A New Approach to Radiation Tolerance for High-Orbit and Interplanetary SmallSat Missions

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This paper describes a new approach to analyzing and achieving high radiation tolerance using commercial-off-the-shelf (COTS) piece-parts in higher radiation operating environments. The approach herein combines empirical single event effects testing (SEE) using, high-energy monoenergetic protons, with an analysis of the expected integral fluence and flux at higher particle energies not typically covered by proton testing. The empirical proton testing and analysis of the operating environment are then combined to estimate an overall likelihood of SEE in a given deep space mission. The concept of “limiting cross-section” is introduced, which places an upper bound on the worst-case susceptibility of any particular COTS device not covered by proton testing. This simple but powerful approach lets designers quickly evaluate the risk to hardware and missions not covered with testing. Lastly, it is observed that for many high-gate-count piece-parts with high-Z materials, which produce high-energy nuclear scattering, the use of proton screening for SEE may be almost entirely sufficient for select deep space environments.

Introduction

Recent years have seen the announcement of a number of deep space nanosatellite missions, targeted at cis-lunar space, near-Earth asteroids, and even Mars (with two nanosatellites—MarCo—even having been launched as of this writing). These missions push small spacecraft design to its limits, as the deep space environment is in many ways less benign than the low-Earth orbit environment: communications require larger apertures and more power, navigation cannot use existing assets such as GPS, propulsion is required for both trajectory control and attitude control system desaturation, and so forth. However, the physics underlying these challenges is well understood, and they can be addressed with conventional small-spacecraft hardware, or with scaled-down “big space” technologies.

However, one significant challenge facing the small-spacecraft designer is the ionizing radiation environment. For LEO missions, there are several existing methodologies that address this. These can be broken down into three major categories:

1. do nothing, and see what happens
2. “careful COTS”, with key components tested, screened, and/or shielded [1]
3. full radiation-hardened design with specially-designed and tested parts

The third option generally results in the highest possible confidence in mission success. However, using radiation-hardened parts can result in very high costs, both direct (such parts can be very expensive to procure) and indirect – they are often less capable, larger, and more power-hungry than state-of-the-art COTS components. Candidate parts are screened for SEE with a variety of heavy ions, chosen to cover the Linear Energy Transfer (LET) spectrum expected to be seen during the mission. This testing strategy is very time- and resource-intensive: parts need extensive preparation such as de-lidding and thinning, and suitable accelerator facilities are costly to use and heavily subscribed.

The “do-nothing” approach can be effective for many low-cost missions, but brings with it attendant risks. Many modern electronic parts can be quite robust against radiation effects, and so a mission which does no mitigation can succeed; however, very little can be said about the probability of success a priori. Even worse, even flight heritage
cannot be relied on without lot control, as part performance can vary between manufacturing lots. Simply put, this approach is playing Russian roulette with nature’s high-energy particle gun.

The “careful COTS” approach introduced by Sinclair and Dyer [1] is a powerful approach that improves the probability of mission success while controlling overall cost. In this approach, key COTS components are screened and tested, while the rest of the spacecraft is designed in such a way that radiation-induced failures have a limited impact on the mission. In order to keep testing costs down, this approach typically screens parts for SEEss using high-energy (50-200 MeV) protons; optionally, TID susceptibility is tested using protons or Co-60 gamma rays.

The rest of this paper describes an extension of the “careful COTS” principle to beyond-LEO missions, and their associated radiation environments. We first present a set of reference orbits for analysis corresponding to multiple regions of interest (LEO, GTO, GEO, and interplanetary space), followed by a brief background on the deep-space radiation environment. We then introduce the proposed radiation mitigation approach using a worked example of a one-year reference mission in lunar orbit, where the proposed methodology in this paper is combined with a set of reference design choices to provide a radiation-robust system example.

Reference Operating Orbits and Environments

Rather than tie the analysis to a particular mission, this paper considers a “generic” small deep-space mission. In this concept, the spacecraft is launched as a secondary payload to LEO, where it undergoes initial commissioning. Following the (brief) LEO phase, an impulsive maneuver is used to transfer the spacecraft into a high (GTO) orbit or comparable highly-eccentric Earth orbit for initial phasing. After a period in GTO, a second impulse is used to transfer into GEO for further operations, or direct injection into near-Earth interplanetary space (representative of missions to cis-lunar space or to a near-Earth asteroid), where is fulfills its nominal mission.

