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INTERPLANETARY EXPLORATION-A CHALLENGE FOR PHOTOVOLTAICS*

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Future U.S. interplanetary missions will be less complex and costly than past missions such as Voyager and the soon to be launched, Galileo. This will be required in order to achieve a balanced exploration program that can be sustained within the context of a limited budget.

Radioisotope Thermoelectric Generators (RTGs) have served as the power source for missions beyond the orbit of Mars. Recent government costing practices have indicated that the cost to the user of these power sources will significantly increase. Solar arrays can provide a low cost alternative for a number of missions. Potential missions are identified along with concerns for implementation, and some array configurations under present investigation are reviewed.

INTRODUCTION

During the first decade of planetary exploration, spacecraft were launched as often as every few months, first to the moon, then the nearby planets, and finally to the outer planets. By the end of the second decade the intervals between launches lengthened to years as a result of budgetary constraints and increased mission complexity. This change has strained the nation's ability to conduct an effective interplanetary exploration effort. Technologies and talents required for various aspects of the spacecraft were required only at sporadic intervals and as a result maintenance of interplanetary spacecraft and scientific capabilities has become difficult.

In 1980 the Solar System Exploration Committee (SSEC), an ad hoc committee of the NASA Advisary Council, was established to examine and review the planetary exploration program. From this review, the SSEC defined an overall program that presented a number of features including a balance of missions between near earth planets, small bodies (asteroids and comets) and the outer planets (ref. 1). Of major importance, the program established a critical level of activity consistent with a realistic sustainable budget, in order to provide for stability. In implementing this, the approach specified highly focussed, less complex mission that could rely heavily on existing technology and hardware inheritance to reduce costs. Whereas the cost of some early missions such as Viking 1 and 2 had exceeded two billion dollars total, the new plan would be based on a total annual funding level of ~$300 M (FY '84). Of this total approximately $60 M/year would be available for the planetary observer program (near earth missions), ~$100 M/year for Mariner Mark II

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program (missions beyond the inner solar system), and the remainder available for mission operations and scientific analysis. (These numbers can be compared with previous years funding levels for interplanetary exploration. For the years 1964-66 and 1972-74, annual funding exceeded $800 M in FY '84$. For 1978, and 1981-84, funding fell below $250 M in FY '84$.)

A critical item in any of these missions is the power source. For missions such as the planetary observers (Venus Radar Mapper, Mars Observer, Mars Aeronomy Orbiter, Venus Atmosphere Probe, Mars Surface Probe, Lunar Geoscience Orbiter) previous planetary experience has demonstrated the suitability of photovoltaics as the primary power source. For missions within the Mariner Mark II program (Comet Rendezvous/-Asteroid Flyby, Comet Sample Return, Multiple Mainbelt Asteroid Orbiter/Flyby, Earth Approaching Asteroid Rendezvous, Saturn Orbiter, etc.) past experience would point to the use of RTGs. However, within the past few years the cost of RTGs has come under examination. Historically, the cost of the fuel for an RTG power source has been "subsidized" by DOE, resulting in a relatively low RTG cost to NASA. This policy is presently under review and not yet resolved. Existing estimates of the RTG fuel costs range up to ~$3500 per thermal watt. If NASA is required to assume these costs or a significant portion of them, the RTG cost per mission could be prohibitive within the context of a constrained budget. This is especially so when qualification and spare unit article costs are included. For this reason a number of missions which normally might be RTG powered are potentially open to photovoltaic power. As shown in figure 1, the solar array cost is a function of the solar distance at which the power is required. Allowing for uncertainties in RTG costs, and array performance, array applications out to 6 AU can be considered as cost effective. With this in mind Mariner Mark II missions can be examined for solar suitability.

