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**National Aeronautics and  
Space Administration**

## **NASA Safety Standard**

# **Guidelines and Assessment Procedures for Limiting Orbital Debris**

Office of Safety and Mission Assurance  
Washington, D.C. 20546

NASA SAFETY STANDARD  
GUIDELINES AND ASSESSMENT PROCEDURES FOR LIMITING  
ORBITAL DEBRIS

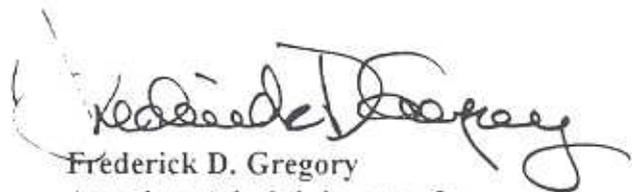
PREFACE

EFFECTIVE DATE: AUG 1 1995

Collision with orbital debris is a hazard of growing concern as historically accepted practices and procedures have allowed man-made objects to accumulate in orbit. To limit future debris generation, NASA Management Instruction (NMI) 1700.8, "Policy to Limit Orbital Debris Generation," was issued in April of 1993. The NMI requires each program to conduct a formal assessment of the potential to generate orbital debris. This standard serves as a companion to NMI 1700.08 and provides each NASA program with specific guidelines and assessment methods to assure compliance with the NMI.

Each main debris assessment issue (e.g., Post Mission Disposal) is developed in a separate chapter. For the reader who needs just an overview of the debris issues, consult the guideline descriptions in chapter 2.

The standard was developed jointly by the Office of Safety and Mission Assurance (Code Q) and the Johnson Space Center Space Physics Branch. Comments, questions, or suggestions concerning this document should be directed to Code QS.



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# 1. INTRODUCTION

## 1.1 SCOPE AND PURPOSE

This document serves as a companion to NASA Management Instruction (NMI) 1700.8, and provides specific guidelines and methods to comply with the NASA policy to limit orbital debris generation. The guidelines serve to help ensure that launch vehicles, upper stages, and payloads meet acceptable standards for limiting orbital debris generation. This document should be used by the program manager or project manager as the primary reference in conducting debris assessments. The standard establishes guidelines and provides supporting analysis tools for: (1) limiting the generation of orbital debris, (2) assessing the risk of collision with existing space debris, and (3) assessing the potential of spacecraft-generated debris fragments to impact the Earth's surface. In addition to guidelines and methods for assessment, this volume provides formats for the debris assessment reports. Two appendices are used to define frequently used terms and to provide summary background information.

Another document, entitled "Reference Manual for Orbital Debris Assessments," provides more in-depth background and technical information. In addition, debris assessment software is available to support the assessment of particular guidelines and to evaluate mitigation measures.

## 1.2 OVERVIEW OF NASA MANAGEMENT INSTRUCTION 1700.8

NASA Management Instruction (NMI) 1700.8 states "NASA's policy is to employ design and operations practices that limit the generation of orbital debris, consistent with mission requirements and cost-effectiveness." The NMI requires that each program or project conduct a formal assessment for the potential to generate orbital debris.

The debris assessment must address the potential for orbital debris generation that results from normal operations and malfunction conditions, and on-orbit collisions. The assessment must also address provisions for postmission disposal. Malfunction conditions refer to those credible failure scenarios or conditions that can result in the direct generation of orbital debris or that can disable the spacecraft to preclude postmission disposal. Examples of orbital debris generated during normal operations include items such as lens covers, shrouds, and staging components that are released into the environment. An on-orbit explosion is an example of debris generation by malfunction. Examples of debris generation by collisions include immediate debris generation by collisions with large objects and by loss of control of a spacecraft or payload as a result of impact with small debris during mission operations.

To satisfy the NMI, the program or project manager may need to plan for such things as:

- Depleting on-board energy sources after completion of mission
- Limiting orbit lifetime after mission completion to 25 years or maneuvering to a disposal orbit
- Limiting the generation of debris associated with normal space operations
- Limiting the consequences of impact with existing orbital debris or meteoroids
- Limiting the risk from space system components surviving reentry as a result of postmission disposal

## 1.3 OBJECTIVES OF DEBRIS ASSESSMENTS

Each program or project should attempt to meet all pertinent guidelines. It is understood, however, that satisfying these guidelines must be balanced with the necessity to meet mission requirements and to

control costs. If a guideline cannot be met because of over-riding conflict with mission requirements or prohibitive cost impact, this should be specifically noted in the assessment with rationale and justification provided.

As a matter of practice, it is desirable for the program or project to work with the Office of Safety and Mission Assurance during the assessment process. Ideally, the program or project should also use the expertise at NASA centers. The Office of Safety and Mission Assurance at each center can direct programs/projects to groups that can provide assistance with debris assessments. These groups have resources for analyzing complex debris problems which may not be covered in the detail necessary in this standard or in the debris assessment software.

## 2. CONDUCTING THE DEBRIS ASSESSMENT: AN OVERVIEW

This section provides an overview of what should be covered by a debris assessment. The detailed guidelines and evaluation methods for specific assessment issues are presented in chapters 3 through 7.

As required by NMI 1700.8, "Policy to Limit Orbital Debris Generation," the debris assessment covers two broad areas: the potential for generating debris during normal operations or malfunction conditions, and the potential for generating debris by collision with space debris (natural or human-generated) or orbiting space systems. These two broad areas are broken down into five issues to be addressed in the assessment:

- Debris released during normal operations
- Debris generated by explosions and intentional breakups
- Debris generated by on-orbit collisions during mission operations
- Safe disposal of space systems after mission completion
- Structural components impacting the Earth following postmission disposal by atmospheric reentry

The assessment will be organized around these issues with specific guidelines associated with each. The objective for each program or project is to assess whether all applicable guidelines have been met.

### 2.1 STRUCTURE OF THE GUIDELINES DOCUMENT

Each of the five assessment issues is covered by a chapter in this guidelines document. Each issue is addressed by specific "guideline" statements. Each guideline statement appears in a bold box at the beginning of a chapter or major section of a chapter.

After the box containing the guidelines, the following sections are presented:

- "Rationale for Guidelines"
- "Method to Assess Compliance with the Guidelines"
- "Brief Overview of Debris Mitigation Measures"

These sections provide information on the reasoning behind the guidelines, a recommended approach for assessing whether a program has met a given guideline, and what kinds of modifications in design or procedures might be considered to bring a program within guidelines. An overview of assessment issues and associated guidelines is presented in table 2-1.

Mitigation measures, analysis support procedures, and technical background are presented in more detail in a companion document, "Reference Manual for Orbital Debris Assessments." Volume I of this document is "Assessment of Debris Mitigation Procedures." Additional technical information is presented in Volume II, "Technical Background for Assessing Orbital Debris Risk." Each volume is organized around the guideline areas used in this standard.

### 2.2 PERFORMING DEBRIS ASSESSMENTS

Each program should address the applicable guidelines in each of the four areas of normal operations, accidental explosions or intentional breakups, debris collision, and postmission disposal. If atmospheric reentry is used as the postmission disposal option (Section 6), the program should address the guideline on reentry risk contained in Section 7. The guidance provided in the sections titled "Method to Assess

**Table 2-1. Debris Assessment Issues and Corresponding Guideline Descriptions**

<b>Debris Assessment Areas</b>	<b>Guideline Description</b>	<b>Comments</b>	<b>Chapter</b>
Release of debris during normal mission operations	<ul style="list-style-type: none"> <li>• Limit number, size, and orbit lifetime of debris larger than 1 mm</li> <li>• Limit lifetime of objects passing through GEO</li> </ul>	<p>Includes staging components, deployment hardware or other objects larger than 1 mm that are known to be released during normal operations</p> <p>Tethers or tether fragments left in orbit are considered operational debris</p>	3
Accidental explosions	<ul style="list-style-type: none"> <li>• Limit probability of accidental explosion during mission operations</li> <li>• Deplete on-board stored energy at end of mission life</li> </ul>	Includes systems and components such as range safety systems, pressurized volumes, bipropellant fuels, and batteries	4.1
Intentional breakups	<ul style="list-style-type: none"> <li>• Limit number, size, and orbit lifetime of debris larger than 1 mm</li> <li>• Assess risk to other programs for times immediately after a test when the debris cloud contains regions of high debris density</li> <li>• No assessment of orbital hazard for breakups occurring below altitude 90 km</li> </ul>	Intentional breakups include tests involving collisions or explosions of flight systems and intentional breakup during space system reentry to reduce the amount of debris reaching the ground	4.2
Collisions with large objects during mission operations	Assess probability of collision with intact space systems or large debris	Collisions with intact space systems or large debris will create a large number of debris fragments that pose a risk to other operating spacecraft. A significant probability of collision may necessitate design or operational changes	5
Collisions with small debris during mission operations	Assess and limit the probability of damage to critical components as a result of impact with small debris	Damage by small debris can result in both mission failure and failure to perform postmission disposal. A significant probability of damage may necessitate shielding, use of redundant systems, or other design or operational modifications	5
Postmission disposal	Remove spacecraft and upper stages from high value regions of space so they will not threaten future space operations	Options are to transfer to a disposal orbit or transfer to an orbit where the space system will reenter within 25 years. Disposal orbits are defined away from LEO, GEO, and semisynchronous (12 hour) circular orbit	6
Debris surviving reentry and impacting in populated areas	Limit number and size of debris fragments that survive uncontrolled reentry	This guideline limits human casualty expectation	7

Compliance with the Guidelines" should be sufficient to carry out the assessment. The models in the debris assessment software support the approach and techniques described in this section. Each program is free to use alternative methods or models that they feel are more suitable for their particular program. If alternative methods or models are used, the program should document such methods or models in the debris assessment report.

Two assessment reports should be completed. The first is prepared at PDR and the second 45 days prior to CDR. The PDR debris assessment should identify debris generation issues and , where possible, assess those issues. The CDR assessment should identify, assess, and resolve all debris issues in detail. Chapter 8 of this document provides the specific information that should be included in a debris assessment report.

Although assessments are prepared only at PDR and prior to CDR, it is advisable for each program or project to consider potential debris issues during concept development (Phase A) and development of preliminary requirements, specifications, and designs (Phase B) to estimate and minimize potential cost impacts.



### **3. ASSESSMENT OF DEBRIS RELEASED DURING NORMAL OPERATIONS**

Debris is often released as an incidental part of normal space operations; this type of debris is referred to as operational debris. Since the release can be planned, it can be done in a manner that does not pose a significant risk to other users of space. Small debris—1 mm in diameter (about 1 mg) and larger for LEO, or 5 cm (about 100 gm) and larger for GEO—is a source of concern because it has enough energy to critically damage an operating spacecraft. Larger debris might collide with other large objects in the environment and create clouds of secondary debris fragments.

The probability of a collision occurring with debris released during normal operations depends on the number and size of the debris and on length of time it remains in orbit. The guidelines, therefore, limit the total number of such debris and their orbit lifetimes. Debris released during normal operations includes debris released during staging and payload separation, deployment, and mission operations; tethers and tether fragments left in orbit at the end of mission are also considered operational debris. Upper stages, payloads, and solid rocket motor debris are not covered by these guidelines.

# GENERAL POLICY OBJECTIVE

## CONTROL OF DEBRIS RELEASED DURING NORMAL OPERATIONS

NASA programs and projects will assess and limit the amount of debris released in a planned manner during normal operations.

## GUIDELINES

3-1. *Operational debris passing through LEO:* For operations leaving debris in orbits passing through LEO, the total amount of debris of diameter 1 mm and larger released during normal operations should satisfy two conditions:

- a. The total area-time product should be no larger than  $0.1 \text{ m}^2\text{-yr}$ . The area-time product is the sum over all operational debris of the debris cross-sectional area multiplied by the total time spent below 2000 km altitude during the orbit lifetime of each debris object.
- b. The total object-time product should be no larger than 100 object-yr. The object-time product is the sum over all operational debris of the total time spent below 2000 km altitude during the orbit lifetime of each debris object.

Note: Tethers and tether fragments are considered operational debris if left in the environment after mission completion.

3-2. *Operational debris passing through GEO:* For operations leaving debris in orbits passing within 300 km GEO altitude, debris of diameter greater than 5 cm will be left in orbit only if it has a perigee altitude low enough that atmospheric drag will lower its apogee altitude to be no higher than 300 km below GEO altitude within 25 years.

## Rationale for Guidelines

For guideline 3-1a, the value of  $0.1 \text{ m}^2\text{-yr}$  is based on the principle that large debris released during normal operations should represent a much smaller risk of collision with other large objects in orbit than do operating spacecraft. Historically, spacecraft have had an average cross-sectional area of about  $10 \text{ m}^2$  and an operational lifetime of three years, giving the average operating spacecraft a  $30 \text{ m}^2\text{-yr}$  area-time product. Adopting a guideline value between  $0.1$  and  $1.0 \text{ m}^2\text{-yr}$  would therefore yield a probability of collision that is  $\sim 1/30$  to  $1/300$  the collision probability represented by an average operating spacecraft. A value of  $0.1$  is used based on mass considerations. Typical operational debris of mass 1 kg and larger, which is large enough to cause complete collisional fragmentation of intact payloads or upper stages, will have an orbit lifetime  $\sim 10$  years under the  $0.1 \text{ m}^2\text{-yr}$  guideline value. If a  $1.0 \text{ m}^2\text{-yr}$  value was chosen, such operational debris would have a lifetime of  $\sim 100$  years, and therefore accumulate in orbit.

Guideline 3-1b limits the total number of debris objects released based on their orbit lifetimes. Based on historical precedent and practice, an acceptable level of risk for released debris damaging another operational spacecraft is  $<10^{-6}$ . The guideline value of 100 object-yr was chosen because debris released

during normal operations following this guideline will have probability on the order of  $10^{-6}$  of hitting and potentially damaging an average operating spacecraft.

Tethers present a much greater risk to operating spacecraft than would be expected from their mass and cross-sectional area, and may have a high probability of being severed and thus left in the environment. Consequently, tethers or tether fragments left in orbit after completion of mission require special consideration for the risk they pose to operating spacecraft.

Debris that is not removed from GEO altitude by energy losses resulting from atmospheric drag, a process requiring a low perigee altitude, will remain in the GEO environment for many thousands of years. Therefore, guideline 3-2 limits the accumulation of debris at GEO altitudes and will prevent the development of a significant debris environment, as currently exists in LEO.

## Method to Assess Compliance with the Guidelines

### Debris Passing Through Low Earth Orbit (Guideline 3-1)

For each of the debris objects released during normal operations, calculate the cross-sectional area and orbit lifetime. The total area-time product (guideline 3-1a) will be

$$\sum_{\substack{\text{debris larger} \\ \text{than 1 mm}}} (\text{cross-sectional area} \times \text{orbit dwell time below 2000 km})$$

“Orbit dwell time below 2000 km” is defined as the total time spent by an orbiting object below an altitude of 2000 km during its orbit lifetime. If the debris is in an orbit with apogee altitude below 2000 km, the orbit dwell time equals the orbit lifetime.

The total object-time product (guideline 3-1b) will be

$$\sum_{\substack{\text{debris larger} \\ \text{than 1 mm}}} (\text{orbit dwell time below 2000 km})$$

The following procedure is used to determine orbit dwell time:

1. Determine the average cross-sectional area, area-to-mass ratio, and initial orbit for each debris piece released. The average cross-sectional area is the cross-sectional area averaged over aspect. For simple convex debris, the average cross-sectional area is 1/4 the surface area. For highly irregular debris shapes an estimate of the average cross-sectional area may be obtained as follows: determine the view,  $V$ , that yields the maximum cross-sectional area and denote the cross-sectional area as  $A_{\max}$ . Let  $A_1$  and  $A_2$  be the cross-sectional areas for the two viewing directions orthogonal to  $V$ . Then define the average cross-sectional area as  $(A_{\max} + A_1 + A_2) / 2$ . Measure all areas in square meters.

The area-to-mass ratio for the debris object is the average cross-sectional area ( $m^2$ ) divided by the mass (kg).

The initial debris orbit is the orbit of the object releasing the debris unless the release occurs with  $\Delta v$  greater than a few meters per second. For debris released with  $\Delta v$  greater than  $\sim 10$  meters per

second, the initial debris orbit may be significantly different from that of the object releasing the debris. The debris assessment software can be used to calculate the initial orbit in this case.

2. Calculate the orbit dwell time for each debris piece released. Either the procedures described in this section or the debris assessment software may be used. If the debris assessment software is used, the solar activity should be set to 130 solar flux units (sfu).

For each debris object:

- (1) Locate the apogee and perigee altitude for the orbit in figure 3-1 for low altitude near-circular orbits or figure 3-2 for highly eccentric orbits. The orbit will generally fall between two area-to-mass ratio contours, referred to as the bounding contours.
- (2) For each debris object not having the reference area-to-mass value of  $0.01 \text{ m}^2/\text{kg}$ , divide the bounding contour values by

$$100 \times (\text{the area-to-mass ratio for the debris object})$$

The modified contour values are the orbit lifetime contour values (figure 3-1) or the orbit dwell time below 2000 km contour values (figure 3-2) for the area-to-mass ratio of that debris object.

