Exhibit 3
FCC Form 442
Orbital Debris Mitigation
HawkEye 360 Pathfinder Cluster

This report is presented in compliance with
NASA-STD-8719.14, APPENDIX A.

Document Data is Not Restricted.
This document contains no proprietary, ITAR, or export controlled information.
DAS Software Version Used In Analysis: v2.0.2

INTRODUCTION

1. HawkEye 360 ("HE360"), a US company headquartered in Herndon, Virginia, plans to launch three experimental microsatellites (the "Hawk satellites") in the 4th quarter of 2017. The satellites will fly in proximate formation and work together to form a single observation platform (the "Pathfinder cluster"). The expected maximum operational lifetime of the satellites is <7 years.

2. The satellites are designed to operate in circular sun-synchronous orbits with a nominal altitude of 575 km and inclination between 97 and 98 degrees and are calculated to re-enter the Earth's atmosphere and burn up completely in 7 years or less. Due to the composition and small size of the satellites (200 mm x 267 mm x 440 mm prism weighing less than 13 kg) the entire satellite will burn up and be consumed due to atmospheric heating. There is 0% probability of human casualty as no large or small pieces of the spacecraft will survive to the Earth's surface.

3. Due to the International Space Station's higher orbital inclination angle and much higher altitude, there is no possibility of collision between the Hawk satellites and the International Space Station.

4. The NASA Debris Assessment Software confirmed that the Pathfinder cluster satisfies all of the Requirements for Limiting Orbital Debris including:
   a. Mission-Related Debris Passing Through LEO
   b. Mission-Related Debris Passing Near GEO
   c. Long-Term Risk from Planned Breakups
   d. Probabiliy of Collision With Large Objects
   e. Probability of Damage from Small Objects
   f. Postmission Disposal
   g. Casualty Risk from Reentry Debris

5. HE360 confirms that the Hawk satellites will not undergo any planned release of debris during their normal operations. In addition, all separation and deployment mechanisms, and any other potential source of debris will be retained by the spacecraft or launch vehicle. HE360 has also assessed the probability of the space stations becoming sources of debris by collision with small debris or meteoroids of less than one centimeter in diameter that could cause loss of control and prevent post-mission disposal. HE360 has taken steps to limit the effects of such collisions through shielding, the placement of components, and the use of redundant systems.

6. HE360 has assessed and limited the probability of accidental explosions during and after completion of mission operations through a failure mode verification analysis. As part of the satellite manufacturing process, HE360 has taken steps to ensure that debris generation will not result from the conversion of energy sources on board the satellites into energy that fragments the satellites. All sources of stored energy onboard the spacecraft will have been depleted or safely contained when no longer required for mission operations or post-mission disposal.

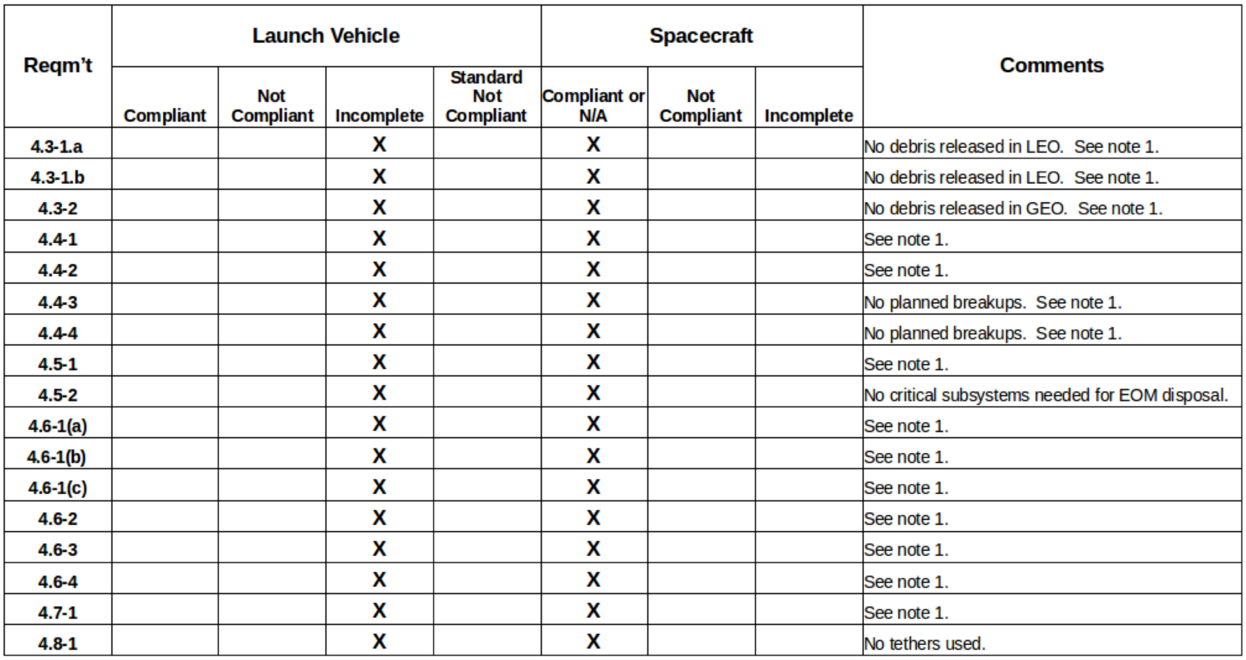
7. HE360 has assessed and limited the probability of the space stations becoming a source of debris by collisions with large debris or other operational spacecraft. HE360 does not intend to place any of the Hawks in an orbit that is identical to or very similar to an orbit used by other space stations, and, in any event, will work closely with the cluster launch providers to ensure that the satellites are deployed in such a way as to minimize the potential for collision with any other spacecraft. This specifically includes minimizing the potential for collision with manned spacecraft.

