Space Environments and Survivability

3

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Just as spacecraft design teams are increasingly approaching the design and construction of a spacecraft as an integrated system, the overall environment and survivability of the spacecraft should be approached in a similar fashion. Typically perceived as either too expensive or design limiting, design for environmental survivability, whether it be from thermal, radiation, atomic oxygen, or spacecraft charging effects, is usually done strictly on an *ad hoc* basis. Unfortunately, 'faster, better, cheaper' (FBC) missions seldom consider anything much beyond thermal effects and, independently, radiation effects on selected parts. The basic requirements however, to significantly reduce the weight/size of a FBC mission and to make use of the latest commercial, off-the-shelf devices [with their often significantly lower radiation and Single Event Effects (SEE) tolerances] mandate that much greater thought be given to multiple uses of the spacecraft design to fulfill multiple environmental survivability functions. The objective of this chapter, after providing an introduction and overview of the space environment and its effects, is to detail the steps required for a systematic approach to space environment survivability that can be achieved with the least impact on the overall design process.

Fortunately, the concepts required to carry out a systems approach to environmental survivability currently exist. For example, both the Galileo Jupiter and the Cassini Saturn missions expended considerable effort in developing the methods necessary to design a thermal protection system that both provided meteoroid protection and limited spacecraft charging effects. In the case of Galileo, extensive effort was spent in developing an integrated radiation-resistance design for the Star Scanner—the designers picked a radiation resistant photomultiplier, substituted mirrors for lenses where possible, and carefully placed additional shielding to provide robust protection (upwards of 10 g/cm^2), to maximize the survivability of this system during Galileo's passage through Jupiter's inner radiation belts. Spot shielding, Error Detection and Correction (EDAC) software, hardening of selected components, Faraday cage shielding of the cabling, and similar techniques were all combined to provide ultra-reliable protection for the Galileo and Cassini systems. In the case of Cassini, attention was also paid to the way the vehicle was oriented in flight so as to limit meteoroid impacts. To a degree, radiation fluxes are also 'oriented'—a factor that can be used to limit impacts on sensitive surfaces. On a case-by-case basis, good tools exist for providing specific types of environmental protection and that, in some instances, allow combining techniques. A well thought out survivable design considers all these components simultaneously.

The next generation of 'microsats', 'cubesats', or 'sciencecraft' will implicitly require a systematic approach to environmental protection if they are to realize meaningful levels of reliability within the size, mass, and power constraints of these concepts. The placement of parts, the selection of environmentally robust software (i.e., EDAC for SEEs), intelligent 'on–off' control of sensitive systems when the spacecraft is in a hazardous environment (many components are 'harder' when turned off), use of intrinsically hard circuit designs as opposed to softer circuit designs, redundancy, utilization of graceful degradation, multiple use of shielding (for thermal, radiation, spacecraft charging, atomic oxygen protection, etc.) are a few of the procedures to be considered. One example brings the point home: on the US Department of Defense Clementine spacecraft, officially called the Deep Space Program Science Experiment (DSPSE), the average shielding was $\sim$100 mils ($\sim$2.5 mm) of aluminum. This implied that the solid-state recorder would be sensitive to approximately 1,000 Single Event Upsets (SEU) per day background due to protons. Indeed Clementine experienced an observable solar proton event

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during its first month of operation. The Clementine solid-state recorder, however, did not see the event and averaged around only 70 SEUs per day over the mission. A careful review of the spacecraft design revealed that the majority of the solid-state recorder components were protected by at least 300 mils (7.6 mm) of shielding—not from the spacecraft but because the boards were closely packed inside their boxes and provided a significant amount of self-shielding. Designers, particularly in the early stages of a mission often fail to take account of such ‘intrinsic shielding’, leading to an erroneous concern for radiation effects.

A proper systematic design approach to environmental survivability requires: (1) a review of the primary environments and interaction(s) of concern and (2) a listing of the general design options for each concern. These options should be cross-correlated with the specific interactions to identify design options common to the different effects. The design is then iterated with changes in the design reflected in quantifiable metrics for each effect—for example, changing a thermal blanket design may change the meteoroid protection and may alter the radiation shielding and spacecraft mass. Changing the position of a star scanner might enhance its radiation protection or alter its thermal load. A systematic design approach needs to identify such ‘cross-correlations’. Ultimately, the goal of an analysis is to identify the minimum number of design procedures that can yield the maximum benefit for several different environmental effects.

The steps taken to limit a particular environment and its effects are typically well understood—Galileo is an example of how different protection methods can be played off against multiple effects. It is also clear that if mass and size are a premium and if environmentally ‘soft’ and advanced technology are synonymous, then the integrated approach is both necessary and a prerequisite if missions are to succeed in the future. Guidelines and methods for approaching the problem systematically are reviewed in this chapter. The objective is to provide an insight into initial integrated environmental survivability design and for establishing the reality of the potential benefits.

3.1 Procedure

The steps for identifying various integrated design trade-offs starts with the definition of the mission: its trajectory, instruments, and requirements. From these the relevant environments and interactions are defined. Based on the top-level interaction(s), the design trade-offs or options are then identified. These are then assessed in terms of relevant selection criteria (say, mass, cost, complexity, software impact, and so forth). The design trade space is optimized and a set of design solutions developed for project consideration. These steps are listed in Table 3.1.

Table 3.1 Integrated environmental design procedure

Step number	Step
One	Identify requirements based on trajectory, instruments, and unique mission constraints
Two	Rate the environments versus the interactions
Three	Identify the design trade-offs for the environments/ interactions of highest concern
Four	Establish mass, cost, complexity criteria metrics for trade-offs
Five	Optimize combinations of design choices
Six	Evaluate resulting designs

3.2 Environments

The space environment is far from benign in its effects on space systems. Given the growing complexity and consequent sensitivity of space systems, an understanding of the space environment and its interactions is the first step in mitigating these effects. Ten types of environment will be considered here. The first, the neutral atmosphere, is primarily responsible for drag, glow, and oxygen erosion. The next two environments, the magnetic and electric fields, are responsible for magnetic torques and induced electric fields. The UV/EUV radiation environment is not only responsible for the formation of planetary ionospheres but also for photoelectrons and long-term changes in material surface properties. The IR environment is a major driver of thermal effects. Four charged-particle environments are considered: the interplanetary environment, the plasmasphere/ionosphere (responsible for ram/wake effects and solar array arcing), the plasmasheet (the primary region for spacecraft charging) and its low altitude extension the auroral zone, and the radiation belts. Although primarily referenced to the Earth, these environments each have their direct corollaries for the other planets as well. Finally, the solid-particle environment (synthetic space debris (unique to the Earth), interplanetary meteoroids, and surface dust) will be discussed (cometary particle clouds and planetary rings must also be considered). The intent is not to provide a detailed description of each environment (which are planet/orbit specific) but rather to provide an overview of their chief characteristics as they apply to environmental interactions. These characteristics are needed in defining the spacecraft effects for the purpose of design trade studies.

3.2.1 Neutral Atmosphere

Typically, the major environment at low altitudes around the planets (except Mercury) and Titan is the ambient neutral atmosphere. Atmospheric drag and ablation are

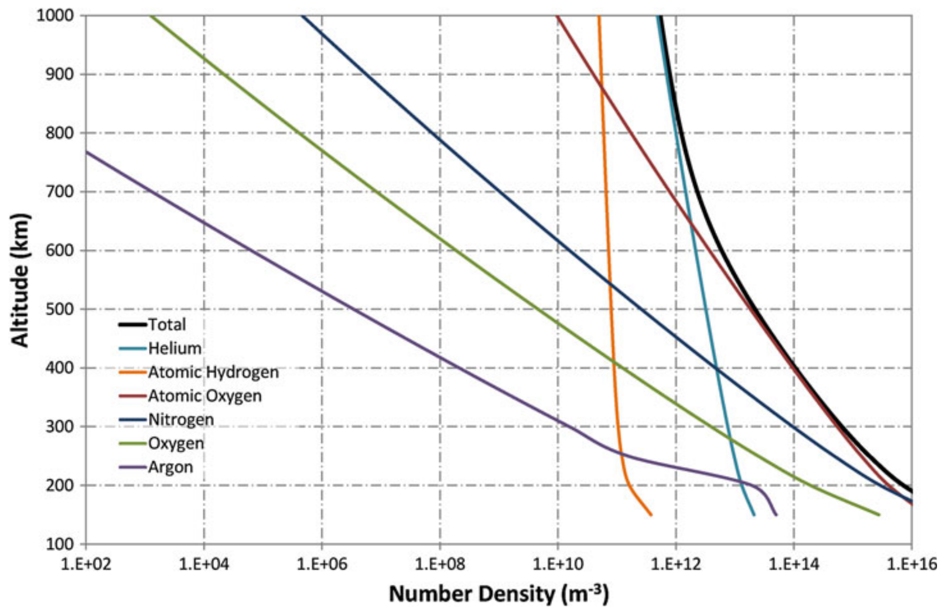

Fig. 3.1 Number density profiles for the Earth's atmosphere based on the US Standard Atmosphere, 1976 [2]

major concerns for this environment. In addition, typical orbital velocities relative to an atmosphere lead to impact energies of multiple eV's—high enough to induce chemical interactions such as oxygen erosion. Neutral particle densities for the Earth range from 10^{10} cm^{-3} at approximately 200 km altitude to 10^6 cm^{-3} or less at 1,000 km altitude, see Fig. 3.1. Atmospheric models that describe the density, composition, and temperature fall into three basic classes: static profiles, global analytic fits, and time-dependent simulations. Examples for the Earth are the US Standard Atmosphere profiles, the Jacchia and MSIS (mass spectrometer and incoherent scatter) analytic models, and various thermospheric global circulation models (TGCMs) [1]. For the Earth at least, there are a number of models. Similar types of models exist for the other planets and some of the moons. Static profiles in particular are readily available for most destinations, but aside from Mars, where several MSIS and TGCM models exist on-line, direct access to more complicated models for the other planets is typically limited. Static models for planets such as Venus and Mars are useful for reentry or atmospheric capture. The effects of the atmosphere on a spacecraft's trajectory, along with a more detailed discussion of atmospheric density models, can be found in the astrodynamics chapter (Chap. 4).

3.2.2 Electric and Magnetic Fields

Electric and magnetic fields exist around most bodies in space. Magnetic fields range from tenths of a gauss near the Earth and ~4–8 gauss at Jupiter's surface down to a few gammas (nanotesla) in the solar wind. Note that one tesla is equal to 10^4 gauss. Ambient or induced electric fields (e.g., $\mathbf{V} \times \mathbf{B}$, see later) range from 0.3 V/m close to the Earth to as much as 60 V/m near Jupiter. For comparison, spacecraft surface charging potentials can reach ~20 kV at the Earth. There are detailed models of the magnetic fields of Earth, Jupiter and Saturn, from which the induced electric fields can be derived [3]. Mars, the Moon and Venus do not have significant magnetic fields, although strong local magnetic field anomalies have been identified at the Moon, while Mercury's field is apparently about 1 % as strong as Earth's but remains subject to further investigation by both the MESSENGER (MErcury Surface, Space ENvironment, GEochemistry and Ranging) and BepiColombo space missions. There are first-order models of the magnetic fields of Neptune and Uranus but these will need to be better developed in the future. Although simple magnetospheric models exist for all the planets, except for the Earth (Fig. 3.2) these need to be made more quantitative to determine actual magnetopause and magnetosheath crossings for instruments. Finally, computer codes capable of tracing out the field lines from the magnetic field models are needed for the radiation belt models (the latter typically require so-called 'B and L' coordinates) and are readily available. Although currently little information exists, the magnetic fields of comets and perhaps asteroids will also need to be defined during early planning of missions to these bodies.

