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Analysis of the charged particle radiation effect for a CubeSat transiting from Earth to Mars

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ABSTRACT

This paper presents a computational estimation of the total ionizing dose from protons and electrons in the Earth's magnetosphere and interplanetary space for a hypothetical CubeSat transiting from Earth to Mars. An initial hyperbolic escape of the spacecraft from Earth's gravitation is assumed, followed by an elliptical transfer from Earth to Mars under the Sun's gravitation. The rapid traversal of the Earth's radiation belt yields a smaller ionizing dose, whereas high-energy solar protons in the interplanetary space have the greatest effect on the ionizing dose during the transfer between the planets. Variation in the heliocentric distance of the spacecraft is considered in the calculation. Calculation of the shielding distributions with Geant4 and the transport of the ionizing particles across the obtained distributions yields an estimation of the total ionizing dose as a function of position within the spacecraft as well as statistical confidence levels. With a moderate confidence level, this calculation shows that a practical exploration of Mars with a CubeSat is possible in terms of the expected total ionizing dose.

1. Introduction

The initial concept of the CubeSat was to develop and utilize a spacecraft weighing in the range of 1–3 kg for use in education [1], but the subsequent evolution of this concept has provided many researchers in the field of space exploration with more frequent and less expensive access to space. Compared to conventional satellites, CubeSat offers the possibility for a smaller, more versatile space vehicle due to the advent of modern technology such as microelectronics and nanotechnology. This small spacecraft has been applied to various disciplines including space science, remote sensing, astrobiology, and engineering [2–9]. While such a cost-effective solution to accessing near-Earth space is a significant step forward, the performance and lifetime expectations for this spacecraft are limited by several constraints such as physical size, available electrical power, accuracy of the attitude control, and absence of propulsion. In particular, the absence of a propulsive device for the spacecraft has severely restricted its range to altitudes below about 1000 km.

However, creative ideas and innovative technology are being pursued to accomplish more challenging missions. For example, the planetary hitchhiker concept [10] involves a CubeSat (daughter-ship) serviced with a propulsive module (mother-ship). In this mission concept, the insertion to Geostationary Transfer Orbit (GTO) is directly made by a conventional launch vehicle. Then the mother-ship brings the daughter-ship to desired interplanetary trajectory by performing several apogee-raising burns with its chemical propulsion system. The mother-ship finally releases the CubeSat (daughter-ship) for scientific experiments once the desired orbit is achieved. A propulsive device specifically designed for CubeSat missions [11] is also being investigated to allow a mission concept with its own propulsion and without the propulsive service platform. Such challenging missions have yet to be realized, but strong demand to explore astrophysical objects, such as the Earth's magnetosphere, Moon, planets, and near-Earth asteroids, will eventually enable us to perform interplanetary investigations with the effective approach of CubeSats.

Modern *in-situ* observations in space have found that the vicinity of Earth is occupied with charged particles, which are often trapped by Earth's magnetic fields [12]. Later observations further showed that these regions are dynamically filled with charged particles that are diverse in energy [13], species [14], and origin [15]. It has been also discovered that the distribution of plasmas in interplanetary space is dominated by the flow of charged particles from the outer atmosphere of the Sun. Extreme augmentation of these interplanetary particles in energy and flux is occasionally

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encountered. For example, during the solar proton events of October 1989, strong enhancements of proton energy up to the GeV range and orders of magnitude increases in particle fluxes were found both on the ground [16] and in interplanetary space [17]. Observations indicate that the occurrence of energetic solar activities such as solar flares and Coronal Mass Ejections (CMEs) is closely related to enhancements of particle energy and flux in interplanetary space.

Therefore, if CubeSats are to be deployed for interplanetary missions, they must survive the strong radiation effects of high-energy charged particles from Earth's intense radiation belts, solar flares, and CMEs. This is one of the most critical issues that must be addressed to extend the applicability of the CubeSat to interplanetary missions. In this paper, we calculate the expected ionizing radiation dose for standardized CubeSat structures on an interplanetary mission to Mars. We consider the trajectory to Mars and the presence of solar particles along the orbit together with populations of trapped particles in Earth's magnetic fields. The current solar proton model is based on measurements made near Earth, and is therefore valid only for predictions at that location. On the other hand, the distance a spacecraft travels to Mars is great enough that heliocentric variation of high-energy proton fluxes must be taken into account. We attempt to estimate the solar proton fluences for the trip. Mars remains one of the focal points of scientific investigation and practical exploration. The result of this calculation will be informative to those designing CubeSat spacecraft for the purpose of undertaking interplanetary missions. In Section 2 we briefly review the calculation process, followed by a description of assumed interplanetary orbits from Earth to Mars. Sections 3 and 4 offer our results and summary, respectively.

2. Calculation method

The analysis starts with numerical modeling of the charged particle populations along the orbit trajectory and geometrical modeling of the spacecraft shielding structure. The shielding distribution is then calculated, followed by the Total Ionizing Dose (TID) at specified locations. Part of the present paper utilizes the method of the previous investigation by Seo et al. [18] and Kim et al. [19]. The following sections provide a summary of the tasks executed at each step of the calculation.

