Non-Nuclear Power Sources for Deep Space

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NON-NUCLEAR POWER SOURCES FOR DEEP SPACE

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ABSTRACT

An analysis is made of the synergism of electric propulsion and non-nuclear power for deep space applications (i.e., to the asteroid belt and beyond). In current generation systems, electric propulsion is usually considered to be impractical because of the lack of high power for deep space, and non-nuclear power is thought to be impractical partly due to its high mass. However, when taken in combination, a solar powered electric upper stage can provide ample power and propulsion capability for use in deep space.

Three decades ago, the mass of solar array/battery systems was so high as to preclude their practical use in deep space applications, but this is no longer the case. The high efficiency of today's solar cells coupled with the remarkable energy storage capability of modern secondary batteries makes this a feasible technology in the near term, as demonstrated by the impressive capabilities of the SCARLET power system as well as other arrays. This paper examines in a preliminary way the possibility of using solar electric propulsion as an upper stage to boost a payload from low earth orbit to a trajectory which would take it economically as far as the orbit of Jupiter (as a hypothetical reference point), while providing the same amount of electric power at Jupiter as could be supplied by two radioisotope thermoelectric generators (RTGs).

BACKGROUND

Solar Concentrator arrays have intriguing benefits in performance and cost for spacecraft applications. The use of contoured Fresnel-type lenses permits high solar light concentration ratios with modest pointing accuracy requirements. This overcomes the difficulties associated with precision mirror type systems developed in the 1980s.

Although the original motivation for considering concentrator systems was performance, there is also a surprising cost advantage. Because only several percent of the array is actually solar cells, this means that the total cross sectional area of solar cells is much less than in typical flat panel arrays. Since the primary cost driver on a solar panel is the cost of the semiconductor materials, relatively high cost (per unit area) multibandgap solar cells can be incorporated on the array while still providing a substantial cost savings.

An important demonstration of this technology is being carried out on systems such as the SCARLET (Solar Concentrator Array with Refractive Linear Element Technology) system, developed under BMDO auspices, to be used on the Deep Space 1 probe as part of the Millennium program for JPL. In addition to an asteroid, comet and Mars flyby, the probe will demonstrate key technologies which will advance the state of the art both for scientific and commercial space missions.

The SCARLET array will use Tecstar cascaded multibandgap concentrator cells, with curvilinear Fresnel lenses to concentrate incident solar radiation. The technology used for SCARLET is capable of being scaled to significantly higher power levels. The advantages include low mass, high structural stability, and reduced number of cells required for equivalent amounts of power compared to conventional arrays.

CONCERNS ABOUT RADIOISOTOPES

Radioisotope thermoelectric generator (RTG) systems have generally been selected for missions only when other systems are absolutely unavailable. The disadvantages of radioisotopes include the need for nuclear safety as another dimension of concern in payload integration; the lack of assured availability of plutonium in the post-cold-war world; the enormous cost of plutonium-238; and the system complexity introduced by the need to continuously cool the system during the pre-launch phase.

NON-NUCLEAR ALTERNATIVES FOR DEEP SPACE

It is now possible to consider the use of non-nuclear technologies for space probes to the orbit of Jupiter and probably well beyond it. The technology for large solar arrays is now sufficient to provide power for a space probe to Jupiter orbit and beyond. Although the solar irradiance at Jupiter is only 51 W/m^2 compared to 1380 W/m^2 near the earth (or about 3.7 percent), this is nevertheless sufficient to power space probes at similar levels as Galileo. The Galileo vehicle used two RTGs capable of producing 285 W(e) initial power, and about 265 W(e) at Jupiter (i.e., 530 W(e) total).[1] Thus, for a space probe requiring two RTGs, the equivalent solar array would have to be sized for at least 14.3 kW(e) in earth orbit. An additional overcapacity should be used to account for possible degradation en route. Additional capacity is required for batteries if substantial operation is required in Jupiter eclipse, or perhaps if high power sensors are required. Thus, a conservative estimate for the total power for the solar array at beginning of life (BOL) may be in the range of 25 kW in order to provide 500 W continuous power at Jupiter.