- (3) Interpolate along a vertical line between the bounding contour values to get the estimated orbit lifetime or orbit dwell time below 2000 km.
3. For guideline 3-1a, multiply each orbit lifetime or orbit dwell time below 2000 km by the cross-sectional area for that debris object and sum the area-time product over all debris released. For guideline 3-1b, sum the orbit lifetimes, or orbit dwell times below 2000 km, over all debris released. Compare the final sums with the guideline.

The debris assessment software can also be used to perform steps 1-3.

### **Special Consideration for Tethers or Tether Fragments Left in Orbit After Completion of Mission (Note to Guideline 3-1)**

For tethers, which are smaller in two dimensions but orders of magnitude larger in the third dimension than normal space structures, the potential to damage operating spacecraft is much larger than would be expected from the tether mass and cross-sectional area. Consequently, programs using tethers must take extra measures to control the potential for damaging other systems. To limit this type of risk to other users of space, tethers left in orbit after completion of mission or tether fragments created when meteoroids or orbital debris sever the tether are considered operational debris.

To limit the risk presented by the tether debris to operating spacecraft, the maximum tether debris length will be limited by its orbit lifetime. The lifetime must take into account any associated end-point vehicles attached to the tether. The relationship between the maximum allowed tether debris length,  $L_{\text{MAX}}$ , measured in kilometers, and orbit lifetime,  $T$ , measured in years, is

$$L_{MAX} \text{ [km]} = 1 / T \text{ [yr]}$$

The orbit lifetime of the tether system will be determined following the same procedure as for evaluation of operational debris for guideline 3-1. In calculating the area-to-mass ratio for orbit lifetime calculations, the average cross-sectional area of the tether system will be the cross-sectional area of the tether measured along its length,  $A_T$  (calculated in step 1 below), plus the cross-sectional area of any attached end-point vehicle(s). The mass of the tether system will either be the tether mass, if the tether is detached from the end-point vehicle(s), or the tether mass plus any attached end-point vehicle(s) mass. End-point vehicle mass and cross-sectional area are considered only if the mission plan calls for the tether to remain attached to the end-point vehicle(s). The effective altitude for the system for orbit lifetime calculations will be the altitude of the midpoint of the tether if it is not attached to an end-point vehicle or the altitude of the center of mass for the system if the tether remains attached to an end-point vehicle or vehicles.

The length of tether remaining in the environment after the end of mission, which will be the length of the tether if the mission plan is to leave the tether in the environment at the end of mission, or the length of tether that is expected to be cut off by meteoroid or orbital debris impact during the mission if the mission plan is to retract the tether at the end of mission, must be no larger than  $L_{MAX}$ .

The length of tether cut during the mission is related to the probability of the tether being cut. To calculate the probability of the tether being cut:

1. Calculate the cross-sectional area of the tether,  $A_T$ , from the tether diameter,  $D_T$ , in meters and length  $L$ , in meters:

$$A_T = D_T \times L \text{ [m}^2\text{]}$$

If the tether cross-section perpendicular to its length is not circular, calculate an effective diameter from the longest tether cross-sectional dimension,  $l_{max}$  and shortest cross-section dimension,  $l_{min}$ , measured in meters from

$$D_T = \sqrt{l_{max} \times l_{min}} \text{ [m]}$$

2. Calculate the minimum cutting impactor diameter for meteoroids or orbital debris. This will be the diameter of the smallest impactor that can cut the tether. Assume this diameter to be  $D_T / 5$ , unless it can be shown that either the tether design or tether materials will allow a larger minimum diameter.
3. Get the environment flux,  $F$ , for the minimum cutting impactor diameter using either figure 5-2 or 5-3.
4. Calculate the probability of the tether being cut,  $P_{CUT}$ , during mission time  $T$ , measured in years, by

$$P_{CUT} = 1 - e^{-FA_T T}$$

The expected length of tether that will be cut off during the mission,  $L_{CUT}$ , will be

$$L_{CUT} = P_{CUT} \times L$$

To satisfy guideline 3-1,  $L$  may not exceed  $L_{MAX}$ , if the mission plan is to abandon the tether in orbit;  $L_{CUT}$  may not exceed  $L_{MAX}$  if the mission plan is to retract the tether at the end of mission.

### **Debris Passing Near Geosynchronous Altitude (Guideline 3-2)**

This assessment determines the maximum perigee altitude for a debris object if it is to have an apogee altitude at least 300 km below GEO altitude after 25 years as stated in guideline 3-2. For this analysis use the following steps:

1. Determine the average cross-sectional area, area-to-mass ratio, and initial orbit for each debris piece released.

(See Step 1 for Debris Passing Through Low Earth Orbit.)

2. For each debris object determine the maximum initial perigee altitude the object can have for atmospheric drag to lower the apogee altitude to 300 km below GEO altitude in 25 years. Figure 3-3 may be used to determine this maximum perigee altitude, as follows:
  - 2a. Draw a vertical line at the initial apogee altitude of the debris orbit.
  - 2b. Pick the points where this line intersects the area-to-mass ratio contours on either side of the area-to-mass ratio of the debris object.
  - 2c. Linearly interpolate in area-to-mass ratio along the line between these points to find the point for the area-to-mass ratio for the debris object.
  - 2d. The perigee altitude corresponding to this point is the highest perigee altitude that the initial debris orbit may have if the apogee altitude is to be lowered to at least 300 km below GEO altitude within 25 years.

For example, assume the debris object has an initial apogee altitude of 1,600 km above GEO and an area-to-mass ratio of  $0.025 \text{ m}^2/\text{kg}$ . Interpolate along a vertical line for apogee altitude of 1,600 km above GEO between contours for  $0.01 \text{ m}^2/\text{kg}$  and  $0.03 \text{ m}^2/\text{kg}$  to locate the point  $3/4$  of the way from  $0.01$  to  $0.03 \text{ m}^2/\text{kg}$ . This orbit has an initial apogee altitude of 1,600 km above GEO and an initial perigee altitude of 400 km; in 25 years, with area-to-mass ratio of  $0.025 \text{ m}^2/\text{kg}$ , the orbit will have an apogee altitude 300 km below GEO altitude.

3. To fall within guidelines, the initial perigee altitude of the debris piece must be no higher than the perigee altitude determined by the interpolation procedure. In the example cited above, the operational debris object can have an initial perigee altitude no higher than 400 km.

## **Brief Summary of Debris Mitigation Measures**

If a program or project does not fall within guidelines, there are a number of mitigation measures that may be taken. These include:

1. Considering, if appropriate, orbit lifetimes for debris released near times of peak solar activity. This option can be investigated if the mission start time is well determined and falls within two years before a peak in the solar activity.
2. Releasing debris in orbits with lower perigee altitude or releasing debris with larger area-to-mass ratio to reduce orbit lifetime.
3. Releasing debris under conditions where lunar and solar perturbations will reduce lifetime.
4. Limiting release of debris by making design changes, changing operational procedures, or confining debris to prevent release into the environment.
5. The allowed length of tether debris can be maximized if the tether is detached from end-point vehicles at the end of mission. If the tether is cut away from the end-point vehicles at the end of mission, the end-point vehicles are treated as payloads or upper stages for disposal guidelines.

Measures to reduce orbit lifetimes are discussed in "Assessment of Debris Mitigation Measures," Volume I of the Reference Manual for Orbital Debris Assessments.

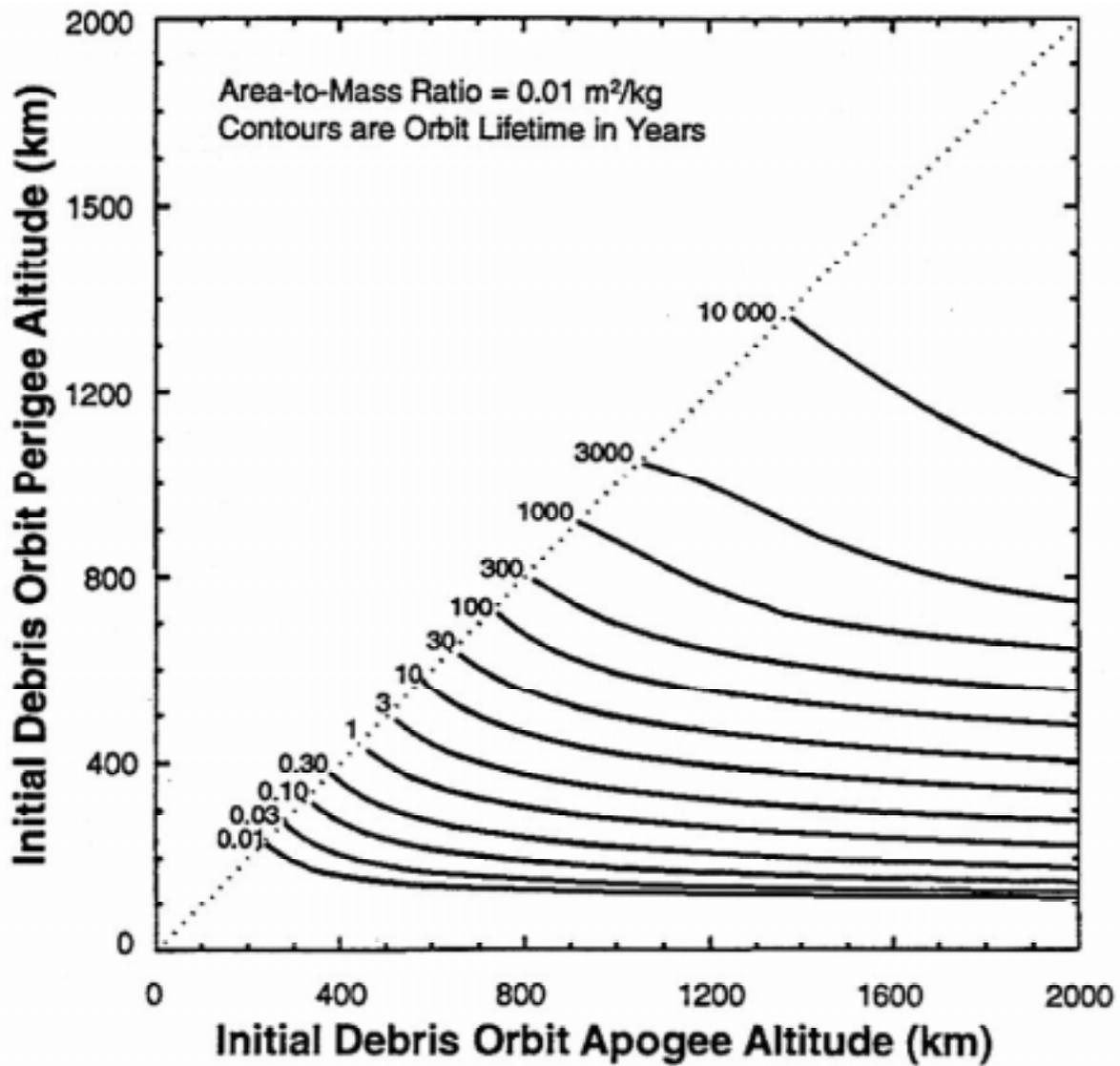


Figure 3-1. Orbit lifetimes for debris released in low altitude, low eccentricity orbits. Radiation pressure effects neglected in orbit lifetime calculations.

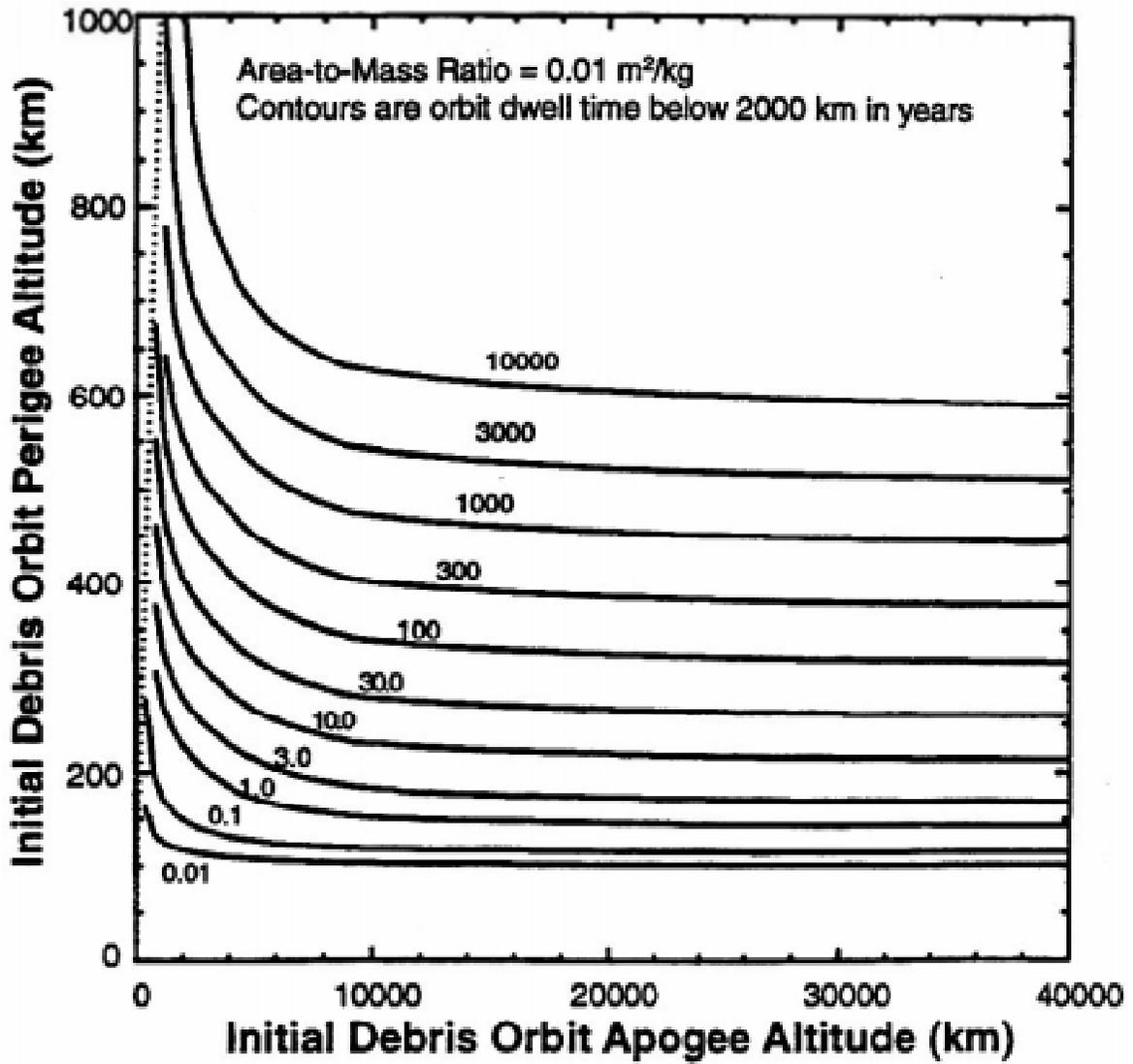


Figure 3-2. Orbit dwell times below 2000 km for debris released in high eccentricity orbit. Radiation pressure effects and lunar and solar gravity perturbations neglected in the calculation of orbit lifetime.

