated by projecting the area covered by the cells onto a plane normal to the direction of incident sunlight and multiplying by the power generated per unit area of normal incidence.

Medium-power (40–50 W) spacecraft, such as the IMP series (IMP A through G), obtain additional surface area for their arrays by using paddles, with the cells bonded to the surface (figure 15). While this technique can provide more area than body-mounted cells, and hence increased power for given spacecraft size, the overall efficiency is not improved unless attitude control is employed to limit shadowing of the array. Actual efficiencies of paddle and body-mounted constructions are generally between 1 and 2%.

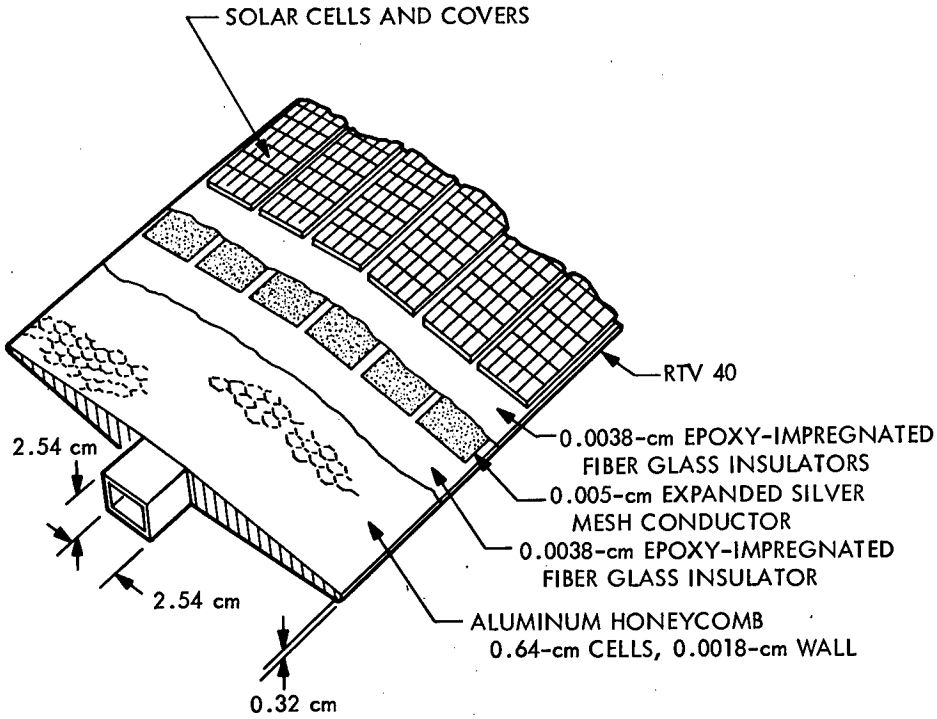

Figure 15.–Construction of the IMP-D solar paddles (ref. 29).

The most efficient array is the oriented panel. By use of attitude-control devices and drive motors controlled by Sun sensors, the panels are continuously kept pointing at the Sun. The full efficiency of the cell is still not obtained since the prolonged exposure to direct sunlight raises the temperature of the array, reducing the efficiency to 6–8% (ref. 30). Large thermal stresses are also encountered when the spacecraft enters the earth's shadow, and the temperature may, depending upon the duration of the eclipse, drop below −100°C. Figure 16, showing early measurements of the array temperature on a Nimbus spacecraft, illustrates the cooling of the array. The increased efficiency of the solar cells at low temperature causes a surge of power upon reexposure to direct sunlight. Protection devices must often be used to prevent burnout of the load or associated circuitry when this occurs.

Invariably, there is a drastic reduction in power output per kilogram as the solar cell array is constructed. A thin cell (0.10 mm) yields about 542 W/kg (246 W/lb); addition of a 0.076-mm coverslide and adhesive cuts this in half; series–parallel connections and addition of a substrate bring the result to about 110 W/kg (50 W/lb); final wiring and accessory hardware will reduce this to around 44 W/kg (20 W/lb) for a best effort. In actual practice, as seen from table 1, many oriented flight system arrays produce between 9 and 16 W/kg (4 and 7 W/lb). This drastic increase in weight as the solar cell is combined into a workable system in conventional designs has stimulated developmental work in deployable lightweight "roll-out" and "fold-out" techniques.

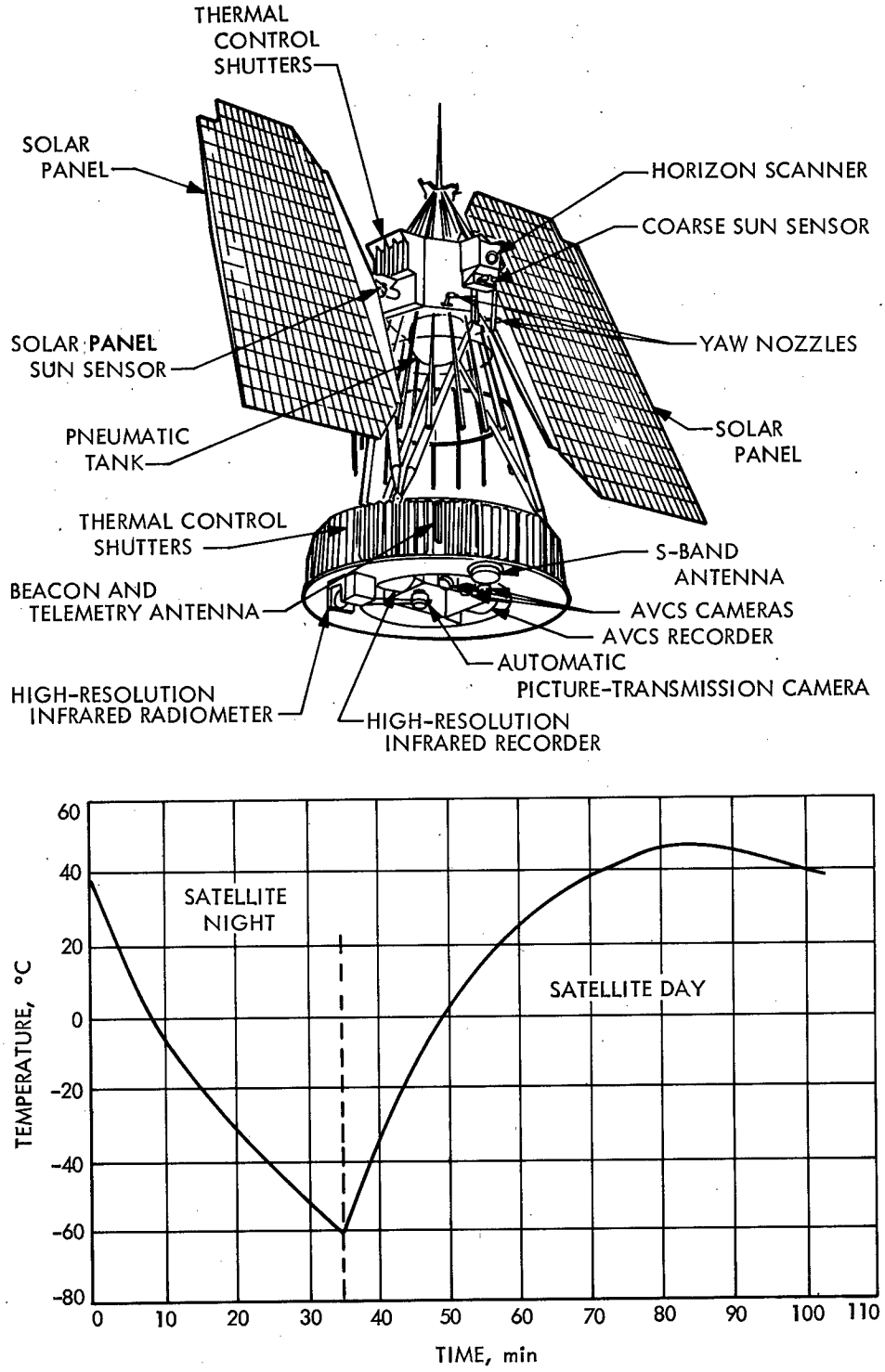

Figure 16.–The Nimbus 2, with solar cell array on two oriented panels, initially provided 450 watts at 40°C. The panel temperature, measured through one orbit, is shown above.