8. The Hawk satellites will perform station-keeping maneuvers to maintain separation between the Hawks in the cluster and sustain the desired geometry. Typical inter-satellite distances between the satellites will be approximately 125 km and maintenance maneuvers will be conducted relatively infrequently – approximately once a week. However, the cluster will not maintain the satellites' inclination angles, apogees, perigees, and right ascension of the ascending node to any specified degrees of accuracy beyond the goals of maintaining the cluster geometry.

9. HE360's disclosure of the above parameters, as well as the number of space stations, the number and inclination of orbital planes, and the orbital period to be used, can assist third parties in identifying potential problems that may be the result of proposed operations. This information also lends itself to coordination between HE360 and other operators located in similar orbits.

1. Self Assessment of the ODAR using the format in Appendix A.2 of NASA-STD-8719.14

A self assessment is provided below in accordance with the assessment format provided in Appendix A.2 of NASA-STD-8719.14.



Reqm't	Launch Vehicle				Spacecraft			Comments
	Compliant	Not Compliant	Incomplete	Standard Not Compliant	Compliant or N/A	Not Compliant	Incomplete	
4.3-1.a			X		X			No debris released in LEO. See note 1.
4.3-1.b			X		X			No debris released in LEO. See note 1.
4.3-2			X		X			No debris released in GEO. See note 1.
4.4-1			X		X			See note 1.
4.4-2			X		X			See note 1.
4.4-3			X		X			No planned breakups. See note 1.
4.4-4			X		X			No planned breakups. See note 1.
4.5-1			X		X			See note 1.
4.5-2			X		X			No critical subsystems needed for EOM disposal.
4.6-1(a)			X		X			See note 1.
4.6-1(b)			X		X			See note 1.
4.6-1(c)			X		X			See note 1.
4.6-2			X		X			See note 1.
4.6-3			X		X			See note 1.
4.6-4			X		X			See note 1.
4.7-1			X		X			See note 1.
4.8-1			X		X			No tethers used.

Illustration 1: Orbital Debris Self-Assessment Report Evaluation: HE360 Pathfinder Mission

1. Assessment Report Format

ODAR Technical Sections Format Requirements:

As HawkEye 360, Inc. is a US company, this ODAR follows the format recommended in NASA-STD-8719.14, Appendix A.1 and includes the content indicated at a minimum in each section 2 through 8 below for the satellites in the Pathfinder cluster. Sections 9 through 14 apply to the launch platform, in this case a Falcon 9, and are not covered here.

2. ODAR Section 1: Program Management and Mission Overview

Project Manager: HawkEye 360, Inc.

Foreign government or space agency participation: The satellites will be launched aboard a Falcon 9 rocket launched from Vandenberg AFB in the USA. No foreign government or space agency participation is anticipated.

