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Harvesting Electromagnetic Energy from Hypervelocity Impacts for Solar System Exploration

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Abstract

CubeSats have opened new opportunities for solar system exploration by decreasing mission development and launch costs. While they have proven successful in the inner solar system, providing sustained power to these spacecraft at large distances from the sun requires novel power generation methods. Solar power is dramatically diminished at these distances and radioisotope thermoelectric generators do not scale well to spacecraft with small dimensions. The space environment could provide a solution; by harnessing energetic hazards in the environments under investigation, the spacecraft could forgo traditional power sources. Radio frequency emissions are abundant in the space environment with sources originating from planetary aurora, particle dynamics in the magnetosphere, and electromagnetic pulses (EMPs) associated with hypervelocity impacts between micrometeoroids and spacecraft. A case study of a mission to the rings of Saturn is presented and analyzed for feasibility. Upper bounds on the power emitted by the hypervelocity impact EMPs are produced by combining results from ground-based experiments and Saturnian environment models. We find that although the kinetic energy of hypervelocity impacts is significant, the conversion efficiency to EMPs is too low for spacecraft to effectively use.

1 Introduction and Motivation

Exploration of the outer solar system is an expensive endeavor and maximizing the science return of the few missions that will make the journey is of paramount importance to mission designers. Nanosatellites (spacecraft between 1-10 kg in mass) such as CubeSats (spacecraft made up of cubic units 10 cm to a side) succeeded in driving down costs for missions to Earth orbit and provided platforms for experimentation with mission concepts, hardware, and payloads. While solo flights to the outer planets may be currently infeasible for them, Cubesats traveling with larger motherships have the potential to advance science while controlling costs.

For example, nanosatellites can enable distributed sensing missions to the outer planets. Magnetospheres are large-scale, complex structures whose properties are constantly in flux. Understanding physical processes that drive the magnetosphere requires resolution of spatial gradients in plasma properties. Distributed sensing missions using swarms of

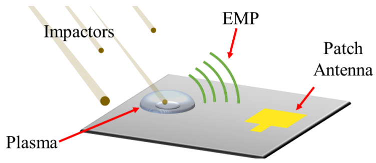

Figure 1. Graphical representation of the HVI energy harvesting concept.

CubeSats as free-flying sensors can produce plasma measurements spread over hundreds of kilometers — a significant improvement over the tens of meters that a boom-mounted sensor would provide. This concept has heritage in the Cluster [1] mission for example.

Power presents a challenge to small spacecraft in the outer solar system. The extreme distance from the sun precludes the use of solar panels for Cubesats, and radioisotope thermoelectric generators are expensive, difficult to handle, and bulky relative to the amount of power delivered. Another approach under consideration is to use the power within the space environment to power these spacecraft. Radio frequency (RF) sources within the space environment present a tempting target considering their abundance.

Hypervelocity impacts (HVIs) are one such source of radio frequency emissions. HVIs on spacecraft surfaces and airless bodies occur frequently in the space environment due to micrometeoroids, dust, and orbital debris (referred to collectively as impactors throughout this paper). As described in detail in [2], HVIs on metallic targets emit an electromagnetic pulse (EMP) which can interfere with spacecraft electronics, resulting in mission anomalies. Given that the emitted energy is significant enough to be destructive, we aim to characterize the amount of energy carried by the EMP and to determine whether it can be harnessed to power nanosatellites. Figure 1 illustrates this concept using a surface-mounted patch antenna.

This study analyzes the amount of expected energy available to spacecraft traversing Saturn's rings from impact EMPs. First, we present general expressions for the upper bound of energy available to a spacecraft given an impactor flux model and EMP measurements from laboratory experiments. Characteristics of the Saturnian space environment are derived from spacecraft data while our own measurements provide the expected EMP energy. Synthesizing these elements resutls in an estimate of the power available to these spacecraft.

2 Methodology

The physics governing the generation of the EMPs are not well understood, nor is the Saturnian impactor mass and velocity distribution completely known. In addition, the design space of harvesting circuit topologies is too vast to enumerate completely or optimize for the harmonic structure of the EMP. A number of simplifying assumptions, therefore, are required to obtain a usable power estimate. Energy conservation allows us to constrain the EMP energy to a fraction of the impactor kinetic energy, resulting in an upper bound on available energy. Using spacecraft geometry and physical models for the EMP energy as a function of impactor properties and the ambient impactor environment, we can compute the expected RF energy flux to a rectifying antenna (rectenna). This analysis assumes that EMPs result from every HVI event, that the EMP radiation pattern is isotropic, and that HVI measurements from lower velocity ground-based experiments can be extrapolated to orbital speed HVI events on spacecraft. Model-specific assumptions are presented in their respective sections.

