NASA/CR-2005-213691

Analysis of Past and Possible Threats Posed by Satellite Breakups and Potential Actions to Minimize Their Risk to Human Tended Vehicles

Leah Holmes

United Space Alliance
Houston, Texas

December 2005

The NASA Scientific and Technical Information Program Office ... in Profile

Since its founding, NASA has been dedicated to the advancement of aeronautics and space science. The NASA Scientific and Technical Information (STI) Program Office plays a key part in helping NASA maintain this important role.

The NASA STI Program Office is operated by Langley Research Center, the lead center for NASA's scientific and technical information. The NASA STI Program Office provides access to the NASA STI Database, the largest collection of aeronautical and space science STI in the world. The STI Program Office is also NASA's institutional mechanism for disseminating the results of its research and development activities. These results are published by NASA in the NASA STI Report Series, which includes the following report types:

- TECHNICAL PUBLICATION. Reports of completed research or a major significant phase of research that present the results of NASA programs and include extensive data or theoretical analysis. Includes compilations of significant scientific and technical data and information deemed to be of continuing reference value. NASA counterpart of peer-reviewed formal professional papers, but having less stringent limitations on manuscript length and extent of graphic presentations.
- TECHNICAL MEMORANDUM. Scientific and technical findings that are preliminary or of specialized interest, e.g., quick release reports, working papers, and bibliographies that contain minimal annotation. Does not contain extensive analysis.
- CONTRACTOR REPORT. Scientific and technical findings by NASA-sponsored contractors and grantees.
- CONFERENCE PUBLICATION. Collected papers from scientific and technical conferences, symposia, seminars, or other meetings sponsored or co-sponsored by NASA.
- SPECIAL PUBLICATION. Scientific, technical, or historical information from NASA programs, projects, and missions, often concerned with subjects having substantial public interest.
- TECHNICAL TRANSLATION. English-language translations of foreign scientific and technical material pertinent to NASA's mission.

Specialized services that complement the STI Program Office's diverse offerings include creating custom thesauri, building customized databases, organizing and publishing research results ... even providing videos.

For more information about the NASA STI Program Office, see the following:

- Access the NASA STI Program Home Page at **http://www.sti.nasa.gov**
- E-mail your question via the Internet: help@sti.nasa.gov
- Fax your question to the NASA STI Help Desk at (301) 621-0134
- Telephone the NASA STI Help Desk at (301) 621-0390
- Write to:

  NASA STI Help Desk
  NASA Center for AeroSpace Information
  7121 Standard Drive
  Hanover, MD 21076-1320

NASA/CR-2005-213691

Analysis of Past and Possible Threats Posed by Satellite Breakups and Potential Actions to Minimize Their Risk to Human Tended Vehicles

Leah Holmes

United Space Alliance
Houston, Texas

National Aeronautics and
Space Administration

Johnson Space Center
Houston, Texas 77058

December 2005

Available from:

NASA Center for AeroSpace Information
7121 Standard Drive
Hanover, MD 21076-1320

National Technical Information Service
5285 Port Royal Road
Springfield, VA 22161

This report is also available in electronic form at http://ston.jsc.nasa.gov/collections/TRS

Table of Contents

Abstract	1
Background	1
SBRAM	2
Historical Study	3
Trends in Pc Due to Breakup Time and Right Ascension of the Ascending Node	5
Trends in Pc Due to Inclination, Eccentricity and Argument of Perigee	8
Uncertainty of Breakup Time	10
Small Burn Maneuver Study	11
Possible Actions	13
Conclusion	16

Tables

3a	Risk of Each Type of Satellite	3
3b	Traits of Satellites	4
3c	Vehicle and Attitude of the Highest Risk for Each Satellite	4
3d	Duration of Threat	5
3e	Strength of Correlation	5
3f	Correlation between Pc and the Orbital Elements	5
4a	Correlation between the Difference in Time from the Original Breakup and Pc for Shuttle free Flight Only	6
4b	Correlation between the Absolute Value of the Difference in RAAN between the Satellite and Asset and Pc for Shuttle Free Flight Only	6
4c	Satellite Altitudes	6
4d	Correlation between the absolute value of the Difference in RAAN between the Satellite and Asset and Pc for Shuttle Free Flight Only	7
4e	The Highest Shuttle Free Flight Pc and the Absolute Value of Δ RAAN It Occurred at for Each Satellite	7

Tables, continued

5a	Correlation between the Absolute Value of the Difference of Inclinations of the Satellite and Asset and Pc for Shuttle Free Flight Only	8
5b	Correlation between the Absolute Value of the Difference of Eccentricities of the Satellite and Asset and Pc for Shuttle Free Flight Only	8
5c	Correlation between the Absolute Value of the Difference of AOPs of the Satellite and Asset and Pc for Shuttle Free Flight Only	9
5d	Satellite Attitudes	9
5e	Highest Risk for Satellite 20390 for Shuttle Free Flight Only	10
6a	Correlation Coefficients between Uncertainty of Breakup Time and Pc for Shuttle Free Flight Only	10
6b	Correlation Coefficients between Uncertainty of Breakup Orbit and Pc for Shuttle Free Flight Only	11
7a	Trends Caused by a Maneuver	12
7b	Most Significant Changes caused a Maneuver	12
7c	Correlation Coefficients between the Max Pc and the Orbit That Each Type of Burn Occurred	12
7d	Correlation Coefficients between the Pc and the Orbit That each Type of Burn Occurred	12
7e	Correlation Coefficients between the 10-Day Integrated Pc and the Orbit That Each Type of Burn Occurred	13
8a	Time of Max Pc	14
8b	Trends in the Number of Times a Satellite Reached a Pc of Its Highest Order of Magnitude within a 10-Day Period While Slipping the Launch Day by Two Days	14
8c	Trends in Growth of Cumulative Pc	15
8d	Correlation between Days after Breakup that Launch Occurs and 10-Day Cumulative Pc	15
8e	Trends in Time That 90% of the 10-Day Cumulative Pc is Reached	15

Tables, continued

8f	ORA Calculations	16
8g	Rescheduling an EVA	16

Appendices List **18**

A.1	Satellite Breakups (1998-2005)	19
A.2	Selected Satellite Breakups (1998-2005)	20
A.3	Comparison of the Asset to the Satellite	21
B.1	Satellite 20390—Russian Rocket Body	22
B.2a	Satellite 20390—Russian Rocket Body	23
B.2b	Satellite 25544—International Space Station	24
C.1	Satellite 20390—Russian Rocket Body	25
C.2	Satellite 20390—Rocket Body	26
C.3	Satellite 20390—Russian Rocket Body	27
D.1	Satellite 20390—Russian Rocket Body	28
D.2	Satellite 20390—Russian Rocket Body	29
D.3	Satellite 20390—Russian Rocket Body	30
D.4	Satellite 20390—Russian Rocket Body	31
E.1a	Satellites 20390, 26960, 26040	32
E.1b	Satellites 20390, 26960, 26040	33
E.2	Number of Times a Single Encounter Pc Reached Its Highest Order of Magnitude for a Specified Satellite	34
E.3a	Satellite 26040—Spacecraft	35
E.3b	Satellite 26040—Spacecraft	36
E.3c	Satellite 26040—Spacecraft	37
E.4a	Optimal Relative Attitude for the Shuttle	38
E.4b	Optimal Relative Attitude for the Station	39
F	Acronym and Shorthand Notation List	40

ABSTRACT

There have been 30 known satellite breakups from First Element Launch in November of 1998 to June of 2005. While breakups have been studied, the threat that they cause to human space flight is not well understood. Previous studies simply assessed the probability of collision per square meter (Pc/m^2) and never considered different particle diameters or vulnerable areas of vehicles. Also, no flight rules exist to aid the crew and flight control team (FCT) in response to a satellite breakup, causing concern for the safety of crew members and vehicles. This study fills the gaps left by previous tests and provides information that will be used to create flight rules that will increase safety during human spaceflight.

There are six parts to the completed analysis. The Historical Study calculates the threats that have been posed to the Space Shuttle, the International Space Station, the combined vehicles while docked, and extravehicular activities. It also determines the duration of the risk and how certain orbital elements are related to the probability of collision (Pc). The Trends in Pc due to Breakup Time and Right Ascension of the Ascending Node Study examines how breakup time and right ascension of the ascending node (RAAN) affect the Pc. The Trends in Pc due to Inclination, Eccentricity, and Argument of Perigee Study investigates how inclination, eccentricity, and argument of perigee (AOP) are correlated to Pc. It also determines the highest threat that the various vehicles can experience. The Uncertainty of Breakup Time Study establishes the sensitivity of the software, Satellite Breakup Risk Assessment Model (SBRAM), to the uncertainty of breakup time and the uncertainty of orbit that the breakup took place. The Space Debris Research Group (KX2) provided this software along with particle diameters that would cause catastrophic failure to specific vulnerable areas of multiple vehicles. The Small Burn Maneuver Study indicates how a maneuver can alter the Pc. Finally, the Possible Action study explores the options the FCT and the crew have to lower the Pc and increase the safety of the vehicle and crew.

Each study contributes a significant factor to the flight rules that are currently being written. The Historical Study, the Trends in Pc due to Breakup Time and Right Ascension of the Ascending Node Study, and Trends in Pc Due to Inclination, Eccentricity and Argument of Perigee Study help estimate the threat of future breakups and verify the Pc thresholds that will mandate actions from the FCT or crew. The Uncertainty of Breakup Time Study helps define requirements for the accuracy of the breakup information to validate the Pc calculations. The Small Burn Maneuver Study and Possible Actions Study determine which actions should be taken and how much each action will reduce the risk in specific occurrences to ensure safety of the crew and asset objects.

1. BACKGROUND

From First Element Launch in November 1998 to June 2005, 30 known satellite breakups occurred. Prior to this study, the catastrophic threat to the Space Station or Shuttle was unknown. Previous Flight Design and Dynamics Division (DM) studies only looked at probability of collision per square meter (Pc/m^2), and never considered different particle diameters associated with catastrophic failure or vulnerable areas of vehicles. No flight rules exist which dictate guidelines on crew or FCT response to a breakup. This causes concern for the safety of the astronauts aboard the Station and the Shuttle. However, before flight rules can be developed, it is necessary to evaluate future threats posed to NASA's human spaceflight vehicles.

One method of threat prediction can be done by determining the threat posed by historical real-world events. The Space Debris Research Group (KX2) developed a software package called Satellite Breakup Risk Assessment Model (SBRAM) "to provide mission planners and decision

makers with a tool to make real-time assessments of the risk to their satellites."[1] SBRAM uses the two-line elements (TLE's) of the asset and parent object and other pertinent inputs to model the breakup and propagate the debris cloud into the future, estimating the risk the breakup poses to the asset. The output is given in Pc/m^2.

Using the output of SBRAM combined with a particular particle diameter and vulnerable area, the probability of collision (Pc) can be calculated. Because breakups are a hybrid threat, two Pc's must be found. First, high Pc's exist at the points where the concentrated portion of the debris cloud intersects the asset object. This is similar to a conjunction with an individual object and the max Pc of a single encounter is necessary to asses the threat. Satellite breakups can also resemble background flux that results from Micrometeoroid Orbital Debris (MMOD). The asset object intersects with the debris cloud multiple times during the debris cloud's lifespan, increasing the total risk with each orbit of the asset object. The integrated, or cumulative, Pc over a specific time period shows this type of threat.

However, a few limitations to the study exist. The analysis only uses a subset of satellite breakups which affects the confidence in the results of the study. Also, SBRAM is a computing intensive application, and a typical 10-day run takes on the order of 25 minutes. The lengthy testing time limited the amount of runs that could be performed. This forced the user to implement a large step size when incrementing test variables. More powerful computers would be able to run SBRAM more quickly. Automating the tests with a scripting language would allow the tests to be run overnight. This is an area of potential future research. Both factors would increase the amount of tests run and decrease the step size in test variables, resulting in more detailed data. Furthermore, time constraints allowed for only one satellite to be tested and verified. Testing all satellites would increase the accuracy of the results.

2. SBRAM

KX2 provided the following input data for each of the 30 investigated breakups: TLE's of the asset and parent object, date and time of breakup, mass of satellite, scaling factor, and type of satellite (rocket body or spacecraft). The scaling factor is a normalized representation of the number of particles generated by a breakup, varying between 0.0 and 1.0 where the value 1.0 signifies the most particles. The solar flux was found from the National Oceanic and Atmospheric Administration (NOAA) website: http://www.sec.noaa.gov/index.html. Also, a total of 25 Monte Carlo iterations are necessary to reduce the Pc variations to a level below two decimal places of precision; and therefore, 25 was used as the input for the number of simulation runs in each test.

For the first run of the Historical Study, the simulation start time was the time of breakup, and the simulation end time was ten days after breakup. This was run with four different minimum fragment sizes associated with catastrophic failure: 0.3 cm, 0.04 cm, 0.16 cm, and 1 cm. Each vehicle and attitude has a different minimum fragment size and vulnerable area that was provided by Dr. Eric Christiansen, who is part of the Hypervelocity Impact Technology Facility (HITF). A SBRAM.log file and "RevbyRev" file are generated with each completed simulation. The log file reports the time-integrated Pc/m^2 and the standard deviation for each iteration of a Monte Carlo run, while the "RevbyRev" file reports the Pc/m^2 for each rev where a possible conjunction occurred and the time it took place after simulation start time.

With this output, calculations dependent on the minimum fragment size and vulnerable area were made to find the Pc of NASA's human spaceflight vehicles. The following are the five conditions that were studied throughout the analysis.

- Pc of Shuttle free flight = Pc/m^2 of the 0.3cm min fragment size * $260m^2$
- Pc of Station free flight = Pc/m^2 of the 0.3cm min fragment size * $125m^2$ + Pc/m^2 of the 1.0cm min fragment size * $600m^2$

[1] Matney, Mark. *An Introduction to the Satellite Breakup Risk Assessment Model (SBRAM)*. Lockheed Martin, 1998.

- Pc of docked ops -XVV attitude (Station body coordinates) = Pc/m^2 of the 0.3cm min fragment size * 300m^2
- Pc of docked ops +XVV attitude (Station body coordinates) = Pc/m^2 of the 0.3cm min fragment size * 500m^2 + Pc/m^2 of the 0.16cm min fragment size * 16m^2
- Pc of EVA = Pc/m^2 of the 0.04cm min fragment size * 3m^2

There are a few limitations to SBRAM. The software is labor intensive because it is only set up to calculate the Pc/m^2 and does not take the vulnerable area into account, forcing the user to input each particle size and area manually. As shown from the equations above, multiple fragment sizes need to be tested for each breakup, indicating that the current version of the software is not user friendly. Also, SBRAM will not complete a test when the orbits of the satellite and asset are identical. To overcome the problem, the RAAN was altered until SBRAM would function properly. Currently, the software is being modified to address these issues. It will become a real time analysis tool.

For each condition besides Station free flight, Pc's calculated were potential Pc's rather than actual. For example, the Shuttle was only on-orbit during one of the 30 breakups. For this breakup, the test was run with respect to the Station and the Shuttle. For the 29 other breakups, the Shuttle free flight Pc represents the threat that the debris cloud would have posed to the Shuttle had it been in orbit at the time of breakup. Also, it was assumed that the orbit of the Shuttle would be the same as that of the Station. While this is not exactly correct, the results will still provide useful information because most of the Shuttle's on-orbit time is spent docked to the Station.

3. Historical Study

A. Study of Past Breakups

This study quantified the threat that each of the 30 breakups posed to various vehicles and attitudes (See Appendix A.1). It was determined that the range of 10-day integrated Pc's was from 0 to 4.95×10^{-4}. The highest 10-day integrated Pc was caused by satellite 20390, a Russian rocket body, in the docked ops +XVV attitude.

Type of Satellite (no. of Pc's calculated)	% of Pc's > 1.0×10^{-5}	% of Pc's between 1.0×10^{-5} and 1.0×10^{-6}	% of Pc's < 1.0×10^{-6}
Ullage Motor (170)	2.9%	19.4%	77.7%
Rocket Body (80)	16.3%	20.0%	63.7%
Spacecraft (60)	23.3%	35.0%	41.7%

Table 3a. Risk of Each Type of Satellite

In general, Pc's of breakups of ullage motors were much smaller than those of rocket bodies or spacecraft (See Table 3a). Historically, less than three percent of the Pc's calculated for breakups of ullage motors were greater than 1.0×10^{-5}. The small risk associated with ullage motors can be explained by the fact that they have a smaller scaling factor, meaning fewer particles are produced by the breakup. Also, the mass of an ullage motor is much smaller than rocket bodies or spacecraft (See Table 3b). Another reason for the difference in risk is that ullage motors generally have an elliptical orbit which lowers the chance of collision. Therefore, ullage motor breakups were deemed a minimal threat to human spaceflight in the past, and will be considered a minimal risk for future flights.

Typical Traits of Each Satellite	Ullage Motors	Rocket Bodies	Spacecraft
Mass	55 kg	900-1760 kg	850-3000 kg
Scaling Factor	0.1	1.0	0.6-1.0

Table 3b. Traits of Satellites

% of Satellites with Highest Specified Pc for each Vehicle and Attitude	Shuttle Free Flight	Station Free Flight	Docked Ops – XVV	Docked Ops +XVV	EVA	Pc = 0 in all categories
10-Day Integrated Pc	0.0%	3.2%	0.0%	74.2%	6.5%	16.1%
10-Day Max Pc	0.0%	6.5%	0.0%	74.2%	3.2%	16.1%

Table 3c. Vehicle and Attitude of the Highest Risk for Each Satellite

A majority of the satellites had their highest 10-day integrated and 10-day max Pc while in the docked ops +XVV attitude (See Table 3c). This is because the attitude has a smaller minimum fragment size associated with catastrophic failure (0.16cm) which means that there is more debris available to damage the Shuttle tiles. The attitude also has a large vulnerable area (500 m^2) associated with minimum fragment size of 0.3cm. Clearly, the study supports the decision to discontinue flying in the ISS +XVV attitude during docked ops.

