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Space Vehicle Propulsion Systems: Environmental Hazards of Low Earth Orbit–Lunar Domain

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The attempt is made to describe various space hazards in the geolunar environment, along with examples of various degradation effects that can arise as a consequence of long-term mission exposure. Although the problem of man-made low-Earth-orbital debris is of immediate urgency for the proposed manned space station, it is not treated here.

Nomenclature

E, F_1, F_2 = three different regions in the ionosphere
F = ratio of cometary to sporadic meteoroid flux

Introduction

THE methodology used for this investigation involved an examination, enumeration, and assessment of the open literature for detailed descriptions of the hazards to be discussed. Given this body of information, the potential of these hazards for degradation, disruption, or possible termination of performance of LOX–LH_2 as well as other rocket engines, was considered. The mode of study was that of a selective literature search, in which candidate documents were selected on a basis of being current, as well as giving adequate quantitative detail. The hazards were then classified as to pervasiveness, abruptness, the capability of producing long-term hardware degradation and deterioration. They were then considered in relation to their specific influence in various mission modes in geolunar space.

The overview of results and conclusions that follow came from approximately 350 citations and 200 extended documents. In view of time constraints, these numbers do not fully represent the total wealth of applicable and potentially useful literature on this subject. Table 1 summarizes the objective and findings of this study.

We now consider the relevant space hazards and locales. For illustrative purposes, Table 2 enumerates these hazards in conjunction with the regions in geolunar space and their relative importance. Given the four dominant hazards—radiation, meteoroids, monatomic oxygen, and thermal gradient and thermal shock—Table 3 displays the space-hazard descriptions and anticipated deleterious effects.

With the condensed information contained in Tables 1–3, we now present a detailed discussion of these hazards.

Radiation

In evaluating the various aspects of the radiation hazard in geolunar space, one must appreciate that there is more than one source and modality associated with this hazard. The source can be broadly classified as cosmic (extra- and intragalactic), solar, and geospheric (due to the Earth itself). The basic modalities are electromagnetic (ionizing and nonionizing), and particulate, which is for the most part ionizing. Nonionizing electromagnetic radiation is classified as thermal and will be discussed in the section under thermal gradient and shock.

For sources and modalities closest to the Earth, a significant source of radiation hazard is the presence of ionized particles produced by the trapping of free solar electrons, protons, and neutrons by the Earth's magnetic field. In consideration of Fig. 1, it is evident that regions of elevated electron[1] density envelop low Earth orbit (LEO) and geosynchronous Earth orbit (GEO). The region of elevated electron density extends virtually to low lunar orbit. It consists of three overlapping broad layers of electron clouds, denoted as the E, F_1, and F_2 layers. Detailed explanation of these classifications is beyond the scope of this paper, and the reader should consult Ref. 1.

Apart from electrons, positively charged particles (protons) are also trapped by the Earth's magnetic field. These particles produce layered altitude distributions of their own, and possess number-flux distributions differing from those of electrons (Fig. 2).

Profiles for these various charged particle distributions are modified by long- as well as short-term solar cycle variations (Fig. 3). Below GEO, symmetric proton fluxes exist and near GEO have been found to vary by a factor of 4 from night to day, whereas closer to the Earth they appear to be insensitive to local-time effects at energies

Table 1 Study of space environmental effects

Study objective
- To evaluate potential space hazards which may be detrimental to the propulsion systems of vehicles that reside in geolunar orbit

Findings
- A number of potential space hazards exist that could affect the space vehicle
  1) Radiation: Hard ionizing electromagnetic radiation (gamma rays); charged particle plasmas (solar wind)
  2) Meteoroids: Particulate masses up to about 1 g
  3) Monatomic oxygen
  4) Thermal gradients and shock
- Detrimental effects of space hazards can largely be avoided by prudent practice in space vehicle and propulsion system design
- Long-term space environmental effects on propulsion system materials require further consideration.

