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THEMIS

THEMIS
Orbital Debris Assessment
Preliminary Design Review (PDR) Deliverable

SAI-RPT-0555

Revision —

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EXECUTIVE SUMMARY

This report documents the compliance of the Time History of Events and Macroscale Interactions during Substorms (THEMIS) Project with the PDR deliverable guidelines of NASA Safety Standard 1740.14, 'Guidelines and Assessment Procedures for Limiting Orbital Debris'. The project has just completed its Mission PDR as of November 14, 2003. A summary of the current Probe compliance status is provided in Table ES-1, and LV compliance is provided in Table ES-2.

Table ES-1 Probe Orbital Debris Guideline Compliance Status

Guideline Number	Description	Met	Not Met	Not Applicable	Impact/ Issue
3-1	Operational Debris - LEO	X			None
3-2	Operational Debris - GEO	X			None
4-1	Accidental Explosion During Mission	Probably			None
4-2	Accidental Explosion After Mission	Probably			Battery depletion
4-3	Intentional Breakup- Long-term Risk			X	
4-4	Intentional Breakup- Short-term Risk			X	
4-5	Intentional Breakup- During Reentry			X	
5-1	Collision with Large Objects	Probably			None
5-2	Collision with Small Objects	Possibly			Shielding
6-1	Disposal – LEO	Possibly			Lunar phasing
6-2	Disposal – Above LEO			X	
6-3	Disposal - 12 Hour Orbits			X	
6-4	Disposal Reliability		X		Single String design
7-1	Reentry Survivability	Probably			None

Table ES-2 Launch Vehicle Orbital Debris Guideline Compliance Matrix

Guideline	Compliance	Possible mitigation strategies
3-1a	Requires modification	▪ Launch date selection ▪ Perigee reduction
3-1b	Compliant	
4-1	Non-Compliant	Requires new hardware
4-2	Compliant	
5-1	Requires modification	▪ Launch date selection ▪ Perigee reduction ▪ Depletion burn
6-1	Requires modification	▪ Launch date selection ▪ Perigee reduction ▪ Depletion burn
7-1	Compliant (not 2nd stage)	Requires new hardware

1.0 THEMIS PROGRAM BACKGROUND and PROGRAM MANAGEMENT

This report documents the compliance of the Time History of Events and Macroscale Interactions during Substorms (THEMIS) Project with the requirements of NASA Policy Directive 8710.3A 'NASA Policy for Limiting Orbital Debris Generation', and NASA Safety Standard 1740.14, 'Guidelines and Assessment Procedures for Limiting Orbital Debris'. THEMIS is manifested for a dedicated launch on a Delta 7920. The second and third stage assessment was performed by the Kennedy Space Center Expendable Launch Vehicle Office and has been incorporated into this document, per Reference 8 in Section 1.5. This particular report is in support of the Mission PDR deliverable. All data in this report represents best available information, but is subject to change as the mission matures.

1.1 PROJECT OBJECTIVES

The THEMIS Mission is the fourth mission in NASA's Medium-class Explorer (MIDEX) program, and was announced on March 20, 2003. THEMIS is a five-satellite mission with the job of determining the causes of the global reconfigurations of the Earth's magnetosphere that are evidenced in auroral activity. THEMIS consists of five small satellites, carrying identical suites of electric, magnetic, and particle detectors, that will be put in carefully coordinated orbits. Every four days the satellites will line up along the Earth's magnetic tail, allowing them to track disturbances. The satellite data will be combined with observations of the aurora from a network of observatories across the Arctic Circle.

Dr. Vassilis Angelopoulos of the University of California, Berkeley, Calif., (UCB) is the Principal Investigator for THEMIS. NASA's Goddard Space Flight Center, Greenbelt, Md., manages the Explorer Program for the Office of Space Science, Washington. UCB performs all mission management functions, mission and ground operations, as well as provides all Instruments. Swales Aerospace provides the five Probe Busses, as well as the Probe Carrier, and performs the Probe integration and mission integration functions.

1.2 MISSION DESCRIPTION

THEMIS will be launched on a Delta II 2925-10, from CCAS, with a daily launch window of 40 minutes, into a transfer orbit of 1.1 x 12 Re near the ecliptic plane. The five THEMIS probes will deploy themselves from the Probe Carrier, and then use on-board hydrazine propulsion to ascend to the following approximate ecliptic plane science orbits: 1.5x30.9 Re for P1; 1.2x19.8 Re for P2; 1.2x12.1 Re for P3 and P4; and 1.1x9.8 Re, for P5. P5 will be raised to 1.1x13.2 Re orbit after the first tail season. The Probe Carrier (a simple mechanical fixture) stays attached to the Launch Vehicle 3rd stage in the original transfer orbit. The mass of each Probe will be less than 134 kg, and the mass of the Probe Carrier (not including the Third Stage) will be less than 137 kg.

The duration of the mission is two years, which includes sufficient time to capture data from two full magnetic tail seasons. During this time there are two crucial alignment periods each year that are the crucial observation period. The first day (called the "Wedding Day") of the first period is February 21, 2007, which is when all the probes must line up in the night side tail, and lasts until April 22. The first day of the second period is August 28, 2007, when the alignment is on the dayside, and lasts until October 27. During the second year, the first day of the first period is March 3, 2008, and lasts until May 2. The first day of the second period is September 3, 2008, and lasts until November 2. Beyond this period, the mission would enter extended operations depending on funding and the health of the Probes. The hardware must meet a two-year life requirement and the each Probe is required to provide enough fuel for the two-year mission, and then perform a depletion burn for subsequent passive re-entry within 25 years. This depletion burn will be timed with the lunar phase to ensure that the final orbits re-enter instead of being raised.

1.3 PROJECT SCHEDULE

The launch of THEMIS is scheduled for August 2006. The Preliminary Design Review (PDR) was held in November 2003. The Critical Design Review (CDR) will be in May 2004, and the first Probe Pre-Environmental Review in May 2005.

1.4 PROJECT MANAGEMENT

The THEMIS project manager is Mr. Peter Harvey at the UCB. The Probe and Probe Carrier manager at Swales Aerospace is Michael Cully.

1.5 REFERENCES

The following documents are referenced within the body of this document:

1. NASA Safety Standard 1740.14 'Guidelines and Assessment Procedures for Limiting Orbital Debris'
2. Draft NASA Standard 8719.14 Guidelines and Assessment Procedures for Limiting Orbital Debris, September 2003
3. KSC/ELV, MER Delta II Final EIS Databook, Vol. II, Chp. 8, October 2001
4. NASA Policy Directive 8710.3A 'NASA Policy for Limiting Orbital Debris Generation'
5. Pichardo, Eugenio, "Launch Vehicle Portion of the Orbital Debris report for the Dawn Mission (PDR)", ELVL-2003-0031604, 17 November 2003
6. Carney, M., "Delta II Delta II 2nd stage Casualty Area", NASA KSC Memo ELVL-2001-0025355, 19 July 2001
7. Johnson, N., "Reentry Risk Calculations", e-mail and spreadsheet sent to B. Beaver, NASA KSC, 7 October 2003
8. Benson, William, "Launch Vehicle Portion of the Orbital Debris report for the THEMIS Mission (PDR)", NASA KSC Memo ELVL-2003-TBD, to be published.

2.0 THEMIS MISSION DESIGN AND OPERATION FACTORS

2.1 THEMIS HARDWARE

2.1.1 PROBE

The Probe primary structure is of all composite/aluminum honeycomb construction, in the shape of a cube, with dimensions of 85.4 cm (33.616") across flats, and 47.9 cm (18.84") high. The lower deck is the primary mounting surface for most of the instruments and probe components, and interfaces directly to the Probe Carrier separation ring. The four side panels hold the body mounted solar arrays, and attach to the upper and lower decks, as well as the four corner panels. The four corner panels serve as instrument mounts and connect to the two adjacent solar panels on either side. A central tube support the upper and lower axial antennas, as well as the mast that supports the S-Band antenna. The central tube extends 51.7 cm (20.36") above the upper deck of the Probe. See Figure 2-1 for clarification.

The lower deck, upper deck, and the solar array panels are honeycomb panels, with aluminum core and graphite composite facesheets (M55J/RS-3). The corner panels are just M55J/RS-3 with no core, and are 2 mm (0.08") thick. The lower deck is 39mm (1.5") thick, with facesheet thickness of 0.4 mm (0.016"), and 3.2 mm (0.125") cell size core. The top deck is 13mm (0.5") thick, with facesheet thickness of 0.2 mm (0.008"), and 3.2 mm (0.125") cell size core. The side panels are 13mm (0.5") thick, with facesheet thickness of 0.2 mm (0.008"), and 4.8 mm (0.1875") cell size core. The central tube is constructed of graphite composite, with a 0.5mm (0.02") thick wall.

The majority of the Probe electronics boxes and instruments are mounted to the lower deck, with a few exceptions. One of the instruments (the SST) is mounted to a corner panel along with the sun sensor. Two magnetometers are mounted on the booms attached to the top deck. The axial booms are mounted inside the central tube and deploy out the top and bottom. Attached to the lower deck in the center of each solar panel are the four EFI instruments. The THEMIS website (http://sprg.ssl.berkeley.edu/themis) can be accessed for additional information. See Figure 2-2 for clarification.

