Apophis T–9 Years: Knowledge Opportunities for the Science of Planetary Defense – Session 5

Low-thrust: the fast & flexible path to Apophis

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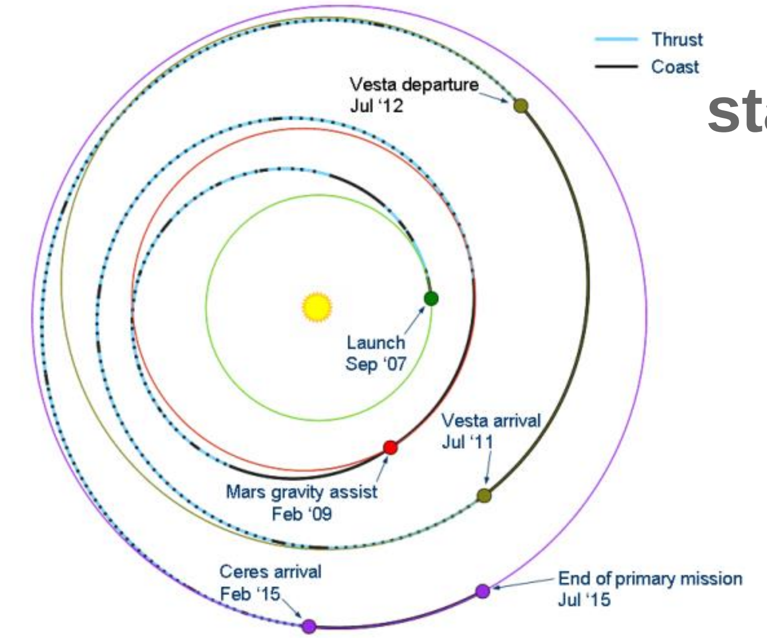
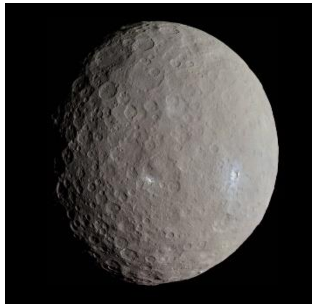
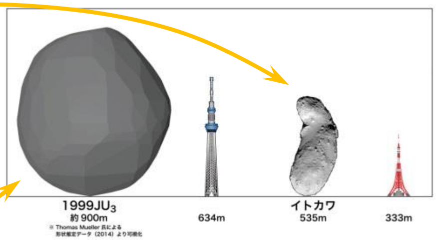


state of the low-thrust art: DAWN – 2-asteroid rendezvous
HAYABUSA &HAYABUSA2 – 1-asteroid sample-return

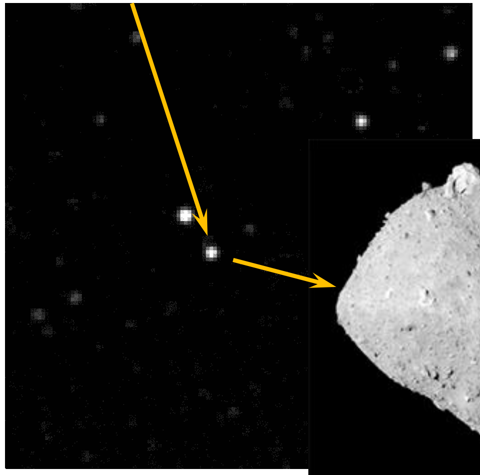
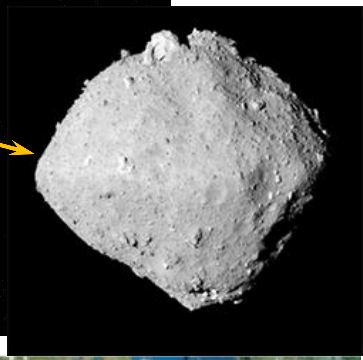



- DAWN orbited (4) Vesta & (1) Ceres
  - **Δv produced ≈ 11 km/s** from 425 kg Xe propellant
  - fly-by at (2) Pallas suggested but too marginal, possible fly-by of (145) Adeona not accepted



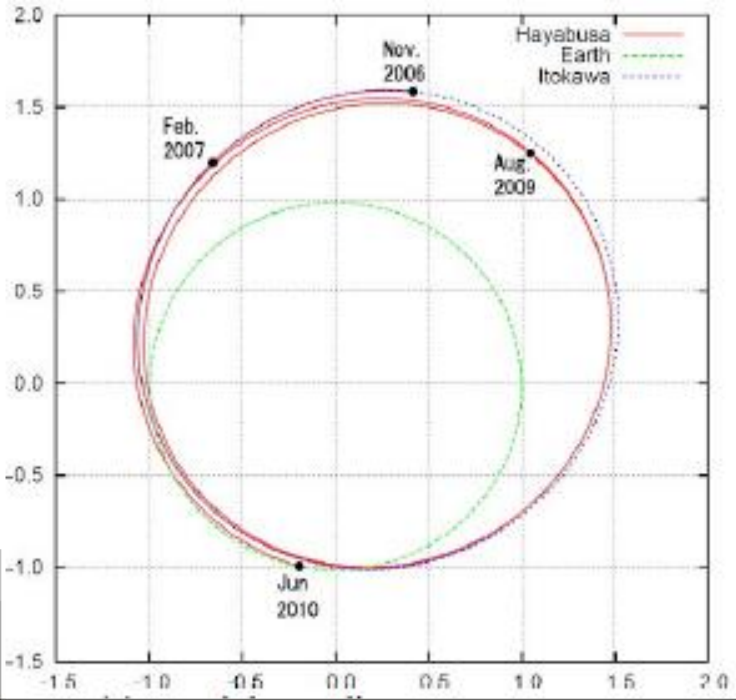
- HAYABUSA returned samples from (25143) Itokawa
  - **Δv capability ≈ 4.0 km/s** from 66 kg Xe propellant
  - total Δv ≈ 0.4 km/s from 67 kg bipropellant
  - **≈ 40 kg** used for SEP during mission, yielding 2.2 km/s

- HAYABUSA2 is currently on the way back from carbonaceous PHA (162173) Ryugu
  - **Δv capability ≈ 3.2 km/s** from 66 kg Xe propellant
  - **up to ≈ 3.5 km/s** if filled up to 73 kg Xe capacity

<u>*common feature*</u>:

high-performance electric propulsion



figures: Kuninaka et al., 2009; Nishiyama et al., 2015 , NASA/DLR, JAXA

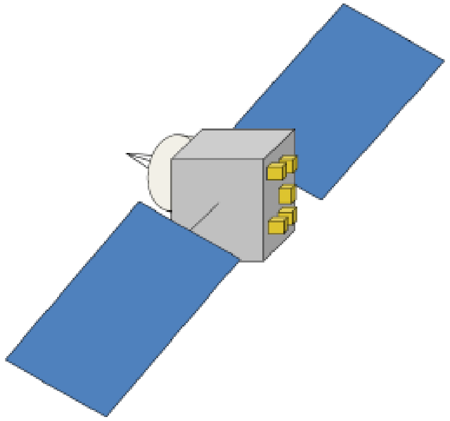
much more is possible by solar-electric propulsion

- DAWN spent some Δv to move between various mapping orbits, the HAYABUSAS to return samples
  - **what‘s science payload to carry?**
  - **what‘s there to repurpose?**
  - **where are the limits?**

SESAME (Maiwald & Marchand, 2016)

- **science payload 33 kg** (orbiter) **+ 5* 4.3 kg** (landers)
- **launch mass 1571 kg**
  - **~3x** HAYABUSA2, though only moderately larger than DAWN
- payload fraction <3.5 %: ≈ no gain by descoping science
- GTOC-5-like trajectory to **5 NEAs** of ≈200 candidate targets
- ***primarily* astrodynamic target selection**
- targets tied to launch date

