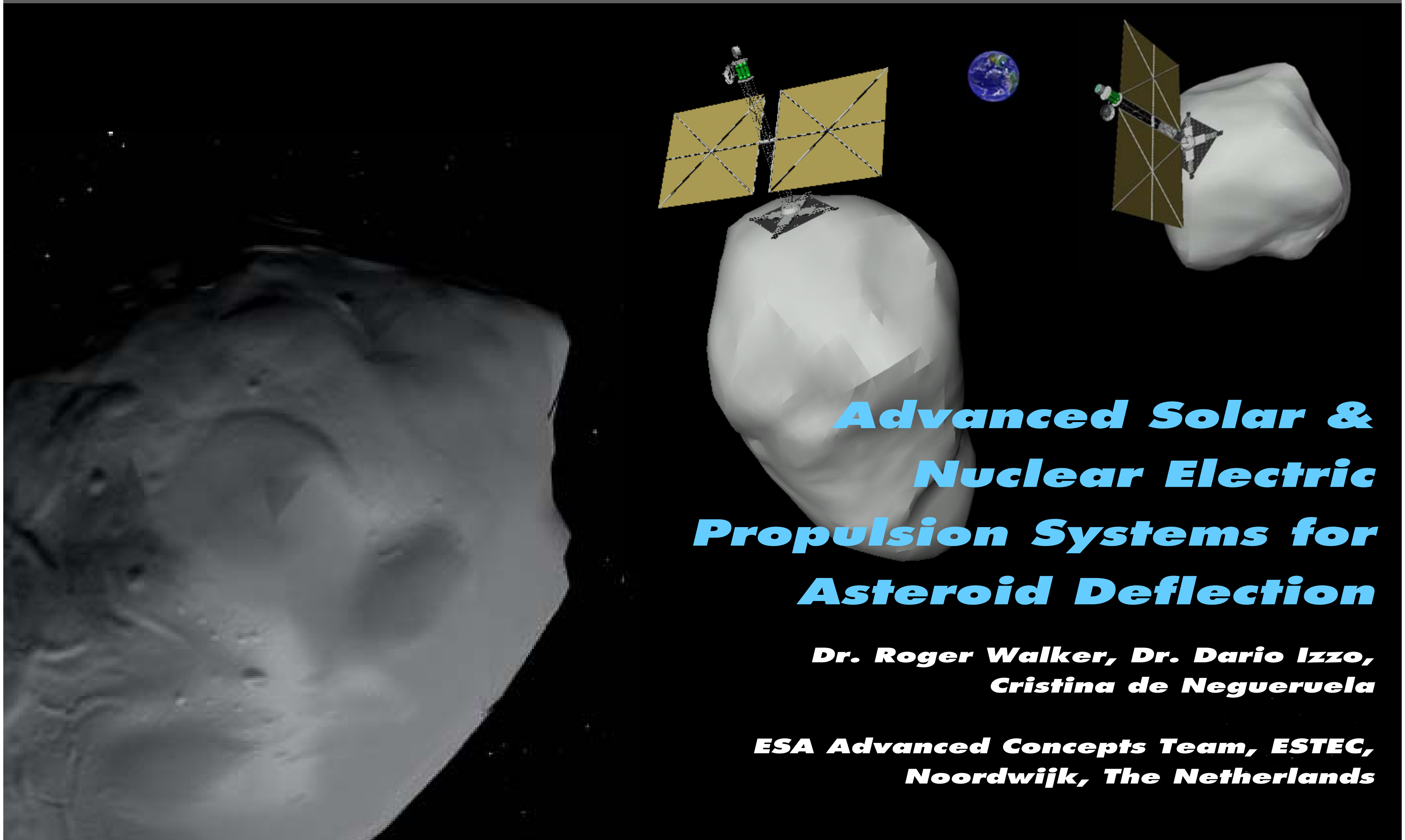
ADVANCED CONCEPTS TEAM



Advanced Solar & Nuclear Electric Propulsion Systems for Asteroid Deflection

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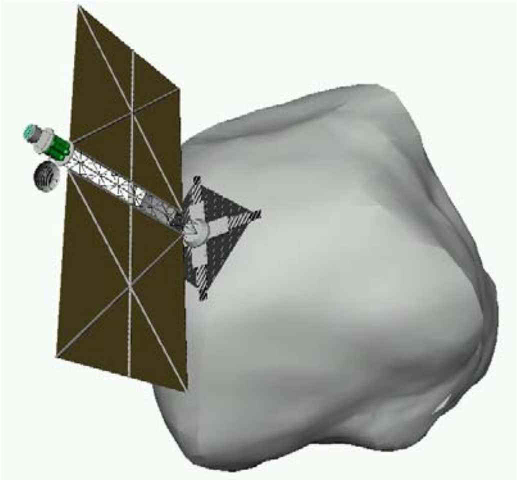
Overview

- Introduction
- Electric Propulsion Deflection
  - EP Asteroid Deflection Formula
  - Mission Model Development
  - Asteroid Deflection Scenario Definition
  - System Trade-offs & Optimisation Analysis
  - Minimum Required Warning Times
  - NEP/SEP System Preliminary Design Concepts
- Kinetic Energy Impactor Deflection Using EP
  - KE Impact Impulsive Asteroid Deflection Formula
  - Low-Thrust Trajectory Optimisation
  - Performance Trade-off vs. EP Deflection
- Conclusions



Introduction

- Near Earth Asteroids (NEAs) pose an impact hazard to Earth
- Low probability, high consequences, similar to other natural hazards
- Space technology is reaching a sufficient level for a deflection capability within the next decades
- Time to start considering the options
- NEA deflection options
  - High-energy impulsive: K.E. impactors (chemical or electric), nuclear stand-off blasts
  - Low-energy long-duration: surface ablation via laser or solar concentrator, mass drivers, surface-attached propulsive devices
- Most attainable in the nearer term: kinetic energy interceptors and surface attached propulsive devices