The amount of energy in the electromagnetic fields E_{EMP} is less than the combined impactor kinetic energy and energy stored in the electrostatic field due to spacecraft charging. Since the charging state is variable, we use the kinetic energy alone as a weak upper limit. Experimental measurements of EMPs can be used to compute a tighter bound. The amount of energy E_c stored by a rectenna with frequency-dependent efficiency $\eta_0(f)$ is

$$E_c = \int_0^\infty \eta_0(f) ESD(f)\, df < \eta_{0,\max} \frac{\Delta\Omega}{4\pi} E_{\mathrm{EMP}} \tag{1}$$

where $ESD(f)$ is the energy spectral density of the incident signal, $\Delta\Omega$ is the solid angle subtended by the antenna elements, and $\eta_{0,\max} = \max_f \eta_0(f)$ is the peak efficiency of the antenna and rectifying electronics. In the far-field, $\Delta\Omega$ would simply exhibit the usual inverse square attenuation with distance. For small spacecraft, however, the rectenna is more likely to be in the near-field. As an example, for an antenna element with height ℓ and diameter d protruding at 90° from the spacecraft, $\Delta\Omega \approx \pi$ sr for nearby impacts and asymptotically approaches $2\ell d/r^2$ as the distance $r \to \infty$.

When more than one impactor impinges upon the surface, the power collected by the rectenna is computed using a model of the impactor flux density $F(m,u)\, du\, dm$ (number of particles with mass m and u impacting per area per second). Deployable structures can increase the number of impacts on the spacecraft and thus the harvested energy. Assuming that the deployable surface is circular with radius a with the antenna at its center, the power per unit impact area can be derived from (1) as

$$\Gamma_{\mathrm{EMP}} < \frac{A_{\mathrm{eff}}}{A} \eta_{0,\max} \int_0^\infty dm \int_0^\infty du\, F(m,u) E_{\mathrm{EMP}}(m,u) \tag{2}$$

where A is the impact area and

$$A_{\mathrm{eff}} = \frac{1}{2} \int_0^a r \Delta\Omega\, dr \tag{3}$$

is the effective power collection area. In the small spacecraft limit, $A_{\mathrm{eff}} = \pi a^2/4 = A/4$.

3 Physical models

Models for the electromagnetic field energy E_{EMP} and the meteoroid flux density $F(m,u)$ are needed to proceed further. The former can be estimated using detections of EMPs from terrestrial impact experiments while the latter is constructed using data from spacecraft measurements.

3.1 Electromagnetic pulse energy

An empirical power law relation between the equivalent isotropic radiated power, the mass m, and velocity u of the impactor is $P_{\mathrm{EMP}} \propto m^\alpha u^\beta$ where $\alpha \approx 1$ and $\beta \approx 3.5$ [3]. This relationship agrees with similar empirical laws describing the amount of charge produced in HVI experiments [4]. We use this proportionality to estimate E_{EMP} over a range of masses and velocities.

$$E_{\mathrm{EMP}} = C m^\alpha u^\beta \tag{4}$$

Data from experiments can be used to determine the constant of proportionality C. We recently conducted a set of experiments using the light gas gun at the NASA Ames Vertical Gun Range (AVGR). In these tests, small, 1.6 mm diameter aluminum spheres were fired in partial vacuum (0.5 Torr) at speeds between 4-6 km/s. Targets were made of various materials and biased to different voltages mimicking spacecraft charging conditions. A suite of RF sensors — including patch antennas, dipole antennas, and a 0.9-1.8 GHz log-periodic monopole antenna (LPMA) — was placed inside the chamber about 1 m from the target. We detected EMPs on two of the thirteen shots ($m = 5.9$ mg, $u = 5.4$ km/s) on aluminum targets biased to -300 V.

Given its broadband characteristics, the LPMA was used to estimate the EMP energy. The signal energy within a short temporal window around the EMP was computed, accounting for amplifier gain, cable losses, and space losses. For these shots, $E_{\mathrm{EMP}} \approx 70$ nJ. Using Equation (4), this results in a proportionality constant $C \approx 1 \times 10^{-15}$ when m and u are measured in SI units. Orbital impactors can travel at speeds up to about tens of km/s, bringing with them comparatively higher EMP energies. In addition, the flux of impactors can be quite high, especially in the rings of Saturn.