TABLE 1a.–Satellite Photovoltaic Power System Characteristics (ref. 31)

Item	Nimbus "B"	EGO	OSO	OAO	UK-1	Explorer 12	Explorer 14	Relay	Tiros	IMP
Type of array[a]	O	O	O	N.O.	N.O.	N.O.	N.O.	N.O.	N.O.	N.O.
Number of solar cells	11 000[b]	32 500	1872	53 000	4256	5600	6144	8400	9120	11 520
Cell mounting	Panels	Panels	Panels	Paddles	Paddles	Paddles	Paddles	Body	Body	Paddles
Weight, kg										
Array	29.1	57.7	2.4	101	4.0	5.0	5.8	11.7	11.1	12.0
Storage	51.3	33.6	13.9	80.4	6.4	2.9	2.9	12.7	18.2	3.0
Power system	80.4	91.3	16.3	181.2	10.4	7.9	8.7	24.4	29.3	15.1
Maximum power developed, W	410	560[f]	31[c]	772[d]	11.7	20.4	34.7	35	51[e]	74.3
Power/weight, W/kg										
Array	14.1	9.7	13.2	8.0	2.9	4.1	5.9	3.1	4.6	6.2
Power system	5.1	6.2	1.9	4.3	1.1	2.6	4.0	1.4	1.7	4.8
Array area, m^2	4.0	7.3	0.4	17.4	1.0	1.4	1.4	1.6	1.6	2.7
Array power/area, W/m^2	102.3	77.5	82.9	44.6	11.8	24.8	24.8	21.5	31.2	28.0
Array efficiency, %	7.3	5.5	5.9	3.2	0.85	1.0	1.8	1.5	2.2	2.0

EGO = Eccentric Orbiting Geophysical Observatory
OSO = Orbiting Solar Observatory
OAO = Orbiting Astronomical Observatory
[a]O = oriented; N.O. = not oriented
[b]2 × 2-cm cells
[c]At 70°C
[d]Average power from maximum to minimum expected operating temperature
[e]At $\pi/4$ rad (45 deg) inclination to Sun
[f]At 60°C

TABLE 1b.–Interplanetary Spacecraft Photovoltaic Power System Characteristics (ref. 32)

Item	M 62 (Venus)	M 64 (Mars)	M 67 (Venus)	M 69 (Mars)	M 71 (Mars)	M 73 Venus/Mercury	VO 75 (Mars)
Weight, kg							
Array	23.6	34.01	28.6	50.34	50.34	26.30[a]	71.65[a]
Storage	14.9	14.97	14.9	16.78	29.45	29.47	68.93[a]
Power system	10.8	14.97	17.2	22.67	21.74	15.42[a]	19.50[a]
Maximum power developed, W							
Earth	200	690	400	864	850[a]	478[a]	1310[a]
Encounter	270	320	600	470	475[a]	600[a]	562[a]
Power/weight,[b] W/kg							
Array	8.74	20.3	14.0	17.16	16.88	18.17	18.28
Power system	7.78	23.0	12.4	21.90	16.60	10.65	14.81
Array area, m^2	2.69	6.5	4.09	7.71	7.71	4.51[a]	14.68[a]
Array power/area, W/m^2	74.35	106.1	97.79	112.06	110.2	105.9	89.24
Efficiency,[c] %	5.5	7.8	7.2	8.3	8.1	7.8	6.61

M = Mariner; VO = Viking Orbiter

[a]Estimated

[b]Assuming power = value at Earth

[c]Using power flux at Earth = 135×10^{-3} W/cm^2

2.7.2 Interconnections

Thin metal interconnectors are used to make electrical connection between individual cells. A major cause of performance loss for solar cell arrays is the breaking of these interconnectors, leading to open strings and loss of current. The failure mechanism is often a cracking of the metal at the connection interface due to stresses either from mechanical vibration or thermal cycling. Test data, such as those shown in figures 17 and 18, are useful to relate breakage to the number and severity of thermal cycles, as well as the material used.

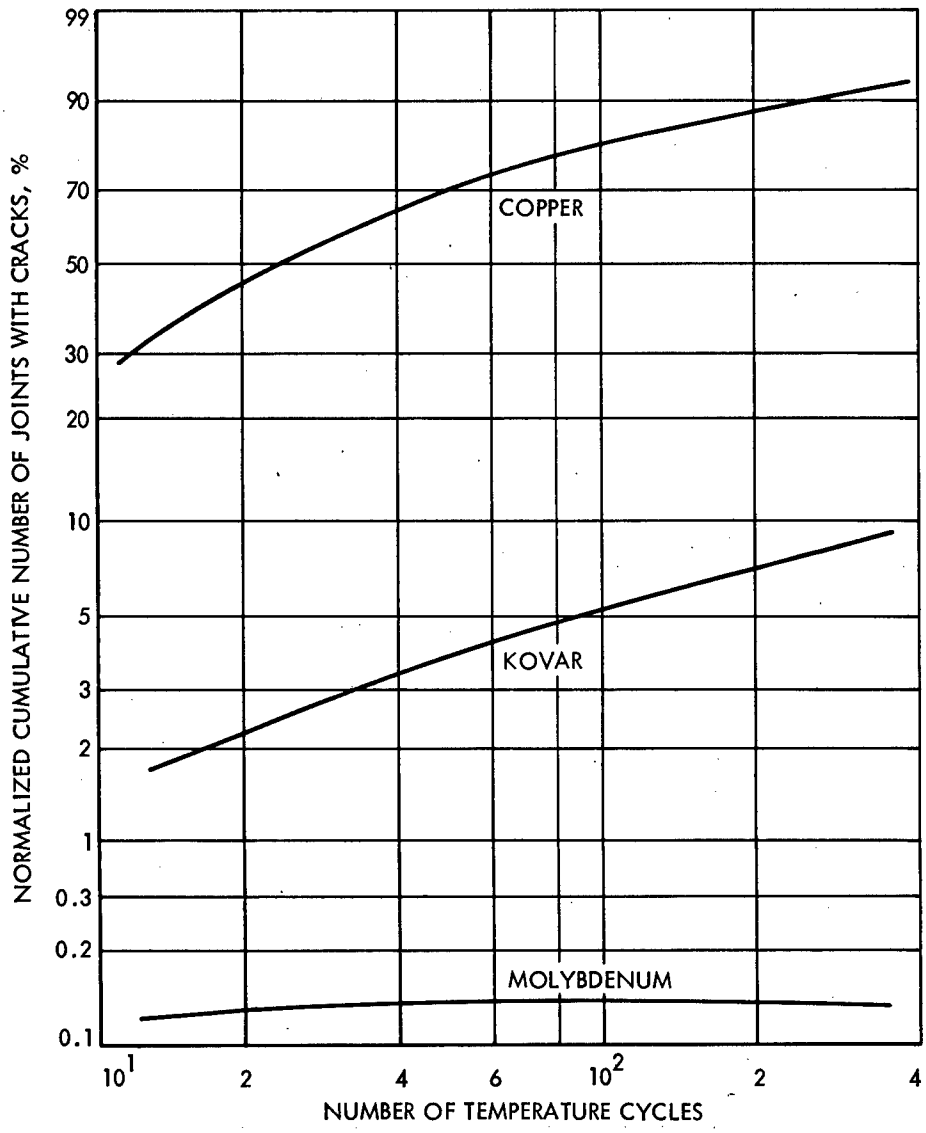

Figure 17.–Cumulative number of joints with cracks (Weibull Probability Scale) vs number of temperature cycles (+75°C to −175°C) for three interconnector materials (ref. 33).