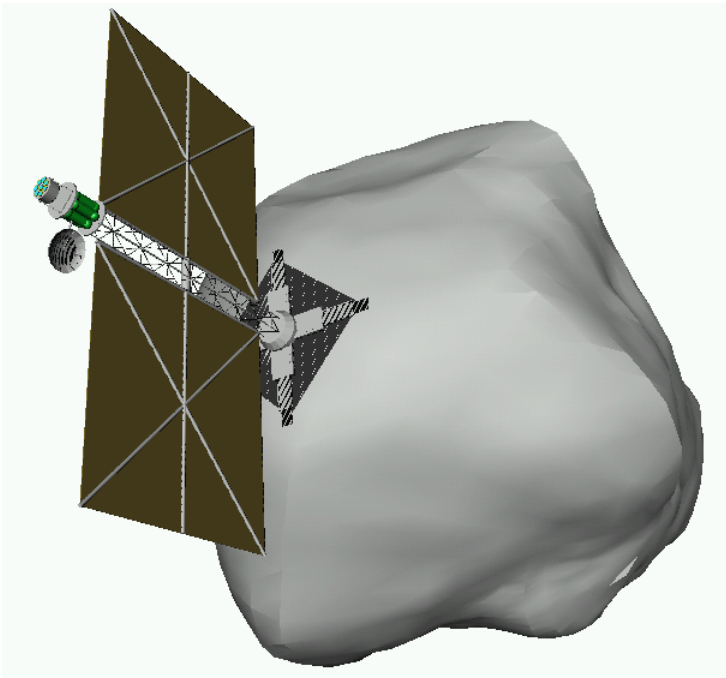
Introduction

- Electric propulsion deflection
  - Rendezvous, land and push technique
  - Challenges: landing/attachment/attitude control of irregular aggregated rotator
  - Benefits: inherent controllability, flexibility, universal, no fragmentation

- Kinetic energy impactor deflection with EP
  - EP used to put impactor spacecraft on high eccentricity heliocentric intercepting trajectory -> very high impact velocity, momentum transfer & impulsive delta-V
  - Challenges: guidance navigation & control to hit target centre of mass at hypervelocity, uncertainties in momentum transfer due to asteroid internal structure
  - Benefits: potential for high deflection performance, efficient use of propulsive energy, no complex close proximity operations or surface interactions

- Both methods require high mission delta-V, moderate thrust (N-level), multi-ton spacecraft
  - High-power & specific impulse electric propulsion systems are the enabler
  - Nuclear fission reactors or Large lightweight solar arrays for power

Electric Propulsion Deflection



Asteroid Deflection Formula

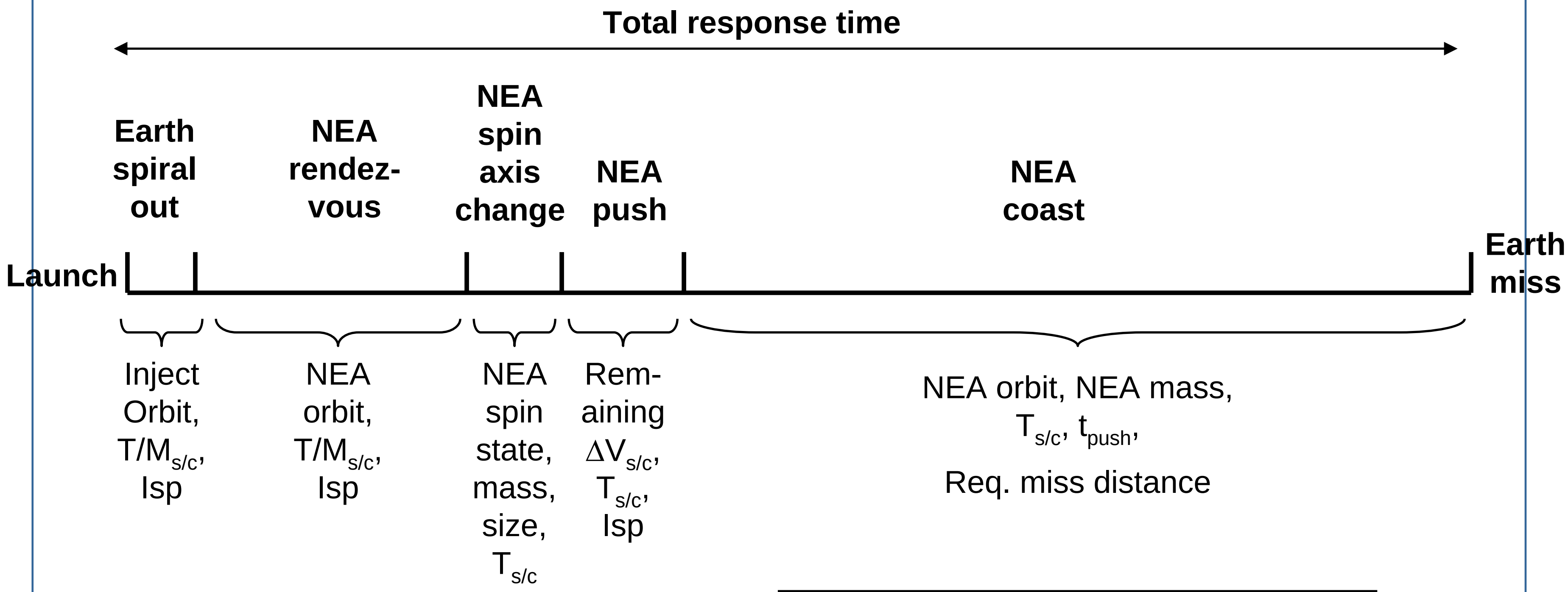
- Introduced in paper AAS 05-141 “On The Deflection Of Potentially Hazardous Objects ” by D. Izzo
- Proven to accurately predict the miss distance induced by the long-duration low-thrust EP deflection method

$$d_{\min} = \frac{3a\gamma V_{Earth}}{\mu} \int_0^{t_p} (t_s - t)\vec{v} \cdot \vec{A} dt$$