The three satellites with the highest Pc's in attitudes other than docked ops +XVV were compared. Satellite 27856 had the highest 10-day integrated Pc and 10-day max Pc for Station free flight. This can be explained by its extremely low altitude (Ha = 263 km). The smaller particles tend to decay faster than the larger ones; and therefore, the majority of the particles posing a threat to the asset within 10 days of breakup were large. Station free flight has a large area associated with a 1.0 cm fragment size, explaining the high Pc's in that category for this satellite. Satellite 19773 had the highest 10-day integrated Pc and 10-day max Pc for EVA. This can be explained by its extremely high altitude (Ha = 35716 km). The smaller particles which are only catastrophically dangerous to EVA tend to decay faster; and therefore, the majority of the particles posing a threat to the asset within 10 days of the breakup will be small and only harmful to an EVA. The last satellite (19856, Ha = 18770 km) follows this same logic for 10-day integrated Pc where its highest value is in the EVA category; however, its 10-day max Pc was highest in Station free flight. While this does not follow the trend that has been discussed, the 10-day max Pc for the EVA category is only slightly lower than the 10-day max Pc in the Station free flight category for this satellite. This signifies that the theory still holds (See Appendix A.1).

B. Duration of Threat

This study determined the length of the risk involved with the nine satellite breakups that resulted in a Pc greater than 1.0 $x10^{-5}$ (See Appendix A.2). The value 1.0 $x10^{-5}$ was chosen because it is currently the threshold for performing an avoidance maneuver for tracked objects. For these satellites, the start day was increased by increments of ten days, and each simulation lasted ten days. This continued until there was a downward trend in both 10-day max and 10-day integrated Pc.

Time of Simulation (Days after breakup)	% of Time the Risk was Greatest at the Specified Time Period	% of Pc's that were Lower than the Original 10-Day Pc for the First Time in the Specified Time Period
0-10 Days	82.2%	N/A
10-20 Days	17.8%	82.2%
20-30 Days	0.0%	16.7%
30-40 Days	0.0%	1.1%

Table 3d. Duration of Threat

After completing the test, the maximum 10-day integrated Pc was again caused by satellite 20390 (5.04×10^{-4}). All satellites had their greatest Pc within the first 20 days with a majority of them actually occurring in the first 10 days. Also, the threat decreased within twenty days of the highest risk (See Table 3d).

C. Trends in Pc

This study established correlations between Pc and four elements: inclination, right ascension of the ascending node (RAAN), eccentricity, and argument of perigee (AOP) (See Appendix A.3). The strength of the correlation is categorized according to table 3e. The study was completed by calculating the difference in each of the previously mentioned orbital elements between the asset and satellite. These values were then compared to the highest 10-day integrated Pc of that satellite.

Type of Correlation	Range of Correlation Coefficient (r)
None	$0 < \lvert r \rvert < 0.1$
Weak	$0.1 \leq \lvert r \rvert < 0.33$
Medium	$0.33 \leq \lvert r \rvert < 0.66$
Strong	$0.66 \leq \lvert r \rvert \leq 1.0$

Table 3e. Strength of Correlation (Definition)

Correlation	*Δ Inclination (Δ Inc)	*Δ RAAN	*Δ Eccentricity (ΔEcc)	*Δ AOP
Between the Pc and the variable	0.08	0.08	-0.44	-0.15

Table 3f. Correlation between Pc and the Orbital Elements
*Δ = Difference in the variable between the asset and satellite

Clearly, no correlations existed when comparing the Pc to Δ Inc or Δ RAAN; however, Pc showed signs of correlation with Δ Ecc and Δ AOP. The negative, medium correlation between Pc and Δ Ecc indicated that a satellite with a circular eccentricity (similar to that of the Station) generally had higher Pc's than one that had an elliptical eccentricity. Also, the negative, weak correlation between Δ AOP and Pc signified that when the Δ AOP was small, the Pc value was high (See Table 3e and 3f).

4. Trends in Pc due to Breakup Time and Right Ascension of the Ascending Node

Without much knowledge about the threat caused by breakups, it is difficult to determine what actions to take in response to a breakup in order to ensure the safety of the astronauts and spacecraft. Different aspects of orbits can change throughout the satellite's lifespan and alter the effect that the breakup would have on human spaceflight. It is important to understand how these variables can affect the Pc. This analysis was performed using breakup time and RAAN as the variables because it is reasonable to believe that a satellite could breakup at any time and at any RAAN. Therefore, the purpose of this portion of the study was to determine any correlations between Pc and the tested variables and establish the highest Pc of a specified orbit.

The breakup of satellite 20390 had the highest Pc in the historical test; and therefore, it was chosen to be used for this portion of the study. All initial parameters, except the variable that was being altered, were fixed at the value they were in the historical study of this satellite breakup. First, a series of tests were performed by varying the time of breakup by increments of one fourth of a revolution (0.015625 decimal days) until a full revolution was reached. In another series of runs, the RAAN was altered in increments of 90 degrees, using values between 0 and 360 degrees. For each test, four runs were made with different minimum fragment sizes: 0.3 cm, 0.04 cm, 0.16 cm, and 1 cm. Because the output was in terms of Pc/m^2, a spreadsheet was used to compute the 10-day integrated and 10-day max Pc of Shuttle free flight, Station free flight, docked ops +XVV, docked ops –XVV, and EVA (See Section 2 for calculations associated with each category). Shuttle free flight was verified using satellite 26040 and 25088. Then to create the highest Pc possible, the Station was run against a hypothetical satellite with the same orbital elements as the Station during 1999 and 2001. Again, the RAAN was altered over 360 degrees to find the maximum Pc**.**

Satellite no.	10-Day Integrated Pc	10-Day Max Pc
Satellite 20390	0.33	-0.01

Table 4a. Correlation between the Difference in time from the original breakup time (Δ Time) and Pc for Shuttle Free Flight Only

For the breakup time test, all values were within one order of magnitude; and therefore, there was no visible trend between 10-day integrated or 10-day max Pc and breakup time (See Appendix B.1). However, a weak to medium correlation existed between 10-day integrated Pc and breakup time when calculated numerically (See Tables 3e and 4a). This can be attributed to the minimal data points collected for the specific study, and further testing would be needed to establish any definitive correlations.

Satellite no.	10-Day Integrated Pc	10-Day Max Pc
Satellite 20390	0.93	0.71
Satellite 26040	0.16	0.07
Satellite 25088	-0.23	0.27

Table 4b. Correlation between the absolute value of the Difference in RAAN between the Satellite and asset (Δ RAAN) and Pc for Shuttle Free Flight Only

There was a strong, positive correlation between Pc and the absolute value of Δ RAAN for satellite 20390 (See Tables 3e and 4b). The highest max Pc was about an order of magnitude greater than the lowest value. The greatest 10-day integrated and max Pc value occurred when the RAAN of the satellite ($RAAN_{SAT}$) was 180 degrees different than the RAAN of the Station ($RAAN_{ISS}$). As $RAAN_{SAT}$ approached $RAAN_{ISS}$ from either side, there was a downward trend in Pc (See Appendix B.2a). This trend is logical because when the $RAAN_{SAT}$ is 180 degrees different than the $RAAN_{ISS}$, the satellite and Station will be traveling directly towards each other when crossing the planar intersections, increasing the chance of collision. When the $RAAN_{SAT}$ equals the $RAAN_{ISS}$, the planes are similar, and the two objects will be moving in the same direction, lowering the chance of collision. The tests were repeated with satellite 25088 and 26040. While there was no visible trend for satellite 26040 or 25088, the Pc was weakly correlated to the absolute value of Δ RAAN. When finding a strong trend in one satellite but not the others, the orbits were graphed, and they indicated a trend between the correlation coefficient and Semi Major Axis (SMA).

Satellite no.	Ha of the Satellite (km)	Hp of the Satellite (km)	Difference of Semi Major Axis (Δ SMA) Between Satellite and ISS (km)
Satellite 20390	487.3	471.9	92.6
Satellite 26040	413.4	410.6	29.8
Satellite 25088	403.3	237.0	-46.9

Table 4c. Satellite Altitudes

Satellite 20390 had a higher altitude than the Station. Satellite 26040 had a similar altitude to the Station. Satellite 25088 had a lower altitude than the Station. The correlation between difference in Semi Major Axis between the satellite and Station (Δ SMA) and the correlation coefficient of the absolute value of Δ RAAN and Shuttle free flight, 10-day integrated Pc was found to be 0.97 (See Table 4c). This correlation was positive and strong, meaning the higher the satellite's attitude is compared to the Space Station, the greater the correlation between Δ RAAN and Pc. However, the correlation was determined from only three data points. In the future, more satellites could be used to verify the validity of the correlation. The altitude of satellite 20390 was about 100 km higher than the Station, indicating that there will be a strong correlation. This is logical because the particles experienced more drag than the Space Station when the breakup occurred, lowering the orbit of the debris cloud to be closer to the orbit of the Station. This increased the Pc's sensitivity to the orbital elements of the breakup creating the strong correlation. However, the altitudes of satellites 26040 and 25088 were similar to or lower than the Station. When the satellites broke up, drag caused the debris clouds to be located below the Station. Therefore, the Pc was not as affected by the RAAN of these satellites as in the previous satellite, explaining the different strength of correlation between satellites 20390, 26040, and 25088 (See Table 4c).

Satellite no.	10-Day Integrated Pc	10-Day Max Pc
ISS (4/18/99)	-0.67	-0.60
ISS (11/21/01)	-0.62	-0.59

Table 4d. Correlation between the absolute value of the Difference in RAAN between the Satellite and Asset (Δ RAAN) and Pc for Shuttle Free Flight Only

The study where the Space Station was run against a hypothetical satellite in its exact orbit produced a medium to strong negative correlation and an apparent visual trend for both of the orbits tested (See Table 3e and 4d). The highest Pc occurred when the $RAAN_{SAT}$ was similar to the $RAAN_{ISS}$, and it was the lowest when the RAAN's were 180 degrees apart (See Appendix B.2b). The trend can be explained by the fact that when the RAAN's of the two objects are similar, the objects are traveling with each other in close proximity, maintaining a constant high Pc. When the RAAN's of the objects are opposite, they only intersect twice during an orbit, lowering the chance of collision.

Satellite no.	Absolute Value of Δ RAAN that Yielded the Highest Pc	Highest Shuttle Free Flight 10-Day Integrated Pc
20390	180	3.81×10^{-4}
26040	110	2.86×10^{-4}
25088	180	3.08×10^{-5}
ISS (4/18/99)	20	1.18×10^{-3}
ISS (11/21/01)	15	3.83×10^{-4}

Table 4e. The Highest Shuttle Free Flight Pc and the Absolute Value of Δ RAAN it Occurred at for Each Satellite

As expected, the satellites with identical orbits to the Space Station had the highest Pc's. This proved that if the breakup has a similar orbit to the asset, the risk will be high. Similarly if the breakup's orbit is very different than the asset, the risk will be much lower (See Table 4e).

For some satellites, SBRAM would not complete the tests when the $RAAN_{SAT}$ was similar to the $RAAN_{ISS}$. This mainly occurred when the Station was run against a satellite with the same orbital elements. This problem is being addressed by the KX2, and the test should be reevaluated once a solution is established.

5. Trends in Pc due to Inclination, Eccentricity, and Argument of Perigee

There is currently little information about the threat that breakups pose, and there are no flight rules to react to them. Once the highest possible Pc is established, there will be a better understanding of the risk posed by breakups, and a procedure mandating action based upon pre-determined threat levels can be developed. The purpose of this part of the study was to determine any correlations between 10-day integrated or max Pc and inclination, eccentricity, and AOP.

Because it had the highest Pc in the historical study, the breakup of satellite 20390 was chosen. All conditions were kept as they were in the original study except the variable that was being altered.

First, variations of the inclination of the satellite's orbit were tested. The values were altered in increments of 15 degrees. Also, a test was run with the inclination matching that of the Space Station (51.6 degrees). Four other satellites were tested in the same manner for inclination.

Then, the eccentricity was adjusted. The values used were 0.1, 0.01, the original eccentricity, and the eccentricity of the Station (0.0005137), and 0.0.

Finally, the Argument of Perigee (AOP) was changed in increments of 60 degrees, only including the values that fell between 0 and 360 degrees. Then, the same test was run with the AOP equaling that of the Station (14.7018 degrees). The results were verified with multiple satellites.

Each test was completed for 4 minimum fragment sizes: 0.3 cm, 0.04 cm, 0.16 cm, and 1.0 cm. The output was given in terms of Pc/m^2; and therefore, a spreadsheet was used to calculate the Pc of Shuttle free flight, Station free flight, docked ops +XVV, docked ops –XVV, and EVA (See Section 2 for specific equations).

Satellite no.	10-Day Integrated Pc	10-Day Max Pc
Satellite 20390	-0.51	-0.58
Satellite 26040	0.07	0.14
Satellite 26960	0.45	-0.13
Satellite 25088	0.09	0.14
Satellite 25942	0.25	-0.17

Table 5a. Correlation between the absolute value of the Difference of Inclinations of the Satellite and Asset (Δ Inc) *and Pc for Shuttle Free Flight Only*

The correlation coefficients between the absolute value of Δ Inc and Pc ranged from -0.58 to 0.45. Each satellite showed a different sign and strength (See Tables 3e and 5a). From this information, it can be determined that any trend is not correlated with inclination, but with a geometry specific to each satellite with respect to the Station. For example, satellite 20390 showed a visual trend. The Pc increased as the inclination of the satellite increased until the value of 68 degrees when the Pc dropped off by an order of magnitude from the highest value; however, these results could not be duplicated with another satellite (See Appendix C.1).

Satellite no.	10-Day Integrated Pc	10-Day Max Pc
Satellite 20390	-0.96	-0.86

Table 5b. Correlation between the absolute value of the Difference of Eccentricities of the Satellite and Asset (Δ Ecc) *and Pc for Shuttle Free Flight Only*

There was a strong negative correlation between the absolute value of Δ Ecc and Pc (See Tables 3e and 5b). This meant that as the absolute value of Δ Ecc increased, the Pc decreased. The visual trend showed that the Pc was relatively level at values around the eccentricity of the Station, and then at an eccentricity of 0.1, there was a drop-off of an order of magnitude (See Appendix C.2). One explanation of the drop-off is the large eccentricity. In this case, changing the orbit to become elliptical caused perigee to be less than 80 km and the debris burned up on reentry, lowering the Pc. If the perigee is large enough so that the debris does not reenter, the orbit becomes much bigger than that of the Space Station. This lengthens the time it takes the debris cloud to complete one orbit, and it may not be at the planar intersections at the same time as the Station due to different velocities. The debris cloud is simply at Station altitudes for a shorter amount of time, lessening the chance of collision. Furthermore, there is a larger area for the debris to spread over, causing the debris to fade into background flux more quickly.

Satellite no.	10-Day Integrated Pc	10-Day Max Pc
Satellite 20390	-0.87	-0.61
Satellite 26040	0.68	0.49
Satellite 26960	0.48	0.29
Satellite 25088	0.13	0.09
Satellite 27053	0.45	0.41

Table 5c. Correlation between the absolute value of the Difference of AOP's of the Satellite and Asset (Δ AOP) *and Pc for Shuttle Free Flight Only*

There were no common visual trends between Pc and the absolute value of Δ AOP for multiple satellites. The correlation coefficients ranged from -0.87 to 0.68 (See Table 5c). This showed that any Pc trend seen in individual satellites was not correlated with the absolute value of Δ AOP, but the particular geometry of the satellite. For example in satellite 20390's case, the Pc was highest when the AOP of the satellite (AOP_{SAT}) equaled that of the Space Station. The Pc decreased as the AOP_{SAT} changed 90 degrees, and then, increased to the point when AOP_{SAT} was 180 degrees different from the Station (See Appendix C.3). This satellite had an altitude that was higher than the Station so that when it broke up and the particles experienced more drag than the Station, the debris cloud decayed to the altitude of the Station (See Table 5d). It is predicted that there is an undetermined range of Δ SMA's that cause this behavior. In these cases, the orbits of the two objects are similar and Pc is highest when the lines of apsides of the two orbits are aligned, as in satellite 20390. In other cases when the Δ SMA does not decay into the specific range, the debris cloud is not near the Station. When the AOP's of the satellite and Station are equal, the lines of apsides are aligned, but the different orbit sizes may not cause any points of intersection, lowering the Pc. As the AOP_{SAT} varies, there is more chance of intersection and the Pc is higher.

Satellite no.	Ha of the Satellite(km)	Hp of the Satellite (km)	Difference of Semi Major Axis (Δ SMA) Between Satellite and Station (km)
Satellite 20390	487.3	471.9	92.6
Satellite 26040	413.4	410.6	29.8
Satellite 26960	680.6	545.6	226.0
Satellite 25088	403.3	237.0	-46.9
Satellite 25942	743.5	731.2	375.6
Satellite 27053	391.6	229.3	-55.0

Table 5d. Satellite Attitudes

Changed Orbital Elements	10-Day Integrated Pc	10-Day Max Pc
No Change	2.36×10^{-4}	5.32×10^{-6}
Inclination (Δ Inc = 11.9°)	3.09×10^{-4}	7.10×10^{-5}
Eccentricity (Δ Ecc = 0.0095)	2.46×10^{-4}	6.95×10^{-6}
AOP (Δ AOP = -20°)	4.15×10^{-4}	1.05×10^{-4}

Table 5e. Highest Risk for Satellite 20390 for Shuttle Free Flight Only

The highest 10-day integrated Pc's remained within one order of magnitude of the original value; however, the highest max Pc's increased greatly as the orbital elements were altered. Neither of the Pc's exceeded the order of magnitude of 1.0×10^{-4} (See Table 5e).

