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# Orbital Debris Environment for Spacecraft Designed to Operate in Low Earth Orbit

Donald J. Kessler, Robert C. Reynolds,
and Phillip D. Anz-Meador

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# Orbital Debris Environment for Spacecraft Designed to Operate in Low Earth Orbit

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## ABSTRACT

The orbital debris environment model contained in this report is intended to be used by the spacecraft community for the design and operation of spacecraft in low Earth orbit. This environment, when combined with material-dependent impact tests and spacecraft failure analysis, is intended to be used to evaluate spacecraft vulnerability, reliability, and shielding requirements. The environment represents a compromise between existing data to measure the environment, modeling of this data to predict the future environment, the uncertainty in both measurements and modeling, and the need to describe the environment so that various options concerning spacecraft design and operations can be easily evaluated.


## INTRODUCTION

The natural meteoroid environment has historically been a design consideration for spacecraft. Meteoroids are part of the interplanetary environment and sweep through Earth orbital space at an average speed of 20 km/s. At any one instant, a total of 200 kg of meteoroid mass is within 2000 km of the Earth's surface. Most of this mass is concentrated in 0.1 mm meteoroids.

Within this same 2000 km above the Earth's surface, however, is an estimated 3,000,000 kg of man-made orbiting objects. These objects are in mostly high inclination orbits and sweep past one another at an average speed of 10 km/s. Most of this mass is concentrated in approximately 3000 spent rocket stages, inactive payloads, and a few active payloads. A smaller amount of mass, approximately 40,000 kg, is in the remaining 4000 objects currently being tracked by U.S. Space Command radars. Most of these objects are the result of more than 90 on-orbit satellite fragmentations. Recent ground telescope measurements of orbiting debris combined with analysis of hypervelocity impact pits on the returned surfaces of the Solar Maximum Mission satellite indicate a total mass of approximately 1000 kg for orbital debris sizes of 1 cm or smaller, and approximately 300 kg for orbital debris smaller than 1 mm. This distribution of mass and relative velocity is sufficient to cause the orbital debris environment to be more hazardous than the meteoroid environment to most spacecraft operating in Earth orbit below 2000 km altitude.

Mathematical modeling of this distribution of orbital debris predicts that collisional fragmentation will cause the amount of mass in the 1 cm and smaller size range to grow at twice the rate as the accumulation of total mass in Earth orbit. Over the past 10 years, this accumulation has increased at an average rate of 5 percent per year, indicating that the small sizes should be expected to increase at 10 percent per year. Reasons that both of these rates could be either higher or lower, as well as other uncertainties in the current and projected environment, are discussed in the section "Uncertainty in Debris Flux." As new data become available, a new environment will be issued.


The authors wish to acknowledge those responsible for original research and data analyses utilized by this work. Dr. Andrew Potter, Mr. John Stanley, Dr. Karl Henize, Dr. Faith Vilas, and Mr. Eugene Stansbery (all at NASA/JSC) obtained and analyzed data obtained using optical telescopes, infrared telescopes, and radar on individual debris fragments. These data were used to evaluate the relationship between the physical size, radar cross-section, and optical brightness. Dr. Gautam Badhwar (NASA/ JSC) analyzed the atmospheric drag characteristics of individual fragments to evaluate the relationship between physical size and mass. Dr. David McKay (NASA/JSC) and Mr. Ron Bernhard (LESC) analyzed and compiled much of these impact data on the recovered surfaces from the Solar Maximum Mission (SMM) satellite.

## DATA SOURCES

The following data sources were considered in the construction of this environmental model:

1. Orbital element sets supplied by the U.S. Space Command (both the cataloged population and those objects awaiting cataloging) for the period between 1976 and 1988

2. Optical measurements by the Massachusetts Institute of Technology (MIT) in 1984 using the telescopes of their experimental test site (ETS) in Socorro, New Mexico

3. Measurements designed to determine orbital debris particle albedo using a ground-based infrared telescope at the Air Force Maui Optical Station/Maui Optical Tracking and Identification Facility (AMOS/MOTIF), U.S. Space Command radars, and both NASA and U.S. Space Command telescopes

4. Analysis of hypervelocity impacts on the surfaces returned by the Shuttle from the repaired Solar Maximum Mission satellite in 1984

5. Mathematical models which consider various traffic models and satellite fragmentation processes to predict the future accumulation of debris

## KEY ASSUMPTIONS AND CONCLUSIONS CONCERNING DATA SOURCES

The following assumptions and/or conclusions were made or reached concerning the above data sources:

1. The flux resulting from the U.S. Space Command orbital element sets is complete to a limiting size of 10 cm for objects detected below 1000 km altitude.

2. During average seeing conditions, the MIT telescopes detected objects to a limiting size of 5 cm in diameter at a rate of two times the rate of 10 cm and larger objects.

3. During optimum seeing conditions, the MIT telescopes detected objects to a limiting size of 2 cm in diameter at a rate of five times the rate of 10 cm and larger objects.

4. The surfaces of the Solar Maximum Mission satellite experienced an orbital debris flux which varies from 20 percent of the meteoroid flux for debris sizes larger than 0.05 cm to a factor of 1000 times the meteoroid flux for sizes larger than 1 μm.

5. The orbital debris flux between 0.05 cm and 2 cm can be obtained by a linear interpolation (on a $\log_{10} F$ (flux) vs $\log_{10} d$ (diameter) plot) of the Solar Maximum Mission satellite surface data and the MIT telescope data.

6. For any given size of orbital debris, the variation of flux with altitude, solar activity, orbital inclination, and the velocity and direction distribution is the same as that predicted by the U.S. Space Command orbital element set data.

7. The accumulation of objects tracked by the U.S. Space Command, when averaged over an 11-year solar cycle, will increase at a rate of 5 percent per year.

8. The accumulation of objects detected by the MIT telescopes and the Solar Maximum Mission satellite surfaces, when averaged over an 11-year solar cycle, will increase at twice the rate of the tracked objects, or 10 percent per year.

## DESIGN STANDARDS

### Recommended Flux for Orbital Debris

The cumulative flux of orbital debris of size d and larger on spacecraft orbiting at altitude h, inclination i, in the year t, when the solar activity for the previous year is S, is given by the following equation:

$$F(d,h,i,t,S) = k \cdot \phi(h,S) \cdot \psi(i) \cdot [F_1(d) \cdot g_1(t) + F_2(d) \cdot g_2(t)] \tag{1}$$

where

$F$ = flux in impacts per square meter of surface area per year
$k$ = 1 for a randomly tumbling surface; must be calculated for a directional surface
$d$ = orbital debris diameter in cm
$t$ = time expressed in years
$h$ = altitude in km ($h \leq 2000$ km)
$S$ = 13-month smoothed 10.7 cm-wavelength solar flux expressed in $10^4$ Jy; retarded by 1 year from t
$i$ = inclination in degrees

and

$\phi(h,S) = \phi_1(h,S) / (\phi_1(h,S) + 1)$

$\phi_1(h,S) = 10^{(h/200 - S/140 - 1.5)}$

$F_1(d) = 1.05 \times 10^{-5} \cdot d^{-2.5}$

$F_2(d) = 7.0 \times 10^{10} \cdot (d + 700)^{-6}$

$p$ = the assumed annual growth rate of mass in orbit,

$g_1(t) = (1 + 2 \cdot p)^{(t - 1985)}$

$g_2(t) = (1 + p)^{(t - 1985)}$

The inclination-dependent function $\psi$ is a ratio of the flux on a spacecraft in an orbit of inclination i to that flux incident on a spacecraft in the current population's average inclination of approximately 60°. Values for $\psi$ are given in figure 1 and tabulated in table 1.