Table 2 Potential space hazards vs locale

SPACE LOCALE	POTENTIAL SPACE HAZARD				
	Radiation (Gamma Rays)	Radiation (Solar Wind)	Meteoroids	Monatomic Oxygen	Thermal Gradients
LEO	◑	◑	◑	●	●
GEO	●	●	◑	○	●
LEO to GEO Transfers	●	◑	◑	◑	◑
Low Lunar Orbit (LLO)	●	◑	●	○	●
LLO to Lunar Surface Transfers	●	◑	●	○	◑
Lunar Surface	●	◑	●	○	◑

KEY: ● Strong Effect ◑ Weak Effect ○ No Effect

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Table 3 Space hazards and potential effects

Space hazard	Description	Potential effects
Radiation (gamma rays)	Extremely hard, ionizing electromagnetic radiation	Degradation of nonmetallic materials
Radiation (solar wind)	Charged particle plasmas (ionizing x-rays, non-ionizing uv radiation, electrons, protons, and neutrons)	Spacecraft system charging affecting electronics, or resulting in abrupt static discharge
Meteoroids	Cometary and asteroidal masses ranging from dust to particulate masses up to 1 g	Surface erosion from very small particles (dust) Structural deformation or compromise from larger particles
Monatomic oxygen	Cometary and asteroidal masses ranging from dust to particulate masses up to 1 g	Deleterious oxidation—reduction of atomic oxygen with metallic propulsion system components
Thermal gradients and shock	Sunrise–sunset thermal disturbances	Catalysis of spacecraft charging enhance corrosive effects of monatomic oxygen Impose mechanical stresses on exterior structure

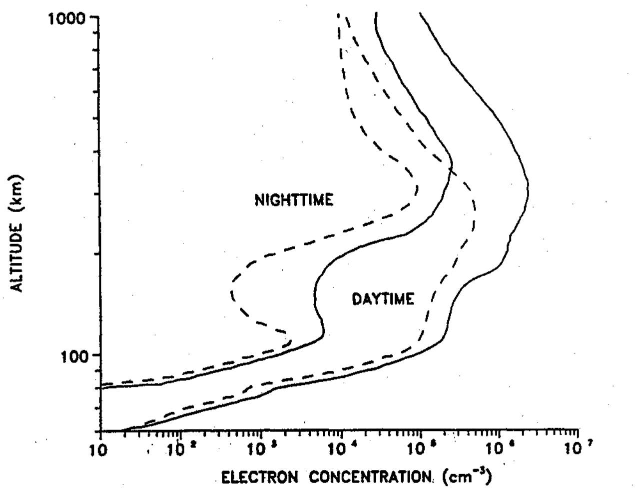

Fig. 1 Typical midlatitude daytime and nighttime electron density profiles for sunspot maximum (solid lines) and minimum (dashed lines) (after Tascione[1]).

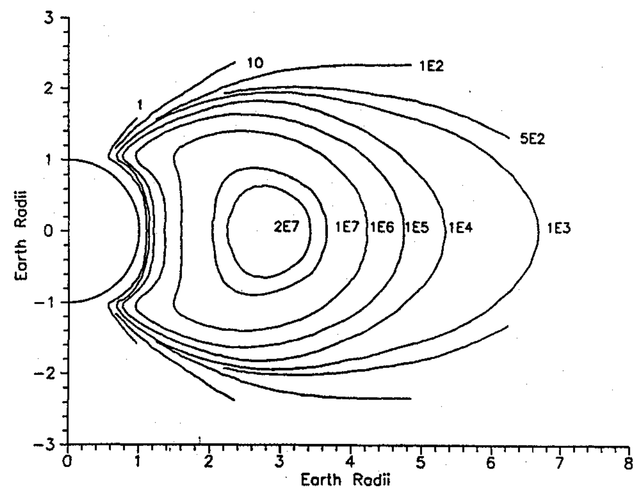

Fig. 2a Distribution of trapped protons with energies greater than 1 MeV (after Tascione[1]).