Schedule of upcoming mission milestones:

Launch: No Earlier Than December 2017

Mission Overview:
The 3 satellites comprising the Pathfinder cluster will be launched into orbit on the Falcon 9 launch vehicle, and will rapidly be deployed from their restraint mechanisms and commissioned. The cluster will then begin payload operations that will continue for at least 2 years.

ODAR Summary: No debris released in normal operations; no credible scenario for breakups; the collision probability with other objects is compliant with NASA standards; and the estimated nominal decay lifetime due to atmospheric drag is well under 25 years following operations (< 7 years, as calculated by DAS 2.0.2).

Launch vehicle and launch site: Falcon 9, Vandenberg AFB

Proposed launch date: No Earlier Than December 2017

Mission duration: Nominal orbit lifetime: 2 years. Maximal orbit lifetime: < 7 years

Launch and deployment profile, including all parking, transfer, and operational orbits with apogee, perigee, and inclination:

The Pathfinder satellites will deploy from a Falcon 9 into an sun-synchronous orbit from which they will naturally decay due to atmospheric drag. The nominal deployment altitude is 575 km.

Nominal Insertion Case:	Apogee: 575 km	Perigee: 575
Inclination:	97 - 98 degrees	
LTDN:	10:30	

The Pathfinder satellites have propulsion for station keeping and cluster formation establishment.

There is no parking or transfer orbit.

3. ODAR Section 2: Spacecraft Description

Physical description of the spacecraft:
The Pathfinder satellites are microsatellites, each with a launch mass of 12.75 kg.

Basic physical dimensions are 200 mm x 267 mm x 440 mm.

The load bearing structure is comprised of two skeleton Magnesium trays, with rails along four corner edges. The solar arrays are body-mounted.

Power storage is provided by 3 prismatic Lithium-Ion cells. The batteries will be recharged by solar cells mounted on the body of the satellite.

Total satellite mass at launch, including all propellants and fluids: 12.75 kg.

Dry mass of satellites at launch, excluding propellant: 12 kg

Description of all propulsion systems (cold gas, mono-propellant, bi-propellant, electric, nuclear): Electrothermal formation-keeping propulsion with H2O working fluid.

Identification, including mass and pressure, of all fluids (liquids and gases) planned to be on board and a description of the fluid loading plan or strategies, excluding fluids in sealed heat pipes: Water and FE36 for pressurization.

Fluids in Pressurized Batteries: None. The satellites use heritage, unpressurized, standard COTS Lithium-Ion battery cells from SAFT.

Description of attitude control system and indication of the normal attitude of the spacecraft with respect to the velocity vector:
Satellite attitude is controlled by magnetorquers and reaction wheels. The nominal attitude is a align/constrain sun-tracking mode where a particular fixed body-frame vector, chosen to maximize power generation, is aligned with the sun, and rotation about the sun vector is constrained to point a second fixed body-frame axis to nadir. Other possible attitude modes include: nadir-pointing, target-tracking, tumble/de-tumble and low/high drag profiles.

Description of any range safety or other pyrotechnic devices: No pyrotechnic devices are used.

Description of the electrical generation and storage system: Standard COTS Lithium-Ion battery cells are charged before payload integration and provide electrical energy during the mission. The cells are recharged by triple-junction GaAs solar cells. A battery protection circuit protects against over and undercharge conditions.

Identification of any other sources of stored energy not noted above: None.

Identification of any radioactive materials on board: None.

4. ODAR Section 3: Assessment of Spacecraft Debris Released during Normal Operation

Identification of any object (>1 mm) expected to be released from the spacecraft any time after launch, including object dimensions, mass, and material: There are no intentional releases.

Rationale/necessity for release of each object: N/A.

Time of release of each object, relative to launch time: N/A.

Release velocity of each object with respect to spacecraft: N/A.

Expected orbital parameters (apogee, perigee, and inclination) of each object after release: N/A.

Calculated orbital lifetime of each object, including time spent in Low Earth Orbit (LEO): N/A.

Assessment of spacecraft compliance with Requirements 4.3-1 and 4.3-2 (per DAS v2.0.2)

4.3-1, Mission Related Debris Passing Through LEO: COMPLIANT

4.3-2, Mission Related Debris Passing Near GEO: COMPLIANT

5. ODAR Section 4: Assessment of Spacecraft Intentional Breakups and Potential for Explosions

Potential causes of spacecraft breakup during deployment and mission operations:
There is no credible scenario that would result in spacecraft breakup during normal deployment and operations.