An average 11-year solar cycle has values of S which range from 70 at solar minimum to 150 at solar maximum. However, the current cycle, which peaks in the year 1990, is predicted to be above average, possibly exceeding 200.

An example orbital debris flux is compared with the meteoroid flux from NASA SP8013 in figure 2 for $h = 500$ km, $t = 1995$ years, $k = 1.0$, $i = 30°$, and $S = 90.0$.

The flux is defined such that the average number of impacts N on a spacecraft surface area of A exposed to the environment for the interval $t_i$ to $t_f$ is given by the following equation:

$$N = \int_{t_i}^{t_f} F \cdot A \, dt \qquad (2)$$

where A is the surface area exposed to the flux F at time t.

The value of k can theoretically range from 0 to 4 (a value of 4 can only be achieved when a surface normal vector is oriented in the direction of a monodirectional flux), and depends on the orientation of A with respect to the Earth and the spacecraft velocity vector. If the surface is randomly oriented, then k = 1. If the surface is oriented with respect to the Earth, then the section "Velocity and Direction Distribution" must be used to calculate a value for k. In general, if the surface area is facing in the negative velocity direction, k = 0. However, if this area is facing in the same direction as the spacecraft velocity vector, and the spacecraft orbital inclination is near polar (which causes more "head-on" collisions), then k will approach its maximum value of approximately 3.5 for the current directional distribution.

The probability of exactly n impacts occurring on a surface is found from Poisson statistics, or

$$P_n = \frac{N^n}{n!} \cdot e^{-N} \qquad (3)$$

Uncertainty in Debris Flux

Factors which contribute significantly to the uncertainty in the orbital debris environment are inadequate measurements, an uncertainty in the level of future space activities, and the statistical character of major debris sources. The environment has been adequately measured by ground radars for orbital debris sizes larger than 10 cm. A limited amount of data using ground telescopes has shown a 2 cm flux which is currently estimated to be known within a factor of 3. Orbital debris sizes smaller than .05 cm have only been measured at 500 km; at this altitude and for these smaller sizes, the environment is known within a factor of 2. Interpolation was used to obtain the flux between 0.05 cm and 2 cm at 500 km, and would be justified if the amount of mass between these two sizes were approximately the same as the mass contributing to the two sizes, or approximately 100 kg to 1000 kg. Mathematical modeling of various types of satellite breakups in Earth orbit make such an assumption seem reasonable. However, other than "reasonableness," there are no data which would prevent the flux of any particle in the size range between 0.05 cm and 2 cm from being as high as the 0.05 cm flux, or as low as the 2 cm flux, that is, vary by as much as several orders of magnitude.

An additional uncertainty from the measurements arises because there are no measurements of debris smaller than 2 cm at other than 500 km altitude. Mathematical modeling concludes that if the debris is in near circular orbits and the source of the debris is at higher altitudes, the ratio of the amount of small debris to large debris should decrease with decreasing altitude. This ratio is assumed constant in the design environment. Consequently, there would be a smaller flux of less than 2 cm debris at altitudes less than 500 km, and a larger flux at altitudes above 500 km than is predicted by this model. However, if the debris is in highly elliptical orbits, then the flux of small debris could be nearly independent of altitude. Consequently, the amount that the flux differs from the design environment could be as high as a factor of 10 (either higher or lower) for every 200 km away from the 500 km altitude, up to an altitude of approximately 700 km. The large number of breakups at altitudes between 700 km and 1000 km and at 1500 km, together with the extremely long orbital lifetimes of fragments in these regions, make any predictions very sensitive to the nature of each of these breakups. The U.S. Space Command data give fluxes at 800 km and 1000 km which are twice as high as predicted by the recommended flux model, as shown in figure 3. For most altitudes between 1000 km and 2000 km, the current flux from objects tracked by the U.S. Space Command is significantly lower than the design environment. However, the large number of breakups at 1500 km could have scattered smaller fragments over this region; in addition, future traffic may increase the flux of larger objects.

Predicting future activity in space is highly uncertain. Since 1966, the non-U.S. launch rate has increased by a average of 10 percent per year; however, U.S. launch rates have decreased at this same rate, leading to a constant world launch rate since 1966. This constant launch rate has led to a linear, or decreasing percentage, growth in the accumulation of objects being tracked by the U.S. Space Command. Averaged over the last solar cycle, this accumulation has grown at an average rate of 5 percent per year. A continued constant launch rate would mean that the value of "p" in the expression for $g_2$ could either decrease from 0.05 with time, or the growth rate could follow the linear functional form $g_2(t) = 1 + p \cdot (t - 1985)$. On the other hand, traffic models which represent the best current estimate of future space activities up to the year 2010, would lead to between a 5 percent and 10 percent per year increase in the amount of U.S. mass to orbit. In addition, some U.S. and world traffic projections would give rise to increases in the accumulation of larger objects in orbit as high as 20 percent per year. While such large increases do not seem historically justified, an upper limit of a 10 percent increase per year, or $p = 0.1$, is not unrealistic. Any larger increases in the use of low Earth orbit would likely include different operational techniques which would invalidate assumptions used to express the design environment.

Predicting the population not tracked by the U.S. Space Command is even more uncertain since we do not even have historical data to extrapolate. However, there are some indicators. Historically, the satellite fragmentation rate has increased with time, indicating that values for $g_1$ would increase with time faster than values for $g_2$. However, actions are currently underway which should reduce the future satellite explosion rate. On the other hand, mathematical models predict that within the very near future, random collisions could become an important cause of satellite fragmentations. Under these conditions, the small debris population would increase at approximately twice the percentage rate of the large population, until a "critical density" of large objects is reached. This critical density corresponds to a value of $g_2$ between 10 and 100 (i.e., the tracked population is 10 to 100 times its 1985 total number). At this time, values for $g_1$ would increase very rapidly with time, independent of values for $g_2$.