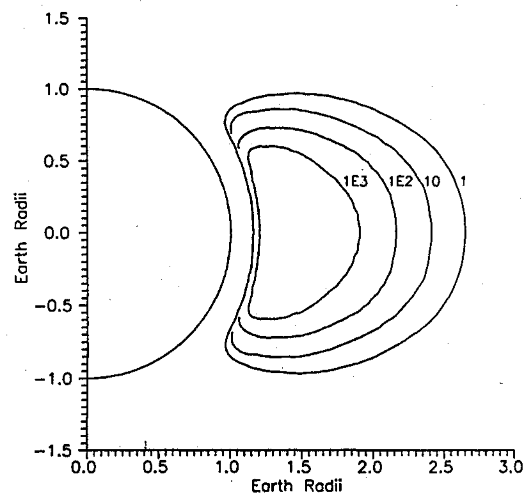

Fig. 2b Distribution of trapped protons with energies greater than 100 MeV (after Tascione[1]).

in excess of 25 MeV. It has been observed, however, that at approximately 3 Earth radii, protons with a 1-MeV peak experience sizable disturbances.

Augmenting the trapping of charged particles in the Earth's magnetic field is the cosmic galactic and intragalactic radiation. In this mechanism primary cosmic rays (i.e., neutrons, protons, and other totally ionized nuclei of energies greater than 1 GeV) are incident on the atomic nuclei composing the outer fringes of the Earth's atmosphere. This bombardment of the outer atmosphere gives rise to a torrential sequence of nuclear events (reactions). A principal product of these events is a so-called neutron albedo flux (reflection flux).

For neutrons associated with the albedo flux, the residence time in the upper reaches of the atmosphere significantly exceeds the half-life of a typical neutron. This provides an adequate time period for the neutrons to decay into H^+ (protons) and e^- (electrons) (one each for each neutron), which also become trapped by the Earth's magnetic field, further contributing to the charged particulate bulk of the Earth's radiation belts. This process is believed to be an important source of protons whose energies are in excess of 50 MeV.

Besides environmental hazards for a crew on a protracted mission, the previously described phenomena cause what are referred to as variation of space drag profile and spacecraft system charging. The first effect—unexpected changes in spacecraft drag profiles—creates uncertainties in the coded signal sequences whose purpose is the maintenance of orbital tracking and shape determination. The second effect, spacecraft system charging, directly threatens the integrity of the spacecraft and its systems. This hazardous condition, a direct result of the radiation environment previously described, is the focal point of this discussion.

Spacecraft system charging has become a matter of much greater concern due to modern low-voltage integrated-circuit and chip technology now incorporated into propulsion, navigational, control, and communication systems. This concern is not one of supposition or hypothesis; it was borne out by an anecdotal experience with the Pioneer spacecraft. Its systems were all but terminated by the effects of charged particles (H^+ and e^-) in the Leviathan charge belts of the planet Jupiter. This direct experience with space charging effects is not limited to Jovian fly-by missions, but was clearly demonstrated

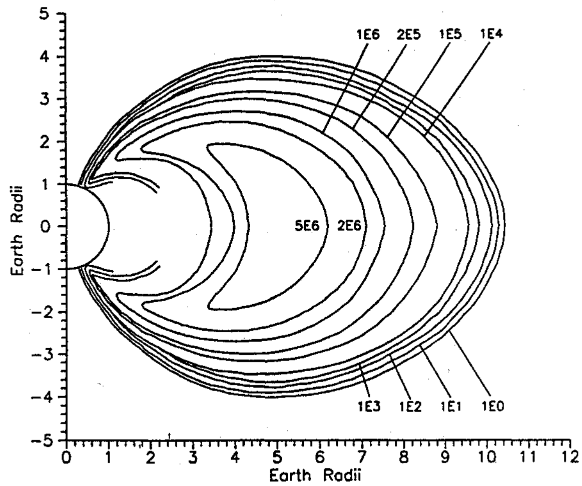

Fig. 3a Distribution of trapped electrons during solar minimum with energy greater than 0.5 MeV (after Tascione[1]).