6. Uncertainty of Breakup Time

Historical data shows that the time of the breakup is not always known. In some cases, a particular orbit can be specified while in other cases, the exact orbit cannot be determined. Based on past data from US Strategic Command (USSTRATCOM), a majority of breakup times are either known within 10 minutes, or known to be during one of several orbits. Also, it was unknown how the software (Satellite Breakup Risk Assessment Model) would react to uncertainty of breakup time. Once this information was found, breakup uncertainty requirements could be established for USSTRATCOM. The purpose of this study was to determine the sensitivity of the Pc to change in breakup uncertainty time.

Satellites 20390 and 26040 were used for the study. Two variables were altered throughout the study: uncertainty of breakup time and the orbit on which the breakup occurred. To simulate different possible orbits of breakup, orbits were subtracted from the breakup time in one orbit increments, starting with the original breakup time (t_B) and continuing through 6 orbits before breakup. Each breakup time was tested for different uncertainty times also. For the t_B through 2 orbits before t_B, the values used were 0, 2, 4, 5, 6, 8, and 10 minutes. For the other breakup times, only 0 minutes, 5 minutes, and 10 minutes were used as uncertainty times. The 10-day integrated Pc and 10-day max Pc were calculated for each test for Shuttle free flight (See section 2 for equation). As the orbit and uncertainty time was changed, both the 10-day integrated Pc and the Pc at the time of original highest risk, $Pc(t_o)$, were recorded and graphed to determine a requirement for uncertainty of data received. Finally, the highest individual encounter Pc and the time that it occurred was recorded along with the corresponding information from the other uncertainty times.

Correlation Coefficient	Original Breakup Time (t_B)		Minus 1 Orbit		Minus 2 Orbits	
Satellite no.	10-Day Integrated Pc	10-Day Max Pc	10-Day Integrated Pc	10-Day Max Pc	10-Day Integrated Pc	10-Day Max Pc
Satellite 20390	-0.10	-0.14	0.69	-0.89	0.78	-0.59
Satellite 26040	-0.84	-0.38	-0.64	-0.71	-0.59	0.10

Table 6a. Correlation Coefficients between Uncertainty of Breakup Time and Pc for Shuttle Free Flight only

Table 6a represents the correlation between the uncertainty of breakup time and Pc for the t_B, one orbit before t_B, and two orbits before t_B. There was no visual trend between Pc and uncertainty of breakup time (See Appendix D.1). All values were within an order of magnitude of one another. The numerical correlations ranged from -0.89 through 0.78 for the two satellites (See Table 6a). It can be determined that any correlations were not due to changing the uncertainty of breakup time, but each correlation was a result of a specific geometry of the satellite's orbit.

Correlation Coefficient	10 Min Uncertainty		5 Min Uncertainty		0 Min Uncertainty	
Satellite no.	10-Day Integrated Pc	10-Day Max Pc	10-Day Integrated Pc	10-Day Max Pc	10-Day Integrated Pc	10-Day Max Pc
Satellite 20390	-0.68	0.00	-0.34	-0.13	-0.76	0.02
Satellite 26040	-0.44	-0.25	0.22	-0.41	0.86	0.53

Table 6b. Correlation Coefficients between Uncertainty of Breakup Orbit and Pc for Shuttle Free Flight only

Table 6b shows the correlation between uncertainty of breakup orbit and Pc for the three different uncertainty times. There was no visual trend between Pc and the orbit on which the breakup occurred (See Appendix D.2). The correlation coefficients for this ranged from -0.76 to 0.86. This signified that any correlation found was due to an orbit's specific geometry and did not depend on changing the uncertainty of orbit of the breakup (See Table 6b).

While tracking the 10-day integrated Pc over different breakup orbits, all Pc's were within +14% or -32% of the average 10-day integrated Pc (See Appendix D.3). Clearly, the uncertainty of orbit did not affect the 10-day integrated Pc, proving that it depends on the geometry of the object's orbit and not the uncertainty time. Each orbit tested in the study showed that the data would be valid. Further research would be necessary to determine if the Pc would be applicable if the uncertainty of breakup orbit was greater than 6 orbits. When following the $Pc(t_o)$, the values ranged from +221% to -77% of the average value (See Appendix D.4). Because the $Pc(t_o)$ fluctuated as the orbit of breakup changed, it was determined that an individual encounter Pc depends on the timing of the orbits. This information must be known within one orbit to ensure its accuracy.

7. Small Burn Maneuver Study

Before flight rules can be written, the way that maneuvers can affect the Pc needs to be understood. A 3 ft/s burn would be optimal because the Orbital Maneuvering System (OMS) engines are unnecessary for that size burn for the Shuttle. Moreover, it is approximately the typical burn that the Station makes during a debris avoidance maneuver (1 m/s). Also, performing the burn in full orbit increments starting at one half of an orbit before the Pc at time of original highest risk, $Pc(t_o)$, will cause the vehicle to miss the breakup both in the radial and down track directions. The purpose of this study was to lower the Pc by maneuvering to avoid an individual encounter with the breakup cloud that had a high Pc.

Again, satellite 20390 was chosen because it had the highest Pc in the Historical Study. The program VEC2TLE was used to input a posigrade 3 ft/s impulsive burn into the TLE of the Station at 0.5 orbits, 1.5 orbits, 2.5 orbits, 3.5 orbits, 4.5 orbits, and 5.5 orbits before the time of highest risk. This was repeated for a retrograde 3 ft/s burn in each case. These new TLE's were used in SBRAM with the simulation start time equaling the time of the burn and the simulation end time equaling 10 days after breakup. The Pc's calculated were the max Pc after the burn took place and the Pc at the time of original highest risk, $Pc(t_o)$, (See Section 2 for the Pc equation). These Pc values were compared to the original 10-day max Pc. The 10-day integrated Pc was also calculated by combining the Pc's that took place before the burn from the original test with the Pc's after the burn from the new data. To verify the data, satellites 26960 and 26040 were also used.

Of all Satellites, % of Burns that Caused the Specified Trend	Max Pc	Pc at Time of Original Highest Risk
Increase by at least 1 OM*	0.0%	0.0%
Increase	30.6%	5.6%
Decrease	69.4%	63.9%
Decrease by at least 1 OM*	0.0%	30.6%

Table 7a. Trends Caused by a Maneuver (Satellites 20390, 26040, and 26960 included)
*OM - Order of Magnitude

The Most that a Pc Changed Due to a Burn	Max Pc	Pc at Time of Original Highest Risk
Most Increase	0.14 OM*	0.02 OM*
Most Decrease	0.20 OM*	7.81 OM*

Table 7b. Most Significant Changes Caused by a Maneuver (Satellites 20390, 26040, and 26960 included)
*OM - Order of Magnitude

A majority of the burns caused a drop in both max Pc and $Pc(t_o)$. Also, none of the burns caused an increase of an order of magnitude for either Pc. While the max Pc only varied slightly and remained within an order of magnitude of its original value, the $Pc(t_o)$ dropped significantly due to over 30% of the burns tested. This showed that the overall max Pc was not affected by a maneuver, but an individual encounter that posed an extreme risk was avoided. In all cases, a burn that was beneficial to the immediate safety of the vehicle was found (See Tables 7a and 7b).

Of all the burns for the three satellites, over one-third caused the time of the new max Pc to occur before the original 10-day max Pc; however, for each satellite a case was found that this did not occur.

Satellite no.	Posigrade Burn	Retrograde Burn
20390	0.09	0.29
26040	-0.73	0.66
26960	0.34	-0.13

Table 7c. Correlation Coefficients between the Max Pc and the Orbit that each type of Burn Occurred

The correlation coefficients for max Pc and orbit of burn were random. Each satellite and type of burn has a different strength and sign of correlation between max Pc and the orbit that the burn occurred (See Tables 3e and 7c). This supported the previous conclusion that max Pc was not affected by a maneuver.

Satellite no.	Posigrade Burn	Retrograde Burn
20390	-0.22	-0.77
26040	-0.68	0.21
26960	-0.86	-0.89

Table 7d. Correlation Coefficients between the $Pc(t_o)$ and the Orbit that Each Type of Burn Occurred

A majority of the correlations between $Pc(t_o)$ and the orbit that the burn occurred were negative and strong (See Tables 3e and 7d). This signified that the earlier a burn is applied, the lower the $Pc(t_o)$ will be. This statement is logical because an early burn allows the vehicle to travel at a different speed and altitude for a longer period of time, causing the two objects to be a greater distance apart at the time of interest.

Satellite no.	Posigrade Burn	Retrograde Burn
20390	-0.10	-0.92
26040	-0.88	-0.90
26960	0.76	-0.89

Table 7e. Correlation Coefficients between the 10-Day Integrated Pc and the Orbit that Each Type of Burn Occurred

There was a strong negative correlation between the time a retrograde burn was executed and 10-day integrated Pc. As a retrograde burn was performed earlier, the Pc decreased. The correlation was random for posigrade burns showing no relationship (Table 3e and 7e). However, the 10-day integrated Pc was not altered by an order of magnitude by a small posigrade or retrograde maneuver; and therefore, a small maneuver would not increase the safety of the vehicle or crew in terms of cumulative Pc.

8. Possible Actions

There are some actions that the crew and flight control team could take to help reduce the probability of collision (Pc). For Shuttle, one would be to delay the launch. This could circumvent a specific max Pc or remove occurrences of Pc's that are of the same order of magnitude as the max Pc. If launch has already occurred, a Minimum Duration Flight (MDF) could be an option, and the crew would only complete the essential actions planned for the flight. This would cause the integrated Pc to decrease, and it may avoid a max Pc. This can be tested in SBRAM by altering the simulation start and stop times to manipulate the length of the mission. Another possibility that is available to both the Shuttle and Station is to maneuver into the attitude which points the least sensitive part of the vehicle in the relative velocity vector with respect to the debris cloud at the time of closest approach to reduce the Pc of an individual occurrence or a group of high risk encounters. The attitude for Shuttle aligns the -X body axis with the relative velocity vector between the Shuttle and the debris cloud (-XRV), and for Station, it is aligning the +X body axis with the relative velocity vector between the Station and the debris cloud (+XRV). This attitude is also called the Optimal Relative Attitude (ORA). An action the crew can take to minimize risk to an EVA would be to reschedule the activity to a lower risk time period. The purpose of this portion of the study was to reduce Pc through various actions.

A. Duration of Mission

First, the duration of a mission was tested to determine how it would affect the Pc. For satellites 20390, 26960, and 26040, the simulation length was changed in increments of 2 days starting with 2 days and ending with 14 days. This was completed for Shuttle Free Flight only to simplify analysis, but the behavior should be the same for all scenarios.

Time Period in Days	% of Satellites that Reached the Max Pc in Each Time Period	% of Satellites that Reached the OM* of Max Pc for the 1st time in Each Time Period
All Pc's = 0	25.0%	25.0%
First 2 Days	10.7%	28.6%
2-4 Days	14.3%	17.9%
4-6 Days	7.1%	3.6%
6-8 Days	25.0%	10.7%
8-10 Days	21.4%	17.7%

Table 8a. Time of Max Pc
*OM = Order of Magnitude

As expected, the integrated Pc increased as each two day increment was added to the simulation. As the exposure increases, the integrated Pc levels off (See Appendix E.1a). The max Pc remained within one order of magnitude of the original value for all mission lengths (See Appendix E.1b). As seen in Table 8a, a majority of the satellites experience Pc's of the highest order of magnitude for the first time in the first four days, explaining the minimal changes in max Pc as length of mission is altered. From this information, it can be determined that a MDF will only reduce the integrated risk of a breakup and could decrease that threat below the specified threshold. However, it will have a minimal effect on the max Pc. Also, increasing the length of a mission by two days will not increase the integrated or max threat by an order of magnitude.

B. Delaying the Launch Day

Another test was performed to study the possibility of delaying the launch date. The time that the max Pc occurred was recorded for each satellite that broke up since First Element Launch in November 1998. Then, the first time and the number of times the Pc of each satellite reached its highest order of magnitude were also recorded for various 10-day time periods. Additionally, the 10-day cumulative Pc was calculated and graphed for multiple satellites, increasing the start day by one half day to represent slipping the launch day. The growth of cumulative Pc was also graphed. Finally, the time that the cumulative Pc reached 80% and 90% of its 10-day value were found.

Specified Trends of Times the Pc Reached its Highest Order of Magnitude as the Launch Day Increases by 2 Days	% of Satellites with the Specified Trend
Downward Trend	62.5%
Rise Then Drop-off	25.0%
Upward Trend	12.5%

Table 8b. Trends in the Number of Times a Satellite Reached a Pc of its Highest Order of Magnitude within a 10-Day Period While Slipping the Launch Day by 2 Days (See Appendix E.2)

All but one of the satellites experienced an eventual downward trend in number of times a Pc of highest order of magnitude was reached in a 10-day time period as the launch day increased (See Appendix E.2 and Table 8b). This means that in almost all historical cases, slipping the launch day after a breakup occurs just prior to the original launch day will reduce the threat due to that breakup.

Trends of the 10-Day Cumulative Pc as Launch Day Slipped by 2 Days	% of Satellites that Followed the Specified Trend (Launch Day ranged from 0-14 Days After Breakup)
Decreased by an OM*	25.0%
Decreased more than 1/2 and OM* but less than an OM*	25.0%
Decreased by less than 1/2 an OM*	37.5%
Increased by less than 1/2 an OM*	12.5%
Increased by more than 1/2 an OM*	0.0%

Table 8c. Trends in Growth of Cumulative Pc (See Appendix E.3a)
*OM = Order of Magnitude

Half of the satellites had 10-day cumulative Pc's that decreased by at least half an order of magnitude as the launch day slipped for two weeks. The Pc increased in only one case (See Appendix E.3a and Table 8c).

Satellite no.	Correlation Coefficient, r
Satellite 20390	0.02
Satellite 26040	-0.93
Satellite 26960	-0.95
Satellite 25088	-0.94
Satellite 27053	-0.96
Satellite 25942	-0.86
Satellite 21798	-0.66
Satellite 27475	-0.97

Table 8d. Correlation between Days after Breakup that Launch Occurs and 10-Day Cumulative Pc (See Appendix E.3a)

All but one satellite experiences a strong negative correlation between day after breakup that the launch occurs and the 10-day cumulative Pc. This means that as the Launch day moves away from the day of breakup, the Pc drops (See Appendix E.3a and Tables 8d).

Trends of What Day 90% of the 10-Day Cumulative Pc is Reached as the Launch Day Increases	% of the Satellites that Follow the Specified Trend
Increase then Decrease	37.5%
Decrease then Increase	25.0%
No Correlation -- Inconsistent	12.5%
No Correlation -- All Values Within One Day	25.0%

Table 8e. Trends in Time that 90% of the 10-Day Cumulative Pc is Reached (See Appendix E.3c)

There was no trend in growth of the cumulative Pc (See Appendix E.3b). Also, there was also no definitive trend in the day that 90% of the 10-day cumulative Pc was reached, thus slipping the launch had no impact on the rate of cumulative Pc growth (See Appendix E.3c and Table 8e).

C. Optimal Relative Attitude (ORA)

The ability of lowering the Pc by maneuvering into the attitude that minimizes the vulnerable area or ORA was tested. Using the satellites 20390, 26960, and 26040, the Shuttle free flight Pc and the Shuttle's ORA Pc were calculated and compared (See Table 8f for equations). This occurs when Shuttle is in the -XRV attitude. A similar test was completed for Station free flight and the ORA for the Station, which occurs in +XRV attitude as discussed earlier.

Vehicle	Attitude	Equation
Shuttle	Free Flight	Pc/m^2 of the 0.3cm min fragment size * 260m^2
Shuttle	-XRV	Pc/m^2 of the 0.3cm min fragment size * 38.5m^2
Station	Free Flight	Pc/m^2 of the 0.3cm min fragment size * 125m^2 + Pc/m^2 of the 1 cm min fragment size * 600m^2
Station	+XRV	Pc/m^2 of the 0.3cm min fragment size * 15m^2 + Pc/m^2 of the 1 cm min fragment size * 184m^2

Table 8f. ORA Calculations

To find the percent that the Pc drops when comparing free flight to ORA, the following equation was used: $percentage = 100 - \left(\frac{Pc(ORA)}{Pc(\,freeflight)} \right) * 100$. When maneuvering from Shuttle free flight to -XRV attitude, there is an 85% decrease for any given Pc due to an individual encounter with the cloud (See Appendix E.4a). When Station is changed from Station Free Flight to +XRV attitude, there is an 80% decrease for any given Pc due to an individual encounter with the cloud (See Appendix E.4b). Maneuvering into ORA would increase the safety of the Shuttle and the Station.

D. Rescheduling an Extravehicular Activity

Finally, every possible six hour cumulative Pc associated with EVAs was calculated during the first 10 days after breakup for satellites 20390, 26040, and 26960. This represents the possible Pc's an EVA could encounter. The minimum and maximum cumulative Pc's were then compared.

Satellite no.	Max 6 hr Cumulative Pc	Min 6 hr Cumulative Pc	Difference in Order of Magnitude
Satellite 20390	1.13×10^{-6}	5.44×10^{-8}	1.3
Satellite 26960	5.98×10^{-7}	1.92×10^{-11}	4.5
Satellite 26040	7.45×10^{-7}	3.38×10^{-8}	1.3

Table 8g. Rescheduling an EVA

Rescheduling an EVA could possibly decrease the 6 hr cumulative risk by 4.5 orders of magnitude. During this study, the Pc dropped by at least 1.3 orders of magnitude. In all cases, it was beneficial to reschedule an EVA with high original risk following a breakup (See Table 8g).