➔ **remember: we're chasing just *one* asteroid!**



Property	Value
Departure Date from Earth	21 March 2023
Arrival @ 5th target	26 February 2030
Total Δv	→ 16.6 km/s
Wet mass	1,571 kg
Xenon fuel mass	451 kg
Bi-propellant mass	125 kg

Table 7: Mission parameters of SESAME

Target Body Data	Absolute Magnitude (H)	Orbit Condition Code (OCC)	Observation opportunities prior launch
$2001QJ_{142}$	23.4	6	2012
$2000SG_{344}$	24.8	2	none
$2009OS_5$	23.6	5	2014-2020
2007YF	24.8	5	2021
$1999AO_{10}$	23.9	6	2018, 2026

Table 8: Mission target data

Body	Arrival Date	Departure Date	Time of Flight (d)	Duration of Stay (d)
Earth	-	21 Mar 2023	316	-
$2001QJ_{142}$	31 Jan 2024	22 Jul 2024	444	173
$2000SG_{344}$	9 Oct 2025	8 Mar 2026	384	150
$2009OS_5$	27 Mar 2027	28 Sep 2027	445	185
2007YF	16 Dec 2028	19 May 2029	283	154
$1999AO_{10}$	26 Feb 2030	4 Sep 2030	314	190

Table 9: Trajectory data.

figures: Maiwald & Marchand, 2016

same idea, different method: Multiple NEA Rendezvous by Solar Sail

- solar sail propulsion is _not_ limited by the amount of fuel carried

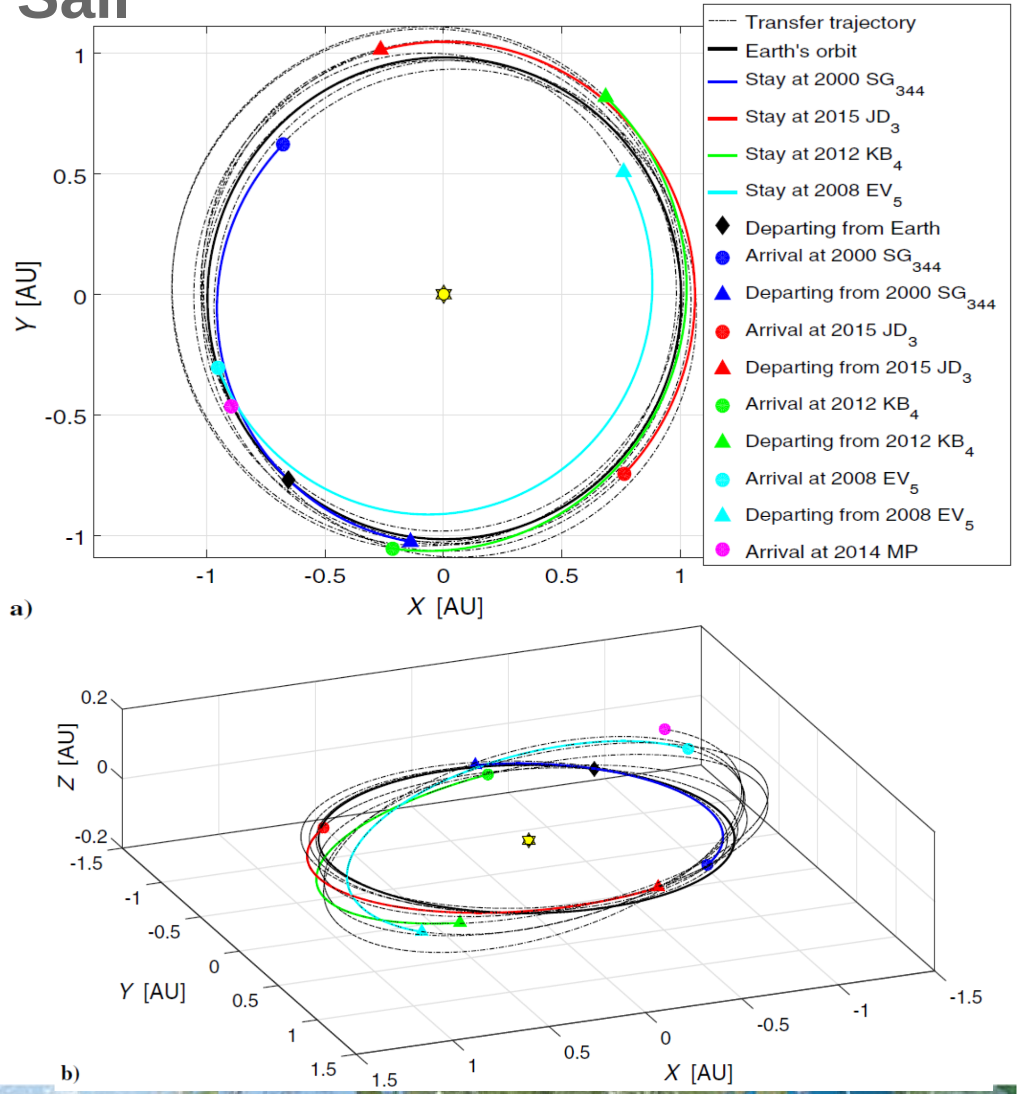
– then what's the next limit? –

how well it is designed, built, tested, flown & fixed

mechanisms have been fixed in space w/o astronauts around: e.g. Voyager 1 scan platform

- recent studies (Peloni et al., 2016-2018) demonstrate
  - 5 NEA stays for ≥100 days, each, in 10 years
  - accumulated **Δv >50 km/s** @ a_c = 0.2 mm/s²
  - ➔ asteroid-oriented target selection is feasible
  - ➔ at-launch & in-flight target change capability

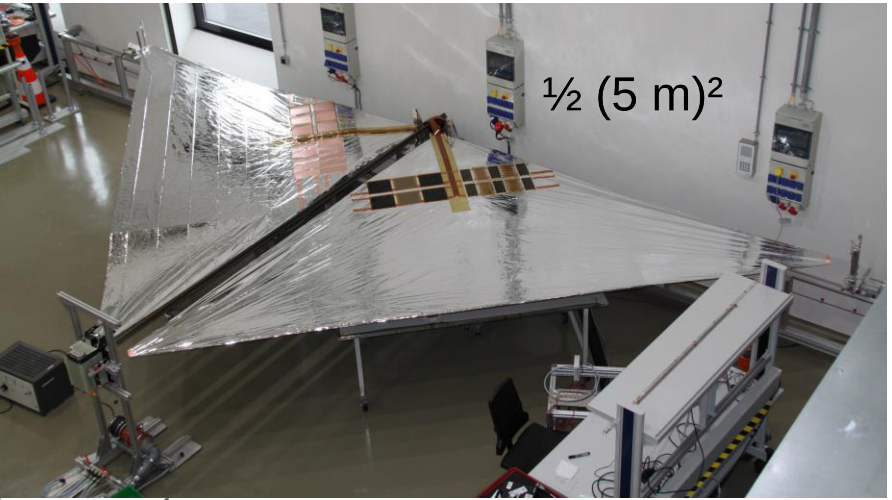
- target-flexible Multiple NEA Rendezvous for planetary science was identified as a mission type uniquely feasible with solar sail already by the GOSSAMER Roadmap Science Working Groups

