SOLAR CELL POWER SYSTEMS FOR SPACE VEHICLES[1]

by
N. W. Snyder[2]
Institute for Defense Analyses
Research & Engineering Support Division
Washington, D. C.
and
R. W. Karcher[3]
General Electric, MSVD
Philadelphia, Pennsylvania

ABSTRACT

On May 3 and 4, 1960 a symposium was held in Washington, D.C. under the sponsorship of the Advanced Research Projects Agency by the Institute for Defense Analyses. Ten programs involving solar cell power systems for current U.S. space vehicles conducted under government sponsorship were discussed by the project engineers who directed these programs. This paper will highlight and summarize the results of that symposium for this session, where possible.

DISCUSSION

It is becoming more apparent that the electrical power supply for space vehicles is an important consideration in the design of that vehicle, particularly where requirements for power are above several hundred watts. Reliability and weight are of prime importance in considering the design of these power supplies. For the short term mission of missiles and rockets, various forms of batteries have served as the

---

[1] Presented at the ARS Space Power Systems Conference, The Miramar Hotel, Santa Monica, California, September 27-30, 1960

[2] Scientific Advisor to the Department of Defense on Advanced Energy Conversion and Space Power

[3] Manager - Electrical Power and Distribution

electrical power supply. In this application, only a minor use of chemical turbo-machinery APUs have been made. Orbiting satellites and long-time space probes require considerably increased levels of energy, and the weight of electro-chemical devices becomes excessive. Solar energy with photovoltaic converters have been used successfully on several satellite designs and are the only power supplies being considered for the immediate future as well.

Silicon (p on n) solar cells have been the energy converter between solar energy and electricity with the storage of energy being accomplished by the nickel-cadmium battery. This paper will be directed toward the general engineering considerations applied to the vehicles discussed in the symposium mentioned above. In some cases, the data is classified and, therefore, has not been included.

At the Santa Monica Space Power Conference, there was a panel discussion of the various problems and to illustrate the scope of vehicles and laboratories involved, a list of these panel members are listed. They also were key participants in the Washington Symposium in May, 1960.

Charles Burrell, Lockheed, MSD, Sunnyvale (Samos, Midas, Discoverer)
Richard Karcher, General Electric, MSVD, Philadelphia (Advent)
Raymond Miller, Space Technology Laboratory, Los Angeles (Pioneer)
Robert Hamilton, Jet Propulsion Laboratory, Pasadena (Ranger)
Walter Scott, Applied Physics Laboratory, Silver Spring, Md. (Transit)
Sherman Winkler, Radio Corporation of America, Princeton (Tiros)
George Hunrath, USASRDL, Fort Monmouth, N.J. (Vanguard, Courier, Pioneer, Tiros, Explorer)

The power level of the solar cell system affects two design factors, the stabilization method and the array design. Vehicles requiring less than one hundred watts have been spin-stabilized. This reduces the complexity of the stabilization sub-systems; however, the increased number of solar cells increases both the weight and cost of the array by a factor of 4, approximately. Vehicles such as Pioneer, Explorer, Tiros, Transit and Courier were designed in this manner. Above 100 watts, the vehicles are designed to be stabilized and employ sun-oriented panels for the mounting of the solar cell. Samos, Midas, Ranger and Advent are examples of space vehicles in this category.

Power at 100 watts is not an exact transition point from non-oriented to oriented solar cell systems. It is the approximate power where the increased weight of the 75% non-working solar cells on a spin-stabilized vehicle becomes greater than the orientation sub-system. Also, the larger area covered by the solar cells on a spin-stabilized vehicle becomes excessive. This increased coverage affects the ability of the vehicle to dissipate heat and provide proper antennae and also restricts location on sensors. Quite obviously the specific mission must be analyzed to determine the most advantageous method of operation if the power level is approximately 100 watts.

Each space vehicle has a prime mission other than that which requires an optimization of the power supply in which case the designer must make a considerable effort to integrate his sub-system into the vehicle in such a manner as not to seriously affect that particular mission. For this reason, it is difficult to compare directly the many designs which are possible. Some general observations may be made which will indicate appropriate design approaches for a wide range of vehicles and missions. An approximate relation between the power level (100 watts) required and the array configuration (non-oriented vs. oriented) has already been indicated.

