Summary

Results of a thermal analysis of a solar thermal engine showed that the fluid chamber temperature was within 40°F of the steady-state temperature of 3850°F after 30 min of operation, was relatively insensitive to changes in cooling duct convection coefficient and absorber surface emissivity, and was very sensitive to the hydrogen mass flow rate.

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Radiation Exposure Comparison of Venus and Mars Flyby Trajectories

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Introduction

FOLLOWING the lead of the Apollo lunar flyby missions that predated Apollo 11, a Venus or Mars flyby mission would serve to verify the ability to sustain astronauts in the interplanetary environment without the additional complexity of a Mars entry and landing. In addition to assessing propulsion and life-support requirements, the mission planner analyzing a trajectory for a crewed Mars transportation and habitation vehicle (TransHab) demonstrator must consider radiation exposure and related hazards. An approximate radiation model has been developed to quantify the radiation hazard of trajectories as a function of time and distance from the sun. Interplanetary spacecraft radiation exposure is approximated by incorporating models for galactic cosmic radiation (GCR) and solar particle events (SPEs). GCR is the ambient radiation experienced from energetic processes of the stars and galaxies outside this solar system. This radiation decreases in intensity with proximity to the sun. SPEs include solar flares and are relatively brief, intense periods of radiation that increase with solar proximity.

Ballistic Earth–Mars–Earth (EME) and Earth–Venus–Earth (EVE) trajectories for the period beginning 1 January 2002 and ending 31 December 2011 have been developed using a hybrid global–local parameter search method.[1] The EVE 2010 and EME 2006 trajectories represented in Fig. 1 were determined to be favorable candidate missions based on their propulsion requirements and time of flight. Risk assessments of the EVE and EME trajectories are given and the Venus flyby trajectory will be shown to be superior to the Mars flyby in terms of radiation exposure.

Table 1 Acute dose effects on humans

Effect	Dose, RAD
Blood-count changes	0–50
10% Chance of vomiting	80–120
25% Chance of vomiting, other symptoms	130–170
50% Chance of vomiting, other symptoms	180–220
20% Chance of death in 2–6 weeks	270–330
50% Chance of death in 1 mo	400–500
Nausea within 4 h, few survivors	550–750
Nausea within 1–2 h, no survivors	1000
Immediate incapacitation	5000

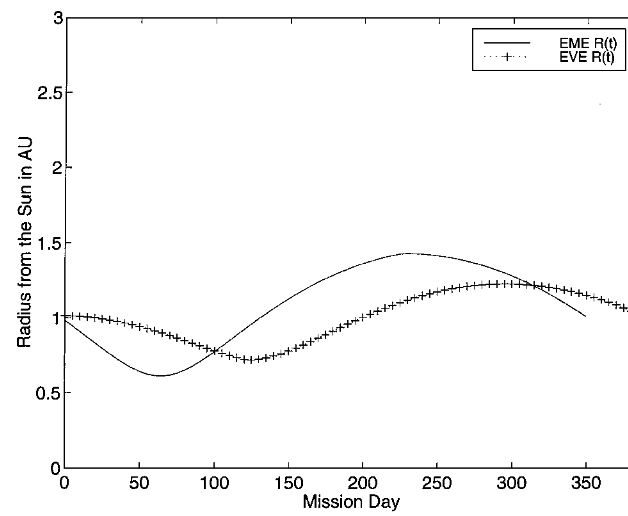

Fig. 1 EME 2006 and EVE 2010 mission radius vs mission day.