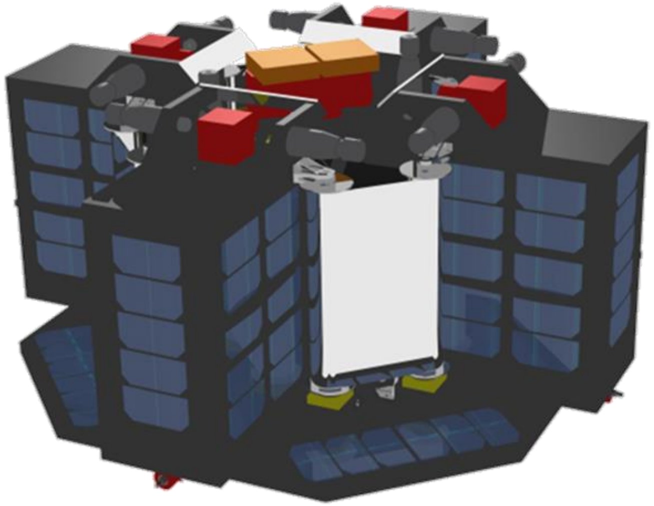


figures: Peloni et al., 2016-2018

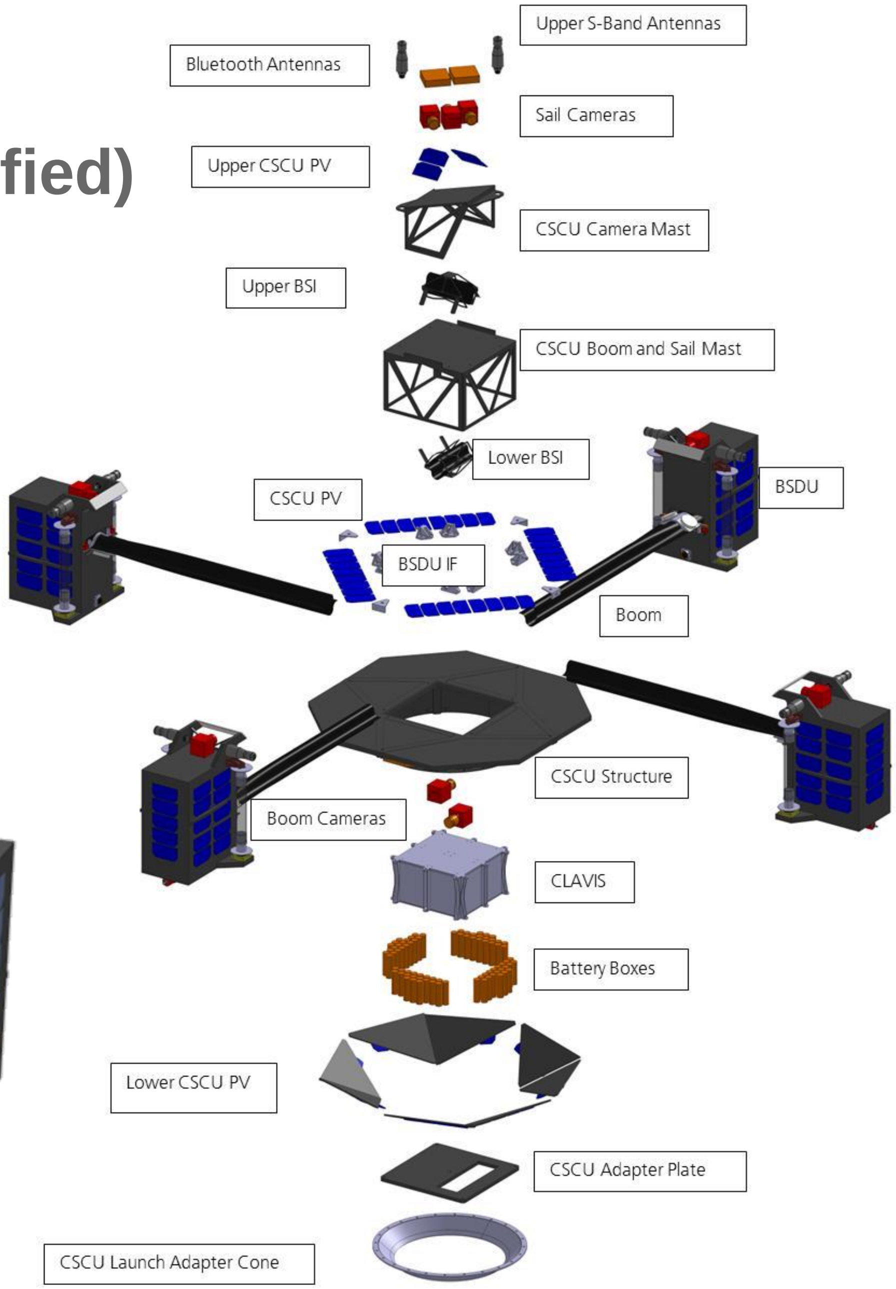
GOSSAMER solar sail technology development (qualified)

- the 3-step DLR-ESTEC GOSSAMER Roadmap to Solar Sailing was set up in 2009 to develop key technolgies for science missions
- 1st step: GOSSAMER-1 EQM was built & qualification tested
- development was stopped after reaching TRL 5
- a PFM design was ready to proceed
- a launch opportunity was available
- all-launchers load envelope



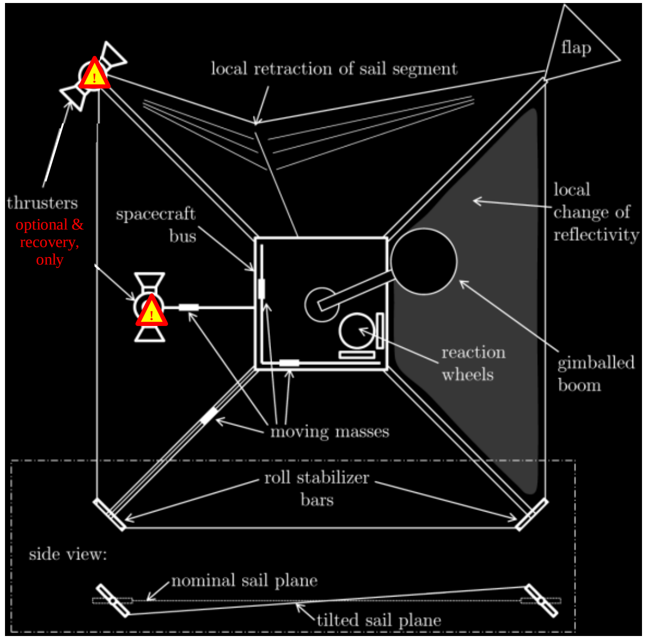


≈30…≤37.6kg, 79·79·50 cm³



figures: Seefeldt et al., 2016/2017

the road ahead for solar sailing (…alas, ’t was one of those… ;-)

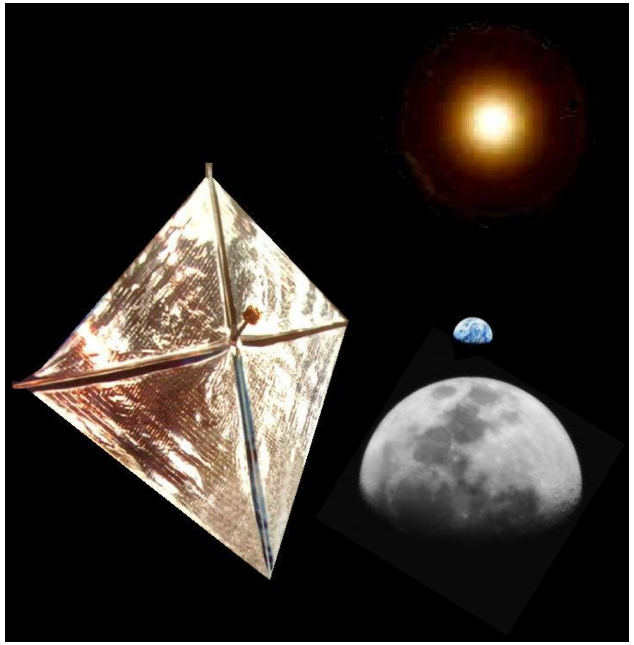


Gossamer-2 – high-orbit attitude & thrust vector control evaluation

- (20…25 m)² sail area
- orbit where solar radiation pressure is dominant – high LEO, MEO, GTO, >GEO
- implementation of several to “all” control methods *and* all relevant mechanisms
- **find out what’s the best …**

the Gossamer Roadmap:
step 2 – control
step 3 – proving the principle
…
…that was the idea…