MARINER MARK II

A number of the Mariner Mark II missions present challenging opportunities for photovoltaics. As shown in figure 2 the solar range of these missions extends well beyond the range of present solar array experience. In addition, array operation is required over a wide variation of solar intensity resulting in a correspondingly large variation in array output. A major concern for photovoltaics under these circumstances is that for conditions of low intensity low temperature (LILT), various losses in cell output can occur. These losses are very irregular and can lead to an unacceptable degree of nonpredictability in array design and operation. Consequently, when LILT losses become appreciable the use of solar arrays may be impractical. As shown in figure 2, the onset of such conditions can be moved to increasing distances by means of solar concentration, although at an increase in structural complexity and array pointing requirements.

Even without the LILT degradation solar arrays present a number of difficulties for interplanetary acceptance. Since solar intensity drops off with the square of distance, a rapidly increasing array area is required to meet power needs at increasing solar distance. With typical interplanetary spacecraft power requirements of a few hundred watts, many tens of square meters of array can be required. This large area impacts launch packaging and deployment, and ultimately leads to a requirement for low array mass density. Additionally, large arrays will compete for limited available spacecraft area with scientific experiments and with required fields of view. Consideration must also be given to spacecraft maneuvering during encounters, to ensure simultaneous array sun pointing, antenna earth pointing, and experiment target viewing. Unlike typical earth orbiting missions, these maneuvers may involve very rapid movements due to high approach velocities.

Since array area must be sized for the worst case, the variation in solar distance means that available power not only will vary widely but for much of the mission will greatly exceed the requirements. Handling such a power variation, and yet maintaining the highest efficiency at critical conditions will not be a trivial matter. Removal of excess power, due to widely varying circuit currents and voltages, needs to be effectively handled in the design of the solar array power system. In some cases combinations of separate and discrete circuits might be utilized at various solar distances and in other cases it may be more effective to allow for circuit reconfiguration during flight or to consider the use of maximum power tracking.

The first planned use of solar power for a Mariner Mark II mission will be unique in that an array will be combined with an RTG. As mentioned earlier the cost of an RTG is quite high. At the same time the present RTG power supply provides for a fixed unit of power, ~250 W. Scaleable RTGs have been proposed but are not presently available. For the Comet Rendezvous/Asteroid Flyby (CRAF) mission present planning indicates that slightly more than 250 W will be required. For this mission, a spare RTG from the Galileo program will be used. Acquiring a second RTG to meet an additional power need cannot be justified on a cost basis. Hence, the idea was proposed of using an add on solar array to make up the difference.

Such an array was initially envisioned to be on the order of 6 m^2, located at a fixed angle on the spacecraft side (fig. 3). After analysis of the array performance, including LILT effects, possible off sun pointing (up to 45^o) and potential shadowing, it was apparent that greater area was required to meet the mission power requirements. Packaging a larger array on the configuration was difficult but resolved by changing the array shape to that of a washer and locating it colinearly with the earth pointing high gain antenna. The maximum size of the array was then established by the shuttle bay. As shown in figure 4, a considerable increase in array area was achieved, although the washer shape will reduce the cell packing density. As an additional benefit the array will maintain close to normal incidence sun-pointing, particularly as the solar distance is increased. This is due to the near coincidence of sun and earth locations for outbound viewing. Thus as array output decreases with increasing solar distance, any off angle pointing loses will also be reduced, maximizing output for the critical power situations. The array/RTG combination provides an interesting fusion of techologies that can meet technical and cost requirements. The tie in of antenna and array functions also provides a way of avoiding the complexity of a separate fully articulated array orientation system. Although the CRAF array is of good size by conventional array standards (~10 m^2), it is useful to consider what a fully photovoltaic powered spacecraft might require. Figure 5 is a schematic for such an array. The spacecraft is dwarfed in comparison. Yet with the trend to larger area arrays for planetary spacecraft and the NASA-OAST funded development of large area deployable arrays such a configuration may not seem unusual within a decade.