$d_{\min}$	minimal distance between the asteroid and the Earth
a	semi-major axis of the asteroid's orbit
γ	non dimensional parameter, depends on encounter geometry
V_{Earth}	Earth velocity at encounter
μ	gravitational parameter of the Sun
t_s	time before impact the strategy is started
t	time counted from ts
$\vec{v}$	asteroid velocity along its unperturbed orbit
$\vec{A}(t)$	deflection strategy applied to the asteroid

Mission Drivers

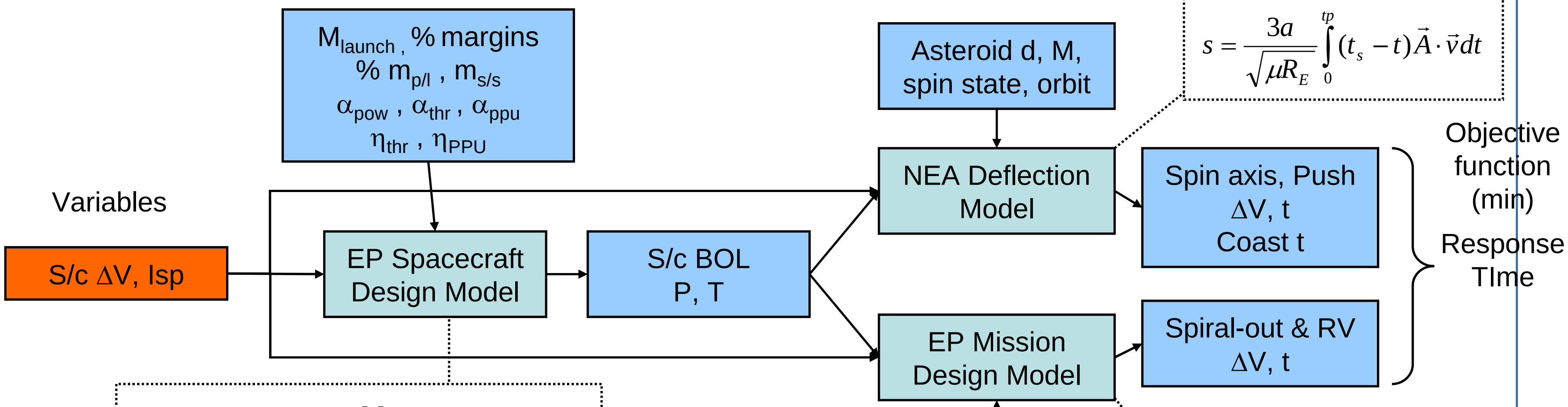
- Mission phases



Trade-off between spacecraft thrust, ΔV, Isp

Mission Model Development

- EP Deflection Toolbox developed by ESA ACT



$$t_{spin} = \frac{\theta_{spin} I_{ast} \omega_{ast}}{r_{ast} T_{sc}}$$

$$s = \frac{3a}{\sqrt{\mu R_E}} \int_0^{tp} (t_s - t) \vec{A} \cdot \vec{v} dt$$

$$P_{BOL} = \frac{M_{p+p}}{(\alpha_{thr} \eta_{ppu} + \alpha_{ppu}) N_{T,per} + \alpha_{pow}}$$

$$M_{p+p} = (M_{wet} - M_f)(1 - m_{s/s} - m_{p/l}) - m_{\tan} M_f$$

$$M_f = M_{wet} \left(1 - e^{-\Delta V / g_o I_{sp}}\right)$$

$$T_{BOL} = \frac{2\eta_{thr}}{g_o I_{sp}} \eta_{ppu} P_{BOL}$$

$$M_{spiral} = M_{wet} \left(1 - \exp\left(\frac{\sqrt{\mu_E}}{g_o I_{sp}} \left(\frac{1}{\sqrt{a_2}} - \frac{1}{\sqrt{a_1}}\right)\right)\right)$$

$$t_{spiral} = \frac{M_{spiral} g_o I_{sp}}{T_{BOL}}$$

M_{RV}, t_{RV} computed by DITAN low-thrust software

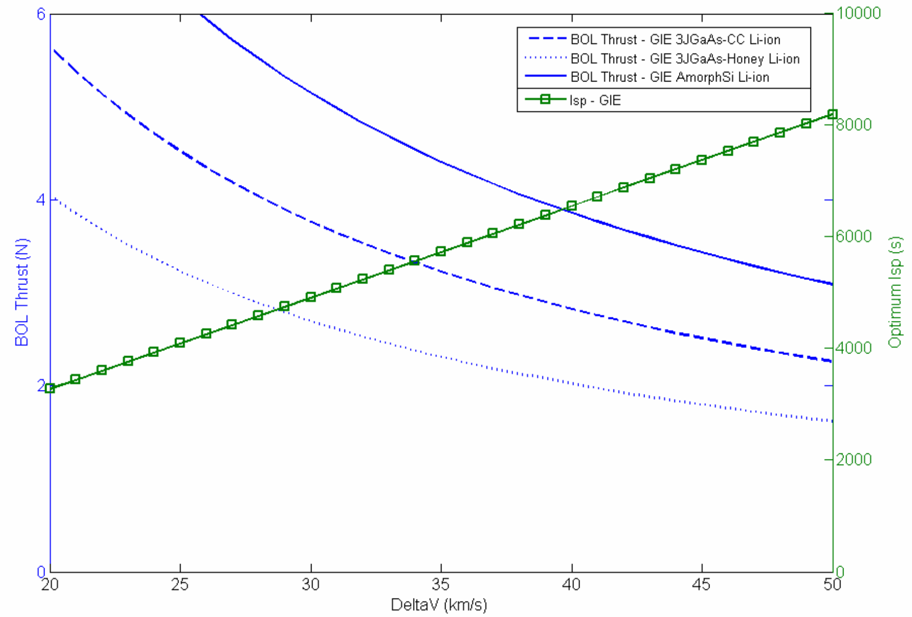
Mission Constraints & Options

Mass budget			
Launch mass	20,900 kg	Proton K into LEO	
Margins	3% launcher	20% system	
Mass fractions	Payload 10% (dry)	Structure 25% (dry)	Tanks 15% (fuel)
Propulsion subsystem			
(1) Gridded Ion Engines	Isp 3000-10000s	19-62 kW/N	7 kg/kW
(2) Hall Effect Thrusters	Isp 1500-3000s	12-24 kW/N	6 kg/kW
Power subsystem			
Solar electric			
(1) 3J GaAs, honeycomb array	33% efficiency	27 kg/kW	
(2) 3J GaAs C-C array	33% efficiency	14 kg/kW	
(3) Amorph Si Thin Film array, Cbooms	12% efficiency	4 kg/kW	
Secondary Power :	Li-ion Batteries (1hr, full thrust)	8.5 kg/kW	
Nuclear Electric			
Specific mass as a function of power	40-50 kg/kW for Power >80 kW		