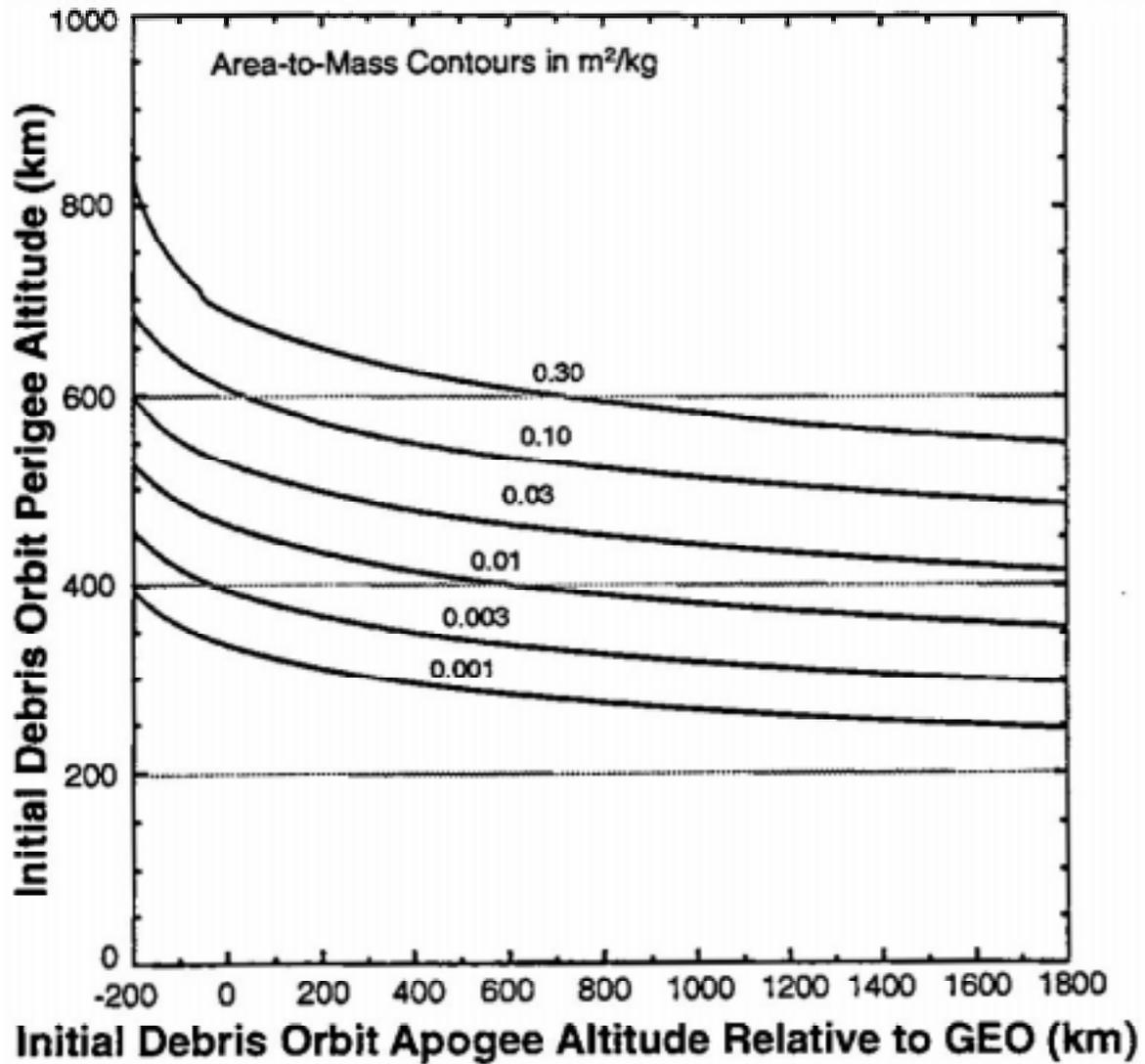


Figure 3-3. Initial orbit apogee and perigee altitude required to lower apogee altitude to 300 km below GEO altitude in 25 years as a function of area-to-mass ratio. Solar radiation pressure and lunar and solar gravity perturbation effects neglected in orbit evolution calculations.

## **4. ASSESSMENT OF DEBRIS GENERATED BY EXPLOSIONS AND INTENTIONAL BREAKUPS**

Explosions have been the primary contributor to the orbital debris environment. Some explosions have been accidental with on-board energy sources providing the energy. However, some intentional breakups have occurred as tests or as a means of disposing of spacecraft.

### **4.1 ACCIDENTAL EXPLOSIONS**

Accidental explosions of spent upper stages have been the primary source of debris in LEO. The source of energy for these events has been the structural failure of pressurized volumes or a failure allowing residual hypergolic bipropellant fuels to mix and ignite. The Delta second stage was a source of such debris. Investigations determined that the cause of the explosion was failure of the common bulkhead separating the two fuel components. This failure allowed residual bipropellants to combine, producing an explosion, the most recent of which occurred after 16 years in orbit. The Delta Project Office changed operating procedures to vent the fuels after completion of missions and no stage which has been vented has exploded. In 1978, a Soviet EKARAN satellite in GEO experienced an explosion, with an over-pressurized battery as the suspected cause. After breakup of the Ariane Spot -1 third stage in Sun-synchronous orbit, the Ariane introduced design modifications to prevent future explosions. In cases where design and operations modifications have been made to remove stored energy sources, accidental explosions have been prevented.

On-board energy sources include chemical energy in the form of fuels and explosives associated with range safety systems, energy in the form of pressurized volumes (as in sealed batteries and thermal control, attitude control, or propulsion systems) and kinetic energy (as with control moment gyroscopes).

## GENERAL POLICY OBJECTIVE

### CONTROL OF DEBRIS GENERATED BY ACCIDENTAL EXPLOSIONS

NASA programs and projects will assess and limit the probability of accidental explosion during and after completion of mission operations.

## GUIDELINES

- 4-1. *Limiting the risk to other space systems from accidental explosions during mission operations:* In developing the design of a spacecraft or upper stage, each program, via failure mode and effects analyses or equivalent analyses, will demonstrate either that there is no credible failure mode for accidental explosion, or if there are such credible failure modes, will limit through design or operational procedures the probability of the occurrence of such failure modes.

Note: As a quantitative reference, when the probability of accidental explosion can be estimated to be less than 0.0001, the intent of the guidelines has been met.

- 4-2. *Limiting the risk to other space systems from accidental explosions after completion of mission operations:* All on-board sources of stored energy will be depleted when they are no longer required for mission operations or postmission disposal. Depletion will occur as soon as such an operation does not pose an unacceptable risk to the payload.

## Rationale for Guidelines

Concerning the note to guideline 4-1: by keeping the probability of accidental explosion less than 0.0001, the average probability of an operating spacecraft colliding with an explosion fragment larger than 1 mm from that space system will be less than  $10^{-6}$  per "average spacecraft". An average spacecraft is a spacecraft of average size with average mission lifetime in circular orbit at an altitude through which explosion debris fragments of size 1mm or larger would pass if an explosion occurred. The average probability of collision is the probability of collision averaged over the altitude that would be covered by the breakup cloud.

Under guideline 4-2, explosions caused by stored energy in the form of fluids such as volatile liquids, bipropellant fuels, or electrolytes in batteries will be prevented by venting or using other depletion procedures. In the past, failure to remove energy sources has resulted in explosions occurring anywhere from hours to years after completion of the mission.

## Method to Assess Compliance with the Guidelines

### Limiting the Probability of Accidental Explosion (Guideline 4-1)

When the space system design is being evaluated by a failure mode and effects analysis or some equivalent analysis, the program or project will identify credible failure modes that lead to accidental explosion and estimate the probability of those failure modes occurring. The risk to operating spacecraft

caused by an accidental explosion is discussed in "Technical Background for Assessing Orbital Debris Risk," Volume II of the Reference Manual for Orbital Debris Assessments.

### **Eliminating Stored Energy Sources (Guideline 4-2)**

During space system design and development, the program will identify sources or potential sources of stored energy and develop and implement a plan for eliminating these sources at the end of mission operations. Safing procedures to be considered might include

- Burn residual fuels to depletion and leave fuel lines with valves open
- Vent pressurized systems
- Leave batteries in a permanent discharge state
- Deactivate range safety systems
- Remove power from control moment gyroscopes

### **4.2 INTENTIONAL BREAKUPS**

Intentional breakups have been used to provide safe reentry of space structures and to conduct on-orbit tests. An understanding of the approach taken in the evaluation for intentional breakups requires an understanding of the development of a debris cloud after breakup.

Immediately after breakup, the debris cloud exhibits large spatial and temporal changes in the concentration of the debris. For example, at the point where the breakup occurred there will be no debris at times, while at other times the debris cloud densities will be orders of magnitude above the background. An operating spacecraft may have a small probability of colliding with the debris if the interaction were to occur randomly, but a high probability of collision if it passes through a region of high density concentration. The test program can avoid having such high risk interactions by controlling the time of the test. However, because of the many perturbations that occur to objects in orbit and the sensitivity to the exact time and location of the breakup event, whether operating spacecraft will pass through regions of high debris density concentration can be determined accurately only a few days before the test. Consequently, the assessment and control of this risk must be performed immediately before the test.

Within a few days after breakup, the debris becomes more uniformly distributed within the cloud and the cloud reaches a state called the pseudo-torus. Within a few weeks to a few months after the test, the debris cloud evolves to a shell configuration. By the time the debris cloud reaches the pseudo-torus state, the probability of collision between the debris cloud and other objects in space can be calculated assuming random encounters. This means that the risk to other users of space can be characterized by quantities such as the area-time and object-time product for the debris cloud. These quantities, which depend on the breakup altitude and the general characteristics of the breakup process, can be calculated early in the development process for the testing program.

## GENERAL POLICY OBJECTIVE

### CONTROL OF DEBRIS GENERATED BY INTENTIONAL BREAKUPS

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NASA programs and projects will assess and limit the effect of intentional breakups on other users of space.

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## GUIDELINES

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- 4-3. *Limiting the long-term risk to other space systems from planned tests:* Planned test explosions or intentional collisions will be conducted at an altitude such that for debris fragments larger than 1 mm: (a) the area-time product does not exceed 0.1 m<sup>2</sup>-yr, and (b) the object-time product does not exceed 100 object-yr. No debris larger than 1 mm will remain in orbit longer than 1 year. This guideline is similar to guideline 3-1 for debris generated during normal operations.
- 4-4. *Limiting the short-term risk to other space systems from planned tests:* Immediately before a planned test explosion or intentional collision, the probability of debris larger than 1 mm from the breakup colliding with any operating spacecraft will be verified to not exceed 10<sup>-6</sup> immediately after breakup when the debris cloud presents regions of high risk for other space systems.
- 4-5. *Limiting the risk to other space systems from breakup as a planned reentry procedure:* The planned destruction of a structure as a routine reentry procedure will occur at an altitude no higher than 90 km.

## Rationale for Guidelines

These guidelines reflect the approach taken within the U.S. space program to limit the debris contribution from on-orbit tests. After the P-78 (SOLWIND) ASAT test, subsequent tests, such as the Delta-180, were reviewed by a safety panel for their near-term threat to operating spacecraft (guideline 4-4) and for their long-term contribution to the orbital debris environment (guideline 4-3).

Debris from intentional breakup released under guideline 4-3 of this section would be no more of a contributor to the long-term growth of the orbital debris environment than debris released under guidelines for normal operations (guideline 3-1). The limit of 1 year for orbit lifetimes for debris larger than 1 mm prevents the accumulation of debris from intentional breakups.

The risk to other users from concentrations within the debris cloud which occur immediately after breakup is limited by guideline 4-4 to no more than the risk represented by other debris deposition events such as release of operational debris.

Guideline 4-5 ensures that all debris from planned satellite destruction as a part of the reentry disposal procedure will deorbit within a few hours.

## Method to Assess Compliance with the Guidelines

### On-Orbit Tests (Guidelines 4-3 and 4-4)

The evaluation procedure for planned test breakups uses guideline 4-3 for long-term planning conducted during program development, and uses guideline 4-4 for near-term planning conducted immediately (a few days) before the test. The objective of the long-term plan is to understand and control the impact of the test on the space environment in general; that of the near-term plan is to control the risk of damage to operating spacecraft.

The steps for performing the evaluation are

1. Define a breakup model for the test. A breakup model describes the debris created in the breakup process in terms of the distributions in size, mass, area-to-mass ratio, and velocity imparted at breakup. A standard breakup model used for debris environment evolution calculations may be acceptable for a test, or the breakup model may require taking into account specific characteristics of the planned test. Standard breakup models or support for defining specific breakup models for a given test may be obtained from the Solar System Exploration Division at NASA Johnson Space Center.
2. Calculate and sum the area-time and object-time products for the debris as defined by the breakup model and the state vector at the time of breakup. This procedure is described in detail in chapter 3. The debris assessment software may be used to calculate initial state vectors for the debris fragments and the resulting orbit lifetimes. Compare these summed products with the guidelines.
3. Verify that no debris larger than 1 mm will have an orbit lifetime greater than 1 year.
4. At the time of the near-term evaluation, conducted a few days prior to the test, use the USSPACECOMMAND element set data to verify that immediately after breakup no operating spacecraft will have a probability of collision greater than  $10^{-6}$  in passing through regions of high density of debris larger than 1 mm. Special software is generally required to analyze the debris cloud characteristics immediately after breakup. For programs requiring element set data or assessment support, contact the Solar System Exploration Division at NASA Johnson Space Center.

### Planned Destruction as a Reentry Procedure (Guideline 4-5)

If the guideline for the breakup altitude of 90 km or below is followed, no additional assessment is required. If the breakup altitude is above 90 km, the disposal procedure is treated as if it were a test breakup and the assessment procedure for on-orbit tests is followed.

## Brief Summary of Debris Mitigation Measures

To lower the risk associated with on-orbit tests:

1. Lower the altitude at which the breakup occurs. This is by far the most effective response for reducing both the near-term and long-term risk to other users of space.

2. Lower the perigee altitude of the orbit of the test vehicle(s).
3. Adjust the time for performing a test by a few minutes to allow spacecraft or large debris to move away from regions of high flux concentration.



## 5. ASSESSMENT OF DEBRIS GENERATED BY ON-ORBIT COLLISIONS

This section covers debris generation by random on-orbit collision during mission operations. This includes both the direct generation of debris by collision between the space vehicle and another large object in orbit and the indirect or potential generation of debris when collision with small debris damages the vehicle to prevent its disposal at the end of mission operations, and making it more likely that the vehicle will be fragmented in a subsequent collision with another large object in orbit.

While it remains intact, a spacecraft or upper stage represents a small collision risk to other users of space; however, once it is fragmented by collision, the collision fragments present a risk to other users that is orders of magnitude larger. Because of the large collision velocities, a debris object much smaller than the spacecraft will cause fragmentation (referred to as catastrophic collision). For purposes of evaluation, debris of diameter 10 cm and larger will be assumed to cause such catastrophic collision.

Catastrophic collision during mission operations represents a direct source of debris, and the probability of this occurring is addressed by guideline 5-1. However, if a spacecraft or upper stage fails to perform postmission disposal it becomes a potential source of debris because a structure that is abandoned in orbit may subsequently experience catastrophic breakup. The probability of such an event occurring as a result of damaging impact with small debris is addressed by guideline 5-2.

### GENERAL POLICY OBJECTIVE

#### LIMIT THE GENERATION OF ORBITAL DEBRIS FROM ON-ORBIT COLLISIONS

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NASA programs and projects will assess and limit the probability of operating space systems becoming a source of debris by collisions with man-made debris or meteoroids.

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### GUIDELINES

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5-1. *Collision with large objects during mission operations:* In developing the design and mission profile for a spacecraft or upper stage, a program should estimate and evaluate the probability of collision with another large object during mission operations.

Note: As a quantitative reference, when the probability of collision with large objects is on the order of or less than 0.001, the intent of the guideline has been met. For programs using tethers, the tether itself need not be considered when estimating the collision probability with large objects.

5-2. *Collision with small debris during mission operations:* In developing the design of a spacecraft or upper stage, a program should estimate and limit the probability of collisions with small debris of size sufficient to cause loss of control to prevent postmission disposal.

Note: As a quantitative reference, when the probability of collision with debris leading to loss of control or inability to conduct postmission disposal is on the order of 0.01 or less, the intent of the guideline has been met.

## Rationale for Guidelines

Guideline 5-1 limits the amount of debris that will be created by collisions between spacecraft or upper stages in LEO or GTO (geosynchronous transfer orbit) and other large objects in orbit. By keeping the probability of collision between a spacecraft or upper stage and other large objects to less than 0.001, the average probability of an operating spacecraft colliding with collision fragments larger than 1 mm from that spacecraft or upper stage will be less than  $10^{-6}$  per “average spacecraft”. An average spacecraft is a spacecraft of average size with average mission lifetime in circular orbit at an altitude through which the fragments from such a collision would pass if the collision occurred. The average collision probability is the probability of collision averaged over the altitude that would be covered by the breakup cloud.

Guideline 5-2 limits the probability of spacecraft and upper stages being disabled and left in orbit at the end of mission, which would contribute to the long-term growth of the orbital debris environment by subsequent collisional fragmentation.

## Method to Assess Compliance with the Guidelines

### Collisions with Large Objects During Mission Operations (Guideline 5-1)

For missions in or passing through low Earth orbit (LEO), the probability of a space system being hit by an intact structure or large debris object during its mission life,  $P$ , can be approximated by\*

$$P = F \times A \times T \quad (5-1)$$

where

- F = cross-sectional area flux for the orbital debris environment, taken from figure 5-1
- A = average cross-sectional area for the space system in  $m^2$
- T = mission duration in years

The orbital debris flux is taken to be 0 for altitudes above 2000 km. The flux for meteoroids 10 cm in diameter or larger is negligible and can be ignored.

For a circular mission orbit the debris flux is taken from figure 5-1 at the mission orbit altitude.

For an eccentric mission orbit, break the orbit into altitude intervals such that the debris flux does not vary by more than a factor of 2 over any interval. Weight the flux in each altitude interval, taken from figure 5-1, by the fraction of time spent by the vehicle in that altitude interval to yield a time-weighted debris flux. The fraction of time spent in an altitude interval is approximated by the ratio of the size of the altitude interval to the difference between the apogee and perigee altitude of the orbit. For eccentric orbits,  $F$  is the sum of the time-weighted debris fluxes. The debris assessment software may be used for a more exact calculation of  $P$  or for an exact calculation of fraction of time spent in an altitude interval.