(↑2009 – ↓2014)



Gossamer-3 – all-up proof test & science mission readiness demonstrator

- (50 m)² sail area
- initial orbit high enough to spiral out (sail up)
- applies best control method(s) of Gossamer-2
- **prove that sails can operate science missions**
  - imager & *maybe* a *tiny* science-like payload, e.g. sail-environment interaction (~1kg total)

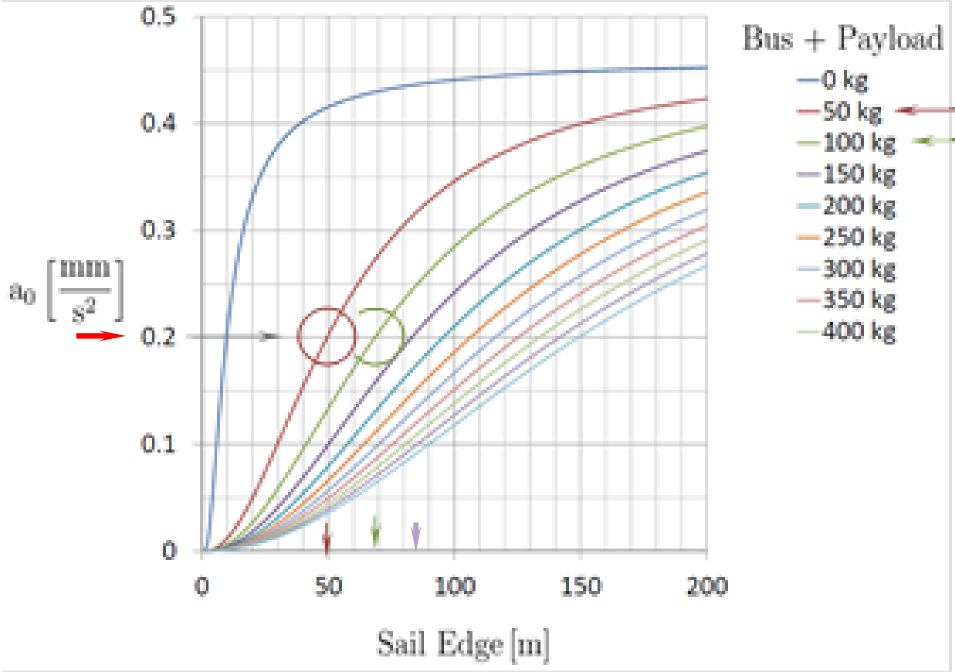
images adapted from Dachwald et al. & ENEAS study
additional images: Moon: Jérôme Salez (wiki), Earth: Bill Anders, Apollo 8, NASA

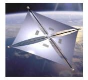
performance: the magic Multiple NEA Rendezvous & landers numbers

- GOSSAMER-1 technology based
  - 0.2 mm/s² & 50 kg bus & payload → (50 m)² membrane
  - 0.2 mm/s² & 100 kg bus & payload → (70 m)² membrane
  - 0.2 mm/s² & 150 kg bus & payload → (85 m)² membrane

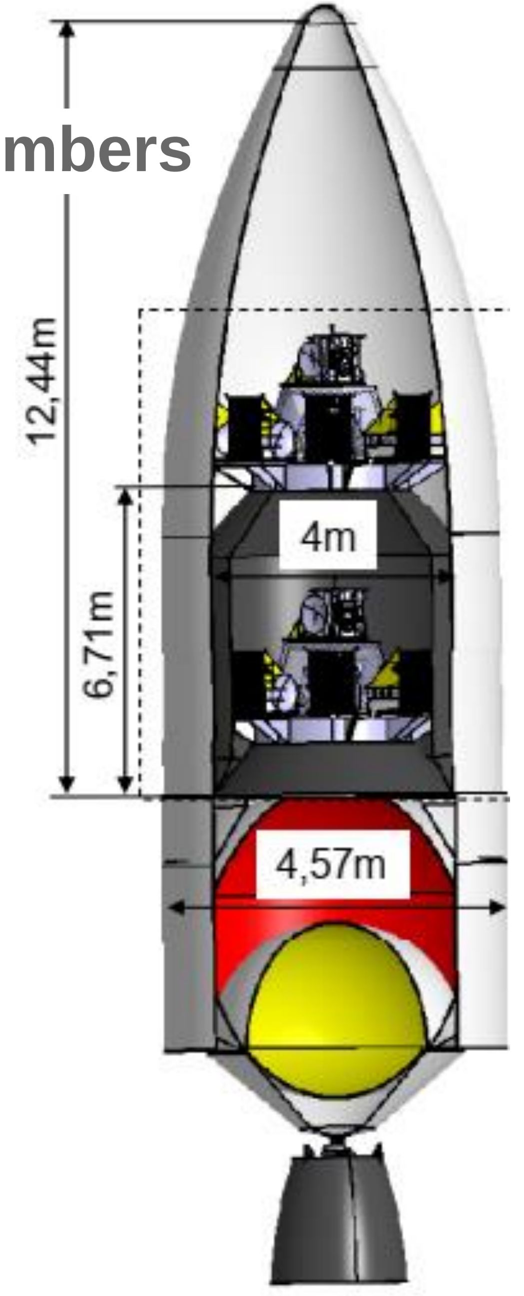
→ ESPA* / ASAP** compatible micro-spacecraft range (* ≤181 kg, 61·61·96 cm³ / ** ≤200 kg , 80·80·100 cm³) →