As mentioned earlier, one problem for the CRAF mission (and other MMII missions) that needed to be addressed was the LILT degradation. The array performance analysis for CRAF included an amount of LILT loss. However, as many have observed, the magnitude of LILT degradation for any single cell is quite unspecified and can vary considerably from cell to cell or for different LILT conditions (ref. 2). Possible mechanisms for LILT losses have been discussed. Some causes, such as low shunt resistance, or non-ohmic contact behavior have been convincingly identified and their incidence can be avoided. Others, such as the "broken knee" (softening of knee of cell I-V curve) phenomenon have not been demonstrably corrected. Since a LILT degradation-free cell does not exist, the initial MMII approach to this problem will be to select cells based on both air mass zero and LILT behavior. Although requirements have not been defined, it is assumed that a certain amount of LILT degradation will be tolerated and included in the array performance design analysis. Although excluding cells with high LILT degradation will reduce the yield of acceptable cells it is presently assumed that a reasonable yield can be achieved. At present, insufficient data is available to determine the accuracy of this assumption. Clearly failure to achieve this will significantly jeopardize use of solar arrays for MMII type missions. A far better solution would be to correct the LILT degradation phenomenon at the cell level. Until that can be demonstrated, a cell selection process is felt to offer the best alternative.

SOLAR ELECTRIC PROPULSION

The use of photovoltaics to power electric/magnetic engines for spacecraft has been evaluated for many years. Recent advances in thruster technology have lead to performance improvements renewing interest in outer planetary mission applications. Advantages of solar electric propulsion (SEP) systems include those of reduced flight time and enhanced spacecraft mass allowance. In order to use solar arrays for thruster power a number of concerns must be addressed, some of which are common to any deep space mission. First is the need for large area arrays. Typical SEP applications are based on the availability of multikilowatt power sources, between 25 and 30 kW at beginning of life. RTGs are not competitive at these levels. Obviously, high specific power (W/kg) becomes important for these power levels. In addition because of the very large size of these arrays methods for achieving lower specific cost ($/watt) will be very important. Although primary thrust performance is achieved near earth it is advantageous to maintain a high power output as long as possible. Consequently maintaining maximum solar array output requires accommodations of cell voltage variation with solar distance and avoidance of severe LILT degradation.

Although arrays of this size are not state-of-the-art, experience gained with the shuttle flown solar array flight experiment (SAFE), and present NASA-OAST programs for large area high performance arrays development are all applicable to SEP.

CONCLUSION

The need to provide for stability in the U.S. planetary exploration program has been addressed by NASA. The means for achieving this relies on the use of less complex, yet scientifically high priority, low cost missions. The potentially high cost of RTG power sources may jeopardize the viability of this approach. Photovoltaic solar arrays offer a low cost solution for powering a number of far earth missions. In order to achieve this it will be necessary to overcome a number of obstacles. Due to reduced solar intensity large area arrays will be required, even for modest power outputs. These large sizes will in turn impact available mass allowances and spacecraft fields of view. The variation in array performance with solar distance must be accommodated in a manner that effectively meets the spacecraft power needs under all circumstances. The degradation in power output due to LILT conditions must be handled in a practical manner, if an outright solution is not feasible. Predictable power output is as important for a mission as is the quantity of that output. Although none of the above obstacles is considered insurmountable it will be necessary to address them during the next few years to establish technology readiness.

As an initial step in using solar arrays at far AU conditions, the Mariner Mark II Comet Rendezvous mission is examining the combined use of an RTG and a solar array. In this case the array is used to provide a modest augmentation in power, avoiding the costly addition of another complete RTG.

As a method of eliminating orientation mechanisms and controls, the array will be a washer shaped structure located colinear with the earth pointing high gain antenna. Use of this concept is dependent on a predictable and minimal LILT array power loss. Its success will provide a demonstration of the suitability of photovoltaic power systems for other interplanetary missions beyond the orbit of Mars.