We normalize radiation exposure in each mission phase to a one-year duration, although it is unlikely that a real mission would spend that long in some of these phases: the GTO environment, particularly, is extremely hostile from a TID standpoint, and any transits through the Van Allen belts should ideally be performed as quickly as possible.

The radiation environment (see below) experiences a temporal variation, which is correlated with the 11-year solar activity cycle. We present results during two representative years: 2019 represents a solar minimum, and 2023 represents a solar maximum. Lastly, while the focus of this analysis has been for the environments noted above, the same methodology can be applied to missions further or closer to the sun (i.e. Mars or Venus, respectively).

Radiation Environment and Models

The radiation environment which is the subject of this paper consists of streams of high-energy particles (electrons, protons, and positively-charged ions), which have several different sources. Before discussing the nature of this radiation, it is important to review some basic definitions; the reader is referred to [2] for more details. The fundamental unit of measurement of this kind of radiation is differential flux, \( \frac{d\Phi}{dE} \), which is defined as the number of particles of a specific type which pass through an area during an interval of time; units of \( \text{cm}^{-2} \text{s}^{-1} \text{MeV}^{-1} \) are typically used\(^1\). By integrating the differential flux over a time, we can compute the differential fluence \( \frac{dF}{dE} \), which is the total number of particles of a specified energy that pass through an area during the interval. The units of differential fluence are \( \text{cm}^{-2} \text{MeV}^{-1} \). We can also define an integral fluence \( F(>E) \), which is the number of particles with energy greater than a threshold that pass through an area (with typical units \( \text{cm}^2 \)):

\[
F(>E) = \int_E^\infty \frac{dF}{dE} dE
\]

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\(^1\) Some models include a directional component in fluxes and fluences, which is apparent by the inclusion of a sr\(^{-1}\) in the units. We follow the guidelines of [3] and assume that fluxes are omni-directional; in that case, a multiplication by \( 4\pi \text{sr} \) is necessary.
Depending on the exact orbit, a spacecraft in deep space will encounter [2]:

1. trapped radiation
2. solar energetic particles (SEP)
3. galactic cosmic rays (GCR)

Trapped radiation refers to radiation which is present near the Earth (trapped by the interaction with the geomagnetic field), and consists of both high-energy protons and electrons. The electrons of the sort that are found in the near-Earth environment are not particularly penetrating, and their effect is principally limited to solar cells, which are necessarily mounted exterior to the spacecraft. However, any substantive shielding – such as that provided by even lightweight spacecraft structure – will absorb electrons before they have a chance to interact with electronic components. As such, they will not be discussed further. Trapped radiation is relevant to spacecraft in medium-Earth orbit (MEO) and high-Earth orbit (HEO), and some low-energy protons will also be seen in GEO.

Solar energetic particles are high-energy protons and ions that are emitted by the Sun, accelerated by the solar magnetic field, and travel along field lines from the Sun outwards into the solar system. The Sun normally emits a very low flux of SEPs; however, when a solar flare occurs, very large numbers of SEPs can be emitted. The flux is dominated by protons; the flux of other ions is typically two orders of magnitude lower at low energies and drops off rapidly with increasing energy. SEPs are seen in MEO and higher orbits, as the Earth’s magnetosphere provides shielding against them in LEO.

Galactic cosmic rays are high-energy positively charged ions of interstellar origin. The flux is again dominated by protons, with helium ions being an order of magnitude less common, and other species being even rarer. As mentioned above, the radiation environment is dependent on the solar activity cycle. During times of solar minimum, the Sun is quiet with few solar flares, and the SEP flux is extremely low. At the same time, the flux of GCRs increases. Conversely, during solar maximum, solar flares occur at unpredictable intervals, generating intense bursts of SEPs, while GCR flux drops. Figure 2 and Figure 3 show the variation in GCR fluence from solar minimum to solar maximum in deep space. As can be seen, while the integral fluence varies only by a factor of two, the differential fluence spectrum is substantially different. During times of solar minimum, the peak fluence is not only much higher, but it is at a lower energy. Since lower energy particles interact with silicon more strongly than high energy particles, this indicates that GCRs will play a more important role during solar minima.

