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Advanced Power Systems for EOS

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Sheila G. Bailey, Irving Weinberg,
and Dennis J. Flood
Lewis Research Center
Cleveland, Ohio

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ADVANCED POWER SYSTEMS FOR EOS

Sheila G. Bailey, Irving Weinberg, and Dennis J. Flood
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

ABSTRACT

The Earth Observing System, which is part of the International Mission to Planet Earth, is NASA's main contribution to the Global Change Research Program. The program opens a new era in international cooperation to study the Earth's environment. Five large platforms are to be launched into polar orbit, two by NASA, two by the European Space Agency, and one by the Japanese. In such an orbit the radiation resistance of indium phosphide solar cells combined with the potential of utilizing 5 micron cell structures yields an increase of 10% in the payload capability. If further combined with the Advanced Photovoltaic Solar Array, the total additional payload capability approaches 12%.

to be serviceable if an effective means of on-orbit servicing becomes available during the life of the mission. The satellite will be launched into polar orbit from Vandenberg Air Force Base on a Titan-IV rocket. The need for global coverage every one to three days dictates a sun-synchronous orbit with a quasi-two-day repeat; a 705 km altitude, 98.2-degree inclination orbit meets this need. It will have a 13:30 local equatorial-crossing time.

The platform science objectives include global quantification of the hydrologic cycle, monitoring of the upper atmosphere and surface radiation flux, a comprehensive study of clouds, the characterization of biological activity and ecosystems through optical measurements, an improved study of atmospheric circulation, and the global determination of surface mineralogy.

INTRODUCTION

The international scientific community is organizing research efforts to advance our knowledge of both natural and human-induced global change. The U. S. Global Change Research Program, a consensus interagency plan, defines the U. S. element of those efforts. Mission to Planet Earth, the central NASA contribution to the this program includes two proposed initiatives: the Earth Observing System (EOS) and Earth Probes. EOS consists of a space-based observing system, a data and information system, and a scientific research program. This represents the initiation of a comprehensive, global observing system with broad and high-resolution spectral and spatial, as well as long-term temporal, coverage of the Earth. The space component will consist of two series of polar-orbiting platforms: EOS-A and EOS-B, with the earliest possible launch of EOS-A in late 1997. EOS will be supplemented by companion European and Japanese platforms, as well as the continuing operational environmental satellites. The 15-year observational period will be achieved using three identical satellites per series, each with a five-year design lifetime.

EOS-A DESCRIPTION

Figure 1 illustrates the EOS-A platform [Ref. 1]. EOS-A has a 5-year design life, a payload mass of 3,000 to 3,500 kg, 6 kW average power for the platform itself and 3.2 kW for the payload, 300 Mbps peak data rate, 30 to 50 Mbps average data rate, and 0.5 terabyte of on-board data storage. The present baseline calls for replacement of the platform after 5 years; however, the platform is designed

POWER SYSTEM WITH SILICON SOLAR CELLS

A currently considered platform design allocates 1407 kg to providing electrical power using large area (8 x 8) cm silicon (Si) cells, 0.2 mm thick, covered with a 0.125 mm thick ceria stabilized glass microsheet. Improvements in the photovoltaic performance of the array would reduce the mass and permit more of the spacecraft mass fraction to be allocated to orbital communications, observations, and other equipment packages.

The large area silicon cell, 10 ohm-cm base resistivity, with dual anti-reflective coating and a back surface field has an average efficiency of 14.2% at 28°C, beginning of life (BOL). These cells are currently under production for Space Station Freedom [Ref. 2].

The array would consist of approximately 58 pairs of panels with each panel containing 200 series wired cells. The panel pair is also wired in series to produce a nominal 160 Vdc output. The array would be designed to produce 12.2 kW at 5 years and 9.76 kW at 7.5 years. The payload requires 3.2 kW average and 4.2 kW peak.

