THEO & MUFN: Defending Earth Against the 2023 PDC Hypothetical Asteroid Impact

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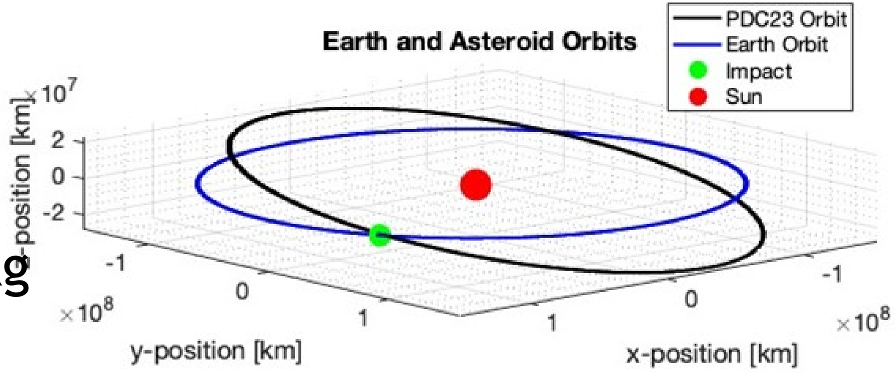
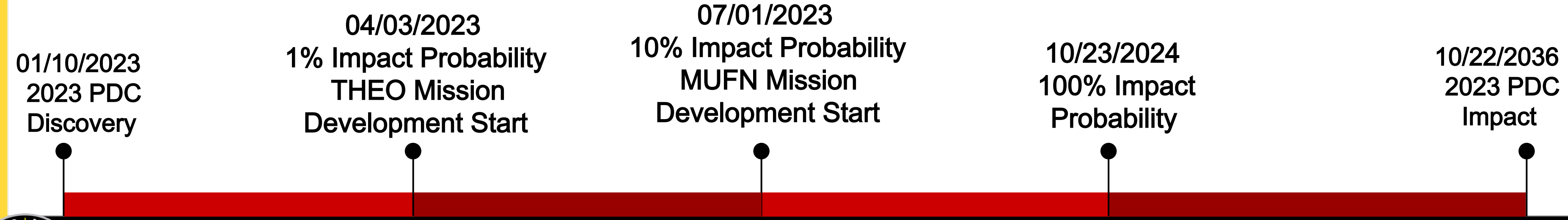
THEO | MUFN
Terrestrial Hazard Exploration Orbiter | Mitigation Using a Fission Nuclear Device

2023 PDC



- Diameter: 295 m - 1119 m
- Mass : 2.42e10 kg - 1.32e10 kg
- Out of phase with Earth
- No close approaches until Earth impact
- Potential to affect over a billion people

  (~20-25% of Earth's population)



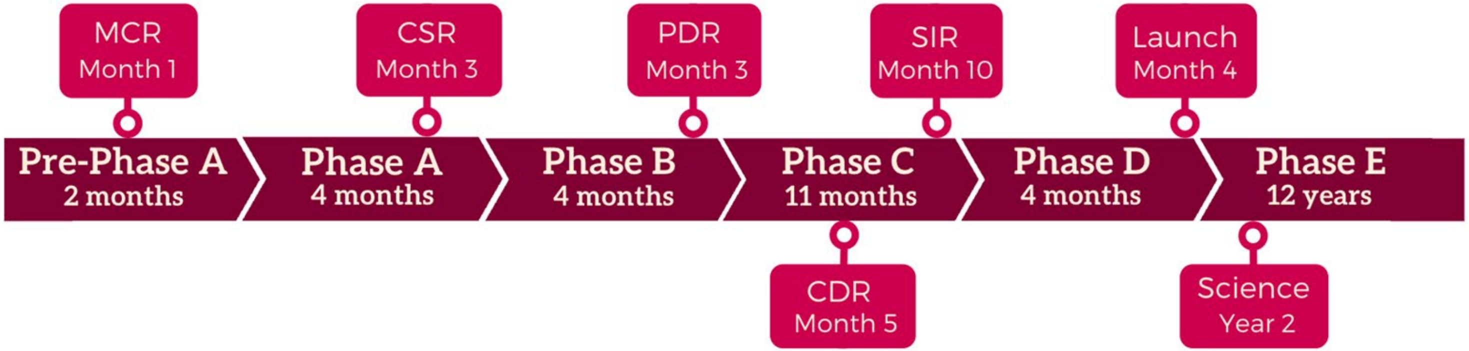
Terrestrial Hazard Exploration Orbiter (THEO)

Reconnaissance Mission

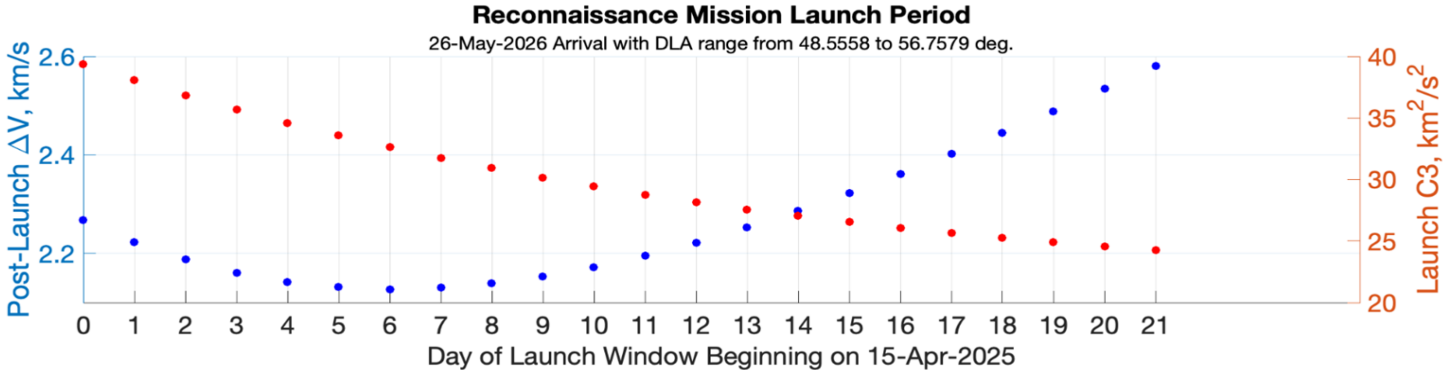
Requirements

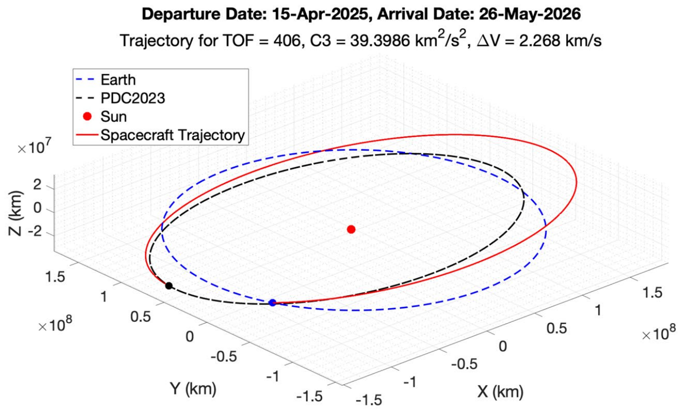
- Rendezvous
- Observe mitigation mission
- 2 year development

#	Requirement
R1	THEO shall launch no later than December 2025
R2	THEO shall arrive at 2023 PDC no later than two years after launch
R3	THEO shall have a wet mass no more than 2,500 kg
R4	THEO shall study the morphological characteristics of 2023 PDC
R5	THEO shall study the gravitational and dynamical properties of 2023 PDC
R6	THEO shall study the elemental composition of 2023 PDC
R7	THEO shall image the space around 2023 PDC
R8	THEO shall have a mission lifetime of 12 years
R9	THEO shall deliver final science data no later than one year after arrival at 2023 PDC



Launch Period

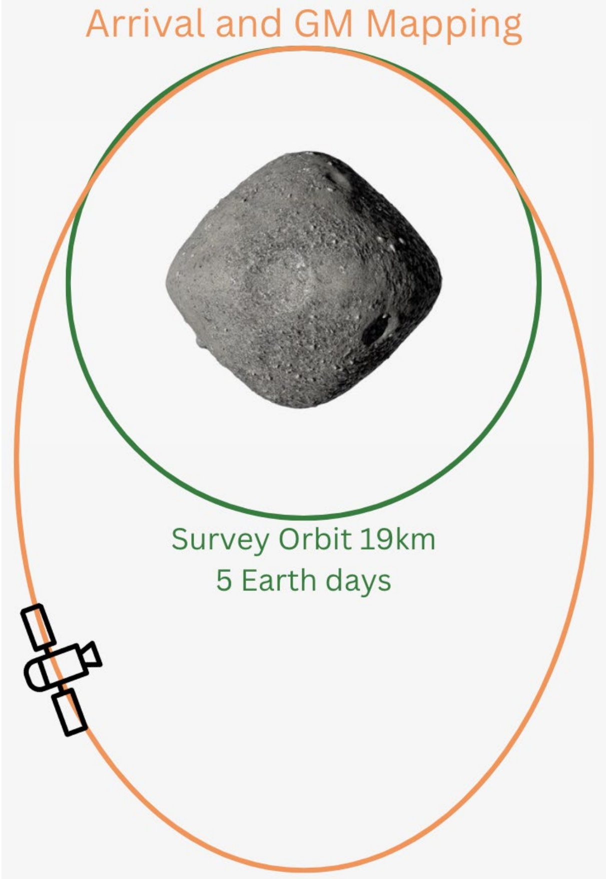




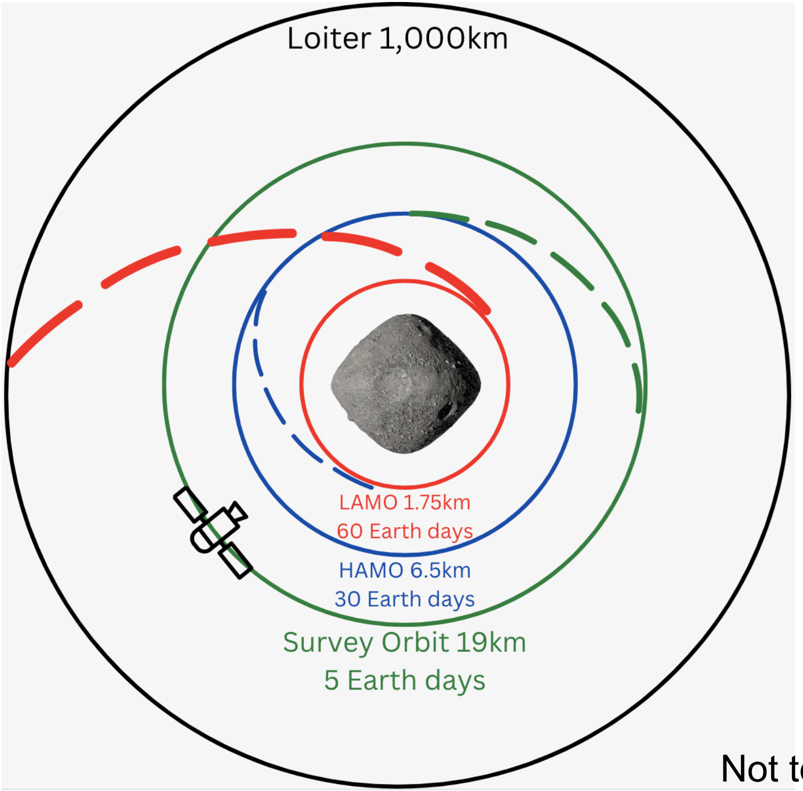
Trajectory Search Parameters:

- **Earth Launch:** 1 January 2025 – 1 January 2026
- **DLA** < |(57°)|
- **C3** < 70 km^2/s^2

Max Post Launch
ΔV ~ 2.6 km/s



- HAMO - High Altitude Mapping Orbit
- LAMO - Low Altitude Mapping Orbit



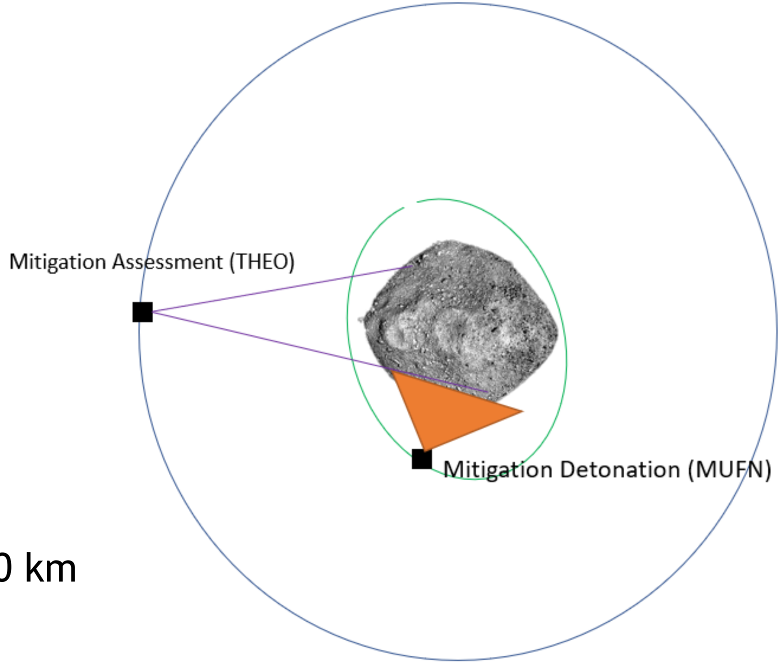
Not to scale

Scientific Instruments

Instrument	Asteroid Property	Orbit Used In
Imagers	Surface Topography Internal Structure	Survey Orbit (Low Resolution Mapping) HAMO
Spectrometer	Mass Mineral Composition	Survey Orbit (Low Resolution Mapping) HAMO LAMO
Laser Altimeter	Shape Model Rotation State Gravitational Attraction	LAMO
X-band Transponder	Gravity Field	GM Mapping Survey Orbit HAMO LAMO

THEO Mitigation Assessment: Overview

- THEO will observe mitigation mission detonations
  - Determine if detonation occurred
  - Assess effects of detonation on orbital parameters
  - Relay data back to Earth
- Safe THEO loiter distance must be established
  - Line of sight required for detonation observation
  - MeV cause internal charging of dielectric surfaces
- Radiation dose analysis highly complex:
  - THEO will be positioned no less than 1,000 km away from detonation location



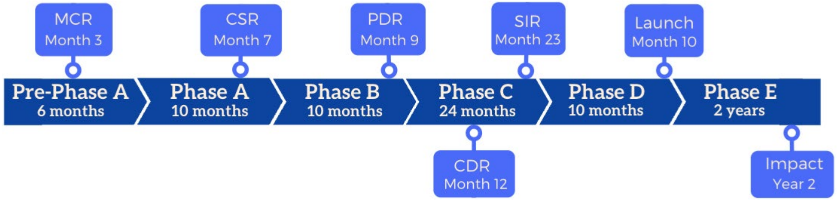
Mitigation Using a Fission Nuclear device (MUFN)