3.2.3 Ultraviolet Radiation

Ultraviolet (UV) and extreme ultraviolet (EUV or XUV) radiation is important for spacecraft interactions as it can change the surface chemistry of materials and causes

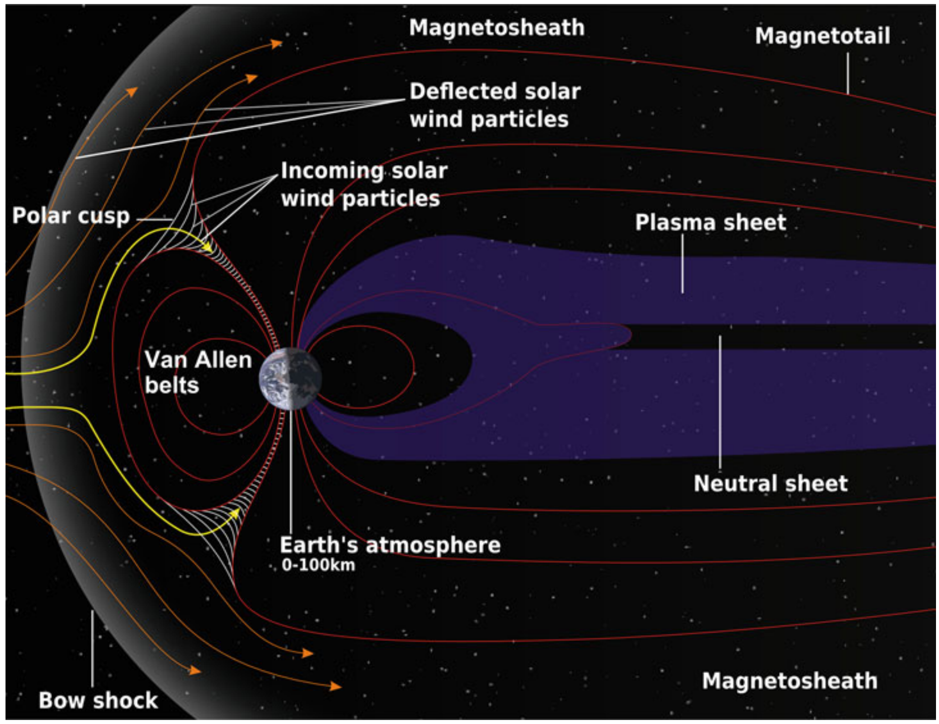

Fig. 3.2 Profile of the Earth's magnetosphere showing the magnetic field lines and structure. *Image* NASA

photoelectron emission. The UV/EUV radiation is the continuum and line spectrum between roughly 10 and 4,000 Å. The solar flux/energy in this spectral range is between 10^7 and 10^{10} photons/(cm^2 s) below 1,000 Å and rises exponentially to 10^{16} photons/(cm^2 s) between 1,000 and 10,000 Å. Note that the Lyman-alpha line at 1,216 Å plays a major role in photoelectron emission. The shortest wavelengths, from 10 to 100 Å, are called X-rays. The solar spectrum at 1 au[1] is illustrated in Fig. 3.3a [4], while Fig. 3.3b presents the ASTM E490-00a(2006) Standard Solar Constant and Zero Air Mass (AM0) Solar Spectral Irradiance, which has an integrated power of 1,366.1 W m^{-2}. The ASTM E490 standard does not cover the complete solar spectrum but does extend from a wavelength of 119.5 μm to 1 m [5]; an ISO standard is also available, see ISO-21348. Note that the solar spectrum can also be represented simplistically as a black-body of effective temperature 5,781 K, whilst the Earth radiates as a blackbody at 254 K. Models of the UV/EUV spectra and the atmospheric attenuation at the Earth and the planets of the UV/EUV are available if attenuation effects for sensors or photoemission are needed—models also exist for estimating spacecraft charging effects during eclipse passage.

3.2.4 Infrared

The infrared (IR) spectrum is between roughly wavelengths of 0.7 and 7 μm and is dominated by the Sun (Fig. 3.3). Other sources of IR are reflected sunlight, atmospheric glow, radiation from planets, and even light from auroral displays. The IR environment is a major source of thermal effects on spacecraft. As in the case of the UV/EUV region, detailed spectra are readily available.

3.2.5 Solar Wind Plasma

The solar wind is a neutral plasma, primarily consisting of electrons, protons, and alpha particles, which flows approximately radially from the Sun at velocities ranging from 400 to 2,500 km/s. Since the Sun rotates in just over 27 days, as the solar wind expands outward the plasma drags the Sun's magnetic field lines out in an Archimedean spiral in the solar equatorial plane (Fig. 3.4). Densities (mean energies) range from around 50 particles cm^{-3} ($\sim$40 eV for ions; $\sim$65 eV for electrons) near Mercury to 0.2 particles cm^{-3} (1 eV for ions; 10 eV for electrons) at Jupiter. Solar wind models are necessary for design purposes ranging from missions near the Sun to the outer solar system—a good example of such models is the NASA Marshall Space Flight Center (MSFC) L2-CPE statistical model [6]. Such models are used to estimate plasma interactions with spacecraft, large solar sails, comets, or asteroids, and for estimating effects on plasma sensors or

[1] An astronomical unit is a unit of length defined as exactly 1.495 978 70691(6) x 10^{11} m, approximatly the average Earth–Sun distance, and is accepted for use with the *Système international d'unités*. The abbreviation is not captialized as it is not named after a person; a.u. and ua are also used alongside the incorrect AU.

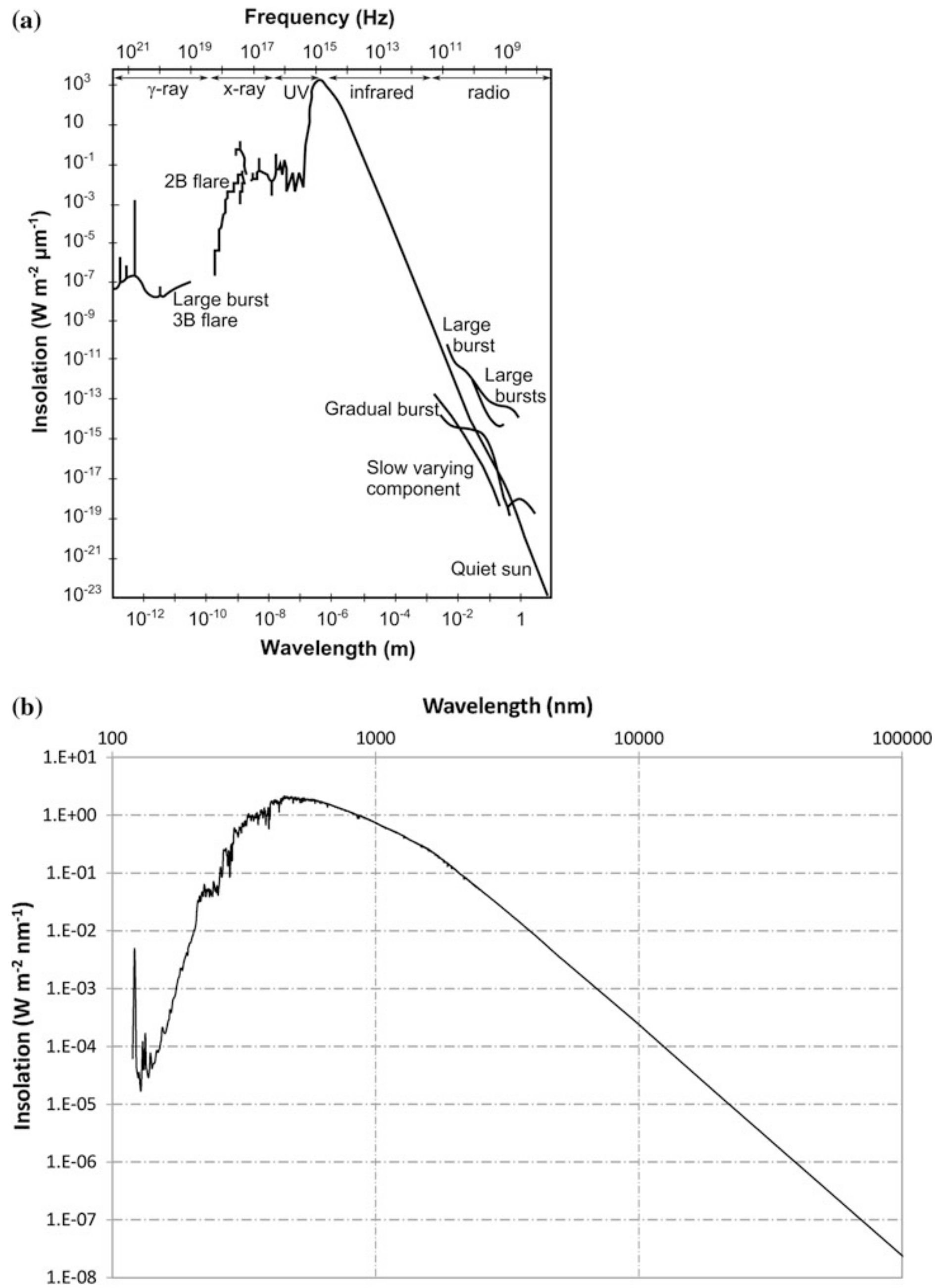

Fig. 3.3 **a** Electromagnetic flux at 1 au showing the frequency range from gamma (γ)-rays/X-rays through visible frequencies to IR and radio waves, reproduced from [4] and **b** the ASTM E490-00a(2006) Standard Zero Air Mass Solar Spectral Irradiance [5]

charging analyses. These plasmas induce spacecraft surface potentials of typically $\sim$10 V—the highest reported surface potential in the solar wind being $\sim$100 V. Plasma interaction models for estimating effects of the solar wind are available for general design purposes—examples are the Nascap-2 K [7] and various particle in a cell (PIC) codes.

3.2.6 Ionospheric Plasma

The ionized component of a planetary atmosphere, the ionosphere, is typically a comparatively dense, 'cold' plasma. For the Earth, the composition varies from an O^+ dominated environment between $\sim$200 and 500 km with a maximum density of about 10^6 cm^{-3}, to H^+ dominated above 1,000–1,200 km with densities from 10^5 cm^{-3} at 500 km, to 10^3 cm^{-3} or less above 2,000 km, see Fig. 3.5. All the planets (and most large moons) have ionospheres with compositions characteristic of their neutral atmospheres. In addition to spacecraft surface charging (typically of little concern compared to auroral-induced charging), ionospheres affect radio wave propagation and are important for their effects on spacecraft communications. Simple static profiles currently exist for all the planets and Titan.