9. Conclusion

With the information gathered in this study, flight rules can now be written to assist the FCT and crew when responding to a breakup. The Historical Study, Highest Pc of a Specified Orbit Study, and Highest Pc Possible Study establish the potential risk of future breakups and indicate what risk is involved with each type of orbit. This helps predict the effects of future breakups and allows Pc thresholds to be established. The Uncertainty of Breakup Time Study determines the requirements that validate Pc data, allowing the information to be used when making an assessment of the situation. Finally, the Small Burn Maneuver Study and Possible Action Study dictate the FCT's and crew's response to future breakups.

This research also gives opportunity to further study and new projects. Once KX2 addresses the limitations of SBRAM, many more tests can be run. For the Highest Pc of a Specified Orbit Study, the case where the Station was run against a hypothetical satellite with the same orbital elements can be completed because the software will allow the RAAN's of the two objects to be equal. Also, a study of a past Soyuz breaking up immediately before or after docking could be run. Station maneuvers could also be tested in the situation. Another way to complete this analysis would be to automate the testing. This would allow a variable to be tested with a much smaller

step size, giving more detailed data about the variable and its connection to the Pc. A possibility for a new project would be to complete a satellite breakup database that would be added to and monitored after each breakup. This database would allow for constant analysis of the most recent data so that the thresholds and the flight rules can be kept current and useful.

Appendix

A. Results for the Historical Study
    1. Satellite Breakups
    2. Selected Satellites Breakups
    3. Comparison of Asset to the Satellite

B. Results for the Trends in Pc due to Breakup Time and Right Ascension of the Ascending Node Study
    1. Variation of Breakup Time Graph (Satellite 20390 – Integrated Pc)
    2. Variation of RAAN Graphs
        a. Satellite 20390 (Integrated Pc)
        b. ISS (4/18/99 – Integrated Pc)

C. Results for the Trends in Pc due to Inclination, Eccentricity, and Argument of Perigee Study
    1. Variation of Inclination (Satellite 20390 – Max Pc)
    2. Variation of Eccentricity (Satellite 20390 – Integrated Pc)
    3. Variation of AOP (Satellite 20390 – Max Pc)

D. Results for Uncertainty of Breakup Time Study
    1. Variation of Uncertainty of Breakup Time (Satellite 20390)
    2. Variation of Breakup Orbit (Satellite 20390)
    3. Ten-Day Integrated Pc for Different Breakup Orbits (Satellite 20390)
    4. Pc at Time of Original Highest Risk for Different Breakup Orbits (Satellite 20390)

E. Results for Possible Actions Study
    1. Variation in Length of Simulation
        a. Integrated Pc
        b. Max Pc
    2. Number of Times the Pc Reached its Highest Order of magnitude for a Specified Satellite
    3. Variation of Launch Day (Satellite 26040)
        a. Ten-Day Cumulative Pc
        b. Growth of Cumulative Pc
        c. Day when 90% of the 10-Day Cumulative Pc is Reached
    4. Optimal Relative Attitude
        a. Shuttle
        b. Station

F. Acronym List

Satellite Breakups (1998-2005)

A.1

			Shuttle Free Flight		Station Free Flight		Docked Ops -XVV Russian Segments Fwd		Docked Ops +XVV US Segments Fwd		EVA	
Sat No.	Date	Type	10-day Integrated Pc	10-day Max Pc	10-day Integrated Pc	10-day Max Pc	10-day Integrated Pc	10-day Max Pc	10-day Integrated Pc	10-day Max Pc	10-day Integrated Pc	10-day Max Pc
20390	4/18/1999	Rocket Body	2.36E-04	5.32E-06	1.94E-04	4.45E-06	2.72E-04	6.13E-06	4.95E-04	1.11E-05	1.60E-05	3.17E-07
26040	11/21/2001	Russian Spacecraft	1.30E-04	3.46E-06	1.24E-04	3.47E-06	1.50E-04	3.99E-06	2.72E-04	7.26E-06	7.82E-06	2.06E-07
26960	12/19/2001	Rocket Body	7.36E-05	3.03E-06	6.21E-05	2.30E-06	8.50E-05	3.49E-06	1.54E-04	6.30E-06	5.84E-06	2.52E-07
25088	11/22/1999	Russian Spacecraft	3.08E-05	1.90E-06	3.29E-05	1.88E-06	3.55E-05	2.20E-06	6.39E-05	3.97E-06	2.08E-06	1.07E-07
27474	6/1/2005	Ullage Motor	2.49E-05	2.54E-06	2.28E-05	2.20E-06	2.88E-05	2.93E-06	5.39E-05	5.37E-06	2.02E-06	1.41E-07
27053	2/28/2004	Russian Spacecraft	1.29E-05	2.16E-06	1.32E-05	1.95E-06	1.49E-05	2.50E-06	2.66E-05	4.42E-06	2.89E-07	4.06E-08
25942	3/11/2000	Rocket Body	1.07E-05	8.79E-07	8.70E-06	1.17E-06	1.24E-05	1.01E-06	2.31E-05	2.02E-06	5.85E-07	9.54E-08
21798	4/15/2004	US Spacecraft	9.27E-06	1.34E-06	9.69E-06	1.61E-06	1.07E-05	1.55E-06	1.90E-05	2.71E-06	7.81E-07	6.00E-08
27475	10/29/2004	Ullage Motor	6.62E-06	7.90E-07	5.26E-06	8.73E-07	7.64E-06	9.11E-07	1.36E-05	1.60E-06	2.57E-07	3.03E-08
22250	*7/14/2001	Ullage Motor	3.56E-06	1.29E-06	3.87E-06	1.01E-06	4.10E-06	1.48E-06	7.75E-06	2.76E-06	4.54E-07	1.05E-07
22250	*7/14/2001	Ullage Motor****	3.08E-06	5.86E-07	3.41E-06	7.64E-07	3.55E-06	6.77E-07	6.56E-06	1.26E-06	2.24E-07	4.27E-08
23631	**11/21/2000	Ullage Motor	3.14E-06	8.98E-07	2.64E-06	9.06E-07	3.63E-06	1.04E-06	6.43E-06	1.83E-06	1.80E-07	3.96E-08
19537	**3/9/1999	Ullage Motor	2.82E-06	7.38E-07	1.98E-06	6.08E-07	3.25E-06	8.52E-07	5.59E-06	1.46E-06	9.55E-08	7.01E-08
21734	10/9/1999	Rocket Body	1.46E-06	3.38E-07	1.05E-06	3.21E-07	1.69E-06	3.90E-07	3.09E-06	8.17E-07	6.26E-08	2.72E-08
27856	12/9/2003	Russian Spacecraft	3.68E-07	1.32E-07	1.07E-06	4.91E-07	4.24E-07	1.52E-07	8.11E-07	3.18E-07	2.25E-08	1.47E-08
20631	**3/28/1999	Ullage Motor	2.58E-07	1.05E-07	1.63E-07	8.18E-08	2.98E-07	1.21E-07	5.19E-07	2.15E-07	8.87E-10	7.56E-10
22032	***2/1/2002	Rocket Body	2.59E-07	2.53E-07	1.25E-07	1.21E-07	2.99E-07	2.92E-07	4.99E-07	4.86E-07	2.61E-10	1.49E-10
23887	**12/13/1999	Ullage Motor	1.38E-07	8.37E-08	8.18E-08	5.57E-08	1.59E-07	9.65E-08	2.69E-07	1.64E-07	6.35E-10	6.27E-10
22925	**9/6/2000	Ullage Motor	9.10E-08	7.06E-08	1.99E-07	1.51E-07	1.05E-07	8.14E-08	2.45E-07	1.93E-07	9.97E-09	3.96E-09
19856	11/13/2003	Ullage Motor	2.26E-08	1.09E-08	1.02E-07	8.97E-08	2.61E-08	1.26E-08	9.94E-08	4.51E-08	1.31E-07	6.78E-08
21766	12/24/2001	Rocket Body	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	6.70E-08	6.70E-08	0.00E+00	0.00E+00
19773	**1/1/2001	Rocket Body	7.49E-09	5.50E-09	3.60E-09	2.64E-09	8.64E-09	6.35E-09	1.71E-08	1.33E-08	2.06E-08	2.06E-08
21226	6/16/2001	Ullage Motor	2.77E-12	2.77E-12	1.39E-12	1.39E-12	3.20E-12	3.20E-12	9.72E-09	9.72E-09	1.53E-13	1.53E-13
19122	7/9/2002	Rocket Body	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	5.29E-09	5.29E-09	0.00E+00	0.00E+00
22274	4/22/2005	Ullage Motor	1.29E-09	1.29E-09	6.20E-10	6.19E-10	1.49E-09	1.49E-09	2.48E-09	2.48E-09	4.32E-14	4.32E-14
19535	8/4/2003	Ullage Motor	7.36E-10	7.36E-10	3.54E-10	3.54E-10	8.49E-10	8.49E-10	1.42E-09	1.42E-09	0.00E+00	0.00E+00
15714	11/29/1998	Ullage Motor	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
18375	4/23/2003	Ullage Motor	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
20583	2/24/2005	Spacecraft	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
21012	2/21/2003	Ullage Motor	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
22066	7/10/2004	Ullage Motor	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00

NOTES

1) All simulations were generated with the ISS as the asset (unless otherwise specified)
2) All simulations start at the breakup time and end 10 days later
3) 10-day Integrated Pc = sum of all individual conjunctions
4) 10-day Max Pc = Maximum individual conjunction Pc experienced in the entire simulation
5) All Pc's are averaged over all of the simulations
6) Breakup Uncertainty Time = 10 minutes
7) Number of simulations for each scenario = 25
8) Number of days for each simulation = 10
9) Pc's over 10^{-5} are highlighted in yellow
10) Pc's over 10^{-4} are highlighted in red

NOTES CONT.

*The time was unknown but before 1738 UTC, 1738 UTC was used
**Time unknown, 0000 UTC was used
***Day and time unknown, Feb 1st and 0000 UTC was used
****This is the one breakup (22250) that the Shuttle was in orbit during breakup time. This simulation was run twice. The first (unmarked) with the ISS as the asset object and the 2nd (marked) with the Shuttle as the asset

Selected Satellite Breakups (1998-2005)

A.2

				Shuttle Free Flight		Station Free Flight		Docked Ops -XVV Russian Segments Fwd		Docked Ops +XVV US Segments Fwd		EVA	
Run No.	Sat No.	Date	Type	10-day Integrated Pc	10-day Max Pc	10-day Integrated Pc	10-day Max Pc	10-day Integrated Pc	10-day Max Pc	10-day Integrated Pc	10-day Max Pc	10-day Integrated Pc	10-day Max Pc
1	27474	6/1/2005	Rocket Body	2.49E-05	2.54E-06	2.28E-05	2.20E-06	2.88E-05	2.93E-06	5.39E-05	5.37E-06	2.02E-06	1.41E-07
2	27474	6/1/2005	Rocket Body	2.00E-05	1.28E-06	1.66E-05	1.64E-06	2.31E-05	1.47E-06	4.29E-05	2.89E-06	1.45E-06	1.79E-07
1	27475	10/29/2004	Rocket Body	6.63E-06	7.90E-07	5.27E-06	8.73E-07	7.65E-06	9.12E-07	1.36E-05	1.61E-06	2.57E-07	3.03E-08
2	27475	10/29/2004	Rocket Body	1.41E-06	5.69E-07	2.15E-06	5.39E-07	1.62E-06	6.56E-07	2.85E-06	1.12E-06	9.29E-08	5.04E-08
1	21798	4/15/2004	US Spacecraft	9.27E-06	1.34E-06	9.69E-06	1.61E-06	1.07E-05	1.55E-06	1.90E-05	2.71E-06	7.81E-07	6.00E-08
2	21798	4/15/2004	US Spacecraft	1.27E-05	1.78E-06	7.42E-06	1.08E-06	1.47E-05	2.06E-06	2.52E-05	3.58E-06	3.08E-07	8.07E-08
3	21798	4/16/2004	US Spacecraft	2.39E-06	3.67E-07	2.34E-06	4.40E-07	2.75E-06	4.23E-07	4.95E-06	7.86E-07	2.96E-07	7.64E-08
4	21798	4/16/2004	US Spacecraft	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	1.65E-07	4.64E-08
1	27053	2/28/2004	Russian Spacecraft	1.29E-05	2.16E-06	1.32E-05	1.95E-06	1.49E-05	2.50E-06	2.66E-05	4.42E-06	2.89E-07	4.06E-08
2	27053	2/28/2004	Russian Spacecraft	3.48E-06	1.68E-06	3.44E-06	1.18E-06	4.02E-06	1.93E-06	7.29E-06	3.52E-06	6.41E-08	1.61E-08
1	26960	12/19/2001	Rocket Body	7.36E-05	3.03E-06	6.21E-05	2.30E-06	8.50E-05	3.49E-06	1.54E-04	6.30E-06	5.84E-06	2.52E-07
2	26960	12/19/2001	Rocket Body	6.10E-05	2.46E-06	4.54E-05	2.10E-06	7.04E-05	2.84E-06	1.27E-04	5.16E-06	2.74E-06	1.02E-07
1	26040	11/21/2001	Russian Spacecraft	1.30E-04	3.46E-06	1.24E-04	3.47E-06	1.50E-04	3.99E-06	2.72E-04	7.26E-06	7.82E-06	2.06E-07
2	26040	11/21/2001	Russian Spacecraft	3.83E-05	1.70E-06	4.14E-05	1.35E-06	4.42E-05	1.96E-06	7.82E-05	3.46E-06	1.47E-06	8.41E-08
1	25942	3/11/2000	Rocket Body	1.07E-05	8.79E-07	8.70E-06	1.17E-06	1.24E-05	1.01E-06	2.31E-05	2.02E-06	5.85E-07	9.54E-08
2	25942	3/11/2000	Rocket Body	5.66E-06	7.26E-07	4.69E-06	6.30E-07	6.53E-06	8.38E-07	1.17E-05	1.50E-06	7.04E-07	6.67E-08
1	25088	11/22/1999	Russian Spacecraft	3.08E-05	1.90E-06	3.29E-05	1.88E-06	3.55E-05	2.20E-06	6.39E-05	3.97E-06	2.08E-06	1.07E-07
2	25088	11/22/1999	Russian Spacecraft	1.12E-05	1.72E-06	9.17E-06	9.95E-07	1.29E-05	1.98E-06	2.29E-05	3.45E-06	7.80E-07	1.28E-07
1	20390	4/18/1999	Rocket Body	2.36E-04	5.32E-06	1.94E-04	4.45E-06	2.72E-04	6.13E-06	4.95E-04	1.11E-05	1.60E-05	3.17E-07
2	20390	4/18/1999	Rocket Body	2.43E-04	4.03E-06	2.11E-04	3.66E-06	2.80E-04	4.65E-06	5.04E-04	8.30E-06	1.92E-05	4.24E-07
3	20390	4/19/1999	Rocket Body	2.21E-04	3.87E-06	1.70E-04	3.09E-06	2.55E-04	4.46E-06	4.51E-04	7.95E-06	1.33E-05	2.31E-07

NOTES

1) When the Pc of run 2 or 3 is larger than that of run 1 it is highlighted in yellow
2) All simulations generate the Pc in respect to the ISS
3) Run 1 starts at the breakup time and ends 10 days later
4) Run 2 starts 10 days after breakup time and ends 20 days after breakup time
5) Run 3 starts 20 days after breakup time and ends 30 days after breakup time
6) 10-day Integrated Pc = sum of all individual conjunctions
7) 10-day Max Pc = Maximum individual conjunction Pc experienced in the entire 10-day simulation
8) All Pc's are averaged over all of the simulations
9) Breakup Uncertainty Time = 10 minutes
10) Number of simulations for each scenario = 25
11) Number of days for each simulation = 10