Radiation Model

Shielding of humans and equipment from interplanetary radiation can be accomplished in a number of ways: 1) dedicated shielding material in specific locations, 2) efficient and thoughtful placement of inert spacecraft mass, and 3) active magnetic field generation shielding in a manner that emulates the Van Allen belt.[2–4] The model developed here uses a uniform water shield of variable thickness in its calculation of exposure to interplanetary radiation. The interplanetary radiation environment is composed of energies from four sources: 1) radiation emitted from onboard medical equipment, power sources, and propulsion systems; 2) steady-state flux of particles, primarily protons, from the sun, known as the solar wind; 3) low flux radiation (~4 particles/cm^2) of energetic ionized nuclei encountered from the high-energy activity of surrounding stellar bodies known as GCR[5]; and 4) intermittent radiation emitted from the sun during SPEs, such as solar flares and coronal mass ejections.[4]

For the purpose of this study, radiation from onboard sources is assumed to be low level and well shielded for the duration of the times of flight of the EVE and EME mission profiles. The steady-state solar flux is also of a negligible magnitude in comparison with the nominal range of GCR radiation. Thus, only GCR and SPE are considered in connection with potential radiation hazards. The effects of radiation can either manifest immediately from exposure to very high doses or may manifest in the long term from the cumulative doses encountered in an astronaut's career or equipment's operational lifetime. A tentative guideline for human exposure to immediate radiation doses over varying lengths of time in operation of the International Space Station (ISS) is given in Table 1 (Ref. 4). A long-term dose guideline for cumulative exposure limits daily, monthly, and annual roentgen equivalent in man (REM) to 25, 50, and 100–500 units, respectively.[5]

The units of radiation effect in these guidelines are the radiation dose (RAD) and the REM. A RAD is equal to 0.01 J/kg and is used primarily as a material-independent measure of radiation. The REM

Received 1 March 2000; revision received 2 June 2000; accepted for publication 2 June 2000.

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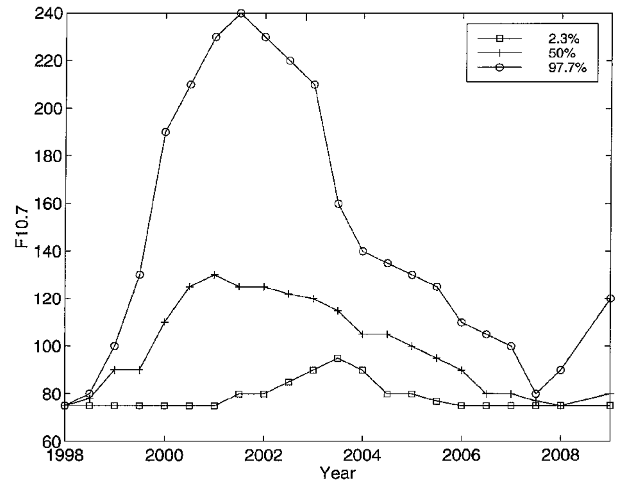

Fig. 2 Solar activity for solar cycle 23.

is a unit of radiation that is equal to the RAD biological tissue is exposed to scaled by a relative biological equivalent (RBE) factor

$$\mathrm{REM} = \mathrm{RAD} \times \mathrm{RBE} \tag{1}$$

For the spectra of radiation included in the GCR and SPE phenomena,[5] the RBE is approximately 10. A quick inspection of these guideline values will indicate that the career ISS dose limits can be reached with a small number of exposures to the relatively unnoticeable blood-count changing levels of 0–50 RAD.

The GCR radiation dosage is modeled as a 60 REM/year at 1 astronomical unit (AU). However, the GCR flux is mitigated by interaction with the solar wind and decreases as distance from the sun decreases. The GCR is modeled with a 6% per AU gradient to approximate this effect:

$$\Gamma_{\mathrm{GCR}} = [60 + 0.06(R - 1)]\Delta T \tag{2}$$

where Γ_{GCR} is the GCR dose in REM and ΔT is the time (years) spent at radius R (AU) from the sun.

SPE occur randomly and deliver a high-intensity radiation to the spacecraft over a finite time. SPE occurrences can be detected by short-term event monitoring and statistically modeled in long-term activity analysis. Event monitoring can be used during a mission to advise shelter for imminent particle activity. Long-term activity analysis seeks to indicate the likelihood of solar events with respect to an 11-year sunspot cycle and can be used for gross mission timing studies. A statistical prediction of solar activity cycle 23 is shown in Fig. 2. The percentage associated with each curve represents the probability with which that curve overestimates the activity for that time. The F10.7 value is the standard unit of 10.7-cm wavelength flux used to measure solar activity. The importance of this F10.7 activity cycle is that it is strongly correlated to sunspot and SPE occurrence.[5] Typical solar cycles peak at approximately 4 years from the previous cycle's minimum value.