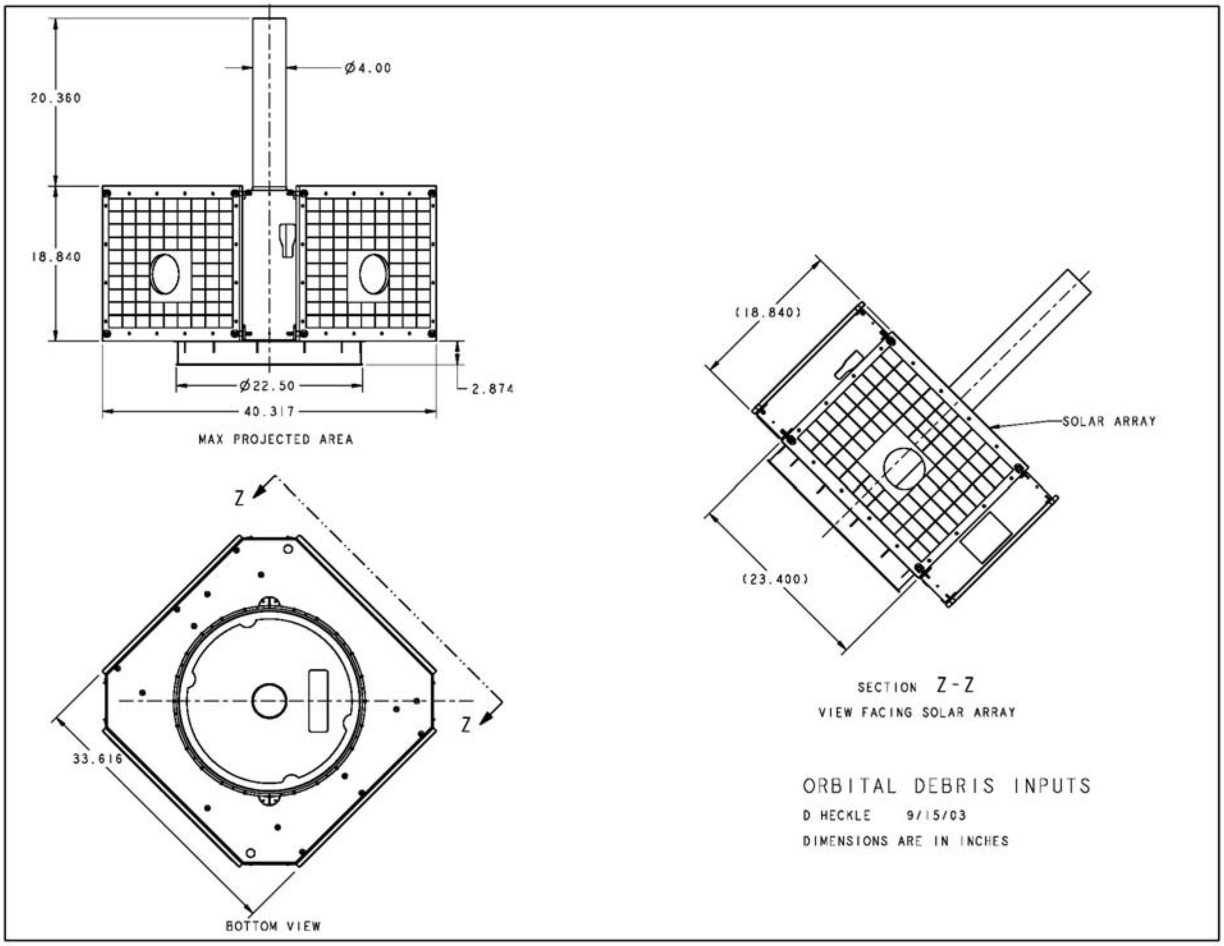

Figure 2-1 THEMIS Probe Structure

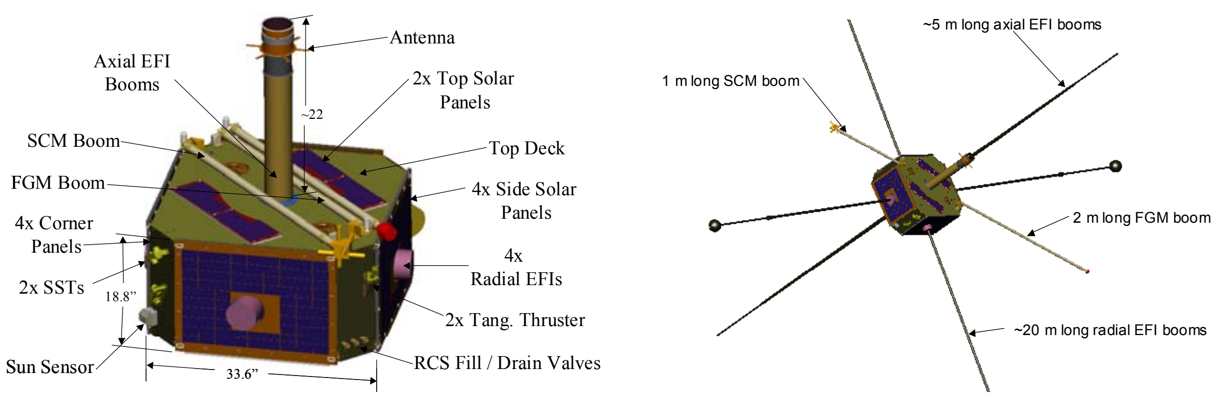

Figure 2-2 THEMIS Probe Stowed & Deployed

The critical Probe control items for maintaining attitude control for disposal purposes are: the Bus Avionics Unit (BAU) box, the Battery, the S-band Transponder, the Sun Sensor, Instrument Data Processing Unit (IDPU), the Flux Gate Magnetometer, and the two (2) Propulsion tanks and lines. The propulsion tanks are Arde Inconel tanks, mounted to the Lower deck, and located in two opposing corners of the spacecraft. This pressurized Hydrazine tank uses Nitrogen as a pressurant. Fuel is transported to the four thrusters through fuel lines on the interior of the lower deck. There are no range safety systems on board the spacecraft.

2.1.2 PROBE CARRIER

The Probe Carrier primary structure consists of the: PAF adapter ring/tube (interface to Delta 3rd stage); center spool (supports top probe); and the main deck (supports the center spool and the lower probes), as shown in Figure 2-3. The PAF adapter ring is machined aluminum with the launch vehicle 3712 standard interface at one end and a simple flange interface to the main deck. The main deck is an assembly of five (5) machined aluminum plates. The deck provides the mounting interface for the separation system of four (4) of the probes, and the attachment for the upper probe center spool. Aluminum struts directly stiffen the outer edges and inner center spool mounting ring of the main deck by connecting them down to the PAF adapter ring. The center spool is an assembly of three simple aluminum machinings, two conical; one cylindrical. The upper conical section interfaces to the bottom ring of the upper probe separation system.

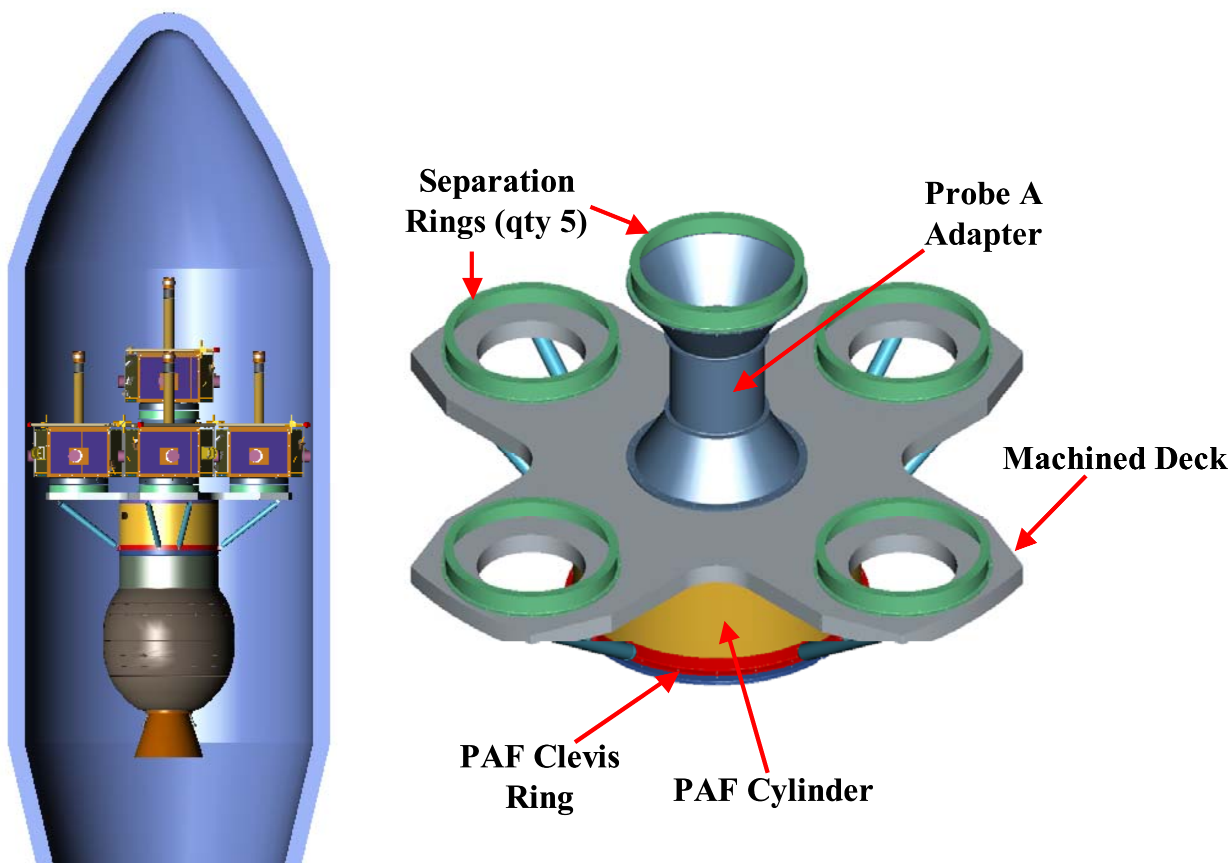

Figure 2-3 THEMIS Probe Carrier With and Without Probes

The PC with dimensions is shown in Figure 2-4. The PAF adapter ring meets the requirements of the Payload Planners Guide for a 3712C PAF, and attaches to the center tube. The center tube is a rolled and rivet cylinder with a 2mm (0.08") wall thickness. The center deck and four outer deck panels at the top of the tube are 70 mm (2.75") thick, with nominal rib and skin thicknesses of 1.5 mm (0.06"), not including interface bosses and blocks. The Inner struts (inside the center tube) are 460 mm (18.1") long, with an OD of 43 mm (1.7"), and a wall thickness of 3.2 mm (0.125"). The Outer struts are 721 mm (28.4") long, with an OD of 57 mm (2.25"), and a wall thickness of 3.2 mm (0.125"). The center spool is built in three sections, with thicknesses of 2.54 mm (0.1"), 2.54 mm (0.1"), and 2 mm (0.08") from bottom to top.