Comparison of the Asset to the Satellite

A.3

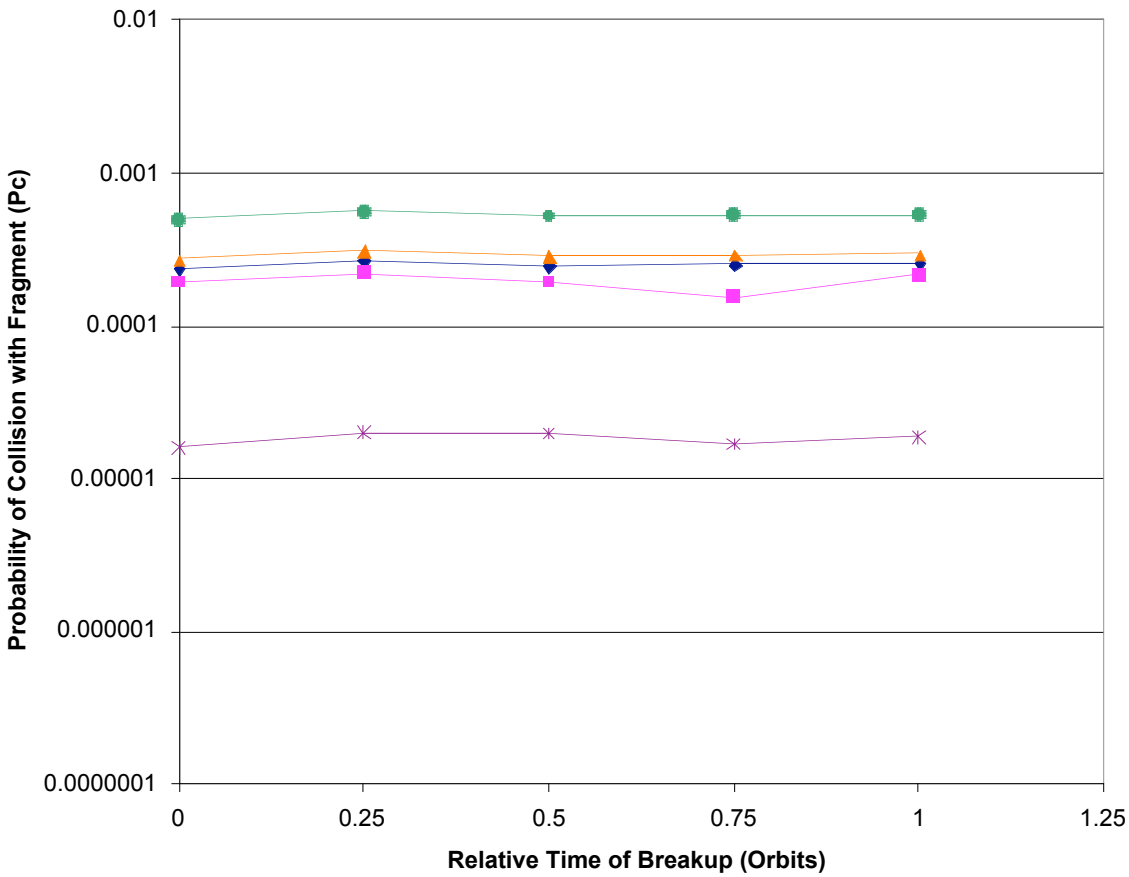
Satellite #	Highest 10-Day Integrated Pc	Difference in Inc. of Asset and Satellite (Degrees)	Difference in RAAN of Asset and Satellite (Degrees)	Difference in Ecc. of Asset and Satellite	Difference in AOP of Asset and Satellite (Degrees)
20390	4.95E-04	22	140	0.00055	41
26040	2.72E-04	13	38	0.00009	103
26960	1.54E-04	46	171	0.00853	5
25088	6.39E-05	13	138	0.01228	48
27474	5.39E-05	12	163	0.04114	54
27053	2.66E-05	13	164	0.01253	41
25942	2.31E-05	47	136	0.00053	37
21798	1.90E-05	47	159	0.00016	21
27475	1.36E-05	12	112	0.04288	175
22250 ISS	7.75E-06	5	146	0.28341	132
22250 Shuttle	6.56E-06	5	146	0.21282	138
23631	6.43E-06	13	135	0.57826	8
19537	5.59E-06	13	124	0.58226	14
21734	3.09E-06	31	109	0.00367	85
27856	1.07E-06	13	84	0.00500	159
20631	5.19E-07	13	83	0.57848	87
22032	4.99E-07	45	177	0.66539	121
23887	2.69E-07	5	177	0.29423	60
22925	2.45E-07	5	51	0.45819	86
19856	1.31E-07	14	67	0.55927	59
21766	6.70E-08	44	55	0.68114	75
19773	2.06E-08	43	137	0.71830	71
21226	9.72E-09	13	21	0.58094	24
19122	5.29E-09	45	109	0.71538	159
22274	2.48E-09	5	83	0.61349	51
19535	1.42E-09	14	101	0.55529	0
15714	0.00E+00	0	132	0.57636	31
18375	0.00E+00	14	138	0.55460	34
20583	0.00E+00	10	21	0.14190	130
21012	0.00E+00	14	124	0.56263	103
22066	0.00E+00	13	17	0.57470	161
Correlation to Pc		0.08	0.08	-0.44	-0.15

Breakup Caused Max Integrated Pc>E-05

Breakup Caused Max Integrated Pc<E-07

Satellite 20390 -- Russian Rocket Body (4/18/99)

Variation in Time of Breakup
10-Day Integrated Pc



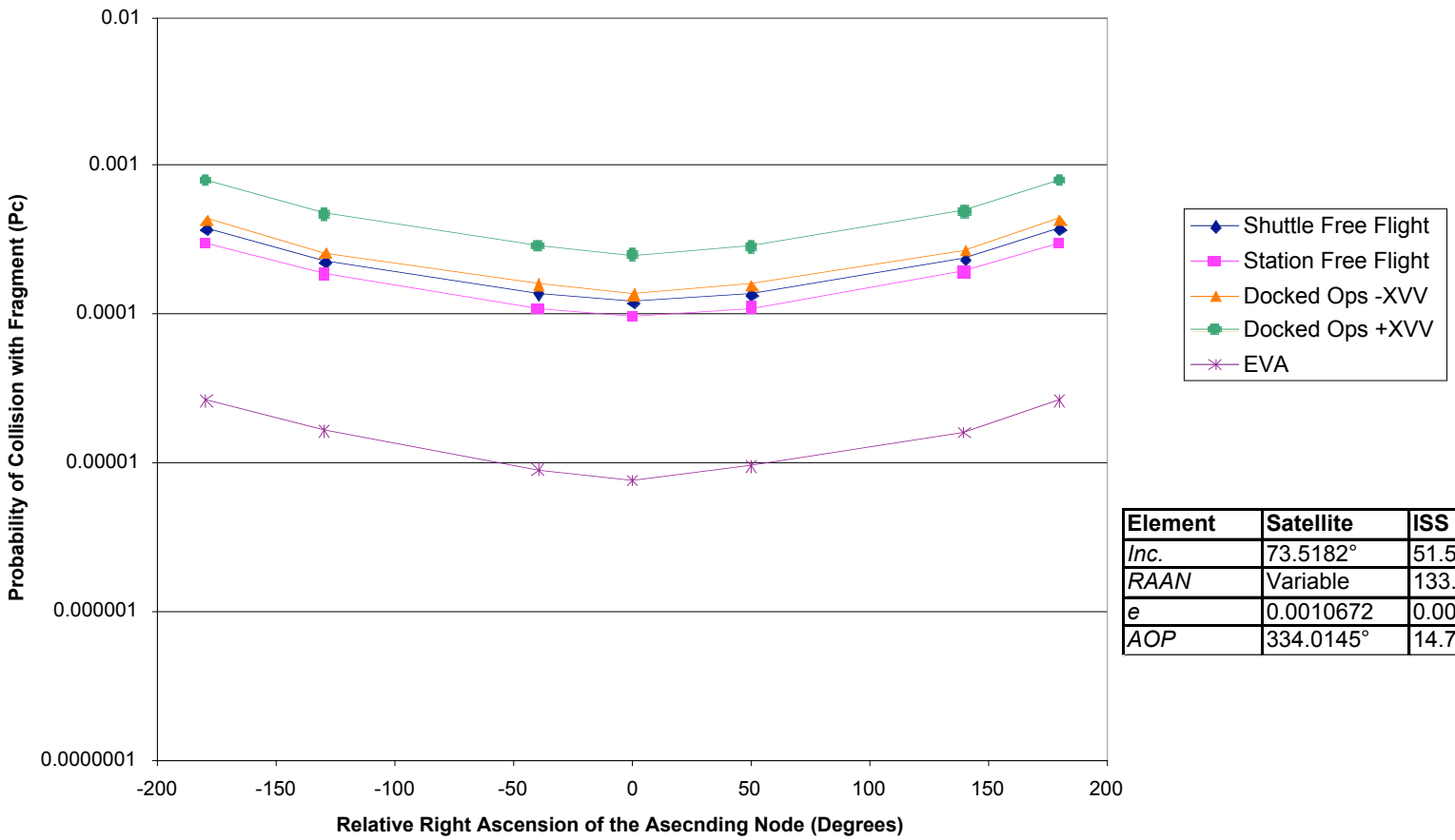
Element	Satellite	ISS
Inc.	73.5182°	51.5915°
RAAN	Variable	133.6202°
e	0.0010672	0.0005137
AOP	334.0145°	14.7018°

NOTE: Original Breakup Time = 108.055 Decimal Days

Satellite 20390 -- Russian Rocket Body (4/18/99)

Variation in Right Ascension of the Ascending Node

10-Day Integrated Pc



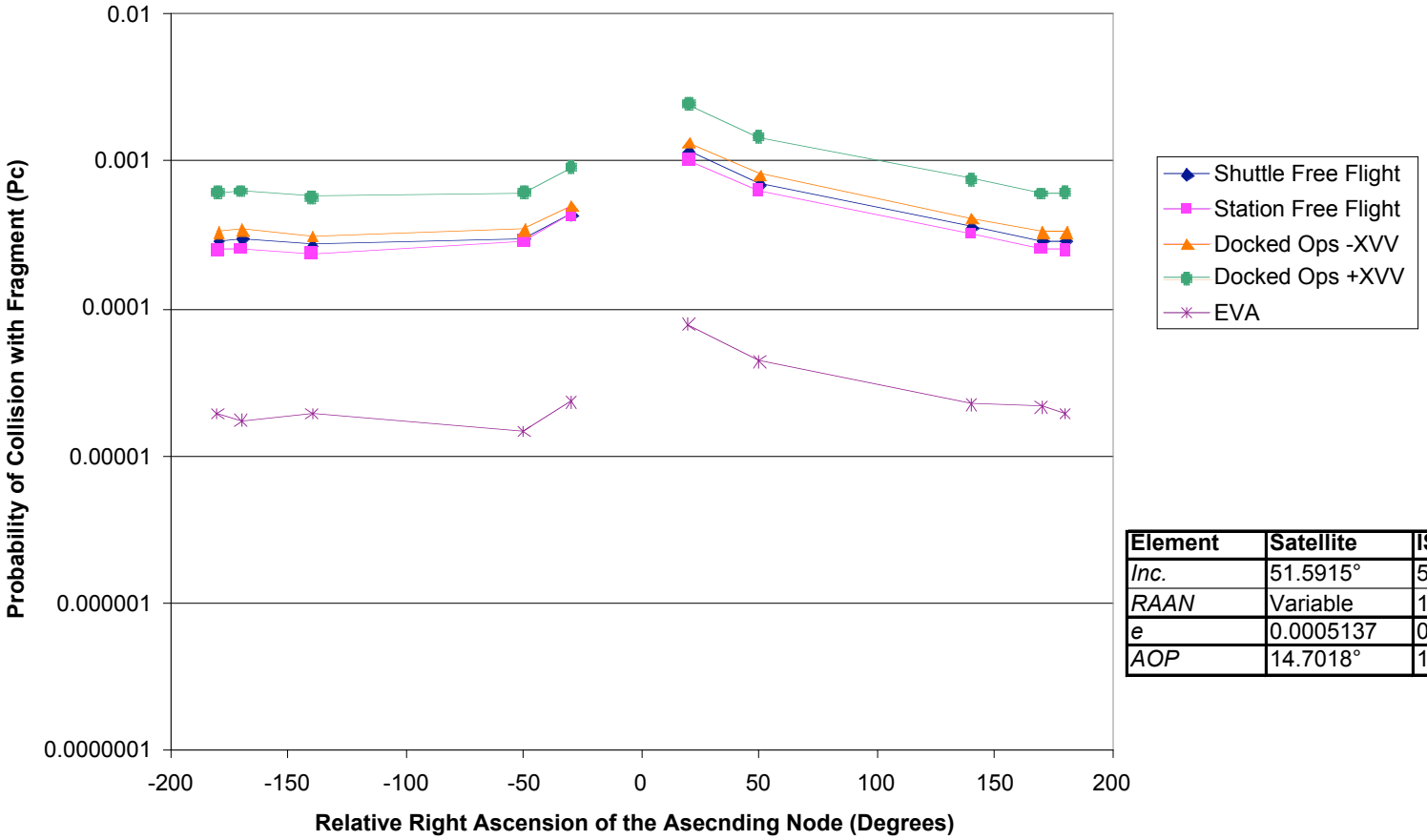
Element	Satellite	ISS
Inc.	73.5182°	51.5915°
RAAN	Variable	133.6202°
e	0.0010672	0.0005137
AOP	334.0145°	14.7018°

B.2b

Satellite 25544 -- International Space Station (4/18/99)

Variation of Right Ascension of the Ascending Node

10-Day Integrated Pc



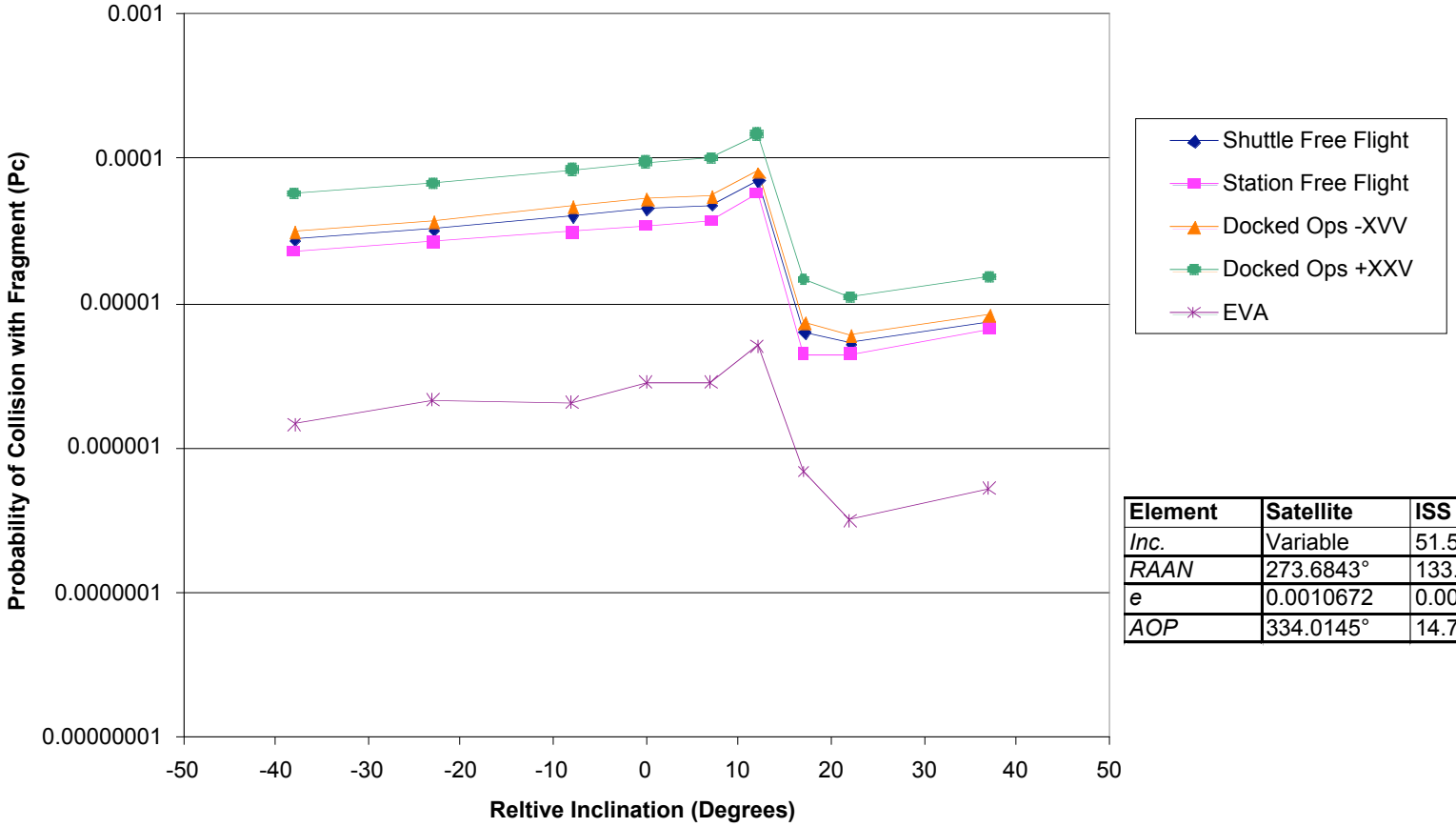
Element	Satellite	ISS
Inc.	51.5915°	51.5915°
RAAN	Variable	133.6202°
e	0.0005137	0.0005137
AOP	14.7018°	14.7018°

C.1

Satellite 20390 -- Russian Rocket Body (4/18/99)

Variation of Inclination

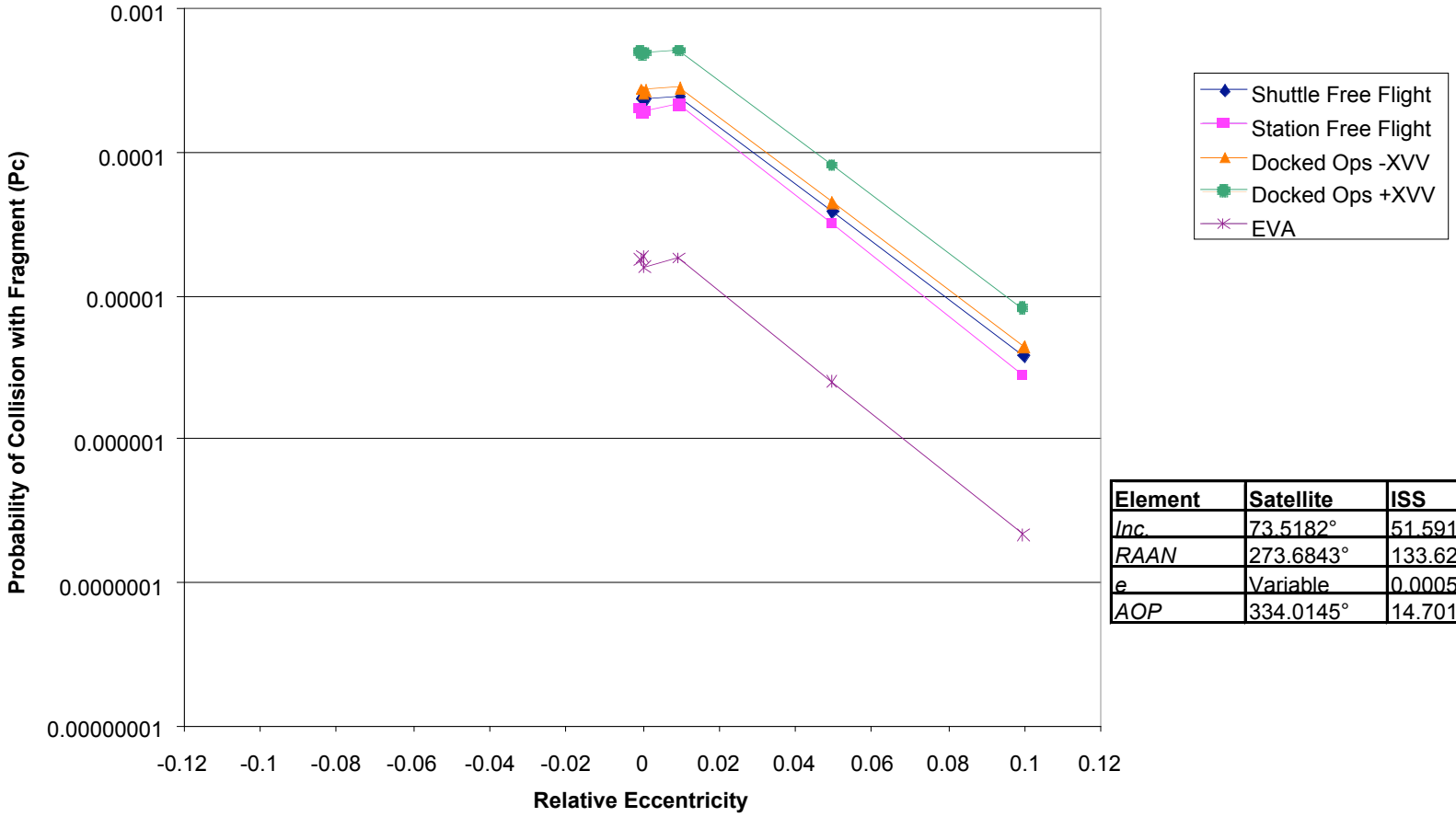
10-Day Max Pc



Element	Satellite	ISS
Inc.	Variable	51.5915°
RAAN	273.6843°	133.6202°
e	0.0010672	0.0005137
AOP	334.0145°	14.7018°