The dose effect for a given magnitude SPE varies with the surface area a sphere with a radius equal to the distance from the sun at which the radiation is encountered. Therefore, the REM encountered at 1 AU was scaled by the inverse square of radius at time of exposure

$$\Gamma_{\mathrm{SPE}} = \Gamma_{1\mathrm{AU}} \big/ R^2 \tag{3}$$

where Γ_{SPE} is the dose in REM encountered at radius R by an event that would have a magnitude $\Gamma_{1\mathrm{AU}}$ at 1 AU. Large amounts of time spent near the sun during high activity periods would not be advisable, but mission hazards may be assessed in terms of the effect of GCR and SPE over a given radius history. The time-varying heliocentric radii of the EVE 2010 and EME 2006 trajectories are given in Fig. 1, and the inverse square of each radius, which is the 1 AU SPE flux scale, can be calculated according to Eq. (3). As the EME trajectory moves to its perisolar point 70 days into the mission, the scaled exposure of a SPE reaches a maximum of 2.6 times greater than the same event experienced at 1 AU. The EVE trajectory hits its closest approach at 125 days for a 1.9 SPE scale factor. In both trajectories, the greatest potential hazard from SPE radiation occurs at the mission perisolar point. Because the SPE is a stochastic event occurring over a finite interval, this hazard must be evaluated in terms of the current solar activity cycle and the likelihood of a SPE near the time of closest solar approach.

Radiation Exposure Analysis

Rather than discuss specific spacecraft physical designs, the radiation hazards of any particular mission will be evaluated in terms of 1) potential SPE radiation exposure to the external surface of a spacecraft mentioned earlier and 2) radiation exposure from both GCR and SPE considering a uniform shielding of 20-cm thickness water[4] to reduce the absorbed REM described by Eqs. (2) and (3). The combined GCR and SPE radiation plot in Fig. 3 contains the mission and monthly dosage limits, the monotonically increasing accumulated GCR dose, the effect of a solar event modeled after the August 1972 SPE, and the combined SPE and GCR potential exposure for both missions. For the radius history of these missions, the gradient of GCR exposure is relatively constant, and the radiation model indicates approximately 30 REM for the EME trajectory and 32 REM for the EVE trajectory. The recommended yearly exposure is multiplied by the mission time (in years) to provide the 48 REM and 53 REM mission limits for the EME and EVE trajectories, respectively.

The SPE exposure curve represents the REM absorption that would be experienced if the August 1972 magnitude SPE occurred on a particular mission day. The EME 2006 trajectory can be seen to be in danger of exceeding the monthly limit from mission day 30 to mission day 100. By combining the EME mission GCR dose of 30 REM with the SPE exposure curve, the combined potential exposure is calculated and labeled as total mission REM. In this illustration, if the single modeled SPE were to occur between 20 and 125 days into the EME mission, both the monthly and yearly mission exposure limits would be exceeded.

Specific examples of the interpretation of Fig. 3 will clarify its utility in analyzing radiation exposure hazards. If the 1972 magnitude SPE were to occur 125 days into the EME mission, the total radiation exposed to the spacecraft over the entire mission would be the accumulated GCR dose of 30 REM and the SPE scaled value of 15 REM for a total of 45 REM, just below the mission 48 REM limit. If it is assumed that approximately 3 REM of GCR radiation is absorbed in the 30 days surrounding the SPE, this scenario would experience 18 REM, satisfying the 25 REM monthly limit. However, if the SPE were to occur at 60 days into the EME mission, the scaled effect of SPE exposure alone of 40 REM would violate the monthly guideline, and the addition of the 30 REM GCR dose would yield a mission exposure of 70 REM, well above the 48 REM guideline for the mission. The continuous effects of GCR and the short-term