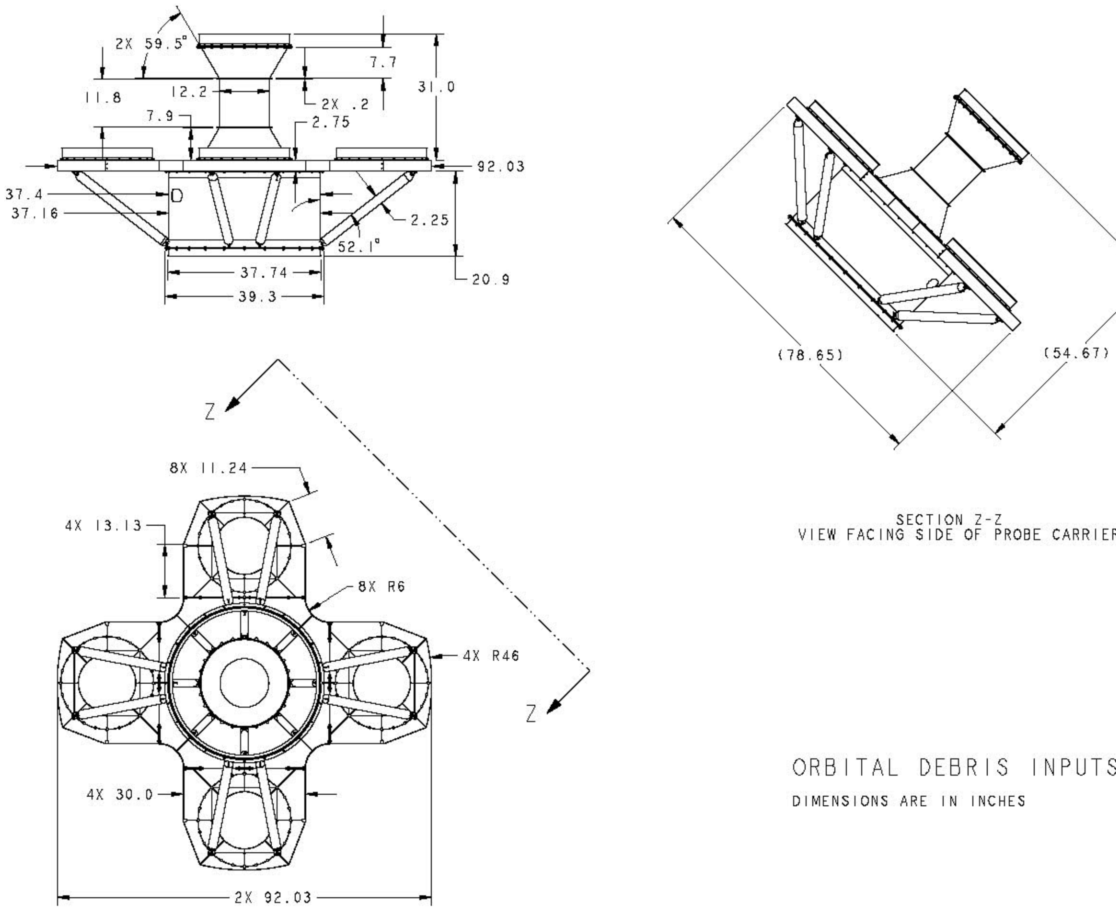

Figure 2-4 THEMIS Probe Carrier Structure

2.1.3 LAUNCH VEHICLE

The THEMIS mission will utilize a Delta 7925 launch vehicle with a 10 foot payload fairing enclosing the Delta II 2nd stage, 3rd stage kick motor and payload during the booster stage flight and the early portion of the Delta II 2nd stage flight. This vehicle consists of a booster stage with a Rocketdyne RS-27A main engine augmented by nine Alliant Graphite Epoxy solid propellant Motors (GEMs), an 2nd stage with an Aerojet AJ10-118K engine, and a Thiokol 3rd stage solid motor stage. The 3rd stage utilizes a 3712C payload attach fitting and a yo-yo despin system.

There are no non-standard stages for this mission. Both the Delta II 2nd stage and the 3rd stage are left in Earth orbit (see Figure 2-5 and Figure 2-6 respectively). After fuel depletion and final venting of the nitrogen and helium tanks, the Delta II 2nd stage will have a dry mass of about 1002 kg. This mass includes the spintable for spin-up of the 3rd stage. After solid rocket motor burnout and probe separation, the 3rd stage and probe carrier will have a combined mass of approximately 258 kg.

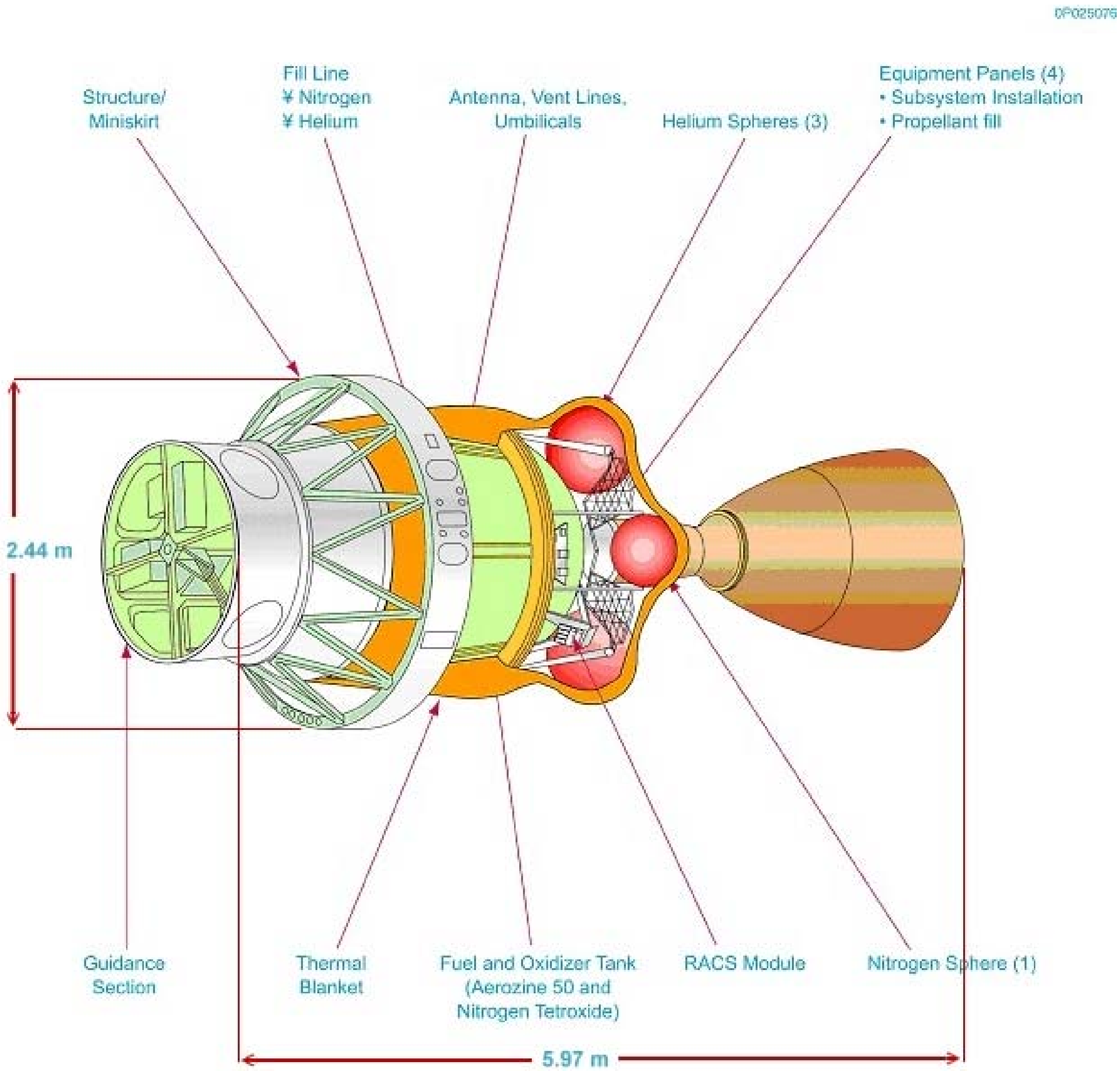

Figure 2-5 Illustration of Delta II 2nd stage

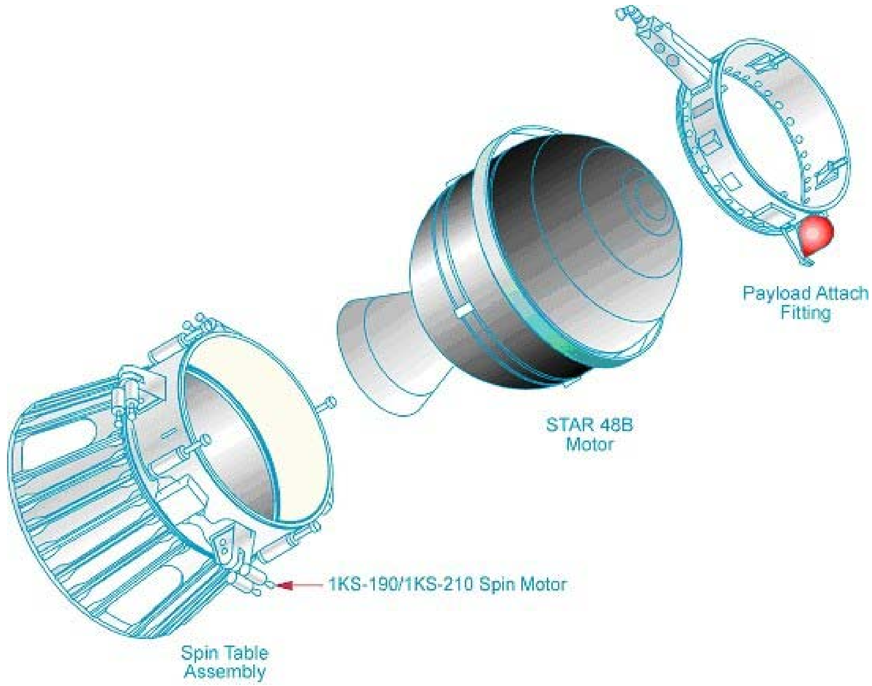

Figure 2-6 Illustration of Delta II Delta II 2nd stage

2.2 THEMIS MISSION PARAMETERS

2.2.1 INITIAL ORBITS

THEMIS will be launched on August 21, 2006 on a Delta II 2925-10, from CCAS, with a daily launch window of 40 minutes. The THEMIS mission will use a three-burn 2nd stage ascent profile to boost a payload consisting of 5 individual probes and a probe carrier into a highly elliptical Earth orbit. After booster stage burn and the initial Delta II 2nd stage burn, the vehicle and spacecraft achieve a circular orbit with a near 100 n.m. altitude and 28.3 degrees inclination. After park orbit coast, a second Delta II 2nd stage burn is conducted to achieve a 185 km perigee altitude and a 637 km apogee altitude. After a second coast period, a third and final Delta II 2nd stage burn is conducted. After the third Delta II 2nd stage burn has been completed, spin-up and 3rd stage separation occurs. The 3rd stage then boosts the payload into its final orbit, which has a perigee of 637 km and an apogee of 70670 km at a 9.0 degree inclination. Launch time and the first Delta II 2nd stage restart for each day of the launch period are designed to achieve the RAAN and argument of perigee targets of 314 and 0.0 degrees respectively. Therefore, the 3rd stage will be left in Earth orbit. Upon separation from the 3rd stage, the Delta II 2nd stage has a 623 km perigee altitude and 2205 km apogee altitude at an inclination of 22.1 degrees.