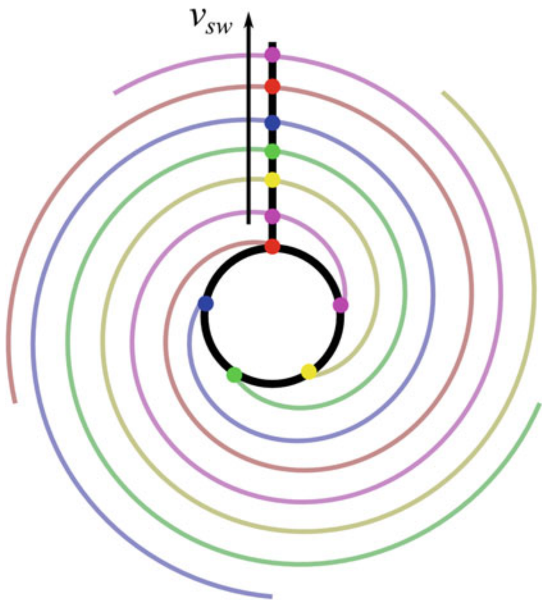

Fig. 3.4 View of the solar wind magnetic field lines showing how they are dragged out in an Archimedean spiral in the solar ecliptic plane as the Sun rotates. v_{sw} is the radial solar wind velocity vector

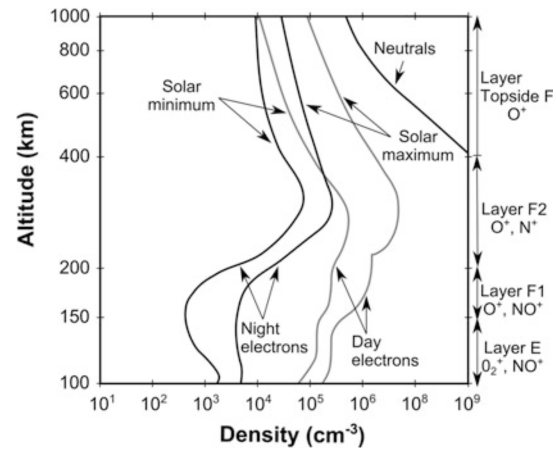

Fig. 3.5 Total ionization profile, with ionospheric layers, adapted from [4]

3.2.7 Aurora Plasma

Above the ionosphere and typically at the magnetic field boundary between high latitude, closed and quasi-closed magnetic field lines is a 'hot' plasma of substantially lower density than the ionosphere but much higher energy (the 'plasma sheet' region in Fig. 3.2). These particles (primarily electrons and protons) precipitate into the atmosphere generating bright arc structures called auroras. Near the Earth's geostationary orbit (the equatorward extension of the auroral plasma), densities are on the order of $\sim 1\ \mathrm{cm}^{-3}$ and mean energies of several tens of keV. This plasma can give rise to surface potentials of 20 kV or more. Auroras are regularly observed at Jupiter and Saturn and there are observations at Uranus and Neptune (Ganymede also has what appears to be auroras). As auroras pose a potential spacecraft charging threat, they need to be considered when evaluating a spacecraft's charging mitigation system.

3.2.8 Trapped Radiation

Superimposed on the closed magnetic field lines of the ionosphere and auroral regimes are the high energy ($E > 100$ keV) trapped electron and proton populations—the so-called van Allen belts. The important components are the electrons with energies between 100 keV to a few MeV and protons with energies from 100 keV to 100 MeV. Jupiter and the Earth have the most damaging radiation belts, though radiation belts exist at Saturn, Uranus, and Neptune. For terrestrial missions, the NASA AE8/AP8 radiation models have been the primary ones used, but the AE9/AP9 models will shortly supersede these. Jovian (GIRE) and Saturnian (SATRAD) radiation models are available from JPL. The Jovian model contains several 'holes'; for example, a lack of a complete statistical understanding and proper modeling of time and pitch angle variations. Preliminary models have also been developed for Uranus and Neptune based on the Voyager flybys. The terrestrial and Jovian radiation belt contours for electrons and protons are illustrated in Fig. 3.6.

3.2.9 Galactic Cosmic Rays

The galactic cosmic ray (GCR) environment consists primarily of interplanetary protons and ionized heavy nuclei with energies from ~ 1 MeV/nucleon to higher than ~ 100 GeV/nucleon. Electrons are also a constituent of GCR, but their measured intensities at energies above ~ 10 MeV are at least 1 order of magnitude smaller than the protons and are usually ignored. The principal element range of interest is from hydrogen to iron. Models of the GCR currently exist for interplanetary space and even for interstellar space (Fig. 3.7) as the Voyager spacecraft are currently crossing into the 'pristine' interstellar medium. Difficulties arise when modeling the detailed spectra for a given orbit within a magnetic field. Models for the Earth are available but the ability to model GCR transport at other planets is limited. For mission design purposes, it is normal to assume a 'worst case' environment (for example, ignoring magnetic shielding). Models of the in situ, trapped heavy ion environments at the Earth and Jupiter are also available for design purposes.

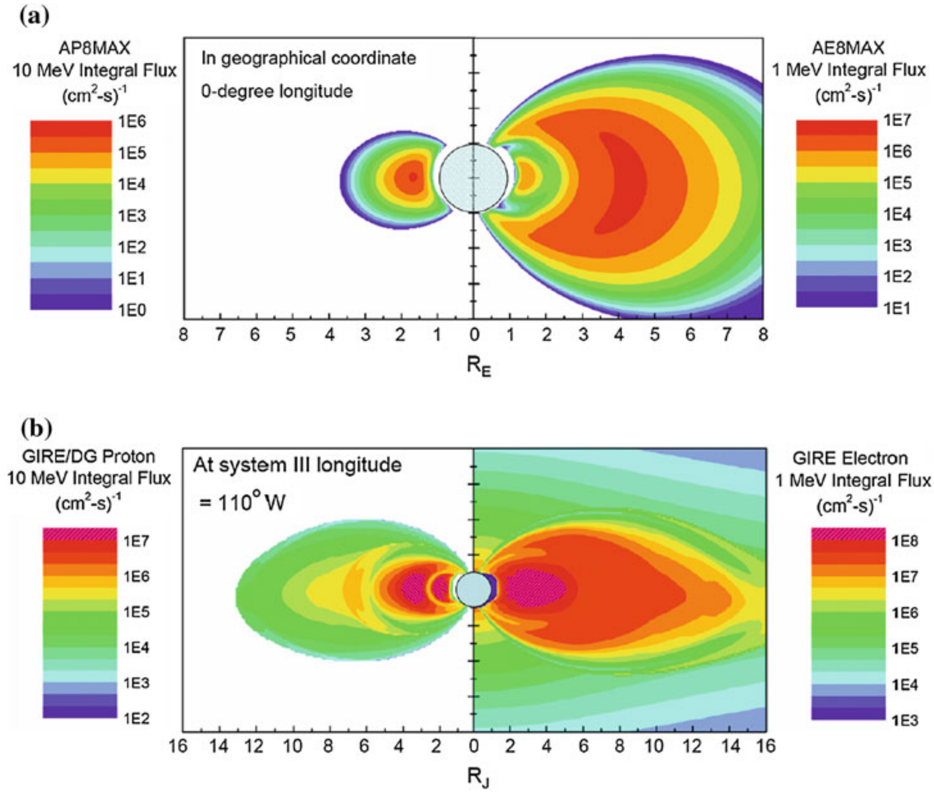

Fig. 3.6 Cross sections of the terrestrial (*top*) and Jovian (*bottom*) radiation belts. Fluxes are for 1 MeV electrons (*right*) and 10 MeV protons (*left*). *Image* I. Jun

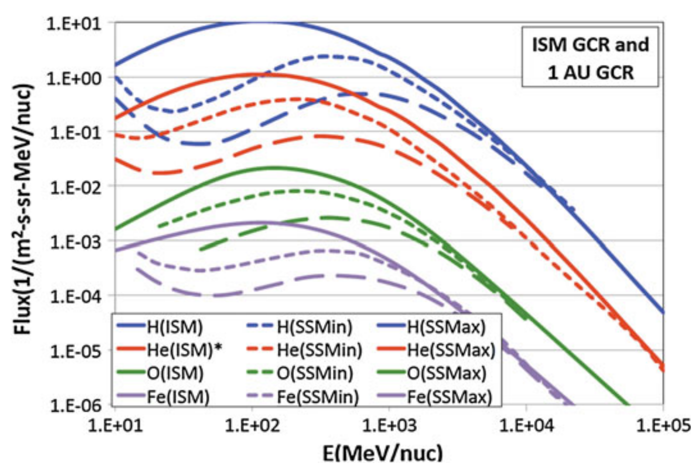

Fig. 3.7 Galactic cosmic rays (GCR) in the interstellar medium (ISM) and at 1 au. Shown are the proton (H), helium (He), oxygen (O), and iron (Fe) fluxes for the ISM (*top curve*), and solar minimum (SSMin-*middle*), and solar maximum (SSMax-*bottom*) conditions

3.2.10 Solar Proton Events

Hydrogen and heavy nuclei in the ∼0.1 to ∼100 MeV/nucleon energy range are ejected during a solar proton event (SPE) or, as it is also called, a solar energetic particle (SEP) event. Intensities are generally a few to several orders of magnitude larger than those of the GCR at these lower energies during these brief events (typically a few days or less in duration). The worst-case solar proton flux is approximately five orders of magnitude larger than the GCR, but becomes 'softer' above ∼100 MeV where the GCR begin to dominate the spectrum.

The energetic particles that make up these events are believed to come from two primary processes: acceleration at the surface of Sun in association with sunspots (so-called solar flares) or at the edge of a rapidly expanding coronal mass ejection (CME) in the solar wind. SPEs created by either process are, after trapped radiation, the major natural radiation of concern to spacecraft designers. Statistical models of the occurrence frequency of the largest events have been developed that can be used to estimate doses for different mission lengths [8].

3.2.11 Meteoroids

Meteoroids are solid particles orbiting in interplanetary space (planetary ring material is a special case of 'meteoroids') and are believed to be either of cometary or asteroidal origin. The mass range is from 10^{-12} g dust grains to 10^{22} g for asteroids and comets (Fig. 3.8). Densities range from 0.5 g/cm^3 (fluffy ice) to between 3.5 g/cm^3 (stony) and 8.5 g/cm^3 (iron/nickel). Impact velocities range from 11 to 70 km/s (the latter particles are believed to be of interstellar origin) with mean values around 20 to 30 km/s. Currently, there are several models available, including the new MSFC Meteoroid Engineering Model (MEM) [9] and the older JPL-developed METeoroid Engineering Model (METEM) [10]. MEM incorporates the latest meteoroid data and is primarily intended for the 1 au environment. METEM provides interplanetary meteoroids from Mercury out to Saturn. The latter has modules for planetary focusing effects and planetary shielding. The METEM model is particularly useful for angular impact estimates and has seen wide use within the community. The database it uses is dated, however, and does not incorporate any of the new data that have become available since its debut.

3.2.12 Synthetic Debris

Space flight operations have led to an artificial shell of synthetic debris around the Earth. This shell of debris poses a greater threat than the natural meteoroid environment within 2,000 km of the Earth. Typical mass densities are 2.5 g/cm^3 and impact velocities are $\sim$10 km/s. The Orbital Debris Engineering Model ORDEM2000 by the Johnson Space Center (JSC) is perhaps the primary debris model and is available to download from the NASA Orbital Debris Program Office at JSC [11]. However, the ESA Meteoroid and Space Debris Terrestrial Environment Reference (MASTER) and Program for Radar and Observation Forecasting (PROOF) models are also widely available and are recommended by the ECSS standards. The ORDEM2000 model provides estimates of the near-Earth debris on a given date and unlike the ESA MASTER model, which is historical, includes spacecraft launch rates and impacts that can be used to project the future debris population. It should be noted however that ESA's Debris Environment Long-Term Analysis (DELTA) tool can be used in conjunction with the MASTER tool to determine future debris trends. Representative debris fluxes are compared with the meteoroid environment in Fig. 3.9.