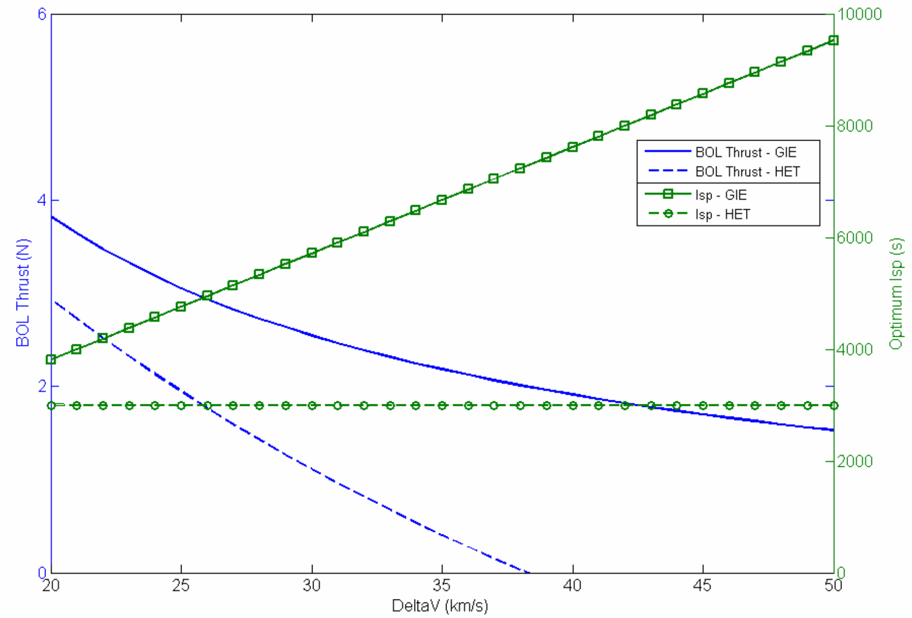
System Trade-off Analysis

- Maximise thrust for the given launch mass & ΔV range
  - Selected Gridded Ion Engines for SEP and NEP propulsion systems
  - Selected Amorphous Si Thin Film solar arrays for SEP power system

SEP



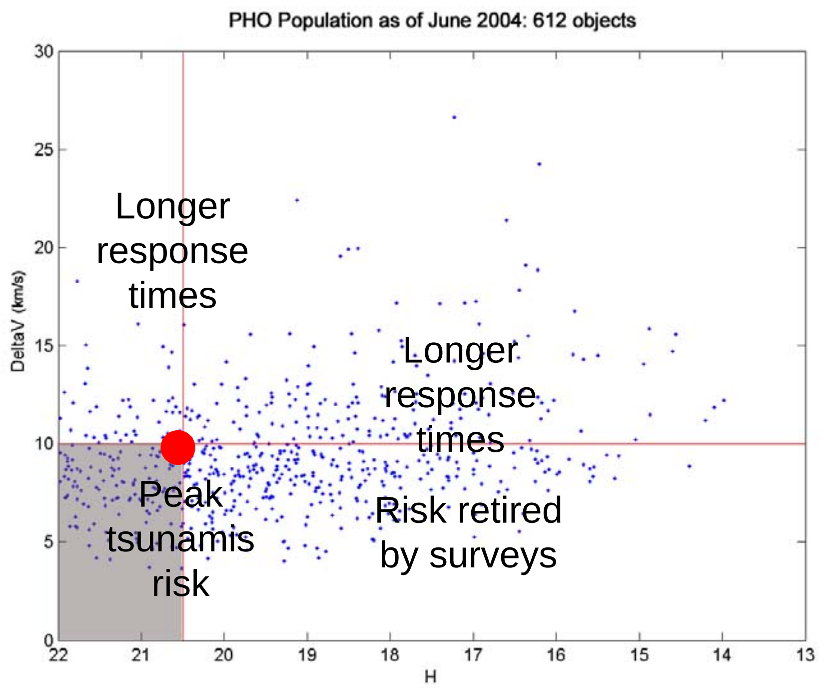
NEP



Spacecraft wet mass: 16220 kg into LEO (w/o margins)

Asteroid Deflection Scenario

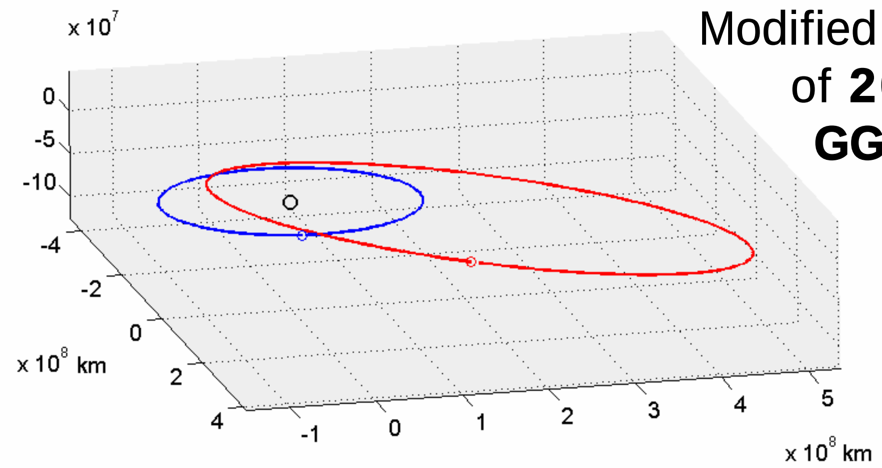
- Capability requirements
  - Object size: Deflection of objects < 200-300m diameter
  - Accessibility: rendezvous Delta-V <10km/s (impulsive)
  - Deflection miss distance: 10,000km minimum