- start from GTO and upper stage disposal orbits of GEO & NavSat



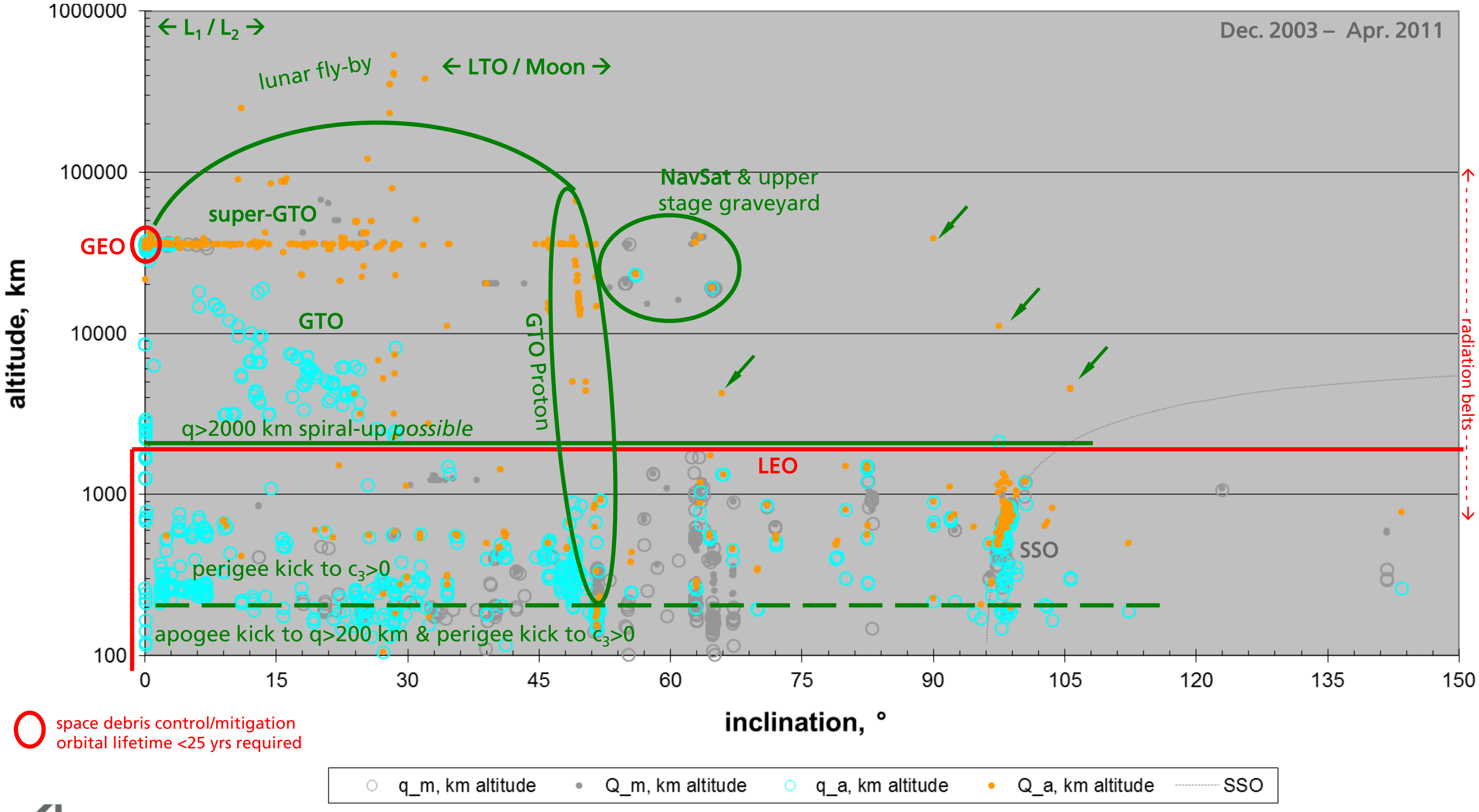
Bus (Gos-1 + X-Band) 	≈ 30 kg
Mother Spacecraft Science Instruments	≈ 10 kg

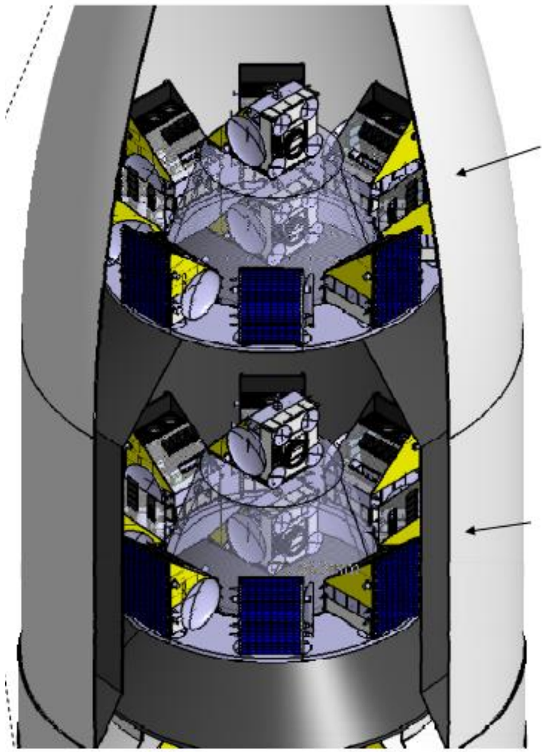
Lander	
PHILAE 	98 kg
SPS Lander 	100kg
1 MASCOT 	10 kg
5 MASCOTs 	50 kg +10 kg support equipment



figures : Seefeldt et al., 2016

where to start from?
– piggy-back launch opportunities for solar sails





space debris control/mitigation
orbital lifetime <25 yrs required

sailing studies to Apophis

- launch opportunities repeat with synodic period of 7.8 years

- many studies assume unrealistically high acceleration
- realistic in now-term/near-term techology: 0.2…<0.3 mm/s²

- *sanity check*:
  impulsive Δv in range of DAWN or SESAME but beyond capability of the HAYABUSAS

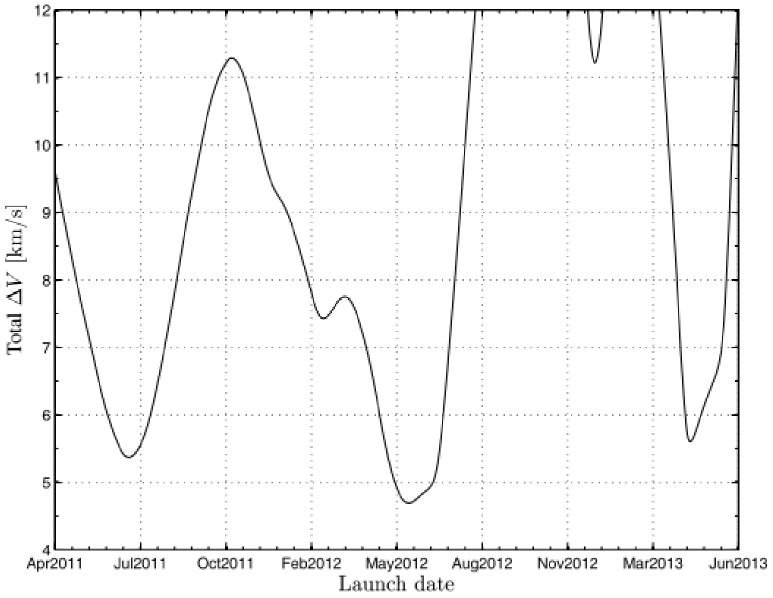


Fig. 8 Required ΔV as a function of launch date for two-impulse maneuver.

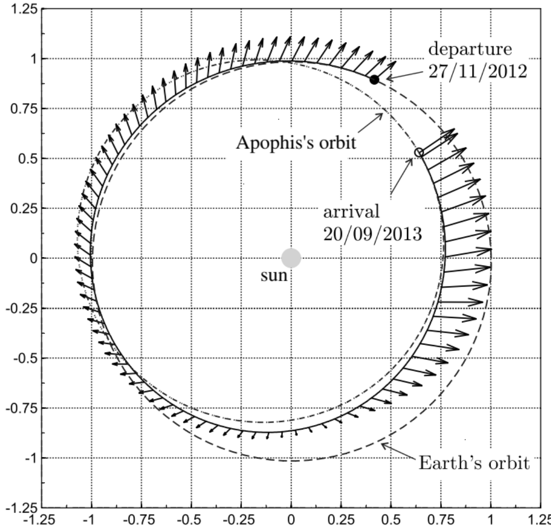


Fig. 6 Projection on the ecliptic plane of spacecraft trajectory and thrust profile for $a_c = 0.5$ mm/s².