The average cross-sectional area is the cross-sectional area averaged over aspect. For a simple convex space system, it is 1/4 the surface area. A simple convex spacecraft body with solar panel wings may be given an average cross-sectional area that is 1/4 the surface area of the spacecraft body plus solar panels.

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\* The exact expression for this probability is  $P = 1 - e^{-FAT}$ , which is approximated by Eqn. 5-1 when the product  $F \times A \times T$  is less than 0.1

For highly irregular spacecraft shapes, an estimate of the average cross-sectional area may be obtained as follows: determine the view,  $V$ , that yields the maximum cross-sectional area and denote the cross-sectional area as  $A_{\max}$ . Let  $A_1$  and  $A_2$  be the cross-sectional areas for the two viewing directions orthogonal to  $V$ . Then define the average cross-sectional area as  $(A_{\max} + A_1 + A_2) / 2$ .

For systems in or passing through geosynchronous orbit (GEO), no assessment is required for collision with intact objects or large debris. Collisions between operating spacecraft will be avoided by preventing radio frequency interference between satellites, which follows the historical practice for controlling operating spacecraft in GEO. The population of large debris objects in GEO is thought to present negligible risk to operating spacecraft and can be ignored.

### **Collisions with Small Debris During Mission Operations (Guideline 5-2)**

Impact with small (millimeter to centimeter or milligram to gram) meteoroids or debris can cause considerable damage because the impacts usually occur at high velocity (~10 km/sec for debris, ~17 km/sec for meteoroids). An obvious failure mode caused by debris or meteoroid impact is for the impact to puncture a hole in some fluid container, causing leakage. However, many failure modes are not so obvious. For example, the shock pressures produced by an impact on the wall of a pressurized tank can damage the tank wall and cause the tank to rupture even though the impactor may not be large enough to puncture the wall. Some electrical component failures have been suspected to have been caused by debris impacts, either directly by severing wires or indirectly by causing electrical components to short circuit when exposed to melted aluminum ejecta or the plasma cloud generated by a nearby hypervelocity impact.

The following evaluation process is used to determine whether damaging impacts with small debris could reasonably prevent successful postmission disposal. The procedure estimates the probability that meteoroid or orbital debris impacts will cause components critical to postmission disposal to fail. If this estimate shows that there is a significant probability of failure, a full penetration analysis should be conducted to guide any redesign and to validate any shielding design. The procedure outlined below should not be used to design shielding.

To estimate the probability that impacts with small meteoroids or orbital debris will prevent postmission disposal:

1. Identify the components critical for postmission disposal and the surface in the component that, when damaged by impact, will cause the component to fail. This surface is termed the "critical surface".

Examples of critical components include propellant lines and propellant tanks, elements of the attitude control system and down-link communication system, batteries, and critical power lines.

The critical surface for a critical component depends on the type of failure for that component. For example, the failure of an unpressurized propellant tank might only result from full penetration of the tank wall; in this case the critical surface would be the interior surface of the tank wall, and the tank wall itself may be treated as part of the material shielding the surface from the environment. However, a pressurized tank may fail from an impact-induced surface flaw or pressure shock on the external surface of the pressure wall; in this case the critical surface would be the external surface of the tank wall. The inner surface of an electronics box would be the critical surface for most electronic boxes.

2. Calculate the at-risk surface area for the critical surface of each critical component,  $A_i$ .

To calculate the at-risk area for a critical surface first determine those parts of the critical surface that will be the predominant contributor to failure. Those will be the parts most exposed to space, and may be considered in two cases. In the case where the critical surface is equally protected by other spacecraft components no part of the surface is the major contributor and the at-risk area is the total area of the critical surface. In the case where some parts of the critical surface are more exposed to space than other parts, the at-risk area is the surface area of those parts of the critical surface most exposed to space.

For example, if an electronics box, having as its critical surface the inner surface of the box, is attached to the outer wall of the vehicle, the at-risk area will be the area of the inner surface of the box on the side attached to the outer wall.

The area at risk is now corrected to give an average cross-sectional area at risk. Correcting the surface areas to average cross-sectional areas has two cases. For vehicles that maintain their orientation relative to the velocity vector the average cross-sectional area at risk will be the area projected in the threat direction. For random tumbling vehicles the area will be 1/4 the projected area with the greatest exposure to space.

3. For each at-risk surface element, identify vehicle components and structural material between the surface and space that will help protect that surface.

Other vehicle components and structural material between a critical surface and the meteoroid/debris environment will shield the surface. Determine the material density and estimate the thickness of each layer of material acting as a shield in the direction where there is least material to act as a shield.

4. Estimate the minimum meteoroid or orbital debris impactor diameter that will damage each at-risk surface element.

To estimate the minimum impactor diameter first estimate the amount of material shielding the surface element from the environment. To do this, estimate the total areal density of vehicle material between the at-risk surface of each component and the environment,  $\sigma_i$ . Areal density of a material is the mass density,  $\rho$ , in grams per cubic centimeter, times the thickness of the material,  $\delta$ , in centimeters;. That is

$$\sigma \text{ [gm/cm}^2\text{]} = \rho \text{ [gm/cm}^3\text{]} \times \delta \text{ [cm]}$$

The units on areal density are grams per square centimeter. The total areal density between the at-risk surface  $i$  and the environment,  $\sigma_i$ , is the sum of the areal densities of the materials along the line connecting the at-risk surface to the environment.

For characteristic meteoroid and orbital debris materials, the minimum meteoroid / orbital debris diameter that will penetrate this areal density,  $d_i$ , will be

$$d_i \text{ [cm]} = K \times \sigma_i \text{ [gm/cm}^2\text{]}, \quad \text{where } K=0.07 \quad (5-2)$$

This calculation of debris diameter is conservative in the sense that it gives a lower bound on the size of debris that might be expected to penetrate the given areal density of material. For specially designed debris shields, K values as large as 0.35 can be achieved for a Whipple shield and 0.70 for a multi-layered, multi-shock shield. The quoted K values are only meant to give estimates of

shielding effectiveness; if there are potential problems with damaging impacts found via this analysis, shielding experts with more rigorous analysis tools should be consulted.

5. Determine the expected number of failures for each critical element,  $h_i$ .

For a vehicle in circular orbit at altitude  $H$ , the expected number of failures is the cross-sectional area flux,  $F(d_i, H)$ , taken from figures 5-2 or 5-3 for the debris size determined in Step 4, multiplied by the planned mission duration,  $T_m$ , in years, multiplied by the area of the at-risk surface,  $A_i$ , as determined in Step 2, multiplied by  $L$  factors, which are correction factors for the vehicle attitude profile as defined in table 5-1 and the associated discussion. Calculate  $h_i$  from

$$h_i = L_{MAN} \times F_{MAN}(d_i, H) + L_{MET} \times F_{MET}(d_i, H) \times T_m \times A_i \quad (5-3)$$

For an eccentric mission orbit, break the orbit into altitude intervals corresponding to the altitudes defined for the orbital debris environment in figure 5-2; note that the meteoroid environment provided in figure 5-3 is independent of altitude. Weight the flux in each altitude interval by the fraction of time spent by the vehicle in that altitude interval to give a time-weighted flux. The fraction of time spent in an altitude interval is approximately equal to the size of the altitude interval divided by the difference between the apogee and perigee altitude of the orbit. For an eccentric mission orbit,  $F_{MAN}$  and  $F_{MET}$  are the sum of the time-weighted debris and meteoroid fluxes respectively. The values for  $L_{MAN}$  and  $L_{MET}$  remain as defined in table 5-1. The debris assessment software may be used for a more exact calculation of  $h_i$  or for an exact calculation of time fraction spent in an altitude interval.

For a vehicle that maintains a constant attitude relative to its velocity vector, the  $L$  value for a critical surface depends on the orientation of the surface as shown in table 5-1:

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**Table 5-1. L Factor Values for Critical Surfaces on Vehicles  
Stabilized Relative to the Velocity Vector**

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Surface <sup>§</sup>	Front	Side	Top	Bottom	Rear
Debris ( $L_{MAN}$ )	3	3	0.01	0.01	0.02
Meteoroids ( $L_{MET}$ )	2	1	2	1	0.2

§: Front = facing direction of motion; Side = perpendicular to direction of motion, surface of Earth; Top = facing the zenith; Bottom = facing the center of the Earth; Rear = facing opposite to the direction of motion

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For surfaces that are not facing one of the orthogonal directions, use an  $L$  value for the closest orthogonal surface; if a surface is not close to any orthogonal surface use the average of the  $L$  values of the bounding orthogonal surfaces. If the vehicle does not maintain a fixed attitude relative to its velocity vector, use an  $L$  value of 1.

6. Calculate the expected number of failures for failure of postmission disposal critical elements,  $F_C$ , by summing the expected number of failures for each element, as determined in Step 5. This sum is expressed as

$$F_C = \sum h_i \quad (5-4)$$

7. Calculate the probability of failure of one or more critical elements,  $P_C$ , as a result of impact with debris by

$$P_C = 1 - e^{-F_C} \approx F_C \quad (5-5)$$

where the approximation in the last step is valid if  $F_C \leq 0.1$ .

## Brief Summary of Debris Mitigation Measures

1. If a LEO program or project has a high probability of colliding with large objects during its mission life, there are several mitigation measures that may be taken. These include
  - a. Changing the planned mission orbit altitude to reduce the expected collision probability.
  - b. Changing the spacecraft design to reduce cross-sectional area and thereby reduce the expected collision probability.
2. There are many mitigation measures to reduce the probability that collisions with small debris will disable the spacecraft and prevent successful postmission disposal. These measures use the fact that the debris threat is directional (for man-made debris, highly directional) and that the directional distribution can be predicted with confidence. Design responses to reduce failure probability include addition of component and/or structural shielding, re-arrangement of components to let less sensitive components shield more sensitive components, use of redundant components or systems, and compartmentalizing to confine damage. Since there are many alternatives to pursue for reducing vulnerability to impact with small debris, some of them requiring in-depth familiarity with hypervelocity impact effects, if a significant reduction in failure probability is required it is advisable to contact a debris group at one of the NASA centers for assistance.

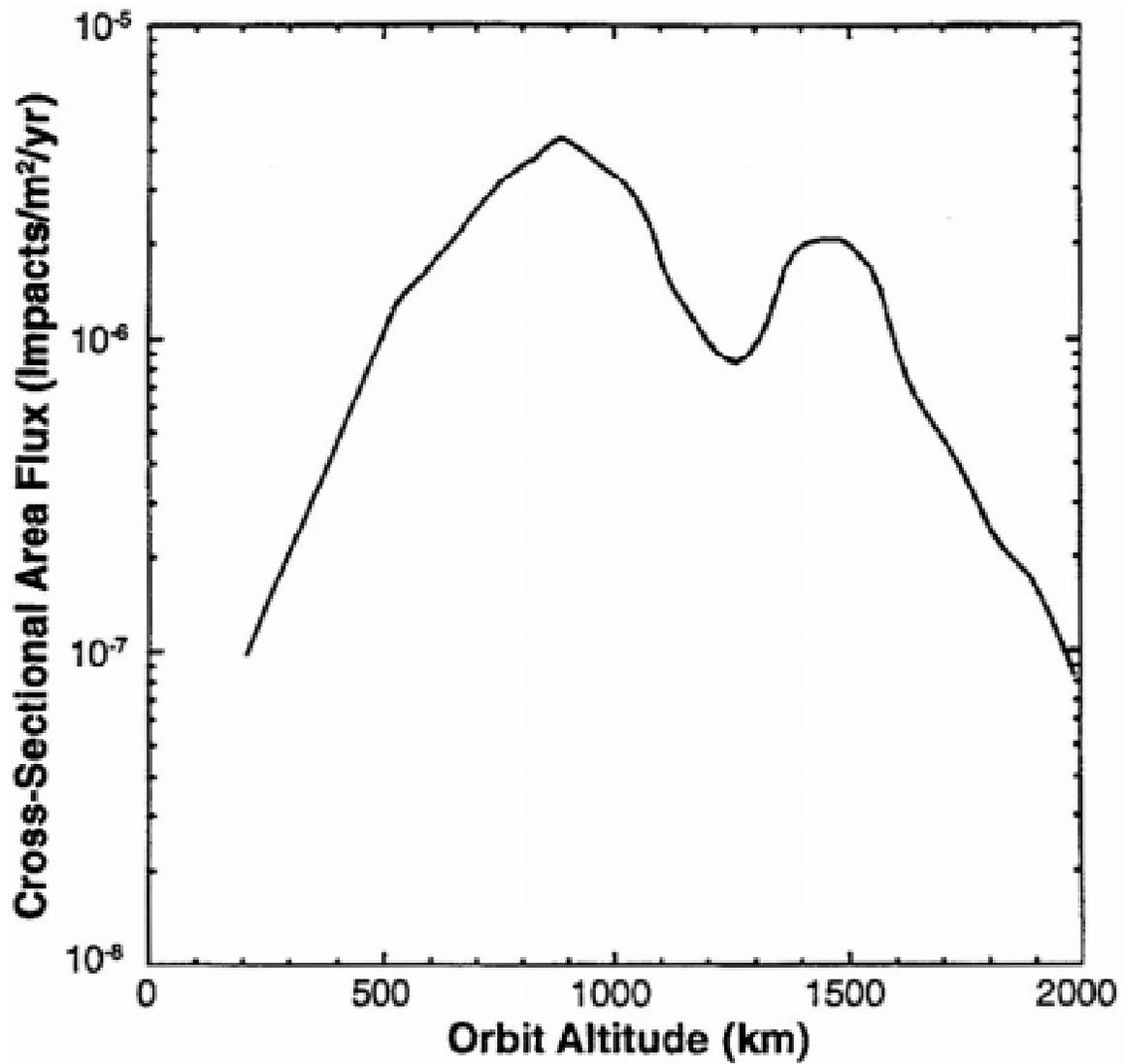


Figure 5-1. Cross-sectional area flux of intact space systems and large orbital debris.

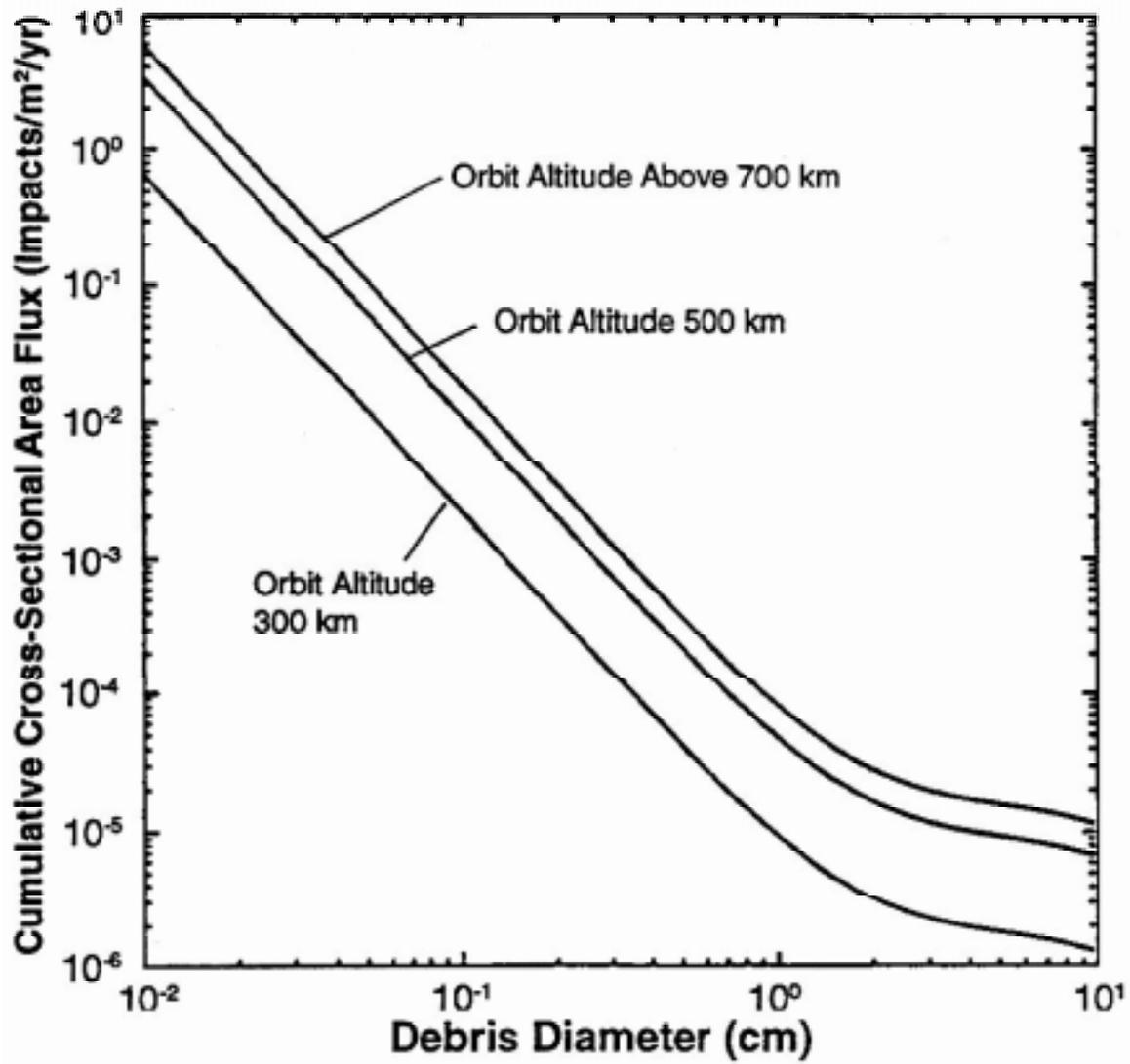


Figure 5-2. Cross-sectional area flux from orbital debris as a function of debris diameter for spacecraft in low Earth orbit.