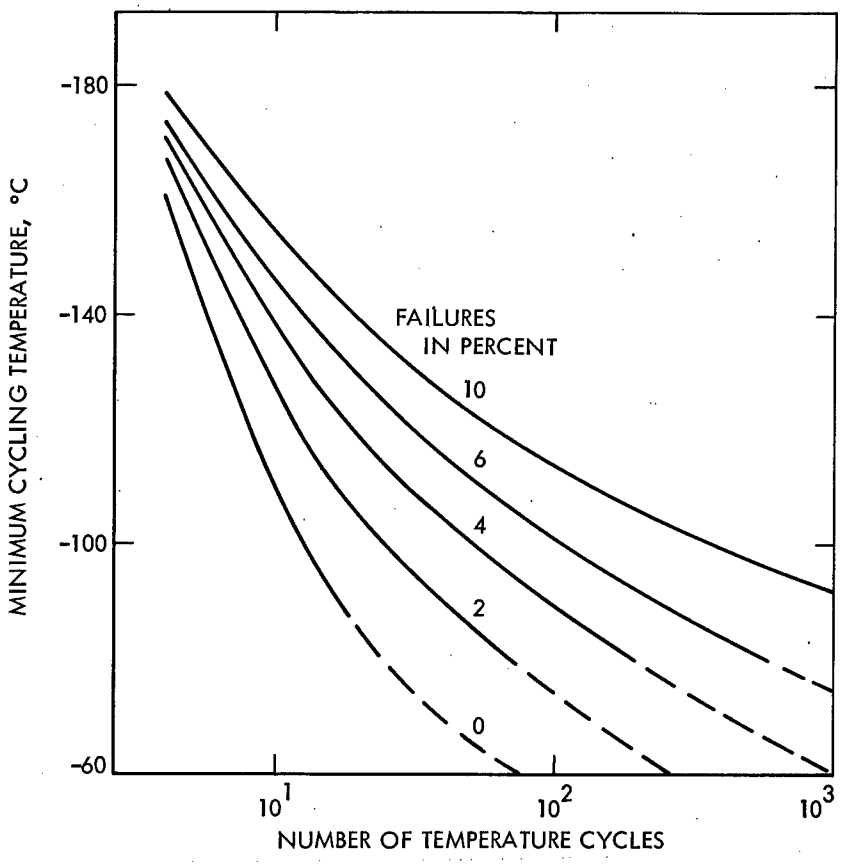

Figure 18.–Minimum permissible cycling temperature vs required number of thermal cycles for constant joint failure levels (for joints with 0.07-mm-thick copper interconnectors) (ref. 33).

The number of materials suitable for interconnections is limited. To avoid stress cracking, the thermal expansion of the interconnector should approach that of silicon. Molybdenum, Kovar (an alloy of 29% nickel, 17% cobalt, and 53% iron), and Invar (an alloy of 35% nickel and 64% iron) have suitable thermal expansion properties. A mesh of either silver or silver-plated molybdenum, which is sufficiently flexible to accommodate much of the strain, is often selected (ref. 25). Copper interconnectors, joined under clean conditions to prevent formation of salt with the solder flux, have been found economical but have thermal excursion limitations. Silver or gold plating of interconnectors is used to reduce resistivity losses.

One type of electrical interconnector, used on a large number of spacecraft, consists of either expanded silver mesh or photo-etched silver mesh. Silver is used because of its ductility and its excellent soldering characteristics to the silver titanium contact. The mesh pattern reduces the problem of matching the thermal coefficient of expansion of the silicon and the stress relief loops required by solid interconnectors, because it is free to expand and contract with three degrees of freedom. It is formed in a simple "S" configuration so as to permit electrical connection to the *n* contact of one cell and the *p* contact of the adjacent cell in series.

Use of very thin material and redundance of interconnecting tabs are common practices that reduce the effects of breakage. Stress relief loops are usually included. As seen from figure 19, the design of a stress relief loop cannot be neglected. This design includes both the stresses due to relative expansion of solar cells and substrate and the stresses induced at the interface of the metal interconnector and the solar cell (ref. 35).

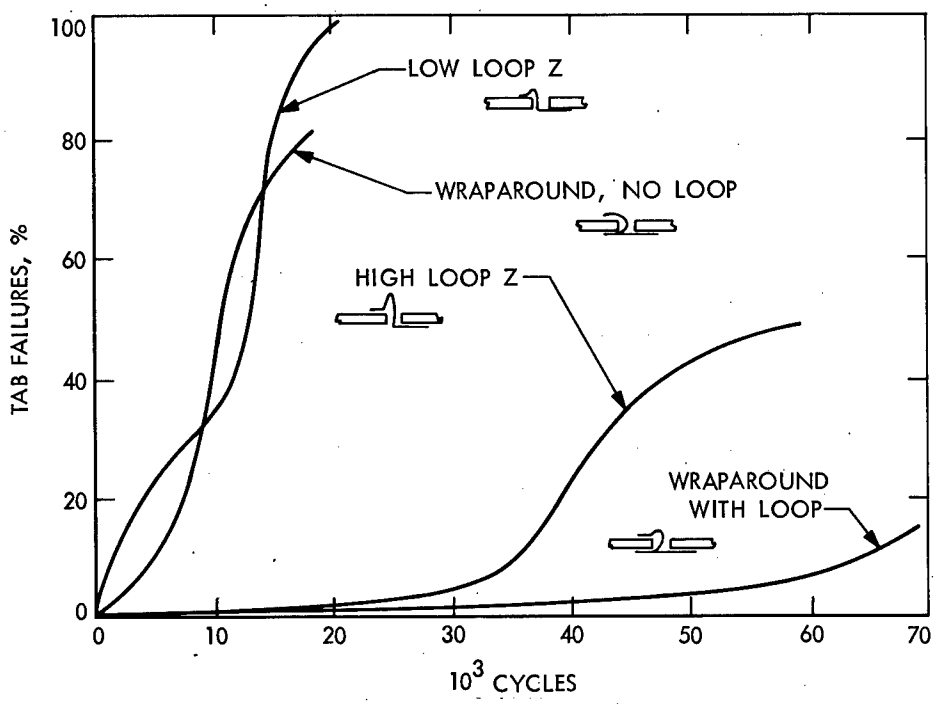

Figure 19.–Stress relief loops, interconnector flexure test results (ref. 34). The high loop is 1.3 mm high and the low loop is 0.4 mm high.

Currently, most interconnections on spacecraft solar cell arrays are held by solder. Resistance soldering, in which a soldering tip briefly touches and heats the parts and the solder alloy, is a versatile approach. Oven soldering, in which each module is assembled with solder in place, clamped, and passed through an oven, lends itself well to rapid production of large arrays and has been used successfully on the Mariner 1967 and 1969 and on the Surveyor 5, 6, and 7 programs.

Solder has limitations: steady temperatures over 125°C, or 10-min transients over 200°C cannot be tolerated (ref. 36); it interacts with some desirable materials; and can result in silicon fracturing during thermal cycling. Solder-dipping to coat Ti–Ag contacts on solar cells is a common practice which protects them from prelaunch corrosion by water vapor. For a solderless interconnection resistant to humidity, another method of contact protection must be devised. The addition of palladium (ref. 7) appears to be such a method and is being used for European Space Research Organization (ESRO) 4. Solderless interconnections can be made by resistance welding or thermocompression.

2.7.3 Electrical Characteristics of the Array

Groups of solar cells of 5 to 80 or more are usually connected to form modules which are the basic building blocks for arrays. The power production of the module is proportional to the total active area of solar cells. For an estimate of total array series resistance, wiring resistances may be neglected: the resistance of the module is then given by R_sS/P, where R_s is the individual cell series resistance (typically 0.25 ohm for 4-cm^2 cell) and where S is the number of cells in series and P the number in parallel. Cells in a module are generally matched, but complications can be introduced if shadowing is possible over part of the module. The power loss in such a case can be significantly greater than that computed on a percent shadow basis (ref. 37). By paralleling a row or several rows of parallel cells with a diode, the effect can be decreased.

The effects of shadowing on the current voltage characteristics of solar cell array circuits have been characterized and used for the development of mathematical models (ref. 38). The models describe circuits of any geometry, with or without shunt or blocking diodes, and can be used in analyses of electrical output of shadowed arrays based on either theoretical or empirical input data.

Each module of a solar cell array is often protected against reverse surges of current by a pair of blocking diodes. (Redundancy of diodes is provided for reliability because these components are susceptible to damage in environmental exposures such as vibration, shock, and thermal cycling.) The modules, mounted on panels, are wired together to form the solar cell array.

Because the power output of a solar cell array can fluctuate greatly in flight, a sensitive load must be protected from excesses and provided with an alternative supply from batteries. Such factors as temperature changes and array illumination angle are evaluated to find what fluctuation can occur. This is the basis for design of the power conditioning system. While a complete description of this system is beyond the scope of this monograph, its major properties should be noted.