Asteroid Deflection Scenario

Semi-major axis, a (AU)	2.143
Eccentricity	0.709
Perihelion, q (AU)	0.623
Aphelion, Q (AU)	3.66
Inclination, I (°)	10.12
Arg. perihelion, ω (°)	95
Ascending node, Ω (°)	13.2
Period (yrs)	3.14
Synodic period (yrs)	1.45
Min. orbit intersect (km)	0

Diameter (m)	200
Density (g/cm^3)	2.4
Mass (Mt)	10
Rotation period (hrs)	9
Rotation pole to orbit plane (°)	40



Modified orbit of **2003 GG21**

Asteroid Deflection Scenario

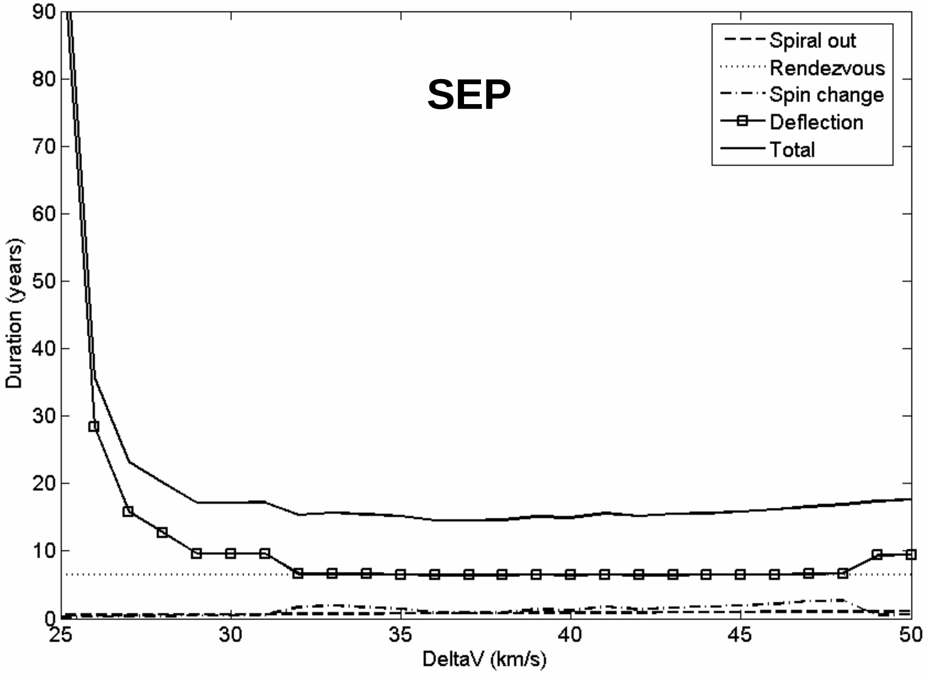
- Asteroid attitude control
  - De-spin for 3-axis control infeasible due to high NEA moment of inertia
  - Use Spin axis control strategy (continuous thrust applied at rotation pole)
  - Time to re-orient spin axis prior to push: $t_{spin} = \dfrac{\theta_{spin} I_{ast} \omega_{ast}}{r_{ast} T_{sc}}$
- Miss distance

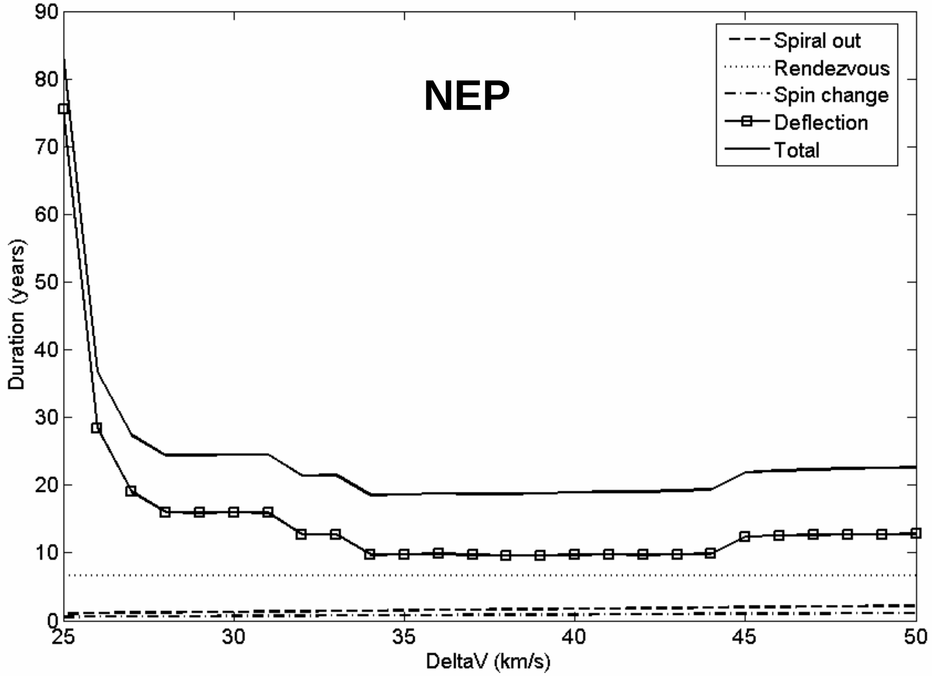
$$s = \frac{3a}{\sqrt{\mu R_{Earth}}} \int_0^{t_a} (t_s - t) \vec{A} \cdot \vec{v} \, dt$$

  - Local maxima at perihelion when acceleration & velocity vectors aligned
- Deflection strategies
  - Inertially-fixed thrust vector (spin axis in-plane)
  - Naturally-precessing thrust vector (spin axis in-plane)
  - Simultaneous ‘torque and push’ (spin axis out-of-plane)
- Inertially-fixed strategy with thrust aligned with perihelion velocity