Summary of failure modes and effects analyses of all credible failure modes which may lead to an accidental explosion:
In-mission failure of a battery cell protection circuit could lead to a short circuit resulting in overheating and a very remote possibility of battery cell explosion. The battery safety systems discussed in the FMEA (see requirement 4.4-1 below) describe the combined faults that must occur for any of seven (7) independent, mutually exclusive failure modes to lead to explosion.

Detailed plan for any designed spacecraft breakup, including explosions and intentional collisions: There are no planned breakups.

List of components which shall be passivated at End of Mission (EOM) including method of passivation and amount which cannot be passivated:
None. The three batteries will not be passivated at End of Mission due to the low risk and low impact of explosive rupturing, and the extremely short lifetime at mission conclusion. The maximum total chemical energy stored in each battery is approximately 92kJ.

Rationale for all items which are required to be passivated, but cannot be due to their design:

The battery charge circuits include overcharge protection to limit the risk of battery failure. However, in the unlikely event that a battery cell does explosively rupture, the small size, mass, and potential energy, of these small batteries is such that while the spacecraft could be expected to vent gases, most debris from the battery rupture should be contained within the vessel due to the lack of penetration energy. This electrical power system has already been flight qualified on the GHGSat-D mission. Further, the battery technology baselined on HawkEye spacecraft has flown on over a dozen UTIAS Space Flight Labs (SFL) spacecraft without failure.

Assessment of spacecraft compliance with Requirements 4.4-1 through 4.4-4:

Requirement 4.4-1: Limiting the risk to other space systems from accidental explosions during deployment and mission operations while in orbit about Earth or the Moon:

For each spacecraft and launch vehicle orbital stage employed for a mission, the program or project shall demonstrate, via failure mode and effects analyses or equivalent analyses, that the integrated probability of explosion for all credible failure modes of each spacecraft and launch vehicle is less than 0.001 (excluding small particle impacts) (Requirement 56449).

Compliance statement:

Required Probability:	0.001.
Expected probability:	0.000.

Supporting Rationale and FMEA details:
Battery explosion:

Effect: All failure modes below might theoretically result in battery explosion with the possibility of orbital debris generation. However, in the unlikely event that a battery cell does explosively rupture, the small size, mass, and potential energy, of the selected COTS batteries is such that while the spacecraft could be expected to vent gases, most debris from the battery rupture should be contained within the vessel due to the lack of penetration energy. Furthermore, each battery has a pressure relief burst disc that prevents catastrophic battery enclosure failure.
Probability: Extremely Low. It is believed to be a much less than 0.1% probability that multiple independent (not common mode) faults must occur for each failure mode to cause the ultimate effect (explosion).

Failure mode 1: Internal short circuit.
Mitigation 1: Qualification and acceptance shock, vibration, thermal cycling, and vacuum tests followed by maximum system rate-limited charge and discharge to prove that no internal short circuit sensitivity exists.
Combined faults required for realized failure: Environmental testing AND functional charge/discharge tests must both be ineffective in discovery of the failure mode.

Failure Mode 2: Internal thermal rise due to high load discharge rate.
Mitigation 2: Cells were tested in lab for high load discharge rates in a variety of flight-like configurations to determine likelihood and impact of an out of control thermal rise in the cell. Cells were also tested in a hot environment to test the upper limit of the cells capability. No failures were seen.
Combined faults required for realized failure: Spacecraft thermal design must be incorrect AND external over-current detection and disconnect function must fail to enable this failure mode.

Failure Mode 3: Excessive discharge rate or short circuit due to external device failure or terminal contact with conductors not at battery voltage levels (due to abrasion or inadequate proximity separation).
Mitigation 4: This failure mode is negated by a) qualification-tested short circuit protection on each external circuit, b) design of battery packs and insulators such that no contact with nearby board traces is possible without being caused by some other mechanical failure, c) obviation of such other mechanical failures by proto-qualification and acceptance environmental tests (shock, vibration, thermal cycling, and thermal-vacuum tests).
Combined faults required for realized failure: An external load must fail/short-circuit AND external over-current detection and disconnect function failure must all occur to enable this failure mode.

Failure Mode 4: Inoperable vents.
Mitigation 5: Battery vents are not inhibited by the battery holder design or the spacecraft.
Combined effects required for realized failure: The final assembler fails to install proper venting.