Figure 20 shows two typical power supply circuits for spacecraft. The solar cell array provides power to the load and a trickle charge to a battery system. Switches determine when extra power is available, to be diverted to the battery for storage. There is a maximum safe potential that can be applied to batteries without causing excessive gassing and pressure buildup. When the potential rises above this, or when the batteries are charged, the surplus is dumped as heat by a power shunt. In many designs, the load can be turned off if the primary voltage becomes too low.

2.8 Testing

The successful performance of a spacecraft power system is dependent on the adequacy of the panel manufacturer's quality assurance program. Qualification tests and acceptance tests have great importance because their results give the most direct evidence of the level of accomplishment in a quality assurance program, prior to the actual flight of the space vehicle. These tests provide the means for assuring that the solar cell arrays will operate satisfactorily for a particular spacecraft and mission. The detailed requirements of these tests are specified in the procurement

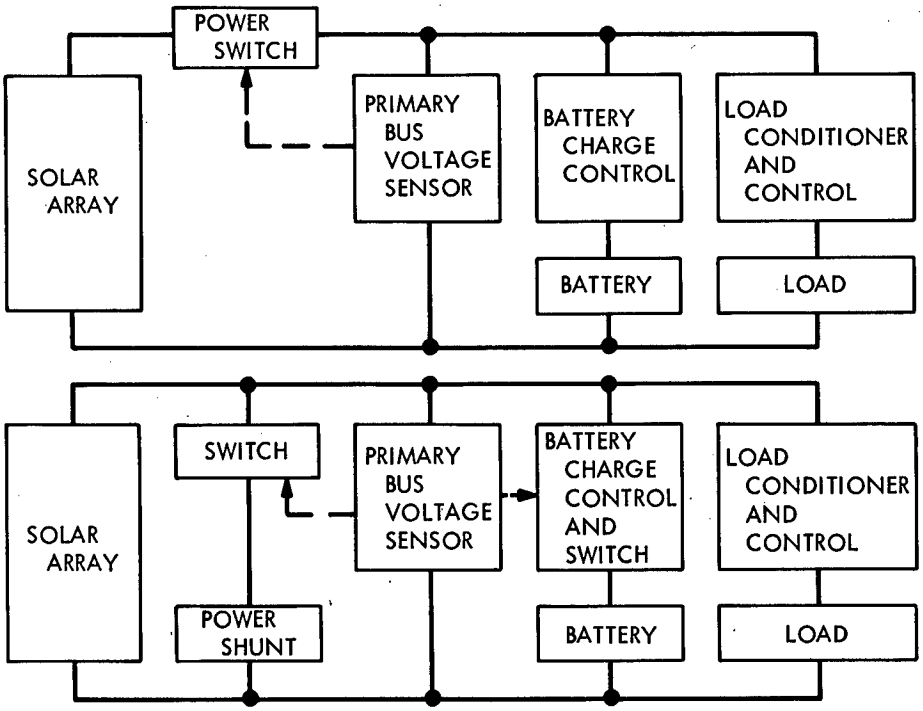

Figure 20.–Schematic diagrams of two typical power supply circuits for satellites (ref. 26).

contract for the solar cell arrays. An example of performance evaluation procedures for use in a program of qualification and acceptance tests is given in reference 39.

Qualification tests are devised and performed to demonstrate the adequacy of a design. Representative samples of the elements of the solar cell array are fabricated or chosen from production lots in accordance with design specification and subjected to laboratory simulation of all environmental conditions anticipated for the flight hardware during shipment, handling, launch, ascent, and orbit or interplanetary trajectory. Some of the qualification tests, particularly in the mechanical area, such as vibration and bending, are designed to impose increasing stress levels or durations until failure of the test item occurs. In such cases, qualification is demonstrated if failure occurs only after a stress level which exceeds by a specified margin the anticipated exposure in the prelaunch, launch, and mission environments.

Even when test-to-failure is not part of the qualification requirement, the tested items that survive are not subsequently installed on the flight vehicle, having served their purpose by demonstrating the strength and weaknesses of the design. Examples of the types of tests commonly required to demonstrate qualification of solar cell array design are given in table 2. The criteria for qualification are minimum acceptance values for the performance characteristics of particular assemblies, subassemblies, and elements following exposure to the specified test conditions. The environmental limits given in table 2 do not pertain to any particular space vehicle or mission but are a composite representation of some of the more severe requirements applied in past programs. Qualification tests are individually tailored to each program or, where appropriate, to specific spacecraft and missions within the program and are developed in detailed specifications.

TABLE 2.–Sample required qualification and test performance elements for solar panels (based on ref. 40)[a]

Element	Required performance
	a. Sample qualification test elements
Humidity	Two cycles simulating prelaunch exposure: Method 507 (MIL-STD-810A)
Acceleration	147 m/s² (15 *g*) for 5 min in direction parallel to axis of launch vehicle
Vibration	Broadband random vibration, overall average 196 m/s² (20 *g*) rms for 3 min in each of three orthogonal directions
Acoustic	Broadband random incidence sound field with an overall sound pressure level of 155 dB
High temperature	2 h at 107°C in dry nitrogen gas, or simulated eclipse conditions
Temperature cycling	Five cycles between −101°C and 43°C (maximum $dT/dt = \approx 4$°C/min)
	b. Sample acceptance test elements
Visual	Check for cleanliness, cracks, excess solder, alignment and fit of coverslides, etc.
Dimension	Fit of mounting holes with mating parts; compatibility with expected maxima of thermal expansion and contraction
Weight and balance	Within tolerances of spacecraft specifications
Electrical	*I–V* curve with 1-Sun illumination; cell-to-substrate resistance >50 MΩ
Vibration	Broadband random vibration, overall average of 98.1 m/s² (10 *g*) rms for 1 min in each of three orthogonal directions

[a]Actual test specifications depend on anticipated mission environment and may differ considerably from these examples.

The electrical output of the test modules or panels is checked before and after each qualification test to measure the change in performance. Performance data are also taken periodically during the tests and typically include four load points on the *I–V* curve, the temperature of the cell, and the short-circuit current. Between performance measurements, the cells are loaded to nearly the maximum power point (see Section 2.3.3).

Test specimens usually are given performance checks while illuminated by an artificial light source. The light sources are specified (often tungsten or tungsten–halide lamps) to insure reproducibility of spectral distribution and intensity and thus enhance the detection of small changes in electrical output that result from the qualification tests. Standard cells having the same spectral response as the production cells are used to check the spatial and temporal uniformity of the light source. Where the test specimens are too large for effective illumination by artificial sources, the performance check is made with sunlight (usually uncollimated) on a comparative basis with a calibrated standard cell (ref. 26). High-altitude locations are preferred for natural sunlight illumination to obtain an approximation to the spectrum of sunlight in space. Precautions are necessary to insure that the illumination for tests does not introduce differential effects on the various cells in the sample; reflection of light from nearby structures, equipment personnel, or the test-bed aircraft, balloon, etc., must be prevented (ref. 39).

After qualification of the design for a solar cell array and its individual components, the parts, subassemblies, and assemblies intended for deployment on the mission vehicle are subjected to acceptance tests. Examples of the types of tests prescribed for acceptance evaluations are given in table 2. The requirements and limits are typical of many programs but do not represent standards; each program has slightly different requirements. Each module intended for flight is acceptance tested to insure that faulty elements are repaired or replaced. Identification of the nature and exact location of a fault may be obtained through visual inspection or one of the various nondestructive detection methods that have been developed (e.g., sequential shadowing of individual cells of an illuminated panel, circuit continuity checks with ohmmeter probes, and various current-detecting techniques). The acceptance test procedures are repeated after repair or replacement of faulty elements.

Caution must be exercised when developing the requirements for acceptance testing to avoid test conditions that would degrade the reliability of the flight hardware.