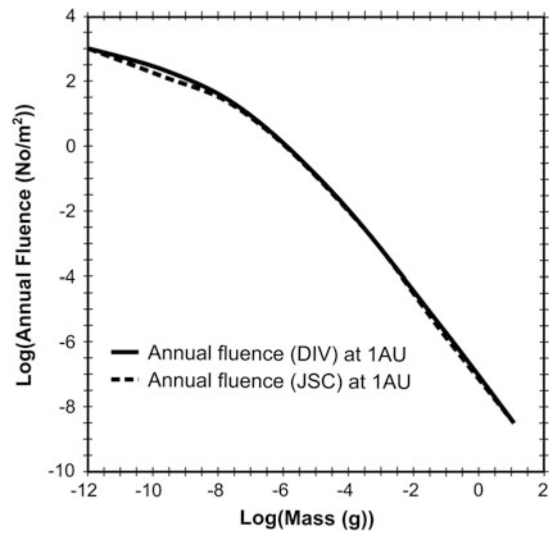

Fig. 3.8 Annual integral interplanetary meteoroid fluencies versus mass at 1 au for two standard meteoroid models

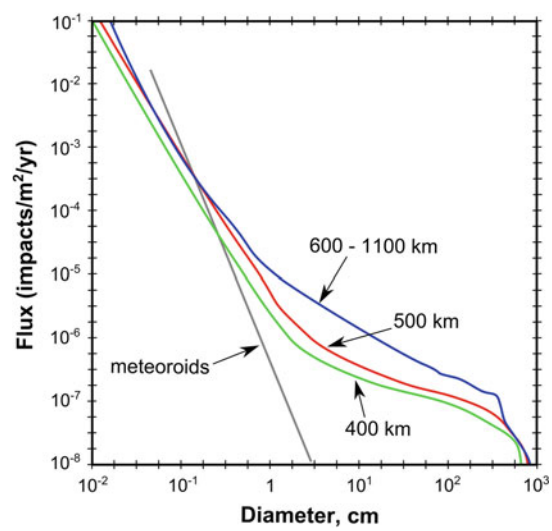

Fig. 3.9 Annual interplanetary meteoroid and space debris fluxes versus diameter at the Earth for various altitudes

3.2.13 Dust

An environment of increasing concern for which there are few models is dust. Significant dust environments have been observed at the Moon, Mars, and comets. Mars dust storms and dust devils and the very 'sticky' lunar dust that astronauts encountered are well-known problem environments. In the past, mission-unique models for specific comet missions have been developed but they are not well defined. Models of the in situ comet dust environment are important as this can seriously affect operations during flybys (hypervelocity impacts) or landings (contamination). 'Dusty plasmas' in this environment—dust that behaves as a collection of charged particles—are another complication. This relatively new environment needs to be carefully considered because of the adhesion of the charged particles on surfaces.

3.3 Interactions

The anticipated sophistication and complexity of future space systems will greatly enhance their sensitivity to environmental interactions, and make what would otherwise have been second-order effects potentially critical problems for survivability. The purpose of this section is to review these interactions and relate them to possible areas of concern for the technologist.

Each category of interaction will be briefly defined in this section and examples provided of potential effects on a spacecraft and its subsystems. In defining these categories, it should be kept in mind that to some degree they overlap as several of the interactions are manifestations of a common underlying phenomenon (i.e., energy deposition or mechanical stress).

3.3.1 Cumulative Radiation Effects

Cumulative radiation effects depend on the type of particles, their energy, and their charge. A high-energy particle can transmute a material (change the atomic species and make the material radioactive), change its atomic structure (displacement damage), or produce free radicals, ions, and electron–hole pairs. Electronic parts and material characteristics thus slowly degrade with time due to these effects. A common measure of damage is total dose. Dose is the amount of energy deposited per unit mass of the absorbing material. An example is the total ionizing dose (TID) which is a measure of the energy deposited in a mass of material creating ionized charge pairs—typical units are 100 ergs/gm or 1 rad. Note that: the material has to be specified, e.g., 'Si' for silicon. For reference, commercial off-the-shelf (COTS) parts are typically 'hard' to sometimes as much as 10 Krads(Si), while space-qualified parts are typically 'harder' than 10 Krads(Si); 'rad-hard' parts are 100 Krad(Si) or higher and parts harder than 1 Mrad(Si) are 'nuclear hardened'.

For engineering purposes, a dose versus depth curve is usually prepared for the design. In the early stages, this is done for a generic mass distribution—solid sphere, spherical shell, flat plate (or slab), two flat plates, etc. A set of calculations for the Clementine lunar missions is presented in Fig. 3.10. As shown in the figures, electrons are much more sensitive to the details of the shielding geometry than ions.

3.3.2 Single Event Upsets

The term single event effects (SEE) encompasses a variety of radiation-induced upsets in microelectronics. Of particular interest are single event upsets (SEU). SEUs are produced in an integrated circuit when a single charged particle passes through the circuit and causes a change in the state of a digital logic element leading to data loss or incorrect commands. As an energetic particle travels through a circuit element, it may deposit energy (producing ionization and a current pulse) sufficient to trigger the element (Fig. 3.11a). The energy loss is principally proportional to the square of the particles electrical charge, Z, but if nuclear interactions occur within the part, this rate can be substantially increased. Hence, more abundant low-Z ions deposit as much energy as less abundant high-Z ions. Figure 3.11b illustrates the actual effects on a Hubble Space Telescope CCD element—note the bright pixels that were 'flipped'. The basic measure of energy transfer is linear energy transfer (LET) typically given in MeV cm^2/mg, which is the energy lost by the particle to the material per unit path length (MeV/cm) divided by the density of the material (mg/cm^3). Hence, multiplying LET by the density of the material being impacted gives the energy deposited per unit length in the material. The SEU rate for each circuit element needs to be evaluated for all the sensitive devices associated with a subsystem on the spacecraft. Figure 3.12 presents representative SPE and GCR fluxes versus LET for varying levels of shielding—shielding has a large effect on SPEs but not much on GCRs; parts with LETs above $\sim$30 must be selected to significantly reduce GCR rates.

3.3.3 Latchup

Another form of an SEE is latchup. The passage of an energetic particle through a sensitive device can sometimes create a transient short circuit or current path. In the case of latchup, this can turn on a parasitic silicon controlled rectifier (SCR) resulting in either a loss of circuit function or thermal runaway from the excessive current. The latter can cause permanent damage. If detected early, both can be mitigated by powering down the device. Given the possibility of permanent damage by this effect, it needs to be evaluated for all potentially sensitive integrated circuits.

3.3.4 Surface Charging/Wakes

Surfaces immersed in a space plasma will charge to a potential relative to the plasma. In sunlight, this is typically a few volts positive due to photoelectron currents. In shadow, to first order, the potential is proportional to the

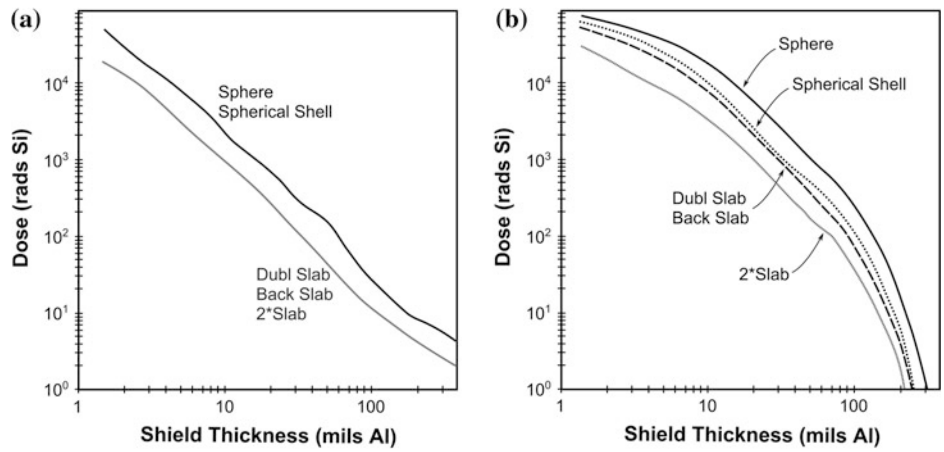

Fig. 3.10 Representative dose/depth curves for the Clementine lunar mission showing the differences between different shielding geometries for protons (**a**) and electrons (**b**). Note that 1 mil = 1/1,000th of an International Inch, which is exactly 25.4 mm

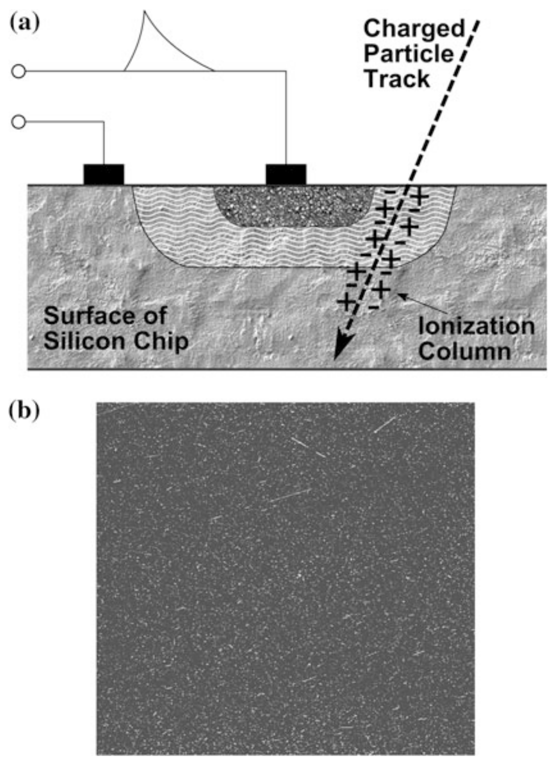

Fig. 3.11 Examples of SEU effects. **a** Schematic illustrates the process of charge deposition in a microcircuit element that leads to a bit flip. **b** CCD image from the Hubble Space Telescope as it passed through the South Atlantic Anomaly showing the effects of SEU events on the CCD pixels

ambient electron temperature (Fig. 3.13a) and current. These potentials can be over 10–20 kV (negative relative to the space plasma ground) and can produce differential potentials over 1,000 V between electrically isolated surfaces. Representative surface potentials in eclipse for the Earth are presented in Fig. 3.13b. As arcing may occur between charged surfaces with potential differences as little as 200 V, this can be a serious environmental concern.

In conjunction with surface charging, a plasmasheath is often created, the scale of which is characterized by the Debye length, denoted as either λ_D or L_D. The formation of this sheath can be understood by considering the effect of placing a surface with no initial net charge into a plasma. At first, more electrons than ions will strike the surface due to the higher thermal speed of electrons; high-energy electrons can penetrate several millimeters, charging internal dielectrics, while lower energy electrons and ions deposit charge on the surface. This electron/ion strike rate imbalance causes the surface to charge negatively until the charge is sufficient to repel further electrons and to attract ions. At equilibrium, the electron/ion currents to the surface balance. A region is formed above the surface within which the positive ions outnumber the electrons, shielding the negative surface potential. Outside the structure, that is the plasmasheath, the ambient plasma does not see the net negative potential.

In the ionosphere the Debye length is typically less than approximately 10 cm, however in the magnetosphere it can be from 0.1 to 1 km. If the Debye length is smaller than the characteristic length of the spacecraft the plasmasheath will provide a conductive path between different parts of the spacecraft, keeping the potential relatively even. However, large potential differences can still occur in the wake region, where the Debye length can be locally large. If the Debye length is larger than the characteristic length of the spacecraft then large potential differences can develop on electrically isolated spacecraft surfaces causing potentially damaging arc discharges that can be a serious concern for solar cell systems, where exposing a semiconductor to sunlight is a required condition of operation. Debye shielding can have a serious effect on particle and field detector instruments, as the presence of the spacecraft alters the very field that the instruments are attempting to measure. As such, instruments are typically placed on electrically isolated structures that extend beyond the Debye shield of the spacecraft and can be biased relative to the spacecraft ground.