Failure Mode 5: Crushing.
Mitigation 6: This mode is negated by spacecraft design. There are no moving parts in the proximity of the batteries.
Combined faults required for realized failure: A catastrophic failure must occur in an external system AND the failure must cause a collision sufficient to crush the batteries leading to an internal short circuit AND the satellite must be in a naturally sustained orbit at the time the crushing occurs.

Failure Mode 6: Low level current leakage or short-circuit through battery pack case or due to moisture-based degradation of insulators.
Mitigation 7: These modes are negated by a) battery holder/case design made of non-conductive plastic, and b) operation in vacuum such that no moisture can affect insulators.
Combined faults required for realized failure: Abrasion or piercing failure of circuit board coating or wire insulators AND dislocation of battery packs AND failure of battery terminal insulators AND failure to detect such failure modes in environmental tests must occur to result in this failure mode.

Failure Mode 7: Excess temperatures due to orbital environment and high discharge combined.
Mitigation 8: The spacecraft thermal design will negate this possibility. Thermal rise has been analyzed in combination with space environment temperatures showing that batteries do not exceed normal allowable operating temperatures which are well below temperatures of concern for explosions. This design has been verified through the GHGSat-D and other SFL missions.
Combined faults required for realized failure: Thermal analysis AND thermal design AND mission simulations in thermal-vacuum chamber testing AND over-current monitoring and control must all fail for this failure mode to occur.

Requirement 4.4-2: Design for passivation after completion of mission operations while in orbit about Earth or the Moon:

Design of all spacecraft and launch vehicle orbital stages shall include the ability to deplete all onboard sources of stored energy and disconnect all energy generation sources when they are no longer required for mission operations or postmission disposal or control to a level which cannot cause an explosion or deflagration large enough to release orbital debris or break up the spacecraft (Requirement 56450).

Compliance statement:
The battery charge circuits include overcharge protection to limit the risk of battery failure. However, in the unlikely event that a battery cell does explosively rupture, the small size, mass, and potential energy, of these small batteries is such that while the spacecraft could be expected to vent gases, most debris from the battery rupture should be contained within the vessel due to the lack of penetration energy. As previously mentioned, the integrated burst disc should prevent any explosion altogether.

Requirement 4.4-3. Limiting the long-term risk to other space systems from planned breakups:

Compliance statement:
This requirement is not applicable. There are no planned breakups.

Requirement 4.4-4: Limiting the short-term risk to other space systems from planned breakups:

Compliance statement:
This requirement is not applicable. There are no planned breakups.

6. ODAR Section 5: Assessment of Spacecraft Potential for On-Orbit Collisions

Assessment of spacecraft compliance with Requirements 4.5-1 and 4.5-2 (per DAS v2.0.2, and calculation methods provided in NASA-STD-8719.14, section 4.5.4):

Requirement 4.5-1: Limiting debris generated by collisions with large objects when operating in Earth orbit:
For each spacecraft and launch vehicle orbital stage in or passing through LEO, the program or project shall demonstrate that, during the orbital lifetime of each spacecraft and orbital stage, the probability of accidental collision with space objects larger than 10 cm in diameter is less than 0.001 (Requirement 56506).

Large Object Impact and Debris Generation Probability:
Collision Probability: < 0.00000;
COMPLIANT.

Supporting Deployment and Collision Risk Analysis
The above collision probability is a product of NASA's DAS 2.0.2 software. This analysis was for the entire 3 satellite cluster and the given probability is the sum of the individual collision probabilities of each of the 3 satellites.

Requirement 4.5-2: Limiting debris generated by collisions with small objects when operating in Earth or lunar orbit:
For each spacecraft, the program or project shall demonstrate that, during the mission of the spacecraft, the probability of accidental collision with orbital debris and meteoroids sufficient to prevent compliance with the applicable postmission disposal requirements is less than 0.01 (Requirement 56507).

Pathfinder is to be deployed into a very low Earth orbit. The density of resident space objects, and therefore the probability of collisions, reduces with altitude below about 800km. Therefore the “nominal insertion” scenario (where satellites are deployed at 575km) represents the highest collision probability insertion scenario and we perform the DAS analysis for this case.