The design environment assumes that the value of $g_1$ increases at twice the percentage rate of $g_2$. This could be expected if the satellite explosion rate continues to increase over the next decade or two. After this time random collisions would cause the rate to continue, independent of actions to reduce the explosion frequency. For values of p greater than 0.1, random collisions would become important in less than a decade, again consistent with the environment assumption. However, if the explosion rate is immediately reduced, and the current rate at which mass is placed into orbit does not significantly increase, then the design environment will predict fluxes for debris sizes smaller than 10 cm over the next 10 to 20 years which are too high by a factor of 2 to 10.

## Average Mass Density

The average mass density for debris objects 1 cm in diameter and smaller is 2.8 g/cm$^3$. The average mass density for debris larger than 1 cm is based on observed breakups, area-to-mass calculations derived from observed atmospheric drag, ground fragmentation tests, and known intact satellite characteristics.

This density has been found to fit the following relationship:

$$\rho = 2.8 \cdot d^{-0.74} \tag{4}$$

<u>Velocity and Direction Distribution</u>

Averaged over all altitudes the non-normalized collision velocity distribution, i.e., the number of impacts with velocities between v and v + dv, relative to a spacecraft with orbital inclination i is given by the following equations:

$$f(v) = (2 \cdot v \cdot v_o - v^2) \cdot (G \cdot e^{-((v - A \cdot v_o)/(B \cdot v_o))^2} + F \cdot e^{-((v - D \cdot v_o)/(E \cdot v_o))^2}) + H \cdot C \cdot (4 \cdot v \cdot v_o - v^2) \tag{5}$$

where v is the collision velocity in km/s, A is a constant, and B, C, D, E, F, G, H, and $v_o$ are functions of the orbital inclination of the spacecraft. The values for these constants and parameters are as follows:

$$A = 2.5$$

$$B = \begin{cases} 0.5 & i<60 \\ 0.5-0.01 \cdot (i - 60) & 60<i<80 \\ 0.3 & i>80 \end{cases}$$

$$C = \begin{cases} 0.0125 & i<100 \\ 0.0125 + 0.00125 \cdot (i - 100) & i>100 \end{cases}$$

$$D = 1.3-0.01 \cdot (i - 30)$$

$$E = 0.55 + 0.005 \cdot (i - 30)$$

$$F = \begin{cases} 0.3 + 0.0008 \cdot (i - 50)^2 & i<50 \\ 0.3 - 0.01 \cdot (i - 50) & 50<i<80 \\ 0.0 & i>80 \end{cases}$$

$$G = \begin{cases} 18.7 & i<60 \\ 18.7 + 0.0289 \cdot (i - 60)^3 & 60<i<80 \\ 250.0 & i>80 \end{cases}$$

$$H = 1.0 - 0.0000757 \cdot (i - 60)^2$$

$$v_o = \begin{cases} 7.25 + 0.015 \cdot (i - 30) & i<60 \\ 7.7 & i>60 \end{cases}$$

When $f(v)$ is less than zero, the function is to be reset equal to zero. An example for $i = 30°$ is given in figure 4.

The user may find it convenient to numerically normalize $f(v)$ so that

$$f'(v) = \frac{f(v)}{\int_0^\infty f(v)\,dv} \tag{6}$$

When normalized in this manner, $f'(v)$ over any 1 km/s velocity interval becomes the fraction of debris impacts within a 1 km/s incremental velocity band. Any average velocity moment may be defined as

$$\overline{v^n} = \int_0^\infty v^n \cdot f'(v)\,dv \tag{7}$$

The direction of impact can be approximated by using this velocity distribution and assuming that it results from the intersection of the spacecraft velocity vector and another circular orbit. That is, all velocity vectors will be in a plane tangent to the Earth's surface, and will appear to be from a direction relative to the spacecraft velocity vector. The direction of the velocity vector is given by the relationship:

$$\cos\theta = -\frac{v}{15.4} \tag{8}$$

where $\theta$ is the angle between the impact velocity vector and the spacecraft velocity vector, and v is the impact velocity. Since a spacecraft velocity of 7.7 km/s was used to calculate relative velocity, this velocity was used to determine the value of 15.4 ($2 \times 7.7$) given in equation 8. A value for k (defined in the section "Recommended Flux for Orbital Debris") is found by integrating over the values of $\theta$ that an oriented surface may be impacted. An example for $i = 30°$ is given in figure 5, where the surface normal vector is located in a plane parallel to the Earth's surface, and has an angle $\gamma$ to the spacecraft velocity vector.

## Uncertainty in Velocity and Direction Distribution

The impact velocity and direction distributions are fundamentally functions of the orbital debris inclination distribution. The inclination distribution changes with time and altitude, and can change significantly as the result of a breakup at any particular altitude. Since the orbits of future breakups cannot be predicted, variables such as the altitude of the spacecraft are of secondary importance. Therefore, the most important variable is the inclination of the spacecraft. However, the velocity distribution will change with time and position in space. These changes could affect the average velocity from the distribution by several km/s.

The fact that orbital debris objects are not in exactly circular orbits will introduce a small uncertainty for most velocities. As a result of the currently small eccentricities of these orbits, the actual direction of impacts are within 1° for most velocities derived from the section "Velocity and Direction Distribution." For low velocities (less than 2 km/s), the uncertainty in direction is much larger, with a significant fraction being more than 20° from the direction derived from the section "Velocity and Direction Distribution." This error in direction can be in the local horizontal plane or can appear as direction errors above or below this plane. High velocity impacts will almost always occur very near to the local horizontal plane and from the forward (downrange) direction; low speed impacts can occur from almost any angle ($0° \leq$ angle $\leq 180°$) in the local horizontal plane as well as at considerable angles ($0° \leq$ angle $\leq 90°$) out of that plane.

## Flux Resulting from Possible Future Inadvertent Breakups

The flux arising from the intentional or inadvertent fragmentation of an artificial Earth satellite in low Earth orbit (LEO) presents a hazard to other satellites. In the region of the breakup, an enhanced flux may be apparent for a considerable period of time, depending upon the altitude of the breakup, and the size and velocity distribution of the debris.

The flux for a particle of mass m may be represented by the equation:

$$F_b = 1 \times 10^{-9} \cdot \phi_b \cdot f \cdot (M/m) \tag{9}$$

where $F_b$ is the flux of impacting fragments per square meter of surface per year, M is the total mass of the parent satellite, m is the mass of individual fragments in the same units as M, and f is the fraction of the total mass going into a fragment size characterized by m. This fraction may be derived from any differential number/mass distribution. The dimensionless quantity $\phi_b$ is a function of distance from the breakup altitude and the velocity of the ejecta from the center of mass; values for $\phi_b$ are given in figure 6.

To obtain values for $\phi_b$, it was assumed that the breakup fragments were ejected in all directions from the center of mass of the parent object with a distribution of velocities. This distribution was assumed to have a "peak" or "most probable" velocity given by $v_p$ with the distribution linearly reducing to zero at $0.1 \cdot v_p$ and $1.3 \cdot v_p$ (i.e., on a number vs. velocity plot, the distribution is shaped like a triangle with the peak of the triangle at $v_p$ and a base range of $0.1 \cdot v_p$ to $1.3 \cdot v_p$). Using this distribution of velocities, new orbits were calculated to obtain flux as a function of altitude. This flux distribution was then normalized and is depicted in figure 6.