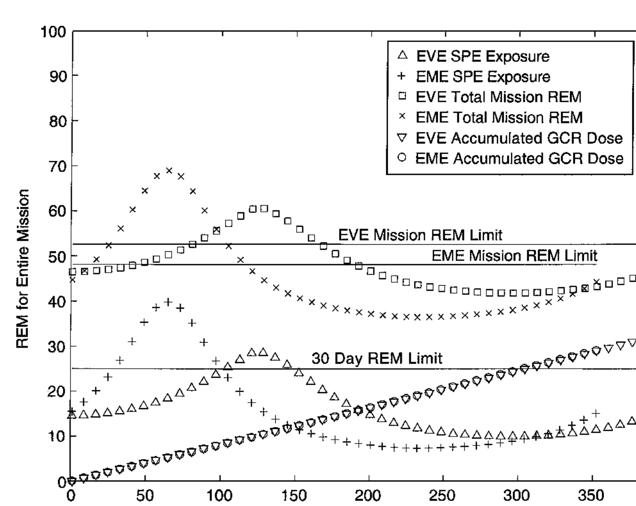

Fig. 3 EME 2006 and EVE 2010 mission radiation.

effects of random SPE must be accounted for simultaneously because the integrated effect of GCR over the mission duration adds an unavoidable offset to the mission radiation exposure that must be considered in tandem with the solar activity cycle and proximity to the sun for the potential of SPE exposure.

Both of the flyby missions considered here pass inside a radius of 1 AU at some point. The EME mission actually passes inside the orbital radius of Venus and has a higher SPE radiation risk (a SPE scale factor of 2.6 for EME vs 1.9 for EVE). Consequently, the EME trajectory has a greater potential than the EVE 2010 scenario for a violation of the mission exposure guidelines. Such a risk would require additional mass for emergency shielding that may prove prohibitive. If the solar activity prediction in Fig. 2 is considering, the 2010 EVE mission would encounter Venus at the very beginning of the maximum activity buildup for solar period 24 and the 2006 EME mission would encounter Mars during an interval of decreased activity. The risk of increased likelihood of an SPE must be weighed in terms of the TransHab distance from the sun and the corresponding scaled exposure magnitude when comparing these two missions. Therefore, the increased risk of SPE occurrence in the EVE mission due to its timing with the beginning of greater solar activity is offset by the small violation (7 REM for the mission limit) of the exposure guidelines that would occur if the large 1972 SPE impacted with the trajectory near EVE mission day 115. One factor that cannot be assessed is the directionality of the SPE, which may or may not intersect a TransHab vehicle following either the EVE or EME trajectory.

Conclusions

An approximate radiation model for the continuous effects of GCR and the finite duration effects of random SPE occurrences has been presented. Ambient solar radiation is not considered because its effect is generally less than that of the GCR, which actually decreases with a small linear slope as a trajectory approaches the sun. A radiation exposure analysis for the EVE 2010 and EME 2006 trajectories was conducted to estimate the relative risks involved with each mission. The radiation model assumed a uniform 20-cm thickness water shield and included the effects of a SPE on the order of magnitude of the large 1972 SPE should such an event occur at different times throughout each mission. The EME trajectory passes closer to the sun during a period of lower solar activity, indicating a greater hazard if a less likely SPE were to occur. The EVE trajectory passes only within a Venus orbital radius of the sun during a time of increased solar activity, indicating less of a hazard should a more likely SPE occur. Given the relative radiation hazards presented here, the EVE 2010 mission is superior to the EME 2006 mission in terms of radiation exposure due to mission duration and proximity to the sun.

Acknowledgments

This work was supported in part by a Graduate Research Fellowship Grant from the National Science Foundation. Special thanks to Bedford Cockrell at NASA Johnson Space Center for suggesting the TransHab problem to us and for his helpful discussions.