Satellite 20390 -- Rocket Body (4/18/99)

Variation of Eccentricity
10-Day Integrated Pc

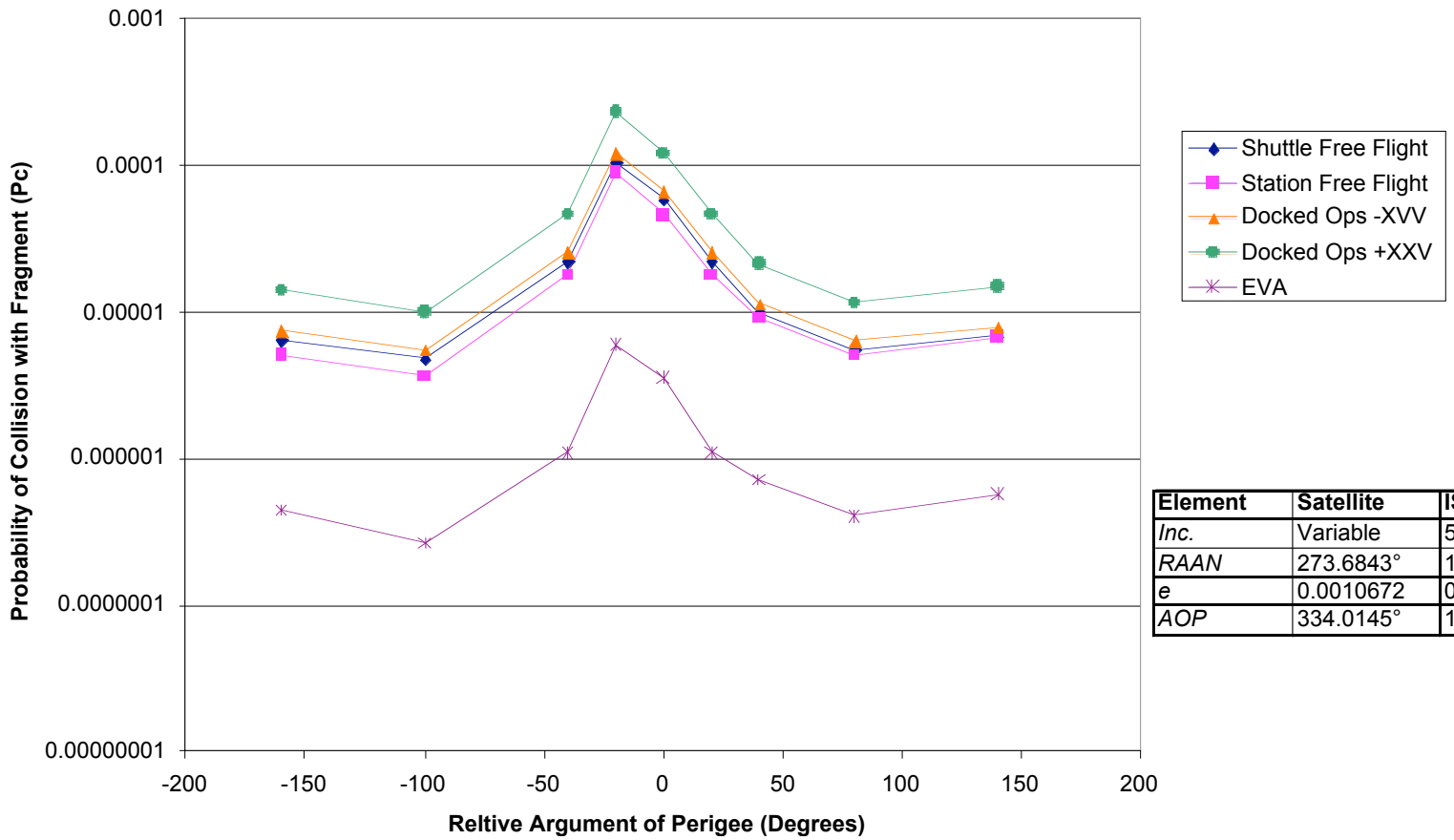


Element	Satellite	ISS
Inc.	73.5182°	51.5915°
RAAN	273.6843°	133.6202°
e	Variable	0.0005137
AOP	334.0145°	14.7018°

Satellite 20390 -- Russian Rocket Body (4/18/99)

Variation of Argument of Perigee

10-Day Max Pc

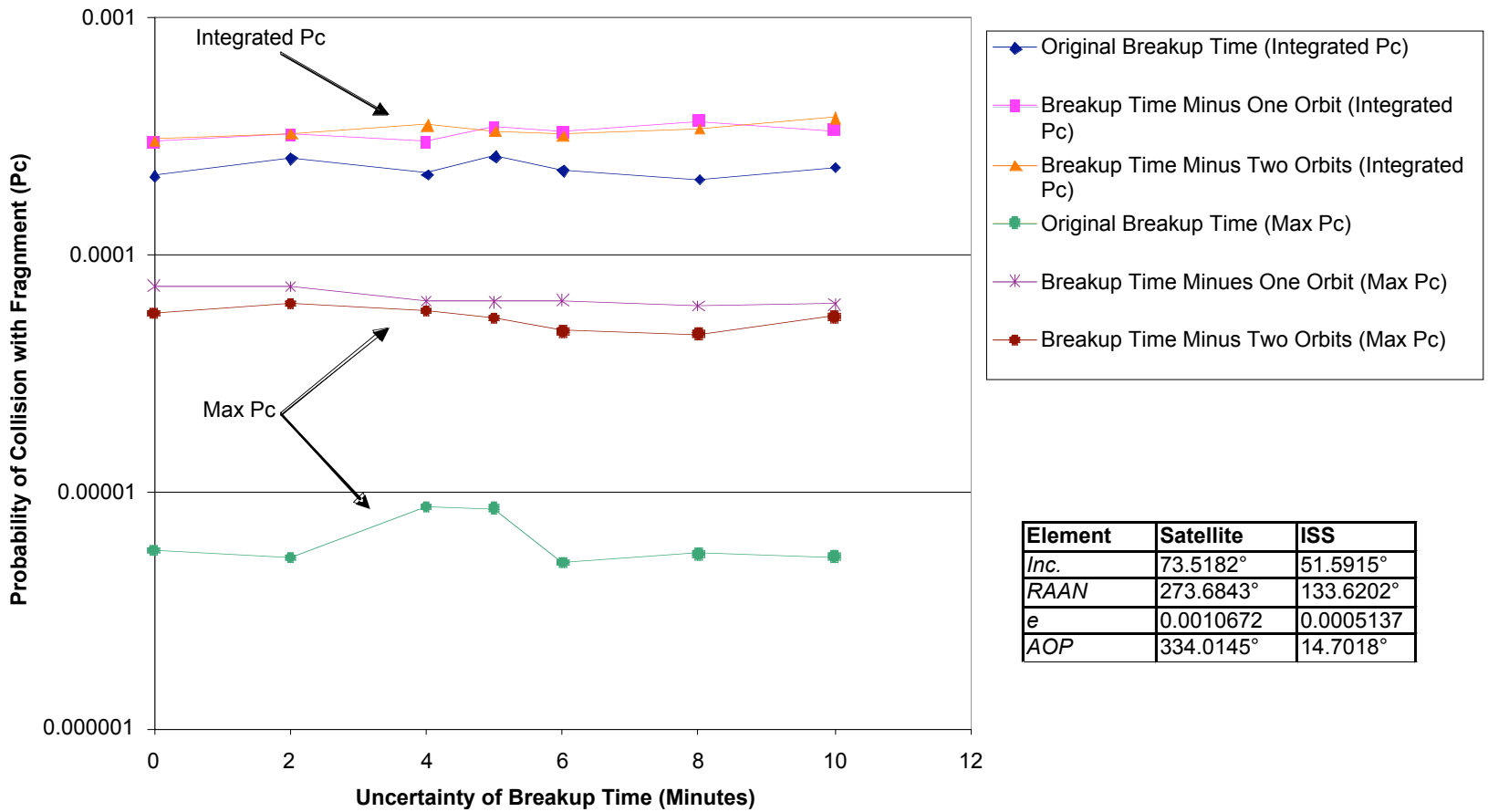


Element	Satellite	ISS
Inc.	Variable	51.5915°
RAAN	273.6843°	133.6202°
e	0.0010672	0.0005137
AOP	334.0145°	14.7018°

Satellite 20390 -- Russian Rocket Body (4/18/99)

Variation of Uncertainty of Breakup Time

Shuttle Free Flight

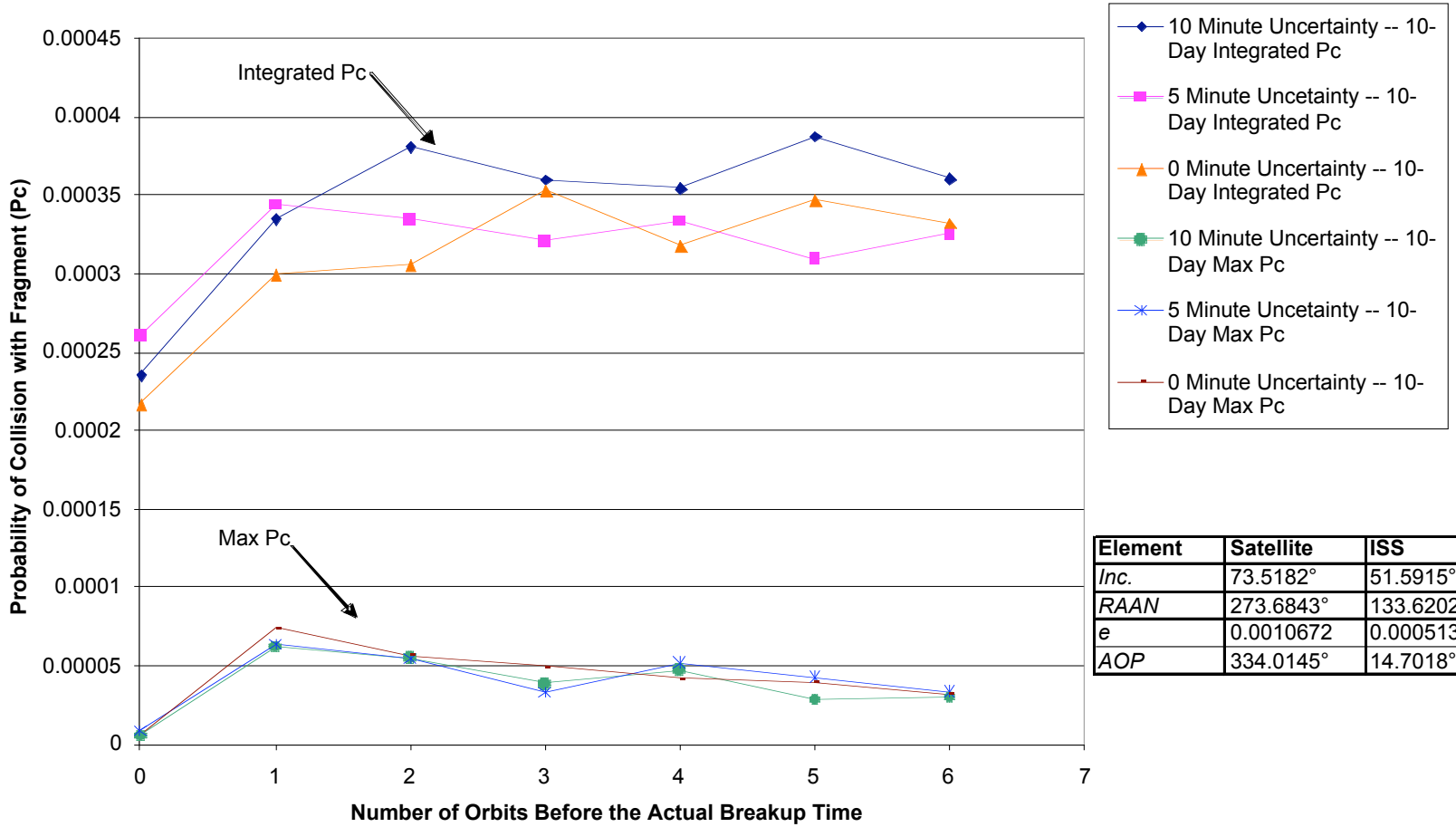


Element	Satellite	ISS
Inc.	73.5182°	51.5915°
RAAN	273.6843°	133.6202°
e	0.0010672	0.0005137
AOP	334.0145°	14.7018°

Satellite 20390 -- Russian Rocket Body (4/18/99)

Variation of Breakup Orbit

Shuttle Free Flight

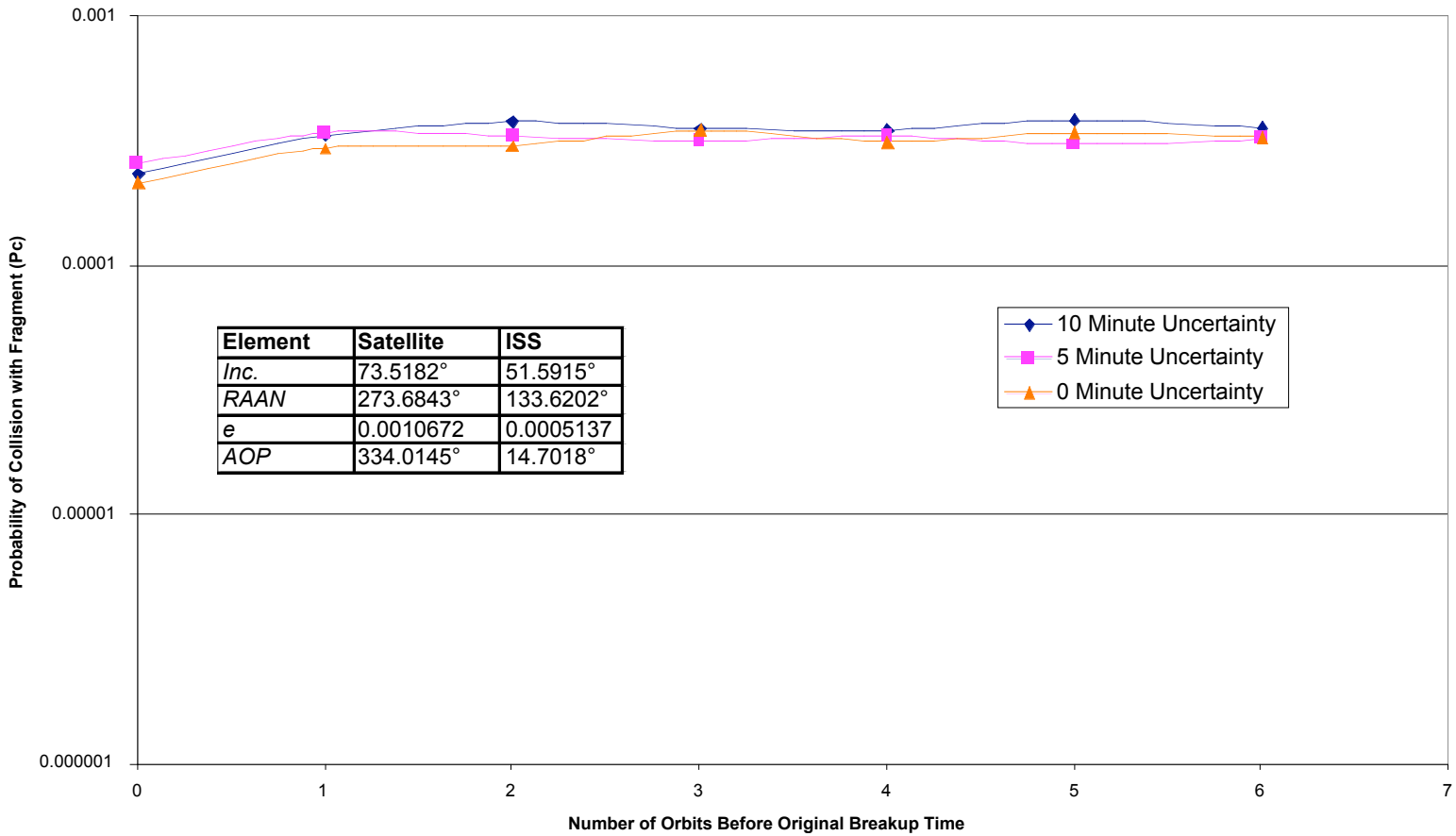


Element	Satellite	ISS
Inc.	73.5182°	51.5915°
RAAN	273.6843°	133.6202°
e	0.0010672	0.0005137
AOP	334.0145°	14.7018°