The ejection velocity should not be confused with the collision velocity. The only time these two velocities would be identical is for the first few days following a breakup, and the object which fragmented is in the same orbit as the satellite at risk. However, the nodal crossing point of all orbits will precess at different rates, so that the collision velocity will increase with time. After a few years, the collision velocity would be close to the general case which depends on the orbital inclination. Inclinations greater than 30° will yield collision velocities of 7 km/s or greater. In general, the collision velocity will be similar to those given in the section "Velocity and Direction Distribution" for most cases.

The time for the flux to decay to $e^{-1}$ its initial value, or its "half- life" H, for a 1 cm aluminum sphere and solar activity of S = 110, is given as a function of altitude in figure 7. When the breakup altitude is above the operational altitude, use the operational altitude to determine the half-life. If the breakup altitude is below the operational altitude, use the breakup altitude to determine the half-life. The half-life is proportional to the particle mass-to-area ratio, so that the half-life of other sizes can be derived. The total number of impacts resulting from a breakup is then

$$N_b = F_b \cdot A \cdot H \tag{10}$$

where A is the surface area of a randomly oriented surface. Given the inclination of the breakup, both velocity and direction could be derived.

## DISCUSSION: AN EXAMPLE OF A FUTURE BREAKUP

When a satellite breaks up in space, its size and velocity distribution are a sensitive function of the type of breakup. If it were a low intensity explosion, nearly all of the fragment mass would be in sizes larger than approximately 10 cm, and the most probable ejection velocity would likely be approximately 50 m/s. The fragments from a hypervelocity collision would include a significant fraction of mass with sizes less than 10 cm. However, the most probable velocities of these fragments would increase with decreasing size. Most of the fragments from a high intensity explosion could go into almost any preferred size, depending on the nature of the explosion.

As an example, assume that half of the mass from a 1000 kg satellite goes into 1 cm fragments. Also, assume that the satellite fragmented at an altitude of 600 km, and that the probable ejection velocity was 150 m/s. The resulting flux of 1 cm fragments at 500 km would be $5 \times 10^{-5}$ impacts/$m^2$-yr. This is larger (by several factors) than the flux predicted at 500 km for 1995, given in the section "Recommended Flux for Orbital Debris." However, assuming no additional breakups occur, this larger flux will effectively last for only 3 years, as shown in figure 7.

## REFERENCES

Many of the assumptions and analyses in this report are based upon unpublished work conducted by the Solar System Exploration Division at NASA's Lyndon B. Johnson Space Center, and upon the unpublished orbital element sets provided by the U.S. Space Command. Published material which could provide the reader additional background material can be found in the following references:

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TABLE 1.- THE FLUX ENHANCEMENT FACTOR Ψ (i)

| Inclination (degrees) | Ψ (i) | Inclination (degrees) | Ψ (i) | Inclination (degrees) | Ψ (i) |
|---|---|---|---|---|---|
| 25 | 0.900 | 58 | 1.075 | 92 | 1.400 |
| 26 | 0.905 | 59 | 1.080 | 93 | 1.440 |
| 27 | 0.910 | 60 | 1.090 | 94 | 1.500 |
| 28 | 0.912 | 61 | 1.100 | 95 | 1.550 |
| 28.5 | 0.9135 | 62 | 1.115 | 96 | 1.640 |
| 29 | 0.915 | 63 | 1.130 | 97 | 1.700 |
| 30 | 0.920 | 64 | 1.140 | 98 | 1.750 |
| 31 | 0.922 | 65 | 1.160 | 99 | 1.770 |
| 32 | 0.927 | 66 | 1.180 | 100 | 1.780 |
| 33 | 0.930 | 67 | 1.200 | 101 | 1.770 |
| 34 | 0.935 | 68 | 1.220 | 102 | 1.750 |
| 35 | 0.940 | 69 | 1.240 | 103 | 1.720 |
| 36 | 0.945 | 70 | 1.260 | 104 | 1.690 |
| 37 | 0.950 | 71 | 1.290 | 105 | 1.660 |
| 38 | 0.952 | 72 | 1.310 | 106 | 1.610 |
| 39 | 0.957 | 73 | 1.340 | 107 | 1.560 |
| 40 | 0.960 | 74 | 1.380 | 108 | 1.510 |
| 41 | 0.967 | 75 | 1.410 | 109 | 1.460 |
| 42 | 0.972 | 76 | 1.500 | 110 | 1.410 |
| 43 | 0.977 | 77 | 1.630 | 111 | 1.380 |
| 44 | 0.982 | 78 | 1.680 | 112 | 1.350 |
| 45 | 0.990 | 79 | 1.700 | 113 | 1.320 |
| 46 | 0.995 | 80 | 1.710 | 114 | 1.300 |
| 47 | 1.000 | 81 | 1.700 | 115 | 1.280 |
| 48 | 1.005 | 82 | 1.680 | 116 | 1.260 |
| 49 | 1.010 | 83 | 1.610 | 117 | 1.240 |
| 50 | 1.020 | 84 | 1.530 | 118 | 1.220 |
| 51 | 1.025 | 85 | 1.490 | 119 | 1.200 |
| 52 | 1.030 | 86 | 1.450 | 120 | 1.180 |
| 53 | 1.040 | 87 | 1.410 | 121 | 1.165 |
| 54 | 1.045 | 88 | 1.390 | 122 | 1.155 |
| 55 | 1.050 | 89 | 1.380 | 123 | 1.140 |
| 56 | 1.060 | 90 | 1.370 | 124 | 1.125 |
| 57 | 1.065 | 91 | 1.380 | 125 | 1.110 |