2.1. Mission orbit to Mars

This hypothetical mission to Mars has two orbital phases, one corresponding to the escape from the Earth's gravitation and the other corresponding to the cruise phase from Earth's orbit to Mars's orbit under the Sun's gravitation. We assume that the initial orbit of the spacecraft is at an altitude of 600 km. From this conventional initial orbit, the spacecraft makes a hyperbolic escape from the Earth's gravitation with assistance from the assumed mother-ship or with its own propulsion. When the spacecraft achieves the asymptotic trajectory away from the Earth's gravitation, the velocity of the spacecraft must be enough to ensure elliptical transfer from Earth's orbit to Mars's orbit. The present analysis is equivalent to utilizing the concept of patched conic approximation [20] without anticipating a high degree of accuracy for the calculated trajectory. More accurate calculation of the trajectory may be achievable, but it is not expected to provide significantly better results than the present approximation.

Because the rotation speed of Earth with respect to the Sun is about 29.78 km/s and the required speed of the spacecraft from Earth to Mars is 32.73 km/s in the rest frame of the Sun [20], the velocity to be attained by the spacecraft at the asymptotic orbit is $v_\infty = 32.73 - 29.78 = 2.95$ km/s parallel to the orbital motion of Earth. The semi-major axis (a), the semi-minor axis (b), and the angle of asymptote (β) corresponding to this asymptotic velocity of the spacecraft should be $a = -45{,}802$ km, $b = 26{,}196$ km, and $\beta = 29.97°$, respectively. The sign of the semi-major axis is negative because the spacecraft has a finite positive total energy along the asymptote. The spacecraft is initially in the ecliptic plane, and remains in that plane throughout the escape. Fig. 1 shows the geometry of the hyperbolic escape. The figure is drawn in the solar ecliptic plane with an origin at the center of Earth. The positive horizontal (left) and vertical axes (down) in the figure correspond to sunward and duskward directions, respectively. Additionally, dashed lines denote the location of the model magnetopause [21], where rough separation of flowing solar plasmas is found from the Earth's magnetic fields. Because the origin of charged particles changes across the magnetopause, we include the contribution of the trapped charged particles in the Earth's magnetic fields only up to the location of the magnetopause. After exiting the magnetopause, the spacecraft is exposed to the flow of charged particles from the Sun.

Outside the magnetopause, whose predicted radius is ~93,017 km (14.6 R_E, where R_E = radius of Earth), the flow of charged particles from the Sun defines the radiation environment of the spacecraft. We assume that the spacecraft begins to follow an interplanetary trajectory from this location under the gravitational influence of the Sun. The previous hyperbolic trajectory allows this elliptical transfer to Mars because the asymptotic velocity of the spacecraft adds to the orbital velocity of Earth to yield the necessary velocity of the elliptical trajectory. The perihelion and aphelion distances of the ellipse are the orbital radii of Earth and Mars, respectively. In order to closely approximate the real interplanetary trajectory to Mars, we use a trajectory similar to that of the Mars Science Laboratory [22]. The spacecraft departed from Earth's orbit on 26 November 2011 and arrived at Mars's orbit on 6 August 2012. The dose rate within the Mars Science Laboratory was recorded during the 254 days of interplanetary travel [25]. The trajectory corresponding to this interplanetary journey is shown in Fig. 2. The figure is given in the solar ecliptic plane with the origin centered at the Sun. The abscissa points toward the vernal equinox as viewed from Earth to Sun. The figure is given in Astronomical Units (AU) which is roughly the average Sun-Earth distance of 1.496×10^6 km.

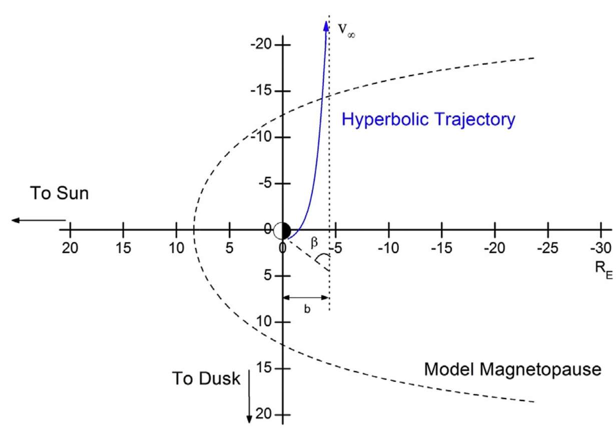

Fig. 1. Hyperbolic escape of a CubeSat from Earth's gravitation. The spacecraft follows a hyperbolic trajectory with its asymptotic speed just sufficient to ensure a proper elliptical transfer from Earth's orbit to Mars's orbit. The asymptotic velocity of the spacecraft corresponds to the velocity difference between the Earth's rotation velocity and the required velocity from Earth to Mars with respect to the Sun. Shown together with the spacecraft trajectory is the location of the modeled magnetopause [22], which roughly separates the flow of solar wind plasmas from the Earth's magnetic fields. After crossing the magnetopause, the spacecraft is exposed to the solar wind plasmas. Therefore, the effect of the trapped population of Earth's radiation belt is calculated only up to the crossing of the magnetopause. The figure is given in the ecliptic plane centered on Earth and viewed from the north.