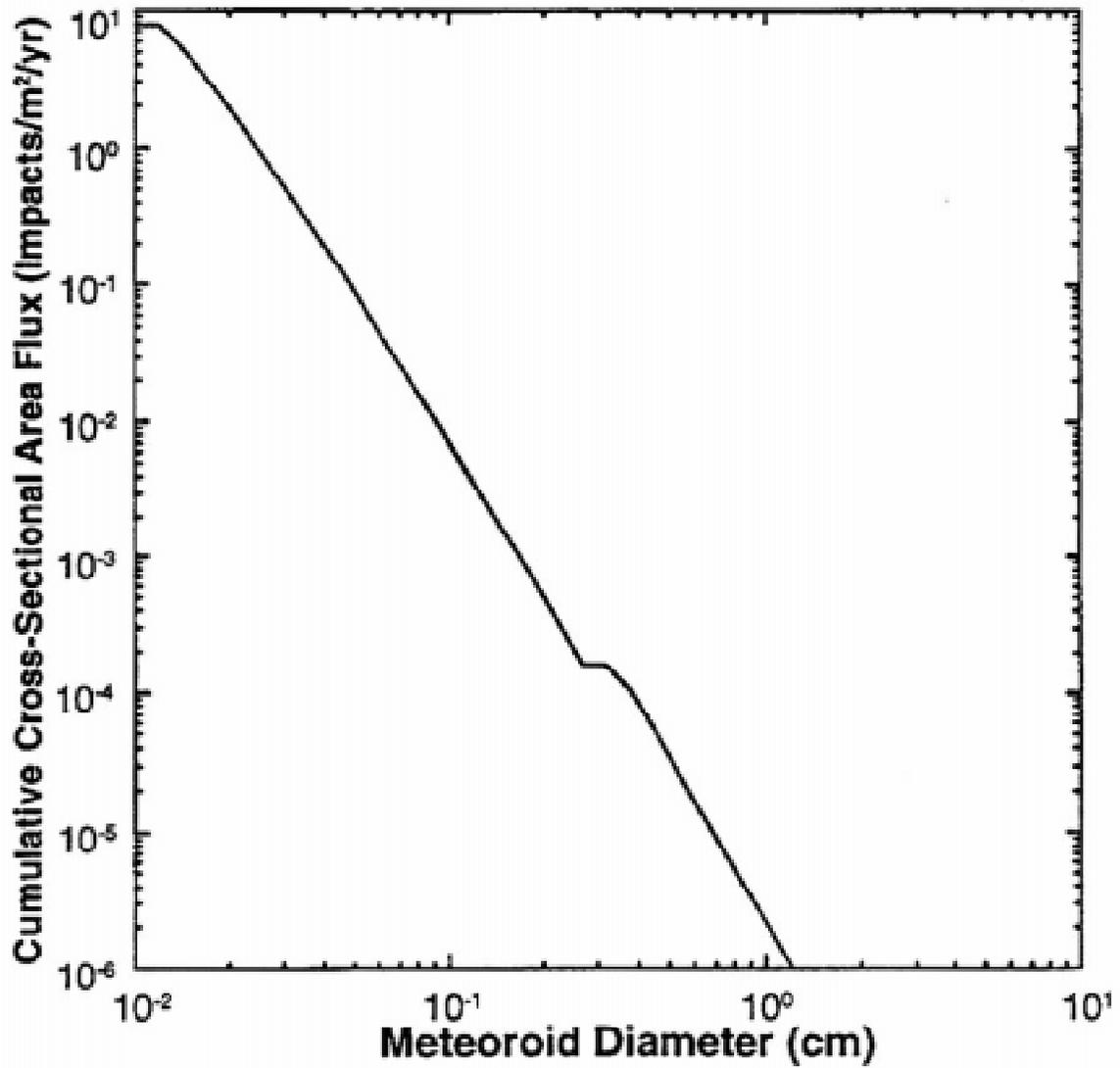


Figure 5-3. Cross-sectional area flux for meteoroids as a function of meteoroid diameter.

## 6. POSTMISSION DISPOSAL OF SPACE STRUCTURES

The historical practice of abandoning spacecraft and upper stages at the end of mission life has allowed roughly 2 million kg of debris to accumulate in orbit. If this practice continues, collisions between these objects will, within the next 50 years, become a major source of small debris, posing a threat to space operations that is virtually impossible to control. The most effective means for preventing future collisions is to require that all spacecraft and upper stages be removed from the environment in a timely manner. Such a requirement, however, would entail great cost in many cases, and there are regions of space where, for the immediate future, disposal of these systems could be made without creating a significant risk to future users. As a result, a variety of disposal options are presented in the guidelines. These guidelines represent an effective method for controlling growth of the environment while limiting the cost impact on future programs.

To provide a context for the guidelines, three high-value regions of space can be identified. These are

- *Low Earth orbit* - The region of space to 2000 km altitude
- *Geosynchronous Earth orbit* - The region of space containing the nearly circular 24-hour orbits. This region has been defined to be within 300 km of the altitude for geosynchronous satellites (from altitudes 35,488 to 36,088 km, centered on 35,788 km).
- *Semisynchronous orbit* - The region of space containing the nearly circular 12-hour orbits. This region has been defined to be within 300 km of the altitude for satellites in circular 12-hour orbits (from altitudes 19,900 to 20,500 km, centered on 20,200 km).

In general, the postmission disposal options are (1) direct retrieval and deorbit, (2) maneuver to an orbit for which atmospheric drag will remove the structure within 25 years, and (3) maneuver to one of a set of disposal regions in which the structures will not interfere with future space operations. Storage orbits in these disposal regions may be used to dispose of space systems at end of mission. These options are summarized in figure 6-1. Most GEO programs will transfer to the super-GEO storage orbit; highly eccentric, high perigee altitude programs might transfer to a sub-GEO storage orbit rather than lowering perigee to reenter. Programs using semisynchronous orbits might use either the low- or high-altitude storage orbit; LEO programs with mission orbit altitudes above 1500 km might choose to transfer to the low-altitude storage orbit rather than transfer to an orbit with a 25-year lifetime. For a program which places a structure in an orbit for eventual atmospheric reentry, there may be restrictions on the disposal maneuver if a significant amount of structure might survive uncontrolled reentry.

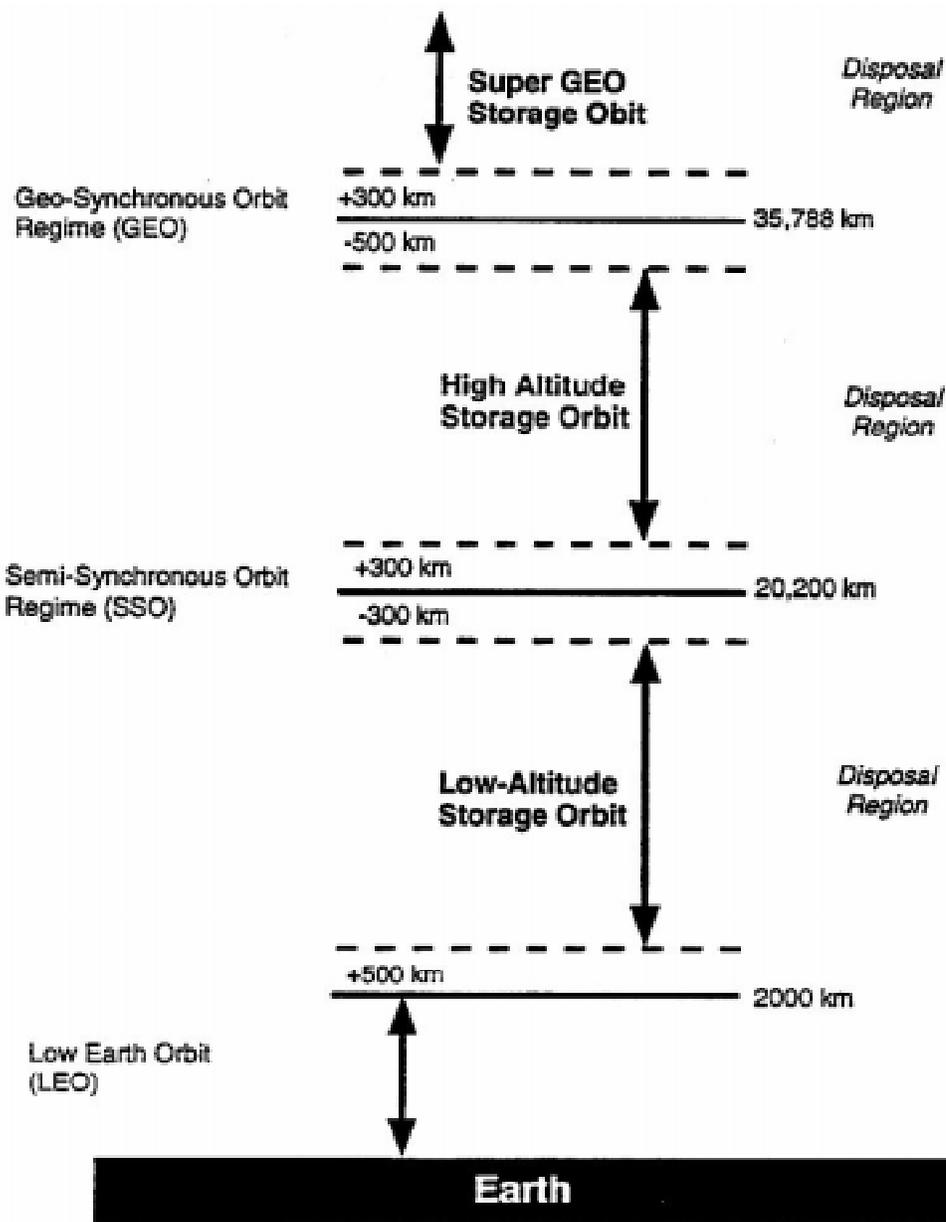


Figure 6-1. Disposal regions and storage orbit options for postmission disposal.

## **GENERAL POLICY OBJECTIVE**

### **POSTMISSION DISPOSAL OF SPACE STRUCTURES**

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NASA programs and projects will plan for the disposal of launch vehicles, upper stages, payloads, and other spacecraft at the end of mission life. Postmission disposal will be used to remove objects from orbit in a timely manner or to maneuver to a disposal orbit where the structure will not affect future space operations.

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## **GUIDELINES**

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- 6-1. *Disposal for final mission orbits passing through LEO:* A spacecraft or upper stage with perigee altitude below 2000 km in its final mission orbit will be disposed of by one of three methods:
- a. Atmospheric reentry option: Leave the structure in an orbit in which, using conservative projections for solar activity, atmospheric drag will limit the lifetime to no longer than 25 years after completion of mission. If drag enhancement devices are to be used to reduce the orbit lifetime, it should be demonstrated that such devices will significantly reduce the area-time product of the system or will not cause spacecraft or large debris to fragment if a collision occurs while the system is decaying from orbit.
  - b. Maneuvering to a storage orbit between LEO and GEO: Maneuver to an orbit with perigee altitude above 2500 km and apogee altitude below 35,288 km (500 km below GEO altitude).
  - c. Direct retrieval: Retrieve the structure and remove it from orbit within 10 years after completion of mission.
- 6-2. *Disposal for final mission orbits with perigee altitudes above LEO:* A spacecraft or upper stage with perigee altitude above 2000 km in its final mission orbit (except for orbits addressed in guideline 6-3) should be disposed of by either of two methods:
- a. Maneuvering to a storage orbit above GEO altitude: Maneuver to an orbit with a perigee altitude above the GEO altitude by a distance of at least  $300 \text{ km} + [1,000 \times \text{average cross-sectional area (m}^2\text{)} / \text{mass (kg)}]$  km.  

A program will use the postmission disposal strategy that has the least risk of leaving the vehicle near GEO in the event of a failure during the disposal process. Because of fuel gauging uncertainties near the end of mission, it is suggested that the maneuver be performed in a series of at least four burns which alternately raise apogee and then perigee.
  - b. Maneuvering to a storage orbit between LEO and GEO: Maneuver to an orbit with perigee altitude above 2500 km and apogee altitude below 35,288 km (500 km below GEO altitude).
- 6-3. *Disposal for final mission orbits that are near-circular 12-hour orbits:* A spacecraft or upper stage with perigee altitude above 19,900 km (300 km below the altitude for 12-hour circular orbits) and apogee altitudes below 20,500 km (300 km above the altitude for 12-hour circular orbits) should be maneuvered to an orbit with perigee altitude above 2500 km and apogee altitude below 19,900 km or to an orbit with perigee altitude above 20,500 km and apogee altitude below 35,288 km (500 km below GEO altitude).
- 6-4. *Reliability of postmission disposal operations:* In developing the design of a spacecraft or upper stage, a program will identify and limit all credible failure modes that could prevent successful postmission disposal.

Note: As a quantitative reference, when the probability of successfully performing the postmission disposal maneuver can be estimated to be 0.99 or greater, the intent of the guidelines has been met.

## Rationale for Guidelines

The intent of guideline 6-1a is to remove spacecraft and upper stages in LEO from the environment in a reasonable period of time. The 25-year removal time from LEO prevents the debris environment from growing over the next 100 years while limiting the cost burden to LEO programs. Spacecraft and upper stages in mission orbits with perigee altitudes below 600 km will usually have orbit lifetimes less than 25 years and will, therefore, automatically satisfy this guideline. This guideline will have the greatest impact on programs with mission orbit perigee altitudes above 700 km, where objects may remain in orbit hundreds of years if abandoned at the end of mission life.

Guideline 6-1a emphasizes the limitations of using drag enhancement devices to reduce orbit lifetime. Drag enhancement will increase the total area of the spacecraft or upper stage and may do little to reduce the probability of hitting large objects in the environment even though the orbit lifetime is reduced. It is, therefore, essential to demonstrate that drag enhancement does not in fact represent an increased risk to other users of space.

As stated in guidelines 6-1b and 6-2b, disposal orbits between LEO and GEO must have perigee altitudes above 2500 km. Objects in these orbits will have a low probability of collision (the current rate is less than 1 per 1000 years). If a collision does occur, very little debris from that collision will come low enough to place spacecraft in LEO at risk. Depending on the number, size, and orbit characteristics of objects using this disposal orbit option, the separation from LEO may need to be increased in the future.

If spacecraft and upper stages are placed in disposal orbits between LEO and GEO, the constraint on apogee altitude in guidelines 6-1b and 6-2b to be no higher than 500 km below GEO altitude prevents lunar and solar perturbations from causing these structures to interfere with GEO satellite operations. If a collision does occur in these disposal orbits, very little debris from that collision will come high enough to place GEO spacecraft at risk.

In guideline 6-1c only 10 years is allowed for planned retrieval after completion of the mission. This time is shorter than the 25 years for orbit decay and atmospheric reentry in guideline 6-1a because retrieval leaves the space system in high value regions of space, whereas transfer to an orbit with reduced lifetime lowers the perigee of the final mission orbit and reduces the fraction of time spent in high value regions of space.

Using guideline 6-2a, the region of space above GEO altitude can be used as a disposal region with little concern for debris buildup because of the low relative velocities, large regions of available space, and relatively low traffic rates in this area. In the near future, the 300 km altitude separation will be sufficient to isolate the disposal region from GEO if steps are taken to remove on-board energy sources after completion of the postmission disposal maneuver (guideline 4-1). However, depending on the level of traffic to GEO and on the characteristic sizes of future GEO satellites, this separation distance may need to be increased in the future. If measures are not taken to prevent explosive structural failure after disposal of GEO systems, a separation distance of ~2000 km will be required to isolate the disposal region from GEO.

The four burns prescribed in guideline 6-2a take into account fuel gauging limitations which are particularly serious at the end of missions. Given that there will be significant uncertainty as to the amount of fuel remaining at the end of mission, there will also be some uncertainty as to whether there is enough fuel to complete the disposal maneuvers. If there is not enough fuel to complete the maneuvers, a plan using four smaller burns to maneuver to the disposal orbit will leave the spacecraft or upper stage

farther from GEO altitude than would a plan using two burns. In planning the postmission disposal, uncertainties in fuel gauging should be considered in the assessment of reliability.