There are a number of models which can be used to predict the radiation environment for a mission. These are integrated into an online tool called SPENVIS [4], which allows a user to define the spacecraft orbit, and to compute unshielded and shielded fluences. This paper used the AP-8 model for trapped radiation, the ESP-PSYCHIC model for SEP, and the ISO 15390 model for GCRs. ESP-PSYCHIC and ISO 15390 fluences are computed for a specified confidence level – this is the probability that the actual fluence will not exceed the model output. This can be used as a design constraint when analyzing the mission.

Figure 1 shows the one-year shielded proton fluence, broken down by particle source, for both a GTO and deep space orbit (the GEO environment is extremely similar to the deep space environment, and has been omitted for clarity). As can be seen, trapped radiation dominates GTO environment. In the deep space environment, the radiation is comprised mostly of solar particles below energies of approximately 100-200 MeV. The high-energy portion of the radiation spectrum is composed of GCRs.
Figure 1: Shielded proton fluence in different environments at solar maximum. Solar particles are at 95% confidence; GCRs are ISO 15390 mean values. Assumes 2.54 mm aluminum shielding.

Figure 2: Integral shielded GCR proton fluences in deep space during solar activity extremes.
Methodology

This paper concerns itself with potentially mission-ending single-event effects. SEEs occur randomly, and are typically characterized by an upset cross-section $\sigma(LET)$ with units of cm$^2$ (for digital memory components, $\sigma$ is often expressed on a per-bit basis, in which case it has units of cm$^2$ bit$^{-1}$), which is computed as

$$\sigma(LET) = \frac{n_{\text{SEE}}}{F}$$

In general, electronic components have a threshold LET, below which the upset cross-section is either zero or too small to measure. Above this threshold, the cross-section rises with increasing LET, until a saturation LET is reached, above which the cross-section remains essentially constant.

Traditionally, cross-section testing is performed using heavy ions. However, high-energy protons can also be used for testing [5] [6]. Protons interact with electronic components principally via elastic and inelastic collisions [7]. In the former case, nuclei of silicon are displaced from the lattice, and deposit their kinetic energy in the device, while in the latter case, the protons are absorbed by the silicon nucleus. The nucleus then fissions into daughter particles whose mass ranges between that of helium and phosphorus. These daughter particles then deposit their kinetic energy in the lattice, potentially causing upsets. Prior work has shown [6] [8] that inelastic proton interactions can produce LETs up to 16 MeV cm$^2$ mg$^{-1}$, though LETs in the range of 10-14 MeV cm$^2$ mg$^{-1}$ are most common in silicon.

However, modern high-gate count electronic piece-parts are not just silicon, and considering their additional materials gives interesting results. Modern COTS components increasingly employ high-Z materials (Cu, Ti, W) in their construction—almost all present-day high-density ICs use W plugs and Cu metallization to improve speed, and these high-Z materials produce much higher-energy nuclear scattering and secondary particles than silicon [9]: for example, the maximum LET for 500 MeV proton collisions with W is ~ 34 MeV cm$^2$ mg$^{-1}$. Lower energy (200 MeV) protons into W produce particles with LET of up to 30 MeV cm$^2$ mg$^{-1}$, while interactions with Cu will result in 25 MeV cm$^2$ mg$^{-1}$ LET.
This suggests a new approach to radiation testing for deep space missions. By testing parts with high-energy protons, data can be obtained on SEE cross-sections across a relatively wide range of the expected LET spectrum. Then, for LETs higher than those obtained via proton testing, an analytic technique is available to bound the probability of an SEE not revealed by testing. This hinges on an observation that the upset cross-section of a device cannot be larger than the physical size of the device itself. In the absence of other data, a conservative estimate of the upset cross-section at high LET can therefore be taken as the area of the device’s active area\(^2\) and combined with the proton testing results to form a complete (and conservative) total picture of device cross-section.

Once the total area of a design is known (by summing the areas of the individual components), the overall system worst-case SEE susceptibility can be estimated by using predicted LET spectra for the mission (or mission segment). The LET spectrum is derived from the expected radiation environment, as shielded by the spacecraft structure. The integral LET spectrum, evaluated at a threshold LET (for example, 20 MeV cm\(^2\) mg\(^{-1}\)), can then be multiplied by the design area to determine what the expected number of SEEs will be.