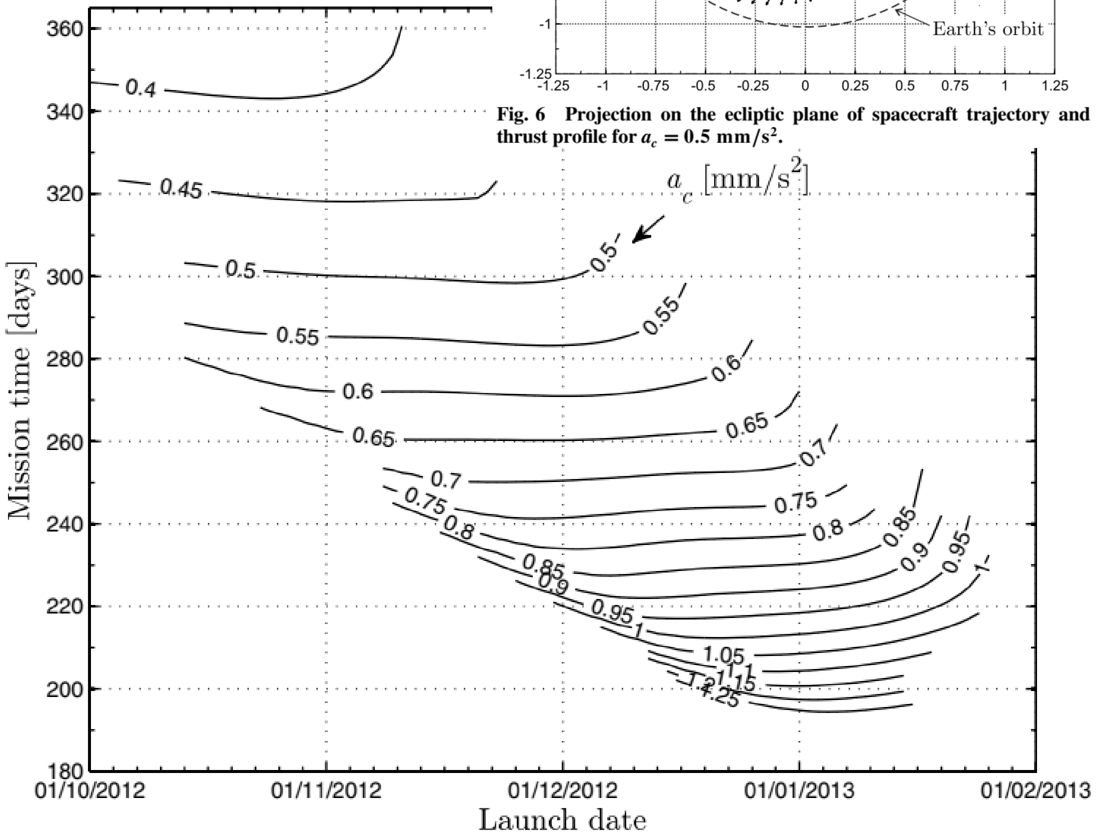


Fig. 4 Launch opportunities in 2012–2013 as a function of a_c.

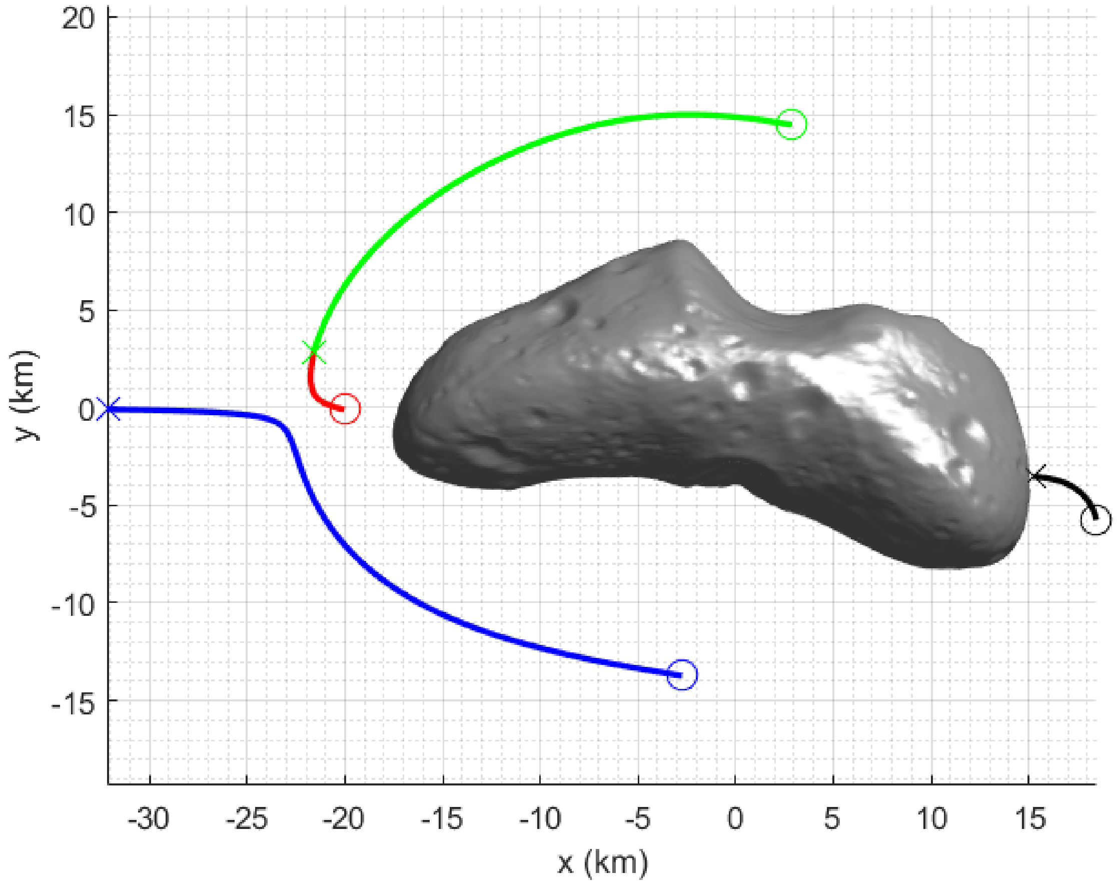
Mengali & Quarta, 2009

Artificial Equilibrium Points (AEP)

- model asteroid: (433) Eros

- sail attitude fixed.
- position of each AEP changes with a_c
  - circles: initial point
  - crosses: final point

- AEPs 1 and 3 “collide and disappear“ at critical value of characteristic acceleration (a_c = 0.87 mm/s^2 for Eros)



$$\alpha = \delta = 0^\circ$$

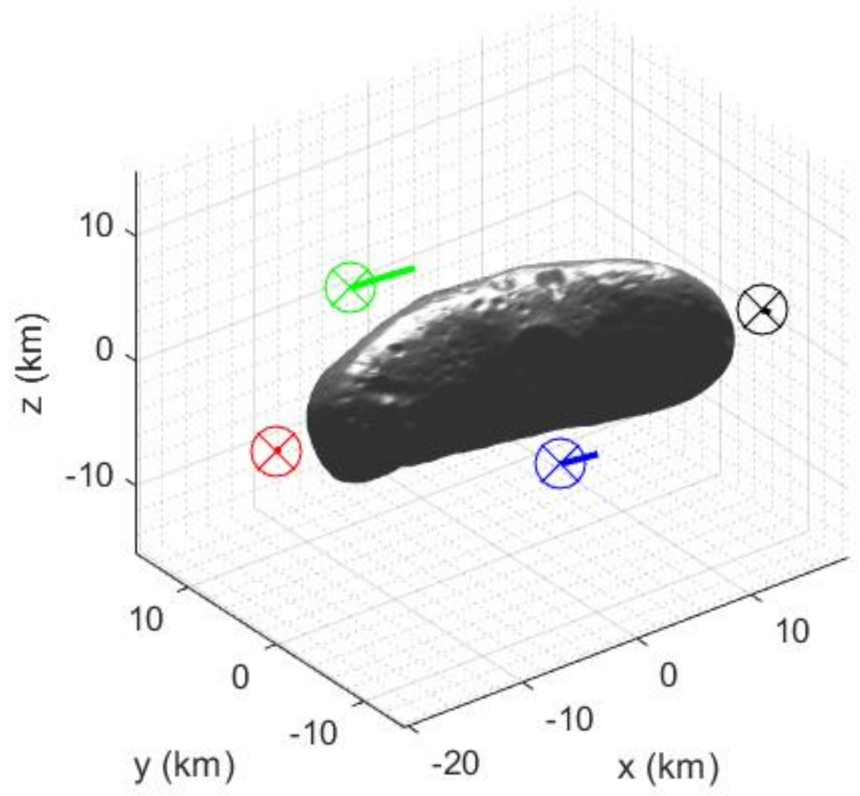
$$a_c \in [0,2]\mathrm{mm/s^2}$$

shifting of AEPs during asteroid rtation

- effect of the Sun rotating around the asteroid in the body-fixed frame

- sail attitude is fixed
- one full rotation of the asteroid
  - circles: initial point
  - crosses: final point