In guideline 6-3, a program with a near-circular, 12-hour orbit will need to maneuver to a disposal orbit above or below the semisynchronous region. The separation from semisynchronous circular orbit is at least 300 km, so lunar and solar perturbations will not cause these objects to interfere with spacecraft operating in 12-hour orbits.

To satisfy guideline 6-4, systems should be removed from useful regions of space with a high probability of success. A reliable propulsion system will have a probability of failure on the order of 0.01. Failure of the propulsion system should be the primary source of failure to perform successful postmission disposal.

## Method to Assess Compliance with the Guidelines

### Limiting Orbit Lifetime Using Atmospheric Drag (Guideline 6-1a)

The amount of time a structure will remain in orbit depends on its final orbit and on the area-to-mass ratio, as discussed in chapter 3. The evaluation in this section follows that developed for guideline 3-1.

The steps in the evaluation are described below.

1. Determine the final mass and average cross-sectional area for the system.

In calculating the final mass of the system, any fuel mass used for postmission disposal and any mass vented as a part of the safing of the structure should be deducted from the final mass. The average cross-sectional area is the cross-sectional area averaged over aspect.

For simple convex objects, the average cross-sectional area is 1/4 the surface area. For a simple convex spacecraft body with solar panel wings, calculate the average cross-sectional area from the surface areas of the spacecraft body,  $A_{\text{body}}$ , and the solar panels  $A_{\text{sp}}$ , as  $(A_{\text{body}} + A_{\text{sp}}) / 4$ .

For highly irregular spacecraft shapes an estimate of the average cross-sectional area may be obtained as follows: determine the view,  $V$ , that yields the maximum cross-sectional area and denote the cross-sectional area as  $A_{\text{max}}$ . Let  $A_1$  and  $A_2$  be the cross-sectional areas for the two viewing directions orthogonal to  $V$ . Then define the average cross-sectional area as  $(A_{\text{max}} + A_1 + A_2) / 2$ . If a structure will be gravity gradient stabilized, the average cross-sectional area perpendicular to the line of flight is used.

The area-to-mass ratio for the system is the average cross-sectional area in square meters divided by the mass in kilograms.

2. Calculate the orbit lifetime in the final mission orbit.

Either the procedures described in this section or the debris assessment software may be used. If the debris assessment software is used, the solar activity should be set to 130 solar flux units (sfu). To calculate orbit lifetime using the figures in the standard:

- a. Locate the apogee and perigee altitude for the final mission orbit in figure 6-2 for low altitude near-circular orbits or figure 6-3 for highly eccentric orbits.
  - b. Interpolate between the area-to-mass ratio contour values on either side of the orbit to get the area-to-mass ratio required for a 25-year orbit lifetime.
  - c. If the area-to-mass ratio for the project space system is less than that determined in step 2b, the orbit lifetime in the final mission orbit will exceed 25 years.
3. Plan to transfer to an orbit with reduced lifetime, if orbit lifetime in final mission orbit exceeds 25 years.

If the orbit lifetime in the final mission orbit exceeds 25 years, determine the decay orbit with a 25-year lifetime for the system being assessed. To do this

- a. Locate the apogee altitude of the final mission orbit on the initial decay orbit apogee altitude axis.
- b. Move vertically from this point to the area-to-mass contour matching that of the system being assessed. If the system area-to-mass ratio does not match a contour value, interpolate along a vertical line between adjacent area-to-mass contours to locate the point on the graph.
- c. The perigee altitude of the selected point is the perigee altitude of an orbit having the same apogee altitude as the final mission orbit and a 25-year orbit lifetime for the area-to-mass ratio of the system being assessed. This orbit requires the minimum propulsion to transfer the system from its final mission orbit to an orbit with a 25-year lifetime. It requires a single retro-burn (i.e., directly opposed to the direction of motion) at apogee of the final mission orbit.

The debris assessment software can also be used to calculate orbit lifetime for a specific orbit and area-to-mass ratio.

Spacecraft using atmospheric drag and reentry for postmission disposal need to be evaluated for survival of structural fragments to the ground. Guidelines for this evaluation are presented in chapter 7 of this standard.

### **Other Postmission Disposal Options (Guidelines 6-1b, 6-1c, 6-2, and 6-3)**

All other disposal options result in the space system being left in long-lifetime orbits that will not interfere with future space operations. A plan for performing the postmission disposal maneuvers should be included in the assessment.

### **Reliability of Postmission Disposal Operations (Guideline 6-4)**

The debris assessment should consider two areas: (1) design or component failure which leads to loss of control during the mission, and (2) failure of the postmission disposal system, including insufficient fuel to complete the disposal operation. Conventional failure modes and effects analysis or equivalent analysis can be used to assess failures which lead to loss of control during mission operations and postmission disposal.

Note: The probability of damage from collision with orbital debris or meteoroids leading to loss of control is analyzed under guideline 5-2.

## **Brief Summary of Debris Mitigation Measures**

For a program or project that elects to limit orbit lifetime using atmospheric drag and reentry, there are several options for reducing orbit lifetime:

1. Lower the initial perigee altitude for the decay orbit.
2. Increase the area-to-mass ratio for the structure using drag augmentation, but be aware of the restrictions imposed in guideline 6-1a.
3. For highly eccentric orbits, restrict the initial right ascension of ascending node of the orbit plane relative to the initial right ascension of the Sun so that the average perigee altitude is lowered.
4. To increase the probability that the postmission disposal maneuver will be successful, the program may want to consider incorporating redundancy into the postmission disposal system.

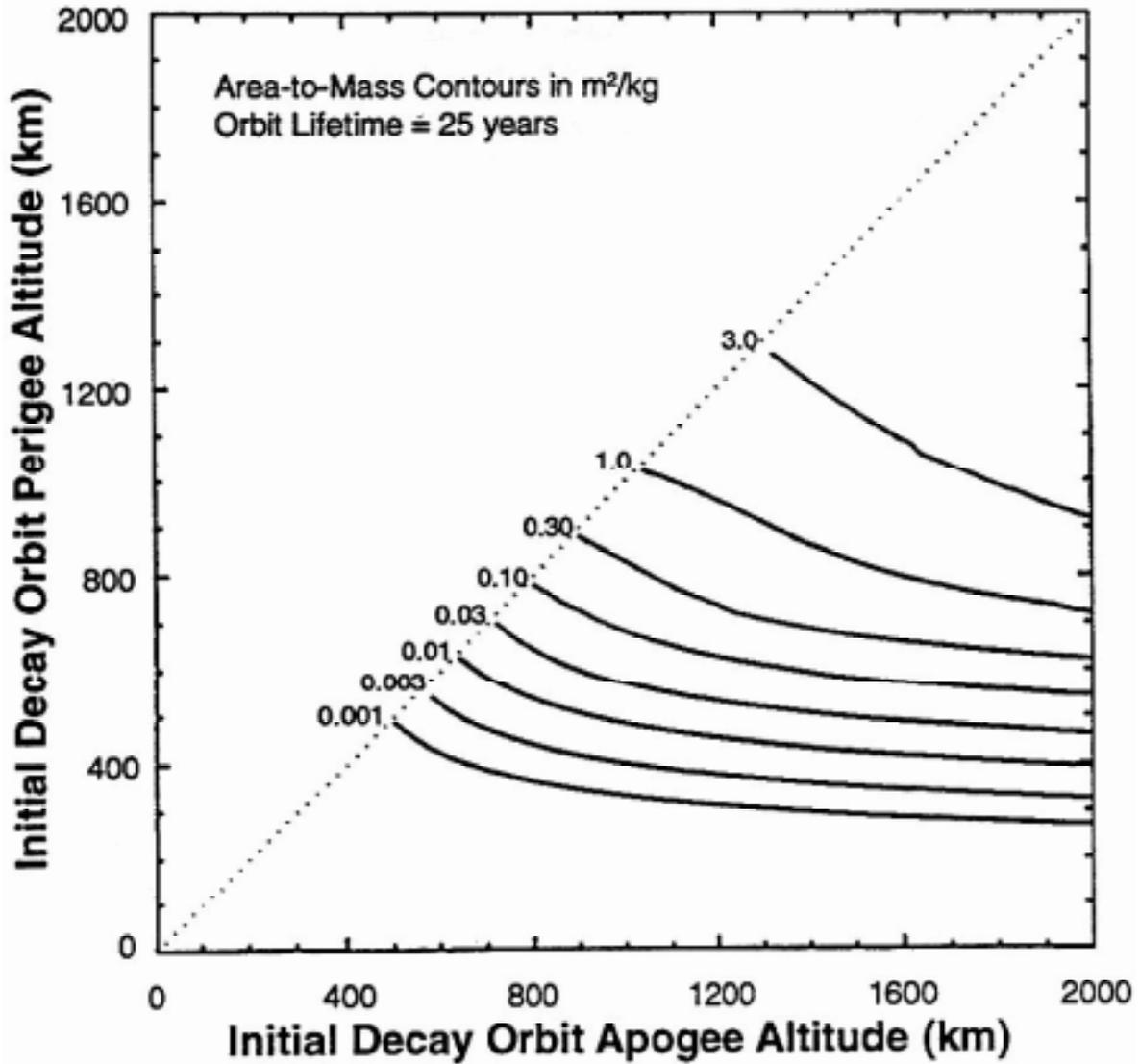


Figure 6-2. Area-to-mass contours for 25-year orbit lifetime for space systems left in low altitude, low eccentricity orbit.

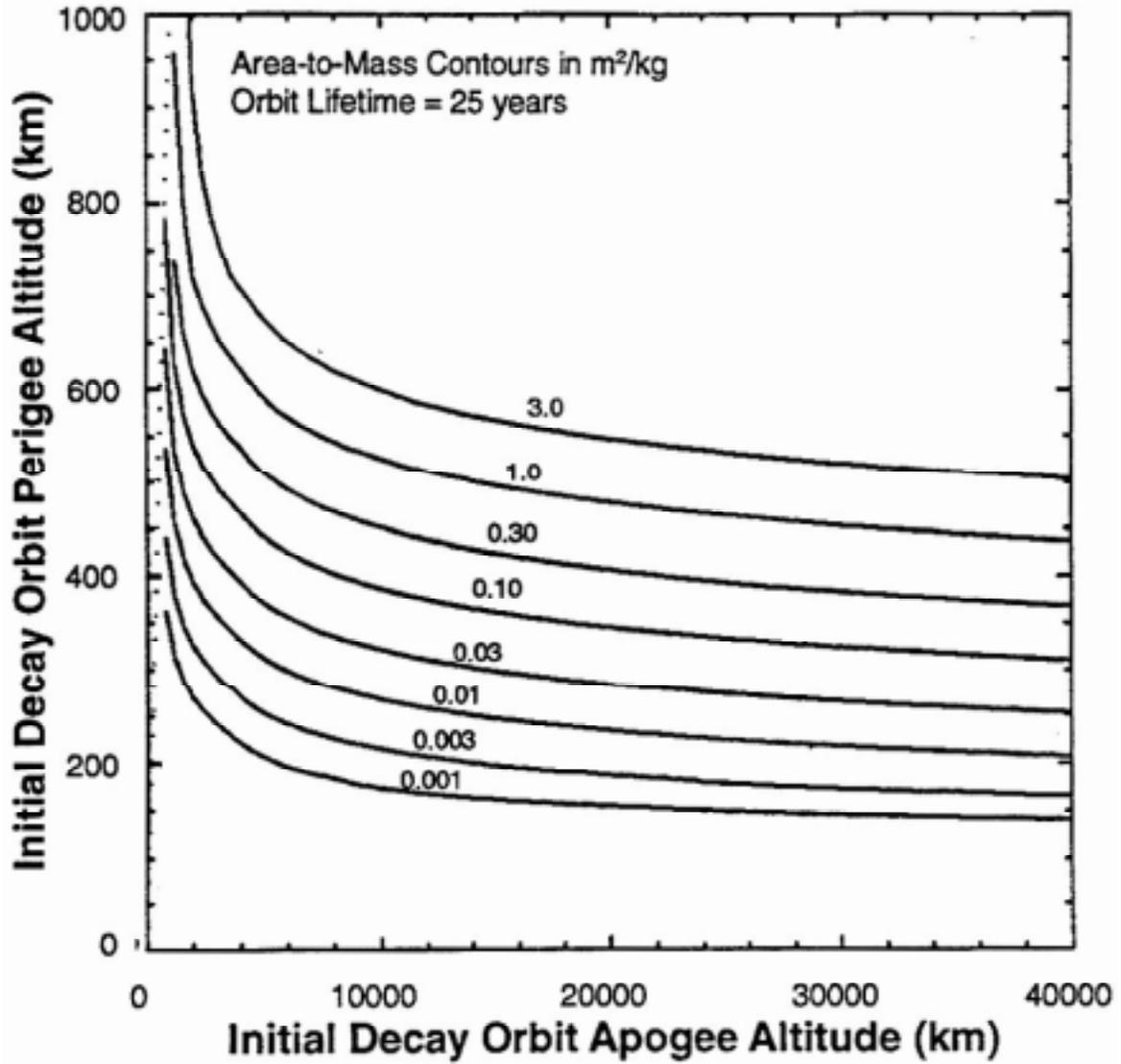


Figure 6-3. Area-to-mass contours for 25-year orbit lifetime for space systems left in highly eccentric orbit. Lunar and solar gravity perturbation effects neglected in orbit lifetime calculation.

## **7. SURVIVAL OF DEBRIS FROM THE POSTMISSION DISPOSAL ATMOSPHERIC REENTRY OPTION**

Programs or projects that use atmospheric reentry to limit the orbit lifetime of their systems in conformance to guideline 6-1 present a potential risk to the Earth's population. This chapter presents the guideline that defines the maximum amount of debris that can survive reentry if the reentry is uncontrolled. Uncontrolled reentry is defined as reentry in which the ground footprint location cannot be determined with sufficient accuracy to guarantee missing landmasses.

### **GENERAL POLICY OBJECTIVE LIMITING THE RISK FROM DEBRIS SURVIVING UNCONTROLLED REENTRY**

NASA programs and projects that use atmospheric reentry as a means to remove space structures from orbit at the end of mission life will limit the amount of debris that can survive uncontrolled reentry. If there is a significant amount of debris surviving uncontrolled reentry, measures will be taken to reduce the risk by establishing procedures or designs to reduce the amount of debris reaching the Earth's surface or to control the location of the ground footprint.

### **GUIDELINE**

7-1. *Limit the risk of human casualty:* If a space structure is to be disposed of by uncontrolled reentry into the Earth's atmosphere, the total debris casualty area for components and structural fragments surviving reentry will not exceed 8 m<sup>2</sup>. The total debris casualty area is a function of the number and size of components surviving reentry and of the average size of a standing individual. This term is defined more precisely in the method to assess compliance section of this chapter.

### **Rationale for Guideline**

The guideline for uncontrolled reentry provides an upper limit of 8 m<sup>2</sup> on the total casualty area of debris that impacts the Earth. An upper limit of 8 m<sup>2</sup> is derived by assuming an average risk of human casualty of 0.0001 per reentry event. However, the risk of a reentry event causing any casualties is actually lower since no correction has been made for the fact that people are usually protected inside buildings or vehicles and will therefore be shielded from reentering debris. To date, no casualties have been attributed to reentering man-made space structures.

## Method to Assess Compliance with the Guidelines

The measure of risk from reentering debris is the debris casualty area. For a piece of debris that survives atmospheric reentry, the debris casualty area is the debris cross-sectional area plus a factor for the cross-section of a standing individual. The total debris casualty area for a reentry event is the sum of the debris casualty areas for all debris pieces surviving atmospheric reentry. Equation 7-1 is used to calculate the total debris casualty area.

An approach for estimating the total debris casualty area ( $D_A$ ) of a space structure from a decaying orbit is summarized in figure 7-1. The following procedure is used to determine if a reentering space structure exceeds the total debris casualty area limits. The parent body is the structure as it exists in orbit. (Note: This procedure has been automated in the debris assessment software)

1. Establish the type of model for the parent body and determine the reference area.

If the dimensions of the parent body are approximately equal in all directions, it should be modeled as a sphere with the diameter,  $D$ , defined to be the largest dimension. The reference area is then

$$A_{\text{ref}} = \frac{\pi D^2}{4}$$

If the parent body is not modeled as a sphere, it should be modeled as an equivalent cylinder. The longest dimension will be the length ( $L$ ), and the largest dimension in the transverse direction will be the diameter of the cylinder ( $D$ ). The reference area is then

$$A_{\text{ref}} = L \times D$$

2. Determine the altitude and total velocity and flight path angle relative to the atmosphere of the parent body at breakup.

As discussed in the Reference Manual for Debris Assessment, experience has indicated that most structures break up at an altitude of approximately 78 km.

Using the equivalent sphere or cylinder dimensions and the mass of the parent body, the trajectory from reentry interface to the breakup altitude should be computed to determine the total velocity and flight path angle relative to the atmosphere at the breakup altitude. The reentry interface altitude is generally taken to be 122 km (400,000 ft).