The mission altitude will influence the system design in several ways. The natural phenomena of charged particle radiation (electrons, protons) varies in intensity as a function of altitude, and affects solar cells unfavorably in proportion to the radiation. The peak intensity of electrons in the 200 to 800 KEV energy range is 10^8 e/cm^{2}sec. For protons in the 40 to 60 MEV, the peak intensities are 2 X 10^4 p/cm^{2}sec and 10^2 p/cm^{2}sec for the inner and outer Van Allen radiation belts respectively. Recent data from the Discoverer flight of November 1960 which was flown during the solar flare of November 12, 1960 roughly estimated that the total dose from the flare over a two day period was approximately 10^{10} protons cm^2 and that the predominant energy was near one Bev. During an active solar flare year, there are usually five such flares (designated three plus intensity) per year. The cyclic nature of such activity indicates that there will be less such flares in the next several years. Glass covers will not protect against the high energy particles from this type of solar flare. Several test cells which have been instrumented will be flown over the next year. In particular, a relatively complete set will be flown on ARENTS (ARPA Environmental Test Satellite) in a 22,000 mile orbit in early 1962.

It has been shown that protection can be provided by glass slides of various thicknesses, depending upon the level of radiation to be expected. A "non-browning" glass is needed in order to transmit solar radiation to the cells as well as provide a surface whose emissivity in the infrared regime is high. At the present time, precise data on space radiation is not available and much investigation by various scientific probes is necessary before the engineer can be sure of his information for purposes of design. The glass slide further protects the cell from micro-meteorite erosion. Little information is available on this parameter. It is presumed that protection against high energy particle radiation is sufficient also for protection against micro-meteorite erosion. For altitudes up to 500 miles, a 6 mil glass cover (Corning microsheet) with an ultra-violet rejection filter with an anti-reflection coating (MgF_2) on the bottom and top of the filter appears standard. At higher altitudes, the glass thickness increases rapidly to as much as 70 mils (see Fig. 1) providing a shield from the more intense Van Allen radiation. In the larger thickness, fused quartz is used and C_eO_2 glass is being considered. The glass thickness is a function of the mass required to absorb and reduce particle energy down to 145 KEV, the value below which silicon cells are not adversely affected, i.e., lattice defects in the p-layer of the cell are minimized. The physical phenomena involving degradation of the solar cell current density is that the minority carrier lifetime in the p-layer is reduced as lattice disturbances are increased. In silicon, 10^{13} to 10^{15} defects will reduce cell efficiencies by approximately 25%. The recent n on p cells (phosphorous diffused surface layer) have shown less degradation to charged particles less than 100 MEV by a factor of 10 to 50 and approximately 2-1/2 for 740 MEV protons. This indicates an important relation of impurity atoms to lattice defects which affects minority carrier lifetimes strongly. The choice of the optical filter is determined by the solar cell optical and thermal response characteristics.

For a single surface array which is sun-oriented, the power density produced is between 6 and 9 watts per square foot, based upon a cell temperature of approximately 100°F under normal solar irradiation, with a packing factor or percent area utilization of 85-90%. The utilized area for spin-stabilized systems is of the order of 25%. Consequently, a high penalty is paid for such a system where power requirements are high. (Greater than 100 watts). A single surface array will weigh in the range of 1 to 3 pounds per square foot. The large variation is caused by the need for protection against charged particle radiation in the Van Allen belts as well as variations in structural support of solar cell arrays. As has been mentioned before, each case is examined on its own merit. Component integration varies greatly with each space vehicle.

It is now important to mention storage of electrical energy for the conditions where continuous power is required even though the satellite may spend a portion of its time in the shadow of the earth or where peak powers, far greater than that which can be provided by the solar cell system at a particular instant, are required for such tasks as telemetry from deep space probes. Although the use of batteries has been widespread over the past one hundred years, maintenance of such batteries has been provided by men. Furthermore, the electrodes have been heavily constructed for long cycle life and hermetic sealing has generally not been required. One of the serious limitations of the present solar cell power systems is the short cycle life of the presently constructed nickel-cadmium battery. Because such batteries with cycle lives from 1 to 5 years have been only recently required, little research has been performed, unfortunately. Failures occur because of overheating, separator and electrode degradation, electrode corrosion, contamination, structural defects and leaks. In the range of several watts up to several hundred watts, other power sources will be seriously considered if the storage battery reliability is not properly solved. Important candidates are the long life radioisotope thermoelectric system (SNAP III type) and the nuclear fission thermoelectric power source (SNAP X type). Every effort will be made to take advantage of the simple solar cell array which utilizes the ever present solar energy available in space by providing a reliable means of storing electrical energy. More detailed discussions are provided in two papers given in the chapter on Electrochemical Cells in Volume III of the American Rocket Society Progress Series by Schulman and Thomas. Furthermore, two other papers in that chapter show interesting possibilities of a gaseous electrode storage battery. In a solar cell power system, the battery is usually 50% or more of the total weight and all of the performance data to date indicates that the storage battery has less than satisfactory reliability. Research already performed on organic polymer separators, hermetic sealing, electrode impregnation, chemical reaction control by additives and precision production control is showing results which would lead the authors to be optimistic about the use of nickel-cadmium batteries for long-life solar cell powered space vehicles. One or two years of research and development should change the present situation markedly. In fact, highly reliable batteries should be available by the end of 1961. The Air Force has made some definite steps in testing procedures, research and production, the Army in research, NASA in research and a special high quality production pilot plant and the Navy in quality control and research. It would behoove the National Science Foundation to support more basic research in electrochemistry in Universities.