Mitigation Mission

Requirements and Mission Parameters

- Five year development
- 2 year lifetime until impact
- **Mitigation Method:** Nuclear Standoff Detonation
- Minimum Required Imparted Change in Velocity on 2023 PDC at 7 years before Earth-impact: **ΔV=25 mm/s**

#	Requirement
M1	MUFN shall neutralise the threat posed by 2023 PDC
M2	MUFN's first launch shall be no later than April 2029
M3	MUFN shall arrive at 2023 PDC no later than two years after launch
M4	MUFN shall have a wet mass no more than 13,000 kg



- **Launch Vehicle:** SpaceX Falcon Heavy
- **Target Launch Date:** 9 April 2028 (beginning of launch period)

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Overview of Nuclear Standoff Detonations

- A NED is detonated at a set distance above the surface of the target asteroid
- Radiation and high-energy particles impact the asteroid and vaporize a thin layer of the surface material
  - Due to the detonation occurring in the vacuum of outer space, there will not be a post-detonation blast wave
- Outgassing occurs from the ablation of surface material and the momentum transfer imparts a change in velocity on the target asteroid

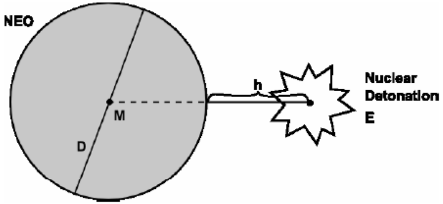

Image from *Evaluating the Effectiveness of Different NEO Mitigation Options*

Sources:
https://history.nasa.gov/conghand/nuclear.htm#:~:text=If%20a%20nuclear%20weapon%20is,as%20usually%20defined%2C%20also%20disappears.
Evaluating the Effectiveness of Different NEO Mitigation Options

MUFN Concept of Operations

1. SpaceX Falcon Heavy launches MUFN spacecraft bus which carries payload of 4 NED-equipped BELASats
2. MUFN rendezvous with 2023 PDC and station keeps above asteroid
3. Single NED-equipped BELASat deploys from MUFN, maneuvers 180 degrees out of phase from MUFN, and detonates at optimal positioning “in front” of asteroid
4. Once debris clears MUFN completes one revolution around 2023 PDC to survey blast-site
5. Steps 4 and 5 repeated until all BELASats have launched and detonated

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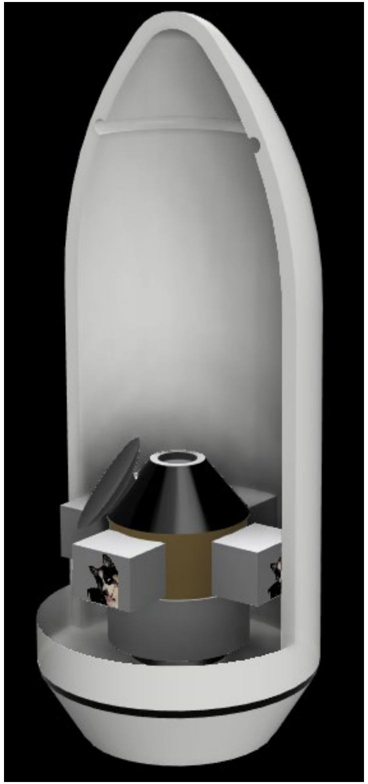
Nuclear Explosive Device Selection

Individual NED Yield (kt)	Number of NEDs per FH	Imparted ΔV per NED (mm/s)	Total Imparted ΔV per FH (mm/s)	Individual NED Mass (kg)	Total NED Mass per FH (kg)	Remaining FH Payload Capacity (kg)	Optimal NED Standoff Distance (m)
NED Options for Successful Deflection with 1 Falcon Heavy Launch Vehicle							
Not Possible							
NED Options for Successful Deflection with 2 Falcon Heavy Launch Vehicles							
80	15	0.84	12.56	44.44	666.67	333.33	63.00
85	15	0.88	13.13	47.22	708.33	291.67	65.00
90	14	0.91	12.79	50.00	700.00	300.00	66.00
95	14	0.95	13.30	52.78	738.89	261.11	68.00
100	13	0.99	12.83	55.56	722.22	277.78	69.00
105	13	1.02	13.29	58.33	758.33	241.67	70.00
110	12	1.06	12.69	61.11	733.33	266.67	71.00
NED Options for Successful Deflection with 3 Falcon Heavy Launch Vehicles							
125	8	1.16	9.28	69.44	555.56	444.44	75.00
130	7	1.19	8.35	72.22	505.56	494.44	76.00
160	7	1.38	9.67	88.89	622.22	377.78	81.00
165	6	1.41	8.47	91.67	550.00	450.00	82.00
205	6	1.64	9.85	113.89	683.33	316.67	87.00
210	5	1.67	8.35	116.67	583.33	416.67	88.00
290	5	2.08	10.41	161.11	805.56	194.44	96.00
295	4	2.11	8.42	163.89	655.56	344.44	96.00
340	4	2.32	9.27	188.89	755.56	244.44	99.00
445	4	2.77	11.08	247.22	988.89	11.11	105.00
450	3	2.79	8.37	250.00	750.00	250.00	106.00

- Imparted ΔV per kg of NED mass is higher for lower nuclear payload yields
  - More “wasted radiation” for larger nuclear detonations
- Each MUFN mitigation spacecraft bus transports four 340 kt NED-equipped BELASats
  - Each BELASat detonation imparts **2.32 mm/s** of ΔV (at optimal standoff distance)
  - 3 Falcon Heavy launches required to impart a total ΔV of **27.8 mm/s** and clear the 25mm/s requirement

MUFN Mitigation Vehicle Design



Properties	Value	Notes
Oxidizer (MON-25) Tank Height (m)	1.7	2 tanks; Capsular design; .75 m diameter; Design for max capacity
Propellant (MMH) Tank Height (m)	1.7	2 tanks; Capsular design; .75 m diameter; Design for max capacity
Helium Tank Radius (m)	0.4	2 tanks; Spherical design; Design for max capacity
Hydrazine Tanks Radius (m)	0.15	2 tanks; Spherical design; Design for max capacity
Engine Height (m)	0.7	2 engines, a primary and backup
NED Bus Height (m)	1.5	Bus and NEDs
Central Truss Structure Height (m)	1.8	Tanks built around the central truss structure and the engines below
Total Height (m)	4	Tanks are arranged a ring around the central truss structure and the NED bus is stacked on top. Maximum Width is 3 meters

Subsystem	***Mass (kg)***	***Notes***
Payload Mass (kg)	1000	Multiple NED devices, bus, comms, computers, and sensors
Fuel Mass (kg)	1600-2900	Spread across all oxidizer and propellant tanks
Inert Mass (kg)	528	Includes: Tanks, engines, thrusters, RTGs, Helium, structure, hydrazine, and thermal control
Total (kg)	3000-4500	LV capabilities vary from 3300 to 7600
ΔV Achieved (km/s)	2.3-3.4	Target ΔV is 2-3.5 km/s. This includes a 20% mass design reserve on the spacecraft.

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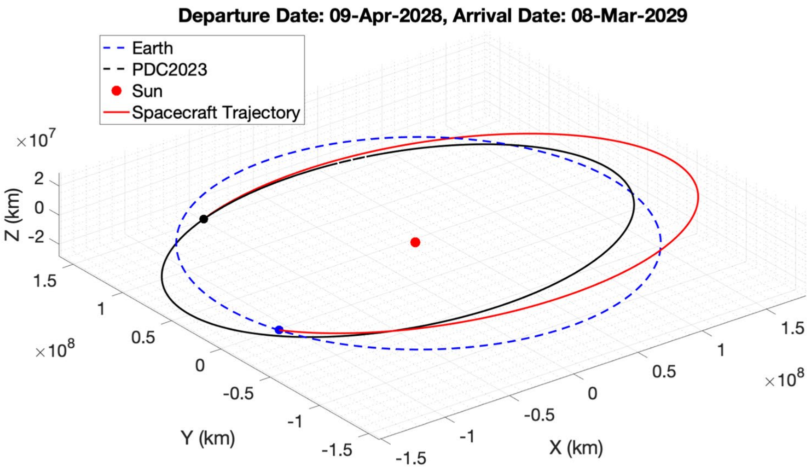
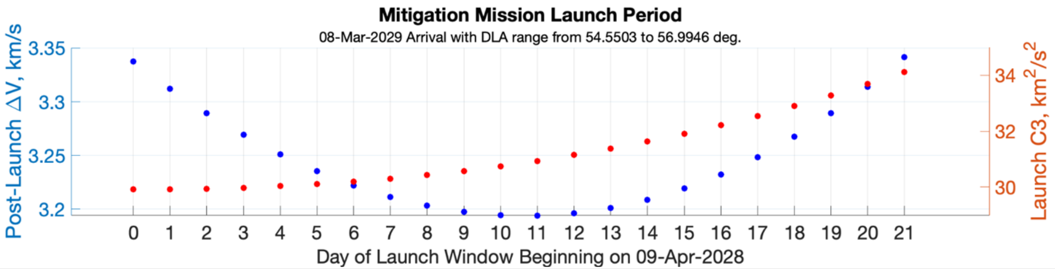
Trajectory Design

- There are 5 launch periods in 2028 and 2029
  - April 2028, May 2028, May 2029, September 2029, October 2029
  - 3 Launches are required
- All launch periods have a DLA < |(57°)|, C3 < 60 km^2/s^2, ΔV < 3.5 km/s





Expected Outcome

- The MUFN program will launch 4 separate Falcon Heavies, each housing a MUFN mitigation spacecraft that contains 4 independent NED-equipped BELASats
  - Only 3 MUFN deliveries are required for mission success
- Each MUFN spacecraft will rendezvous with 2023 PDC and deploy the BELASats in series for detonation
- After the post-detonation debris clears, MUFN performs a surveying revolution then deploys the next BELASat
  - Ejecta from detonation clears in approximately 2 hrs
- Total of **27.8 mm/s** of ΔV imparted from 3 Falcon Heavy launches
  - Option to impart **37.1 mm/s** of ΔV with redundant 4th Falcon Heavy launch

Optimal miss distance = 3500 km, all launch periods result in successful deflection

Backup Section

THEO - Science Traceability Matrix

Near Earth Object Characterization

<table>
<tr><th>Science Goals</th><th>Science Objectives</th><th>Planetary Defense Operations</th><th>Physical Parameters</th><th>Observables</th><th>Instruments</th><th>Instrument Requirements</th><th>Top Level Mission Requirements</th></tr>
<tr><td rowspan="4">What are the charactersitics of 2023 PDC? With the aim of gathering information to inform planetary defense activities and design a mitigation mission to neutralize the threat posed by 2023 PDC.</td><td rowspan="4">Determine high-level NEO charactersitics to inform mitigation options for planetary defense operations.</td><td>Earth Impact Effects Modeling
Disaster Response Planning
Mitigation Mission Planning</td><td>Mass
Mineral Composition</td><td>Reflected Sunlight
X-Ray Emission
Long Term Astrometry</td><td>IR/Visual Spectrometer
X-Ray Spectrometer
Imagers</td><td rowspan="4">1. Imager (Otical, IR)
FOV: 4x4 deg
Focal Length: 125 mm
F-number: 3.3
Resolution (IFOV): 0.068 mrad
IFOV: 14.0 arcsec
Filter Wheel: 0.4 - 5 microns

2. IR/Visual Spectrometer
Wavelength: 0.4 - 4.3 microns
FOV: 4-mrad diameter circle

3. X-Ray Spectrometer
Focal Length: 20 cm
FOV: 27.6deg
Energy Range: 0.5 - 7 keV
Energy Resolution: <260 eV @5.9 keV

4. Laser Altimeter (LIDAR)
Range (km): 0.01 - 10
Accuracy (cm): 6(L), 31(H)
Resolution (cm): 0.1 (bit), 1.1 (L), 2.6 (H)
Divergence (micro rad): 100(L), 200(H)
Pulse Rate (Hz): 10000 (L), 100 (H)
Pulse energy (mJ): 0.01 (L), 0.7 (H)

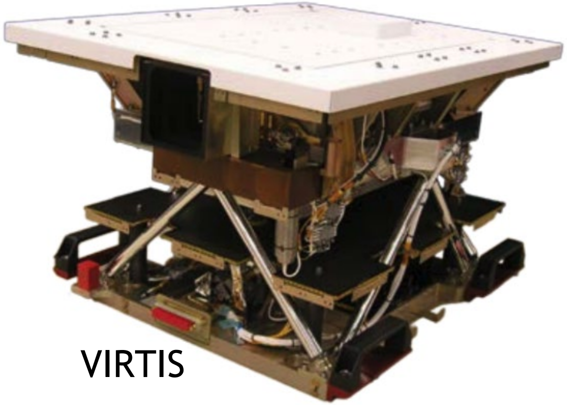
5. X-Band Transponder
External Sync Frequency: 125Hz nominal</td><td rowspan="4">Determine physical parameters of 2023 PDC to inform planetary defense activities and the design of a mitigation mission.

Observe the mitigation mission and gather data about the mitigation method used to inform future planetary defense activities.