The debris assessment software provides the capability for calculating the relative velocity and flight path angle at breakup given the breakup altitude. These values should be used unless a reentry breakup analysis has been conducted specifically for the structure in question.

3. Identify the components within the parent body.

If the parent body is larger than 0.5 m in any dimension and consists of multiple components, it will break up into components of significant size during reentry. Each of these components must then be evaluated separately. The design of the structure must be reviewed and all components that are larger than 0.25 m in any dimension must be identified.

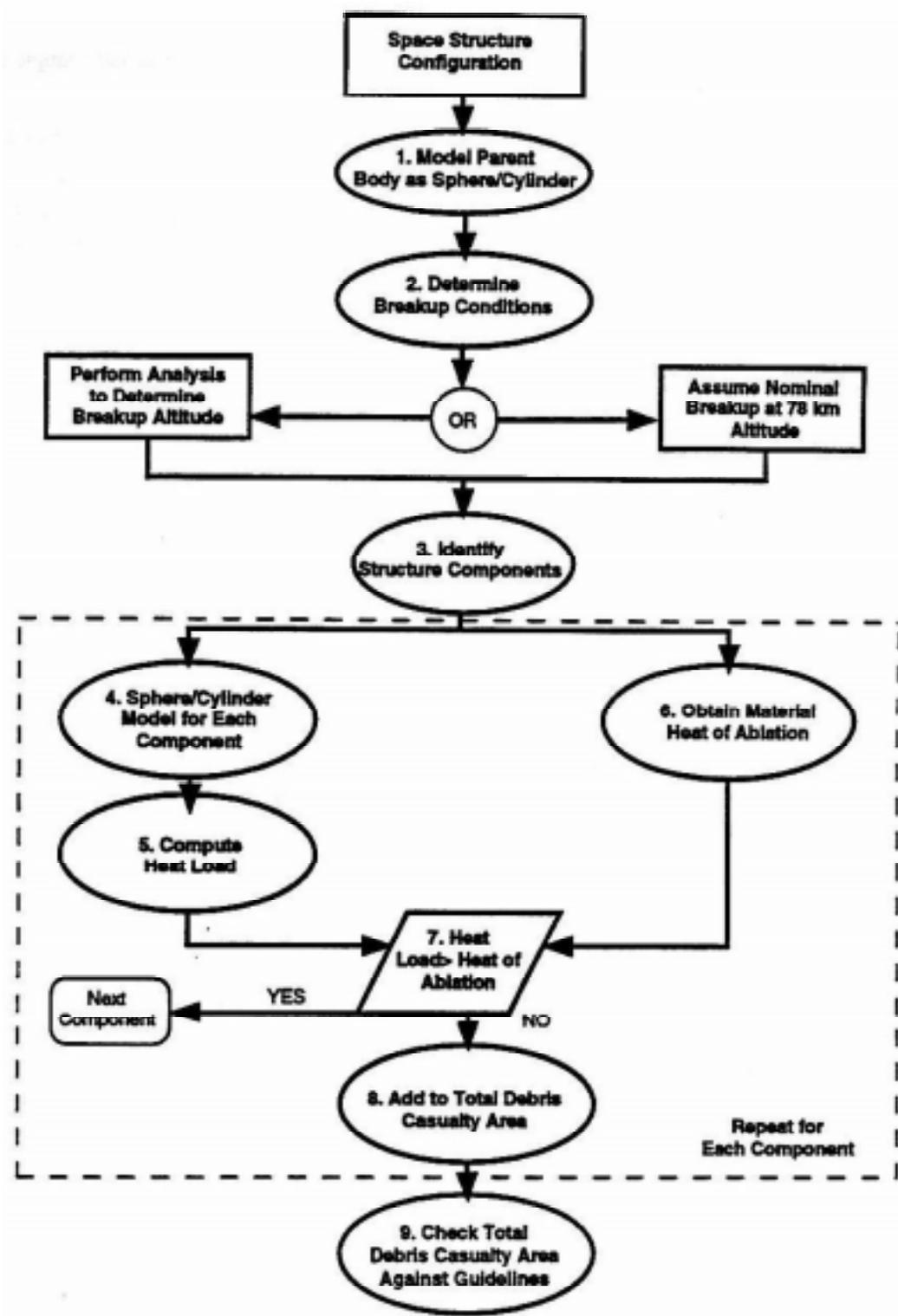


Figure 7-1. Flow chart for calculating total debris casualty area,  $D_A$ , for uncontrolled reentry.

If the structure is smaller than 0.5 m in any dimension, the parent body is considered a single piece of reentering debris and step 4 may be skipped.

4. Model each of the components coming out of the breakup as equivalent spheres or cylinders and determine the predominant material of each component.

Using the procedure described in step 1, model each of the components as described in step 3 as equivalent spheres or cylinders. A flat plate is modeled as a cylinder of having as diameter the largest plate dimension and length the thickness of the plate. Calculate the reference area for each component using the rules from step 1.

Review the design of each component and identify the predominant material. The total mass of each component must also be specified.

5. Determine the integrated heat load experienced by each component.

Using the equivalent sphere or cylinder dimensions, the mass, and the initial trajectory conditions, compute the average heat load to each component. Software such as the debris assessment software is required for this step.

6. Determine the specific heat of ablation,  $h_a$ , of the predominant material of each component.

Using the material properties,  $h_a$  is computed by

$$h_a = c_p \times (T_m - T_i) + h_f \quad [\text{J/kg}]$$

where

$c_p$  = specific heat capacity (J/kg-°K)

$T_m$  = melt temperature (°K)

$T_i$  = initial temperature (°K)

$h_f$  = heat of fusion (J/kg)

Table 7-1 provides specific heat capacity, heat of fusion, and heat of ablation (assuming an initial temperature of 300°K) for many spacecraft materials.

7. Determine the reentry survivability of each component.

Let

$H$  = the heat load per unit area experienced by a reentering space structure (J/m<sup>2</sup>)

$M$  = component mass (kg)

$A_s$  = surface area of component (m<sup>2</sup>)

A necessary and sufficient condition for a structure to survive reentry is

$$H < M \times h_a / A_s$$

**Table 7-1. Selected Properties for Materials Commonly Used in Spacecraft Fabrication**

No.	Material	$\rho$ (kg/m <sup>3</sup> )	$c_p$ (J/kg-K) (Avg)	$k$ (W/m-K) @300K	$h_f$ (J/kg)	$\Delta H_{ox}$ (J/kg-O <sub>2</sub> )	T melt (K)	$h_a$ (J/kg)
1	SS 21-6-9	7832.8	439.2	12.44	286098	16816980	1728	924302
2	Al 2024-T8xx	2803.2	972.7	154.64	386116	34910934	856	926937
3	Gr/Ep	1550.5	879.3	4.92	23	12305703	700	351731
5	Beryllium	1842.1	2635.3	169.49	1093220	32073679	1557	4405792
7	Copper	8938.0	430.7	395.88	205932	9832002	1356	659740
8	UZrH	6086.8	418.7	10.00	131419	0	2144	903688
10	Gold	19300.1	139.5	316.54	64895	0	1336	209417
18	Lead	11677.1	134.7	35.68	34968	8669002	600	64368
19	Silver	10491.8	232.8	429.58	105833	1943838	1234	323268
24	Strontium	2594.9	736.9	35.41	95599	36999682	1043	642960
25	Titanium (6 Al-4 V)	4437.0	805.2	7.15	393559	32480264	1943	1716421
29	Barium	3491.9	284.7	18.04	55824	34594598	983	250380
44	Al 5052	2684.6	900.2	137.03	386116	34910934	880	908235

8. Compute the total debris casualty area. The total debris casualty area in square meters,  $D_A$ , is calculated as follows:

$$D_A = \sum_{i=1}^N (0.6 + \sqrt{A_i})^2 \quad (7-1)$$

where N is the number of objects that survive reentry .

If the procedure described in this section is used to model reentry survivability,  $A_i$  is the reference area in square meters of the  $i$ th piece predicted to survive reentry, as determined in step 4. If a reentry model is used that accounts for mass loss during reentry,  $A_i$  will be the calculated average cross-sectional area of that piece at ground impact.

The average cross-sectional area of a standing individual, viewed from above, was taken to be 0.36 m<sup>2</sup>. The 0.6 term is then the square root of this area.

9. Determine if the total debris casualty area exceeds the guideline limits.

If  $D_A$  exceeds 8 m<sup>2</sup>, procedures should be initiated to decrease the amount of debris that could potentially survive reentry or to control the ground impact point for the debris.

Since the procedures outlined in this section are approximate, a more rigorous reentry survivability analysis may be conducted to determine the pieces that survive reentry and contribute to the debris casualty area. If this is done, then the models and analysis procedures should be documented in the debris assessment report.

## Brief Summary of Debris Mitigation Measures

If the amount of debris surviving reentry exceeds the guideline, then either the ground impact point should be controlled by the postmission disposal maneuver or measures should be taken to reduce the amount of debris surviving reentry. Control of the ground impact point requires control of the location of the postmission disposal burn and also requires that the postmission disposal orbit have a low (typically negative) perigee altitude. Options to consider include:

1. Performing a controlled reentry.

Maneuvering the structure at the end of mission to a disposal orbit with a perigee altitude low enough to control the location of the reentry and ground impact points. This option was adopted by the STS program ("Space Shuttle: Flight and Ground Systems Specifications," NASA Johnson Space Center, NSTS 07700, Volume X, Revision J, June 1990). The guidelines in this document are

- a. The reentry debris impact footprint will be no closer than 370 km, or 200 nautical miles (nm), from foreign landmasses, 46 km (25 nm) from U.S. territories and the Continental United States (CONUS), and 46 km (25 nm) from the permanent ice pack of Antarctica.
- b. Authorities for shipping lanes and airline routes in the area of the reentry footprint will be notified of the event.

These guidelines would be acceptable for debris surviving in controlled reentry.

2. Using materials that are less likely to survive reentry.

The material properties of the components have a significant effect on reentry survivability. While thermophysical and physical properties such as thermal conductivity, specific heat capacity, heat of fusion, melt temperature, and density influence the reentry survivability behavior, the heat of ablation is the best indicator of the component's ability to survive reentry. Materials with a low heat of ablation can be used to reduce the debris that survives reentry. If a material with a lower heat of ablation can be substituted in the design, the debris area can possibly be reduced. Volume II of the Reference Manual for Orbital Debris Assessment discusses the effect of material properties on the reentry demise.

3. Decreasing the effective drag at the reentry interface by decreasing the frontal area of the object.

This decreases the area-to-mass ratio and results in a decreased probability of reentry survival. Thus, design practices or operational procedures that decrease the area-to-mass ratio during reentry can reduce the amount of debris which survives reentry. Once the reentry process has begun, procedures to reduce the structure frontal area can reduce the reentry debris area.

4. Causing a structure to break up at higher altitude.

Even structures with large area-to-mass ratios can survive reentry if they are protected from the reentry environment until late in the reentry process. Design practices that release the hardware earlier in the reentry trajectory reduce the probability of its survival.

5. Maneuvering the structure at the end of the mission to a disposal orbit where reentry will not occur (chapter 6, "Postmission Disposal of Space Structures").

Measures to mitigate the risks of reentry are discussed more fully in the Reference Manual for Orbital Debris Assessment.



## 8. FORMAT FOR ASSESSMENT REPORTS

Two stand-alone debris assessment reports should be submitted during program or project development. The initial report should be submitted at preliminary design review (PDR) with a final assessment 45 days prior to critical design review (CDR). The reports are to be approved by the sponsoring Associate Administrator and then coordinated with the Office of Safety and Mission Assurance. The purpose of the report submitted at PDR is to identify debris issues early in the development cycle where resolutions are least costly to implement. The report submitted prior to CDR will document the position of the program or project relative to the guidelines to limit orbital debris generation.

The suggested format of these reports parallels the format of this standard, so that information developed using the standard can be easily integrated into the reports.

### 8.1 FORMAT FOR REPORT ISSUED AT PDR

Prior to PDR, a preliminary debris assessment should be conducted to identify areas where the program or project might contribute debris and to assess this contribution relative to the guidelines in so far as is feasible.

The debris assessment report should be organized as follows:

#### **Section 1: Brief Background on Program and Program Management**

To include

- Mission Description
- Program/Project objectives
- Program/Project schedule
- Responsible program or project manager

#### **Section 2: Description of Design and Operations Factors**

##### *2.1 Hardware*

To include (as available)

- Physical description of main structure
- Description of surfaces/materials exposed to space
- Description of spacecraft components most sensitive to debris impact
- Description and location of pressurized volumes
- Description of on-board propellants
- Description and location of fuel storage and transport systems
- Description of range safety systems
- Description of systems containing stored kinetic energy

## 2.2 Mission Parameters

To include

- Number of spacecraft
- Launch date and time
- Mission orbit (apogee/perigee altitude, inclination)
- Flight attitude

## Section 3: Assessment of Debris Released During Normal Operations

### 3.1 Debris Released During Staging, Payload Separation, or Payload Deployment

To include

- Preliminary description of debris released and associated orbits
- Preliminary estimate of area-time and object-time products

### 3.2 Debris Released During Mission Operations

To include

- Preliminary description of debris released and associated orbits
- Preliminary estimate of area-time and object-time products

## Section 4: Assessment of Orbital Debris Generated by Explosions and Intentional Breakups

### 4.1 Explosions from On-Board Stored Energy

To include

- General description of systems or components containing stored energy
- General plan for depleting stored energy sources after completion of mission

### 4.2 Intentional Breakups

To include

- General description of object being fragmented
- Description of energy source
- Description of orbit in which breakup will occur, and location and altitude of breakup
- Description of breakup model
- Preliminary estimate of area-time and object-time product of breakup fragments
- Preliminary plan for assessing risk to other operating spacecraft from the debris cloud formed immediately after the test

## Section 5: Assessment of Debris Generated by On-Orbit Collisions

### 5.1 Assessment of Collisions with Large Objects During Mission Operations

To include

- Estimate of probability of impact with large objects, based on planned mission

## 5.2 *Assessment of Collisions with Small Debris During Mission Operations*

To include

- Identification of systems or components most vulnerable to debris impact
- Preliminary assessment of shielding requirements, design considerations

## **Section 6: Description of Postmission Disposal Procedures and Systems**

To include

- Planned option for postmission disposal
- Description of disposal procedures and systems
- Identification of obstacles to successful postmission disposal

## **Section 7: Assessment of Survival of Debris from the Postmission Disposal Atmospheric Reentry Option**

To include

- Initial assessment of structures which will survive uncontrolled reentry
- Conservative estimate of total debris casualty area for debris surviving uncontrolled reentry
- Preliminary plan for atmospheric reentry if it appears that the guideline of 8 m<sup>2</sup> will be violated

## 8.2 FORMAT FOR REPORT ISSUED PRIOR TO CDR

Forty-five days prior to CDR, another debris assessment should be completed. This report should comment on changes made since the PDR report. The level of detail in this report should be consistent with the available information on design and operations.

When there are design changes after CDR that impact the potential for orbital debris generation, an update of the debris assessment report should be prepared, approved, and coordinated with the Office of Safety and Mission Assurance.