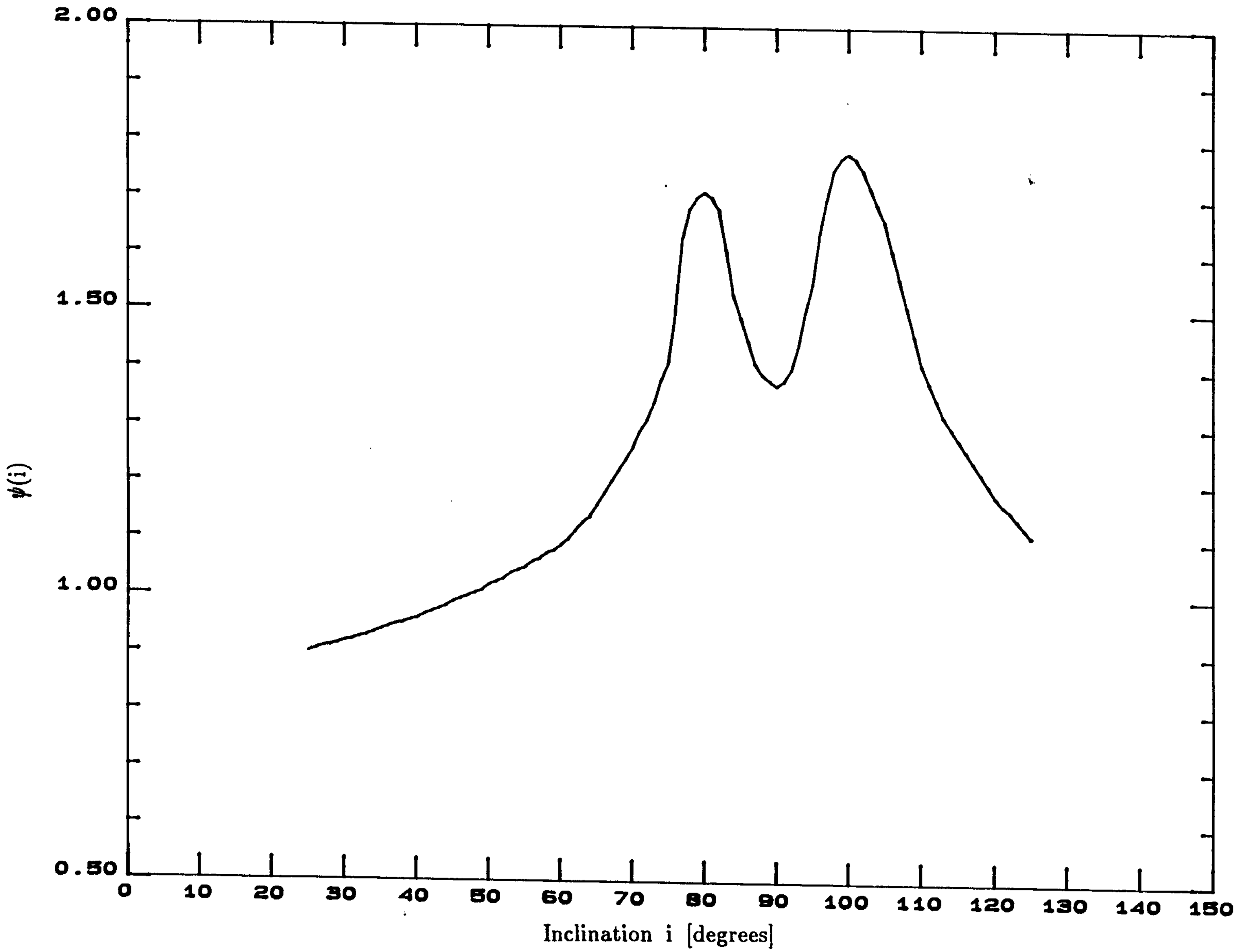


Figure 1.- The inclination scaling factor Ψ (i).

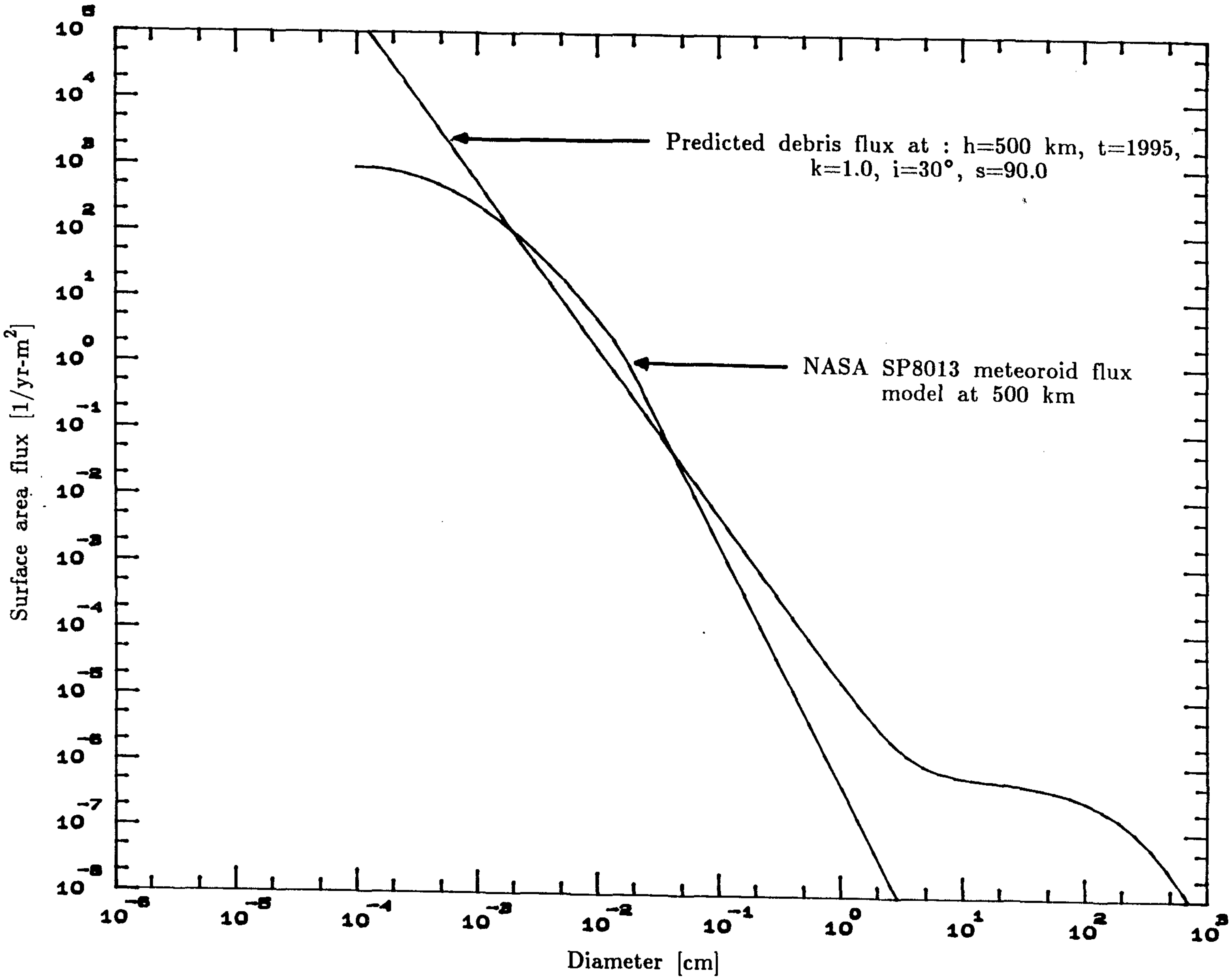


Figure 2.- A comparison of the predicted debris flux and the NASA SP8013 meteoroid flux model (including Earth shielding) at 500 km.