System Optimisation

- Total response time vs spacecraft ΔV





Optimum: ΔV=38 km/s, Isp=6215 s

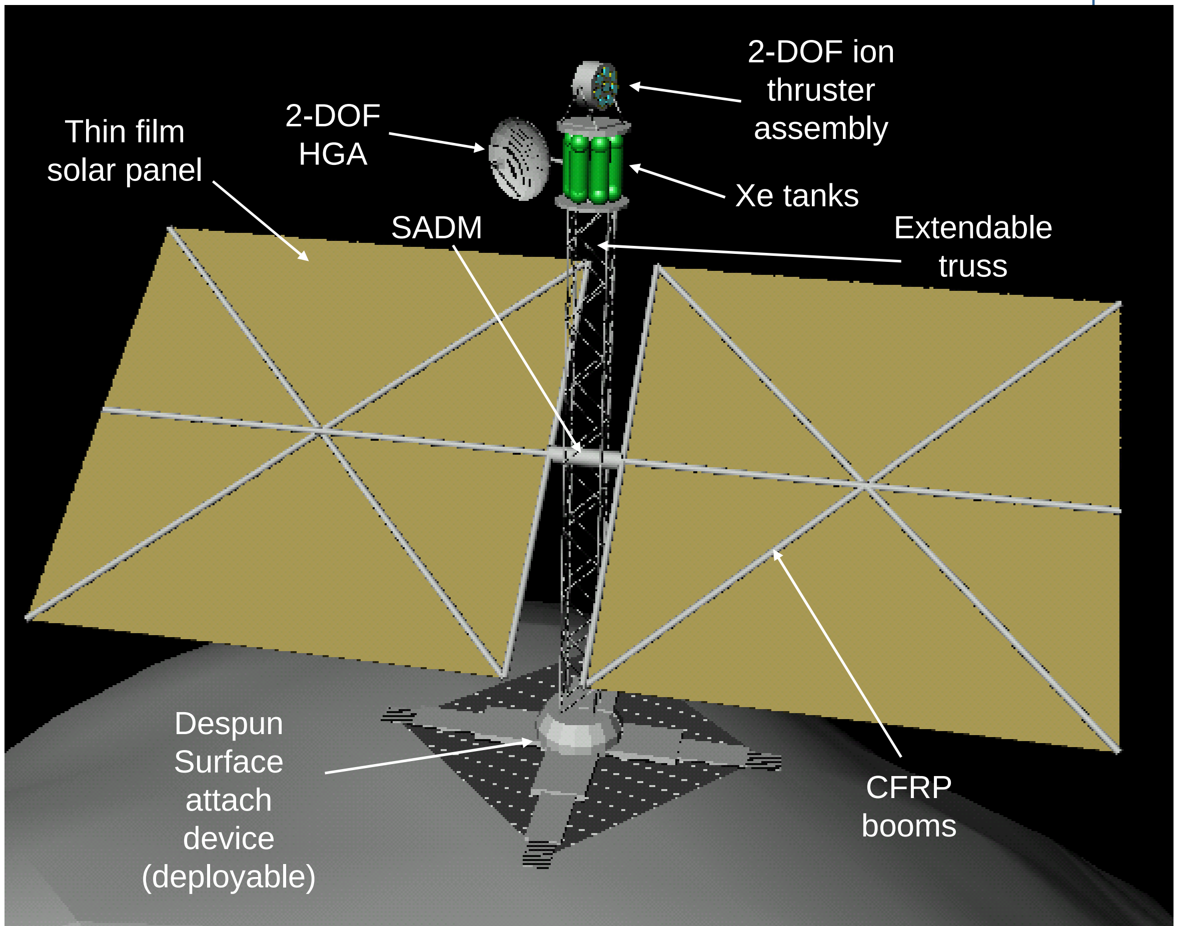
Total response time: 14.6 years

Optimum: ΔV=38 km/s, Isp=7235 s

Total response time: 18.7 years

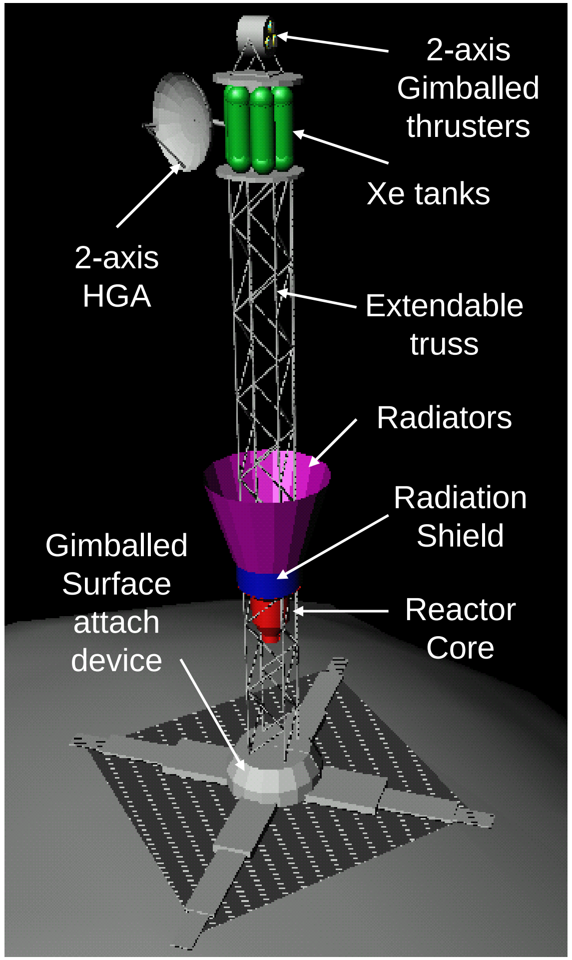
Preliminary SEP Spacecraft Design

- Mass
  - 16220kg wet, 7522kg fuel, 8696kg dry
  - 2112kg power, 2060kg propulsion
- Dimensions
  - 35m deployed length
- Power s/s
  - Power 165kW (1AU), 295kW perihelion, 19kW aphelion
  - 1400 m^2 solar array area
- Propulsion s/s
  - 4N thrust (1AU), Isp 6215s, 8x40kW ion thrusters, 6 Xe tanks
- Comms
  - X/Ka dual band, 2.5m HGA >100kbps @ 2AU range (Ka)
- ACS
  - 2-DOF gimballed main thrusters, reaction wheels for fine pointing