Batteries used in low altitude orbits are discharged approximately 10% only for reliability purposes thus imposing a great weight penalty. In a low orbit (300 miles to 500 miles) a battery must be charged and discharged every 90 minutes. Thus, for a long-life power system, these batteries must cycle say between 5000 and 20,000 times depending on the lifetime required for the vehicle. Although reliability of other components in a space vehicle may not reach these standards of lifetime at the present time, one must not relax on the development of the power supply support system, because improvements will occur in the wide variety of technologies involved in the contemplated space vehicles.

For the case of a 24 hour orbit (22,000 miles altitude), the shadow period is only 75 minutes out of 24 hours and then only in a fraction of the year period. Therefore, a deep discharge of 60% to 70% is because of the slower charge rates possible (causing much less damage to the cells) of approximately 8 to 16 hours, rather than the 30 or 40 minutes, as found with the low orbit case. The weight of cells required is also considerably less for the same energy storage with 1 to 2 watt-hours per pound possible for a 10% discharge and 6 to 10 watt-hours per pound possible for a 60 to 70% discharge.

CORRELATIONS

As in all scientific and engineering technology, one attempts to correlate information judiciously even though it is scarce. Use of a graph might indicate a trend which is useful for predicting a physical phenomena or in component design. An attempt was made to correlate solar cell system parameters with the data available and previously appraised over the past year or two by space vehicle power plant designers. Figure 1 represents the experience involving design of the vehicles Vanguard, Courier, Pioneer, Tiros, Explorer, Transit, Ranger, Advent, Samos, Midas and Discoverer, as well as the experience in certain cases of speculative designs of vehicles not now under government funding. Figure 1 specifically shows the effect of altitude on several important parameters involved in the design of solar cell power systems. These parameters are watts per square foot of the array (only includes the solar cells and its structural support), percent depth of discharge of the batteries (nickel-cadmium), glass cover thickness (6 mil is Corning microsheet and larger thickness are fused quartz), pounds per kilowatt output of the whole system, pounds per kilowatt for the batteries, and pounds per square foot of the solar cell array. The curves were based on several spin-stabilized vehicles at the 400 mile altitude and in oriented array designs for the 22,000 mile altitude. The specific weight curves show a trend of decreasing values as the altitude increases. This, for the most part, can be attributed to the reduced battery weight. Note the relative position of curve of pounds per kilowatt for the batteries as compared to the system pounds per kilowatt curve. The relative percentage of the battery weight to the total system weight is approximately 50% at the 22,000 mile altitude and approximately two-thirds of the system weight at the 300 mile altitude. Flight at higher altitudes permits a greater percent depth of discharge with a consequent increased utilization of the batteries for storing energy. As one would expect, the specific weight for the batteries in pounds per kilowatt when plotted versus altitude shows a marked decrease with altitude but eventually flattens somewhat because of the limitations on depth of discharge. Eventually this curve must drop to zero since no electrical storage is needed above a certain altitude because shadows are not encountered. This case is invalid if power peaks greater than the solar cell output are required, for instance, for telemetry.

There are two data points not in general agreement with the curves drawn. The first point, dealing with the pounds per kilowatt for the total system and shown considerably above the curve at 300 miles altitude, is a plot based on information concerning the Transit vehicle. It appears as though the vehicle requirements forces a weight penalty on the design of the solar array. (Note the structural members supporting the solar cells and circumscribing the equator of the vehicle as described in the paper on the development of the power system for the Transit satellite by Scott.) The second data point contradicting the plotted curves is that of percent depth of discharge for the batteries on the Courier vehicle. While the standard practice for vehicles at the 400 mile altitude has been to use 10% depth of discharge, the designers of Courier have used 25%.

Meeting Notes of the May 3-4 Solar Cell Power System Symposium can be obtained by those with proper classified status (SECRET) from the Technical Information Officer, Fred A. Koether, of the Institute for Defense Analyses, Advanced Research Projects Agency, The Pentagon, Washington, D.C. Ask for the Solar Cell Power Systems for Space Vehicles - Meeting Notes prepared under the direction of Nathan W. Snyder (173 pages). Also available is a chart of the parameters involved in the design of several of the U.S. space vehicles which is also classified SECRET.

FIGURE 1.

SOLAR CELL POWER SYSTEM PARAMETERS
BASED ON DESIGN POINTS FOR SEVERAL SPACE VEHICLES