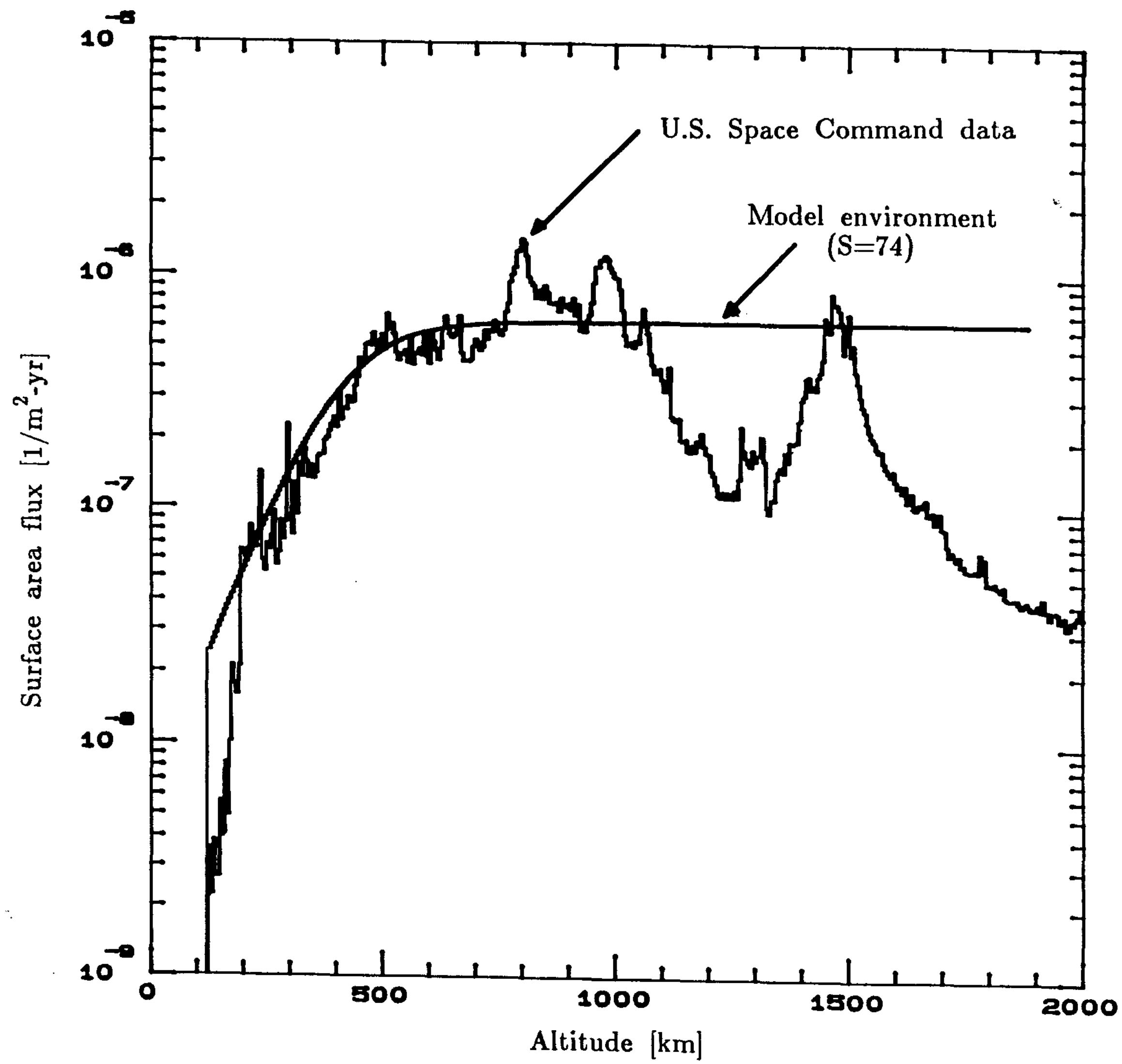


Figure 3.- Flux resulting from all objects tracked by the U.S. Space Command compared to the model environment for 10 cm objects in January 1987. An orbital inclination of 60° was assumed.