The silicon cells are mounted on a light flexible kapton substrate to decrease the array mass, however, it also leaves the rear surface of the cell much more vulnerable to radiation. The equivalent shielding thickness for the rear cell surface of a back surface field (BSF) cell .2 mm thick is .05 mm [Ref. 3]. In an orbit of 98.2° at 705 km this results in a 5-year total equivalent

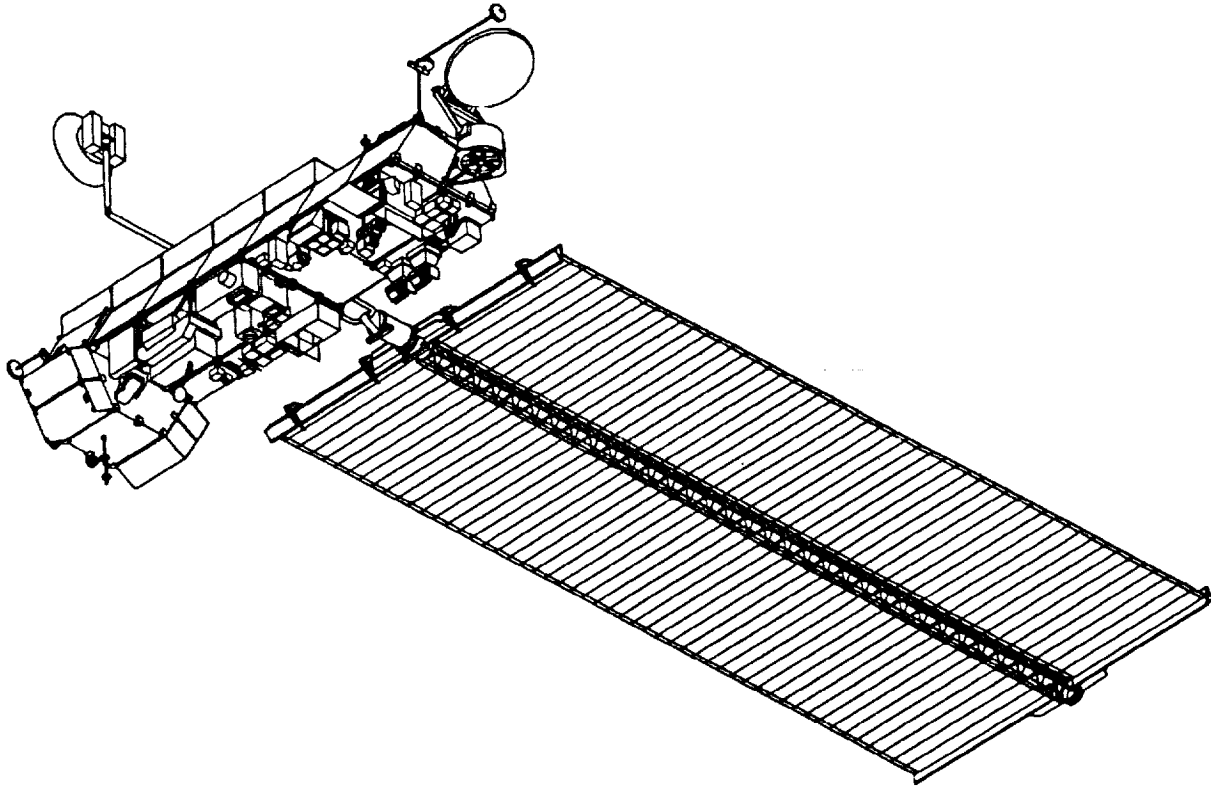


Figure 1. EOS-A satellite

fluence of $1.39 \text{ E}14 \text{ e}^-/\text{cm}^2$ for protons and $4.7 \text{ E}12 \text{ e}^-/\text{cm}^2$ for electrons [Ref. 4] (see Figures 2 and 3). The resulting degradation factor (efficiency EOL/ efficiency BOL) is .82 after five years.