An array of this power would have been unthinkable in the 1960s because efficiencies were only a few percent, while structures were much heavier and cells degraded more quickly. However, it is clearly possible with today's advanced technology.[2]

The weight penalty for a large array is substantial. For example, the SCARLET array is designed for 48 W/kg. Thus a 25 kW array might weigh about 500 kg (plus an additional mass for batteries and power conditioning). This compares to about 112 kg for two RTGs. Thus the mass penalty seems prohibitive, as the mass of the power supply plus propulsion would exceed the mass capability of the Inertial Upper Stage (IUS). However, this objection might change with the inclusion of electric propulsion.

ELECTRIC PROPULSION

The availability of ~25 kW(e) in earth orbit raises the interesting possibility of coupling electric propulsion units to this "free" electric power. If electric propulsion is used to raise the probe from low-earth-orbit to an earth-escape trajectory, the system could actually save on low-earth orbit mass. Electric propulsion could be used by itself in a spiral trajectory orbit raising maneuver to earth escape velocity, or it could be used in conjunction with a chemical upper stage (either solid rocket or liquid), which would boost the payload to an elliptical orbit.

For example, Space Power Inc. of San Jose, CA is currently marketing Closed Electron Drift Hall Effect Thrusters (HET) to 4.5 kW(e) which produce a specific impulse of 1900 s with 56% efficiency. Key advantages of the HET are (1) long operating life, and (2) reliable and predictable performance and (3) excellent I_{sp} and efficiency.

This generation of thrusters offers better performance combined with longer life and much lower electromagnetic interference (EMI) than its predecessors. Over 12000 hours of life tests have been performed on individual HET thrusters in ground tests in addition to over 3000 hours of operation in space.

TRANSFER TRAJECTORIES

Our concept is to begin the Earth-Jupiter trip with a swing-by near the Sun—close to the orbit of Venus and perhaps even closer if thermal loads can be tolerated. During the solar swing-by, much more power will be produced by the solar panels, allowing the spacecraft's velocity to be increased significantly. The outbound leg of the journey can, therefore, be made much more quickly than with the classical trajectory. For constant power, an optimum low-thrust trajectory tends to have two legs with thrust programs that are nearly mirror images of each other in order to match the boundary conditions at both ends of the journey. A typical trajectory for low thrust appears schematically below, along with the proposed solar swing-by trajectory. It is anticipated that the first leg of the solar swing-by option will consist mainly of gathering velocity while heading toward the sun and then back out to approximately one Earth's orbital radius (i.e., 1 AU). The resulting trajectory will have a large radial component at that point. The second leg, from about 1 AU to Jupiter, will consist mainly of adjusting the velocity to match boundary conditions at Jupiter.

The concentration of light energy on the cell is not a feasibility issue. Predecessors to the SCARLET project such as SCOPA were based upon solar concentration ratios nearly an order of magnitude higher than encountered in SCARLET. However, the associated temperature rise is more of a concern, since efficiency will be reduced. The SCARLET cells operate at about 68 C (i.e., in earth orbit). Near Venus, the temperature would rise to about 125 C, according to informal estimates.[4] The tandem $GaInP_2/GaAs$ cells have an efficiency correction of about 0.2 %/C. Thus at 125 C, the cells would produce about 89% of their nominal efficiency at 68 C. The total array power near Venus orbit would be about 42 kW for a nominal 25 kW array.

To attain even higher temperatures, additional cooling capability would probably be necessary. The main challenges for operation at 125 C and above will be to develop a high temperature electrical contact and bonding agent for the cells, as well as high thermal conductivity facesheets. For example, current facesheets have a thermal conductivity of 384 W/mK. An improved facesheet can be developed on the basis of carbon/carbon composites using Pyrograf™-I carbon fiber. Composite thermal conductivities as high as 950 W/mK have been achieved with carbon-carbon.[5]

The use of fins can also be considered as a means for optimizing the efficiency or for reducing the cell temperature without greatly affecting the mass. This can be attractive from a mass standpoint because the mass-normalized thermal conductivity of Pyrograf™-I carbon composite is about ten times higher than that of copper.[6]

In addition, high temperature, high electrical conductivity silicone adhesives are being developed using silver additives, as well as Pyrograf™-III nanofibers which maintain high electrical conductivity at 125 C and above.

This suggests that enhanced performance can be achieved in trajectories close to the sun.