3. CRITERIA

Spacecraft solar cell arrays shall be designed so that, in combination with any other source provided, the required electrical power shall be available to meet all mission objectives under all specified operational conditions. The design of the array and the associated systems of the spacecraft must take into account all factors that can degrade reliable performance. These factors include the external environment, the various interfaces between the array and the spacecraft, possible interference between the array and other subassemblies of the spacecraft, the characteristics of incident light throughout the mission, and the functional characteristics of the array. Demonstration that the design of the solar cell array is entirely suitable for its intended mission should be made by a suitable combination of analytical studies and experimental tests.

3.1 Assessment of Design Objectives

Prerequisites to an evaluation of design objectives include definitions of the electrical power requirement for the mission and the constraints on design because of orbit parameters (or trajectory), spacecraft size and configuration, and the launch vehicle. These definitions are basic to the explicit criteria applicable to a solar cell array and shall be determined prior to the selection of a specific design approach.

3.1.1 Power Requirements

A conservative estimate of the electrical power needs of the spacecraft should be made as the initial step in the design process. The estimate should include:

- The length of the mission and the power required to perform the last operation in the mission.

- The frequency, durations, and level of peak power demand.

- The frequency, durations, and requirements of planned "powered-down" configurations of the spacecraft.

- Required output voltages and load impedances for all planned operating modes.

The total power requirement estimated at the beginning of the spacecraft design process should be adjusted to anticipate an increase of at least 10% to accommodate the usual "growth" of power-consuming operations and equipment into the final configuration for the mission. It should also be recognized that battery charging efficiency may be 65% or lower in the later stages of the mission.

The initial assessment of required solar cell area in the array should be made from the estimated power requirements and with due consideration for the efficiencies available with various proven designs of cell configurations. As the design progresses, the area and configuration evaluations must be iterated to account for performance-degrading effects of mission and spacecraft characteristics treated in the following parts of this section.

3.1.2 Trajectory or Orbit Characteristics

The available input power to the solar cell array is dependent on the intensity, angle of incidence, and periods of occultation of solar illumination. This input should be evaluated through analysis of the trajectory or orbit of the mission.

The orbit or trajectory parameters should also be applied, together with available information on the spatial distribution of cosmic and trapped-particle radiation, to evaluate the exposure of the solar cell array to this cause of damage throughout the mission.

3.1.3 Spacecraft Design Constraints

The interface conditions between the solar cell array and other elements of the spacecraft systems should be specified, where possible, by interface definition documentation. It shall be demonstrated that both the array and the spacecraft elements with which it interfaces will perform satisfactorily when the specified conditions are met.

It shall be demonstrated that the performance of the array will not be adversely affected by the presence or the operation of other spacecraft systems. Factors to be considered include:

- Electrical interfaces with spacecraft electronics and power conditioning equipment.

- Reflections from the spacecraft, which may enhance power output but may cause localized heating.

- Shadowing of the array by booms, antennas, or other parts of the spacecraft.
- Mechanical interfaces, including stability, rigidity, relative motions, deployment, alignment, and access.
- Radiation interfaces with radioactive devices (nuclear power or calibration sources).
- Spacecraft attitude control and, where applicable, array orientation, requirements, capabilities, and method of operation (including efflux from thrusters).

3.1.4 Launch Vehicle Constraints

The launch vehicle for the mission which the array will power must be taken into account because of constraints it imposes with respect to:

- Total weight limitation for the array.
- Dimensional limitations on the launch configuration of the array.
- The characteristics of the vibration, thermal, acoustical, and gravitational stresses that will be imposed on the array during launch and separation from the launch vehicle.

It shall be demonstrated that the array design is compatible with the specifications for the interface between spacecraft and the launch vehicle and that the array will survive the launch without degradation of subsequent performance in the mission.

3.2 Components of the Solar Cell Array

It shall be demonstrated that the elements selected for assembly into the solar cell array comprise an optimal configuration which meets the power requirements defined for the mission and accommodates the constraints of environmental stresses, interfaces with the spacecraft/launch vehicle, and program limits of cost and time.

3.2.1 Coverslides

The coverslides specified for the array should provide the following features:

- Dimensional tolerance which insures complete coverage of the solar cells but avoids excessive overhang that can result in large losses by breakage during assembly or by thermal expansion during flight conditions.
- An effective antireflective treatment of the outer surface to minimize loss of solar illumination in the wavelength range of 0.4 to 1.1 μm.

- An ultraviolet filter to reduce photodegradation of underlying adhesive to an acceptable level.

- An effective shield that will protect the solar cells from excessive degradation by corpuscular radiation and that will not itself degrade to cause excessive light transmission losses throughout the mission.

3.2.2 Adhesives

The adhesives used to bond coverslides to solar cells and the cells to the substrate shall be demonstrated to have adequate bonding strength after exposure to the prelaunch, launch, and mission environments. The adhesive for the coverslide should be selected to insure that darkening, embrittlement, and other forms of degradation will not cause unacceptable losses of array performance in the mission. The adhesive for the solar cell should be selected to insure a high reliability of adhesion during the spacecraft mission. Both adhesives should be selected to provide sufficient flexibility to permit different thermal expansion of mated surfaces.

The factors which most commonly cause difficulty and which adhesives must withstand without excessive degradation include:

- High humidity in the prelaunch environment.

- Ultraviolet and corpuscular radiation.

- High and low temperatures during storage, launch, and the mission.

- Repairability – replacement of either cell or cover, or both.

The handling and application of adhesives should follow procedural requirements that will insure that their optimal properties are achieved in the solar cell array. These requirements shall take into account the following:

- The shelf life of the adhesive.

- The "pot-life" of the adhesive (i.e., the length of time it is permitted to stand ready for use in a container).

- Temperature and humidity exposure of adhesives from the time they are used until the array is launched on its mission.

- Risks of contamination during application.

- Chemical compatibility between different types of adhesives and between adhesives and other materials used in the array (particularly during application and cure of adhesive compounds).

3.2.3 Solar Cells

It shall be demonstrated that the photovoltaic cells selected for the array are adequate to provide the required end-of-mission power. The characteristics of the solar cell shall be compatible with the following constraints:

- When in combination with the coverslide specified for the design, the cell shall be resistant to mission-impairing degradation from the expected corpuscular radiation environment.
- Dimensional tolerances shall be specified to optimize the trade-off between cost and difficulty of fitting coverslides.
- Cell area should be selected to minimize cost, but without unduly complicating installation of the cells.
- Cell thickness should be selected to optimize the trade-off between power output per unit weight and area and resistance to degradation by corpuscular radiation.
- The type of cell (base material, dopant, polarity, resistivity, etc.), and the design of its electrical grid and contact should be chosen to insure performance at the required output to the end of the mission while minimizing costs.

3.2.4 Circuitry

The circuitry that connects solar cells to form modules and connects modules to form the array shall be demonstrated to cause the cells to operate at the voltage of maximum power output at the end of the mission and shall have a geometry such that magnetic fields are minimized. Resistance between cells and between modules should be kept to a minimum. The series–parallel circuitry should be designed to minimize degradation of array performance in the event of electrical short or open circuits between cells. If the array will be only partially illuminated at times, the circuitry should prevent shunting through the darkened portion. Wiring should be insulated from the panel and should be protected with appropriate covering where abrasion may be a problem.

3.2.5 Mounting and Connecting Hardware

The electrical and mechanical links interconnecting cells and cell modules shall be demonstrated to withstand repeated thermal expansion and contraction associated with the expected maximum thermal excursions without impairment of array performance. The electrical interconnections shall be designed to minimize the effect of an open and short circuit because of imperfections in solder joints or environmental damage (e.g., corrosion or meteoroid strike). Redundancy of interconnectors should be used to insure reliability.

3.2.6 Substrate

The substrate should provide a surface that accommodates good adhesive bonding of the cell and a support that prevents deleterious flexure of the cells. The material and design should also be selected with consideration for desired structural properties and the total weight penalty to the spacecraft. Consideration should be given to the nature and bonding of the dielectric insulating layer with respect to stability and integrity, as dictated by mission requirements.

3.3 Testing

A program of tests shall be performed which satisfactorily establishes that the performance parameters of the solar cell array and all of its elements are within specifications under all expected operating conditions. The test procedures shall not introduce extraneous operating conditions or environments. Visual defects, in an otherwise acceptable solar cell array, should normally not be a cause for rejection. The test plan should provide for the following where applicable:

- Component and subassembly qualification tests.
- Calibration test.
- Qualification tests of a specimen array in simulated launch and mission environments.
- "End-to-end" field tests. Comparison of outputs from each section of the array.
- Acceptance test and criteria for the array and its elements.
- Prelaunch go/no-go test.