POWER SYSTEM WITH INDIUM PHOSPHIDE SOLAR CELLS

Indium Phosphide (InP) solar cells are excellent candidates for space power generation. Total area efficiencies over 19%, air mass 0 (AM0), are now routinely achieved for cells 4 cm^2 in area [Ref.5]. Figures 4 and 5 illustrate the superior radiation resistant properties of InP [Refs. 6,7, and 8]. In addition a small module containing InP solar cells has shown no degradation after four years in space [Ref. 9]. A large-scale commercial process has been developed to produce high efficiency diffused junction InP solar cells and one thousand of these cells are the power source for a lunar orbiter on board the Japanese ISAS scientific satellite "MUSES-A", which was

launched January, 1990. However there was no telemetry data from this satellite after one month.

Recent data have shown that a fluence of 10 MeV protons produces damage in InP that can be approximated by a fluence of 1 MeV electrons, about 650 times that of the 10 MeV proton fluence [Ref. 8]. This compares to 3500 for silicon and 1000 for gallium arsenide [Ref. 10]. Utilizing this relative damage coefficient and the silicon equivalent fluences yields a degradation factor of .997 for InP cells with .05 mm coverglass after 5 years in orbit. The total annual equivalent 1 Mev electron fluence from trapped protons and electrons is $2.3 \text{ E}13 \text{ e}^-/\text{cm}^2$. However, InP cells, on board the LIPS III satellite with 12 mil coverglass, in a nearly circular orbit of 1100 kilometers altitude with an inclination of 60° for four years have a radiation environment of approximately 1.8 times the total fluence of five years in the EOS orbit (after adjusting for the differences in coverglasses) and have exhibited no measurable degradation [Ref. 9]. Annealing under continuous minority carrier injection by sunlight may explain the apparent lack of degradation for these cells [Ref.7].

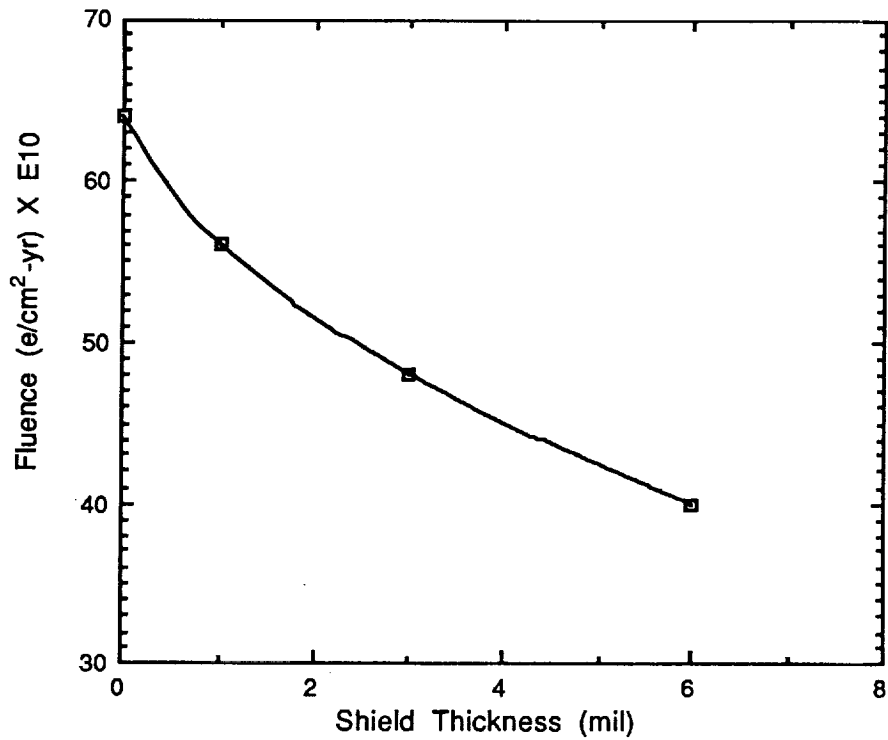


Figure 2. Annual equivalent 1 MeV electron fluence for silicon from trapped electrons, 705 km altitude, 100° inclination, as a function of shield thickness