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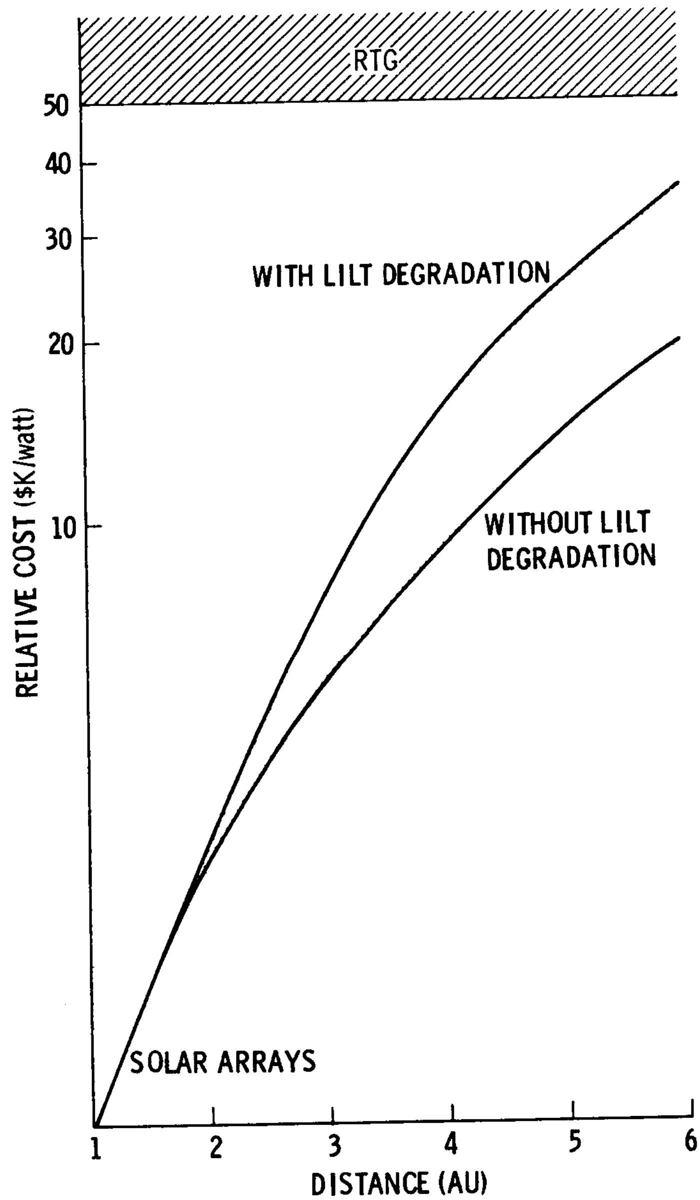