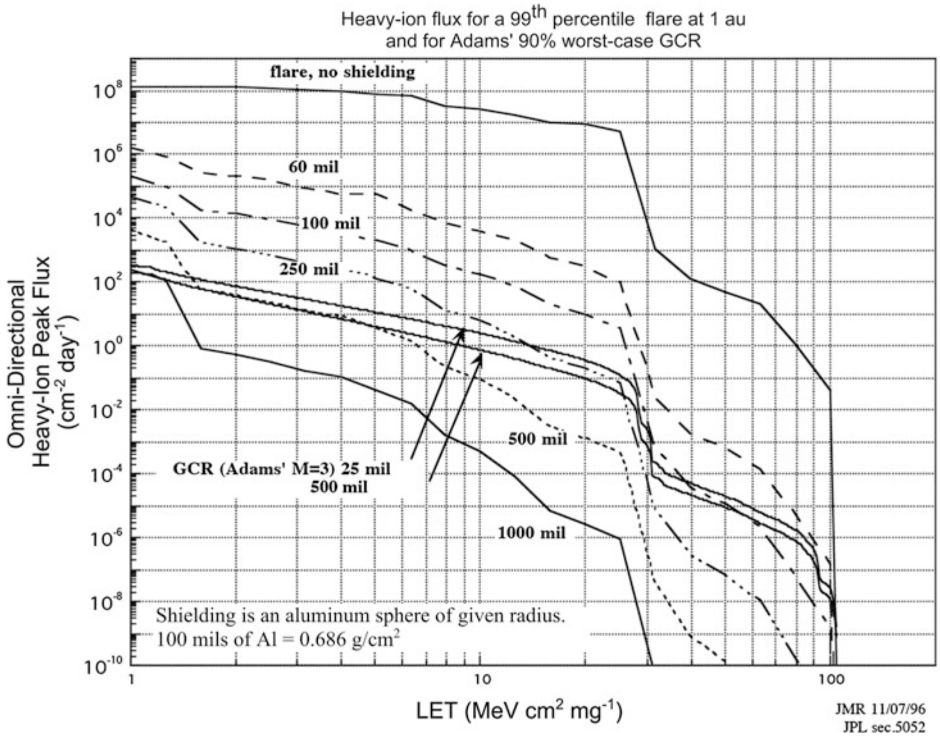

Fig. 3.12 Annual GCR and solar proton event fluxes for various shielding thicknesses. *Image* J. M. Ratliff

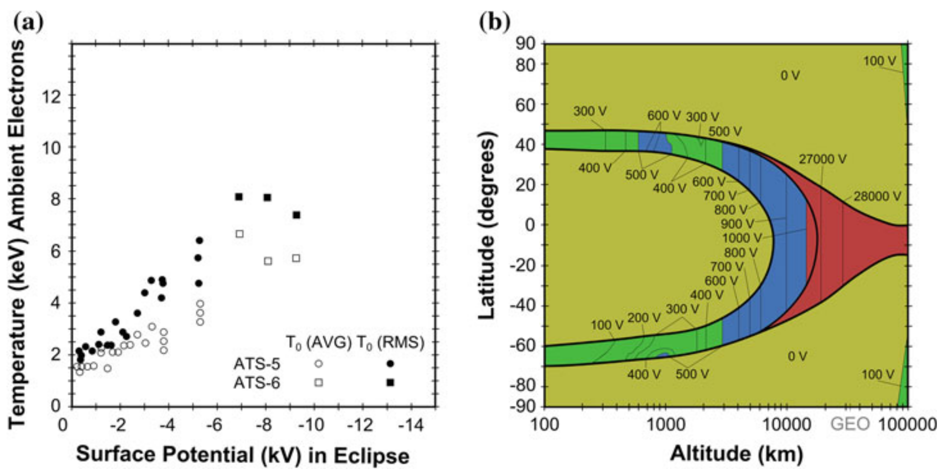

Fig. 3.13 Surface charging effects for the Earth. **a** Comparison of observed surface potentials in solar eclipse at geosynchronous orbit for the ATS-5 and ATS-6 spacecraft versus plasma temperature. **b** Estimates of the surface potential in eclipse versus position for the midnight meridian

Though normally of little concern, in moderate to dense plasmas like the ionosphere the vehicle's velocity relative to the plasma can produce a charged wake structure around the vehicle (and sensors) altering the currents and electromagnetic fields around it. Ionospheric ions typically have thermal velocities lower than the orbital velocity; as such, the motion of the structure causes the plasma density to build up in the ram direction, with a consequently low-density region occurring in the structure's wake. Density deviations can be several orders of magnitude from the ambient and the Debye length in wake can be locally very large, which tends to be negatively charged as plasma electrons typically have a greater velocity than ions. In addition to distortions in particle and field measurements, the dense plasma can also lead to enhanced power loss (positive surfaces that draw electrons) to arcing (negatively charged surfaces). The International Space Station (ISS) flies plasma contactors to minimize these effects.

3.3.5 Internal Charging

In addition to surface charging, internal electrostatic charging/discharging (IESD) (also called buried charging) is a very real concern—particularly at the Earth (Fig. 3.14) and Jupiter. High energy electrons (100 keV or higher) can easily penetrate spacecraft surfaces and deposit charge on or in internal surfaces, but protons of the same energy are stopped

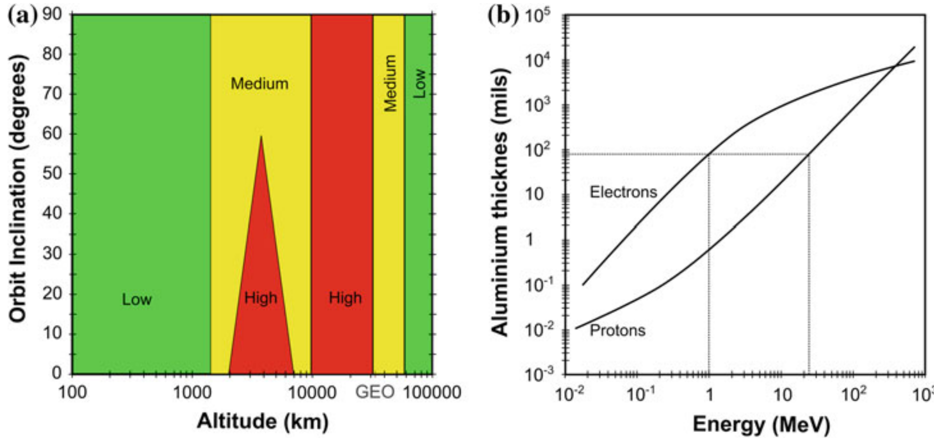

Fig. 3.14 Internal charging effects for the Earth. *Left* provides estimates of regions of IESD concern for circular orbits. *Right* gives the mean penetration depth of electrons and protons in aluminum. As illustrated, a 1 MeV electron penetrates as deeply as a proton of over 20 MeV (as there are many times fewer protons at the higher energy, negative charge builds up leading to IESD). Note that 1 mil = 1/1,000th of the International Inch, which is exactly 25.4 mm

at the surface (Fig. 3.14)—the resulting differential charging can lead to arcing inside the normal Faraday cage of the spacecraft. At least 2.5 mm of aluminum shielding is typically needed in the Earth's environment to prevent this effect. Methods for estimating the effects of internal and surface charging and mitigating their effects are presented in [12].

3.3.6 Power Loss

Although normally of limited concern, as power systems (particularly solar arrays and electrodynamic tethers) approach 200 V or more in operating voltage, positive potential surfaces can experience parasitic power losses. Any exposed positive surfaces (perhaps pin holes in insulation such as produced by micrometeoroid impacts) will attract electrons—for potentials of approximately 200 V or higher, the electrons will receive sufficient energy to generate dense clouds of secondary electrons. This plasma cloud effectively defeats the insulation and results in high ambient electron currents and power loss over positively charged surfaces.

3.3.7 Lorentz Effect

A conducting body crossing a magnetic field will experience an induced electric field proportional to the cross (or vector) product of the instantaneous velocity, **V**, and the magnetic field, **B**, that is (**V** × **B**)—the Lorentz effect. In low Earth orbit, this can be as high as 0.3 V/m. Much higher values are experienced at Jupiter—approximately 60 V/m over the polar caps for the Juno mission. At the Earth, induced voltages of 10 V have been seen on the Shuttle and over 100 V on the ISS. For a 100 km long tether (possible with today's technology), potentials of approximately 10,000 V could be generated and the tether used as a power source (in return for a decrease in altitude). Given the varying nature of the potentials across a structure as it rotates, **V** × **B** potentials can be very annoying for some spacecraft.

3.3.8 Surface Damages

Arc crazing/blow-off, sputtering, ablation due to the neutral atmosphere, EUV-induced chemical changes, radiation damage (especially for Teflon), and other effects can seriously damage exposed spacecraft surfaces. Oxygen erosion in low Earth orbit has been found to be a serious problem for many organic compounds (Kapton in particular) and a few metals (silver and osmium). Even a few days spent in low Earth orbit can seriously damage some types of surface and, in the case of the Long Duration Exposure Facility (LDEF), entire surface samples were found to disappear after a few years of exposure. Finally, micrometeoroid impacts can fracture solar array cover glasses, penetrate cabling, and similarly degrade surfaces. Such degradation will lead to long-term decay in surface properties and must be considered in the selection of surface materials and appropriate coatings or shielding.

3.3.9 Contamination

Outgassing, thruster firings, gas leaks, water dumps, erosion of surfaces, flaking of paints, and long term curing of epoxies can all contribute to the contamination environment around a spacecraft. Charging can lead to enhanced deposition rates on some surfaces while EUV and radiation can alter the chemical effects of the contamination. From changes in alpha/epsilon to the glint of small contaminate particles in the field of view of a sensor or the degradation of optical transmission properties, contamination is a serious problem. Water in particular is a pervasive and potentially highly damaging contaminant (nitrogen purges and expensive ground handling techniques are consequences). Control and limitation of such contamination effects is an important factor that needs to be included in a survivable design.

3.3.10 Atmospheric Glow

Although it has only been detected at the Earth so far, serious optical contamination in the form of a visible glow on surfaces facing into the spacecraft velocity vector has been observed for orbits of 800 km altitude or lower. Figure 3.15 shows an example of this glow along the vertical stabilizer and Orbital Maneuvering System (OMS) pods for the Shuttle. The phenomenon may result from the interaction of atomic oxygen with spacecraft surfaces, as the glow intensity appears to vary with the atomic oxygen density. The interaction generates optical emissions, primarily in the orange range of the spectrum (apparently consistent with the emission spectrum of NO_2) that can contaminate sensitive IR sensors. As the glow appears to come primarily from surfaces in the ram direction and to be enhanced during thruster firings, careful placement of optical sensors and timing of thruster firings may need to be considered in the mission design.