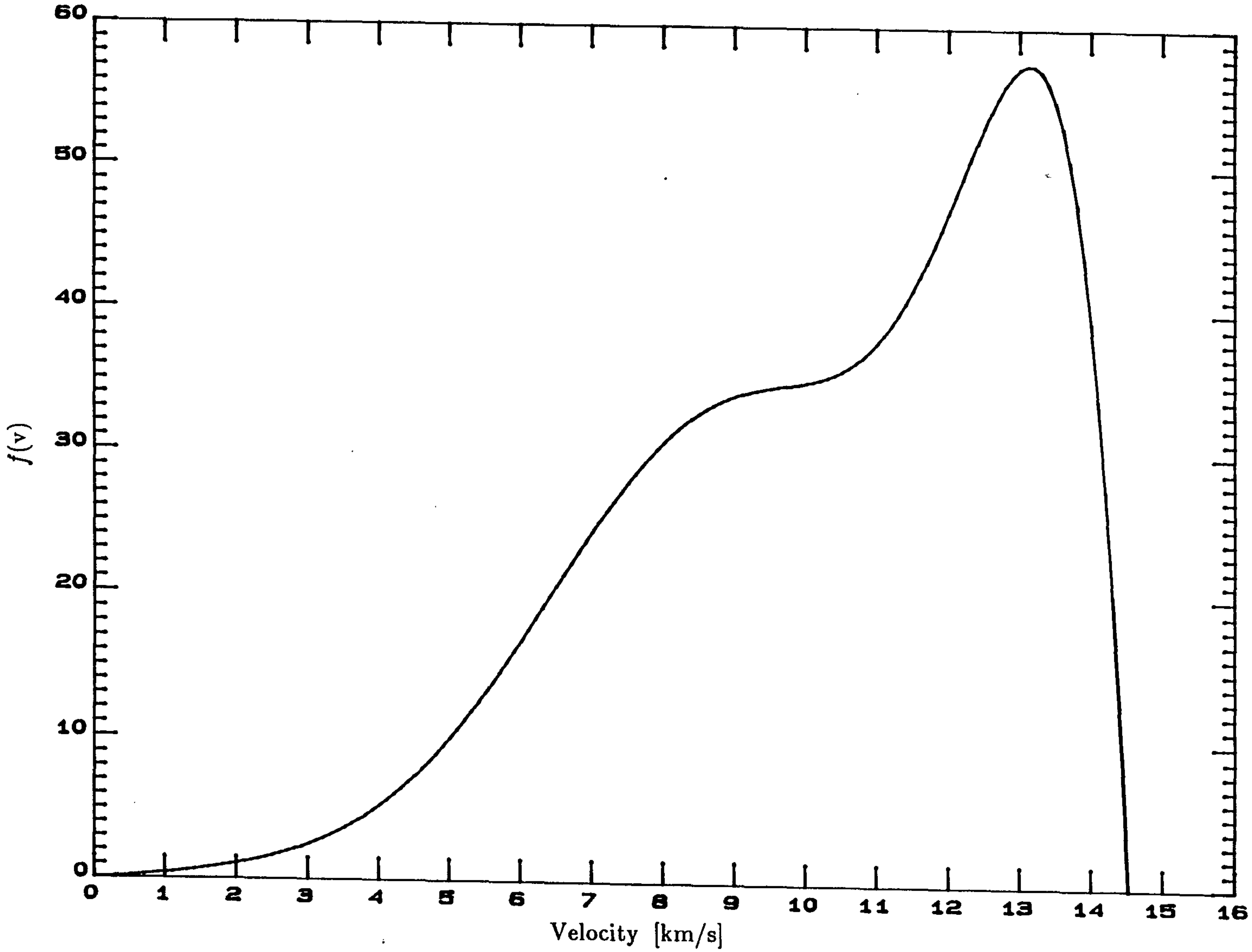


Figure 4.- The un-normalized collision velocity distribution for an inclination at 30°.

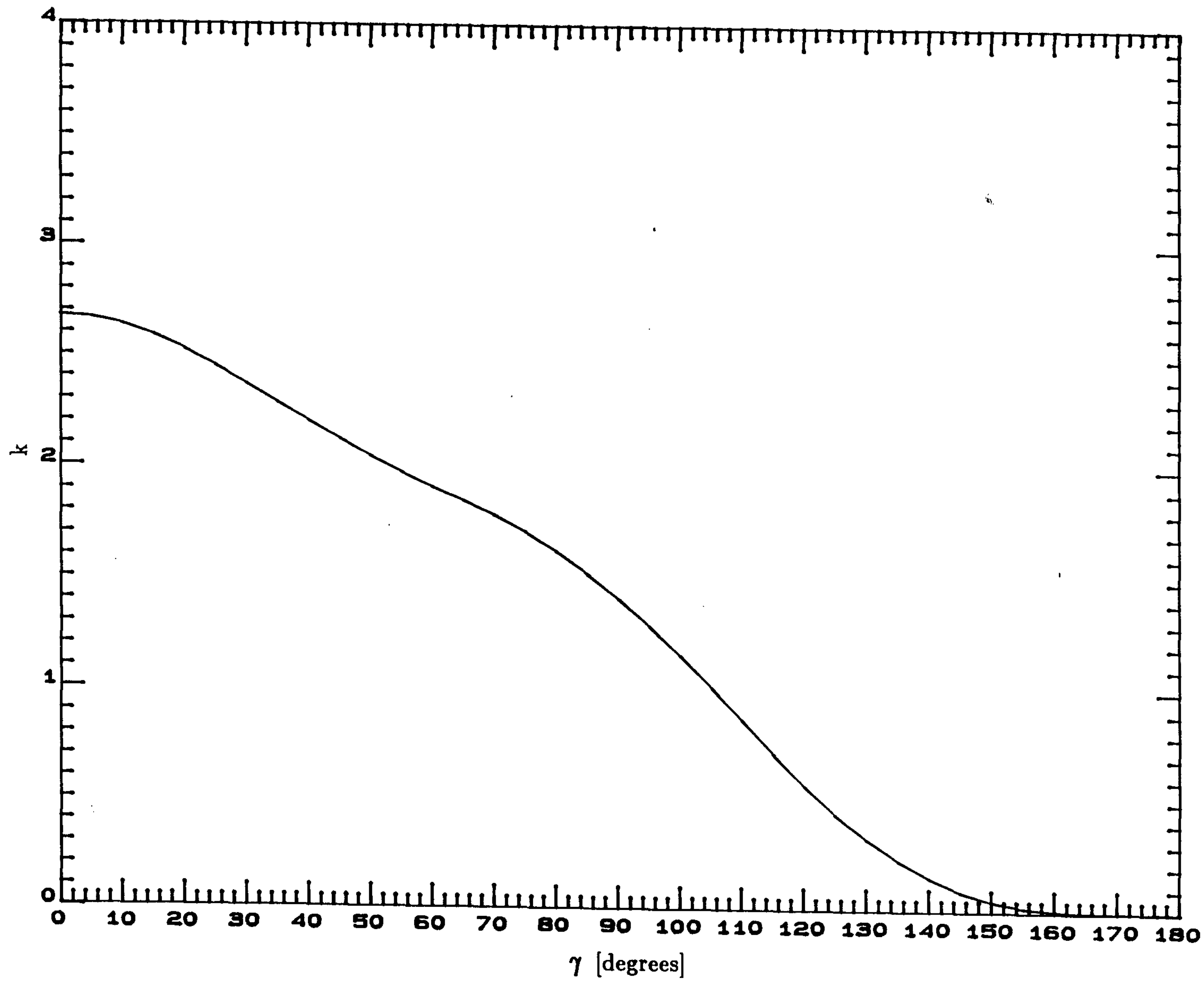


Figure 5.- The *ratio of the flux on a spacecraft* surface whose normal vector is oriented γ degrees to the spacecraft velocity vector to the flux on a randomly-oriented surface for an inclination at 30°.