FIGURE 1. SOLAR ARRAY COST vs SOLAR DISTANCE

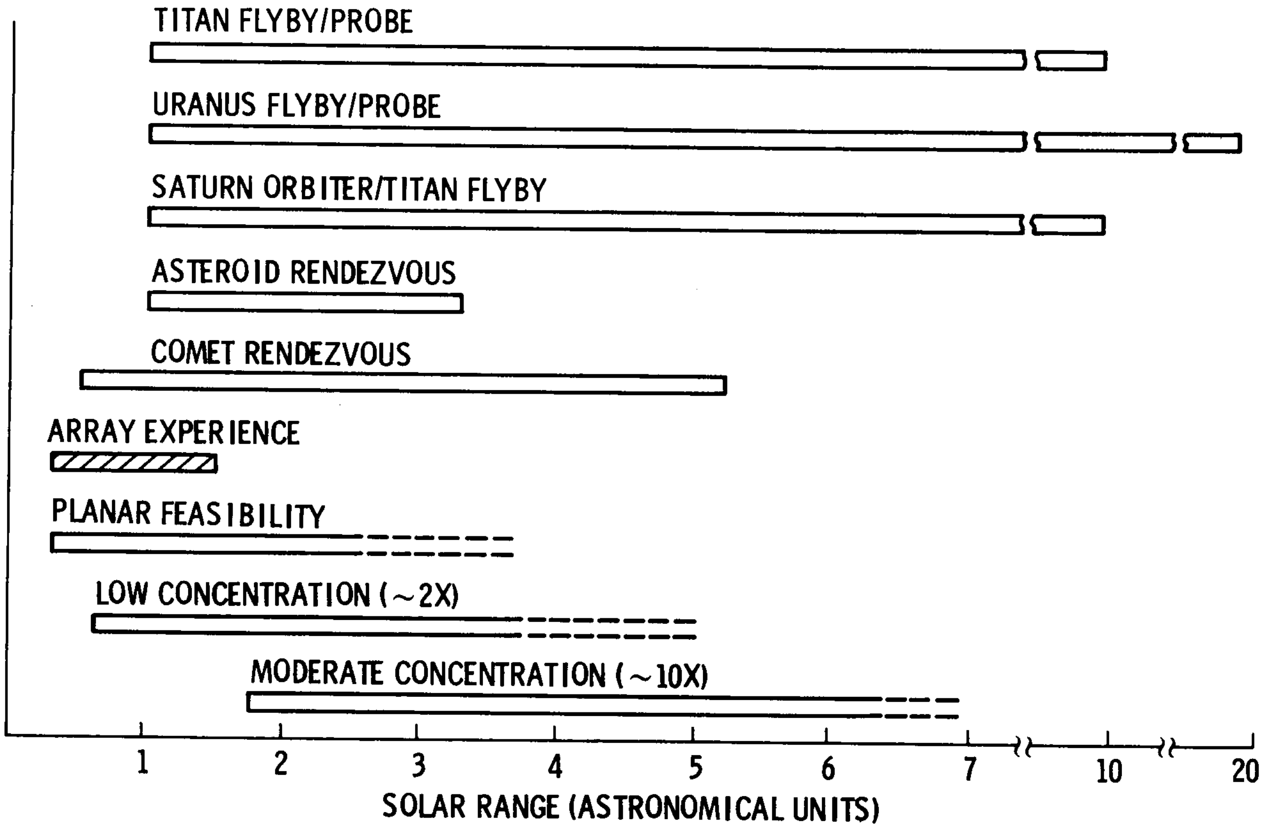

FIGURE 2. MARINER MARK II MISSIONS vs SOLAR DISTANCE

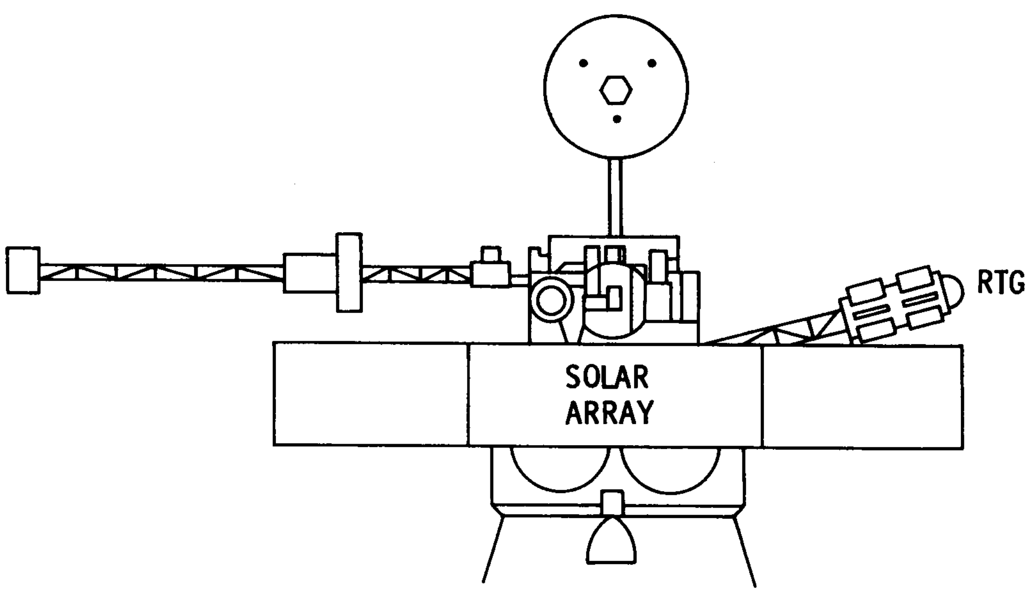

FIGURE 3. INITIAL MMII CRAF SPACECRAFT SOLAR ARRAY CONFIGURATION