This analysis can be used to place a confidence bound on the number of upsets by using the correct radiation spectrum model. As described above, both the SEP and GCR models present in SPENVIS incorporate confidence bounds; unfortunately, the trapped radiation models do not, though those are only applicable to HEO.

As an example, Table 1 (excerpted from the Appendix) shows the expected integral LET fluences at 14 and 30 MeV cm\(^2\) mg\(^{-1}\) for a number of mission segments, at the 95% confidence level. As can be seen, the quiet-Sun fluences are approximately 3.6 times lower than during active-Sun times. Since SEPs are generated in large quantities only during solar flares, it is useful to examine what proportion of the active-Sun fluence is due to SEPs. This information is also shown in Table 1. As can be seen, the vast majority of high-LET fluence is due to solar flares, and there are very few potential SEE-causing particles when no active flares are present. This implies that a very large fraction of the risk in high orbits and deep space can be mitigated with a robust protection measure against solar flares.

Table 1 illustrates another striking result: that if we can confidently assert that high-energy proton testing uncovers SEE to LET approaching 30 MeV cm\(^2\) mg\(^{-1}\), the remaining fluence at higher LET is sufficiently small that not even the radiation environment analysis need necessarily be performed. Or said differently, it can be argued that for high-Z components that produce substantial secondary high-energy particle spectra, a sufficient amount of the integral fluence expected for interplanetary missions is adequately covered for such devices via radiation testing.

### Table 1: Yearly fluences (cm\(^2\)) (95% confidence) at LET > 14 and 30 MeV cm\(^2\) g\(^{-1}\) with 2.5 mm aluminum shielding. Note that 2019 is solar minimum, and 2023 is solar maximum.

<table>
<thead>
<tr>
<th>Orbit</th>
<th>Year</th>
<th>Fluence &gt; LET 14</th>
<th>Non-SEP fluence &gt; LET 14</th>
<th>Fluence &gt; LET 30</th>
<th>Non-SEP fluence &gt; LET 30</th>
</tr>
</thead>
<tbody>
<tr>
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<td>2019</td>
<td>154</td>
<td>154</td>
<td>5</td>
<td>3</td>
</tr>
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<td>Deep Space</td>
<td>2019</td>
<td>226</td>
<td>226</td>
<td>4</td>
<td>4</td>
</tr>
<tr>
<td>GTO</td>
<td>2023</td>
<td>560</td>
<td>22</td>
<td>3</td>
<td>1</td>
</tr>
<tr>
<td>Deep Space</td>
<td>2023</td>
<td>826</td>
<td>32</td>
<td>4</td>
<td>1</td>
</tr>
</tbody>
</table>

### Worked Example: 1 Year Lunar Mission using *Deep Space Xplorer* Platform

With the above methodology established, in this section we analyze an example lunar mission. In this reference case, the *Xplorer* low-cost microspacecraft developed by Deep Space Industries (DSI) is used. The avionics suite for this mission segment has been developed by Deep Space Industries (DSI). The avionics suite for this mission segment has been developed by Bonin and Stras.

\(^2\) The device area can be estimated using a number of sources: radiographic (X-ray) imaging can show the physical silicon die inside plastic microcircuits with good accuracy. It is possible to decapsulate components, although this can involve the use of hazardous chemicals. Finally, measurements of the physical package can be made, though this is most accurate for “chip-scale” packages.
spacecraft has been developed, tested and analyzed jointly by Deep Space Industries and Canadensys Aerospace Corporation using the methods outlined in this paper.

**Xplorer Overview**

The Xplorer platform (Figure 4, [10]) is designed to execute missions lasting 1-3 years in interplanetary space, with all system-level margins and performance calculated for a three-year end-of-life. Deep Space Xplorer is an approximately 300 kg, ~1 m x 1 m x 1 m microspacecraft with more than 5 km/s of delta-V capacity available using a proprietary green bipropellant system. The overall form-factor is designed to be compatible with commercial rideshare launches to LEO or GTO using an ESPA-Grande secondary payload adapter, though implementations requiring less than its full delta-V capability can realize lower launch mass and/or increased payload capacity.