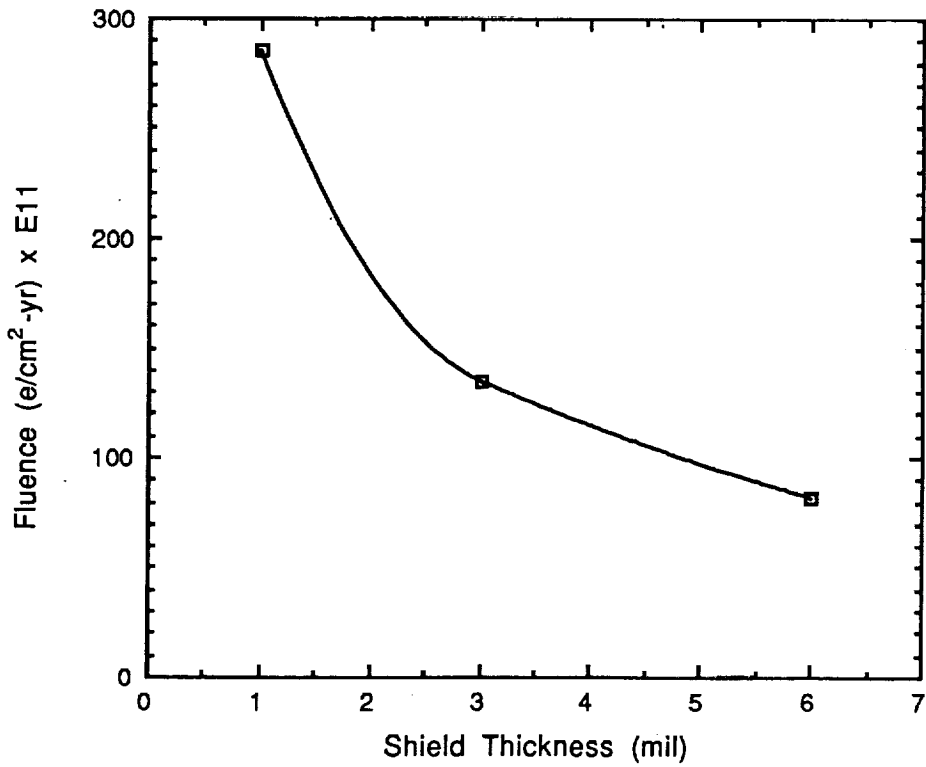


Figure 3. Annual equivalent 1 MeV electron fluence for silicon from trapped protons (P_{max} , V_{oc}), 705 km altitude, 100° inclination, as a function of shield thickness