Assuming an average available electric power of 30 kW during the time that the thruster is inside the orbit of the earth, the I_{sp} of 1900 s and efficiency of 56% means that a thruster burn consuming 10,000 kg of propellant (i.e., about 90% of the energy required for a Jupiter rendezvous, as shown below) could be accomplished in three years. This time could be substantially reduced if the array can operate successfully even closer to the sun, or if gravitational assists can be successfully used.

Of course, outside the orbit of the earth, the power system output (and hence the propulsive thrust capability) will decrease.

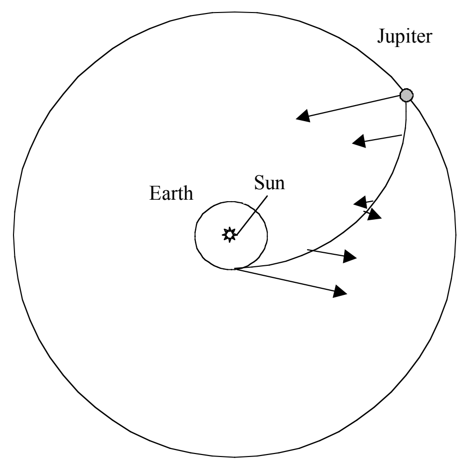

FIGURE 1. CONVENTIONAL CONSTANT LOW THRUST TRAJECTORY FOR JUPITER RENDEZVOUS.

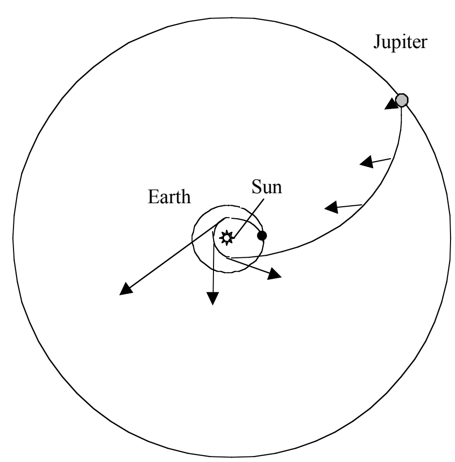

FIGURE 2. LOW-THRUST TRAJECTORY FOR SOLAR-ELECTRIC POWER USING A SOLAR SWINGBY.

OPERATIONAL CONSIDERATIONS

As an example of a deep space vehicle, the Galileo system can be considered as an excellent example. Galileo consisted of an "orbiter" system which orbited Jupiter as well as a system containing a descent module, referred to as the "probe." At launch, the orbiter mass was 2223 kg, including a 118 kg science payload and 925 kg of propellant. The probe mass was 339 kg, including 121 kg for the descent module itself, and 30 kg for a science payload. In all, the mass was 2562 kg. The total mass in low earth orbit for Galileo and the Inertial Upper Stage was 17304 kg.

For the purposes of a Jupiter mission, it is assumed that 20 km/s total delta-v would be required. For a payload envelope of 17304 kg, a 1900 s I_{sp} capability means that 11386 kg of propellant would have to be consumed, leaving 5917 kg for the mass of the probe plus dry mass of the upper stage. The mass for current-generation thruster, power processing unit, flow control unit, and miscellaneous structure is 21.2 kg for a nominal 4.5 kW thruster (4.5 kg/kW). Thus, a peak power of 170 kW would require 765 kg of thruster subsystem mass, and probably less. Assuming tanks, regulators and valves amount to 10% of the propellant mass (very likely a pessimistic assumption), it is possible to assign a mass of 1150 kg for the tankage subsystem. This results in a mass of 4000 kg for the probe. This compares favorably with the dry mass of 1637 kg for Galileo, and suggests that more than adequate margin exists.

If the payload margin is used for battery storage, it may be possible to contemplate flyby missions to the outer planets.

CONCLUSION

Solar power cannot be considered independently for use at Jupiter and beyond because of its mass, and electric propulsion cannot be considered independently because of its high power (energy) requirement. However, with today's technologies, the combination of high solar power and electric propulsion can be considered for deep space missions, and can potentially eliminate the need for radioisotopes for deep space missions.

Solar power technology can be based on the SCARLET array and can probably be further enhanced. In particular, this can be accomplished by improving thermal contact and heat removal methods to accommodate higher thermal loads. This might permit higher power operation near the sun.

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