The five THEMIS probes will deploy themselves from the Probe Carrier, and then use on-board hydrazine propulsion to place themselves in their initial orbits:

Table 2-1 Initial Orbits

Probe	Ra [Re]	Rp [Re]	Inc [deg]	Date
P1	11.725	1.100	7.000	August 21, 2006
P2	11.725	1.100	4.000	August 21, 2006
P3	12.019	1.200	9.000	August 21, 2006
P4	12.019	1.200	9.000	August 21, 2006
P5	12.478	1.350	9.000	August 21, 2006

2.2.2 MISSION ORBITS

The initial orbits for P3 and P4 do not require further adjusting to get ready for Wedding Day, but probes P1, P2, and P5 all do. The approximate orbits for the first night side Wedding Day are achieved by December 23rd, as shown below:

Table 2-2 P1, P2, & P5 Large Orbit Adjustments

Probe	Ra [Re]	Rp [Re]	Inc [deg]	Date
P1	30.943	1.500	10.858	December 23, 2006
P2	19.756	1.168	8.929	December 23, 2006
P5	10.042	1.350	9.000	December 23, 2006

Due to lunar effects, Probes P1 and P2 will require adjustments every month or so to fine-tune their orbits prior to Wedding Day. The final orbits for P1 and P2 for the first wedding day are:

Table 2-3 P1 & P2 Orbits, First Night Side Wedding Day

Probe	Ra [Re]	Rp [Re]	Inc [deg]	Date
P1	31.645	1.500	7.000	January 28, 2007
P2	19.770	1.168	7.000	January 28, 2007

Due to the highly elliptical nature of these initial orbits, especially for P1 and P2, the orbits are constantly being changed depending on the lunar phasing. Orbit maneuvers to adjust lunar phase and maintain the science orbits have been budgeted in the fuel calculations.

In preparation for the first dayside Wedding Day, the orbits are moved from where they have drifted to:

Table 2-4 First Day Side Wedding Day

Probe	Ra [Re]	Rp [Re]	Inc [deg]	Date
P1	31.412	1.500	7.000	June 29, 2007
P2	19.858	1.168	7.000	June 29, 2007
P3	12.019	1.750	4.000	April 22, 2007
P4	12.019	1.750	4.000	April 22, 2007
P5	13.100	1.350	9.000	April 22, 2007

In preparation the second night side Wedding Day, the orbits are moved to:

Table 2-5 Second Night Side Wedding Day

Probe	Ra [Re]	Rp [Re]	Inc [deg]	Date
P1	30.943	1.500	7.000	January 3, 2008
P2	19.756	1.168	7.000	January 3, 2008
P3	11.470	1.750	4.000	January 3, 2008
P4	11.470	1.750	4.000	January 3, 2008
P5	11.869	1.350	9.000	January 3, 2008

In preparation for the second dayside Wedding Day, the orbits are moved to:

Table 2-6 Second Day Side Wedding Day

Probe	Ra [Re]	Rp [Re]	Inc [deg]	Date
P1	31.412	1.500	7.000	July 5, 2008
P2	19.858	1.168	7.000	July 5, 2008
P3	11.470	1.750	4.000	May 2, 2008
P4	11.470	1.750	4.000	May 2, 2008
P5	10.871	1.350	9.000	May 2, 2008

2.2.3 FINAL ORBITS

Following the end of the second dayside period, each probe will perform a depletion burn on or about November 2, 2008, for subsequent passive re-entry within 25 years. This depletion burn will be timed with the lunar phase to ensure that the final orbits re-enter instead of being raised. The orbits following the depletion burns are:

Table 2-7 Final Orbits, Following Depletion Burns

Probe	Ra [Re]	Rp [Re]	Inc [deg]	Date
P1	31.877	1.500	7.000	November 2, 2008
P2	19.858	1.168	7.000	November 2, 2008
P3	12.146	1.100	4.000	November 2, 2008
P4	12.146	1.100	4.000	November 2, 2008
P5	12.146	1.100	9.000	November 2, 2008

2.2.4 ATTITUDES

The attitude of each probe during the mission is defined by the Probe Z axis relative to the spin axis, and in turn the spin axis relative to the ecliptic plane. The requirement is for the spin axis to be tilted 10 ± 3 degrees relative to the ecliptic plane normal, and for the Probe Z axis to be within 5.6 degrees of the probe spin axis. Occasionally, the spacecraft will slew 90 deg to perform orbit maneuvers, but upon completion of the maneuver will return to the default attitude.

3.0 ASSESSMENT of DEBRIS GENERATION DURING NORMAL OPERATIONS

3.1 PROBE

THEMIS probe operation scenarios preclude debris generation during all phases except for separation from the PC. All other deployments are purely mechanical, and result in no debris. Each Probe is separated from the PC by firing a pyrotechnic bolt to release a clamp-band. Tests show that the pyrotechnic bolt cutter devices do not release any debris larger than 1 mm, and a debris booty will be used to contain all of those debris. Therefore, the probe pyrotechnic devices meet both guidelines 3-1 and 3-2 by virtue of all debris being less than the minimum sizes of 1mm and 5mm respectively.

3.2 LAUNCH VEHICLE

By design, no debris is released during normal Delta II 2nd stage operations. All stage separation mechanism hardware is retained or caught. However, the 3rd stage yo-yo weights used to de-spin the spacecraft are jettisoned prior to payload deployment to dump excess angular momentum. It was assumed that each of the yo-yo weights would consist of a cylindrical mass of 1.29 kg attached to a 14.6 meter long cable. These assumptions are based on data from the MER-B mission and will change as the mission matures. The yo-yo weights will be in an orbit with a perigee altitude of 637 km and an apogee altitude of 70670 km. Due to the highly elliptical nature of the orbit, the yo-yo weight orbital lifetime is highly dependent on lunar and solar perturbations. A RAAN sweep is being conducted to bound the range of possible orbital lifetimes for the given perigee. The next study that is planned is to evaluate changing the perigee altitude from 637 km to one that meets the orbital lifetime guideline. The yo-yo weight cable length always meets guideline 3-1 so long as orbital lifetime is below 25 years as it is less then 40 meters in length (1/25 km).

4.0 ASSESSMENT of DEBRIS GENERATED by EXPLOSIONS and INTENTIONAL BREAKUPS

4.1 ACCIDENTAL EXPLOSIONS

4.1.1 PROBE

Accidental explosion of any THEMIS Probe components during the operational mission or after completion of the mission is highly unlikely. Three items were considered in this assessment: the battery and the two propulsion tanks. The risk of accidental explosion during the mission will be calculated as the design matures, and will include the probability of impact with small debris (see Section 5.2).

The battery is baselined to be a 10.5 A-hr, 28 V (8 cells @ 3.6V/cell) Yardney Lithium-Ion battery, but could also be an AEAT battery since there is no procurement contract at this time. This cells used in the Yardney are identical to the cells used in the fully qualified Mars Exploration Rover battery. Risk of explosion exists for this battery if it is severely overcharged, which leads to thermal runaway. Yardney / Lithion has presented test results which showed that a 4.7 Volt overcharge of a 7 Ah Lithium-ion cell (nominal 3.6V) causes the cathodes to undergo an exothermic reaction with the electrolyte which can produce enough internal pressure to open the burst disk and cause fire and/or explosion. Nominally, the battery is not pressurized, just sealed to contain the electrolyte.

This will be mitigated during the operational mission through active charging control internal to the batteries. Once a vendor is chosen, the possibility of a battery control electronics failure will be explored. Possible mitigation measures that may be implemented are a battery cell or pack mounted thermal or pressure fuses (causing an “open” circuit as thermal runaway begins).

Following the completion of mission operations, the plan is to deplete the energy stored in the battery. The details of this implementation will depend on the battery vendor chosen. The risk of accidental explosion due to small debris is unknown at this time, but is probably less than with some other type of batteries since these cells aren’t pressure vessels. If there is no risk of explosion due to penetration, then one solution might be to just draw the battery down as much as possible by shunting most of the array strings, and keeping a load on it.

The propulsion tanks are made by Arde (P/N D4899) and are a new part developed for THEMIS based on modifications to two previous designs. Each tank is 36.83 cm (14.34”) in diameter, constructed of Inconel I718, and is held at the top and bottom (polar mounts), with a side mount outlet port. The two tanks are interconnected and do not contain any internal diaphragms. The MEOP is 400 psi, with a Proof pressure of 600 psi, and Design Burst of 800 psi, and has been designed to leak before burst at MEOP.

Following mission completion, THEMIS will be de-orbited until all fuel is expended, thus eliminating all on-board fuel. Since there is no internal diaphragm, and the tanks are interconnected, all fuel and pressurant will vent to space.

4.1.2 PROBE CARRIER

There are no stored energy devices on board the PC, so there is no risk of accidental explosions.