Small Object Impact and Debris Generation Probability:

Collision Probability (single satellite):	0.00014;	COMPLIANT.
Collision Probability (complete system):	0.00042;	COMPLIANT

7. Assessment of Spacecraft Post-mission Disposal Plans and Procedures

6.1 Description of spacecraft disposal option selected: The satellite will de-orbit naturally by atmospheric re-entry within 7 years of deployment.

6.2 Plan for any spacecraft maneuvers required to accomplish post-mission disposal:
Rapid atmospheric decay is likely.
The nadir pointing or velocity vector alignment requirements determine the ballistic coefficient up until the perigee altitude is approximately 200km. After this point, the satellites may be allowed to tumble, and assuming minimum drag area reentry will occur within one week from this altitude.

6.3 Calculation of area-to-mass ratio after post-mission disposal, if the controlled reentry option is not selected:

Spacecraft Mass: 12.75 kg

Cross-sectional Area:

Maximum Drag Area: 0.157 m^2 (drag area)
Average Drag Area: 0.130 m^2 (drag area)
Minimum Drag Area: 0.054 m^2 (drag area)

Area to mass ratio:

Maximum Drag Area: 0.0131 m^2/kg
Average Drag Area: 0.0108 m^2/kg
Minimum Drag Area: 0.0045 m^2/kg

6.4 Assessment of spacecraft compliance with Requirements 4.6-1 through 4.6-5 (per DAS v 2.0.2 and NASA-STD-8719.14 section):
Requirement 4.6-1: Disposal for space structures passing through LEO:
A spacecraft or orbital stage with a perigee altitude below 2000 km shall be disposed of by one of three methods:
(Requirement 56557)

a) Atmospheric reentry option:

- Leave the space structure in an orbit in which natural forces will lead to atmospheric reentry within 25 years after the completion of mission but no more than 30 years after launch; or
- Maneuver the space structure into a controlled de-orbit trajectory as soon as practical after completion of mission.

a) Storage orbit option: Maneuver the space structure into an orbit with perigee altitude greater than 2000 km and apogee less than GEO - 500 km.

8. Direct retrieval: Retrieve the space structure and remove it from orbit within 10 years after completion of mission

Satellite Name(s)	**Hawk-1, Hawk-2, Hawk-3**
Nominal Orbit	575 x 575 km
Min Lifetime *	2 years
Max Lifetime	7 years
Post-ops life	4 – 5 years

* Min and Max lifetimes take into account variation of operational modes and space weather uncertainty to bound the orbit lifetime

DAS Analysis: The Pathfinder satellites' satellite reentry is COMPLIANT using method "a".

Requirement 4.6-2. Disposal for space structures near GEO.
Analysis: Not applicable.

Requirement 4.6-3. Disposal for space structures between LEO and GEO.
Analysis: Not applicable.

Requirement 4.6-4. Reliability of Postmission Disposal Operations
Analysis: Not applicable.

8. ODAR Section 7: Assessment of Spacecraft Post-mission Dis Assessment of Spacecraft Reentry Hazards

Assessment of spacecraft compliance with Requirement 4.7-1:

Requirement 4.7-1: Limit the risk of human casualty:
The potential for human casualty is assumed for any object with an impacting kinetic energy in excess of 15 joules:
a) For uncontrolled reentry, the risk of human casualty from surviving debris shall not exceed 0.0001 (1:10,000) (Requirement 56626).

9. ODAR Section 8: Assessment of Spacecraft Reentry for Tether Missions

Analysis performed using DAS v2.0.2 shows that no part of the satellite is expected to survive reentry, and therefore that the risk of human casualty is ~ 0.

Requirements 4.7-1b, and 4.7-1c below are non-applicable requirements because the Pathfinder satellites do not use controlled reentry.

4.7-1, b) **NOT APPLICABLE**. For controlled reentry, the selected trajectory shall ensure that no surviving debris impact with a kinetic energy greater than 15 joules is closer than 370 km from foreign landmasses, or is within 50 km from the continental U.S., territories of the U.S., and the permanent ice pack of Antarctica (Requirement 56627).

4.7-1 c) **NOT APPLICABLE**. For controlled reentries, the product of the probability of failure of the reentry burn (from Requirement 4.6-4.b) and the risk of human casualty assuming uncontrolled reentry shall not exceed 0.0001 (1:10,000) (Requirement 56628).

10. ODAR Section 9: Assessment for Tether Missions

Not applicable. There are no tethers in the Pathfinder mission.

END of ODAR for Pathfinder