Preliminary NEP Spacecraft Design

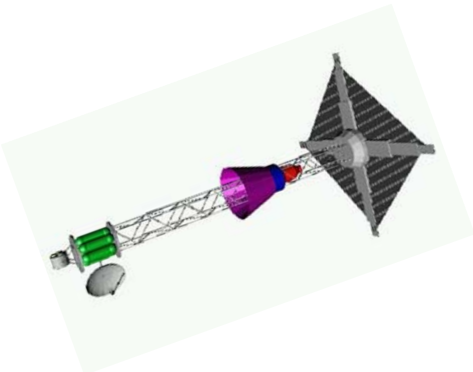
- Mass
  - 16220kg wet, 6723kg fuel, 9495kg dry
  - 4125kg power, 663kg propulsion
- Dimensions
  - 35 m deployed length
- Power s/s
  - Power 95kW (constant), 65m^2 radiator area
- Propulsion s/s
  - 2N thrust (constant), Isp 7235s, 3x40kW ion thrusters, 6 Xe tanks holding 900L each
- Comms & ACS
  - As SEP
- Payload
  - Imagers, radar tomographer, IR spectrometers
- Surface attach device
  - 1m Helical screw into regolith, Long multi-jointed legs, flexible webbing

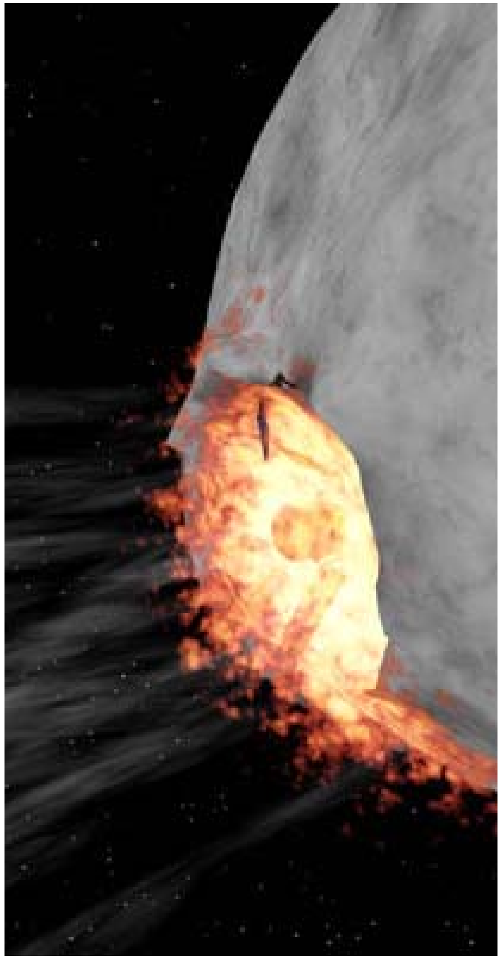


Study Conclusions

- Maximum deflection capability for EP deflection assessed
  - 15-20 ton spacecraft launched into LEO
  - 100 kW-class power levels, N-level thrust
  - 10 megaton asteroid (approx. 200 m size), 10,000 km miss distance
  - 10-20 years response time depending on asteroid orbit & rotation
  - Effective considering typical warning times of 10-50 years
- Comparison between SEP and NEP
  - Shorter response times for SEP due to low asteroid perihelion
  - Expected to be much closer for perihelion close to Earth
- Technology needs
  - Large gridded ion thrusters & PPUs, 40-50 kW, 6000-7500 s specific impulse
  - Large 2-DOF gimballed ion thruster assembly
  - Large deployable amorphous silicon thin film arrays with high packing density
  - Nuclear reactor system, 100 kWe
  - Lightweight, long extendible truss structures
  - Large deployable articulated surface attachment devices with central helical screw

Kinetic Energy Impact Deflection Using EP





Asteroid Deflection Formula

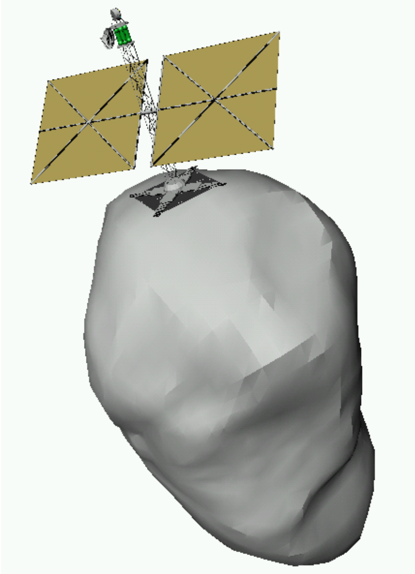
- Impactor vehicle on an interplanetary intercept trajectory
- Assuming a perfectly inelastic impact: $m\vec{v}_{s\backslash c} + M\vec{v} = (m+M)(\vec{v} + \Delta\vec{V})$
- And introducing the impact efficiency η (depends on surface/internal properties of the asteroid) – (we assume a very conservative η =1, i.e. no ejecta)

$$\boxed{d_{\min} = \eta\gamma \frac{3aV_{Earth}}{\mu} \frac{mt_s}{m+M} \vec{v} \cdot \vec{U}}$$

$d_{\min}$	minimal distance between the asteroid and the Earth
a	semi-major axis of the asteroid's orbit
γ	non dimensional parameter, depends on encounter geometry
V_{Earth}	Earth velocity at encounter
μ	gravitational parameter of the Sun
t_s	time before impact the strategy is started
η	impact efficiency
$\vec{v}$	asteroid velocity along its unperturbed orbit
$\vec{U}$	relative velocity vector
m	impactor mass
M	asteroid mass