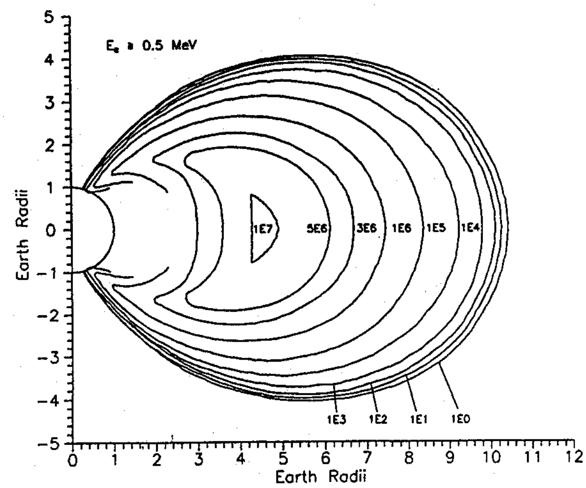

Fig. 3b Distribution of trapped electrons with same energy during solar maximum (after Tascione[1]).

by the Earth-orbiting satellite ATS-6. This spacecraft developed and recorded potential differences with respect to the surrounding medium as high as 20,000 V.

Such a buildup of large static charges can render important sensor systems inoperable. An even greater danger lies in abrupt differential discharge, aggravated, in some mission configurations, by differential external surface temperature gradients. These discharges can produce a myriad of degrading events, including structural damage and damage to components vital to the propulsion system. Abrupt differential discharges involving the peripheral structure of spacecraft have also been observed to be responsible for various system malfunctions.

A curious aspect of spacecraft charging in the geolunar domain is that it is most likely to occur in GEO. As an explanation, the suggestion has been made that this may result from our mission modes for data acquisition tasks concerning the Earth, inasmuch as we have a number of data acquisition satellites in GEO. The geometric shape of the Earth's magnetosheath shows that mission orbits of altitude greater than 4 Earth radii should experience more incidents of the space charging effect because of the position and orientation of the sun with respect to Earth and because of charged-particle streaming. In regions at this extreme altitude, the magnetosheath of the Earth forms a tail analogous to that of a comet. Approximate positioning of a typical GEO is shown with this tail overlaid in Fig. 4.

Efforts to deal with spacecraft system electrification are well evidenced in the literature. One such document[2] describes a lengthy protocol that details in depth various methods for avoiding spacecraft system charging. As an example, it describes the NASA charging analyzer program. This computer program analyzes proposed

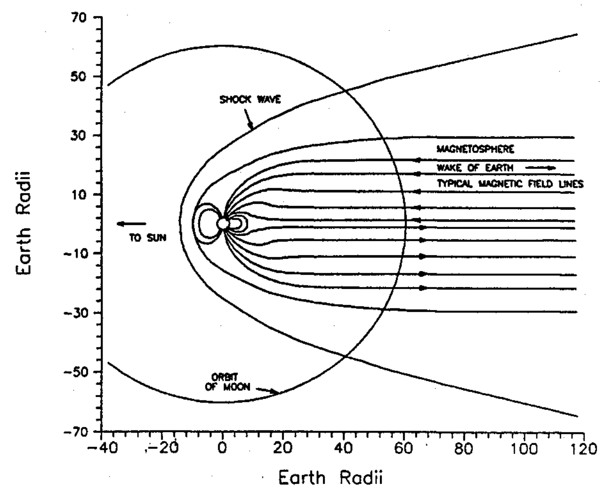

Fig. 4 Cross section of the magnetosphere showing the approximate size in terms of number of Earth radii (Earth radius R = 6370 km). Geostationary orbits are at 6.6 Earth radii (after Tascione[1]).

spacecraft designs, and then gives specific recommendations for optimal procedures for spacecraft hazard reduction.