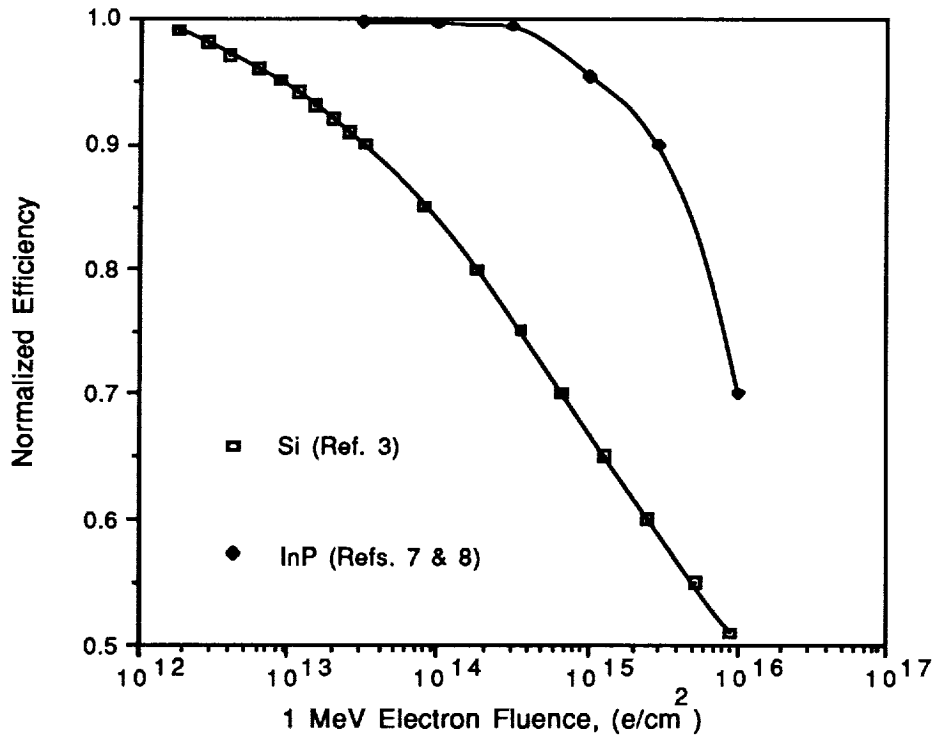


Figure 4. Normalized efficiency after 1 MeV electron irradiation

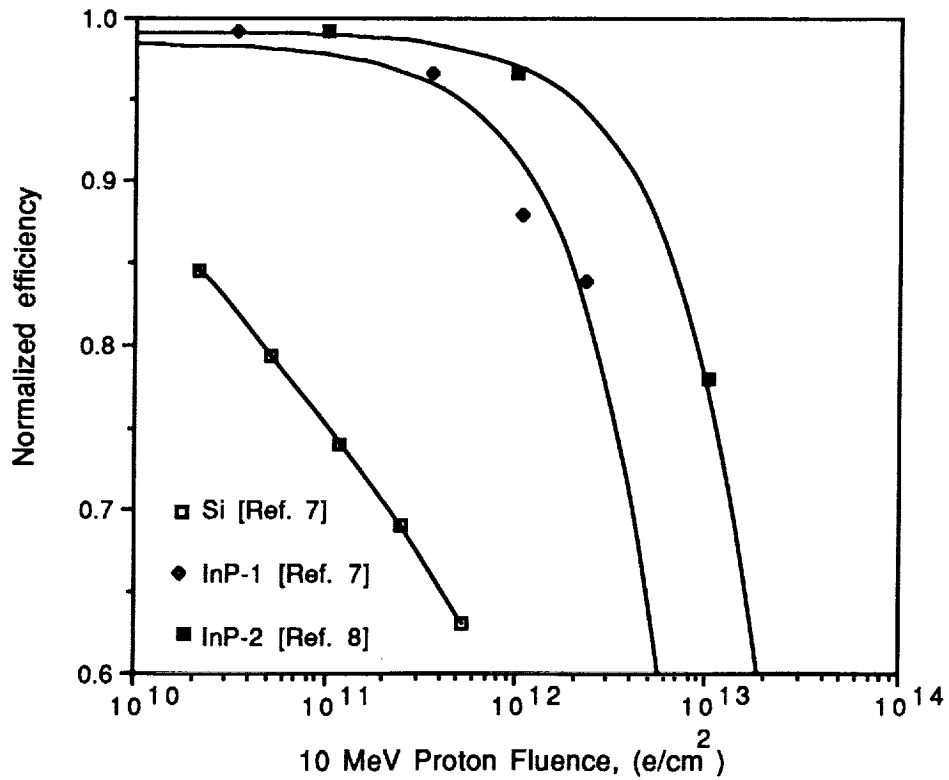


Figure 5. Normalized efficiency after 10 MeV proton irradiation

The recent achievement of 19.1% AMO efficiencies with 4 cm² InP cells [Ref. 5] indicates that increased efficiencies and larger areas are indeed feasible. However, one impediment to the development of InP space cells has been the relatively high cost of InP wafers. Two potential solutions to this problem are either heteroepitaxial InP solar cell structures on silicon substrates [Ref. 11] or ultra thin InP solar cells produced by the CLEFT [Ref. 12] or peeled film techniques [Ref. 13] and bonded to a silicon substrate. Modeling calculations indicate that efficiencies of 22.5% are feasible in the laboratory [14]. Production capability is therefore assumed to be able to achieve a 20% efficient, 5 micron InP solar cell structure on a 55 micron silicon substrate. This cell will be used to demonstrate the mass saving on both a Space Station Freedom array described earlier and a retractable version of the Advanced Photovoltaic Solar Array (APSA) [Ref. 13] under development at NASA's Jet Propulsion Laboratory.