- as expected, these points are the same as the Sun returns to the initial point



$$\alpha = \delta = 0^\circ$$

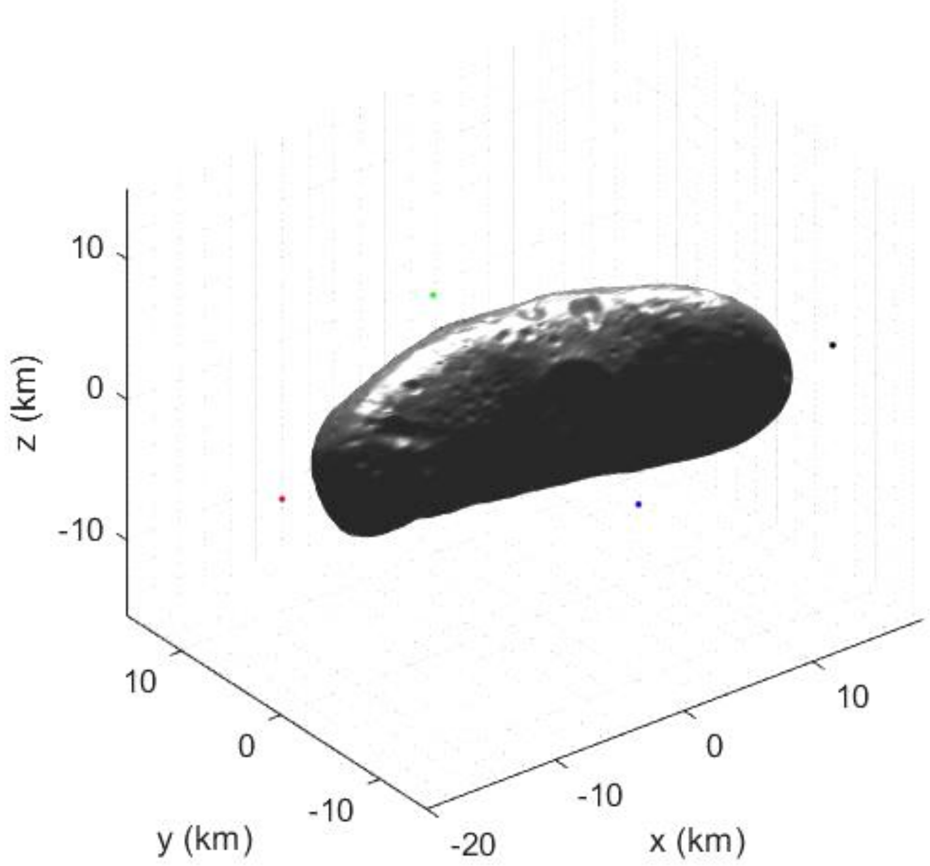
$$a_c = 0.2\ \mathrm{mm/s^2}$$

accessible volume for AEP hovering

- video shows sail performance and sail attitude create the range of positions available for AEPs

- time frozen at the initial time so that the position of the Sun remains fixed
- AEPs 1 and 3 collide at $a_c = 0.87$ mm/s^2 and so their associated range of AEPs disappear

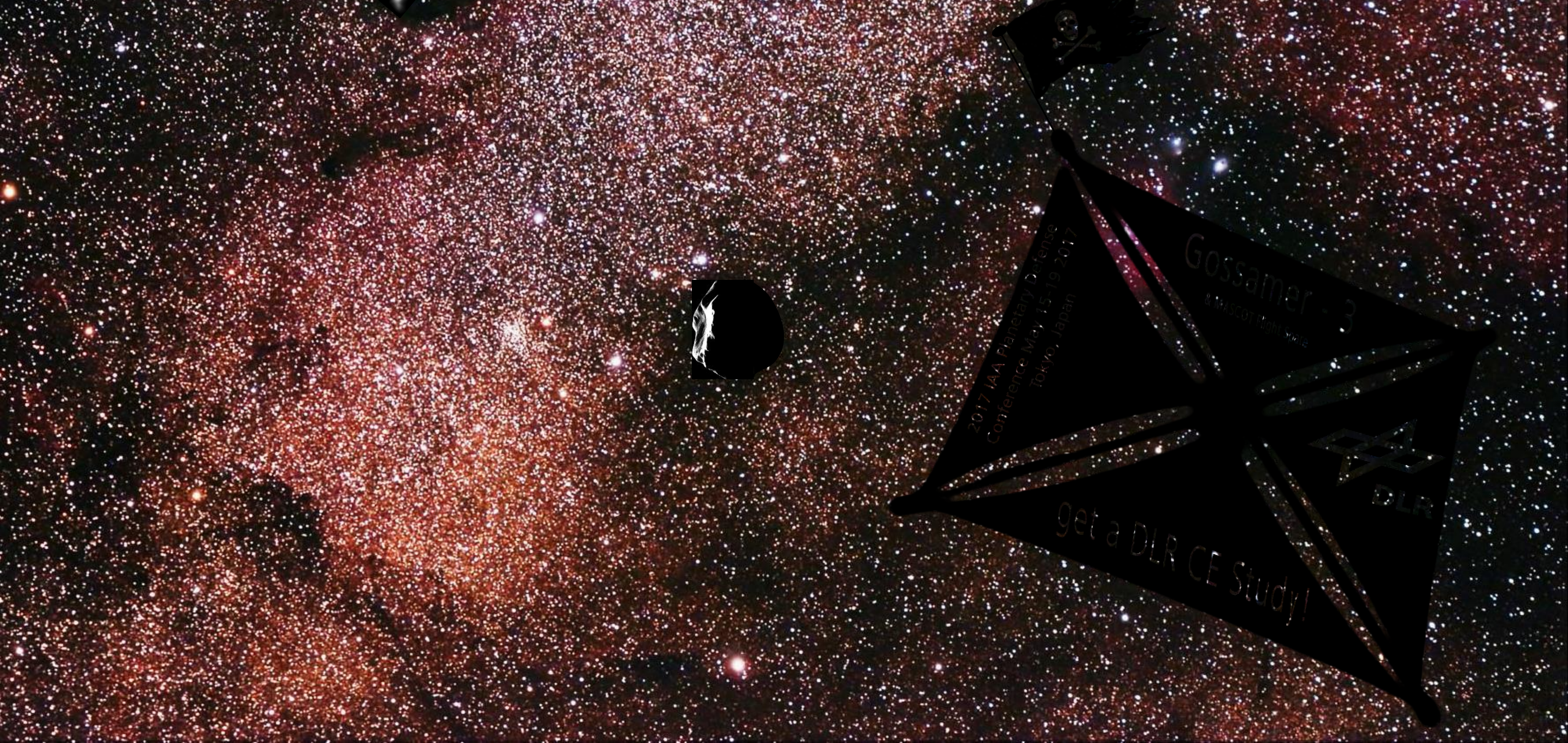
- Note: only small number of sail attitudes shown, lines outline volume of possible AEPs.



$$\alpha \in [0,90]^\circ, \qquad \delta \in [0,360]^\circ$$

$$a_c \in [0,2]\mathrm{mm/s}^2$$

thank you for your attention! – as m any questions? as you like, please ☺



Apophis Small SEP Performance Summary

Technology Status

- SEP is already very flexible:
  wide operation possibilities with established systems →
- Performance is „freely chosen“ (within limits) based on a few high-impact parameters.