Another document,[3] of a more general nature, concludes with specific recommendations for passive and active strategies for the reduction of this hazard:

> The methods of reducing space electrification and protecting the onboard systems against the effects of static electricity can be divided into passive and active. Passive methods are currently the principal ones used aboard geostationary satellites and other spacecraft. They include special methods of designing the spacecraft so that the units and sub-units have good electrical contact with the metal hull of the spacecraft. Special attention is paid to the reliability of the electrical connection of the shielding jackets of the numerous electrical cables to the hull, etc. The spacecraft design should have the minimum possible number of apertures, cracks, sharp projections, to lower the likelihood of formation of greatly different electric fields.

Therefore, fiber-optic communication systems within a spacecraft have an excellent prospect for enhancing resistance of the electronic modules of a spacecraft to electromagnetic noise.

Also addressed were design details and suggestions, including mention of a specific case of satellite system malfunction (the West European satellite Marex 1), which was allegedly due to insufficient attention to the question of spacecraft charging during the design phase.

Charge–discharge effects are further enhanced by cosmic rays. Charged particles in this extraordinary energy range can easily penetrate a spacecraft, and in doing so, alter on-board computer memory, thereby evoking spurious commands.

To further investigate the phenomena, a low-voltage solid-state-instrumented vehicle (Scatha—spacecraft charging at high altitude) has been flown to test various conjectures and hypotheses concerning the effects of the spacecraft charging situation. It has been discovered that photoemission due to charged plasma bombardment (solar wind) plays a significant role in the effect. In addition, direct cosmic-ray insult has been documented by Cunningham.[1]

In that there are limitations on on-board shielding due to launch weight constraints, great attention must be paid to orbital mission trajectory shapes in order to minimize the residence time in an environmentally hostile locale. Consistent interpretations of all observations on this phenomenon are still in the process of being formulated.

We now turn our attention to another space hazard, that of cometary and asteroidal meteoroids.

Meteoroid and Micrometeoroid Hazard

The two principal sources of meteoroids are 1) debris left by comets in their paths about the sun and 2) debris from the asteroid belt lying in an orbital band between Mars and Jupiter. The objects of class 1 have a definite annual pattern (are seasonal), whereas those in class 2 make random appearances in space and time (they are said to be sporadic).

Of these two extraterrestrial debris populations entering the near-Earth–lunar environment, as best we can tell, 90% are of cometary origin, and the remaining 10% are of asteroidal origin. In assessing the risk of direct collision in this environment, the accepted opinion is that impaction of a mission-oriented spacecraft system by an object of large size and velocity is of such low likelihood that it can be ignored. The remaining objects that pose a threat are divided into two populations: those of stream and sporadic meteoroids, with the environment being pictured as a superposition of a constant stream on the sporadic population. Eighteen distinct stream meteor showers per year have been discovered and documented in the literature.[4]

Investigators in this field of astronomical research have developed a specialized parameter, the so-called activity factor, which is a measure of annual sporadic flux change. By definition it is the ratio of the cometary flux component to the sporadic.[4] As a consequence of the relative predictability of the average cometary flux, coupled with the smooth and predictable change in the activity factor F, the maxima of F are avoidable, particularly for planetary missions, by judicious choice of launch parameters. Such adjustments are possible because the mean positions of the intersections of the orbital planes of these objects with the Earth's orbit are at fixed angular directions with respect to the ecliptic.

This passive measure leaves, for practical purposes, only the sporadic component to be dealt with. In quantifying the sporadic flux, objects are classified with respect to size and density, as well as a joint distribution of particle flux and velocity. These parameters are expressed in the form of empirical probability functions for practical risk assessment.

It is clear that for the refinement of such empirical relations, experimental data-seeking spacecraft—of which LDEF (long-duration exposure facility) is a typical example—must be launched, retrieved, and evaluated. In virtue of such activities detailed study of meteoroid damage risk can be undertaken, by classification in terms of the 1) amount, 2) physical pattern, and 3) various mechanisms of damage, with the objective of developing strategies of damage risk minimization.[5]

One mode of meteoroid damage that must not be overlooked is that which can be incurred on lunar descent and ascent, and in low-lunar-orbit missions. Furthermore, one has an additional potential for damage in virtue of lunar ejecta, which comprise secondary particles ejected from the lunar surface due to continual meteoroid bombardment. This environment can be placed in the same empirical form as that for the more direct fluxes discussed previously. The explicit mathematical expressions for the various meteoroid fluxes are given in detail in Ref. 4.