3.2 Saturn environment characteristics

Equation (5) gives the in-plane density profile of impactors within Saturn's rings derived from Cassini Radio and Plasma Wave Science data [5].

$$n(\rho) = n_0 \left(\frac{\rho}{3.95}\right)^p \tag{5}$$

where $n_0 = 0.035\,\mathrm{m}^{-3}$, ρ is the orbit radius measured in Saturn radii, and $p = 16 \pm 2$ for $\rho < 3.95$ and $p = -10 \pm 1.6$ for $\rho > 3.95$. The peak density at $\rho = 3.95$ coincides with the orbit of Enceladus, indicating a promising destination for harvesting energy. Above and below the ring plane, the density falls, following roughly a Lorenztian distribution in the plane normal coordinate.

The density given in Equation (5) is properly an integration over the impactor mass and velocity distribution. Gurnett [6] approximates the mass distribution using Voyager 2 data from a ring crossing at 2.88 Saturn radii. At peak intensity during the crossing, several hundred impacts occurred every second. This distribution takes the form of an inverse cube law.

$$\frac{dn}{dm} = n\frac{2m_0^2}{(m_0+m)^3} \tag{6}$$

where m is the impactor mass, m_0 is the mean impactor mass and n is the density at the orbit radius of interest. Though m_0 is quoted as 1.44×10^{-13} g by Gurnett [6], a more recent estimate [7] places the mean mass at 7.7×10^{-11} g, though the discrepancy could be due to the location of the ring crossings used in the estimation. Combining these two models results in an estimate of the impactor distribution at the orbit of Enceladus.

Finally, the velocity distribution of the impactor must be estimated. For a particle traveling at $\vec{u}_{\mathrm{imp}}$, the impact speed is given by $\Delta u = \|\vec{u}_{\mathrm{sc}} - \vec{u}_{\mathrm{imp}}\|$ where $\vec{u}_{\mathrm{sc}}$ is the velocity of the spacecraft. Even for impactors above and below the ring plane, the angular momentum vector averaged over the distribution should be oriented along the normal to the ring plane. Therefore, as an approximation, the mean velocity vector should lie in this plane as well. We estimate the mean speed to be the circular orbit speed with radius equal to the projected radius in the ring plane. In this model, the impactors orbit on average like concentric cylinders, rotating at $\omega = \sqrt{\mu/\rho^3}$ where μ is the Saturnian gravitational parameter and ρ is the projected radius in the ring plane. Although this model tends to overestimate the velocity in polar orbits, the densities in polar regions are sufficiently tenuous that the estimates are not significantly affected.

An upper limit on the impact speed is found when the impactor and spacecraft are traveling in opposite directions. Further, the speed of any impactor in the ring population is bounded from above by the escape speed $\sqrt{2\mu/r}$. Therefore, the upper limit on Δu is given by $\Delta u_{\mathrm{lim}} = u_{\mathrm{sc}} + \sqrt{2\mu/r}$.

Using either model for Δu, the velocity dependence can be approximated as a delta function, peaked at the speed Δu. With this approximation, the impactor flux distribution can be written as

$$F(m,u) = nu\frac{2m_0^2}{(m_0+m)^3}\delta\left(u - \Delta u\right). \tag{7}$$

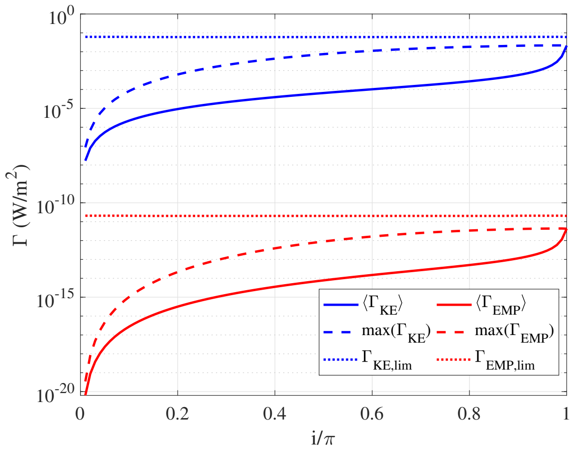

Figure 2. Orbit averaged (solid), maximum (dashed), and upper bound (dotted) estimates of power per unit area incident on the spacecraft due to the impactor kinetic energy (blue) and EMP energy (red) incident on the spacecraft.

4 Results and Discussion

Equipped with these physical models, we can estimate the utility of using impact EMPs as power for spacecraft for several orbits around Saturn. As a threshold for usefulness, we use 77 mW: the power used by the sprite spacecraft deployed from Kicksat [8].

Substituting the models from Equations (4) and (7) into Equation (2), the expression for the power per unit area Γ_{EMP} satisfies

$$\Gamma_{\mathrm{EMP}} < \frac{A_{\mathrm{eff}}}{A}\eta_{0,\max} n m_0 C(\Delta u)^{9/2}. \tag{8}$$

The remaining parameters A_{eff}/A, and $\eta_{0,\max}$ still require specification. For small spacecraft, the ratio of areas can be roughly estimated as $1/4$ in the $a \to 0$ limit from Equation (3). The efficiency $\eta_{0,\max}$ is assumed to be 10%, but in general depends upon the precise antenna and rectifier topology.