4. RECOMMENDED PRACTICES

The solar cell array is a major structural element of many spacecraft. It is therefore recommended that design of the solar cell array be instituted early in the development of a spacecraft design for a given mission. During the early phase, it is preferable to have several alternative designs which may differ in such details as size and type of solar cell, covers, circuitry, array shape, orientation, etc. These competing designs should be developed and analyzed in sufficient detail to identify which alternative is superior in meeting the design objectives.

A first decision point is the selection of the array geometry. Body-mounting onto the spacecraft is the simplest array, since it requires no extending apparatus. Extended arrays, with one or two degrees of freedom, have the advantage of better alignment with the Sun. These more complicated designs are warranted only when a body-mounted array will not suffice; the associated requirements for attitude control, as well as the initial unfolding from the as-launched configurations, contribute to reliability problems for oriented arrays on Earth-orbiting spacecraft.

Parametric studies are recommended for the early stages of the design effort. Components of solar cell arrays, like the rest of the spacecraft, are generally not "mass-produced," but are custom-made to specification from the spacecraft builder. Accordingly, a broad range of variations in design parameters is possible. To insure that the results of a parametric study will be useful, they should be discussed with manufacturing sources, who can provide information on the limits to the ranges of values for the parameters.

The efforts of the array design group should not cease when the design is "frozen." The preparation of formal specifications, following the design through the construction phase and test evaluation, are appropriate functions for this group.

4.1 Assessment of Design Objectives

Good design of a solar cell array for a specific spacecraft and mission is usually measured by the adequacy of the electrical power available to the spacecraft throughout the mission life. Constraints on the array design derive from features of the spacecraft design and the nature and duration of the mission: there may be weight limitations or upper bounds to the area of the solar cell array; there may be requirements for accommodating extended booms or antennas; the cost may become a critical factor; and, finally, specifications concerning radiation torques or on-board sources of magnetic field may be set. It is obviously important that the existence of any of these needs be determined early in the design of the solar cell array.

To insure that the array design is adequate, an evaluation should be made of the initial output of each candidate design. The effects of the launch environment on each design and the subsequent effects of the space environments should then be evaluated.

4.1.1 Power Requirements

To determine the area of solar cells required, determine the power output per unit area (power output per cell times number of cells that can be accommodated in unit area, taking into account required spacing between cells) under a nominal 0.15-mm coverslide, at the end of mission life. The calculation should include degradation factors related to (1) corpuscular radiation on the coverslide, adhesive, and solar cell, (2) the maximum solar cell array temperature expected at the end of mission life, (3) the probability of micrometeoroid and meteoroid impacts, if applicable, (4) losses attributable to the procedures of matching cells, assembly of the array, and testing, (5) prolonged thermal stresses throughout the mission, (6) and the effects of solar intensity U on power output at the end of mission. The required area of solar cells is then equal to the power requirements of the spacecraft and battery system, divided by the power per unit area. If this required area is greater than can be allowed in the spacecraft design, increase the estimated coverslide thickness or solar cell thickness, or lower its base resistivity (whichever appears more acceptable), and repeat the calculation above.

If this approach does not result in a sufficiently powerful solar cell array of acceptable area, the power requirements must be reduced, the mission shortened, or alternative power supplies provided. If the required area is less than was permitted, a thinner solar cell may be considered to reduce array weight.

4.1.2 Trajectory or Orbit Characteristics

The trajectory of a deep space probe and the orbit of a satellite provide information on three factors that affect the performance of the solar cell array. These are (1) illumination intensity, (2) exposure to degrading radiation, and (3) temperature (given normal thermal absorptivity α and emissivity ϵ).

Satellites in orbit around the earth receive illumination of about 140 mW/cm^2 of area facing the Sun. This figure varies a few percent during the year, as described in references 41 and 42, and rapidly goes to zero (neglecting the small illumination by stars) when the earth comes between the Sun and the spacecraft. It is important to ascertain, from orbit parameters, the duration and frequency of these eclipses. In low orbits, the spacecraft may be in eclipse up to 40% of the time, and the solar cell array will have to supply over half its output during illumination into batteries for power during the eclipse.

The orbit also determines the exposure of the solar cell array to protons and electrons. From the orbit, a fluence of particles (particles/cm^2) and an energy spectrum for protons and for electrons striking the solar cell array during the mission life can be estimated (ref. 43). These data should be used to determine the electrical efficiency of each candidate design at the end of the mission. Usually, the selection of coverslide and of solar cell composition are based largely on this calculation.

4.1.3 Spacecraft Design Constraints

Spacecraft design constraints should be included from the start in design of solar cell arrays. Frequently, these constraints determine the basic configuration of the array; for example, spin orientation of a spacecraft in a synchronous earth orbit results in a tilt of its axis so that a solar cell parallel to the axis of symmetry will experience a changing angle of incidence of sunlight. Cells mounted on the lateral surface of such a satellite would receive a maximum of 92% of the solar intensity at local noon near the time of solstices of earth's orbit. This is shown in figure 21. If, under these circumstances, the body-mounted array is not sufficient for the power requirement, a paddle array may be necessary. (In a marginal case, as with an Intelsat spacecraft, the lateral surface may be extended as a short "skirt.") The relative geometry of the spacecraft, earth, and sun positions as well as the eclipses of earth satellites and the eclipse seasons are treated in detail in reference 44. This treatment provides a method for calculating the available Sun illumination for a given spacecraft orbit and can also be used as a basis for selecting the orbit that will maximize spacecraft illumination.

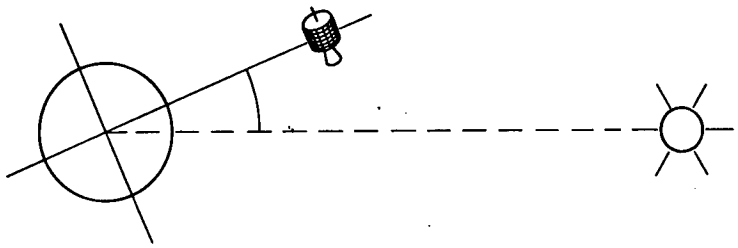

Figure 21.–Spin-stabilized synchronous satellites may have their lateral surfaces at an angle of 23 deg with respect to the earth–Sun line near the times of winter (and summer) solstice. (Cosine of 23 deg is 0.92.)

4.1.4 Launch Vehicle Constraints

The launch vehicle imposes constraints on the dimensions and weight of a solar cell array. If an extendible paddle or oriented array is required, provision must be made for folding or otherwise reducing the array to fit the inside dimensions of a launch vehicle nose cone during launch and for extending the array once it is in space. Depending on the match between the body of the spacecraft and the envelope size, this design requirement may lead to stringent limitations in permissible solar cell array area.

The launch vehicle characteristics also establish a maximum for the weight of the spacecraft for a given trajectory or orbit. Depending on the weight budget for the spacecraft, there may be a stringent constraint on the allowable solar cell array weight. Weight optimization studies dealing with alternatives of materials for all elements of the array, thickness of coverslides, and total number of cells should be performed if such a condition exists (see Section 2.5).

Launch introduces a severe level of vibrational, thermal, acoustical, and gravitational stresses. Each launch vehicle provides its own peculiar combination of stresses. Testing is recommended to ensure that the launch environment does not reduce solar array performance to an unacceptable level. Likewise, testing is recommended to ensure that the maneuvers up to the achievement of a stable orbit (including stage separation and spacecraft antenna and array deployment) do not unduly impair solar array performance (see Section 2.8).

4.2 Components of the Solar Cell Array

If the solar cell array is to function properly within the mission constraints imposed upon it, its individual components must be carefully designed, selected, and assembled.

4.2.1 Coverslides

Coverslides should cover the active area of the front surface completely. Since the cutting and placement of the coverslides involves tolerances, the coverslide should be slightly larger than the

cell to insure complete coverage in all cases. However, too large a coverslide can result in a high degree of breakage, especially for thin (0.152-mm) coverslides.