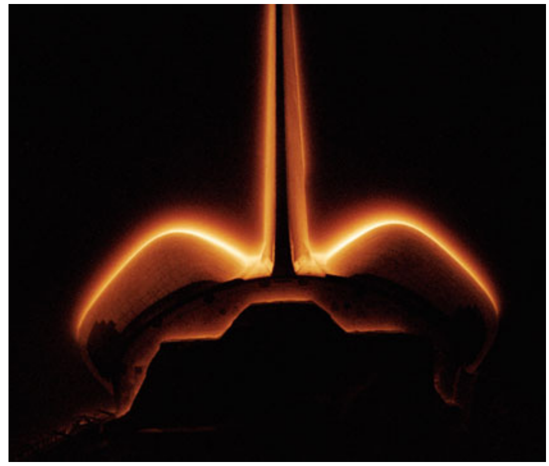

Fig. 3.15 Space Shuttle Columbia during a night pass on STS-62, March 1994. Image documents the glow phenomenon surrounding the vertical stabilizer and the Orbital Maneuvering System (OMS) pods of the spacecraft; NASA Photo ID: STS062-42-026. *Image* NASA

3.3.11 Particle Impacts

Hypervelocity impacts from a few km/s and up between meteoroids or synthetic space debris and spacecraft can be devastating. Interplanetary meteoroid impact velocities average between 20 and 30 km/s (impact velocities as high as 500 km/s, however, may occur near the Sun during a close perihelion passage) whereas space debris impacts are typically 10 km/s. At the Earth, particles with velocities of approximately 70 km/s or greater are believed to be of interstellar origin. Effects range from pitting to complete penetration of walls or even total destruction of a spacecraft. Wiring and pressure vessels (for example, crew quarters or fuel tanks) are particularly sensitive to these effects. Meteoroid shielding is thus a very important consideration for many missions—particularly those to the outer planets where even small pits in the engine nozzles or fuel tanks could lead to catastrophic failures. Externally exposed long cable runs and wire antennas are of particular concern as these are usually thin but very long leading to large areas (greatly increasing their likelihood of getting hit) which could be severed by relatively small particles. Figure 3.17 illustrates the effects of a hypervelocity particle impact on a plate—the particle comes in from the left and exits on the right. Note how the debris cloud expands in a roughly spherical shape (Fig. 3.16).

3.3.12 Torques

The effects of small forces on the stability of spacecraft are well known to cause degradation in pointing accuracy and mechanical deformation. Thermal effects (for example, expansion/contraction of booms), light pressure, gravity gradients, atmospheric drag, meteoroid impacts, and magnetic torques can all cause instabilities. Even arc discharges can impart a measurable impulse. The potential torques on a spacecraft associated with 500 km/s impacts near the Sun may be particularly critical for this class of missions. The sensitivity of the spacecraft to such torques needs to be evaluated for each class.

3.3.13 Thermal

Thermal effects (specifically, the effects of varying temperature and thermal radiation on components) are probably the most important environmental concern for spacecraft. Of particular concern are issues associated with the thermal protection system, as it is often intimately involved with the design of the exterior surfaces of the spacecraft. To reduce mass, design efforts should concentrate on integrating the thermal blanket layout with the meteoroid and spacecraft charging mitigation systems. Typical areas of concern are the conductivity of the thermal blankets, nuclear battery [i.e., radioisotope power source (RPS)] or nuclear reactor

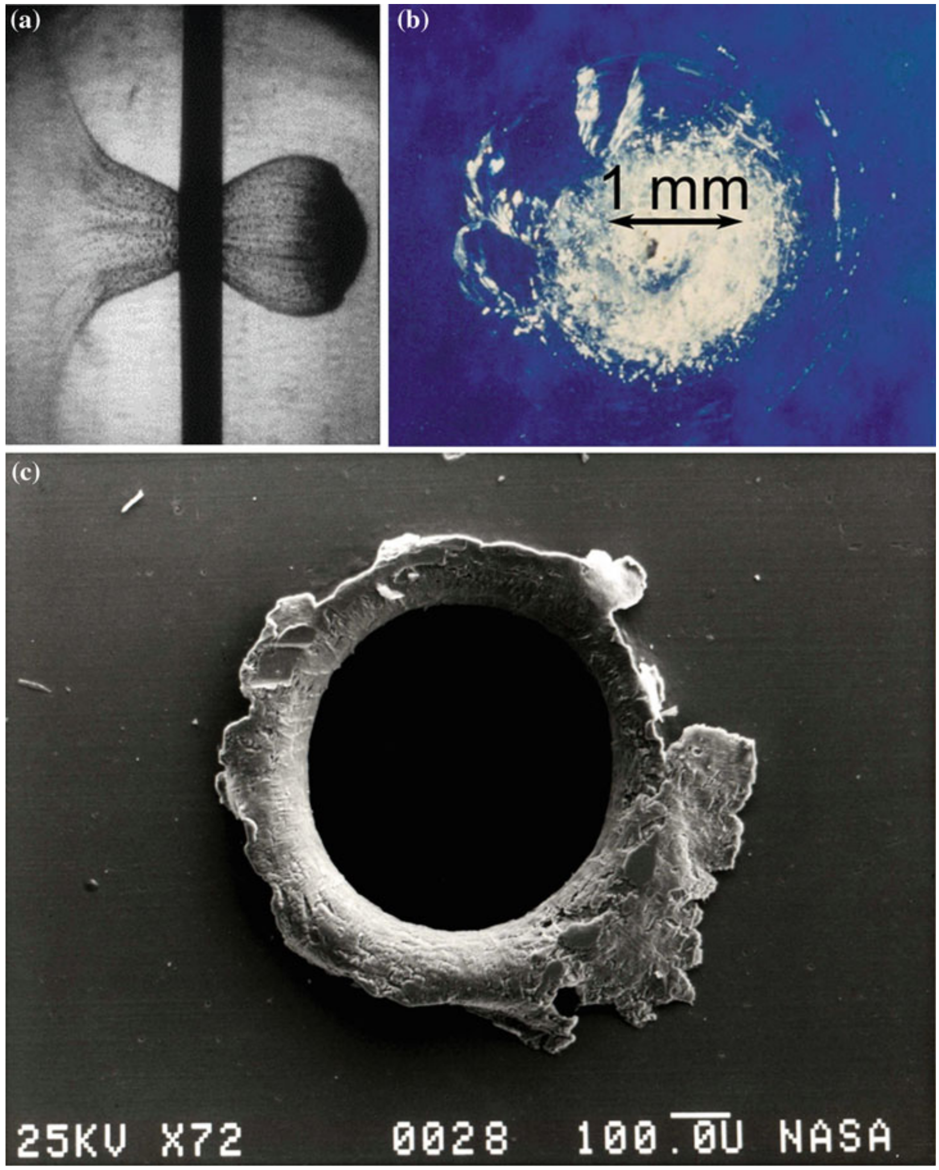

Fig. 3.16 Effect of hypervelocity impacts; **a** the cloud of particles produced by a hypervelocity impact, **b** a window pit from orbital debris on the Space Shuttle Challenger during STS-7 and **c** a view of an orbital debris hole made in the panel of the Solar Maximum Mission, SolarMax, satellite

placement, optical surface reflectors (OSR), thermal sensors (which can pick up stray EM pulses), and, for missions close to the Sun, heat shields. As white paints and Kapton-based thermal blankets, the most common thermal control solutions, are typically non-conductive and sources of arc discharges, there can be complex trade-offs when carrying out a survivability design.

3.3.14 Other Effects

Density fluctuations in the ionosphere lead to enhancements and depletions in the electron density along a radio frequency (RF) propagation path. These processes can distort the phase and amplitude of a signal to and from a spacecraft. In addition to the Earth, typical ionospheric properties and actual measurements are available for several of the planets and the Sun (radio occultation measurements being a common source). Auroras, the galaxy, and the Sun are all natural sources of background RF that can further hamper communications in space. Except in special cases, they are considered to be of secondary importance to typical electromagnetic compatibility (EMC) sources on the spacecraft itself and are ignored for design purposes.

The ambient environment has numerous sources of stray light. Besides the Sun itself which causes significant noise across the entire electromagnetic spectrum, stars, starlight, gegenschein (German for ‘counter shine’; a faint brightening of the night sky in the region of the antisolar point), the zodiacal light (a faint, roughly triangular, whitish glow seen in the night sky that appears to extend up from the vicinity of the Sun along the ecliptic or zodiac), atmospheric glow, the

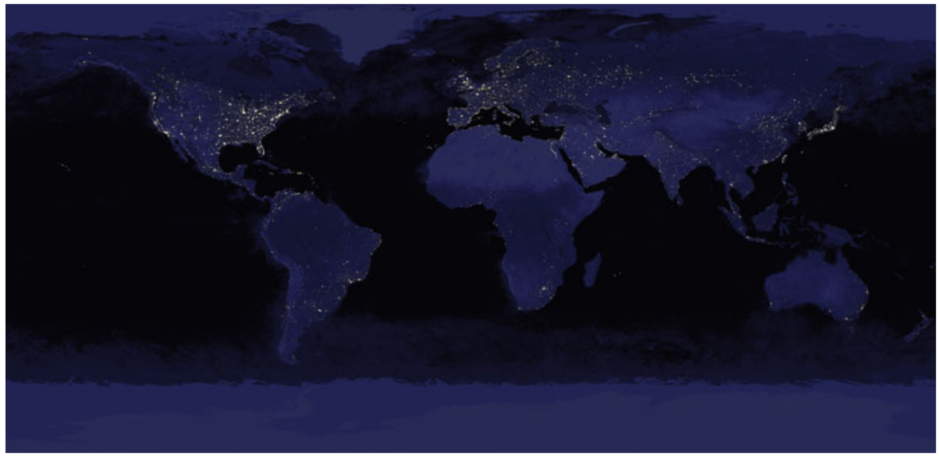

Fig. 3.17 Image of Earth's city lights created with data from the Defense Meteorological Satellite Program (DMSP), Operational Linescan System (OLS). Originally designed to view clouds by moonlight, the OLS was used to map the locations of permanent lights on the Earth's surface. *Image* NASA-GSFC

auroras, the equatorial electrojet, the polar cap aurora, and moonlight all contribute background light in the UV, EUV, and IR. Lately, at the Earth even city lights and oil flares have become a serious concern in making ground observations from space; see Fig. 3.17 and other representative images from the Defense Meteorological Satellite Program (DMSP) which have been particularly useful in bringing this problem to the attention of the public. These effects are considered to be of secondary importance in an integrated design (Fig. 3.18).

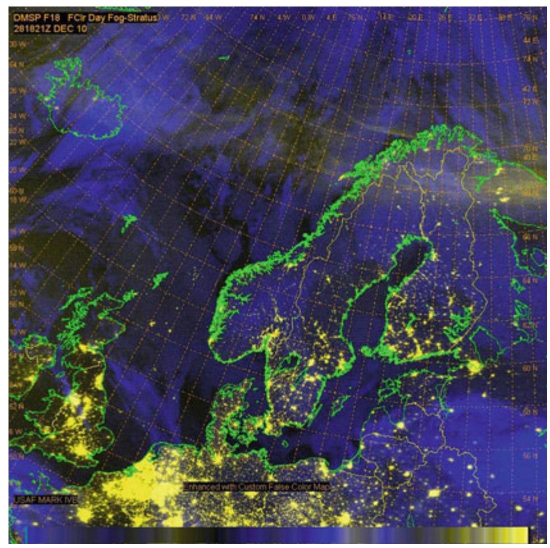

Fig. 3.18 A Defense Meteorological Satellite Program (DMSP) image from December 28, 2010 showing auroras over the polar regions north of Scandinavia outshone by the city lights of Europe. The image uses both nighttime visual and infrared imagery from the DMSP satellites F17 & F18. *Image* US Navy Fleet Numerical Meteorology and Oceanography Center

Rapidly oscillating fields on surfaces can cause serious interactions with space plasma. In particular, depending on the electron density, the mechanical spacing of the elements, and other factors, a resonance phenomenon called multipacting can be induced between the surfaces. Briefly, electrons accelerated into the surface by the time-varying positive component of the field can generate secondaries. These secondaries in turn (if the spacing and timing are correct) can generate more secondaries when they impact, causing a plasma avalanche. The cloud of electrons can lead to significant losses in signal strength and drain power from a transmitter. Whereas power loss is primarily a direct current (DC) process, this is an alternating current (AC) effect. The simulation of this phenomenon, unlike many other environmental effects, is fairly straightforward with the ambient environment playing only a secondary role once the process is initiated as the secondary electrons created quickly outnumber the ambients. Even so, systems must take this effect into account and be tested under ionospheric plasma conditions and, if possible, under the appropriate neutral atmosphere conditions before flight.