For specific applications of the aforementioned expressions, the flux on the lunar surface can be expressed in so-called unperturbed form. Perturbation in the flux expressions[4] arises from 1) gravitational focusing by the Earth and 2) shielding due to eclipse by the moon or Earth, or possible self-shadowing. The fluxes for the stream and sporadic environment must, for an individual mission, be corrected by appropriate defocusing and shielding factors before introduction into the spacecraft design process. Two exceptions to this apply 1) in the case of the flux associated with a specific stream and 2) in the case of eclipse relative to a specific stream; in the latter case the flux is set equal to zero.

Active Measures for Meteoroid Damage Management

Although it is not possible to address this topic extensively here, it can be discussed in outline form.

Generally, the topic can be divided into spacecraft response to impaction, and the types of resulting damage.[5] The factors governing vehicle response are 1) material composition, 2) temperature, 3) severity of stress incurred, 4) thickness, and 5) number of plates constituting the structure and type of fabrication. For a comprehensive discussion and study of meteoroid damage one first classifies damage by its modes, and then by its specific description. Damage modes for practical purposes fall into six categories: 1) catastrophic rupture, 2) leakage, 3) deflagration, 4) vapor flash, 5) reduction of structural strength, and 6) erosion. Given one of the above modes of damage, it is customary to describe from a lexicon of descriptors the damage that has occurred. Damage descriptors in common use are 1) partial penetration and/or surface damage, 2) perforation, 3) local deformation, spall fracture, or reattached spall (back surface), 4) secondary fracture, and 5) catastrophic rupture.

Employing all of the above, damage is evaluated by analytical methods and empirical criteria in conjunction with physical testing.[5] Recently developed methods take advantage of newly developed materials and then, via sophisticated optimization methods, give proper design tradeoffs leading to damage risk minimization.

All the above criteria have been objectively quantified in a recent work.[6] The study incorporates a space-debris environmental model into an overall optimization methodology. This is accomplished by utilizing various standard engineering models for the evaluation of protective structural design requirements for hypervelocity impaction.

The results of this extensive study[6] indicate that consideration of the space platform structure configuration and materials used can partially offset the risks due to dramatic increases of man-made debris in LEO. Clearly, this active method of damage management is applicable to missions in high Earth orbit and lunar missions as well.

We now consider another hazard, which is peculiar to LEO: that of monatomic, or nascent, oxygen. For this hazard, like the meteoroid hazard, there is promise of reduction by new material development and fabrication techniques.

Monatomic Oxygen

In LEO, it has been discovered that indigenous monatomic oxygen is highly reactive with the outer hull of a spacecraft. Such oxidation–reduction chemical reactions can damage structures that are vital to the internal functions of the spacecraft system.

The source of monatomic oxygen in LEO is photodissociation of O_2 and H_2O vapor at the edge of the Earth's atmosphere. Through carefully planned exploratory missions, one can accumulate the data and experience required to better use existing materials for hull fabrication, as well as initiate development of new, resistant materials. Thus, one can in effect set up an interactive data acquisition and material development loop, the results of which should converge on optimal engineering design tradeoffs for proposed missions. Such a process has five distinct phases: 1) fabrication of ground-based mono-oxygen sources, 2) fabrication of ground-based simulated LEO and other environments, 3) investigation of various deleterious reaction mechanisms for selection of suitable candidate materials already available, 4) utilization of findings in phase 3 for determination of reaction cross sections for use in synthesis of more acceptable materials, and 5) design development, and calibration of flight hardware so that LEO and other experimental mission results can be compared with results obtained via ground-based simulated environments. This group of process elements has been abstracted from a detailed study of Refs. 7–11.