Follow 2023 PDC after the mitigation mission to study the effects of the mitigation mission on 2023 PDC's properties to ensure mitigation mission success.</td></tr>
<tr><td>Earth Impact Effects Modeling
Disaster Response Planning
Mitigation Mission Planning</td><td>Binarity</td><td>Space around 2023 PDC</td><td>Imagers</td></tr>
<tr><td>Earth Impact Effects Modeling
Disaster Response Planning
Mitigation Mission Planning</td><td>Strength
Internal Structure
Porosity
Rotation State</td><td>Surface Topography</td><td>Imagers
IR/Visual Spectrometer
Laser Altimeter</td></tr>
<tr><td>Mitigation Mission Planning</td><td>Gravity Field</td><td>Doppler Shift from spacecraft's radial velocity vector component relative to the Earth</td><td>X-Band Transponder</td></tr>
</table>

Science Instruments: COTS

- Malin Space Science Systems
  - IR Camera: ICAM IR3S
  - Visible Wavelength Camera: ECAM C50
  - Heritage from Mars missions and OSIRIS-REx

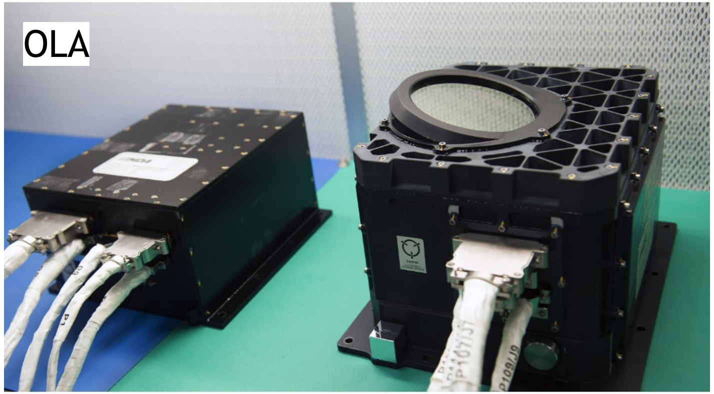


VIRTIS

- Leonardo Airborne & Space Systems
  - VIR Spectrometer: VIRTIS
  - Heritage from Rosetta, Dawn, Venus Express

Science Instruments: Heritage

- OSIRIS-REx build to print
    - Laser Altimeter: OLA
    - X-Ray Spectrometer: REXIS



Preliminary Mass Budget

- Dry mass: 730 kg
- THEO mass at launch: 1,956 kg
- Max C3 70 km^2/s^2
- Max launch using a Falcon Heavy Expendable is 2770 kg
- Total Mission Delta V: 2.6003 km/s

Subsystem	% Margin	Mass [kg]
Science Instruments	75	122
Structure	25	155
Thermal	50	37
Power	35	116
TT&C	75	50
GNC	30	43
Propulsion	25	207
Interplanetary Travel Propellant	30	1225
PDC 2023 Proximity Operations Propellant	400	1

ANSI/AIAA Mass Properties Control for Space Systems

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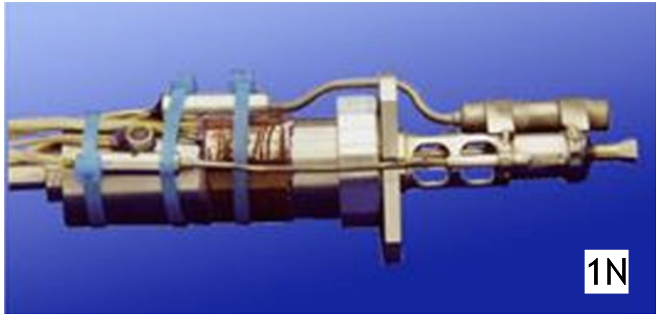
Propulsion System

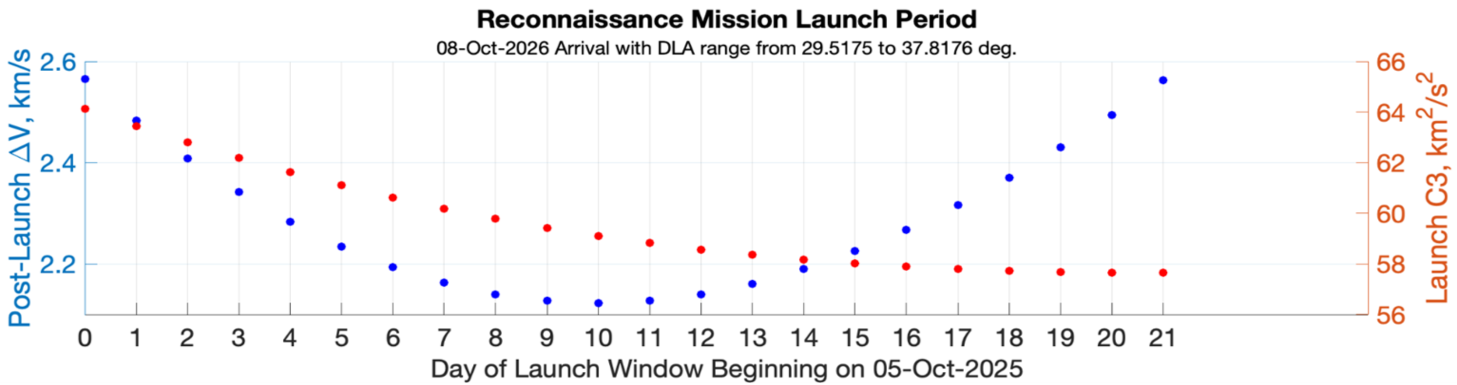
- MMH-MON3 Engines
- Aerojet Rocketdyne
    - R-4D-15
        - High thrust applications
        - 445 N nominal steady state thrust
        - 320 s Isp
        - 4 Thrusters
    - MR-103G 1N
        - Attitude and station keeping maneuvers
        - 0.19 - 1.13 N Thrust
        - 202 - 224 s Isp
        - 12 Thrusters
- Ariane Group
    - Propellant Tanks
        - 700-1108 Litre



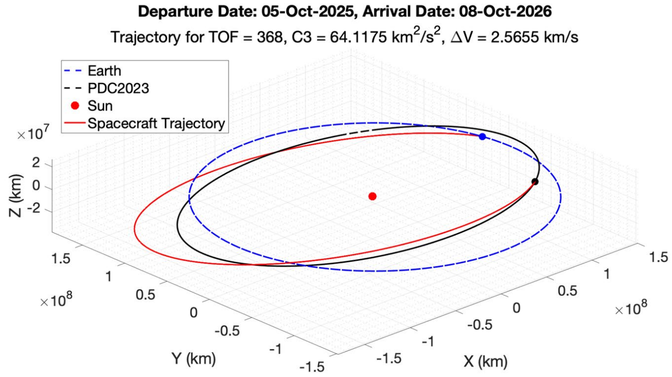
Spacecraft Design (Continued)

- **Early Encounter: Locate 2023 PDC**
  - 2023 PDC will be a faint object with high variability of brightness due to its rotation and shape irregularities, as well as its phase angle with the SUN and spacecraft close to 90 degs
  - SC is maintained in inertial 3-axis stabilized attitude. No maneuvers conducted.
  - Boresight of the navigation camera pointed towards the expected position of the asteroid at the beginning of the rendezvous phase.
- **Approach**
  - Decreasing magnitudes of maneuvers of the approach as the distance to the target decreases.
- **Orbit Maintenance**
  - Orbit insertion for proximity operations and station-keeping at an offset distance for recording of MUFN

Backup Launch Period



- **Earth Launch:** 01/01/2025-01/01/2026
- **DLA** < abs(57°)
- **C3** < 70 km2/s2
- **Max DeltaV** ~ 2.6 km/s



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PDC Arrival Maneuver Sequencing

Trajectory Description	Transfer Type	Scientific Requirement	Burn Time [min]	ΔV (km/s)
Deep Space Maneuver 1 (~1 week from arrival)	-Reduction in speed	-	18	1.0
Deep Space Maneuver 2 (~2 days from arrival)	-Reduction in speed	-	10	0.75
Deep Space Maneuver 3 (~12 hr from arrival)	-Reduction in speed	-	8	0.75
Elliptical GM Orbit (7 km apoapsis radius, e = 0.2)	-Elliptical capture orbit maneuver	-Determine GM of PDC	1	0.1

PDC Arrival Maneuver Sequencing

Trajectory Description	Transfer Type	Scientific Requirement	Burn Time [s]	ΔV (km/s)
Survey Orbit (19 km altitude)	-Circularize orbit at periapsis -Hohmann transfer, decrease radius	-Visual and IR images -Spectrometers -Low resolution mapping -Search for debris and secondary bodies -Gravity field measurements -20-Day Orbit	3	8.98e-05
High Altitude Mapping Orbit (6.5 km altitude)	-Hohmann transfer, decrease radius	-Imagers and spectrometers -Gravity field measurements -30-Day Orbit	0.15	4.19e-05
Low Altitude Mapping Orbit (1.75 km altitude)	-Hohmann transfer, decrease radius	-Gamma ray spectrometer -Laser altimeter -Gravity field measurements -60-Day Orbit	0.26	7.77e-05
MUFN Observation Orbit (1,000 km altitude)	-Hohmann transfer, increase radius	-	0.28	1.21e-05
THEO Station Keeping (1,000 km alt over 5-year period)	-	-	-	0.053

Spacecraft Design (Continued)

Thermal Control System (TCS)

- THEO shall be almost 1AU from the sun throughout its trajectory: 1367 W/m^2 of solar flux
- Passive control: thermal surface finishes and multilayer insulation
- Active control: heaters and radiators

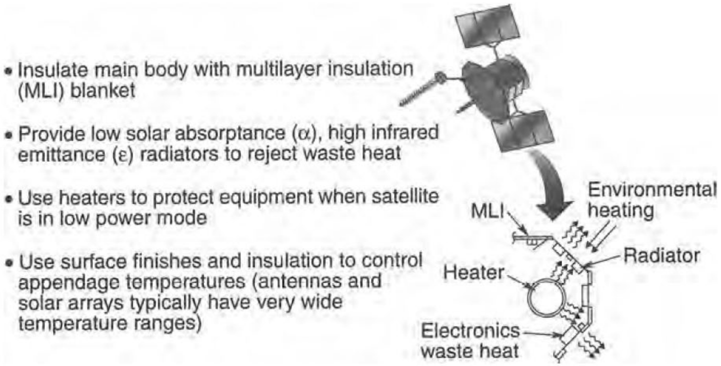

Fig. 3.2. Three-axis-satellite thermal control.

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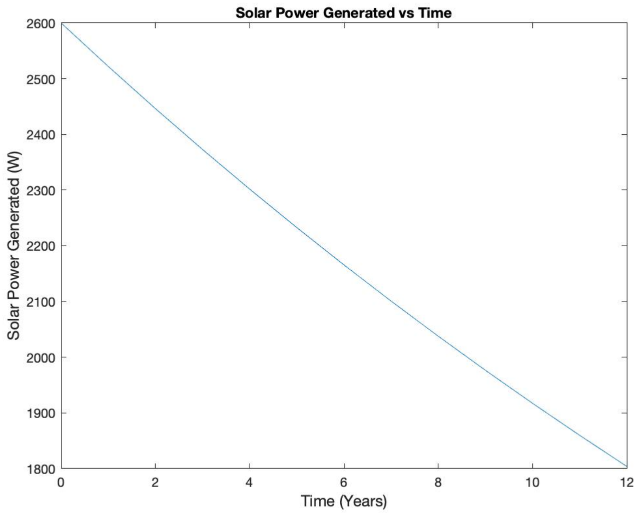
THEO Power Budget

Subsystem	Components	Expected Peak Power Usage		
		Power (W)	Duty Cycle	Avg Power
Science	IR Camera	2	1	2
	Optical Camera	1.6	1	1.6
	IR/Visual Spectrometer	36	1	36
	X-Ray Spectrometer	12.4	1	12.4
	LIDAR	59	1	59
	DVR	40	1	40
Thermal	Radiators	100	3	300
Comms	Ka, X-Band Transponder (Receiver, Transmitter)	15.8	2	31.6
	Ka, X Band Power Amplifier	172	2	344
	Ultrastable Oscillator (Spacecraft Clock)	5	2	10
	Antenna	0.9	1	0.9
	Tranceiver	15	1	15
GNC	Sunsensor	0.7	6	4.2
	Star Tracker	5.6	3	16.8
	Inertial Measurement Unit	13.5	2	27
	3-DOF Gyro	45	2	90
	Computer (Power Management/CDH)	200	1	200
	Control Actuators	17	2	34
Propulsion	R-4D-15 (High Thrust, Aerojet Rocketdyne)	140	1	140
	MR-13G 1N (Attitude, Aerojet Rocketdyne)	17	1	17
	Subtotal Power (W)			1381.5
	Power Design Margin (%)			20
	Total Power (W)			1657.8

Power System - Solar Array

- Selected Solar Array: Spectrolab NeXt Triple Junction (XTJ) generating at 366 W/m^2 at 1 AU.
- To meet the expected power requirement of 1700 W, considering solar degradation rate of 3%, solar panel area required: 7.10 m^2
- The 2 solar panel arrays are gimballed and placed opposite each other on THEO bus.



Power System - Battery

- Selected battery: Lithium ion from Hayabusa mission
- The rated capacity of the cell was 13.2 Ah/500g when the cell was discharged with 2.64 A (0.2 C) at 20°C.
- The battery consisted of 11 cells connected in series. The battery was charged by supplying it with a charge current of 500 mA through the battery charge regulator.

Image Source: JAXA

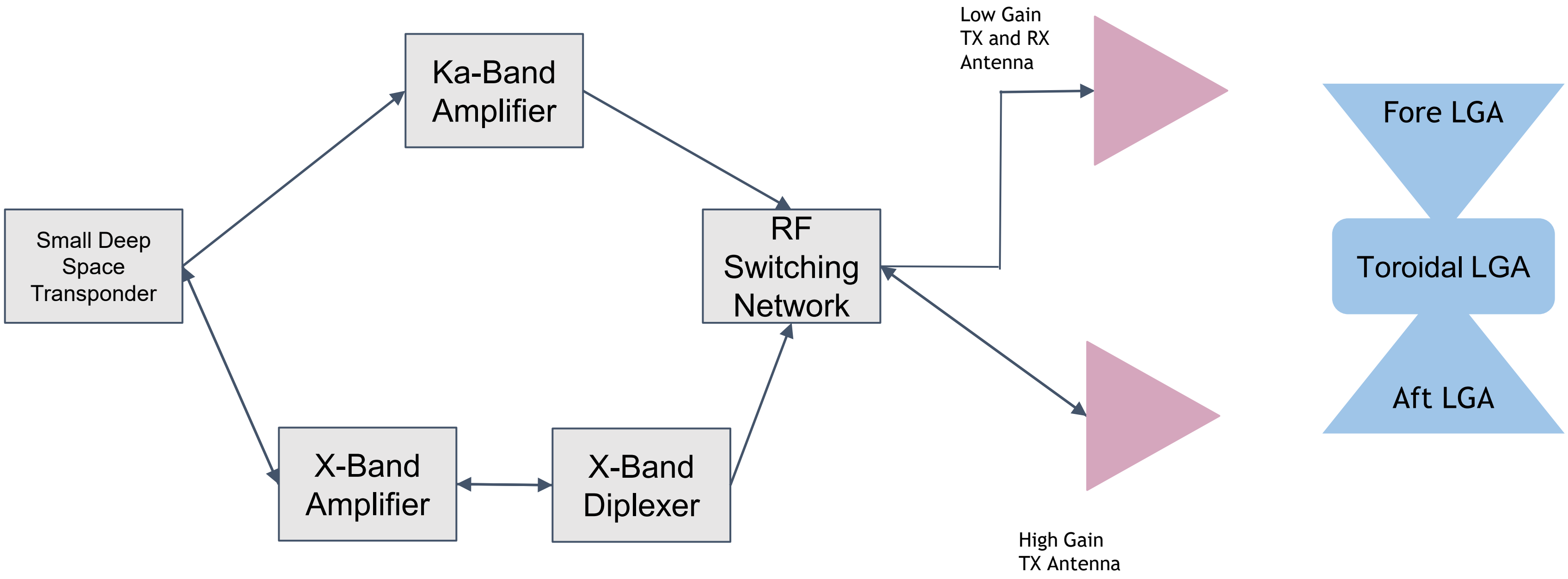
Source:
Yoshitsugu S., Charge/discharge performance of lithium-ion secondary cells under microgravity conditions; Lessons learned from operation of interplanetary spacecraft Hayabusa (2013).

Communications System

A dual X-band and Ka-band system, using Small Deep Space Transponder (SDST), are chosen for redundancy.