COMPARISON AND CONCLUSION

In a relatively high radiation environment InP has a clear advantage in radiation tolerance. Figure 6 compares the degradation in normalized efficiency as a function of the number of years in an EOS orbit for a silicon solar cell with a 5 mil cover glass and an InP solar cell with a 2 mil cover glass. The 5-year degradation factor of silicon is .82 compared to .997 in indium phosphide for the EOS orbit under consideration. A comparison of temperature coefficients, -0.042%/°C for InP and -0.066%/°C for Si,

and assuming a 60°C operating temperature, a 1.47% decrease is expected in InP and a 2.1% decrease in Si. In addition the expected increase in temperature due to radiation degradation gives approximately a .07% advantage to InP at the 5-year design point.

A major advantage in the utilization of InP occurs because the increased optical absorption of InP permits a cell design requiring only 5 microns of solar cell material without a loss of efficiency. A 55 micron silicon substrate and 50 micron coverglass were chosen to be compatible with the APSA program design. Without considering the impact of a reduced array size on other components of the electrical power system, simply changing the cells of the proposed silicon array to indium phosphide without altering any other structural components saves 298 kg. This would represent a 10% increase in payload capability.

The APSA wing is an ultra-lightweight flexible foldout blanket with a weight optimized deployment mechanism consisting of a fiberglass trilongeron lattice mast deployed from a cylindrical aluminum cannister and deployment actuator [Ref. 15]. Using both the APSA array and InP cells secures a gain of 352 kg, which is approximately an 11.7% increase in payload capability.

The preceding results provide a compelling reason to pursue the use of InP solar cells for space applications. To date the space use of these cells has been limited but the data obtained does reflect the excellent radiation resistance exhibited in the laboratory.

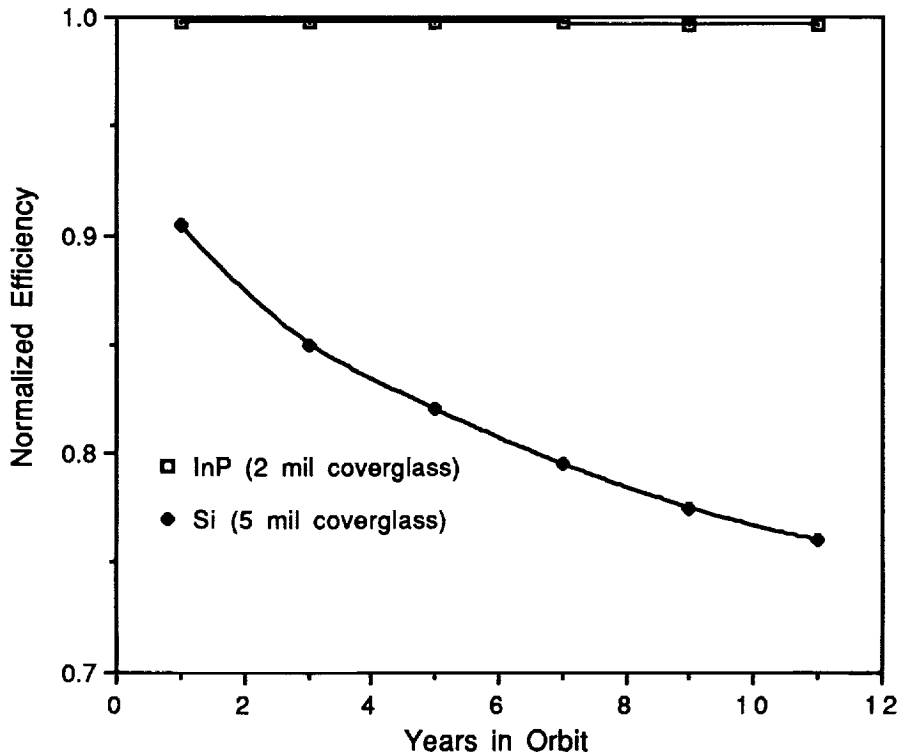


Figure 6. Normalized efficiency as a function of the number of years in a 705 km polar orbit

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