The density and velocity difference are functions of the orbit used by the spacecraft. For simplicity, consider a circular orbit with radius equal to that of the semi-major axis of Enceladus and inclination varying between 0° and 180°. With the approximations used, the peak power for the equatorial, prograde orbit is zero, then increases steadily with inclination as Δu grows. Using the concentric cylinder model, Δu is found to be

$$\Delta u = \sqrt{\frac{\mu}{r}}\sqrt{1 + \frac{r}{\rho} - 2\left(\frac{r}{\rho}\right)^{\frac{3}{2}}\cos i} \tag{9}$$

where r is the orbit radius, $\rho = r\sqrt{1-\sin^2 i \sin^2 \nu}$, i is the inclination and ν is the true anomaly. Most of the energy harvesting will occur as the spacecraft crosses the ring plane where the debris is most dense. The absolute upper

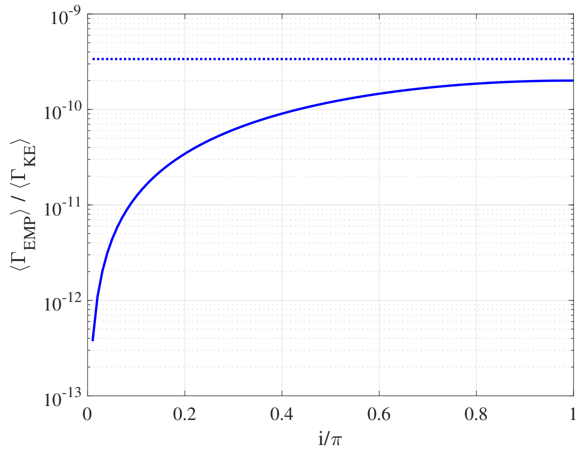

Figure 3. EMP conversion efficiency for each orbit. The dotted line represents the upper bound on this efficiency computed using Δu_{lim}.

limit on Δu for this orbit type is $\Delta u_{\text{lim}} = (1+\sqrt{2})\sqrt{\mu/r}$. Inserting the spacecraft in an elliptical orbit provides a slight enhancement; the upper limit at periapsis becomes $\Delta u_{\text{lim,e}} = (\sqrt{1+e}+\sqrt{2})\sqrt{\mu/r}$ where $0 < e < 1$ is the orbit eccentricity. Therefore, the absolute upper limit for any orbit is $\Delta u_{\text{lim}} = 2\sqrt{2\mu/r}$ during ring plane crossings.

Figure 2 shows the power per unit area averaged over the circular orbit $\langle \Gamma \rangle$ from the kinetic energy of the impactor $\Gamma_{\text{KE}} = \frac{1}{2} m_0 (\Delta u)^3$ and due to the EMP Γ_{EMP}. Γ_{KE} can be thought of as the maximum power available to be collected from HVIs in these orbits, from the EMP or through other means. The maximum power seen by the spacecraft $\max(\Gamma)$ at any point in the orbit is also shown, along with the upper limit on the power Γ_{lim} for any orbit. Figure 3 shows the ratio $\langle \Gamma_{\text{EMP}} \rangle / \langle \Gamma_{\text{KE}} \rangle$, which serves as an average conversion efficiency. The retrograde orbit $i = \pi$ results in the highest available power and average conversion efficiency. If kinetic energy could be harnessed in its raw form with perfect efficiency, the maximum power captured would be 22 mW/m^2 while the amount captured from the EMP is only 4.3 pW/m^2. For comparison, the expected solar power at Saturn is about 4.2 mW/m^2 assuming 30% efficient solar cells facing sunward. It is clear that, to attain 77 mW sustained power, the spacecraft would require an infeasibly large impact area or number of rectennas.

Although Γ_{EMP} is linearly dependent on m_0, the conversion efficiency is dependent on Δu alone. Since this efficiency increases by powers of 3/2 with the relative speed, the expected power generation increases with more energetic velocity distributions, but is limited by the escape velocity.

5 Conclusions

Compared with other solar system environments, the large density of impactors in the Saturnian rings coupled with high relative speeds of transiting spacecraft result in kinetic energy fluxes to the spacecraft surface on the order of the solar constant at Saturn. However, we have shown that the conversion rate from kinetic to electromagnetic energy of the EMP generated in HVIs is far too low to power even minimal electronics on small, free flying spacecraft. Other sources of power in the environment — such as other natural radiators or spacecraft charging — may prove more effective in realizing the distributed sensing architecture with nanosatellites.

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