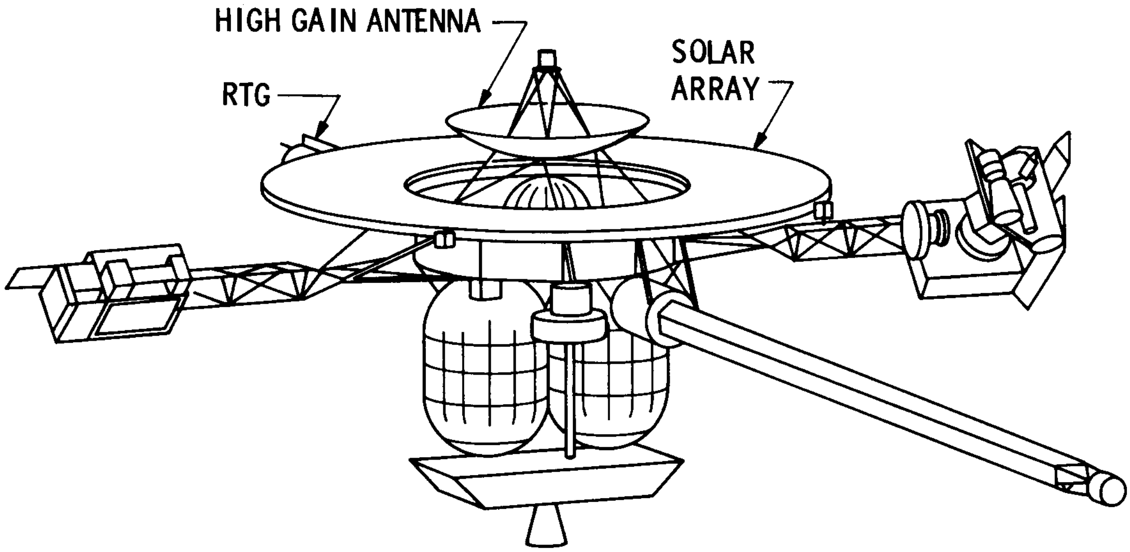

FIGURE 4. ADVANCED CONCEPT SPACECRAFT SOLAR ARRAY CONFIGURATON

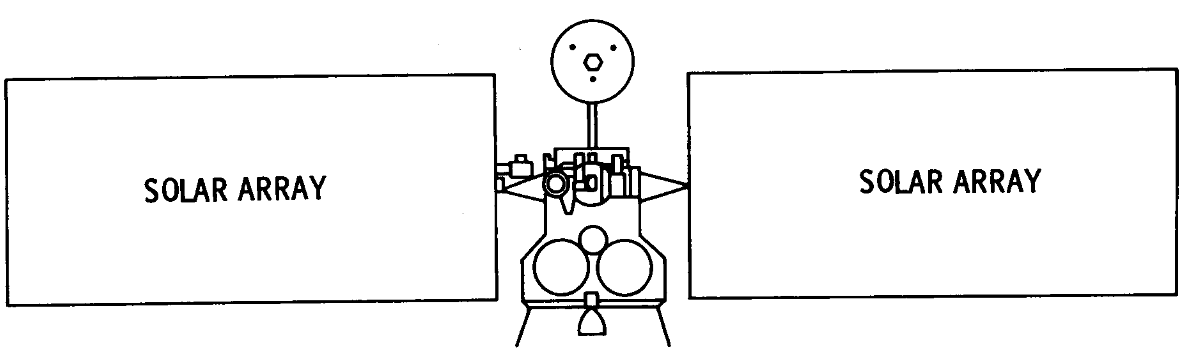

FIGURE 5. CONCEPTUAL MMII CRAF SPACECRAFT FULLY SOLAR POWERED