Band	34-Meter DSN Frequency Parameters
Ka-Band, Downlink	31.80 - 32.30 GHz
X-Band, Uplink	7.145 - 7.235 GHz
X-Band, Downlink	8.2 - 8.6 GHz

Communications System



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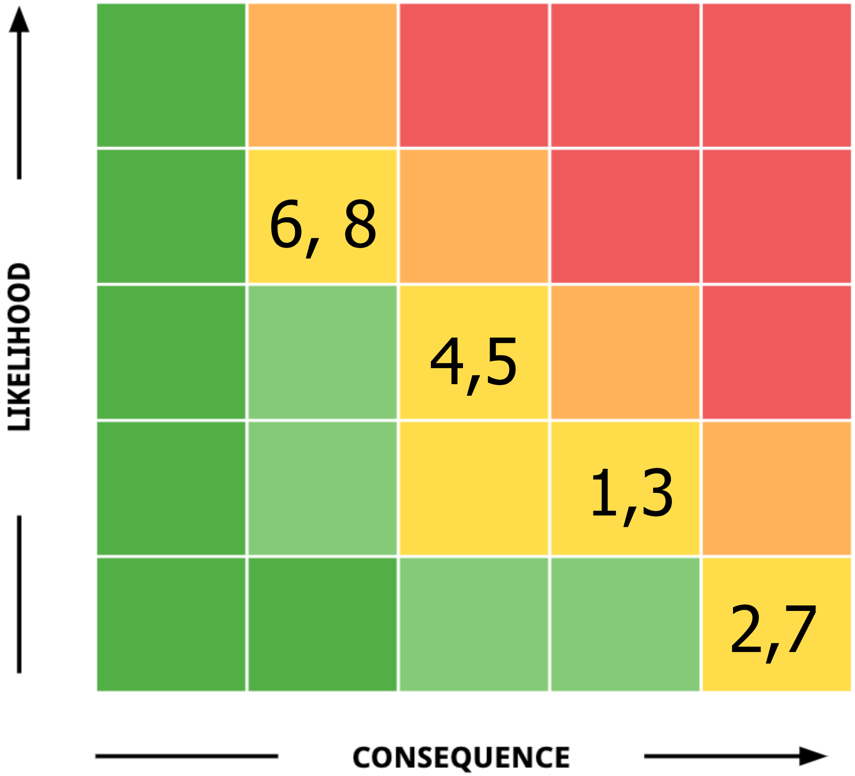
Spacecraft Design (Continued)

Guidance, Navigation, and Control (GNC)

- Heritage from SMART-Nav and DRACO from NASA's DART mission
- Continuously point solar array assembly towards sun
- Guarantees link with Deep Space Network
- 3 major phases:
  - Early Encounter
  - Approach
  - Orbit Maintenance

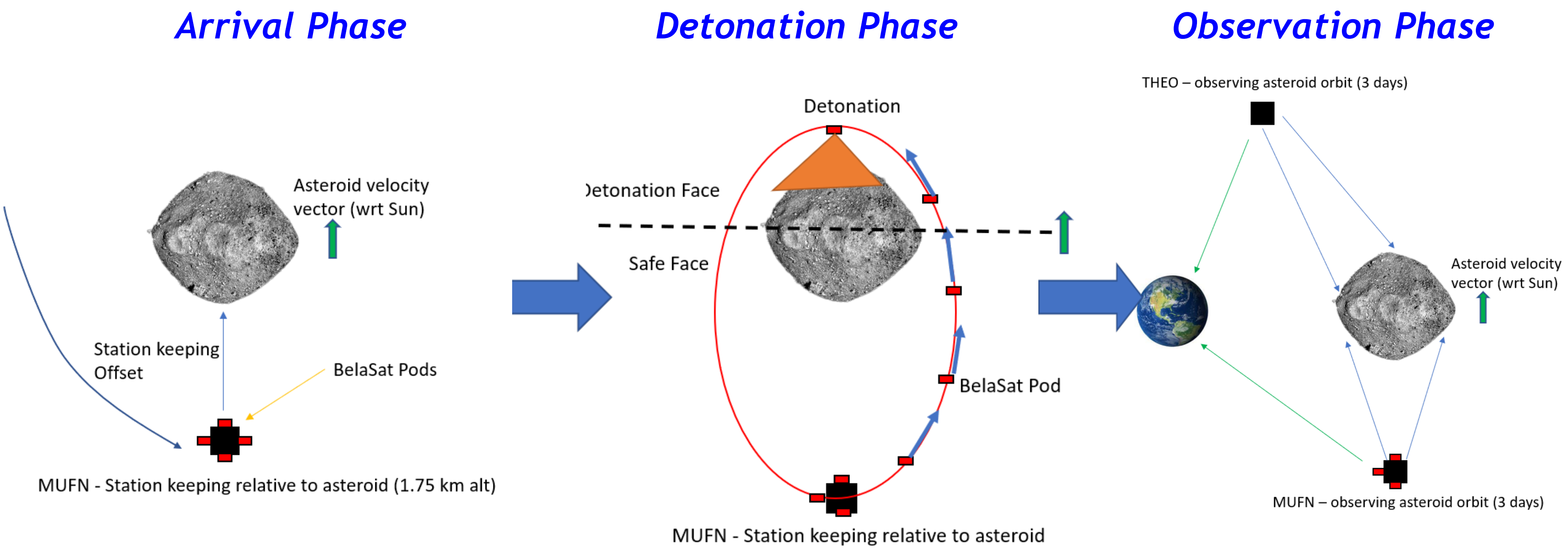
Risk Chart



Risk #	Description
1	Short development phase
2	Launch vehicle failure
3	Critical spacecraft component failure
4	Rendezvous difficulties
5	Orbit difficulties
6	Budgetary constraints
7	Separation system from launch vehicle
8	Failure after culmination of science mission

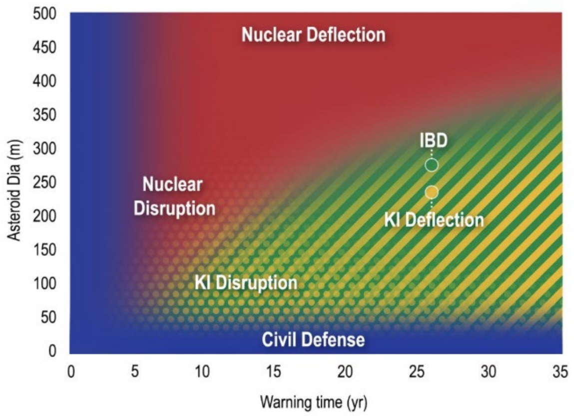
MUFN Backup Slides

CONOPS Visualization



Decision Process for Mitigation Method

- Applying NEO size and time constraints on the mitigation mission leaves nuclear detonations and kinetic impactors as the only viable options
- Kinetic impactors were initially investigated and ruled out
  - It was found that in excess of 100 SLS Block 2 launches would be required to deliver enough mass to divert the asteroid at its upper mass bound



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Nuclear Standoff Detonation Effectiveness

- Research supports nuclear standoff detonations having a high probability of success against almost all asteroid types, compositions, and rotational speeds
- Nuclear detonations are least effective against metallic compositions and rubble-pile type asteroids
  - Estimated that only about 8 percent of asteroids are of metallic composition (M-type)
  - Research has shown that the mitigation method is still effective against rubble-pile asteroids, but increased porosity can significantly reduce the surface melt depth from the blast and produce asymmetric ejecta

	Type			Composition					Rotational Speed		
Mitigation Option	Rubble-Pile	Mono-lithic	Unknown Type	Carbon-aceous	Silic-aceous	Metallic	Icy	Unknown Comp.	Slow Rotation	Fast Rotation	Unknown Rotation
Kinetic Impactor	0.1	1.0	0.3	1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.0
Standoff Nuclear Detonation	0.3	1.0	0.8	1.0	1.0	0.6	1.0	0.9	1.0	1.0	1.0
Chemical Rocket	0.1	1.0	0.3	1.0	1.0	1.0	1.0	1.0	1.0	0.5	0.8
Gravity Tractor	1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.0
High Isp Rocket	0.1	1.0	0.3	1.0	1.0	1.0	1.0	1.0	1.0	0.5	0.8
Mass Driver	0.1	1.0	0.3	1.0	1.0	0.3	1.0	0.9	1.0	0.5	0.8

Mitigation method success ratings (0-1.0) from *Evaluating the Effectiveness of Different NEO Mitigation Options*

Sources:
Evaluating the Effectiveness of Different NEO Mitigation Options
Spacecraft Mission Design for the Optimal Impulsive Deflection of Hazardous Near-Earth Objects (NEOs) using Nuclear Explosive Technology
Influence of Porosity on Impulsive Asteroid Mitigation Scenarios

A Note on International Policy

- The Limited Test Ban Treaty prohibits any nuclear explosion in outer space, regardless of its intended purpose
- The Outer Space Treaty prohibits placing a nuclear weapon in orbit, installing it on a celestial body, or stationing it in space in any other manner
- A State has absolute liability for damage done by any space object for which it is a launching State, including cases where an asteroid is insufficiently deflected and impacts at a different location

Source:
The Legal Aspects of Planetary Defense:SMPAG Ad-Hoc Legal Working Group Key Report Conclusions

Requirement for Change in International Policy

Local scale is the size of a metropolitan area. Regional scale is state, province, or smaller country sized.

Diameter of Impacting Asteroid	Type of Event	Approximate Impact Energy (MT)	Average Time Between Impacts (Years)
5 m (16 ft)	Bolide	0.01	1
10 m (33 ft)	Superbolide	0.1	10
25 m (80 ft)	Major Airburst	1	100
50 m (160 ft)	Local Scale Devastation	10	1000
140 m (460 ft)	Regional Scale Devastation	300	20,000
300 m (1000 ft)	Continent Scale Devastation	2,000	70,000
600 m (2000 ft)	Below Global Catastrophe Threshold	20,000	200,000
1 km (3300 ft)	Possible Global Catastrophe	100,000	700,000
5 km (3 mi)	Above Global Catastrophe Threshold	10,000,000	30 million
10 km (6 mi)	Mass Extinction	100,000,000	100 million

2023 PDC Impact Corridor:

- At its upper size estimates, 2023 PDC has the potential to affect over a billion people (~20-25% of Earth's population)
  - Millions potentially affected by tsunamis resulting from an ocean strike
  - Impact ejecta can have long-lasting global climate effects
- The United Nations Security Council (UNSC) has the power to supersede rules of international law through a decision, which requires the votes of nine out of fifteen Members and no opposing vote by one of the Permanent Five (P5) Members of the UNSC

Sources:
Report on Near-Earth Object Impact Threat Emergency Protocols
https://cneos.jpl.nasa.gov/pd/cs/pdc23/PDC23-ImpactRisk-Epoch1.pdf
The Legal Aspects of Planetary Defense:SMPAG Ad-Hoc Legal Working Group Key Report Conclusions

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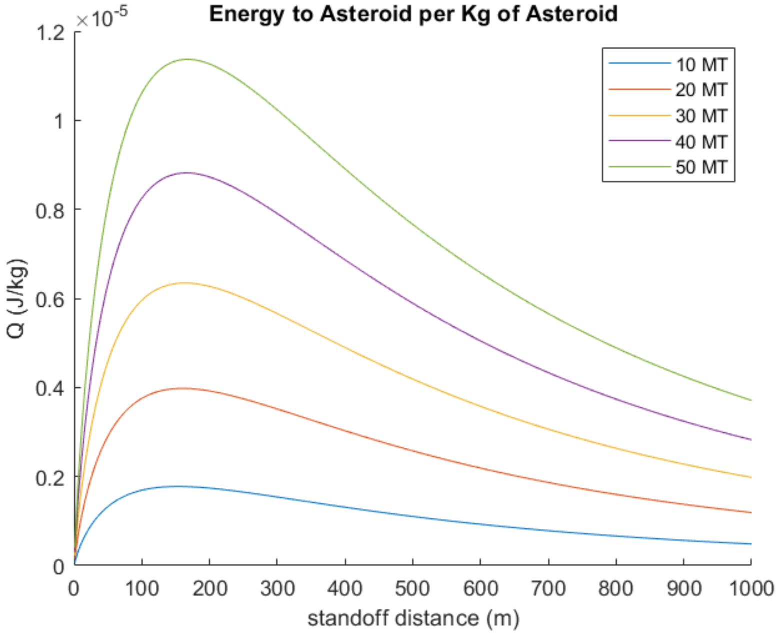
Important Consideration: Disruption

Need to avoid accidental disruption:

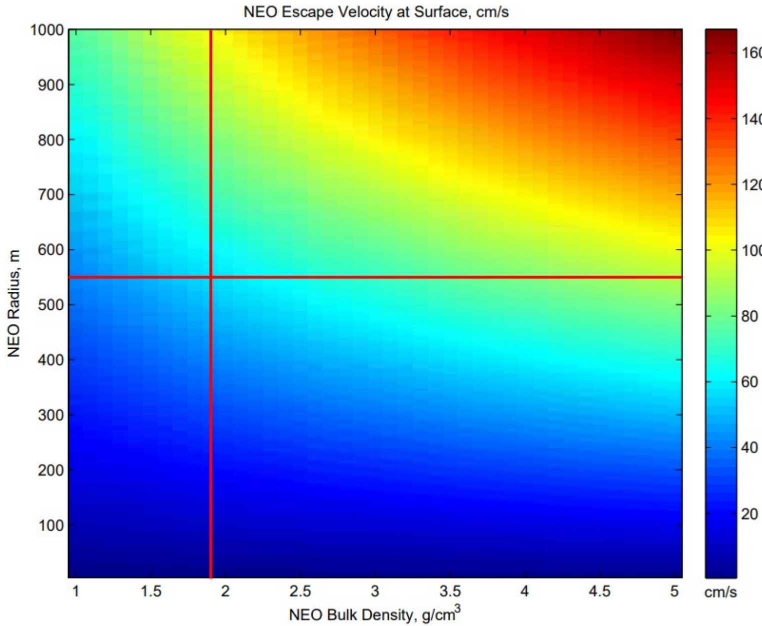
- Heuristic used: energy imparted to asteroid shall not exceed 100-1000 J/Kg
- Assuming change of KE ~ to energy imparted



Conclusion: not at risk of accidental disruption even with total needed yield detonating at once

Disruption Analysis Based on Escape Velocity

- To avoid disruption of 2023 PDC, the ΔV imparted by each NED must not exceed ~10% of the asteroid's escape velocity
    - 2023 PDC escape velocity: 561 mm/s
    - 10% Requirement: 56 mm/s
- The ~10% escape velocity threshold is over 2x the <u>total</u> required ΔV for deflection
    - No risk of disruption