4.1.3 LAUNCH VEHICLE

All possible scenarios for orbital stage breakup are the same as those for orbital stage explosion. Hence this section addresses failure modes which might lead to an accidental explosion in Earth orbit. The analysis of these failure modes has been done for the MER-A and B missions for nuclear safety purposes, and is documented in reference 3. Table 4-1 and Table 4-2 summarize the relevant credible failure modes and the associated probabilities of failure, P_E, *throughout the mission* for the 2nd and 3rd stages respectively. However, the failure probabilities that are relevant to guideline 4-1 must be calculated based only on the period of time that the stages are actually in Earth orbit (not during ascent). This is accomplished through the use of K-factors.

Table 4-1 2nd Stage Failure Probabilities

Failure Event	P_E
Stage 2 Liquid Rocket Engine Catastrophic Failure	3.1500E-04
Stage 2 Purely Structural Failure	1.2000E-06
Stage 2 Oxidizer Tank Failure	2.4000E-06
Stage 2 Fuel Tank Failure	2.4000E-06
Stage 2/3 Failure to Stage	9.0800E-05

Table 4-2 3rd Stage Failure Probabilities

Failure Event	P_E
Star 48B Purely Structural Failure	6.0100E-07
Star 48B Case Rupture	3.5400E-04

A K-factor profile for an event depicts how the probability of event occurrence varies with time. Thus it enables calculation of the total probability of the event over any selected time interval. The K-factor for the 2nd stage is calculates as follows: The K-factor profile is a step function with a value of 3 for the first 10 seconds of each burn and 1 for the remainder of the burn. The step function value is zero between burns or whenever the engine is shut down. A_T is the total area under the step function, P_{PUA} is the probability per unit area for the relevant K-factor profile, and P_E is the probability of occurrence of the relevant event. Therefore,

$$P_{PUA} = P_E/A_T \text{ , a constant value for a given } A_T \text{ or K-profile.}$$

The relevant probability of the failure event, $P_{E(<T)}$, occurring while in Earth orbit is a function of P_{PUA} and the area, $A_{<T}$, under the step function that corresponds to the time that the stage is in Earth orbit.

$$P_{E(<T)} = P_{PUA} \text{ X } A_{<T}$$

For the THEMIS mission with three 2nd stage burns plus a depletion burn,

$$A_T = [3 \text{ X } 10 + (D_{B1} - 10)] + [3 \text{ X } 10 + (D_{B2} - 10)] + [3 \text{ X } 10 + (D_{B3} - 10)] + [3 \text{ X } 10 + (D_{BD} - 10)]$$

For the time interval that the 2nd stage is in Earth orbit, $\mathbf{A_{<T}} = A_T - A_1$. The K-factor calculations for the 2nd stage and the resulting relevant probability are listed in Table 4-3 and Table 4-4

respectively. The total relevant 2nd stage failure probability is 1.6294E-04, which is greater than the maximum acceptable probability of 1.0E-4 listed in guideline 4-1.

Table 4-3 2nd Stage Relevant Failure Probabilities

Failure Event	Probability	P_{PUA}	$A_{<T}$	$P_{E(<T)}$
Stage 2 Liquid Rocket Engine Catastrophic Failure	3.1500E-04	6.0137E-07	207.26	1.2464E-04
Stage 2 Purely Structural Failure	1.2000E-06	2.2910E-09	207.26	4.7482E-07
Stage 2 Oxidizer Tank Failure	2.4000E-06	4.5819E-09	207.26	9.4964E-07
Stage 2 Fuel Tank Failure	2.4000E-06	4.5819E-09	207.26	9.4964E-07
Stage 2/3 Failure to Stage	9.0800E-05	1.7335E-07	207.26	3.5928E-05
Total 2nd Stage Failure Probability	--	--	--	1.6294E-04

Table 4-4 2nd Stage K-factor Calculations

Burn	Total Time	Time in Orbit	A_T	$A_{<T}$
DB1	296.54	0.00	316.54	0.00
DB2	34.64	34.64	54.64	54.64
DB3	100.62	100.62	120.62	120.62
DBD	12.00	12	32.00	32.00
Total	443.80	147.26	523.8	207.26

The 3rd stage K-factor is a step function with a value of 1 throughout the single 3rd stage burn. Since this burn occurs entirely within the Earth's orbit, no adjustments to the 3rd stage P_E's are necessary. Therefore the total 3rd stage failure probability is simply the sum of the individual 3rd stage P_E's and is equal to 4.1180E-04. This also violates the maximum acceptable probability of 1.0E-4 listed in guideline 4-1.

2nd stage stored energy sources include propellant fuel (Aerozine 50) and oxidizer (nitrogen tetroxide), propellant pressurant gas (helium), compressed gas (nitrogen) for settling and Attitude Control System, and the batteries for 2nd stage avionics power. The exact time of engine restart for the depletion burn is driven by visibility from a ground tracking station and the orbit geometry. The 2nd stage will be reoriented and its main engine restarted for the depletion burn after separation from the 3rd stage and completion of the collision avoidance maneuver about one minute later. The bipropellant valve is left open after the depletion burn to vent all of the propellant pressurant gas along with any remaining propellant. The valves on the Attitude Control System (ACS) and settling motors are opened immediately after the depletion burn to vent all of the nitrogen on board. (Depending on the exact time of the depletion burn, exhaustion of all propellants and gases will then be completed between L + 5500 seconds and L + 9500 seconds, the longest Delta II mission duration on record.) Power to the RIFCA guidance computer is left in the "ON" state in order to expend the batteries for total power depletion and final shutdown of the 2nd stage. Final shutdown occurs well after exhaustion of all gases, and within 9740 seconds or less after launch, thus completing 2nd stage passivation. These times will be refined when the exact sequence is completely defined within a year from launch.

There are no designed orbital stage breakups. Provided that the RIFCA guidance computer on the 2nd stage is functional, the RIFCA will issue the applicable commands per the mission timeline sequence in order to execute the fuel depletion burn, stage blowdown for venting of propellants and gases, and battery power depletion. Updated with flight history, the probability of failure of the RIFCA is 4.9 X 10^{-5}. A failed RIFCA could compromise the programmed sequence of engine burns and blowdown of gases and remaining propellants, and in some cases may result in an explosion (with significantly less than 4.9 X 10^{-5} probability). Failure to restart the 2nd stage engine may be caused by failure of the bipropellant valve or by loss of pressure in the pressurant gas or in the propellant tanks. The total probability of failure to restart the 2nd stage engine is 3.6 X 10^{-5}. The propellant in the tanks can only be vented via the bipropellant valve. Should the 2nd stage fail to restart, loss of pressure or the failed bipropellant valve could compromise full venting of pressurant gases and propellants, and in some cases may result in an explosion (with significantly less than 3.6 X 10^{-5} probability).

The 3rd stage stored energy sources are the nitrogen tank and battery used by the Nutation Control System (NCS). NCS blowdown occurs prior to spacecraft separation and the NCS electronics remain active until the battery is exhausted. No other depletion or stage passivation events occur.

The probability of explosion of the 2nd and 3rd stages in Earth orbit exceeds guideline 4-1's threshold value of .0001. To bring this parameter into compliance with the guideline, redesign of the 2nd stage, 3rd stage and related hardware would be required. Fuel depletion and passivation of pressurized and powered systems do comply with guideline 4-2.

4.2 INTENTIONAL BREAKUPS

THEMIS Observatory operations scenarios preclude intentional breakups, as do Delta II operations, thus debris generation due to intentional breakups does not apply.

5.0 ASSESSMENT of DEBRIS GENERATED by ON-ORBIT COLLISIONS

5.1 ASSESSMENT of COLLISIONS with LARGE OBJECTS DURING MISSION OPERATIONS

5.1.1 PROBE

The probability of any one THEMIS probe having a collision with large objects is less than 2.22E-6. This probability of failure meets the guideline specified in document 1740.14 of being less than 0.001, but does not fully take into account the changes in the orbits following the depletion burns. This was accounted for by using the most conservative cross sectional area, as well as the peak cross sectional area flux number for all altitudes below 2000km, as well as roughly doubling the duration of the de-orbit compared to the calculations shown in Section 6. For each probe, the probability was calculated using the formula

$$P = F \times A \times T$$

as shown in Table 5-1. The average cross sectional area, "A", was calculated from Figure 2-1, which shows the primary structure without the deployed booms. The upper left area is 18.840 x 40.317 + 4 x 20.36 + 22.5 x 2.874 = 905.68 in2. The area in the bottom left is 33.616 x 33.616 = 1130.035456 in2. The area on the right is 18.840 x 33.616 + 20.360 x 4 + 22.5 x 2.874 = 779.43 in2. Since the first two areas are the largest, and the upper left is orthogonal to itself, these areas were then used in the following equation to calculate the average cross-sectional area:

$$(A_{max} + A_1 + A_2) / 2 = A_{average}$$

$$(1130 + 906 + 906) / 2 = 1471 \text{ in}^2 * (.0006452 \text{m}^2/\text{in}^2) = 0.949 \text{ m}^2$$

This value is shown in the table at the bottom right next to the "A". The "F" value of 4.5E-6 is the peak cross sectional area flux number for large orbital debris, which was used for all altitudes below 2000 km since it is conservative. The "T" for each probe was determined by calculating the total time each probe spends below 2000 km altitude for the baseline 2 year mission as well as the conservative 10 year de-orbit time (more than twice the anticipated time) described in Section 6.1. This number for each probe is shown in the column titled "Total Time Alt. <2000km, years". It was calculated by first determining the total time each probe is in an orbit with a perigee below 2000 km ("Time In Orbits Per. <2000km, Total yrs"). This was multiplied by the fraction of time within that orbit that the probe spends below 2000 km ("Hrs/orbit <2000km").