Satellite 20390 -- Russian Rocket Body (4/18/99)

10-Day Integrated Pc for Different Breakup Orbits

Shuttle Free Flight

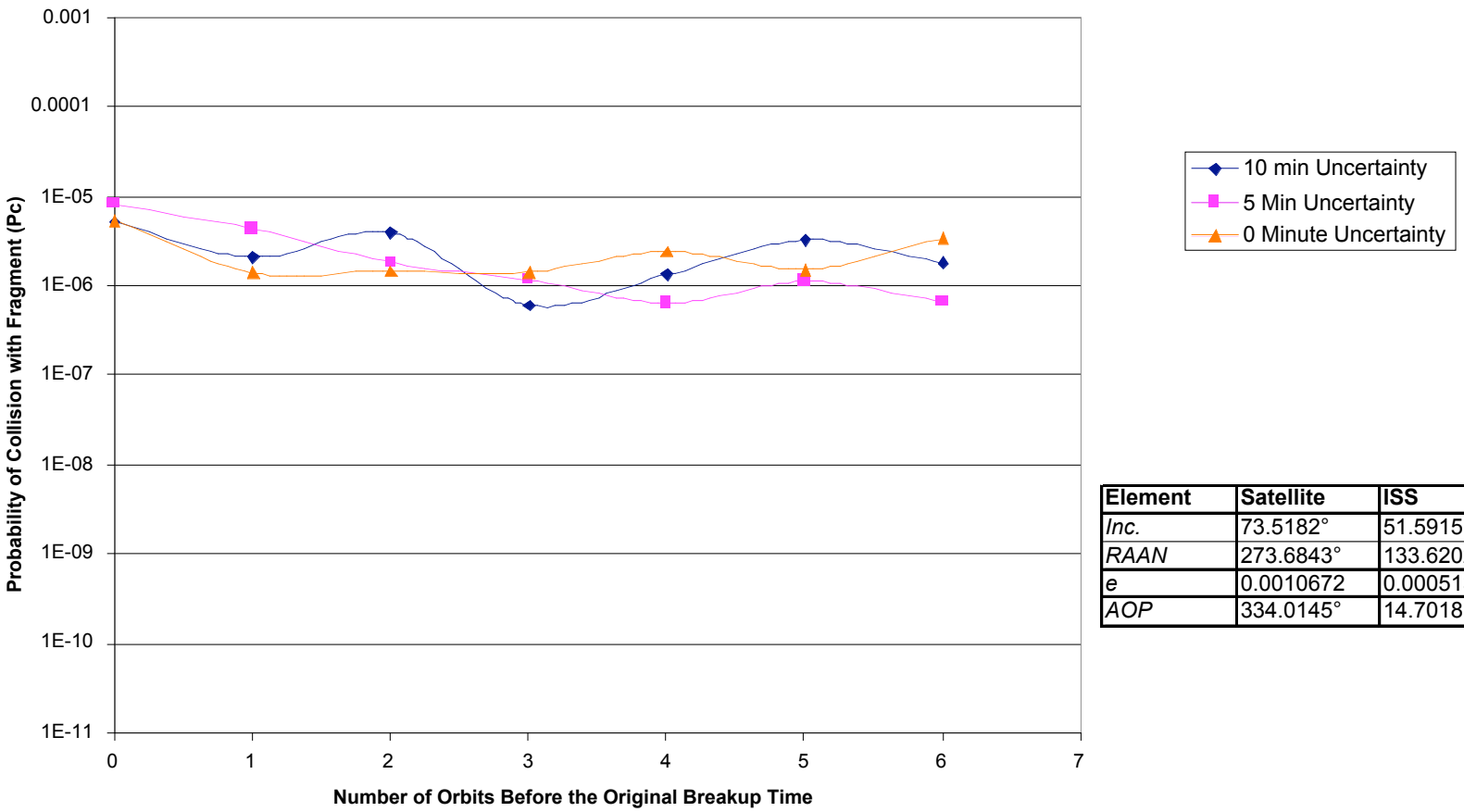


Element	Satellite	ISS
Inc.	73.5182°	51.5915°
RAAN	273.6843°	133.6202°
e	0.0010672	0.0005137
AOP	334.0145°	14.7018°

Satellite 20390 -- Russian Rocket Body (4/18/99)

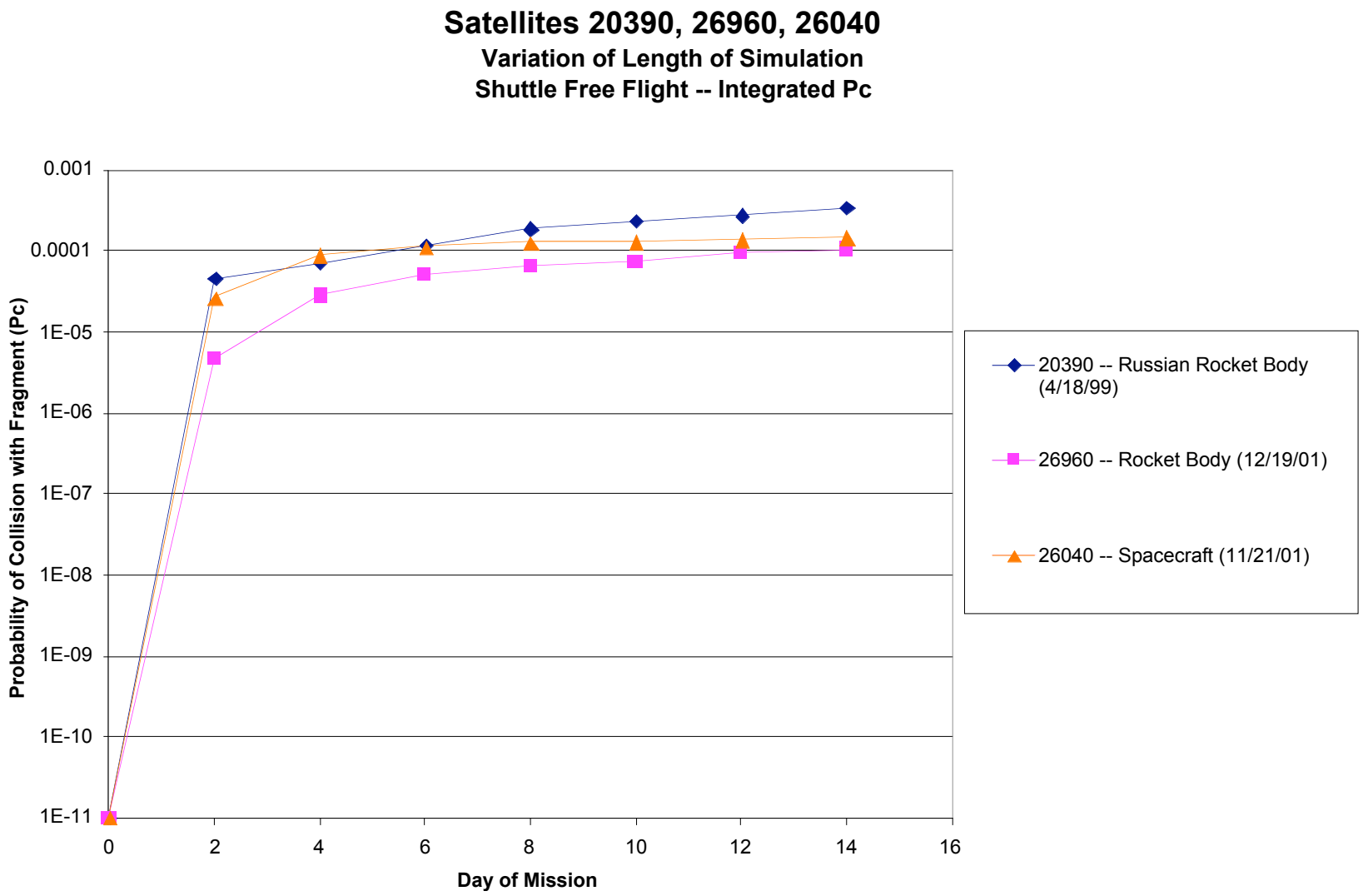
Pc at the Time of Original Highest Risk for Different Breakup Orbits

Shuttle Free Flight

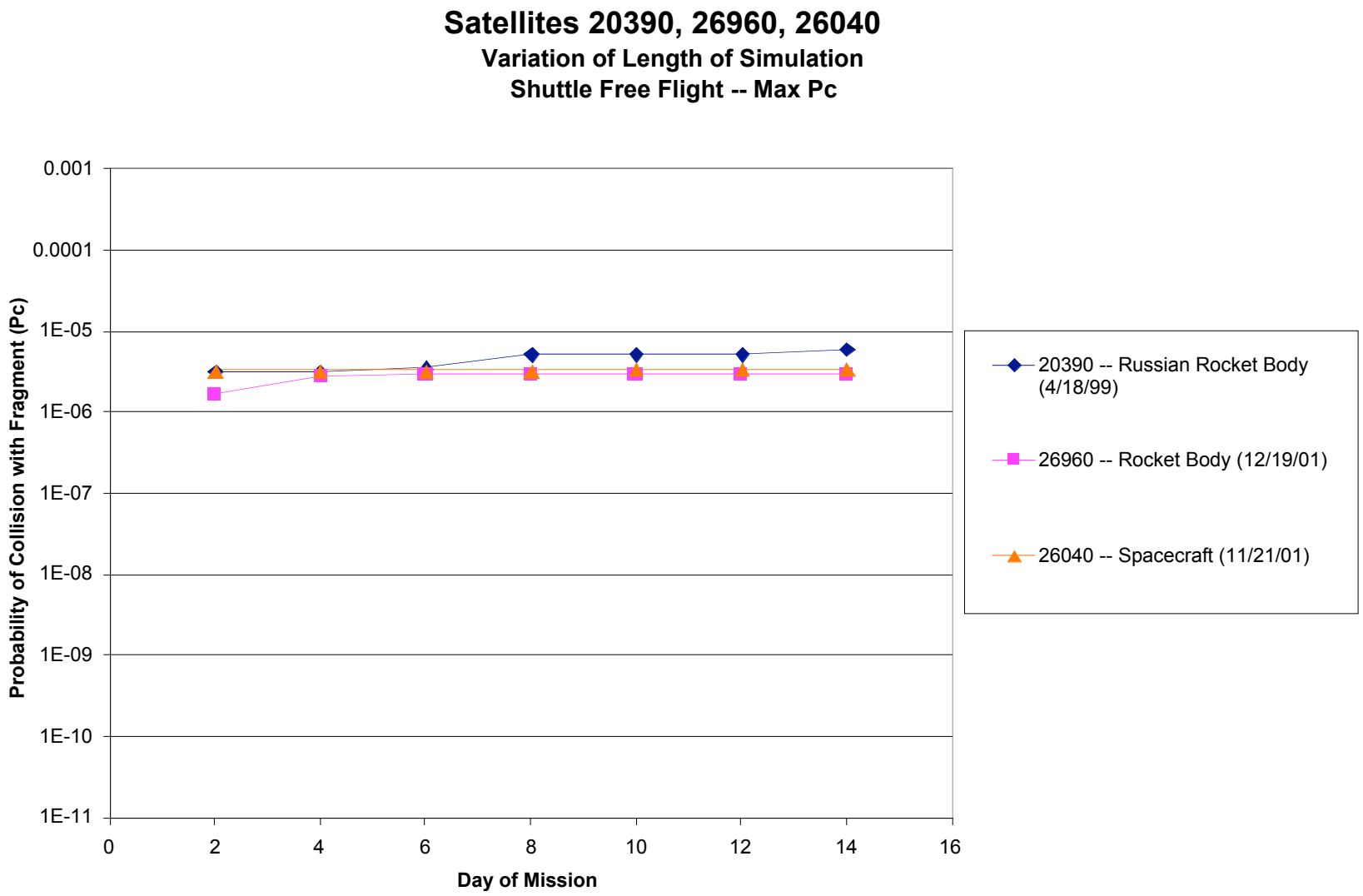


Element	Satellite	ISS
Inc.	73.5182°	51.5915°
RAAN	273.6843°	133.6202°
e	0.0010672	0.0005137
AOP	334.0145°	14.7018°

E.1a

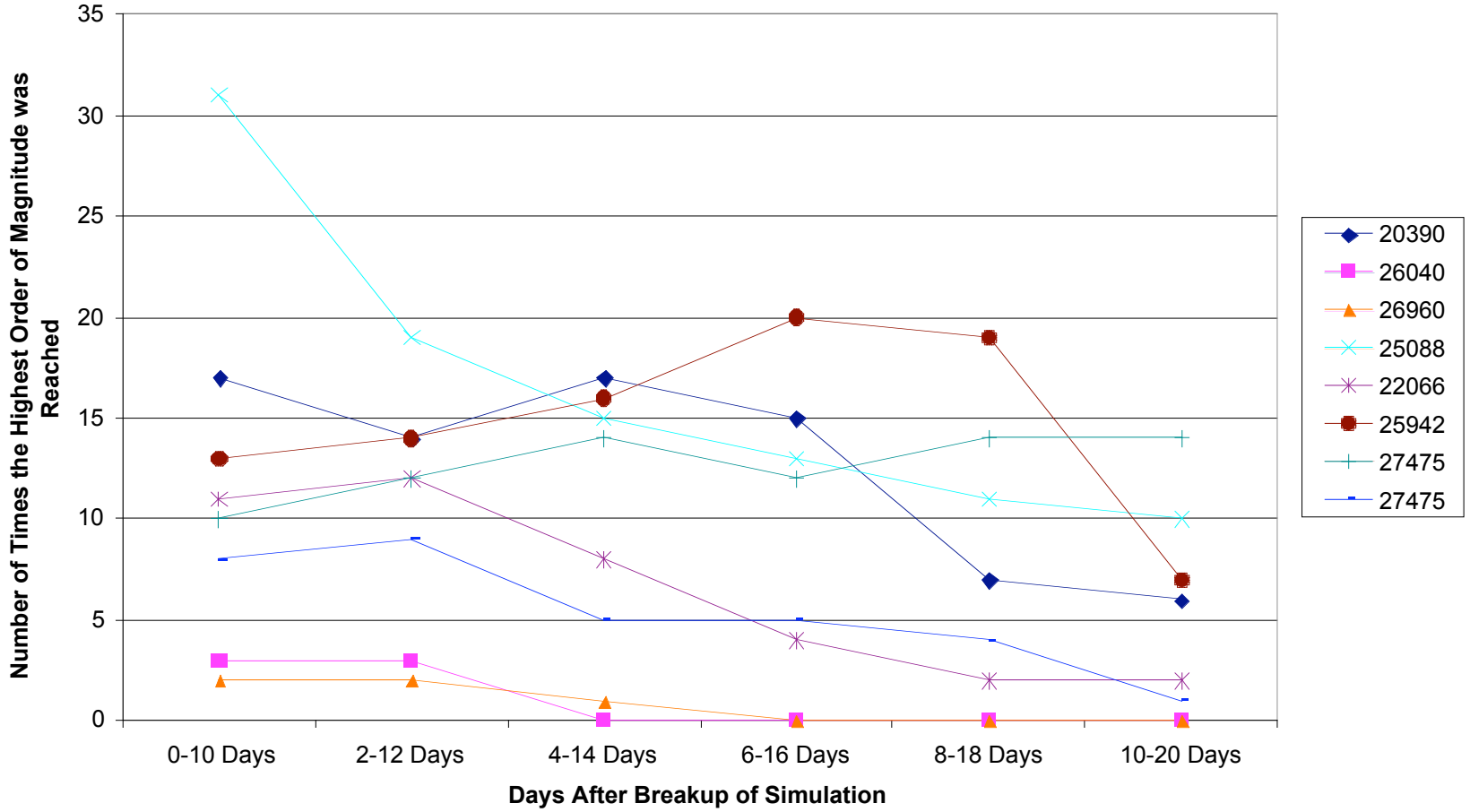

Satellites 20390, 26960, 26040
Variation of Length of Simulation
Shuttle Free Flight -- Integrated Pc
Probability of Collision with Fragment (Pc)
0.001
0.0001
1E-05
1E-06
1E-07
1E-08
1E-09
1E-10
1E-11
0
2
4
6
8
10
12
14
16
Day of Mission
20390 -- Russian Rocket Body (4/18/99)
26960 -- Rocket Body (12/19/01)
26040 -- Spacecraft (11/21/01)

E.1b


Satellites 20390, 26960, 26040
Variation of Length of Simulation
Shuttle Free Flight -- Max Pc
Probability of Collision with Fragment (Pc)
0.001
0.0001
1E-05
1E-06
1E-07
1E-08
1E-09
1E-10
1E-11
0
2
4
6
8
10
12
14
16
Day of Mission
20390 -- Russian Rocket Body (4/18/99)
26960 -- Rocket Body (12/19/01)
26040 -- Spacecraft (11/21/01)

E.2

Number of Times a Single Encounter Pc Reached Its Highest Order of Magnitude for a Specified Satellite