An extremely important preliminary result[11] from land-based testing indicates that most hydrocarbons and active metals are highly reactive, whereas materials containing silicones, fluorides, oxides, and noble metals are largely inert in the presence of a simulated LEO monatomic oxygen environment. Anecdotal results from exposure of Kapton[11] to a simulated monatomic oxygen environment indicate mass loss amounting to one in ten atoms as a result of the initiated chemical reactions. It should be emphasized that such erosion is intensified by the presence of nonionizing and ionizing radiation as well as meteoric dust bombardment.

Thermal Gradient and Shock

The final space hazard we now consider is that of thermal gradients and shock. This hazard is of importance in the geolunar domain in virtue of light–shadow and shadow–light traversals of a spacecraft as a consequence of an eclipse of the sun by the Earth or moon. In addition, through changes of attitude of a system with an irregularly shaped geometry, self-shadowing can occur.

This hazard has more than one effect: it is a catalyst for 1) spacecraft charging in geosynchronous orbit, 2) enhancement of corrosive effects of monatomic oxygen in LEO, and 3) mechanical stressing of exterior structures, particularly solar panels of alar design.

In a recent paper[12] an informative study of the results of thermal gradients and thermal shock are given. This document focuses on an in-depth analysis of data gleaned from the TOPEX satellite. These analyses are specifically directed to the impulsive torques initiated on the large wing solar array when the vehicle exists the Earth's shadow. In addition, a companion torque initiated by rapid cooling of the array on entry into the Earth's shadow is studied. Particular attention was paid to 1) roll–yaw perturbation, 2) thermodynamic and mechanical analysis of sunrise and sunset disturbances, and 3) the effects of these factors on the vehicle.

At this point it is clear, particularly in LEO, that the hazards just discussed will act in unison. In fact, there is every reason to believe that the superposition of these degrading influences will act in a nonlinear, synergistic, manner. Superposition of these various effects will pose a significant challenge to even the most sophisticated damage risk minimization techniques.

LEO Debris

Although the intent of this work was the study of naturally occurring space hazards, it is necessary to make some mention of the manmade LEO debris situation. In examination of the debris–meteoroid hazard that presents itself, particularly to a long-term space station mission, one finds that the natural meteoroid flux is virtually (but not altogether) ignorable when one considers prompt impact damage in LEO. What follows here is a brief overview of what the authors have been able to glean from recent reports on the subject.

In the past the basic philosophy of minimizing spacecraft meteoroid and debris collision incidents[13] has been as follows: consider a proposed mission in the light of an accepted model of manmade debris and meteoroids, then use the model at some statistical confidence interval for collision events, and finally add appropriate shielding and/or equipment redundancy, depending on whether the mission is manned or unmanned. On the basis of several anecdotal findings it has been concluded that for a protracted mission in LEO, viz., a space station, the orbital debris environment must be reformulated for meaningful implementation by vehicle designers. Although the natural meteoroid flux is fairly stable, the standard artificial debris model up to circa 1985 will grossly underpredict the risks of operating a manned space station in the present-day LEO environment. At the time of this writing, work on the reformulation of the artificial-debris environment is underway, along with rethinking methods of assessing crew and vehicle risk. As a consequence of the present state of affairs, this topic cannot be adequately treated in this paper.

Summary

The goal of this study has been to describe various space hazards present in the geolunar environment. In so doing there has been a description of various deleterious effects that can occur to missions in the geolunar region.

The major strategy in dealing with these conditions can be divided into distinct parts: 1) the synthesis of new materials and 2) the implementation of advanced minimax techniques in spacecraft fabrication. The reduction of risk from radiation hazards can be further achieved by the exploration of new onboard circuit and signal technologies and architectures, in particular those based on fiber-optic transmission.

Concluding, it should be reemphasized that new analytical tools must be implemented for the best risk-reduction design tradeoffs possible.

Acknowledgments

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