Table 5-1 Large Object Probability Calculation

Probe	Orbit Description	Ap. Alt Km	Per. Alt Km	Duration Yr	Time In Orbits Per. <2000km Total yrs	Orbit Calculation	Hrs/orbit <2000km	Total Time Alt. <2000 km years	Large Debris Collision Probability
P1	Initial	68404.05	637.8	0.339726	0.339726027	Use Orbit 1	1.17	0.016561425	7.07E-08
	Adjust	190976.454	3189	0.09863					
	First Night	195453.81	3189	0.416438					
	First Day	193967.736	3189	0.515068					
	Sec. Night	190976.454	3189	0.50411					
	Sec. Day	193967.736	3189	0.328767					
	Deorbit	196933.506	3189	10	10	Use Orbit 2	1.43	0.297927908	1.27E-06
	TOTAL								1.34E-06
P2	Initial	68404.05	637.8	0.339726	0.339726027	Use Orbit 1	1.17	0.016561425	7.07E-08
	Adjust	119625.768	1071.504	0.09863	0.098630137	Use Orbit 2	1.43	0.002938467	1.25E-08
	First Night	119715.06	1071.504	0.416438	0.416438356	Use Orbit 2	1.43	0.012406861	5.30E-08
	First Day	120276.324	1071.504	0.515068	0.515068493	Use Orbit 2	1.43	0.015345328	6.55E-08
	Sec. Night	119625.768	1071.504	0.50411	0.504109589	Use Orbit 2	1.43	0.015018832	6.41E-08
	Sec. Day	120276.324	1071.504	0.328767	0.328767123	Use Orbit 2	1.43	0.00979489	4.18E-08
	Deorbit	120276.324	1071.504	10	10	Orbit 2	1.43	0.297927908	1.27E-06
	TOTAL								1.58E-06
P3	First Night	70279.182	1275.6	0.668493	0.668493151	Use Orbit 1	1.17	0.03258861	1.39E-07
	First Day	70279.182	4783.5	0.70137					
	Sec. Night	66777.66	4783.5	0.328767					
	Sec. Day	66777.66	4783.5	0.50411					
	Deorbit	71089.188	637.8	10	10	Orbit 1	1.17	0.487493552	2.08E-06
	TOTAL								2.22E-06
P4	First Night	70279.182	1275.6	0.668493	0.668493151	Use Orbit 1	1.17	0.03258861	1.39E-07
	First Day	70279.182	4783.5	0.70137					
	Sec. Night	66777.66	4783.5	0.328767					
	Sec. Day	66777.66	4783.5	0.50411					
	Deorbit	71089.188	637.8	10	10	Orbit 1	1.17	0.487493552	2.08E-06
	TOTAL								2.22E-06
P5	Initial	73206.684	2232.3	0.339726					
	First Night	57669.876	2232.3	0.328767					
	First Day	77173.8	2232.3	0.70137					
	Sec. Night	69322.482	2232.3	0.328767					
	Sec. Day	62957.238	2232.3	0.50411					
	Deorbit	71089.188	637.8	10	10	Orbit 1	1.17	0.487493552	2.08E-06
	TOTAL								2.08E-06
							Notes:	F=	4.50E-06
								A=	0.949

5.1.2 LAUNCH VEHICLE

At the end of its 3rd burn, the 2nd stage orbital elements are as summarized in Table 5-2. The resulting orbital lifetime was calculated to be 678.5 years using the JSC Orbital Debris Program Office's Debris Assessment Software (DAS 1.5.3). To meet the 25 year orbital lifetime guideline (6-1), a depletion burn will have to be designed to reduce orbital lifetime, or the perigee altitude must be reduced. Since the launch vehicle contractor, Boeing, has not received Authority to Proceed (ATP), and the spacecraft design has not yet been finalized, it is unknown at this time which option will be pursued in order to meet guideline 6-1. Therefore, compliance with guideline 5-1 for the 2nd stage cannot be calculated at this time. The THEMIS program will address this issue as the mission matures.

Table 5-2 2nd Stage Orbital Elements after 3rd Burn

Apogee Altitude (Km)	2205.08
Perigee Altitude (Km)	623.00
Inclination (deg)	22.11
Argument of Perigee (deg)	344.99
Average Cross Sectional Area (m^2)	13.7787
RAAN (deg)	312.19

The 3rd stage orbital elements after depletion are summarized in Table 5-3. Since DAS cannot calculate the collision probability for orbits that exceed an apogee of 2000 km and it is also not capable of determining orbital lifetime for such a highly eccentric orbit, the resulting orbital lifetime and dwell time are being calculated by the JSC Orbital Debris Program Office. Due to the highly elliptical nature of the orbit, the orbital lifetime is highly dependent on lunar and solar perturbations. A RAAN sweep is being conducted to bound the range of possible orbital lifetimes for the given perigee. The next study that is planned is to evaluate changing the perigee altitude from 637 km to one that meets the orbital lifetime guideline. Once the lifetime is determined, the probability will be calculated for that orbit.

Table 5-3 3rd Stage Orbital Elements after Burnout

Apogee Altitude (Km)	70670.50
Perigee Altitude (Km)	637.66
Inclination (deg)	9.00
Argument of Perigee (deg)	0.00
Average Cross Sectional Area (m^2)	5.177
RAAN (deg)	314.00

5.2 ASSESSMENT of COLLISIONS with SMALL DEBRIS DURING MISSION OPERATIONS

The "Guideline 5-2" methodology described in the document "NASA Safety Standard 1740.14" was used for this calculation, with the details provided in Table 5-4. Several points need to be made about the calculations contained in the table. First, the "Debris Diameter" was calculated by multiplying the "Shield Den(sity)" by a "K" value of 0.35 (which corresponds to a Whipple shield) for all components internal to the bus, with the external components (thrusters and Sun Sensor) using a K value of 0.07 since they are outside the probe body. This method was used on EO1, and was justified since all boxes are internal to the spacecraft, and are thus protected by honeycomb panels and sometimes also MLI. The value of the MLI shielding (equivalent to 0.456 grams/cm^2) was provided during EO-1 by the Orbital Debris Program Office at JSC, but should be confirmed for this mission with JSC, along with the overall approach.

Table 5-4 Small Object Probability Calculation (For Probe P2)

Component	X (cm)	Y (cm)	Area (cm^2)	Critical Direction	Shield Den. (g/cm^2)	K Factor	Debris Dia (cm)	Area Flx Man (#/m^2/yr)	Area Flx Met (#/m^2/yr)	Life yr	Lman Factor	Life yr	Lmet Factor
Propulsion tank 1	18.415	radius	1065.3525	Bottom	0.59100608								
Propulsion tank 1	18.415	radius	1065.3525	Top	0.52350304	0.35	0.18322606	0.004	0.0006	0.00979	3	2	1
Propulsion tank 1	18.415	radius	1065.3525	Side	0.57550304								
Propulsion tank 2	18.415	radius	1065.3525	Top	0.52350304	0.35	0.18322606	0.004	0.0006	0.00979	3	2	1
thrusters	2.54	10.16	25.8064	Top	0.611886	0.07	0.04283202	0.2	0.2	0.0335	3	2	1
Propellant Lines	0.635	304.8	193.548	Top	0.63859664	0.35	0.22350882	0.0033	0.00025	0.0335	3	2	1
Battery	10.16	22.86	232.2576	Top	0.41548304	0.35	0.14541906	0.007	0.002	0.0335	3	2	1
BAU	13.97	21.59	301.6123	Top	1.41750304	0.35	0.49612606	0.0033	0.00025	0.0335	3	2	1
S-band Transponder	13.97	17.78	248.3866	Top	0.41548304	0.35	0.14541906	0.007	0.002	0.0335	3	2	1
Gyro 1 (estimate)	1.9	radius	11.341149	Top	0.33750304	0.35	0.11812606	0.015	0.006	0.0335	3	2	1
Gyro 2 (estimate)	1.9	radius	11.341149	Top	0.33750304	0.35	0.11812606	0.015	0.006	0.0335	3	2	1
Sun Sensor (estimate)	4.5	4.5	20.25	Top	0.33750304	0.07	0.02362521	1	2	0.0335	3	2	1

Total Number of Failures 0.0018 Probability of Failure 0.0018

Shielding Material Thickness in cm												
Part	Lower Deck	Top pnl	Side pnls	Solar Cells	MLI	Thruster	Lines	Bat/S-b Wall	BAU Wall	Gyro Wall	SS Wall	Tank wall
Thick(cm)	0.08128	0.04064	0.04064	0.02	From	0.0762	0.07112	0.127	0.5	0.1	0.1	0.0508
Material	Gr Comp	Gr Comp	Gr Comp	Glass	EO1	Steel	Steel	Al	Al	Al	Al	Inconel
	M55J/RS3	M55J/RS3	M55J/RS3					7075-T7351	6061	6061	6061	I718
Den(g/cm^3)	1.661	1.661	1.661	2.6		8.03	8.03	2.74	2.7	2.7	2.7	8.1931
Shield Den					0.456							
Note: Thickness of decks and panels is total for both facesheets.												

The propellant tank is calculated on three lines to evaluate which direction was the worst. Since the calculation was the worst for the top deck, all other components were evaluated for that direction, assuming that their largest surface area was exposed in that direction. In addition, the duration of exposure to LEO orbital debris was calculated for P2, since it is the only probe exposed to LEO at perigee for the entire two year mission, resulting in a total exposed time of .0335 years. It was assumed that the exposure to the meteoroid environment was continuous for the 2 years, which is the primary cause of the probability of failures.

The result of the calculation is that there is a probability of failure during the nominal 2-year mission life of 0.0018. This probability of failure meets the guideline specified in document 1740.14 of being less than 0.01.

Since the Probe Carrier and Third Stage do not perform any post mission disposal actions, they do not need to meet this guideline.

6.0 ASSESSMENT of POSTMISSION DISPOSAL of SPACE STRUCTURES

6.1 DISPOSAL for FINAL MISSION ORBITS PASSING THROUGH LEO

The THEMIS Probes will be disposed of according to Option 6.1a, Atmospheric Reentry, and meets the guideline of reentry in less than 25 years following mission completion. This analysis was performed initially and documented in the CSR, and has not been redone. The following was taken from Section M6. The Launch Vehicle information was generated by KSC.

6.1.1 PROBE

Final end-of-mission orbital parameters, derived from full propagation of multi-body nominal mission analyses, were subsequently enveloped by successive iterations, to determine lifetimes, using the NASA Assessment Software, Analytical Graphics Inc.'s (AGI) Satellite Tool Kit (STK) Lifetime Tool, Goddard Trajectory Determination Software (GTDS). Once the orbits were decided, the final propagated end-of-mission orbit was used as an input into the re-entry software (either DAS Orbit Evolution Tool or GTDS when deemed more appropriate) to verify that reentry occurs within 25 years.