A number of measures are employed in Xplorer to provide radiation robustness, not the least of which is the bulk shielding offered by housing most avionics in its central structure behind its liquid fuel and oxidizer tanks. All piece-parts are de-rated to at least ECSS standards or further (for example, power MOSFETs are never operated above 20% rated drain-to-source voltages, empirically eliminating the risk of SEB or SEGR). At a unit level, every spacecraft load is protected from catastrophic single-event latchup using fast-acting current-limiting resettable solid-state switches. Additionally, all volatile memory and registers on computers and embedded devices is protected from SEUs using TMR error detection and correction, and on most units, memory scrubbing also records discrepancies to provide actual SEU measurements. Additional robustness is given by the use of cold spare redundancy for critical components (i.e. on-board computers), which cannot latch up when unpowered, and which can be brought into service in the event of an unrecoverable hard fault in the powered unit. Finally, hardware decoded commands are available to power cycle all spacecraft systems in the rare event of an unresponsive OBC or power system.
For radiation tolerance, we employ the methodology described in earlier sections. In this approach, all vulnerable electronics are tested to a minimum total dose of 20 krad(Si) using energetic (480 MeV) proton testing, which also provides their empirical single event effects (SEE) cross-section. While proton testing does not provide Linear Energy Transfer (LET) up to maximum levels anticipated for interplanetary space (~37 MeV cm$^2$ mg$^{-1}$), it does uncover SEE out to LET approaching 20 to 25 MeV cm$^2$ mg$^{-1}$ (particularly for modern high gate count semiconductor devices commonly employing high-Z Cu, Ti and W materials, which yield energetic nuclear scattering). above which fluence is typically small.

Figure 6 presents integral flux versus LET for an interplanetary mission at 1 AU solar distance, generated using SPENVIS. Proton testing provides empirical SEE cross-section over the majority of LET with high flux. To assess the risk at higher LET and lower flux, we integrate over mission life to determine the expected fluence, noting that the SEE cross section of any particular device cannot exceed its physical size, as noted earlier. In Figure 6, fluence per year$^3$ of operation exceeding LET covered by proton testing is approximately 4.1e4 m$^{-2}$ sr$^{-1}$, or about 0.5/mm$^2$/year. or put a different way, a “typical” 10x10 mm integrated circuit may expect to see only one high-LET particle per week at worst, allowing an upper limit to be placed on SEE probability not directly tested using heavy ions$^4$.

![Figure 6: Linear Energy Transfer (LET) Spectra vs. Integral Flux for 1AU Interplanetary Reference Case.](image)

Figure 6: Linear Energy Transfer (LET) Spectra vs. Integral Flux for 1AU Interplanetary Reference Case. 2.5 mm Al equivalent shielding assumed.

Similar arguments may be used to assess susceptibility to solar flares—in the case of a flare in the reference case above, SPENVIS provides a peak integral flux of 3.7 m$^2$ sr$^{-1}$ s$^{-1}$, which over a one-day event results in approximately 4 particles/mm$^2$ fluence. Again, this is the maximum cross-section any device can have, providing an upper limit on SEE vulnerability.

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$^3$ Figure 9 assumes 2.5 mm Al equivalent shielding. No sectoring analysis is performed orientations.

$^4$ Assumes fluence is the cross section (i.e. that 100% of particles trigger an SEE), which is a very conservative assumption.
In the event of a solar flare, Xplorer also incorporates a unique power mode we call “bunker mode”. In this mode, the entire spacecraft except for a small circuit in the power system is shut down and held in an unpowered state for several hours to protect against solar flares. The bunker mode state and duration is trigger by a hardware decoded ground command. This eliminates the risk of catastrophic latch-up and reduces the impact of TID and displacement damage associated with the event, provided early warning is available. While for “typical” doses the empirical plus analytical approach in this paper may be used, for SEP events it is highly recommended that additional precautions are adopted.

In summary: a combination of testing, consideration of the expected operating environment, and design robustness allow a radiation tolerance approach that has a large fraction of the reliability achieved with conventional approaches, but at a much smaller overall cost and price.