BUS Capability & Implementation

High-Impact Parameters:

- Available solar power:
  thrust & I_{sp} scale directly *but differently* with input power →
- Beam voltage: thrust ←**TRADE**→ I_{sp}

Programmatic Considerations & Trades

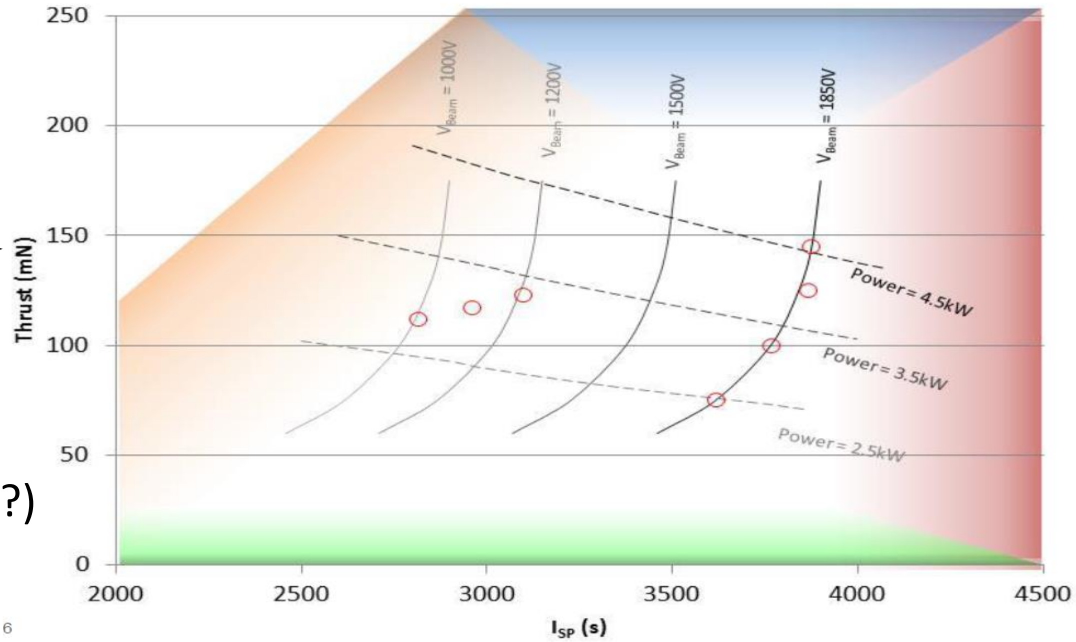
- Choice is based on desired mission outline (go for higher I_{sp} or thrust?)
  → Available propellant fixed? (tank & spacecraft size, mass?)
  → Available time?
  → If SmallSat and lots of time → go for high Isp → low-power ok → small design impact → fast development & low-cost achievable
  → If SmallSat and no time → go for high Thrust → high power needed → design impact → fast development but high cost

typical example: QINETIQ

Thruster	T5 [1-25mN]	T6 [30-230mN]	T7 [30-290mN]
Power class	1 kW	5 kW	7 kW
Mass	2 kg	8.5 kg	13 kg
Grid diameter	10 cm	22 cm	30 cm
	GOCE 2009	BepiColombo 2018	development within H2020

[all up to 3000-4000 Isp]

QinetiQ T6 Performance Map (≈TRL 8)



Apophis Small SEP Performance Summary

Apophis Mission 2029

- → SEP will be
  - → more COTS (thus more available and lower in cost)
  - → more effective (reduced heat dissipation)
  - → more reliable (less degradation)

- → Biggest limiting factor (today and in the future) is electrical power, NOT the technology.
- → Cost might always be between 10-30% of total.

ESKIMO SmallSat Performance Examples

- → 150kg class ESKIMO (as is) reaches ΔV=3.5km/s (3500 I_{sp}, 14kg propellant)
  - ... 12mN using 600W (2x 300W) currently-developed SEP. 50kg P/L. ~~10-15M€ incl. Launch (2029-2026).

- → 150kg class ESKIMO (P/L=Propellant+) reaches ΔV=10.5km/s (3500 Isp, 40kg propellant)
  - ...12mN using 600W (2x 300W) currently-developed SEP. 25kg P/L. ~~12-17M€ incl. Launch (2029-2026).
  - ...24mN using 1200W (4x 300W) currently-developed SEP. 25kg P/L. ~~18-25M€ incl. Launch (2029-2026).

THRUST ↑ - Transfer-Time ↓
Available TIME and therefore THRUST is the SEP system driver.

The SEP system itself is NOT the cost-driver, but the Solar Generator (THRUST scales with power). Solar Generator designs (and cost?) do not evolve much anymore.
Assumption: Double power → Double thrust → 1.5x cost? (Assuming Solar Generator is ~~50% of cost) → Trades are difficult and depend on hard requirements. Many design constellations are possible.

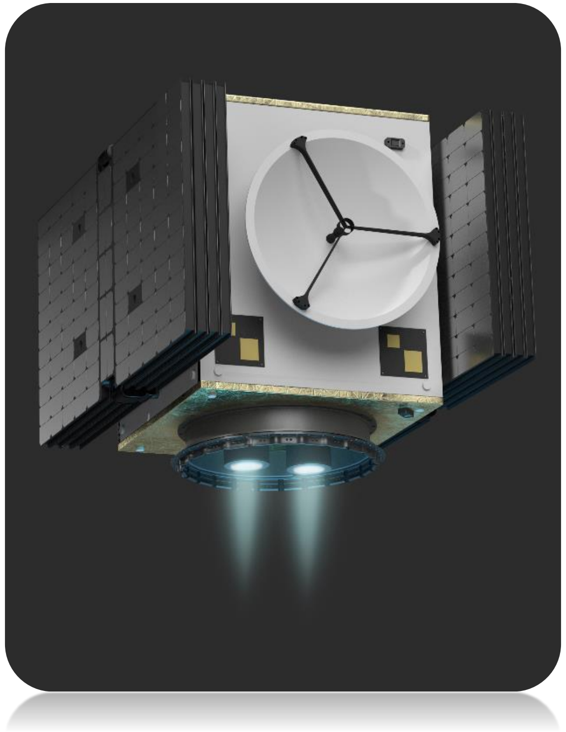
In summary, SEP has much potential and a huge mission-performance-range between a very fast and low-cost COTS-bolt-and-fly design and a well-optimized, dedicated spacecraft.

DLR

LEVITY: ENABLING THE NEXT SPACE MARKETS

ESKIMO: The Next Generation Small Satellite Platform

- 150kg-class, 1m³ small satellite platform
- Specifically designed to address orbits beyond GEO including Moon missions.
- Electric propulsion system, flexible payload accommodation and -interfacing
- The platform can also act as a kickstage, transporting small satellites and offering deep space relay services at the target environment.
- ➢ A convenient ready-to-use platform for any commercial and scientific mission within low-mass, low-cost concepts.



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100W Guaranteed to Payload at Any Time

More than 10 GB of Data Per Day from Low Lunar Orbit
Using Commercial Ground Stations

Deep Space Radiation Tolerance

A plug-and-play solution for instruments

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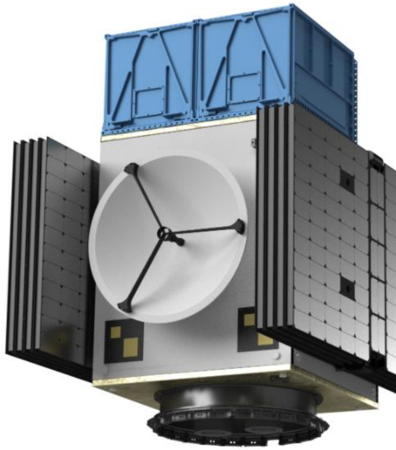
PAYLOAD ENVELOPE INCL. ADAPTERS
300l | 50kg | 32U



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