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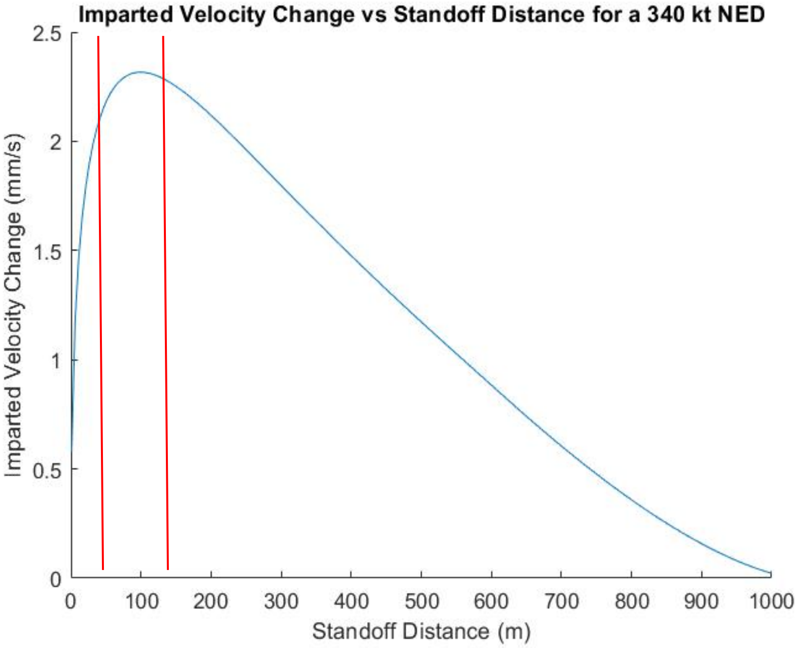
Important Consideration: Detonation Positioning



Acceptable error in detonation standoff:

-30 m to + 60 m

Imparts ΔV greater than 2.2 mm/s with a single detonation

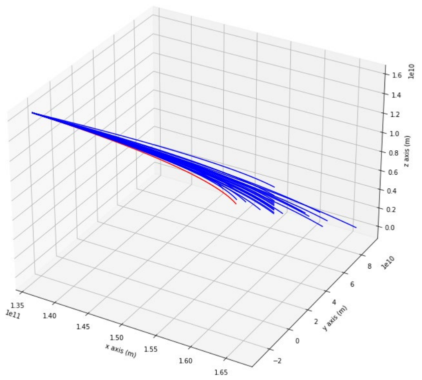
Detonation standoff can be tuned to observed effectiveness real-time

Consideration: Blast Ejecta

Assumptions:

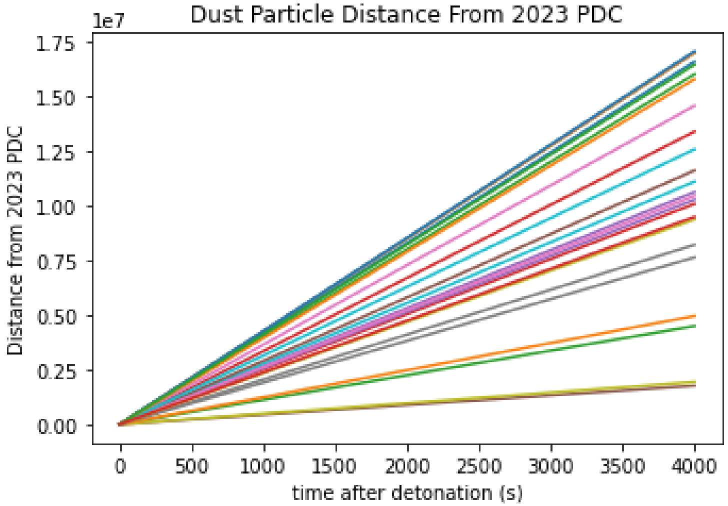
- Cannonball model
- Coefficient of reflectivity of 0.5
- Always in sunlight

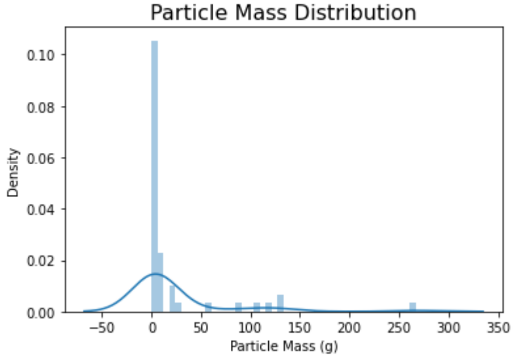
Plot of system bodies orbits

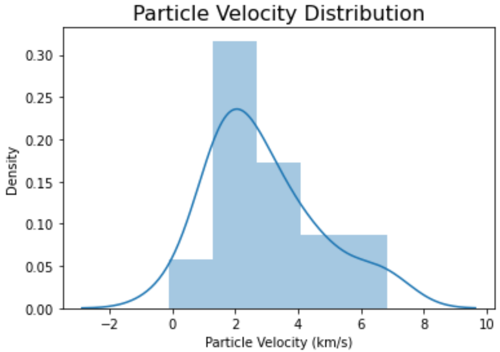


Time = 28 hours

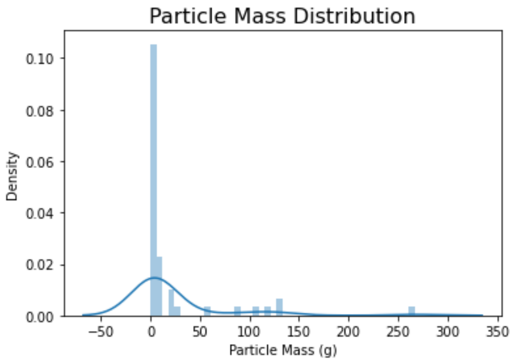
Model shows that ejecta clears in 2-4 hours - represents a safe inter-arrival window between NED detonations

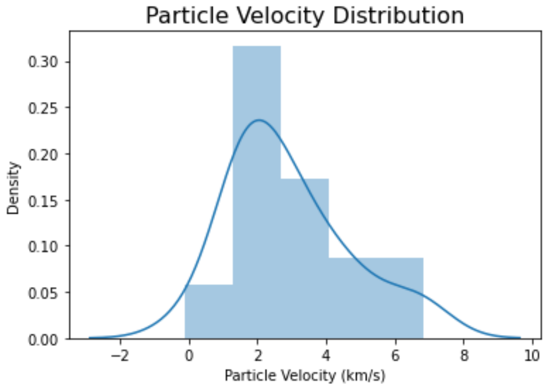






Ejecta Simulation





Particle Velocity Distribution:

- normal , mean = 3 km/s, sd = 1.5 km/s

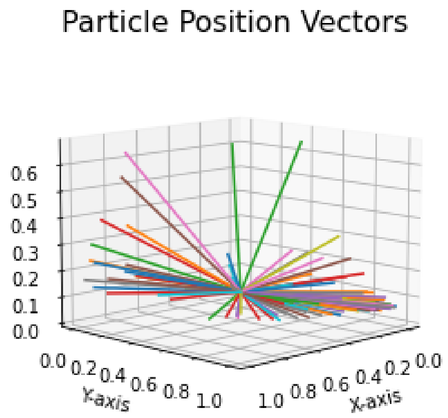
Particle Mass Distribution Distribution:

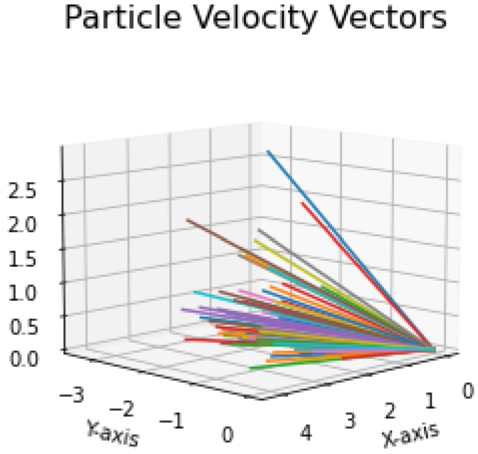
- exponential , mean = 3 grams

Detonation R and V vectors:

- Modeled along track detonation (most effective)
- Simulated Particles originate at 2023 PDC surface with randomized initial r and v vector:

SRP and solar gravity taken into account





Detonation Effectiveness Assessment

Objective: measure imparted ΔV

- Inform Future Detonations
- Gauge effectiveness

Strategy: Optically track 2023 PDC from MUFN, radiometrically track spacecraft from Earth provides sufficient ΔV assessment [Bhaskaran et al]

- Landing radio beacon on 2023 PDC deemed too complex
- Tracking purely from Earth (Optical, Radar) takes too long

Requirement: Track 2023 PDC for 3 days following detonation - 3 day separation between detonations

- ~ sub millimeter uncertainty in imparted ΔV [Bhaskaran et al]

* Bhaskaran et al study 390 m diameter asteroid

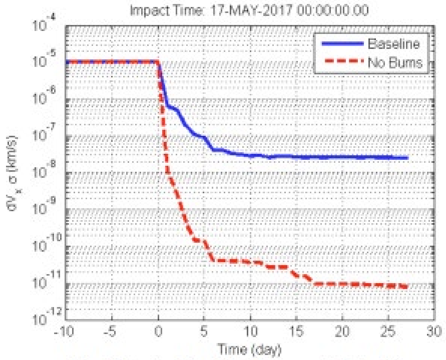


Fig. 5. Asteroid delta-V uncertainty (X)

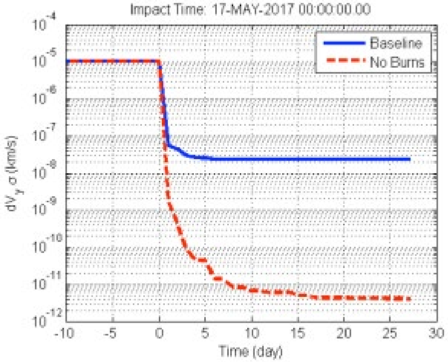


Figure 6. Asteroid delta-V uncertainty (Y)

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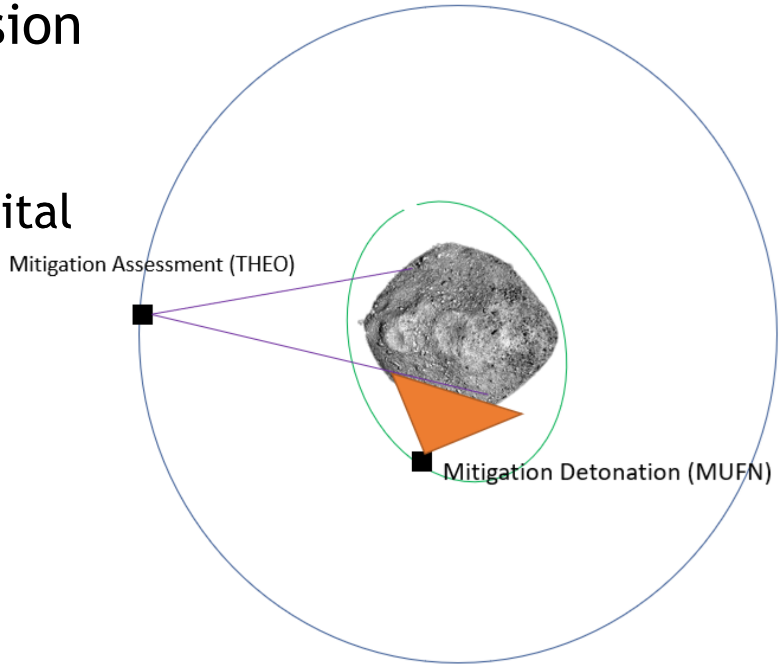
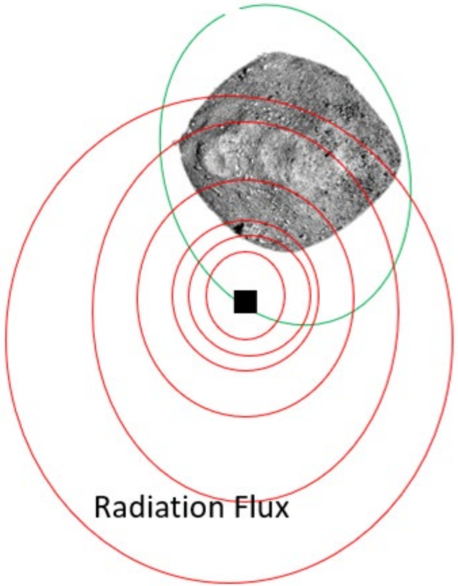
THEO Mitigation Assessment: Overview

- THEO will observe mitigation mission detonations
  - Determine if detonation occurred
  - Assess effects of detonation on orbital parameters
  - Relay data back to Earth
- Ability of THEO to survive detonations must be ensured
  - Ability to assess effectiveness of mitigation missions is paramount



THEO Mitigation Assessment: Considerations

- Safe THEO loiter distance must be established
  - Line of sight required for detonation observation
  - MeV cause internal charging of dielectric surfaces
    - Loss of attitude control
    - Degradation of optical devices
    - Etc
- Trapping of charged particles not a factor
  - Dose governed by inverse square law
- Radiation dose analysis highly complex:
  - THEO will be positioned no less than 1,000 km away from detonation location
  - 1e-10 x MeV flux of detonation



Mitigation of 2023 PDC: Required ΔV

Approximate ΔV needed to estimate nuclear yield:

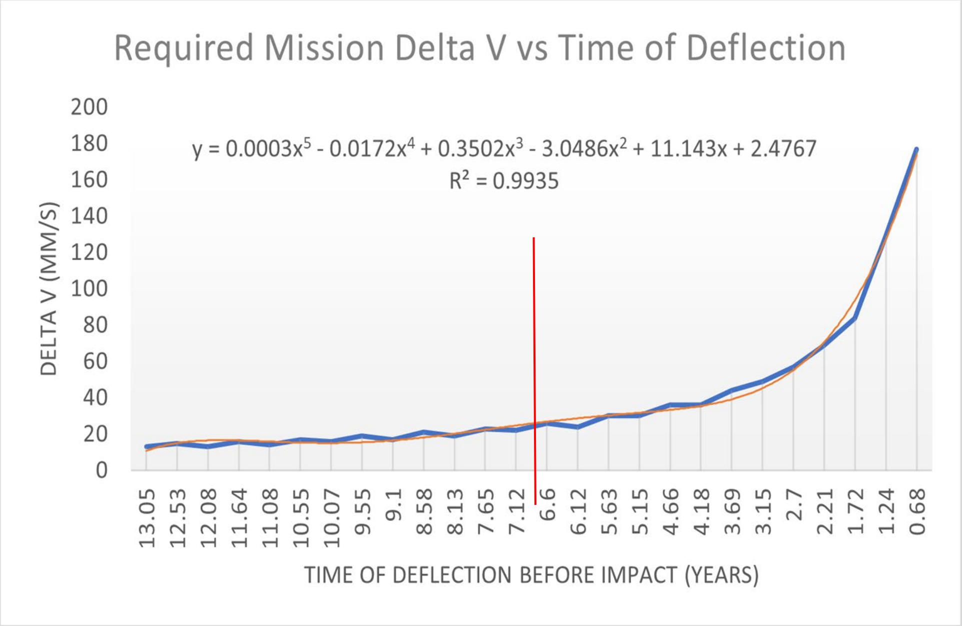
Using CNEOS NEO Deflection App:

- 7 Years before impact
- Parameters closely matched 2023 PDC
- Polynomial curve fit added

To deflect 2023 PDC outside of Earth's B-Plane:

25 mm/s

Provides reference ΔV for following design decisions



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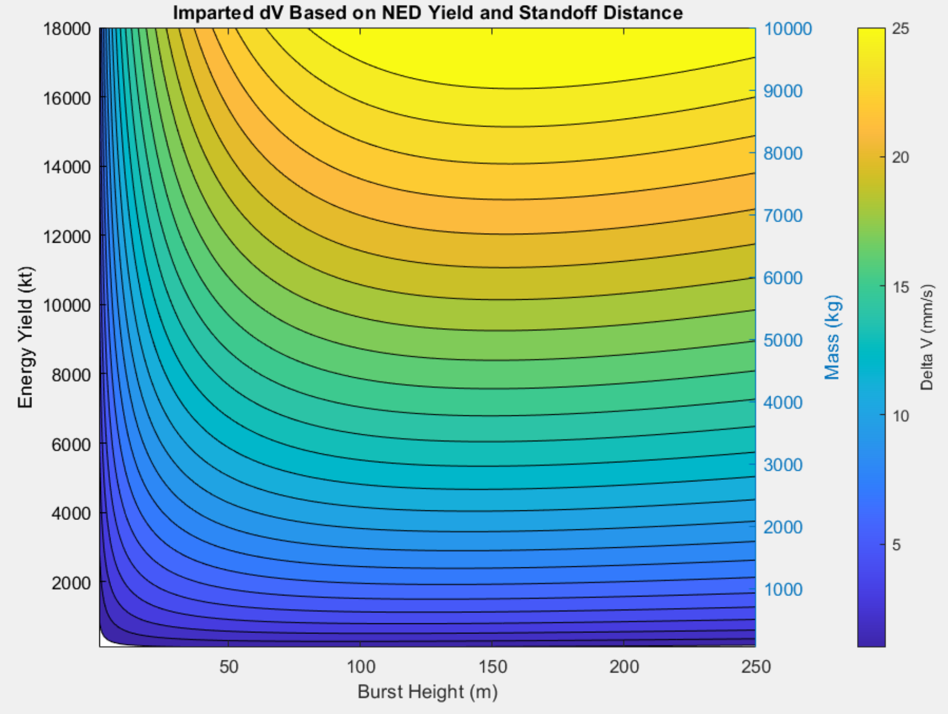
NED Optimization Overview

Objective: Maximize imparted ΔV per kg of payload

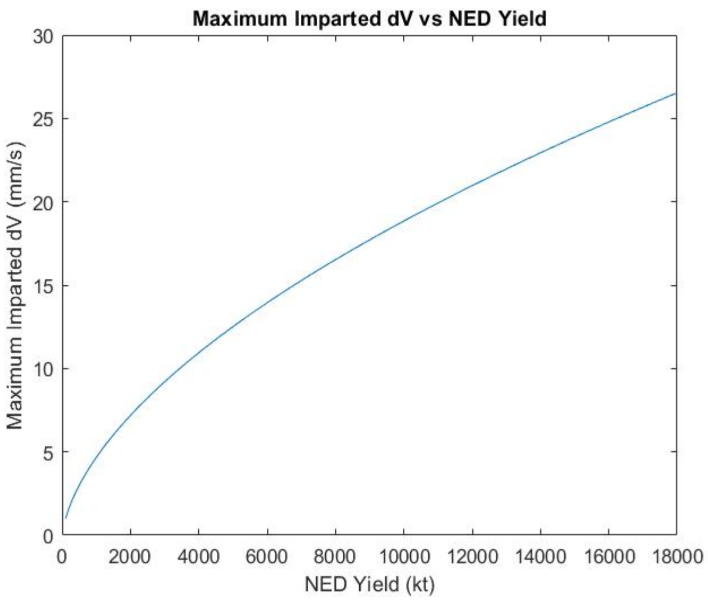
Constraint: Minimum of 25 mm/s of Total Imparted ΔV Required

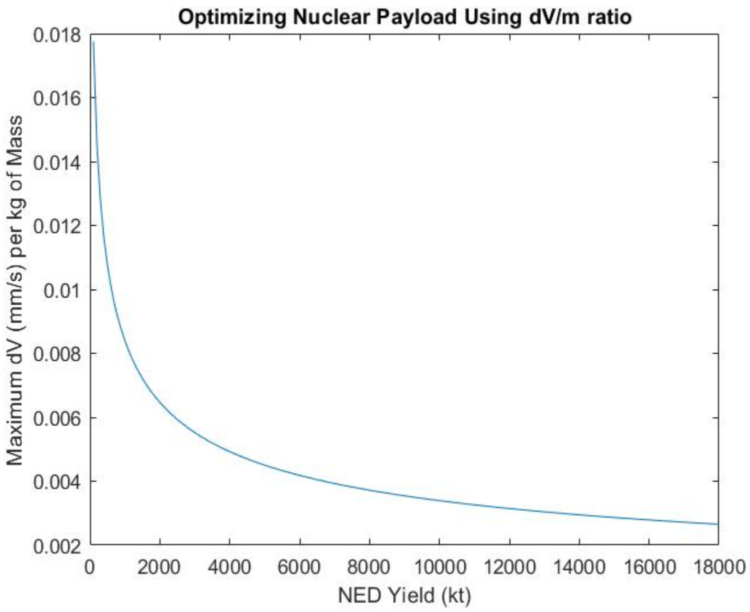
Constraint: 30% Mass Margin Payload Capacity of 1000 kg



- PDR design used 3 mitigation vehicles, each carrying a single 2.7 Mt NED
    - 3 Falcon Heavy Launch vehicles required
    - Each NED imparts 8.6 mm/s of ΔV per detonation
    - Total imparted ΔV of 25.8 mm/s
- Evaluation of the total number of NEDs delivered with each launch vehicle

NED Optimization Criteria





- Imparted dV per kg of NED mass is higher for lower yields
  - Beneficial to target yields less than 3 Mt

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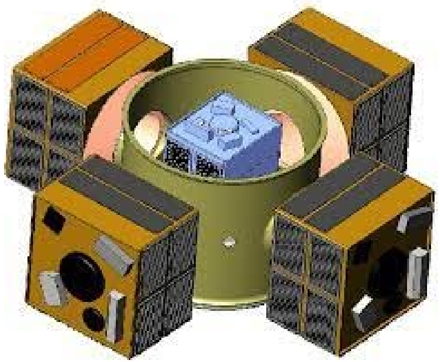
BELASat-MUFN Integration

- BELASat compatible with an ESPA-grande class satellite bus
- ESPA form factor provides a common commercial off-the-shelf (COTS) framework
- Enables flight proven / COTS separation mechanism

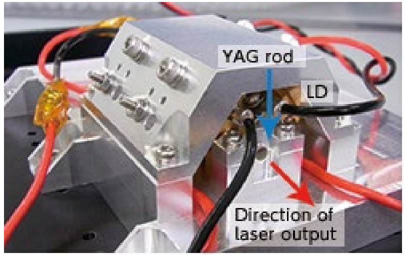


Internal ESPA ring hosts all MUFN power and avionics systems (option to stack multiple rings adds modularity to design)

Image Sources:
Space access for small satellites on rideshare missions with ESPA and ESPA-derived payload adapters

BELASat Design: Instrument Suite

- Focus on reducing payload mass and minimizing potential points of failure
- Hayabusa 2 laser altimeter used for measuring distance to asteroid surface and has been successfully mission tested
    - Compact Laser Altimeter, currently being developed by APL, a viable option that has increased range accuracy and resolution with reduced instrument mass and power consumption
- Asterix 1000 three-axis inertial measurement unit (IMU) used for attitude determination



Hayabusa 2 Laser Altimeter

Astrix 1000 IMU

Image Sources:
https://www.isas.jaxa.jp/e/forefront/2013/mizuno/02.shtml
https://satsearch.co/products/airbus-defence-and-space-astrix-1000

BELASat Reaction Control System

Propellant	Molecular Weight (Kg/Kmole)	Density (g/cm^3)	Specific Thrust (s)	
			Theoretical	Measured
Hydrogen	2.0	0.02	296	272
Helium	4.0	0.04	179	165
Nitrogen	28.0	0.28	80	73
Ammonia	17.0	Liquid	105	96
Carbon dioxide	44.0	Liquid	67	61

- Cold-gas thrusters used for attitude control while in orbit around 2023 PDC
  - Nitrogen gas used for high reliability and minimized storage tank mass
  - 1.83 kg of propellant required for BELASat proximity operations
    - Tripling propellant requirement (5.49 kg) to ensure proximity mission success

Source:
Cold Gas Propulsion System – An Ideal Choice for Remote Sensing Small Satellites

BELASat Power Requirements and Mass Breakdown

- Required instrument operating time of 3 hrs
  - Designing to an operating time of 9 hrs
- Redundant 5.18 kg Lithium-Ion batteries are sufficient for powering BELASat from deployment to detonation
- Off-the-shelf ESPA-grande satellite bus incorporates thermal control for BELASats and associated instruments
- Each BELASat has a total wet mass of 249.65 kg

Component	Power Consumption (W)
Laser Altimeter	17.9
Astrix 1000 IMU	13.5
NED Infrastructure	10.0
Iris V2 Deep Space Transponder	26.0
Dragonfly GECKO Imager	4.5
Total	71.9

Component	Mass (kg)
Laser Altimeter	3.5
Astrix 1000 IMU	4.5
Li-ion Batteries	10.36
Filled N2 Propellant Tank	5.73
8 Thruster Propulsion System	4.0
Iris V2 Deep Space Transponder	1.2
Omni Antennas	0.06
Dragonfly GECKO Imager	0.4
Satellite Frame & NED Hardware	30.0
340 kt NED	188.9
Data Processing Unit	1.0
Total	249.65

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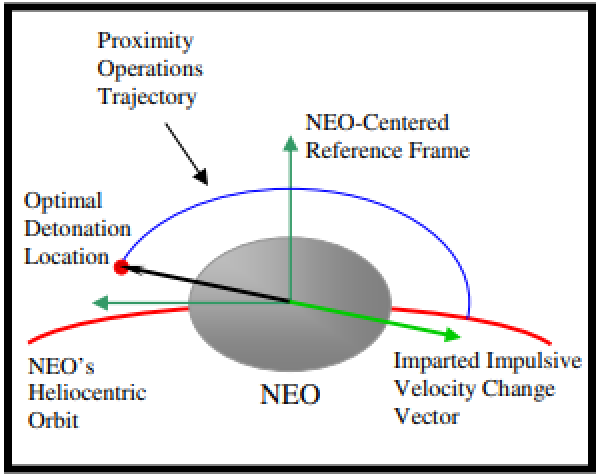
Spacecraft Design: Guidance and Navigation

2023 Proximity Operations Navigation:

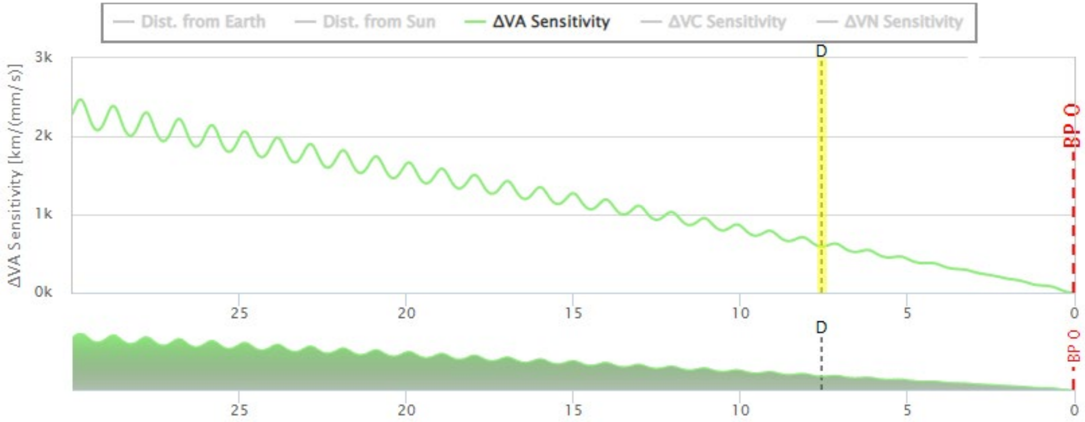
- **Objective:** place spacecraft at optimal s
  and location for detonation
- Required Instruments:
  - Laser Altimeter→Measures distance to a
    equivalent)

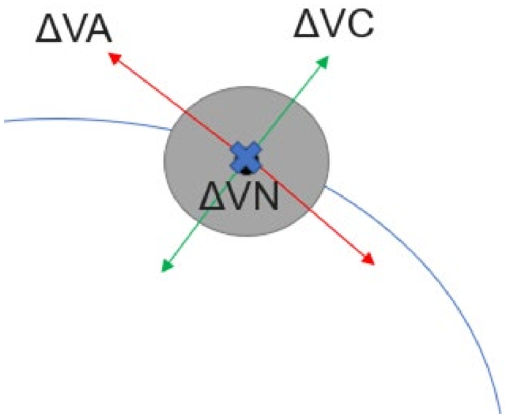


Source: Barbee, Fowler, Davis, Gaylor

Mitigation of 2023 PDC: ΔV sensitivity

- ΔV applied parallel to the velocity vector is 3 orders of magnitude more effective than any other direction
- For this reason, the mitigation mission detonation will be oriented to apply the ΔV in this direction





Source: CNEOS deflection app

Spacecraft Design: Communications

Link	Band	Data-Rate	Link Margin	Antenna
MUFN - DSN	X / KA Band	2 kbps	3.2 dB	2 m HGA
DSN - MUFN	X / KA Band	100 kbps	7.5 dB	11.3 m DSN
BELASat - MUFN	X Band	Up to 250 Mbps	4 dB	Omni
MUFN - BELASat	X Band	Up to 250 Mbps	4 dB	Omni
THEO - MUFN	X - Band	~10 kbps	>3 dB	Omni
MUFN - THEO	X - Band	~10 kbps	>3 dB	Omni
BELASat - THEO	X - Band	~10 kbps	>3 dB	Omni
THEO - BELASat	X - Band	~10 kbps	>3 dB	Omni

[3]: Barbee, Wie, Steiner, Getzandanner, 2013

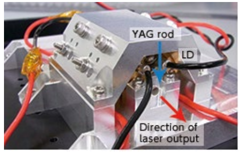
Spacecraft Guidance and Navigation: Component/Capability Suite

MUFN Component	Mass (Kg)	Power (W)
Camera (Optical Link) - ECAM C50	0.4	2.5
Computer Components	1	5
2x Omni Antenna (Radio Link)	0.06	/
Laser Altimeter, IMU	6	20
TOTAL	68.7	187.8

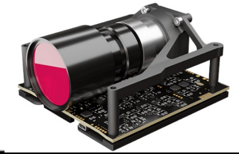
BELASat Component	Mass (Kg)	Power (W)
Camera (Natural Feature Tracking) - GECKO Imager	0.4	4.5
2x Omni Antenna (Radio Link)	0.06	/
Laser Altimeter, IMU	6	20
Computer Components	0.2	1
TOTAL	2.26	26

Astrix 1000 IMU

Hayabusa 2 Laser Altimeter



Dragonfly GECKO Imager



Spacecraft Design: Nuclear Payload Environmental Control and Protection

Requirements:

- Maintain NED at room temperature (nominal spacecraft temperature)
- Protect NED from radiation dosage and possible debris impacts