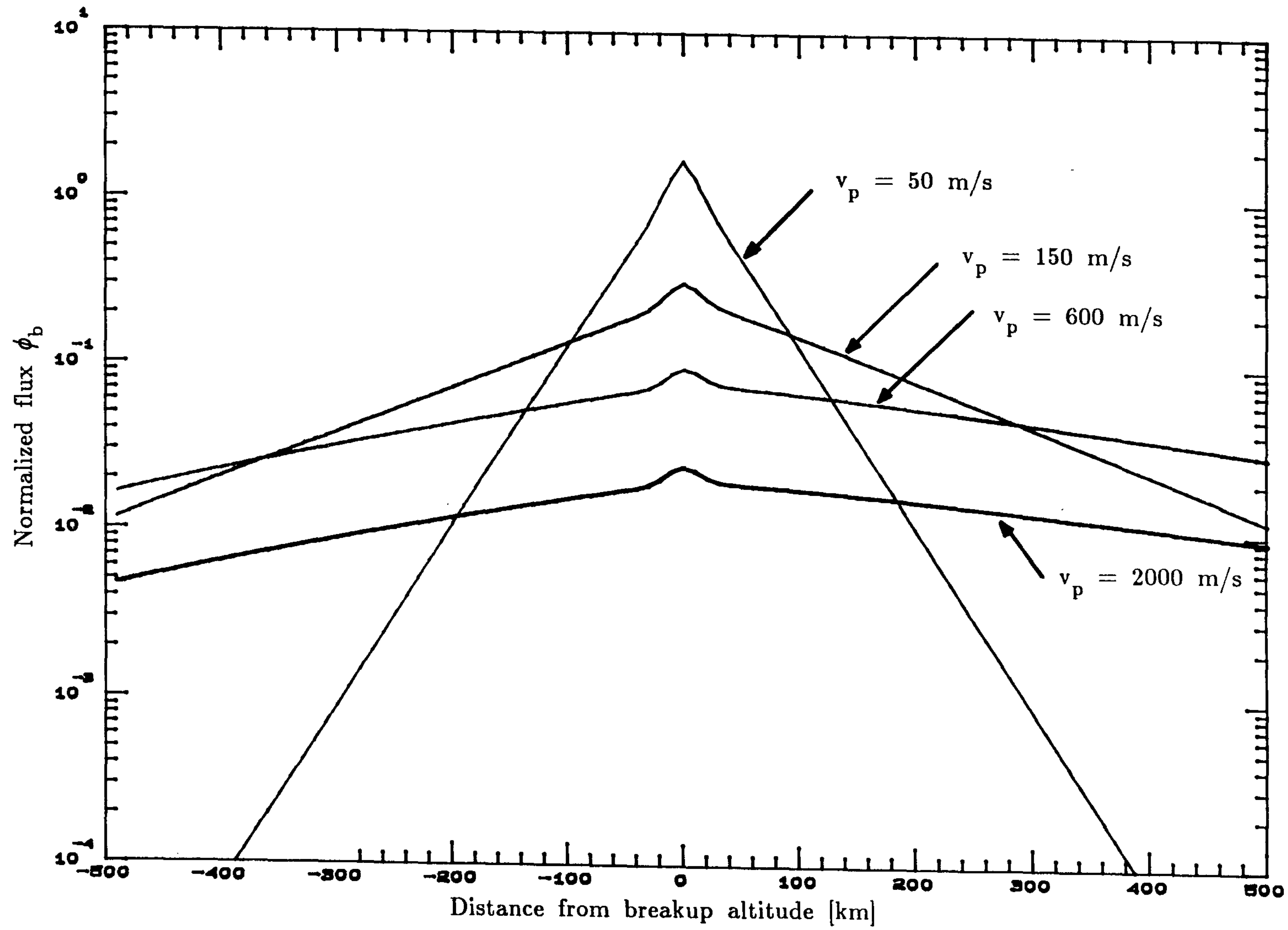


Figure 6.- The debris flux distribution as a function of the most probable fragment ejection velocity.

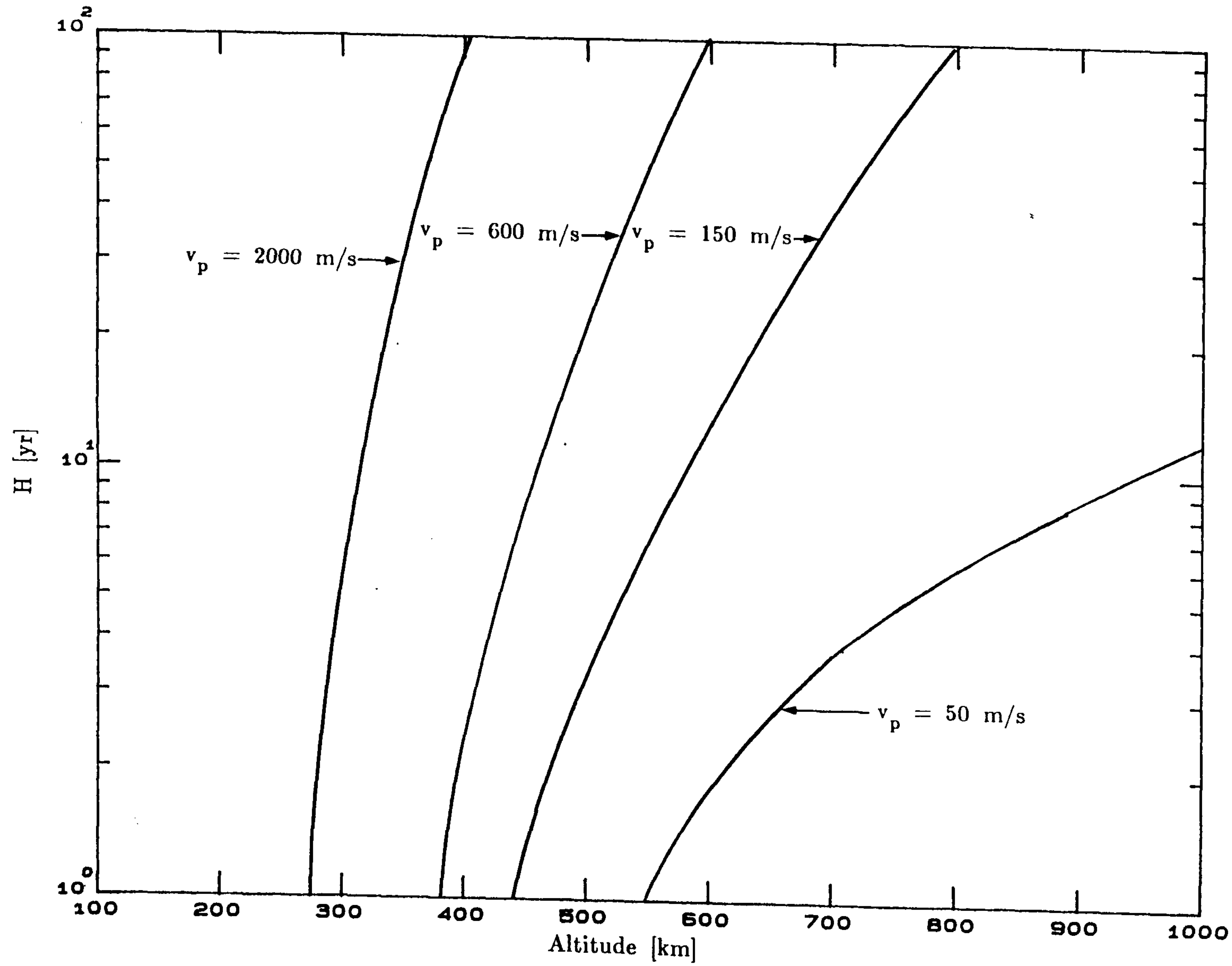


Figure 7.- Flux half-lives, defined as the time necessary for the flux to decrease to a factor of 1/e its original value. To determine the flux half-life from a breakup at an altitude other than the operational altitude, use the lower of the two altitudes.

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15. Supplementary Notes

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Abstract

The orbital debris environment model contained in this report is intended to be used by the spacecraft community for the design and operation of spacecraft in low Earth orbit. This environment, when combined with material-dependent impact tests and spacecraft failure analysis, is intended to be used to evaluate spacecraft vulnerability, reliability, and shielding requirements. The environment represents a compromise between existing data to measure the environment, modeling of these data to predict the future environment, the uncertainty in both measurements and modeling, and the need to describe the environment so that various options concerning spacecraft design and operations can be easily evaluated.

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