Coverslides should be resistant to ultraviolet and corpuscular radiation. Corning 7940 and, to a lesser extent, Corning 0211 satisfy this criterion. Corning 0211 with 1 or 2% cerium added has recently been shown to have an improved resistance to radiation (ref. 45). Other glasses should be used only when their radiation darkening is both tested and tolerable.

An antireflective filter such as magnesium fluoride is recommended for the outer surface of the coverslide.

An ultraviolet filter should be placed on the inner surface of coverslides to protect the adhesive against darkening. However, where microsheet is selected, this filter may be omitted since the coverslide has an absorption band that offers reasonable protection against ultraviolet.

When the use of microsheet is desired because cost is of major concern, an analysis should be performed to insure that transmission losses during the spacecraft mission will not excessively degrade the performance of the array. It should be determined whether the installation of additional cells to compensate for the added degradation is necessary, and, if so, whether such a solution is preferable to the use of some material other than microsheet.

4.2.2 Adhesives

The adhesives used for bonding coverslides to solar cells and solar cells to substrate should be selected from among the silicone-type formulations that have been proved effective for solar cell arrays in space flight missions. The required characteristics of adhesives for both of these applications were reviewed in Section 2.6, where examples of commercially available formulations that meet the requirements were cited.

The distinction in requirements between these applications is that (1) transparency of the cured adhesive is of great importance in the coverslide/solar cell bond but is not a requirement for the solar cell/substrate bond and (2) the important properties for the solar cell/substrate bond include thermal conductivity, dielectric strength, and the development of a strong but resilient bond.

Epoxy-type adhesives have been used in solar cell arrays and actually provide for stronger bonding. However, they are more susceptible to darkening than are the silicone-type adhesives and hence are restricted to cell/substrate bonding. The extra strength of the epoxy bond may be a disadvantage since it is often necessary to partially disassemble and repair an array during fabrication and after acceptance testing.

Candidate adhesives should be subjected to qualification tests under environmental conditions (shock, vibration, thermal cycling, corpuscular and ultraviolet radiation, temperature extremes, and flexure) when applied between specimens of the coverslide, solar cell, and substrate representative of the array in which they will be used. This procedure can be used to evaluate the suitability of candidate adhesives to provide the necessary bonding in the actual flight hardware

under the environmental conditions anticipated during manufacture, testing, transportation, launch, and space flight.

Adhesive materials intended for use in the mission should be afforded special handling prior to and during application and cure to insure that uncontrolled environmental conditions or excessive aging do not cause degradation of the materials which would limit their performance in the mission. The shelf-life and pot-life limitations recommended by the manufacturers of adhesives should not be exceeded. Storage conditions for the adhesive before its use, as well as after its application in the solar cell array, should conform to the manufacturers' recommendations, which usually call for a cool, dry, dark environment. To safeguard against contamination or other impairment to the development of desired bonding characteristics, dust and humidity conditions should be controlled in the work area where adhesives are being applied and cured.

4.2.3 Solar Cells

The selection of solar cells for the array depends on the requirements of the mission and the relative merits and costs of what is available. Therefore, the total array should be analyzed using representative solar cells of several types and costs to find the preferable practical alternative.

Selection of a solar cell design also involves specifying the allowable manufacturing tolerances for the cells. At least one designer of solar cell arrays has addressed this question (ref. 46). The cost per solar cell depends on the percentage of the manufactured cells falling within performance tolerances.

Costs and adaptability to the required array shape appear to be the dominant factors in the selection of solar cell size. Larger solar cells appear to offer savings in array assembly costs. To prevent overload of grid lines, a 2- or 3-cm-wide solar cell is recommended. Solar cells up to 10 cm long are available, and a long solar cell, consistent with the dimensions of the panel, is recommended to reduce panel assembly cost, provided that the thermal expansion characteristics are compatible with mission requirements.

A trade-off study is recommended of the relative cost and desired reliability of an array, including the costs of solar cells, interconnections, and assembly, for several alternatives of individual cell area. Such a study should not overlook possible constraints on area selection because of panel assembly jigs and tooling availability.

Solar cell thickness can be from a low of about 0.15-mm (limited by problems in slicing thinner pieces of silicon) to a high of about 0.40 mm (thicker cells do not give appreciable increases in output). The selection of thickness is often based on availability; however, thicker cells yield more power per unit area, while thinner cells provide more power per unit of weight and retain a greater percentage of their output under exposure to corpuscular radiation.

The electrical contacts of a solar cell have been the subject of extensive studies. Presently preferred practice is to use titanium–silver contacts and grid lines with solder over the silver.

Solderless contacts experience corrosion during preflight storage; introduction of a thin solder coating of palladium gives indications of reducing such degradation if properly controlled.

The solar cell most commonly used at present, a p-Si base of 10 ohm-cm resistivity with an n-Si surface layer about 0.5 μm thick, is the result of extensive optimization studies of base resistivity, polarity, and junction depth. However, as shown by figure 5, choice of 2 ohm-cm resistivity results in a slightly more efficient cell when radiation degradation is not severe. The cross-over point occurs between 1 and 3×10^{15} electrons/cm^2 of 1 MeV, or equivalent. In marginal cases, a trade-off study is recommended.

4.2.4 Circuitry

The individual solar cells must be connected into an array so that their outputs couple to provide the required amount of power at the specific potential of each load. The cells are connected in parallel and in series to achieve this power. A group of cells connected as a unit is often called a "module," and modules are interconnected to form the array. All circuitry should be connected as carefully as possible to minimize resistance between cells and between modules. Using presently accepted practices, the power losses due to soldering solar cells into modules and the introduction of resistances between modules is usually limited to about 1%.

The circuitry of the array determines the operating point on the I–V curve for the solar cells. Circuit design, therefore, involves determining how many cells should be in series and how many parallel circuits are required. Frequently, the array is designed for a spacecraft operating supply of 28 V, determined by spacecraft requirements; up to 37 V may be required for battery charging when the batteries are also required to deliver 28 V.

The array should be designed so that each solar cell operates at the voltage of maximum power output at the end of the mission life. Frequently, this voltage differs from the voltage for maximum power on the as-manufactured cells. A calculation of the best voltage (or load point) for the cells is therefore the basis for design of the circuitry of the array.

The series connection of solar cells into strings to build up voltage increases the deleterious effect of an open circuit. It is therefore advantageous to intraconnect between cells of parallel strings as much as possible.

When strings may operate under different conditions, such as strings of solar cells on opposite sides of a cylindrical spacecraft, or where one string may be in shadow while another is not, intraconnections are not advisable. In such cases current from the illuminated string would be lost by shunting through the darkened string. Wherever such shunting is possible, isolation diodes are recommended. Such diodes provide low-resistance paths to current flow from the illuminated strings, but impede leakage currents through dark strings. An example of this is shown in figure 22.

The figure, intended only to illustrate the use of isolation diodes, omits much of the circuitry of power conditioning. This may be considered as the variable "load" R_L, and not part of the solar array.

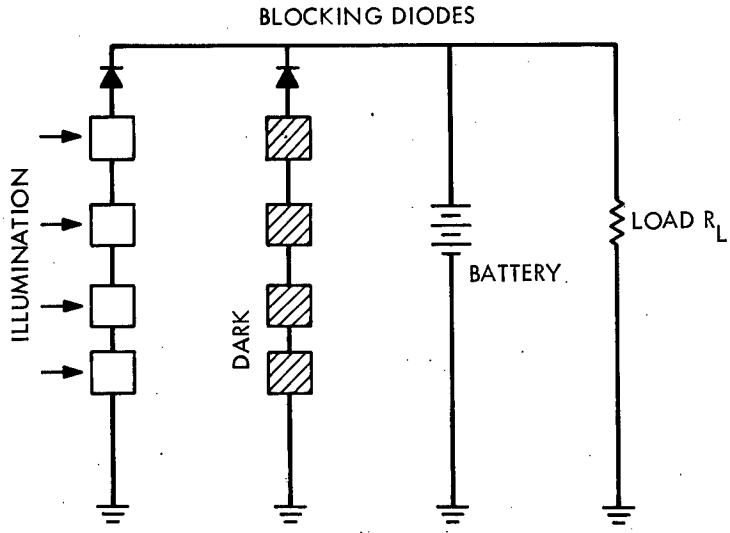

Figure 22.–Two parallel strings of solar cells, with one dark and producing no current. The isolation diodes prevent it from shunting current from the load R_L.