Additional Design Guidelines

A number of additional options are of course available to the designer that can supplement (or supplant) the proposed methods in this paper. One option available to the designer is to add shielding around sensitive or critical electronics. As radiation is transported through the spacecraft structure, it loses energy, and some of it is absorbed. The result of this is that the particle’s effective LET is reduced. With reference to Table 2 in the Appendix, we can see that providing 7.6 mm (total) of aluminium shielding is very effective at reducing the fluence of high-LET solar and trapped particles, reducing it by approximately an order of magnitude. As a more viable alternative, electronics can be surrounded by spacecraft propellant tanks, as we do with Xplorer, to achieve equivalent or greater shielding about most of the whole sphere. However, shielding is unfortunately not effective against the very high energy GCRs. This can be seen in Figure 7 and Figure 8, which show SEP and GCR spectra as a function of shielding thickness. Due to the lower energies of GCRs, shielding against them is somewhat more effective during times of solar minimum.

![Figure 7: SEP fluences for a deep-space mission as a function of shielding thickness.](image-url)
If the fluence reduction offered by shielding is not sufficient, or the mass required for the shielding is not available, the designer can also try to reduce the effective area of the sensitive electronics. With reference to Table 1, we can see that a substantial portion of the high-LET fluence is due to solar-flare-generated particles. While flares are, in general, not predictable, they can be observed as they occur at the Sun, and their propagation to the spacecraft can be computed. By shutting down as much of the spacecraft as possible during the time that flare ejecta pass through the spacecraft, the probability of an upset can be greatly reduced. We have demonstrated a circuit with an effective area of less than 1 mm$^2$ which is capable of shutting a small spacecraft down, and then powering it back on after the flare has passed, at the cost of some overall spacecraft availability.

Finally, recent trends in microelectronics are such that the probability of a catastrophic component failure due to latchup are dropping [11] [12]. As supply voltages in modern electronics drop to below 1.4 V – the level required to sustain a latchup – heavy-ion test data on commercial electronics show threshold latchup LETs in excess of 70 MeV cm$^2$ g$^{-1}$, which are essentially indicators that the devices are latchup-immune. It is dangerous to extrapolate to untested devices, of course, but with proper mission, system, and subsystem design, the designer should be able to isolate and recover from any SEEs.

Conclusions

The analysis method presented in this paper allows a designer to quantitatively trade off mission risk against cost for small deep-space missions. By reducing or eliminating the need for expensive heavy ion testing, and substituting cheaper proton testing, we can lower the cost of mission assurance, while still being able to bound the risk. This allows for intelligent allocation of resources, and also simplifies the process of insuring the mission as needed.

Key observations and lessons from this paper are summarized as follows:
1. There is no substitute for understanding the expected operating environment of a space mission—for a large fraction of missions, proton testing is sufficient to assess device cross-section, with higher-energy fluences being very small
2. For high-Z piece-parts, proton testing may be deemed adequate even for higher LET integral fluence and flux environments
3. The cross-section of a given electronics piece-part cannot be larger than the physical size of the device; therefore, if that limiting cross section is sufficiently small, the device can be considered to have negligible SEE risk
4. Additional design measures (low-power solar flare modes, functional and actual redundancy, smart and selective shielding) can be introduced for relatively low-cost or added complexity, to the benefit of radiation robustness
5. A very large fraction of the risk in high orbits and deep space can be mitigated with robust protection measures against solar flares.

Appendix

This section contains a summary of one-year fluences, computed at a 95% confidence interval, for the various orbits and mission dates discussed elsewhere in this paper. The data is provided to allow the designer to quickly determine SEE rates. For missions which spend their entire lifetimes in the given orbits, the yearly fluence is as shown. If the spacecraft performs manoeuvres, then a piecewise scheme should be used, with fluences scaled according to the time spent in each segment.

The “GTO” orbit is a 300 x 36000 km (altitude) equatorial orbit. The “deep space” orbit is a near-Earth orbit at 1 AU from the Sun.

Table 2: Yearly fluences for a selection of orbits and shielding thicknesses. LETs are in MeV cm$^2$ g$^{-1}$. Numbers in parentheses are fluences excluding SEPs.

<table>
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<th>Orbit</th>
<th>Year</th>
<th>Shielding [mm Al]</th>
<th>F(&gt; LET 20) [cm$^2$]</th>
<th>F(&gt; LET 30) [cm$^2$]</th>
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<td></td>
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<td></td>
<td></td>
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