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Payload Deployment by Reusable Launch Vehicle Using Tether

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Introduction

THE advent of communication satellites in polar orbits of 600–1200-km altitude and the development of the international space station (ISS) at 400-km altitude and 51.6-deg inclination have created an urgent need for low-cost access to such lower Earth orbits. The space tether can be a viable alternative to provide an answer to this need. The momentum transfer is one of the several important tether applications earlier proposed and analyzed.[1] Colombo et al.,[2] Bekey,[3] and Bekey and Penzo[4] have shown that tethered systems provide significant impulse saving for transfer missions from circular orbits. Kyroudis and Conway[5] have stated the advantage of an elliptically orbiting tethered dumbbell system for satellite transfer to geosynchronous altitude. The effects of various tether deployment schemes as well as the out-of-plane libration on payload orbit raising are studied by Kumar et al.[6]

Recently, Bekey[7] has shown the advantage of payload deployment by reusable launch vehicle (RLV) with tether compared to the RLV with upper stages. However, the system equations of motion have not been considered in his study and therefore the system parameters obtained by Bekey so as to deploy the payload at specified low-Earth circular orbit may differ from the results of the actual simulation study. The present investigation is undertaken with a view to circumvent these limitations as well as to carry out the detailed analysis.

Equations of Motion

The governing nonlinear, coupled ordinary differential equations of motion[8] of the system (Fig. 1) obtained by applying the Lagrangian formulation approach are as follows:

$$\ddot{R} = \dot{\theta}^2 R - \frac{\mu}{R^2} + \frac{3}{2}\frac{\mu}{R^2}\frac{M_e}{M}\frac{L}{R^2}(1 - 3\cos^2\beta\cos^2\eta)$$

$$\ddot{\theta} = -2\frac{\dot{R}}{R}\dot{\theta} - \frac{3}{2}\frac{\mu}{R^3}\frac{M_e}{M}\frac{L}{R^2}\sin 2\beta\cos^2\eta$$

$$\ddot{\beta} = 2\frac{\dot{R}}{R}\dot{\theta} - (\dot{\theta} + \dot{\beta})\left[(2 + \mu_c)\frac{\dot{L}}{L} - 2\dot{\eta}\tan\eta\right] - \frac{3}{2}\frac{\mu}{R^3}\left(\sin 2\beta - \frac{M_e}{M}\frac{L}{R^2}\sin 2\beta\cos^2\eta\right)$$

$$\ddot{\eta} = -(2 + \mu_c)\frac{\dot{L}}{L}\dot{\eta} - \frac{1}{2}(\dot{\theta} + \dot{\beta})^2\sin 2\eta - \frac{3}{2}\frac{\mu}{R^3}\cos^2\beta\sin 2\eta$$

$$\begin{aligned}\ddot{L} = {} & \left[(\dot{\theta} + \dot{\beta})^2\cos^2\eta + \dot{\eta}^2 - \frac{\mu}{R^3}(1 - 3\cos^2\beta\cos^2\eta)\right]L \\ & - \mu_c\frac{L}{2}\left[3\frac{\dot{L}^2}{L^2} + (\dot{\theta} + \dot{\beta})^2\cos^2\eta + \dot{\eta}^2 - \frac{\mu}{R^3}(1 - 3\cos^2\beta\cos^2\eta)\right] - \frac{\mathrm{EA}}{M_e}\varepsilon_t U(\varepsilon_t) \\ & - 2\zeta\frac{\mathrm{EA}}{M_e L_0}(\dot{L} - \dot{L}_0)\end{aligned} \tag{1}$$

where $\mu_c = (L/M_e)(\mathrm{d}M_e/\mathrm{d}L) = (1/M_e)(m_t/3)[(m_2 + m_t/2) - 2(m_1 + m_t/2)]$; ε_t = tether strain, $L/L_0 - 1$; $U(\varepsilon_t) = 1$ if $\varepsilon_t \geq 0$, 0

Received 28 October 1999; revision received 19 September 2000; accepted for publication 12 October 2000.

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