Debris Assessment Software was used to determine end-of-mission reentry requirements for all major components of the THEMIS mission. The expected masses used for the Probes and the Probe (with Delta 3rd Stage spent properties included) were 78.46 kg and 282 kg respectively. Area-to-mass ratios were 0.0162 and 0.0183 respectively, based on assumed average areas of 5.177 m^2 for the Carrier, and 1.269 m^2 for the Probes. Probes 1 and 2, the apogee altitudes were above what was supported in the Orbit Evolution Tool determining lifetime (the limit is 100,000 km). For these orbits, the GTDS software was most appropriate determine end of life, using Earth oblateness, solar, lunar, and aerodynamic drag perturbations allowing the orbit integrator to run for a given object until impacting the Earth's surface. As evident from Figure 6-1 lunar phase is affecting the orbit decay much more than RAAN or APER. Each probe depletion burn and final orbit maneuvers will be designed to adjust to the lunar phase and lead to decay that meets the guideline, with a goal of re-entry in <10 years (blue curves) for the RAAN and APER elements of THEMIS.

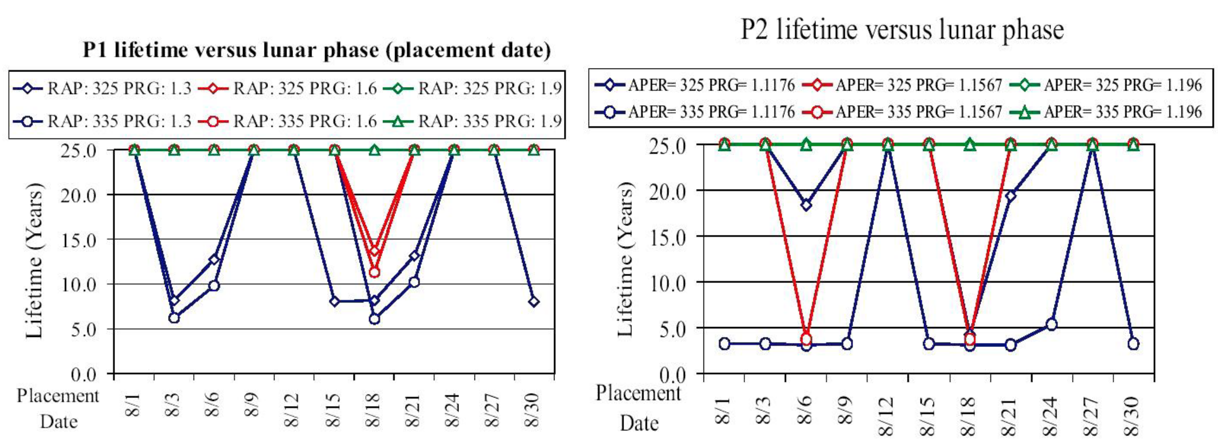

Figure 6-1 Probe P1 and P2 Lifetime versus Lunar Phase

For Probes 3-5, the DAS Orbit Evolution Tool varied right ascension to account for dispersion that might affect lifetime, as shown in Table 6-1. Worst-case variance in lifetime is 19% on Probes 3 and 4. Maximum computed lifetime is for Probes 1 (11.8 years) and 2 (10.8 years), Probe 5 reenters passively within 5 years. The propellant budget includes sufficient baseline propellant to perform the end of mission lifetime maneuvers during the depletion burn.

Table 6-1 Lifetime Calculation for Probes 3, 4, and 5

Object	RAAN (degrees)	Argument of Perigee (Degrees)	R Apogee (Earth radii)	R Perigee (Earth Radii)	Inclination (degrees)	Epoch (EOM)	DAS Lifetime (yrs)	GTDS/ GMAN Lifetime (yrs)
Probe 3	200	0	12.146	1.1	9	8/21/2008	3.04	1.6
	204	0	12.146	1.1	9	8/21/2008	2.96	3.0
	208	0	12.146	1.1	9	8/21/2008	2.71	4.4
	212	0	12.146	1.1	9	8/21/2008	2.71	5.8
	216	0	12.146	1.1	9	8/21/2008	2.63	7.2
	220	0	12.146	1.1	9	8/21/2008	2.55	8.5
Probe 4	200	0	12.146	1.1	9	8/21/2008	3.04	1.6
	204	0	12.146	1.1	9	8/21/2008	2.96	3
	208	0	12.146	1.1	9	8/21/2008	2.71	4.4
	212	0	12.146	1.1	9	8/21/2008	2.71	5.8
	216	0	12.146	1.1	9	8/21/2008	2.63	7.2
	220	0	12.146	1.1	9	8/21/2008	2.55	8.5
Probe 5	200	0	12.146	1.1	4	8/21/2008	3.61	1.6
	204	0	12.146	1.1	4	8/21/2008	3.53	3.0
	208	0	12.146	1.1	4	8/21/2008	3.45	4.4
	212	0	12.146	1.1	4	8/21/2008	3.29	5.8
	216	0	12.146	1.1	4	8/21/2008	3.2	7.2
	220	0	12.146	1.1	4	8/21/2008	3.12	8.5

6.1.2 LAUNCH VEHICLE

The selected option for orbital disposal of the Delta II 2nd stages is a post-mission orbit that will passively lead to atmospheric reentry. At the end of its 3rd burn, the 2nd stage orbital elements are as summarized in Table 5-2. The resulting orbital lifetime was calculated to be 678.5 years using the JSC Orbital Debris Program Office's Debris Assessment Software (DAS 1.5.3). To meet the 25 year orbital lifetime guideline (6-1), a depletion burn will have to be designed to reduce orbital lifetime, or the perigee altitude must be reduced. Since the launch vehicle contractor, Boeing, has not received Authority to Proceed (ATP), and the spacecraft design has not yet been finalized, it is unknown at this time which option will be pursued in order to meet guideline 6-1. The THEMIS program will address this issue as the mission matures.

The 3rd stage orbital elements after depletion are summarized in Table 5-3. Since DAS cannot calculate the orbital lifetime for such a highly eccentric orbit, the resulting orbital lifetime and dwell time are being calculated by the JSC Orbital Debris Program Office. Due to the highly elliptical nature of the orbit, the orbital lifetime is highly dependent on lunar and solar perturbations. A RAAN sweep is being conducted to bound the range of possible orbital lifetimes for the given perigee. The next study that is planned is to evaluate changing the perigee altitude from 637 km to one that meets the orbital lifetime guideline. Therefore, it is intended for the 3rd stage to meet guideline 6-1 by making either a careful selection of launch date, or lowering perigee.

The post fuel depletion Delta II 2nd stage and 3rd stage orbits pass through LEO. Therefore, guidelines 6-2 (for structures near GEO) and 6-3 (for structures between LEO and GEO) are not applicable. If a depletion burn is used to lower the 2nd stage perigee to reduce orbital lifetime, the probability of success for such a maneuver has been shown to be greater than 90% in reference 5, thus satisfying guideline 6-4.

6.2 RELIABILITY of POSTMISSION DISPOSAL OPERATIONS

Initial reliability calculations for the two years mission show a reliability of 0.83 for the probe. As the mission design matures, the FMEA and reliability analysis for the critical components will be performed again, and documented prior to CDR.

7.0 ASSESSMENT of SURVIVAL of DEBRIS from the POST-MISSION DISPOSAL ATMOSPHERIC REENTRY OPTION

7.1 PROBE

This section assesses Guideline 7-1, the potential risk to the Earth's population due to passive reentry, and was performed using the NASA Debris Assessment Software, version 1.5.3. Results of the analysis for a Probe are shown in Table 7-1, and shown to have a debris casualty area of 2.28253 m^2. For the PC/3rd Stage, the results are shown in Table 7-2 and shown to have a debris casualty are of 3.85636 m^2. Both of these meet the guideline of total debris casualty area being less than 8 m^2.

Each Probe is considered a distinct reentry event as their lifetimes vary widely, spanning almost a ten year period. The total dry mass current best estimate is shown in the first line for the parent object to calculate trajectory. The sum of the listed components is 61.801 kg, and the missing mass consists of 5.118kg for RCS lines and small components, 1.737kg of thermal blankets and heaters, and 1.25kg of harness. The primary structure was modeled as flat plates, with Gr/Ep properties since that is the facesheet material, with aluminum honeycomb and small titanium inserts. Almost all the other components were modeled as solid aluminum since that is the primary structural material. One exception is the battery, which probably will have a stainless steel casing on each cell but which cannot be determined at this time since a vendor hasn't been selected. The other exception is the axial booms which are housed in a Gr/Ep tube, but contain other materials.

For the Probe reentry event, only the propellant tanks and the battery were shown to survive. The propellant tanks survive due to the high melting point of their material (Inconel). The lithium ion battery survives because it was modeled as being solid stainless steel, as that material is typically used for the cell cases.