3.3.15 Interactions Versus Environments Trade Matrix

The matrix in Table 3.2 compares the key design environments for spacecraft with the critical interactions. For every mission the technologist needs to identify the relevant mission environments and then determine the specific interactions of concern for that mission and its different phases. This has been done for three representative cases: a mission to Europa (E or e), an outer solar system mission to the dwarf planet Pluto (P or p), and a solar probe-like mission (S or s). Also indicated is whether the interaction is a major (capital letter) or minor (lower case letter) concern. Such assessments are very dependent on the specific mission but are helpful in identifying the critical interaction concerns. As such, this should be the first step in developing an integrated, survivable design.

3.4 Design Options

In this section the design trade space, namely the design options available to the reliability engineer for mitigating specific interactions will be described. Examples range from shielding for meteoroid impacts and blankets for thermal control to the careful selection of conductive materials for spacecraft charging and radiation protection. In particular, Tables 3.3 and 3.4 list some of the major design options for mitigating space environment effects. Representative mitigation options are discussed in more depth in the following sections.

3.4.1 Shielding

Probably the best-known environmental mitigation method is shielding. A number of shielding methods exist for mitigating specific interactions. For radiation and hypervelocity impacts in particular, shielding has proven to be a primary means of limiting their effects. Indeed, as the major objective is typically to place physical mass between the impactor (atomic or particulate) and the target, the methods for mitigating both are similar. The first step is typically to account for the intrinsic spacecraft body shielding. Where necessary, sensitive elements such as radiation-soft electronic components can be protected by bulk shielding, perhaps by encasing assemblies or spacecraft subsystems within a box of high-Z materials—referred to as a doghouse or vault. Or, for the case of hypervelocity impacts, by a specially designed, multi-layer meteoroid Whipple shield for crew quarters or fuel tanks. Special shielding plates can be placed between assemblies or slices in a stack. A specific technique for particularly radiation-soft parts is a so-called spot shield—this can be effective in conserving shielding mass. For other interactions there are similar techniques. In the case of electromagnetic compatibility (EMC) and electromagnetic interference (EMI), often termed EMC/EMI, an electrically conductive Faraday cage is usually employed. Baffles and deflectors are used to limit thruster contamination while in-flight removable covers may be used for science instruments.

3.4.2 Positioning

Closely tied to the use of shielding is positioning—the placement or orientation of critical components or surfaces to minimize the effects of the environment. Examples are the placement of radiation sensitive devices as far as possible from a nuclear power source or the orientation of a surface relative to the velocity vector to minimize the meteoroid fluence. Each of these forms of positioning can yield significant reductions in specific effects and may have little or no impact on a system design. Examples are listed in Table 3.3.

3.4.3 Material Properties

Proper material selection for a specific environment can significantly improve the lifetime and reliability of a space system. Careful selection of materials can both limit and prevent many types of environmental interaction. As discussed earlier, there may be complex trade-offs between conductivity requirements, thermal paints, and radiation sensitivity. The steps required to identify the appropriate materials for mitigating interactions are listed in Table 3.4.

3.4.4 Electronic Parts Selection

Electronic parts are a critical component of a spacecraft design. Parts engineers need to evaluate a range of parameters in identifying the appropriate components for a specific mission. A parts engineer must trade cost and availability versus the class of a part—for example, radiation-hard, space-qualified (Class S or Class B) parts versus commercial parts. Note that failure rates of Class S parts are generally about a quarter of the rate for Class B parts [13]. The relationship between the classes is detailed in Table 3.5 [13]. The parts engineer must trade the known greater cost and limited availability of radiation-hard parts against the costs of radiation testing or the advantages of more capable commercial (non-radiation-hard) parts. Note that typically redundancy is not a justification for using a lower class part as it provides a lower reliability payoff at the point where it is needed. Maverick parts, production flaws, and other uncertainties, however, justify redundancy for critical circuits in high-reliability, long-life applications to protect against random failures [13]. For long-life, the use of high-reliability hardware, Class S parts, and redundancy in critical applications can provide an optimum and cost-effective approach. Parts also need to be evaluated for their electrostatic discharge (ESD), sensitivity. Increasingly, checks must also be made for counterfeits and, in the US and Canada, technologists should review parts lists for Government-Industry Data Exchange Program (GIDEP) alerts.[2]

3.4.5 Circuit/System Design

Careful circuit and system designs can be used to limit and mitigate the effects of radiation and thermal effects—there are for example circuits that are designed to compensate as

[2] See http://www.gidep.org/.

Table 3.2 Key environments and interactions matrix for three representative missions to Europa (E, e), Pluto (P, p) and a Solar Probe (S, s)

Environments	Interaction												
	Cumulative radiation effects	Single event upsets	Latch-up	Surface charging/wakes	Internal charging	Power loss	**V** × **B**	Surface damage	Contamination	Glow	Particle impacts	Torques	Thermal
Neutral atmosphere													
E, **B** fields				ES			e					E	
Ultraviolet radiation				S				S	S				
Infrared radiation								S					S
Solar wind plasma				s								s	
Ionospheric plasma				S		e	e						
Aurora plasma				E									
Trapped radiation	E	Ep	E		E								
Galactic cosmic ray		P	P										
Solar proton events	Sp	Sp	Sp										
Meteoroids						s					SEP	S	
Debris											s		
Dust											sep		

Upper-case letters are major effects and lower-case are minor effects. Assessment varies depending on spacecraft design

Table 3.3 Examples of different placement and orientation techniques for mitigating spacecraft/environment interactions

Placement	(1)	Place systems close to or far away from thermal sources for heat control
	(2)	Place radiation sensitive systems far from radiation sources such as an RPS, reactor, or RHUs
	(3)	Place contamination-sensitive devices out of the line of sight of known contamination sources
	(4)	Orient optical sensors flying in low Earth orbit so that they don't look over surfaces prone to glow
	(5)	Put radiation and meteoroid sensitive systems as close as possible to the center of the shielding protection system
Orientation	(1)	Orienting large, flat surfaces relative to the velocity vector in low altitude orbits to maximize or minimize drag.
	(2)	Orienting the more meteoroid or debris impact sensitive surfaces away from the maximum anticipated angle of fluence
	(3)	Orientation of current loops or the spin axis relative to the magnetic field to control magnetic torqueing
	(4)	Orientation of oxygen erosion sensitive surfaces away from the vehicle normal while in low Earth orbit
	(5)	Orientation of a large array or space tether relative to the magnetic field and velocity vector to alter the induced electric fields
	(6)	Orientation so that sunlit and shadowed surface combinations that may cause arc discharges are minimized or that differential charging due to shadowing is minimized
	(7)	Orientation of a thermally sensitive surface in or out of sunlight to enhance heating or cooling

Table 3.4 Steps for selecting materials to limit environment interactions

Material selection	(1)	For charge mitigation, assess conductivity of internal and external materials
	(2)	To limit radiation effects on materials, assess the long term radiation response of the materials, particularly materials directly exposed to environment and those that are lightly shielded (e.g., behind thermal blankets)
	(3)	Assess the degradation from meteoroid, debris, and dust hypervelocity impacts
	(4)	For thermal control, assess absorptivity, emissivity, and transparency of materials
	(5)	Avoid materials with adverse outgassing properties
	(6)	Assess compatibility of materials at interfaces

Table 3.5 Difference between Class S and Class B parts [13]

Issue	Class S	Class B	Impact
Wafer lot acceptance	Required	–	Uniformity and pedigree traceability
Certification of production facilities	To specific assembly lines	To technologies and general facilities only	Burn-in and screening value relates to consistency of original product
Precap internal inspection	100 %	Sampled	Significant driver on level of reliability—criteria much more stringent in MIL-M-38510H
Particle impact noise detection (PIND) testing for loose particle detection	Required	–	Loose metallics in zero g field can cause failures
Serialization	Required	–	Traceability lost
Interim electrical test between test phases	Required	–	Potential of passing over problems and their causes
Burn-in	240 h	160 h	Later problem discovery
Reverse bias burn-in	Required	–	Impurity migration not detected
Interim electrical test after reverse bias burn-in	Required	–	Effects of reverse bias burn-in may be masked by subsequent actions
Radiographic inspection	Required	–	Observation of latent defects
Non-destructive 100 % bond pull test	100 %	Sampled	Parts with mechanical deficiencies get into equipment

their parts properties drift out of specification. Error detection and correction (EDAC) software is of special value in mitigating the effects of memory bit flips (SEU). Memory sparing and scrubbing are also useful in protecting against single event upset (SEU), single event latchup (SEL), and generic part failures. Designing to worst-case parametric degradation values over voltage, temperature, life, and radiation is also of value. Where possible, a designer should introduce system redundancy at functional, subsystem, or system levels. Single-point failures and effects need to be given special consideration in the system design. It is strongly recommended to perform a failure mode, effects, and criticality analysis (FMECA), parts stress analysis, worst-case analysis, and/or voltage, temperature, and frequency margin tests. In the case of spacecraft charging, the NASA Handbook 4002A [12] may be followed. Finally, all spacecraft circuitry should be analyzed and tested for electromagnetic compatibility (EMC) and electromagnetic interference (EMI), EMC/EMI.

3.4.6 Grounding

Proper grounding methods need to be considered in any electrical design. Standard techniques are described in NASA Handbook 4001 [14]. Typical methods are to provide a ground reference or resistive bleed path for circuit elements at all times—designers should choose system electrical and electronic grounding architecture to avoid structure currents and ground loops.

3.4.7 Trajectory

An obvious method for limiting environmental concerns is through careful orbital trajectory selection. A mission planner could consider optional trajectories that minimize meteoroid, radiation, and charging exposure. The Juno mission is a case in point because the spacecraft trajectory was selected to be highly eccentric and to pass over the poles in such a way that for much of the mission it avoids Jupiter's intense radiation belts. Similarly, the Voyager and Cassini trajectories were selected to pass through the gaps in the Saturnian rings to avoid particle impacts.

3.4.8 Operational Procedures

As in careful trajectory selection, operational procedures can significantly mitigate environmental effects. Specific procedures can be simulated using a testbed and the effects of part, assembly, functional, and spacecraft subsystem failures can be evaluated versus mission timeline. Fault tree analyses can be used to develop fault protection software. Vehicle operational procedures, such as orientation (say, relative to the Sun for thermal protection), can be implemented to limit specific effects during certain mission phases. For example, Cassini was flown in a fixed orientation during the cruise phase to provide a reduction in impacts on its rocket nozzles by a factor of approximately four. Finally, operational modes like hot (electronics 'on') versus cold (electronics 'off') and sparing can be used.

3.4.9 Construction Methods

Proper construction methods are a clear necessity for high reliability and for preventing design failures. These go hand in hand with correct handling procedures (see ISO 9000 practices for proper handling techniques). For complex space systems, the engineer must be acutely aware of how the electrical harnessing layout is constructed and the layout of grounding wires as these can contribute to ground loops and EMC/EMI concerns. Construction techniques need to minimize/limit the contamination of sensors, optics, paints, and coatings. A particularly dramatic example occurred for the first Shuttle tether experiment, on-board STS-75 in 1996. The electrical conductor of the tether was a copper braid wound around a nylon string, encased in Teflon-like insulation, with an outer cover of kevlar. All of this was then placed inside a nylon sheath. Apparently during construction, in winding up the tether on a spool, a wire filing was inadvertently forced into the cable insulation. As the tether was deployed, the imperfection caused a short between the Shuttle and the tether causing an arc discharge that severed the tether.