Strategy:

- Active heating
- External Aluminium whipple shields to protect critical components
  - Whipple shields protect critical components (NED) from high speed debris
  - Extra Aluminium lowers electrical component radiation dose
  - Will use metallic foam casing around NED to prodive whipple shield and radiation dose protection
  - 15 kg allotted for NED protection

Spacecraft Design: Power Subsystem

- MUFN will utilize an RTG for system power needs:
  - Eliminates need for deployable solar arrays which are more susceptible to damage from ejecta
  - Constant power source regardless of sun orientation
  - Provide heating capability if/when necessary
  - Extensive deep space flight heritage (New Horizons)
- Assumed power consumption of 700 W [3] for all subsystems at peak consumption (power budget will be refined going forward)
  - 3x General Purpose Heat Source (GPHS) RTG ~ 5 kg fuel
  - Provides 750 W continuous power

[3]: Barbee, Wie, Steiner, Getzandanner, 2013

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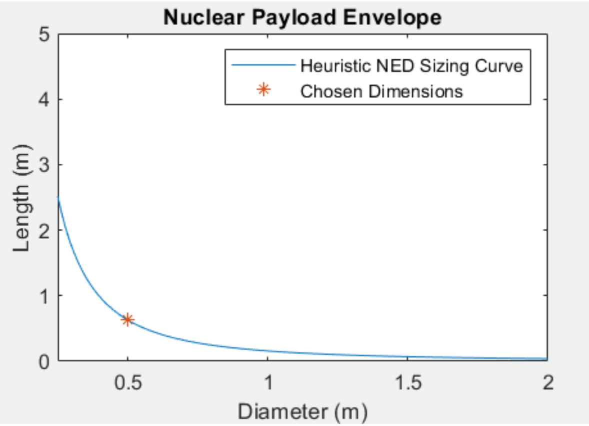
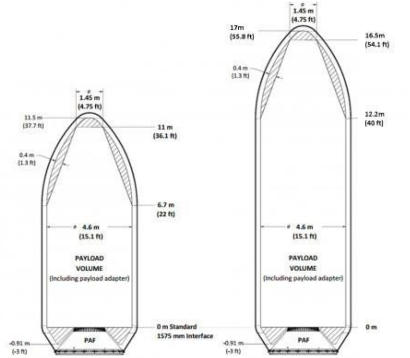
Spacecraft Design: Nuclear Payload

- Standard and extended payload fairings are available from SpaceX for Falcon Heavy
    - Both fairings have an outer diameter of 4.6 meters
    - Standard fairing has a height of 13.2 meters and the extended fairing has a height of 18.7 meters
- Standard fairing sufficient for housing MUFN mitigation bus and four 340 Kt NED-equipped BELASats





From *Falcon User's Guide* (September 2021)

PDC 2023 Proximity Ops: NED Transfer Orbit

MUFN:

- Stationkeeping 1.75 km above 2023 PDC surface - opposite detonation face

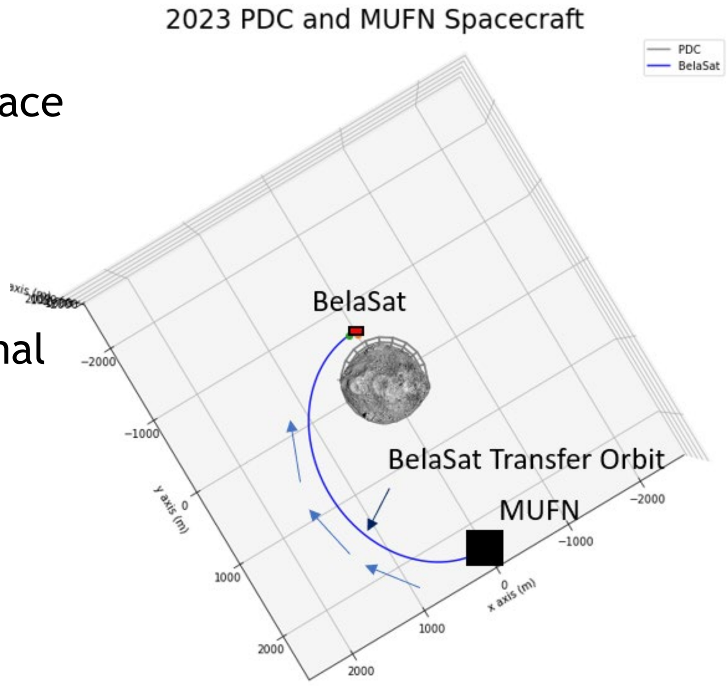
BELASat:

- Will separate from MUFN every 3 days
- 4x separations in total for each MUFN
- Maneuver to elliptical transfer orbit to optimal standoff distance

Effectiveness Assessment

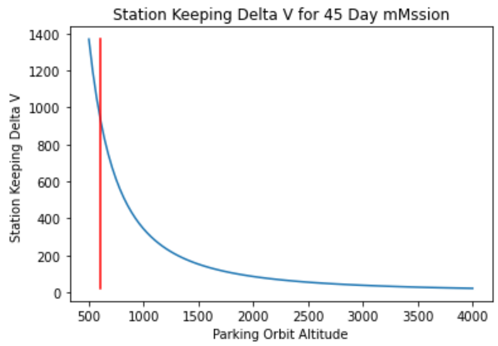
- 2023 PDC is tracked for 3 days by MUFN / THEO

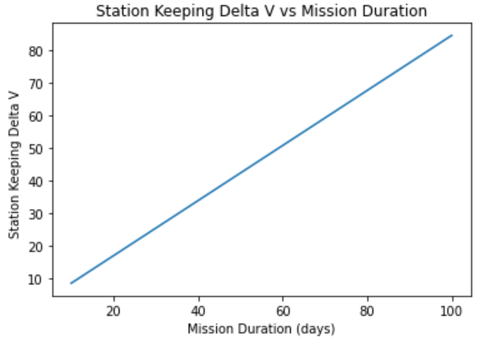
Result: MUFN will be 180 deg out of phase with BELASat at time of detonation - safe



MUFN Stationkeeping Altitude Selection

- THEO will conduct detailed low altitude mapping at altitude of 1.75 km prior to MUFN arrival
  - Precise knowledge of orbital perturbations, surface features (to be used in navigation)
- Lower altitude stationkeeping results in higher mission delta V costs for both MUFN and BELASat
- Lower altitude results in safer position for MUFN to occupy during detonation





Will Select 1.75 km stationkeeping altitude above 2023 PDC for MUFN spacecraft

Proximity Operations ΔV Budget

MUFN:

- Designed for *16 day Mission* (12 day primary, 4 day backup)
- Stationkeeping ΔV: 24 m/s
- Margin for maneuvering: 30% (debris, changing parking orbit, asteroid irregularities)
- Total: 32 m/s

BELASat:

- Pre-detonation maneuver: 0.11 m/s
- Margin for maneuvering: 100% (debris mitigation, change of required standoff distance)
- Total: 0.22 m/s

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Spacecraft Design: Communications

DSN - MUFN link: X-Band

- Positioning data, health data, amring sequence, position imagery

MUFN - THEO link:

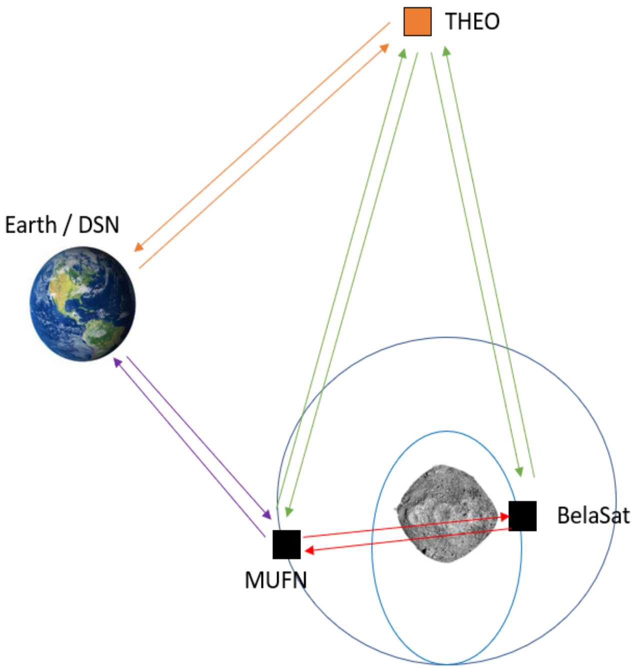
- Positioning data, health data, amring sequence, position imagery

MUFN- BELASat link:

- Positioning data, health data, amring sequence, position imagery

BELASat - THEO link:

- Positioning data, health data, amring sequence, position imagery
- Omnidirectional Antenna



Link Budget Assumptions:

- 3 dB implementation error, 3 dB pointing error 3 dB weather margin (where applicable)
- 3 dB minimum requirement for each link
- Bit Error Rate of 10e-5
- Antenna temperature of 290 K (~room temperature), antenna efficiency of 0.5

Spacecraft Design: Comms Component Suite

MUFN Component	Mass (Kg)	Power (W)
Deep Space Transponder	3.2	15.8
Low Noise Power Amplifier	5	172
Antenna: Omni (2x), 2m HGA	50	/
Wiring + Switches	1	/
TOTAL	68.7	187.8

BELASat Component	Mass (Kg)	Power (W)
Iris V2 Cubesat Deep Space Transponder	1.2	26
Omni - Antenna (2x)	0.06	/
Wiring + Switches	1	/
TOTAL	2.26	26


Credit: NASA


Credit: GomSpace


Credit: GomSpace

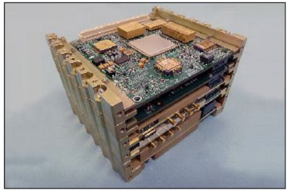


Figure 1: Illustration of the Iris V2 radio (image credit: JPL)
Credit: JPL

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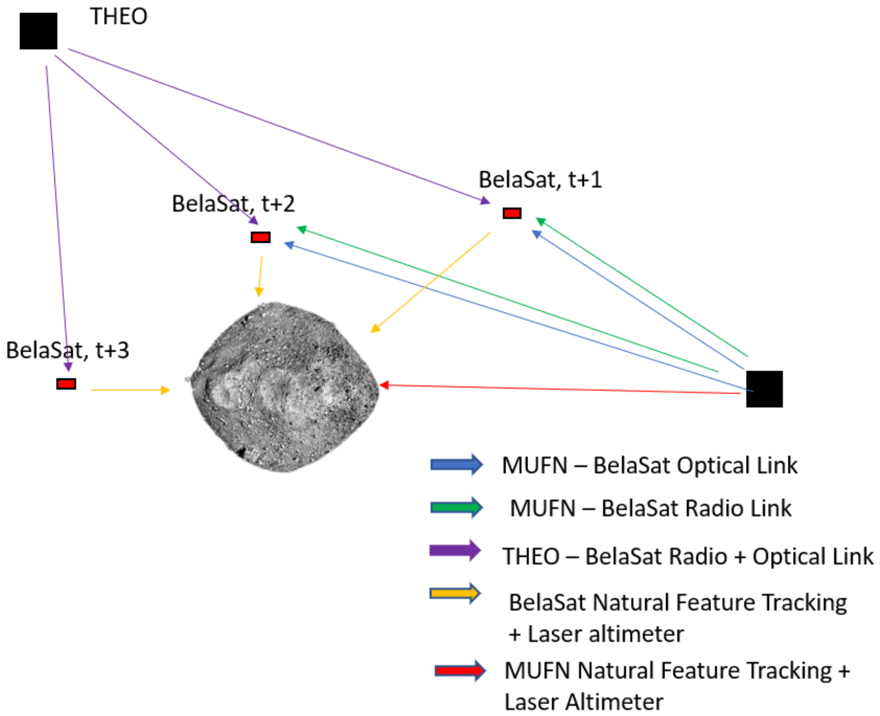
Detonation Maneuver Guidance and Navigation

Radio and visual link between MUFN / THEO to BELASat gives BELASat position relative to 2023 PDC

- Spacecraft RX antennas
- THEO / MUFN Optics

BELASat Natural Feature Tracking + laser altimetry provides additional guidance mechanism

- Shape model uploaded to BELASat prior to mission
- Natural Feature Tracking modeled of OSIRIS-REX capabilities



Spacecraft Design: Subsystem Summary

MUFN Subsystem	Primary Components	Mass (kg)	Power (Watts)
Power	RTG (2)	114	600 (provided)
Propulsion (dry)	Thrusters, Engine, Tanks	267	222
Thermal	Radiators, Mylar blankets	30	100
Separation	4x RocketLab mkII Motorized Lightbands	40	30
Communications	2x omni antenna, 1x 2m HGA	69	188
Guidance and Navigation	Camera, IMU, Star tracker, sun tracker, computer	8	30
TOTAL		528	500

Spacecraft Design: Propulsion

- Single stage MMH-MON25 vehicle
  - A primary R-42 and a backup R-42 derivative engines (Aerojet-Rocketdyne) with 890 N of thrust per engine
  - This engine will be altered to increase expansion ratio, improve specific impulse in a vacuum (340s target), and optimize for the adjustment from MON3 to MON25
  - Helium pressurized hypergolic system
    - Only requires power to valves (90 watts), no turbopumps
  - Tanks are based on Ariane Group Model OST 25/3
    - 331 liter bipropellant Tank designed for hydrazine, MMH, and MON
    - Tank length will be adjusted to accommodate a 30% increased in volume
- 8 MR-103J 1N hydrazine monopropellant-catalyst pulsed thrusters and 8 backups
  - Configuration enables yaw, pitch, and roll control
  - Heaters and values require power (132 watts total)

Spacecraft Mass Summary

Subsystem	***Mass (kg)***	***Notes***
Payload Mass (kg)	1000	Multiple NED devices, bus, comms, computers, and sensors
Fuel Mass (kg)	1600-2900	Spread across all oxidizer and propellant tanks
Inert Mass (kg)	528	Includes: Tanks, engines, thrusters, RTGs, Helium, structure, hydrazine, and thermal control
Total (kg)	3000-4500	LV capabilities vary from 3300 to 7600
ΔV Achieved (km/s)	2.3-3.4	Target ΔV is 2-3.5 km/s. This includes a 20% mass design reserve on the spacecraft.