Table 7-1 Debris Casualty Area from Decaying Probes

UNCONTROLLED REENTRY FROM DECAYING ORBITS								
*** Parent Object Data is in line 1 ***								
Total Debris Casualty Area.... 2.28253007 m^2								
Object Surface Identification	Type	Diameter (m)	Length (m)	Height (m)	Mass (kg)	Material Type	Demise Altitude(km)	Casualty Area(m^2)
Parent	Box	0.854	0.854	0.479	70	Al 7075-T735x	77.9982	0
Side Panel 1	Flat Plate	0.854	0.479	0	1.629	Gr/Ep	77.4334	0
Side Panel 2	Flat Plate	0.854	0.479	0	1.629	Gr/Ep	77.4334	0
Side Panel 3	Flat Plate	0.854	0.479	0	1.629	Gr/Ep	77.4334	0
Side Panel 4	Flat Plate	0.854	0.479	0	1.629	Gr/Ep	77.4334	0
Bottom Plate	Flat Plate	0.854	0.854	0	11.55	Gr/Ep	75.2889	0
Top Plate	Flat Plate	0.854	0.854	0	3.691	Gr/Ep	77.1465	0
Sep. Ring	Box	0.854	0.854	0.479	1.715	Al 7075-T735x	77.2621	0
prop tank 1	Sphere	0.3683	0	0	2.45	Inconel 718	0	0.8582
prop tank 1	Sphere	0.3683	0	0	2.45	Inconel 718	0	0.8582
Sun Sensor	Box	0.045	0.045	0.045	0.25	Al 7075-T735x	72.6743	0
Gyros	Box	0.105	0.051	0.013	0.12	Al 7075-T735x	74.7988	0
BAU	Box	0.216	0.14	0.0889	2.05	Al 7075-T735x	67.4879	0

UNCONTROLLED REENTRY FROM DECAYING ORBITS								
*** Parent Object Data is in line 1 ***								
Total Debris Casualty Area.... 2.28253007 m^2								
Object Surface Identification	Type	Diameter (m)	Length (m)	Height (m)	Mass (kg)	Material Type	Demise Altitude(km)	Casualty Area(m^2)
Transponder	Box	0.1872	0.14	0.0607	2.94	Al 7075-T735x	61.7089	0
BERB	Box	0.05	0.05	0.05	0.695	Al 7075-T735x	66.3657	0
battery	Box	0.1016	0.2286	0.1016	2.95	SS 304L	0	0.5661
FGM	Cylinder	0.0634	1.902	0	1.348	Al 7075-T735x	76.3976	0
ESA/IDPU	Box	0.19	0.301	0.245	7.237	Al 7075-T735x	62.4448	0
SST	Box	0.0554	0.107	0.0965	1.28	Al 7075-T735x	67.2343	0
SST	Box	0.0554	0.107	0.0965	1.28	Al 7075-T735x	67.2343	0
SCM	Cylinder	0.0374	1.12	0	1.5	Al 7075-T735x	73.9301	0
EFI Radial 1	Box	0.23	0.128	0.1118	1.955	Al 7075-T735x	68.1436	0
EFI Radial 2	Box	0.23	0.128	0.1118	1.955	Al 7075-T735x	68.1436	0
EFI Radial 3	Box	0.23	0.128	0.1118	1.955	Al 7075-T735x	68.1436	0
EFI Radial 4	Box	0.23	0.128	0.1118	1.955	Al 7075-T735x	68.1436	0
EFI Axial	Cylinder	0.1067	1.0414	0	3.959	Gr/Ep	76.0625	0
Totals:					61.801			2.2825

7.2 LAUNCH VEHICLE

The Delta II 2nd stage has considerable heritage and flight history, and is non-compliant with an estimated casualty risk of .000182 (see reference 6). Compliance with the .0001 guideline would require redesign of the stage.

The JSC Orbital Debris Program Office is currently scheduled to conduct a generic casualty area analysis for the Delta II 3rd stage. Pending completion of that analysis, KSC conducted a preliminary casualty area analysis using the data listed in Table 7-2 and DAS 1.5.3. Since the sum of the major component masses do not account for all the mass in the 3rd stage and probe carrier, the additional, miscellaneous hardware was modeled as 0.1 m diameter aluminum spheres. The only pieces of equipment that survive reentry to Earth impact are the Star 48 rocket motor case assembly and the probe carrier main deck. The values for the Star-48 motor case assembly are believed to be conservative due to the fact that a significant portion of its mass actually consists of other materials besides titanium that are not as likely to survive reentry, such as the rubber liner. The entire case assembly was modeled as such a titanium sphere since the rubber liner is encapsulated within the titanium shell. Even in the presence of these conservative assumptions, the reentry risk was determined to be 9.25746E-5 based on a casualty area of 5.5906 m^2 and a population density of 16.55898/km^2, which meets the guideline value of .0001. The population density is a function of orbital inclination and the year of reentry (2031 to be conservative).

Table 7-2 Casualty Area of the Delta II 3rd Stage with attached Probe Carrier

UNCONTROLLED REENTRY FROM DECAYING ORBITS								
*** Parent Object Data is in line 1 ***								
Total Debris Casualty Area.... ... 5.59066153 m^2								
Object Surface Identification	**Type**	**Diameter (m)**	**Length (m)**	**Height (m)**	**Mass (kg)**	**Material Type**	**Demise Altitude(km)**	**Casualty Area(m^2)**
Parent	Cylinder	1.22	2.5	0	303.6434	Titanium	77.9855	0
Star 48B Case	Cylinder	0.928	1.25	0	86.99	Titanium	0	2.8124
Nozzle Assy	Cylinder	0.769	1.566	0	69.7851	Gr/Ep	73.3346	0
NCS N2H4 Tank	Sphere	0.1091	0	0	0.8006	Al 6061-T6	62.9657	0
PAF adapt ring	Cylinder	0.94	0.658	0	16.56	Al 6061-T6	71.2503	0
3d Stage Compon	Box	0.1	0.1	0.1	4.8081	Al 6061-T6	51.7548	0
3d Stage Compon	Box	0.1	0.1	0.1	4.8081	Al 6061-T6	51.7548	0
3d Stage Compon	Box	0.1	0.1	0.1	4.8081	Al 6061-T6	51.7548	0
3d Stage Compon	Box	0.1	0.1	0.1	4.8081	Al 6061-T6	51.7548	0
3d Stage Compon	Box	0.1	0.1	0.1	4.8081	Al 6061-T6	51.7548	0
PCA Deck Center	Flat Plate	1.0668	1.0668	0	20.673	Al 7075-T735x	0	2.7782
PCA Petal 1	Flat Plate	0.635	0.762	0	11.9567	Al 7075-T735x	62.5743	0
PCA Petal 2	Flat Plate	0.635	0.762	0	11.9567	Al 7075-T735x	62.5743	0
PCA Petal 3	Flat Plate	0.635	0.762	0	11.9567	Al 7075-T735x	62.5743	0
PCA Petal 4	Flat Plate	0.635	0.762	0	11.9567	Al 7075-T735x	62.5743	0
PCA Strut 1	Cylinder	0.057	0.721	0	1.03	Al 7075-T735x	74.7045	0
PCA Strut 2	Cylinder	0.057	0.721	0	1.03	Al 7075-T735x	74.7045	0
PCA Strut 3	Cylinder	0.057	0.721	0	1.03	Al 7075-T735x	74.7045	0
PCA Strut 4	Cylinder	0.057	0.721	0	1.03	Al 7075-T735x	74.7045	0
PCA Strut 5	Cylinder	0.057	0.721	0	1.03	Al 7075-T735x	74.7045	0
PCA Strut 6	Cylinder	0.057	0.721	0	1.03	Al 7075-T735x	74.7045	0
PCA Strut 7	Cylinder	0.057	0.721	0	1.03	Al 7075-T735x	74.7045	0
PCA Strut 8	Cylinder	0.057	0.721	0	1.03	Al 7075-T735x	74.7045	0
PCA Adapt Tube	Cylinder	0.46	0.7874	0	13.9657	Al 7075-T735x	69.8583	0
PCA PAF Cylinder	Cylinder	0.9586	0.5309	0	14.7617	Al 7075-T735x	72.4282	0
Totals:								5.5906

8.0 CONCLUSION

8.1 PROBE

For the Probe, the compliance with the orbital debris guidelines has been divided into three categories as presented below in Table 8-1. Compliant is denoted with an X. "Probably" denotes a high degree of confidence that the final design will comply with the guideline. "Possibly" indicates that more work is needed to determine if the current design complies with the guideline, and indicates that there are issues to be explored to ensure that the final design is in compliance. At this time the only area of non-compliance is the reliability calculation, which is due to the fact that the mission is a single string design. This number will be re-evaluated once the design matures, and all efforts will be made to achieve the highest reliability possible within the MIDEX mission constraints.

Table 8-1 Probe Compliance Status

Guideline Number	Description	Met	Not Met	Not Applicable	Impact/ Issue
3-1	Operational Debris - LEO	X			None
3-2	Operational Debris - GEO	X			None
4-1	Accidental Explosion During Mission	Probably			None
4-2	Accidental Explosion After Mission	Probably			Battery depletion
4-3	Intentional Breakup- Long-term Risk			X	
4-4	Intentional Breakup- Short-term Risk			X	
4-5	Intentional Breakup- During Reentry			X	
5-1	Collision with Large Objects	Probably			None
5-2	Collision with Small Objects	Possibly			Shielding
6-1	Disposal - LEO	Possibly			Lunar phasing
6-2	Disposal - Above LEO			X	
6-3	Disposal - 12 Hour Orbits			X	
6-4	Disposal Reliability		X		
7-1	Reentry Survivability	Probably			None

8.2 DELTA II 3RD STAGE/PROBE CARRIER

Per the preceding discussions, the relevant orbital debris guidelines can be divided into three categories: those where the mission is compliant, those where the mission can be made to be compliant with some modifications and those where the mission will be non-compliant due to hardware limitations. THEMIS meets 3-1b and 4-2 without qualification. 3-1a, 5-1 and 6-1 require some combination of launch time selection to reduce orbital lifetime, reduction in target orbit perigee and design of a 2nd stage depletion burn. 4-1 and 7-1 cannot be met by the THEMIS mission at all. Risk mitigation strategies will be developed and implemented to address these issues as the mission matures.

Table 8-2 Orbital Debris Guideline Compliance Matrix

Guideline	Compliance	Possible mitigation strategies
3-1a	Requires modification	▪ Launch date selection ▪ Perigee reduction
3-1b	Compliant	
4-1	Non-Compliant	Requires new hardware
4-2	Compliant	
5-1	Requires modification	▪ Launch date selection ▪ Perigee reduction ▪ Depletion burn
6-1	Requires modification	▪ Launch date selection ▪ Perigee reduction ▪ Depletion burn
7-1	Compliant (not 2nd stage)	Requires new hardware

APPENDIX A - DEFINITIONS

ABBREVIATION	DEFINITION
ACS	Attitude Control System
EFI	Electric Field Instrument
FDC	Fault Detection and Correction
FGM	Flux Gate Magnetometer
GSFC	Goddard Space Flight Center
GMAN	General MANeuver Program
GTDS	Goddard Trajectory Determination System
MSASS	Multimission Spin Axis Stabilized Spacecraft Attitude Determination System
PAF	Payload Attach Fitting
STK	Satellite Tool Kit
THEMIS	Time History of Events and Macroscale Interactions during Substorms
UCB	University of California, Berkeley