4.2.5 Mounting and Connecting Hardware

Each solar cell/coverslide unit must be supported and electrically interconnected to form the solar cell array. The structure of the solar cell array should be designed for reliability under all environments encountered.

Flat mounting or flexible shingling of solar cells is preferred over rigid shingling; these methods permit expansion and contraction under thermal cycling without excessive danger of breakage. The interconnecting parts should be designed to permit maximum thermal expansion and contraction while maintaining electrical contact with the solar cells. Where the expected change in solar cell length due to thermal cycling can not be accommodated in the interconnector design, selection of a shorter solar cell is indicated. At least two interconnections are recommended for each electrical contact to reduce the probable number of open strings due to micrometeoroid strikes, etc. Soldering is recommended for interconnections to solar cells, as this is current practice, although ultrasonic bonding and thermocompression or welding may, if current studies are successful, show advantages. Soldering should not be used where temperatures over 125°C are to be sustained or where temperatures over 150°C are encountered for short periods (~10 min).

Materials such as aluminum, copper, silver, molybdenum, Kovar, and Invar have been considered as interconnections. The last two mentioned are not magnetically "clean," a disadvantage in some missions.

The general recommendations for design are:

(1) Where practical, use thin interconnection material to minimize thermal and flexural fatigue, but take care to insure low series resistance.

(2) Use liberal stress relief for both mechanical and thermal stresses to avoid undue stresses on all joints.

(3) Determine whether adequate bonding controls and inspection criteria are being used.

(4) Insure that redundancy is incorporated on both the front surface and the rear surface of the cell contacts.

(5) Keep the areas of soldered or welded joints as small as possible to reduce thermal stress.

4.2.6 Substrate

The substrate supporting the interconnected solar cells is frequently of aluminum- or Fiberglas-faced honeycomb. Attention should be given to matching the thermal expansion coefficient of the substrate and the silicon solar cells. Materials such as Fiberglas and some plastics used as facing for a honeycomb substrate have the advantage of thermal expansion properties compatible with those of silicon and thus eliminate the need for additional insulation.

4.3 Testing

To insure that the solar cell array meets the design objectives, a systematic test and evaluation program should be followed through manufacturing, assembly, mating to the spacecraft, and launch preparations. The conditions specified for qualification tests should be commensurate with the objective of demonstrating that the solar cell array is adequately designed to withstand the worst environmental and operational conditions that can reasonably be anticipated during manufacture, assembly, shipment, flight preparations, launch, and the full mission.

Launch and flight conditions that may cause degradation of performance are identifiable from the characteristics of the launch vehicle and the prescribed trajectory and orbit; they include vibration, acceleration, temperature extremes, thermal shock and cycling, proton and electron irradiation, ultraviolet radiation, meteoroid impact, and structural flexure.

The prelaunch environment often represents the limiting case for environmental hazards. For example, the array may be subjected to more severe mechanical shock and vibration during manufacture or transit than during launch. The degrading exposures to contamination and varying levels of humidity occur prior to launch and are similarly important considerations to be accounted for in the qualification of solar cell array design.

Mechanical, electrical, and environmental testing of solar cells should be performed at both qualification and acceptance levels to insure that the array is ready to perform after exposure to the conditions expected during launch and in flight. The testing program should accommodate the

particular requirements of the planned mission. Standardization of tests has not been practical; however, the following general recommendations should be considered:

- After each test in the qualification sequence, inspect the assembly by means of electrical performance checks (particularly to determine changes in series resistance) and visual examination for broken parts, cracked interconnections, and delamination of cells or coverslides (this is commonly done either with the naked eye or magnification up to 10×).

- After environmental testing, determine whether coverslides have separated from the cells, or cells have been loosened from the substrate (visual inspection at 10× and 30× magnification will generally reveal this kind of discrepancy).

- Perform visual examination and electrical performance checks before and during acceptance testing. (In addition to searching for physical failures, the visual examination should include a search for poor alignment and insufficient cell coverage by the coverslide.)

- Include temperature cycling as part of environmental testing. The tests should comprise 10 cycles with temperature limits set at a range that is 20 centigrade degrees more severe than the expected orbital temperature range and with the time rate of change of temperature in cycling set at the maximum dT/dt expected in the mission.

- During electrical testing, use a solar simulator to illuminate the cells. Measurements should be corrected to account for differences between the spectra and intensities of the test illumination source and the Sun. Standard cells, calibrated in high-altitude balloon flight, should be used for monitoring the light source output during testing.

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SP-8003 (Structures)	Flutter, Buzz, and Divergence, July 1964
SP-8004 (Structures)	Panel Flutter, July 1965
SP-8005 (Environment)	Solar Electromagnetic Radiation, revised May 1971
SP-8006 (Structures)	Local Steady Aerodynamic Loads During Launch and Exit, May 1965
SP-8007 (Structures)	Buckling of Thin-Walled Circular Cylinders, revised August 1968
SP-8008 (Structures)	Prelaunch Ground Wind Loads, November 1965
SP-8009 (Structures)	Propellant Slosh Loads, August 1968
SP-8010 (Environment)	Models of Mars Atmosphere (1967), May 1968
SP-8011 (Environment)	Models of Venus Atmosphere (1968), December 1968
SP-8012 (Structures)	Natural Vibration Modal Analysis, September 1968
SP-8013 (Environment)	Meteoroid Environment Model—1969 (Near Earth to Lunar Surface), March 1969
SP-8014 (Structures)	Entry Thermal Protection, August 1968
SP-8015 (Guidance and Control)	Guidance and Navigation for Entry Vehicles, November 1968
SP-8016 (Guidance and Control)	Effects of Structural Flexibility on Spacecraft Control Systems, April 1969
SP-8017 (Environment)	Magnetic Fields—Earth and Extraterrestrial, March 1969
SP-8018 (Guidance and Control)	Spacecraft Magnetic Torques, March 1969
SP-8019 (Structures)	Buckling of Thin-Walled Truncated Cones, September 1968
SP-8020 (Environment)	Mars Surface Models (1968), May 1969
SP-8021 (Environment)	Models of Earth's Atmosphere (120 to 1000 km), May 1969
SP-8022 (Structures)	Staging Loads, February 1969
SP-8023 (Environment)	Lunar Surface Models, May 1969
SP-8024 (Guidance and Control)	Spacecraft Gravitational Torques, May 1969
SP-8025 (Chemical Propulsion)	Solid Rocket Motor Metal Cases, April 1970

SP-8026 (Guidance and Control)	Spacecraft Star Trackers, July 1970
SP-8027 (Guidance and Control)	Spacecraft Radiation Torques, October 1969
SP-8028 (Guidance and Control)	Entry Vehicle Control, November 1969
SP-8029 (Structures)	Aerodynamic and Rocket-Exhaust Heating During Launch and Ascent, May 1969
SP-8030 (Structures)	Transient Loads From Thrust Excitation, February 1969
SP-8031 (Structures)	Slosh Suppression, May 1969
SP-8032 (Structures)	Buckling of Thin-Walled Doubly Curved Shells, August 1969
SP-8033 (Guidance and Control)	Spacecraft Earth Horizon Sensors, December 1969
SP-8034 (Guidance and Control)	Spacecraft Mass Expulsion Torques, December 1969
SP-8035 (Structures)	Wind Loads During Ascent, June 1970
SP-8036 (Guidance and Control)	Effects of Structural Flexibility on Launch Vehicle Control Systems, February 1970
SP-8037 (Environment)	Assessment and Control of Spacecraft Magnetic Fields, September 1970
SP-8038 (Environment)	Meteoroid Environment Model—1970 (Interplanetary and Planetary), October 1970
SP-8040 (Structures)	Fracture Control of Metallic Pressure Vessels, May 1970
SP-8041	Captive-Fired Testing of Solid Rocket Motors, March 1971
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SP-8044 (Structures)	Qualification Testing, May 1970
SP-8045 (Structures)	Acceptance Testing, April 1970
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SP-8071 (Guidance and Control)	Passive Gravity-Gradient Libration Dampers, February 1971

NASA-Langley, 1971 —— 3