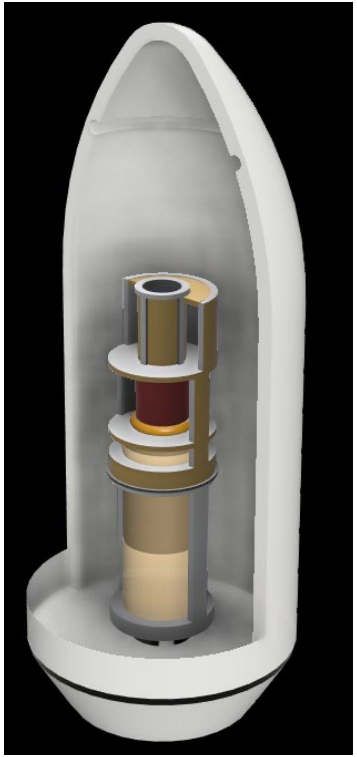
Spacecraft Design: Vehicle Stackup

Properties	Value	Notes
Oxidizer (MON-25) Tank Height (m)	1.7	2 tanks; Capsular design; .75 m diameter; Design for max capacity
Propellant (MMH) Tank Height (m)	1.7	2 tanks; Capsular design; .75 m diameter; Design for max capacity
Helium Tank Radius (m)	0.4	2 tanks; Spherical design; Design for max capacity
Hydrazine Tanks Radius (m)	0.15	2 tanks; Spherical design; Design for max capacity
Engine Height (m)	0.7	2 engines, a primary and backup
NED Bus Height (m)	1.5	Bus and NEDs
Central Truss Structure Height (m)	1.8	Tanks built around the central truss structure and the engines below
Total Height (m)	4	Tanks are arranged a ring around the central truss structure and the NED bus is stacked on top. Maximum Width is 3 meters

MUFN Mitigation Vehicle Redesign

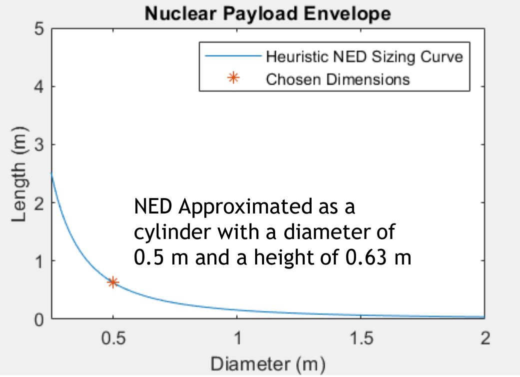
PDR MUFN Vehicle:

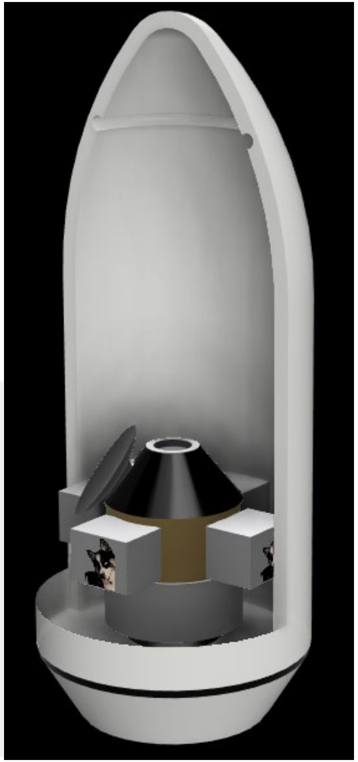
- 2 stages
- Cylindrical fuel tanks
- Vertical stack design
- Single 2.7 Mt NED



CDR MUFN Vehicle Bus:

- 1 stage
- Capsular and spherical fuel tanks
- ESPA ring for BELASat mounting
- Four 340 kt NEDs





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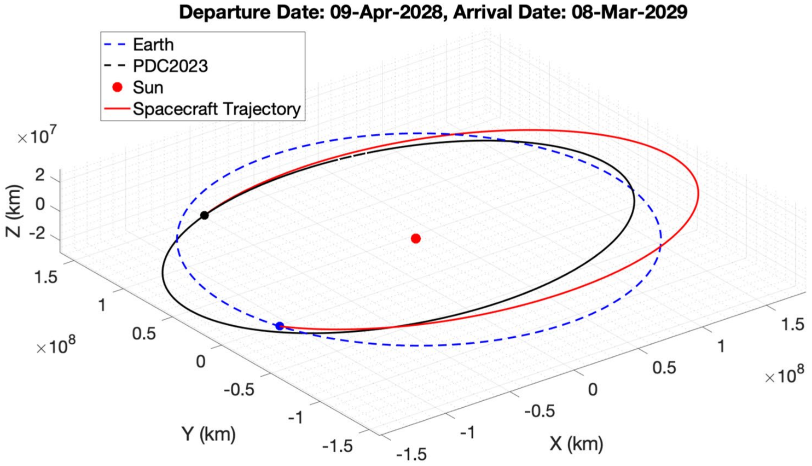
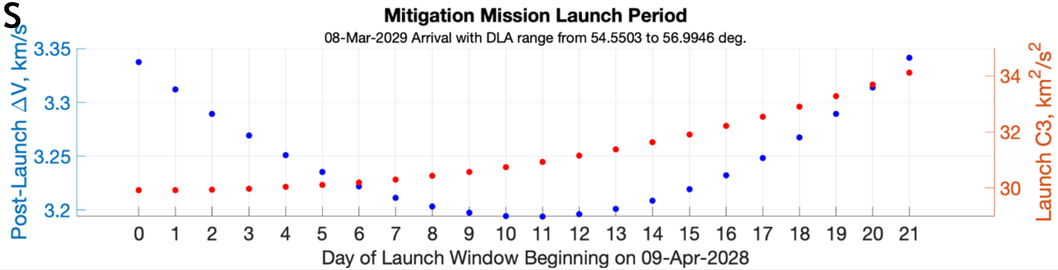
Trajectory Design

- There are 5 launch periods in 2028 and 2029
  - April 2028, May 2028, May 2029, September 2029, October 2029
  - 3 Launches are required
- All launch periods have a DLA < abs(57°), C3 < 60 km2/s2, DeltaV < 3.5 km/s





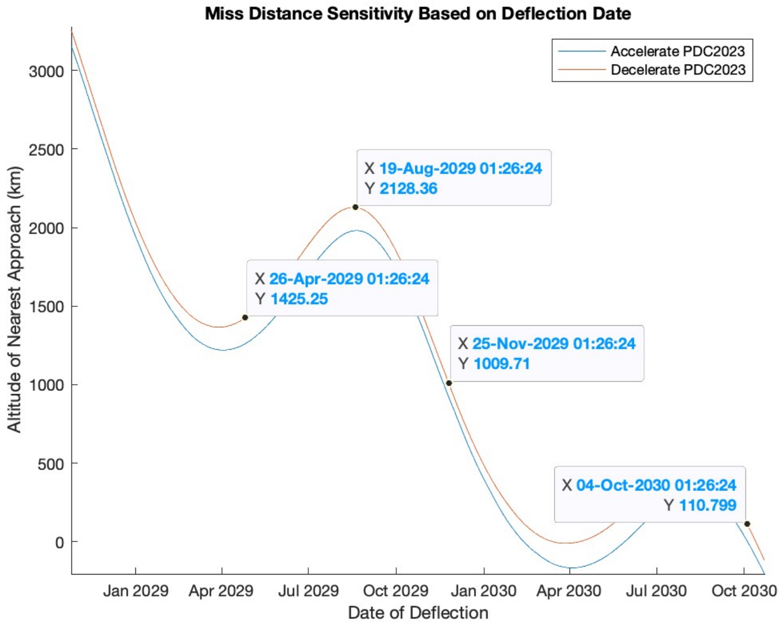
PDC Arrival Deep Space Maneuver

- Deep Space Maneuvers vary depending on the trajectory
  - Below is the estimated variation in parameters depending on the required ΔV for the selected trajectory

Trajectory Description	Transfer Time	Burn Time (min)	ΔV (km/s)
Deep Space Maneuver 1	2 weeks prior to intercept	20-40	1.0-2.0
Deep Space Maneuver 2	2 days prior to intercept	10-20	0.5-0.75
Deep Space Maneuver 3	2 hours prior to intercept	8-17	0.5-0.75
Arrival Maneuver	10 minutes prior to intercept	1	0.1

Deflection Mission Simulation

- 9 body deflection simulation applying single impulse at a given date
- Deceleration is always superior to acceleration in this collision deflection scenario
- Arrival Date of 26-Apr-2029
  - Deflecting immediately results in 1430 km altitude flyby
- Primary Deflection period 18 weeks after arrival
  - 2120 km altitude flyby
- 25-Nov-2029 Last chance to maintain deflection above 1000 km
- 04-Oct-2030 last deflection date to avoid atmospheric interface*
- Used this simulation to optimize the detonation periods



*This analysis does not include use of single contingency NED

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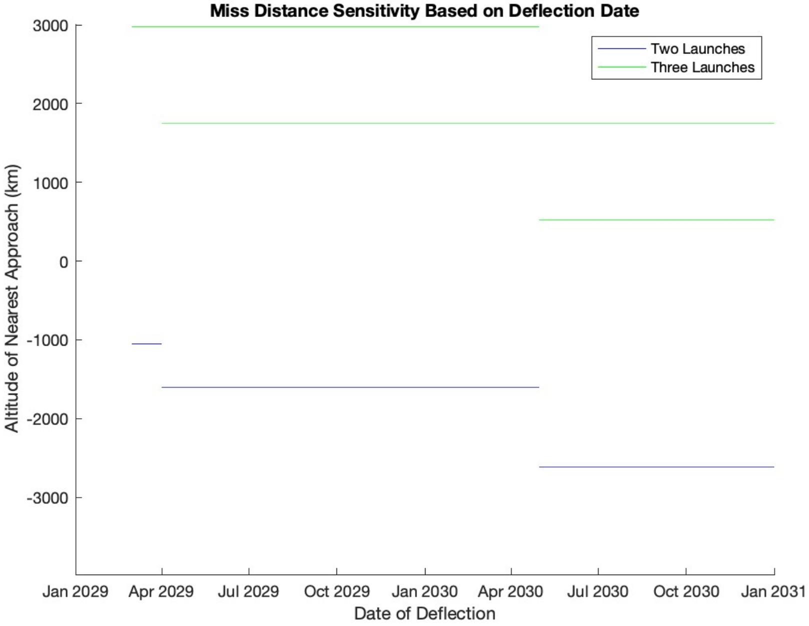
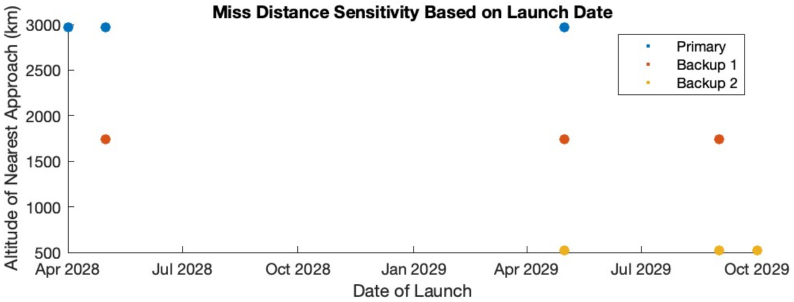
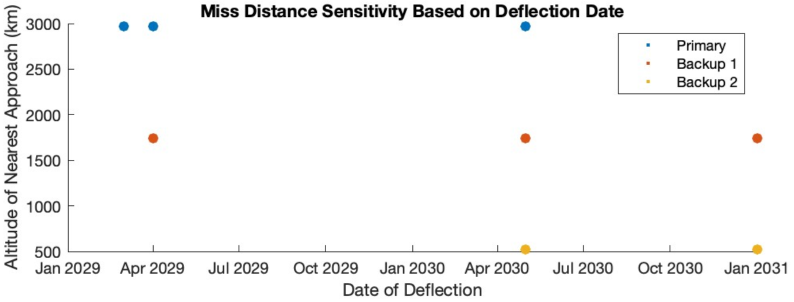
Deflection Mission Margin

- 9 body deflection simulation applying multiple small impulses





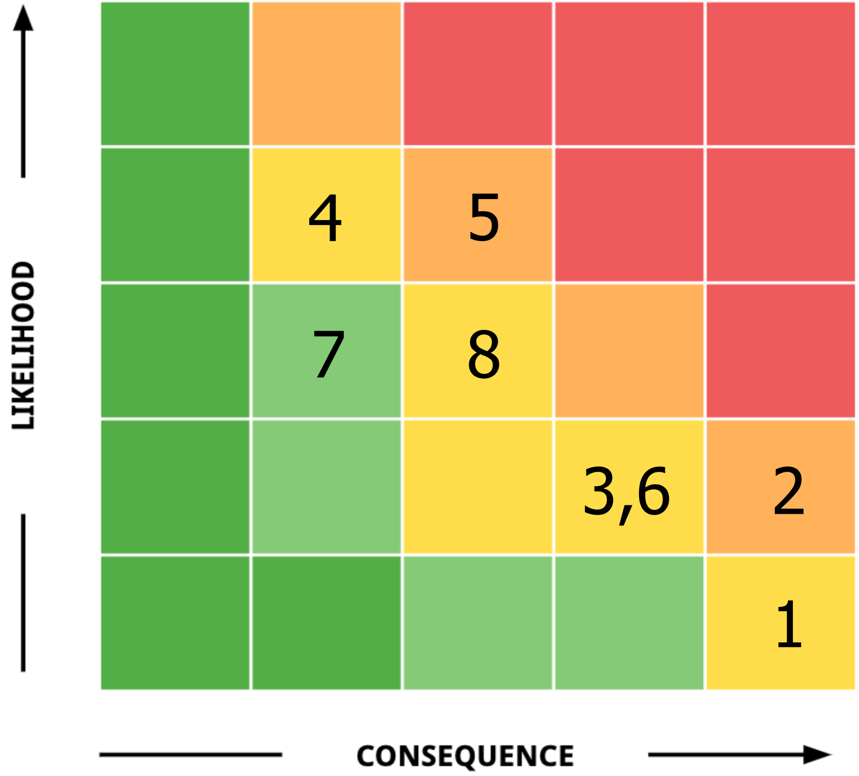


Launch Vehicle and Spacecraft Performance

- Falcon Heavy selected for all 5 launch periods
- Performance based on nominal C3 capability with variable penalties for DLA between 28 and 57 degrees
- The spacecraft and Falcon Heavy were simulated such that the Falcon Heavy may only provide the C3 and the spacecraft may only supply the rendezvous dV.

C3(°)	DLA(°)	dV (km/s)	Prop. Mass (kg)	Total Mass (kg)	dV Mass Margin (%)	Max LV Mass (kg)	LV Mass Margin (%)
19.0	52.7	3.4	2890.0	4524.3	80.0	7632.1	59.3
56.0	49.9	2.3	1620.0	3254.3	80.0	3349.2	97.2
34.0	57.0	3.4	2890.0	4524.3	80.0	5490.6	82.4
26.0	56.1	3.2	2630.0	4264.3	80.0	6542.4	65.2
24.0	57.0	3.2	2630.0	4264.3	80.0	6808.9	62.6

Timeline and Risk Chart



Risk #	Description
1	Launch vehicle failure
2	Critical spacecraft component failure
3	THEO failure
4	Budgetary overruns
5	Mission timeline difficulties
6	2023 PDC exceeds 90th percentile
7	Policy difficulties
8	Accidental asteroid disruption

MUFN Future Work

1. Thermal Management System Analysis: refine active and passive cooling system design
2. NED Environmental Protection refined design
3. Refine launch periods and contingency launch results
   a. Current launch period estimates are conservative and can be expanded
   b. Combine the DLA, C3, and dV into day-by-day parameters to be input into the rocket model
      i. At present the maximum for each value across the launch period is used
   c. Identify additional launch days and identify additional launches that can occur in extended launch periods based on Falcon Heavy launch cadence