The debris assessment report should be organized as follows:

### **Section 1: Brief Background on Program and Program Management**

To include

- Mission description
- Program/project objectives
- Program/project schedule
- Responsible program or project manager

## **Section 2: Description of Design and Operations Factors**

### *2.1 Hardware*

To include

- Physical description of main structure
- Description of surfaces/materials exposed to space
- Description of spacecraft components most sensitive to debris impact
- Description and location of pressurized volumes
- Description of on-board propellants
- Description and location of fuel storage and transport systems
- Description of range safety systems
- Description and location of systems containing stored kinetic energy

### *2.2 Mission Parameters*

To include

- Number of spacecraft
- Launch date and time
- Mission orbit (apogee/perigee altitude, inclination)
- Flight attitude

## **Section 3: Assessment of Debris Generated During Normal Operations**

### *3.1 Debris Released During Staging, Payload Separation, or Payload Deployment (Guidelines 3-1 and 3-2)*

To include

- Description of debris to be released, including size, mass, cross-sectional area, initial orbit, orbit lifetime
- Calculated area-time for debris larger than 1 mm ( $\text{m}^2\text{-yr}$ )
- Calculated object-time for debris larger than 1 mm (yr)
- Calculated time for removal of debris from GEO altitude to at least 300 km below GEO altitude
- Source for analysis if not this standard or the debris assessment software

### *3.2 Debris Released During Mission Operations (Guidelines 3-1 and 3-2)*

To include

- Description of debris to be released, including release time, size, mass, cross-sectional area, initial orbit, orbit lifetime
- Maximum total cross-sectional area of debris in orbit at any given time ( $\text{m}^2$ )
- Calculated area-time for debris larger than 1 mm ( $\text{m}^2\text{-yr}$ )
- Calculated object-time for debris larger than 1 mm (yr)
- Calculated time for removal of debris from GEO altitude to at least 300 km below GEO altitude
- Source for analysis if not this standard or the debris assessment software

## **Section 4: Assessment of Debris Generated by Explosions and Intentional Breakups**

### *4.1 Explosions From On-Board Stored Energy (Guidelines 4-1 and 4-2)*

To include

- Description of failure modes leading to explosion
- Description of systems involved in explosive failure, including the following:
  - Fluid: mass, chemical composition, pressure, energy density
  - Structure: size, materials, thickness, location relative to direction of motion, shielding from environment, most probable failure modes
- Estimated probability of explosion if quantified in assessment
- Detailed plan for safing structure after completion of mission

### *4.2 Intentional Breakups (Guidelines 4-3, 4-4, and 4-5)*

To include

- Description of energy source, total energy content
- Altitude and location of breakup
- Description of model for breakup
- Description of mission including state vector at explosion
- Calculated area-time for debris larger than 1 mm ( $m^2\text{-yr}$ )
- Plan for assessing risk to operating spacecraft from the debris cloud formed immediately after the test
- Documentation of dry run for risk analysis using predicted event time and current USSPACECOMMAND catalog elements

## **Section 5: Assessment of Debris Generated by On-Orbit Collisions**

### *5.1 Assessment of Collisions with Large Objects During Mission Operations (Guideline 5-1)*

To include

- Estimated probability of collision with intact space systems or large debris
- Plan for limiting probability, if applicable

### *5.2 Assessment of Collisions with Small Debris During Mission Operations (Guideline 5-2)*

To include

- Description of primary mission failure modes from meteoroid or orbital debris impact
- Description of design measures taken to protect against impacts with meteoroids or orbital debris, if applicable

## **Section 6: Assessment of Postmission Disposal Procedures and Systems**

### *6.1 Description of Postmission Disposal Option and Disposal System (Guidelines 6-1, 6-2, and 6-3)*

To include

- Statement of disposal option exercised
- Disposal plan and description of supporting systems (final orbit parameters,  $\Delta v$  requirement, disposal system design, etc.)
- Source for analysis if not this standard or the debris assessment software

*6.2 Assessment of Potential Failures that Prevent Successful Postmission Disposal (Guideline 6-4)*

To include

- Description of primary failure modes leading to loss of control during mission operations—from design or from impact with small debris
- Assessment of failure of the postmission disposal system to work properly

**Section 7: Assessment of Survival of Debris from the Postmission Disposal Atmospheric Reentry Option (Guideline 7-1)**

To include

- Verification that surviving debris is within guidelines
- Source for analysis if not this standard or the debris assessment software

**APPENDIX A**  
**DEFINITION OF TERMS**

## APPENDIX A

### DEFINITION OF TERMS

*Apogee* - The point in the orbit that is the farthest from the center of the Earth. The apogee altitude is the distance of the apogee point above the surface of the Earth.

*Apsis (pl. apsides)* - The point in the orbit where a satellite is at the lowest altitude (perigee) or at the highest altitude (apogee). The line connecting apogee and perigee is the *line of apsides*.

*Argument of perigee* - The angle between the line extending from the center of the Earth to the ascending node of an orbit and the line extending from the center of the Earth to the perigee point in the orbit measured from the ascending node in the direction of motion of the satellite.

*Ascending Node* - The point in the orbit where a satellite crosses the Earth's equatorial plane in passing from the southern hemisphere to the northern hemisphere.

*Cratering flux* - The number of impacts per square meter per year of debris objects which will leave a crater at least as large as a specified diameter.

*Debris flux* - The number of impacts per square meter per year expected on a randomly oriented planar surface of an orbiting space structure.

*Debris flux to limiting size* - The number of impacts per square meter per year of debris objects of a specified diameter or larger.

*Delta-v* - The change in the velocity vector caused by thrust measured in units of meters per second.

*Eccentricity* - The apogee altitude minus perigee altitude of an orbit divided by twice the semimajor axis. Eccentricity is zero for circular orbits and less than one for all elliptical orbits.

*f10* - An index of solar activity; a 13-month running average of the energy flux from the Sun measured at 10.7 cm, expressed in units of  $10^4$  Janskys.

*Geosynchronous orbit (GEO)* - An orbit with a period equal to the sidereal day. A circular GEO orbit with 0° inclination is a geostationary orbit, i.e., the nadir point is fixed on the Earth's surface. The altitude of a circular GEO orbit is 35,788 km. When GEO is referred to as an altitude it is that of circular GEO orbit.

*Geosynchronous transfer orbit (GTO)* - A highly eccentric orbit with perigee at LEO altitude and apogee near or above GEO altitude.

*Inclination* - The angle the orbit plane makes with the equatorial plane.

*Jansky* - A unit of electromagnetic power density equal to  $10^{-26}$  watts/m<sup>2</sup>/Hz.

*Line of apsides* - The line connecting the apogee and perigee points in an orbit. This line passes through the center of the Earth.

*Line of nodes* - The line formed by the intersection of the orbit plane with the Earth's equatorial plane. This line passes through the center of the Earth. The ascending node is the point where a satellite crosses the equator from the southern hemisphere to the northern hemisphere.

*Low Earth orbit (LEO)* - The region of space below the altitude of 2000 km.

*Meteoroids* - Naturally occurring particulates associated with solar system formation or evolution processes. Meteoroid material is associated with asteroid breakup or material released from comets.

*Orbit lifetime* - The length of time an object remains in orbit. Objects in LEO or passing through LEO lose energy as they pass through the Earth's upper atmosphere, eventually getting low enough in altitude that the atmosphere removes them from orbit.

*Orbital debris* - Man-made particulates released in orbit. In this document, only debris of diameter 1 mm and larger is considered.

*Penetration debris flux* - The number of impacts per square meter per year that will penetrate a surface of specified orientation with specified materials and structural characteristics.

*Perigee* - The point in the orbit that is nearest to the center of the Earth. The perigee altitude is the distance of the perigee point above the surface of the Earth.

*Right ascension of ascending node* - The angle between the line extending from the center of the Earth to the ascending node of an orbit and the line extending from the center of the Earth to the vernal equinox measured from the vernal equinox eastward in the Earth's equatorial plane.

*Semimajor axis* - Half the sum of the distances of apogee and perigee from the center of the Earth. Half the length of the major axis of the elliptical orbit.

*Semisynchronous Orbit (SSO)* - An orbit with a 12-hour period. A circular SSO is at altitude 19,133 km.

*Solar flux unit (sfu)* - Equal to  $10^4$  Janskys measured at a wavelength of 10.7 cm.

*Space debris* - Either meteoroid or orbital debris.

*Vernal equinox* - The direction of the Sun in space when it passes from the southern hemisphere to the northern hemisphere (on March 20 or 21) and appears to cross the Earth's equator. The vernal equinox is the reference point for measuring angular distance along the Earth's equatorial plane (right ascension), and one of two angles usually used to locate objects in orbit (the other being declination).

## **APPENDIX B**

### **BACKGROUND ON ORBITAL DEBRIS**

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### **BACKGROUND ON ORBITAL DEBRIS**

#### **I. Understanding the Threat of Collision**

Collectively referred to as space debris, objects in orbit can be divided into two categories: natural objects, or meteoroids, which are associated with the solar system; and man-made objects, or orbital debris, which result from operations in orbit about the Earth. Space debris ranges from dust-speck-size objects smaller than a micron to meter-size objects and larger. Almost all space debris larger than about 5 mm is orbital debris.

The threat of collision with orbital debris is an issue of growing concern as historically accepted practices and procedures allow man-made objects, some having the potential to explode, to accumulate in orbit. In the past, explosions have been the primary source of debris and are likely to continue to be so for the immediate future. However, current immediate modeling indicates that even if there is no increase in the number of launches per year, and spacecraft and upper stages continue to be left in orbit at the end of their mission, within the next 50 years collisions between large objects will become the major source of debris. Collisional processes will lead to a large increase in the amount of orbital debris capable of damaging or disabling operating spacecraft. The models currently being used to predict the orbital debris environment indicate that even if the number of launches per year increases only slightly (by approximately 5 launches per year) collisions between large objects in orbit will become a significant source of debris within the next 30 years.

The greatest risk in not controlling the debris environment is the onset of these collisions between large objects. There are two reasons for this. First, once collisions begin to occur, it will be almost impossible to halt the process and they will occur with increasing frequency—a process referred to as collisional cascading. Second, the energies in collisional breakup are much larger than in explosive breakup, in the megajoule (a few kilograms of TNT) to gigajoule (a few metric tons of TNT) range. This energy comes from the very large amount of chemical energy used to get objects into orbit. This large amount of expended energy creates many more debris fragments in all size ranges and spreads the debris over many hundreds of kilometers of altitude. This debris may hit other satellite surfaces, carrying impact energies of hundreds of megajoules per kilogram of impactor mass. At these energies, debris less than 1 mm in diameter, typically about 1 mg of mass, can penetrate an unshielded spacecraft surface and damage sensitive surfaces such as optics or thermal radiators; debris less than 1 cm (1 gm) can penetrate even a heavily shielded surface; and debris as small as 10 cm (1 kg) can cause a spacecraft to break up into debris fragments.

#### **II. The Presidential Directive to Limit Orbital Debris Generation**

On February 11, 1988, President Reagan issued a Presidential Directive on national space policy which included a requirement to limit the accumulation of orbital debris. This directive was the foundation for a coordinated effort among U.S. agencies and other nations to increase the understanding of the hazards caused by orbital debris and to establish effective techniques to manage the orbital debris environment. This effort has matured into the establishment of an International Technical Working Group on orbital debris. A strong consensus was reached within this working group which helped influence all space faring nations to voluntarily establish actions to limit orbital debris.

### III. A Theoretical Perspective on Managing Orbit Debris

Even though access to space and operations in space require a large expenditure of energy, any object left in space after it has performed its desired functions will still contain residual energy. This energy is comprised in large part of *kinetic energy*, and may also include additional *stored energy*, both chemical and mechanical. The kinetic energy results from both the high orbital velocity and the fact that objects are generally in different orbit planes. In low Earth orbit (LEO), the region of space up to 2,000 km altitude, average kinetic energy is about 50 megajoules per kilogram of mass in orbit.\* The stored energy (in the form of momentum devices, residual fuel, pressurized containers, or batteries, for example) is minimal by comparison—usually much less than a megajoule per kilogram. These sources of stored energy, however, can cause their associated structures to fragment, producing numerous smaller fragments, each still containing the 50 megajoules of kinetic energy per kilogram of fragment mass. The large number of fragments, combined with the relatively high kinetic energy for each fragment, creates a much larger risk to other spacecraft.

Fundamentally, the process of managing orbital debris is a process of managing this residual energy. Stored energy can be depleted before ending operations. Kinetic energy management, on the other hand, requires eliminating either mass or relative velocity. Since in most cases it is not possible to cause objects to orbit in a way which reduces the relative velocity, mass must not remain in a region where it can affect other space operations either directly or by fragmentation. In some cases, mass will be removed from orbit by natural forces, but in most removal must be planned. Even after debris control measures are instituted, however, there will be a residual debris environment from mass left in orbit before management of the environment began, from accidents occurring in orbit, and from mass being removed from the environment within the guidelines established for debris management. (The guidelines were designed to prevent growth of the debris environment, while minimizing the cost of compliance.) Consequently, spacecraft must protect themselves from becoming debris sources as a result of colliding with some of this mass. This will require adding "energy dissipation" shields in some cases to protect critical spacecraft components.

Therefore, a program mitigates its orbital debris contribution by controlling the energy it contributes to the orbital debris environment. The short-term environment can be managed by managing the stored chemical and mechanical energy within a spacecraft. This requires reliable designs to prevent explosions during operations. To prevent explosions after completion of mission operations, residual energy such as pressure, fuel, or mechanical energy must be vented or depleted. If spacecraft remain in the environment long enough, they will eventually be converted into fragments as a result of collisions. Consequently, the long-term management of the environment requires that objects be removed from useful orbital regimes at the end of mission life. This also means that objects must have sufficient reliability against orbital debris and other hazards to ensure that they can be removed before any fragmentation occurs.

### IV. Orbital Debris Modeling—Predicting the Probability of Collision

Over the last 10 years NASA has had a program which characterizes the current and future environment and which consists of a combination of models validated by measurements. A better understanding of the consequences of past operations has resulted from this program. We now know that within the approximately 2,000 km altitude LEO region there are billions of very small orbital debris fragments (0.1 mm and smaller, produced from solid rocket motor firings and degradation of spacecraft surfaces) that can erode spacecraft surfaces. More than 1 million of these objects are larger than 1 mm and can cause operational failure if they strike sensitive areas on a spacecraft. Any orbital debris 1 mm and larger comes from the more than 100 accidental and intentional fragmentations that have occurred since the inception of the space program. There are about 150,000 orbital debris objects larger than 1 cm that will

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\* This is roughly 25 times the energy content of TNT.

cause operational failure to a spacecraft regardless of where they strike; and there are about 15,000 objects in LEO larger than 10 cm which will catastrophically fragment any spacecraft they strike.

In the current environment, spacecraft failures caused by collisions with debris occur with probabilities of tenths to hundredths of a percent per year, depending on the mission orbit altitude and on the size of debris that could cause critical damage. Current models, however, project that unless measures are taken to limit the generation of debris, over the next 100 years the probabilities will grow at the rate of a few tenths to a few percent per year and will continue to increase rapidly. Thus the hazard imposed by debris will become almost as significant a cause of spacecraft loss as component failure, even after routine techniques are adopted to protect spacecraft. The guidelines defined in these volumes provide limits on debris generation that will halt the growth of the debris environment and ensure that debris does not become an increasingly significant cost factor for future space operations.

#### **V. Orbital Debris Measurements—Validating the Models**

Until recently comprehensive ground-based measurements could be made only for objects larger than 10 cm, with the primary source for this data being the satellite catalog maintained by USSPACECOMMAND. Therefore, the roughly 6,000 objects which are located in LEO and listed in the catalog account for only a small fraction of the debris which concerns spacecraft operators. However, even for this small fraction, only five percent of the objects in the catalog are operating spacecraft. Over half of the objects are breakup fragments from on-orbit explosions, which makes elimination of explosions the most effective near-term method for controlling the debris environment. About 40 percent of the objects in the catalog are spacecraft and upper stages that have been left in orbit at the end of their operational life. These objects, and others like them if future practice does not forbid it, are the greatest concern for long-term control of the debris environment since they can be turned into millions of fragments larger than 1 mm if they collide with other large structures. Contributing to the concern is the potential of many of these objects to have long orbit lifetimes. Spacecraft and upper stages are being abandoned at a rate greater than atmospheric drag is able to remove them. As a result there is a continuing growth in the mass and target area of man-made material in orbit and an increasing potential for the generation of debris that will inhibit future space operations.

For the geosynchronous Earth orbit (GEO) region, approximately 35,800 km altitude, the same issues are present. Measurement in this region is much more difficult, however, and the man-made debris environment is not as well characterized. It is thought that this environment is significantly less severe than LEO. Collision velocities in GEO are more characteristically 0.5 km/s rather than the 10 km/s in LEO. Consequently, a similar threat is posed by a collision with 1 cm debris in GEO as is posed by a collision with 1 mm debris in LEO. Breakups have occurred in GEO, but whether or not they occurred with a frequency comparable to that in LEO is currently being investigated.

The region between LEO and GEO contains mainly upper stages and debris objects in geosynchronous transfer orbit (GTO) or spacecraft in high-inclination, high-eccentricity mission orbits such as the Soviet communication satellites. However, there are several programs using near-circular orbits with 12-hour periods and there are no measurements indicating breakups of objects in this orbit within this altitude range.

#### **VI. Placing Others at Risk**

The growth in the amount of debris in space and the rapidly increasing hazard it poses demand that users of space exercise responsibility in both the design and operational phases of space missions. To decrease risks to others, space users must avoid the following risk-creating events:

- *Explosions in orbit.* Explosions produce a large number of debris fragments capable of causing single-event failure of an operating spacecraft, as well as a still larger number of smaller debris fragments capable of degrading the performance of a spacecraft. The velocities imparted to the debris on breakup may create a significant risk to spacecraft operating hundreds of kilometers above or below the breakup altitude, and may place debris in orbits with very long lifetimes.
- *Damaging collisions with debris during mission operations.* This most likely will occur with a small piece of debris, leading to loss of control of the spacecraft. However, it could be a collision with a large piece of debris, leading to catastrophic breakup. This is primarily a problem for programs having large spacecraft with long mission lives.
- *Failure to remove a structure from a high value region of space at the end of useful life.* Failure to remove from orbit nonfunctional objects, each of which becomes a potential source of small debris that could affect future space operations in that region, is the operational procedure in most cases today.
- *Leaving operational debris in the environment.* Such debris fragments, while small in number, are generally larger than 1 cm and represent a risk of single-event failure to operating spacecraft. These objects will remain in orbit for months to years if left at low altitude, for tens to hundreds of years if released at altitudes typical for Sun-synchronous missions, and for a virtually unlimited period of time if released above this altitude.
- *Impacting the Earth's surface.* This danger occurs when components or structures from a spacecraft or upper stage survive atmospheric reentry.