Trade-off Analysis vs EP Deflection

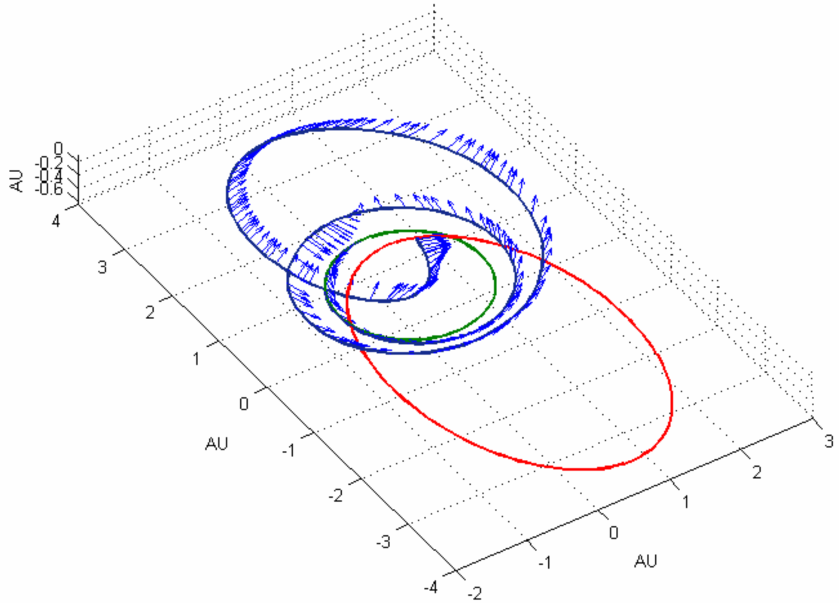
- Based on previous ACT internal study on “Concepts for Near-Earth Asteroid deflection using spacecraft with advanced nuclear and solar electric propulsion systems” published in JBIS 2005
- Asteroid
  - 2003 GG21
  - Mass 10^{10} kg
  - Diameter 200 m
  - Density 2.4 g/cm3
- Spacecraft
  - Nuclear Electric propulsion spacecraft, T=2N, I_{sp}=6700s
  - Wet mass 18000 kg
  - C3=0 reached after spiral out phase (2000kg of fuel used)
- Deflection Strategies
  - Kinetic impactor with EP
  - EP rendezvous land and push



- For each strategy: optimisation of the heliocentric trajectory and assessment of the overall deflection capabilities

Kinetic Impactor with EP Results

- Objective function to maximise: $\vec{v} \cdot \vec{U} = v_{ast}^2 - \vec{v}_{ast} \cdot \vec{v}_{s\backslash c}$



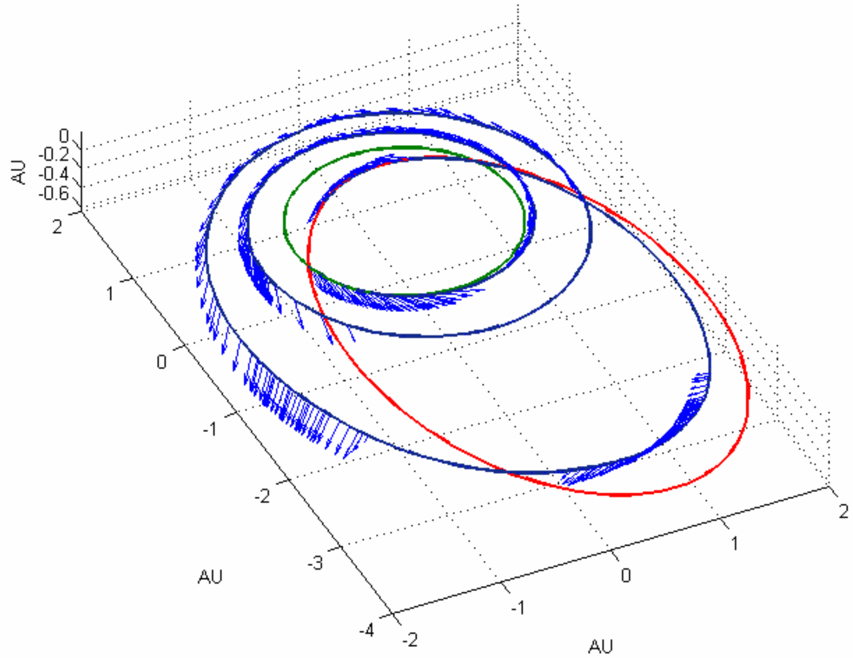
Kinetic Impactor Scenario		
Departure Epoch (Modified Julian Date)	6202	MJD
Interception Epoch	7993	MJD
Avoided Impact Epoch	9210	MJD
Heliocentric phase Duration	4.9	years
Final Mass	11852	kg
Final $\vec{v} \cdot \vec{U}$	1630	$km^2 \backslash s^2$
Final $\vec{v}$ (heliocentric)	[5, 42, 1.5]	km\s
$\vec{U}$ (heliocentric)	[47, 30.7, -2.1]	km\s
Obtained miss-distance	43851	km
Minimal Earth-Sun distance	.22	AU

- Possible improvement maximising $mt_s \vec{v} \cdot \vec{U}$

EP Deflection Results

- Objective function to maximise: $\int_0^{t_p} (t_s - t)\vec{v} \cdot \frac{\vec{T}(t)}{M} dt$
  (related to final mass at rendezvous)
- Heliocentric transfer optimised respect to mass and final relative velocity 0 to achieve capture

Long Duration Thrust Scenario		
Departure Epoch	5937	MJD
Rendezvous Epoch	8035	MJD
Avoided Impact Epoch	9210	MJD
Heliocentric Phase Duration	5.7	years
Final Mass	12985	kg
Obtained miss-distance	3297	km



Study Conclusions

- Assuming an advanced spacecraft design, an optimisation of the heliocentric trajectories has been performed for both a kinetic impactor and a “rendezvous and push” missions, both powered by EP
- The resulting miss-distance has been evaluated via the derived asteroid deflection formulas
- Same spacecraft achieves much larger deflection of the asteroid when using its high specific impulse engines to accelerate toward a maximum momentum exchange impact, rather than rendezvousing with the asteroid and pushing <u>for the chosen test case</u>
- Many more test cases need to be run for the wide variety of PHO orbits