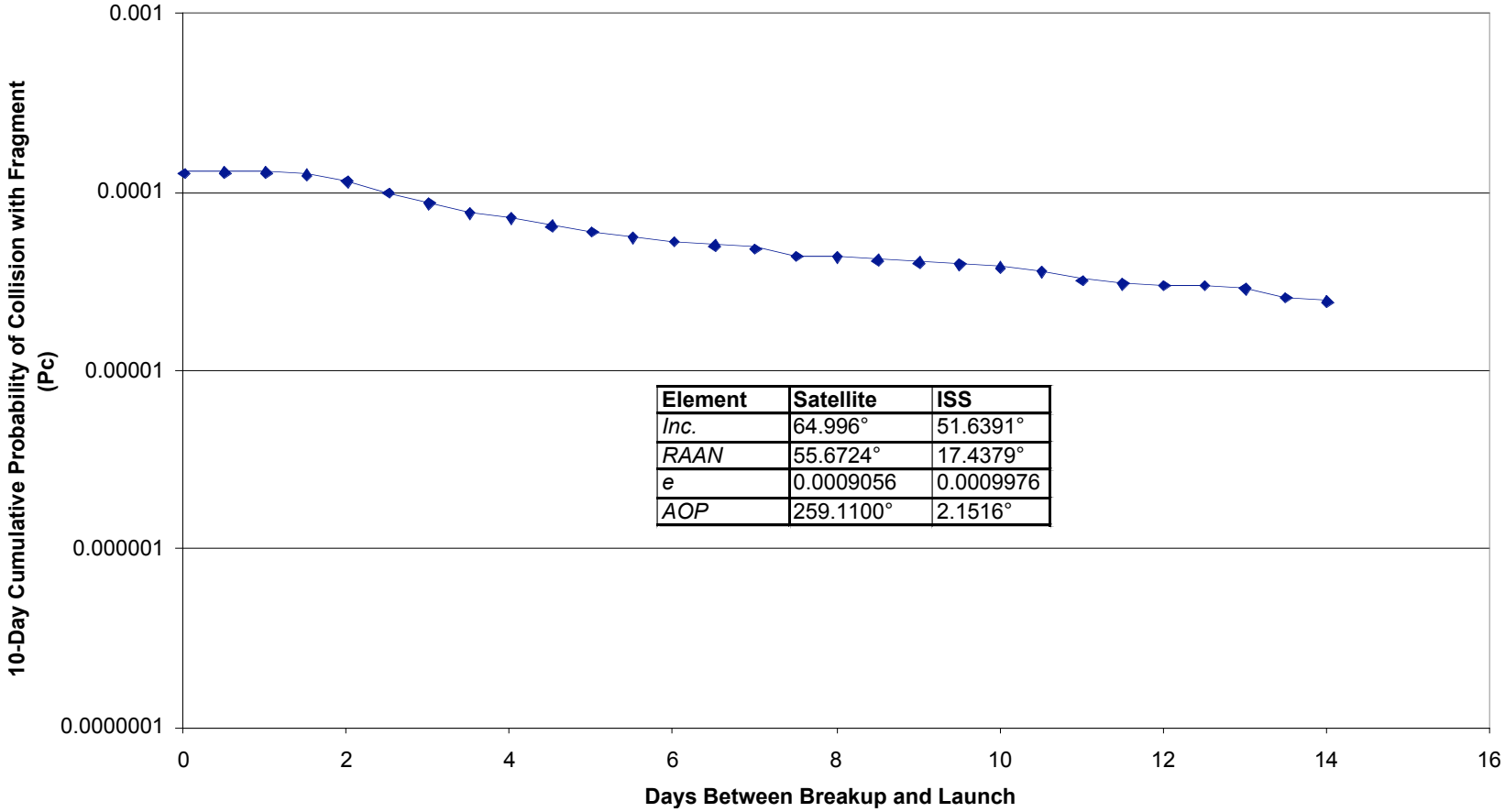


E.3a

Satellite 26040 -- Spacecraft (11/21/01)

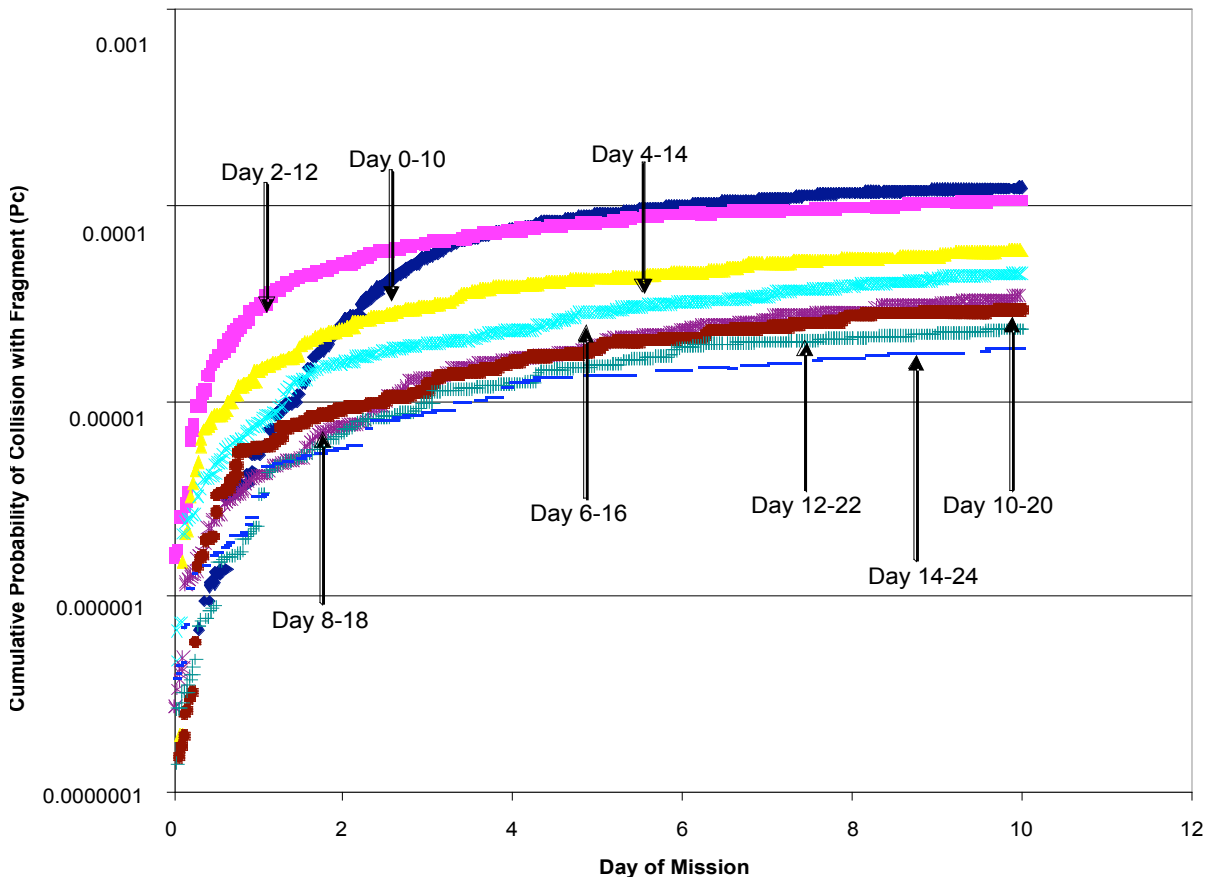
Variation of Launch Day

Shuttle Free Flight



Element	Satellite	ISS
Inc.	64.996°	51.6391°
RAAN	55.6724°	17.4379°
e	0.0009056	0.0009976
AOP	259.1100°	2.1516°

Satellite 26040 -- Spacecraft (11/21/99)
Variation in Launch Day
Shuttle Free Flight



Days of Mission Relative to Breakup Day

- day 0-10
- day 2-12
- day 4-14
- day 6-16
- day 8-18
- day 10-20
- day 12-22
- day 14-24

Element	Satellite	ISS
Inc.	64.996°	51.6391°
RAAN	55.6724°	17.4379°
e	0.0009056	0.0009976
AOP	259.1100°	2.1516°

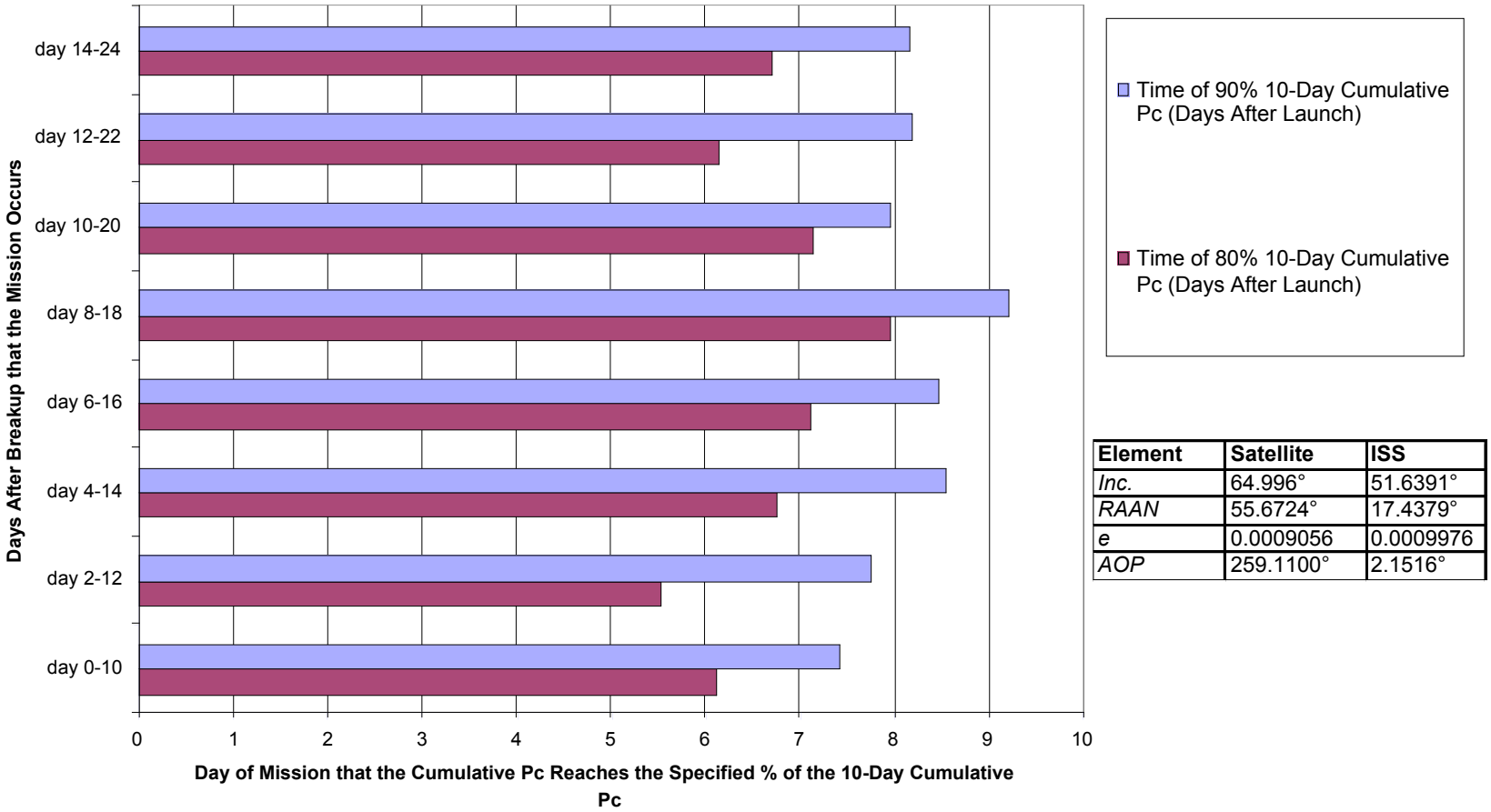
Note: Purpose of graph is to show that there is no trend in slope of the curves

E.3c

Satellite 26040 -- Spacecraft (11/21/99)

Variation of Launch Day

Shuttle Free Flight



Element	Satellite	ISS
Inc.	64.996°	51.6391°
RAAN	55.6724°	17.4379°
e	0.0009056	0.0009976
AOP	259.1100°	2.1516°

Optimal Relative Attitude (ORA) for the Shuttle

E.4a

	Shuttle Free Flight	Shuttle Attitude -XRV, Optimal Relative Attitude (ORA)
Satellite #	**10-Day Max Pc**	**10-Day Max Pc**
20390 -- Russian Rocket Body (4/18/99)	5.32E-06	7.87E-07
26960 -- Rocket Body (12/19/01)	3.46E-06	5.12E-07
26040 -- Spacecraft (11/21/01)	3.03E-06	4.48E-07

NOTES

1) All simulations generate the Pc in respect to the ISS
2) All simulations start at the breakup time and end 10 days later
3) 10-day Integrated Pc = sum of all individual conjunctions
4) 10-day Max Pc = Maximum individual conjunction Pc experienced in the entire simulation
5) All Pc's are averaged over all of the simulations
6) Breakup Uncertainty Time = 10 minutes
7) Number of simulations for each scenario = 25
8) Number of days for each simulation = 10
9) Pc's over 10^{-5} are highlighted in yellow
10) Pc's over 10^{-4} are highlighted in red

Optimal Relative Attitude (ORA) for the Station

E.4b

	Station Free Flight	Station in +XRV, Optimal Relative Attitude (ORA)	Difference in Order of Magnitude
Satellite #	**10-Day Max Pc**	**10-Day Max Pc**	**10-Day Max Pc**
20390 -- Russian Rocket Body (4/18/99)	4.45E-06	8.88E-07	0.70
26960 -- Rocket Body (12/19/01)	3.47E-06	7.54E-07	0.66
26040 -- Spacecraft (11/21/01)	2.30E-06	4.33E-07	0.72

NOTES

1) All simulations generate the Pc in respect to the ISS
2) All simulations start at the breakup time and end 10 days later
3) 10-day Integrated Pc = sum of all individual conjunctions
4) 10-day Max Pc = Maximum individual conjunction Pc experienced in the entire simulation
5) All Pc's are averaged over all of the simulations
6) Breakup Uncertainty Time = 10 minutes
7) Number of simulations for each scenario = 25
8) Number of days for each simulation = 10
9) Pc's over 10^{-5} are highlighted in yellow
10) Pc's over 10^{-4} are highlighted in red

Appendix F
Acronym and Shorthand Notation List

Δ AOP – Difference in AOP between Asset and Satellite
Δ Ecc – Difference in Eccentricity between Asset and Satellite
Δ Inc – Difference in Inclination between Asset and Satellite
Δ RAAN – Difference in RAAN between Asset and Satellite
Δ Time – Difference in Time from Original Breakup Time
AOP – Argument of Perigee
AOP_{ISS} – AOP of the ISS
AOP_{SAT} – AOP of the Satellite
DM - Flight Design and Dynamics Division
FCT – Flight Control Team
HITF – Hypervelocity Impact Technology Facility
ISS – International Space Station
KX2 – Space Debris Research Group
MDF – Minimum Duration Flight
MMOD – Micrometeoroid Debris
NOAA – National Oceanic and Atmospheric Administration
OM – Order of Magnitude
ORA – Optimal Relative Attitude
Pc – Probability of Collision
$Pc(t_o)$ – Pc at the Time of Original Highest Risk
Pc/m^2 – Probability of Collision per Square Meter
RAAN – Right Ascension of the Ascending Node
$RAAN_{ISS}$ – RAAN of the ISS
$RAAN_{SAT}$ – RAAN of the Satellite
SBRAM – Satellite Breakup Risk Assessment Model
Scientific Notation: n.n E-n = n.n x 10^{-n}
SMA – Semi Major Axis
ΔSMA – Difference in SMA between Asset and Satellite
t_b – Original Breakup Time
TLE – Two-line Element
USSTRATCOM – US Strategic Command
-XRV – Shuttle's ORA (the Shuttle aligns the -X body axis with the relative velocity vector between the Shuttle and the debris cloud)
+XRV – Station's ORA (the Station aligns the +X body axis with the relative velocity vector between the Station and the debris cloud)

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

Public reporting burden for this collection of information is estimated to average 1 hour per response, including the time for reviewing instructions, searching existing data sources, gathering and maintaining the data needed, and completing and reviewing the collection of information. Send comments regarding this burden estimate or any other aspect of this collection of information, including suggestions for reducing this burden, to Washington Headquarters Services, Directorate for Information Operations and Reports, 1215 Jefferson Davis Highway, Suite 1204, Arlington, VA 22202-4302, and to the Office of Management and Budget, Paperwork Reduction Project (0704-0188), Washington, DC 20503.

1. AGENCY USE ONLY *(Leave Blank)*	2. REPORT DATE	3. REPORT TYPE AND DATES COVERED
	December 2005	NASA Contractor Report

4. TITLE AND SUBTITLE
Analysis of Past and Possible Threats Posed by Satellite Breakups and Potential Actions to Minimize Their Risk to Human Tended Vehicles

5. FUNDING NUMBERS

6. AUTHOR(S)
Leah Holmes

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)

Lyndon B. Johnson Space Center
Houston, Texas 77058

8. PERFORMING ORGANIZATION REPORT NUMBERS
S-966

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)

National Aeronautics and Space Administration
Washington, DC 20546-0001

10. SPONSORING/MONITORING AGENCY REPORT NUMBER
CR-2005-213691

11. SUPPLEMENTARY NOTES

12a. DISTRIBUTION/AVAILABILITY STATEMENT

Available from the NASA Center for AeroSpace Information (CASI)
7121 Standard
Hanover, MD 21076-1320
Category: 13

12b. DISTRIBUTION CODE

13. ABSTRACT *(Maximum 200 words)*
There have been 30 known satellite breakups from First Element Launch in November of 1998 to June of 2005. While breakups have been studied, the threat that they cause to human space flight is not well understood. Previous studies simply assessed the probability of collision per square meter (Pc/m2) and never considered different particle diameters or vulnerable areas of vehicles. Also, no flight rules exist to aid the crew and flight control team (FCT) in response to a satellite breakup, causing concern for the safety of crew members and vehicles. This study fills the gaps left by previous tests and provides information that will be used to create flight rules that will increase safety during human spaceflight. .

14. SUBJECT TERMS	15. NUMBER OF PAGES	16. PRICE CODE
satellite breakup, human tended vehicles, SBRAM, shuttle, satellite	50	

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	Unlimited