3.4.10 Interactions Versus Design Options

Given the various interactions and the design options available, a correlation matrix can be constructed that relates the appropriate effect with a means for mitigating that effect. Table 3.6 is such a matrix and provides an example of how the engineer might weight the comparative values of a design option versus specific interactions. Note that a '**3**' represents a principal method for mitigating an effect and should be given careful consideration in mission design, whilst '*1*' represents a method with minor effect. In an actual case, a mission-specific assessment should be made for the various spacecraft designs being evaluated.

3.5 Design Factors

The final step in determining the proper mix of design options and mitigating techniques for a given mission is to identify mission-specific design factors that must be considered by a project. While these factors, described in the following, are straightforward, the project management must carefully weigh their comparative value or impact on a specific project—they are not necessarily strictly engineering issues. As an example, the use of a radioisotope power source demands a strong technical justification, and the mission must undergo the Nuclear Safety Launch Approval process. A project must carefully weigh the advantages of the power source versus the additional costs of using it *in lieu* of a solar array.

Table 3.6 Design option space versus interactions, note assessment depends on spacecraft design

Interactions	Design options								
	Shielding	Positioning	Material properties	Electronic parts	Circuit/ system design	Grounding	Trajectory	Operational procedures	Construction methods
Cumulative radiation effects	**3**	**3**	2	**3**	**3**		**3**	*1*	
Single event upsets	2	*1*		**3**	**3**		2	2	
Latch-up	2	*1*		**3**	**3**		2	2	
Surface charging/ wakes	**3**	**3**	**3**	*1*	**3**	**3**	**3**	2	**3**
Internal charging	**3**	2	**3**	2	**3**	**3**	**3**	2	**3**
Power loss	**3**	**3**	**3**						*1*
V × **B**		2				**3**	2		
Surface damage		**3**	**3**						*1*
Contamination	**3**	**3**	**3**			2		2	**3**
Glow		**3**					2	**3**	*1*
Particle impacts	**3**	**3**	2				**3**		
Torques		**3**	**3**				2	**3**	
Thermal	**3**	**3**	**3**				*1*	**3**	

A '3' represents a principal method for mitigation, whilst '*1*' represents a method with minor effect

3.5.1 Cost

The most obvious mission factor to be considered is cost in its various forms. In evaluating every design trade, there are inevitably monetary costs to be compared. The monetary costs are of course very dependent on the mission requirements and the class of the mission—typically, the higher the required reliability of the mission (Class A being the highest, Class D or now E being the lowest), the higher the monetary costs.

3.5.2 Mass

Most missions, because of launcher constraints, are driven by limited mass requirements. Radiation shielding (Europa) or the requirement for redundant systems (Pluto) would likely drive mass requirements in the environmental effects arena.

3.5.3 Power

Although power requirements and the power source (say, solar arrays versus radioisotope power sources) would be major issues for many programs, only the decision on operating voltage will likely have a direct impact on environmental interactions. Many of the other design trades will be affected but will likely not be as critical for the space environment effects considered.

3.5.4 Complexity

Increased complexity may well be a major fallout from any environmental survivability trade. Key impacts would be in more elaborate shielding design (both radiation for Europa and thermal protection for Solar Probe), careful positioning (placement of systems on Europa and Solar Probe would be particularly constraining), special material selection (the Solar Probe heat shield), advanced EDAC software (the beacon mode reliability), enhanced redundancy (Pluto), and elaborate circuit design (Europa radiation hardening).

3.5.5 Reliability

Reliable operations over a decade or more (for example, the Pluto New Horizons mission) or in extreme environments (like the harsh radiation environment at Europa) require special care in the areas of EDAC software, redundancy, and parts hardness. Long-term reliability ultimately leads to the need for a complete systems approach.

3.5.6 Availability

Availability encompasses multiple issues. The first is the well-known issue of parts or design availability—can the necessary parts or a usable design be found? The second issue is system availability to the operators—if this design fix is employed, will it lead to increased down time? That is, if the spacecraft is required to point a certain way to avoid meteoroid impacts would that limit the useful scientific data received?

3.5.7 Usability

In the case of usability, after applying a particular design trade, the designer should determine how it would influence the ease of operation of the vehicle. Would the design fix make it impossible to perform a certain series of operations? Would it rule out operations that might be critical to meeting the mission requirements? Furthermore, the designer should consider whether it is desirable to allow a spacecraft to do something potentially harmful to itself, or whether it is better to simply protect against this through diligent operations procedures. The use of operations procedures in place of physical limitations typically increases the likelihood of recovery from non-nominal scenarios, hence increasing usability.

3.5.8 Special Issues

Although a catch-all, the primary ‘special issues’ being considered here are the politics of radioisotope power supplies, their environmental issues, and the launch vehicle limitations imposed by such programs. Other examples are missions such as Europa and Pluto that place unique planetary contamination requirements on the spacecraft that may affect specific environmental design choices.

3.5.9 Design Options Versus Design Factors

Given the various design options available, a correlation matrix can be constructed that relates the design options and factors. Table 3.7 compares the overall trade space or set of design options with the various factors assuming a representative mission set (Europa, Pluto, and Solar Probe). The specific ratings provided are for a representative mission set where the major design factors would likely be radiation and meteoroid shielding (Europa), material development (Solar Probe heat shield), software development (for a long-term autonomous beacon mode), redundancy (for the Pluto mission), parts hardness (Europa), and trajectory (all three missions would require complex trajectory calculations).

3.6 Designing for Survivability

To summarize, environmental interactions can have serious negative consequences for a mission’s survivability. Systematic consideration of the available mitigation techniques can limit these problems and lead to a much more reliable and often less expensive design. Tracing the pathway from environment to interaction to design options and then evaluating the options based on programmatic factors, however, can be an involved process. The rewards, though, should be obvious—a better-optimized design based on cost, mass, and reliability trades. At the least, by following such a process, the major space environment concerns can be identified early in a program when mitigation can be done relatively inexpensively. This chapter has provided a systematic method for considering the many trade-offs that need to be included. In particular, filling in Tables 3.2, 3.6, and 3.7 provides a formal means of carrying out a first-order evaluation of the ‘tallest tent poles’ in the optimization of a spacecraft design and provides a starting point for identifying the principal mitigation methods.

Table 3.1 summarizes the overall procedure for carrying out an integrated spacecraft design to optimize survivability in the space environment. The principal point to take away is that the designer must consider all the possible environments of concern and their effects early in the design process. Failure to do this can significantly increase the cost and schedule in developing a viable mission concept—it has been said that if finding and addressing an issue in the design phase costs one USD, addressing it in the construction phase will cost ten USD, while addressing it during launch preparations will cost 100 USD. Having to address it during flight may mean loss of the mission.

3.7 Suggested Resources for Space Environment and Survivability

To conclude this chapter, some of the main published resources that the reader should consider in developing a space environment survivability evaluation of a spacecraft are listed below. These are primarily reference books aimed at summarizing the environments and their effects in broad terms. WIKIPEDIA and other online sites are also listed. Unfortunately, and fortunately, these latter sites are periodically updated and thus are subject to change. Of particular

Table 3.7 Design options versus factors/criteria that must be considered in selecting between the options, note assessment depends on spacecraft design

Design options	Factors							
	Cost	Mass	Power	Complexity	Reliability	Availability	Useability	Special issues (RPS)
Shielding	**3**	**3**	2	**3**	**3**		**3**	**3**
Positioning	2	*1*	*1*	**3**	*1*	2	**3**	**3**
Material properties	**3**	*1*		2	2	**3**	*1*	**3**
Electronic parts	2			2	3	2	**3**	
Circuit/system design	**3**	2	**3**	**3**	3	2		**3**
Grounding	2		2	**3**	2		2	
Trajectory	*1*		2	2	2			**3**
Operational procedures	**3**		*1*	2	2		**3**	**3**
Construction Methods	*1*		2	**3**	2		*1*	2

A '3' represents a major, whilst '*1*' represents a minor effect

note, however, are the Space Environments and Effects homepage supported by NASA MSFC and the Space Environment Information System (SPENVIS) website supported by ESA. These two sites provide access to key environment and interaction programs for actually computing the environmental properties and their effects on space missions. Finally, the author strongly recommends that all environmental and survivability analyses start with a visit to your organization's Reliability Engineering professionals.

3.7.1 Further Reading

- Garrett, H.B., and C.P. Pike, eds. "Space Systems and Their Interactions with Earth's Space Environment." Prog. Astronaut. Aeronaut. 71, 1980.
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- Tribble, A. The Space Environment: Implications for Spacecraft Design. Princeton, NJ: Princeton University Press, 1995.
- Hastings, D., and H.B. Garrett. "Spacecraft-Environment Interactions." Atmospheric and Space Science Series, ed. A.J. Dessler. Cambridge, England: Cambridge University Press, 1996.
- Pisacane, V.L. "The Space Environment and its Effects on Space Systems", AIAA Press, Reston, VA, 2008.
- Garrett, H. B., and Whittlesey, A. C., "Guide to Mitigating Spacecraft Charging Effects", JPL Space Science and Technology Series, J. H. Yuen, Editor-in-Chief, John Wiley and Sons, Inc., Hoboken, NJ, 221 pages, 2011. (Note: also published as "Mitigating In-Space Charging Effects—A Guideline", NASA-HDBK-4002A, 3 March 2011.)

3.7.2 Further Online Reading

Note that these sites and addresses are subject to change; hence, the title could also be used as a search term.

Title (search term)	Address
Space Environment Information System (SPENVIS)	http://www.spenvis.oma.be/
CRÈME Homepage (CREME96)	https://creme.isde.vanderbilt.edu/
NASA Technical Standards Program	http://standards.nasa.gov/
Space Engineering Standards (JPL)	http://engineer.jpl.nasa.gov/standards.html
Space Engineering Practices (JPL)	http://engineer.jpl.nasa.gov/practices/
Geomagnetic Field Models	http://www.ngdc.noaa.gov/geomag/
International Geomagnetic Reference Field	http://www.ngdc.noaa.gov/IAGA/vmod/igrf.html
International Meteor Organization Index	http://www.imo.net/
Debris Models	http://orbitaldebris.jsc.nasa.gov/
Government—Industry Data Exchange Program (GIDEP)	http://www.gidep.org/
NASA National Space Science Data Center (NSSDC)	http://nssdc.gsfc.nasa.gov/
Today's Space Weather	http://www.swpc.noaa.gov/
The NASA Space Weather Bureau	http://spaceweather.com/
National Geophysical Data Center	http://www.ngdc.noaa.gov/
USGS Geomagnetism Program	http://geomag.usgs.gov/
The Aurora	http://www.geo.mtu.edu/weather/aurora/

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Title (search term)	Address
DMSP Auroral Photos (Latest Aurora)	http://www.ngdc.noaa.gov/dmsp/
Recent Satellite Outages and Failures	http://sat-nd.com/#FAILURES
Meteor Showers	http://www.meteorblog.com/
NASA MSFC Space Environments and Effects (SEE)	http://see.msfc.nasa.gov/
JPL Homepage	http://www.jpl.nasa.gov/
ESA Space Debris	http://www.esa.int/esaMI/Space_Debris/

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12. H. B. Garrett and A. C. Whittlesey, "Mitigating In-Space Charging Effects—A Guideline," NASA, Washington, DC NASA-HDBK-4002A, p. 181, Mar. 3, 2011.
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