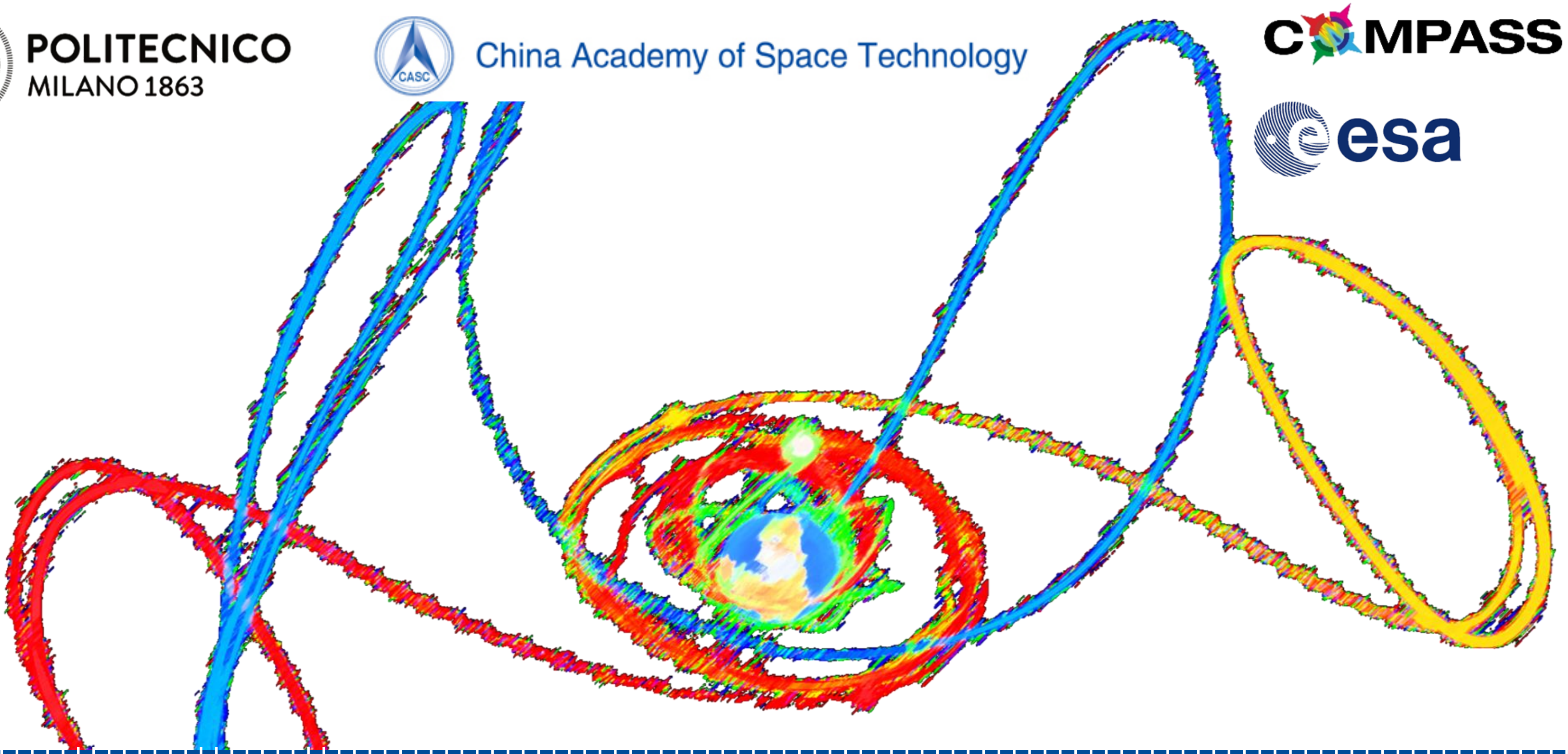


Planetary protection: design and verification

Camilla Colombo, Politecnico di Milano
Email: camilla.colombo@polimi.it

DFH - China Academy of Space Technology, Beijing

Planetary protection

INTRODUCTION

Introduction

Humans now routinely venture beyond Earth and send spacecraft to explore other planets.

With this extraordinary ability comes great responsibility:

- we must ensure that we do not bring back anything harmful from other worlds
- we must make sure that we do not introduce terrestrial biological contamination to other planets and moons that have potential for past or present life.

http://exploration.esa.int

Committee on Space Research (COSPAR) planetary protection policy
United Nations Outer Space Treaty

Introduction

Planetary protection framework

- Since 1958 (year after Sputnik) concern that initial exploration of the Moon and other celestial bodies might compromise future scientific exploration
- Ranger missions in 1961 first used planetary protection requirements
- Since then, all planetary missions had to implement planetary protection measures at different degrees
- Legal framework in the United Nations Treaty on Principles Governing the Activities of States in the Exploration and Use of Outer Space, including the Moon and other Celestial Bodies (Outer Space Treaty)
- Spacecraft have to control
  - Forward contamination
  - Backward contamination

➢ G. Kminek. ESA planetary protection requirements. Technical Report ESSB-ST-U-001, European Space Agency, February 2012.

Introduction

Planetary protection requirements for forward contamination

For interplanetary missions and missions at Libration Point Orbit, planetary protection analysis need to be performed

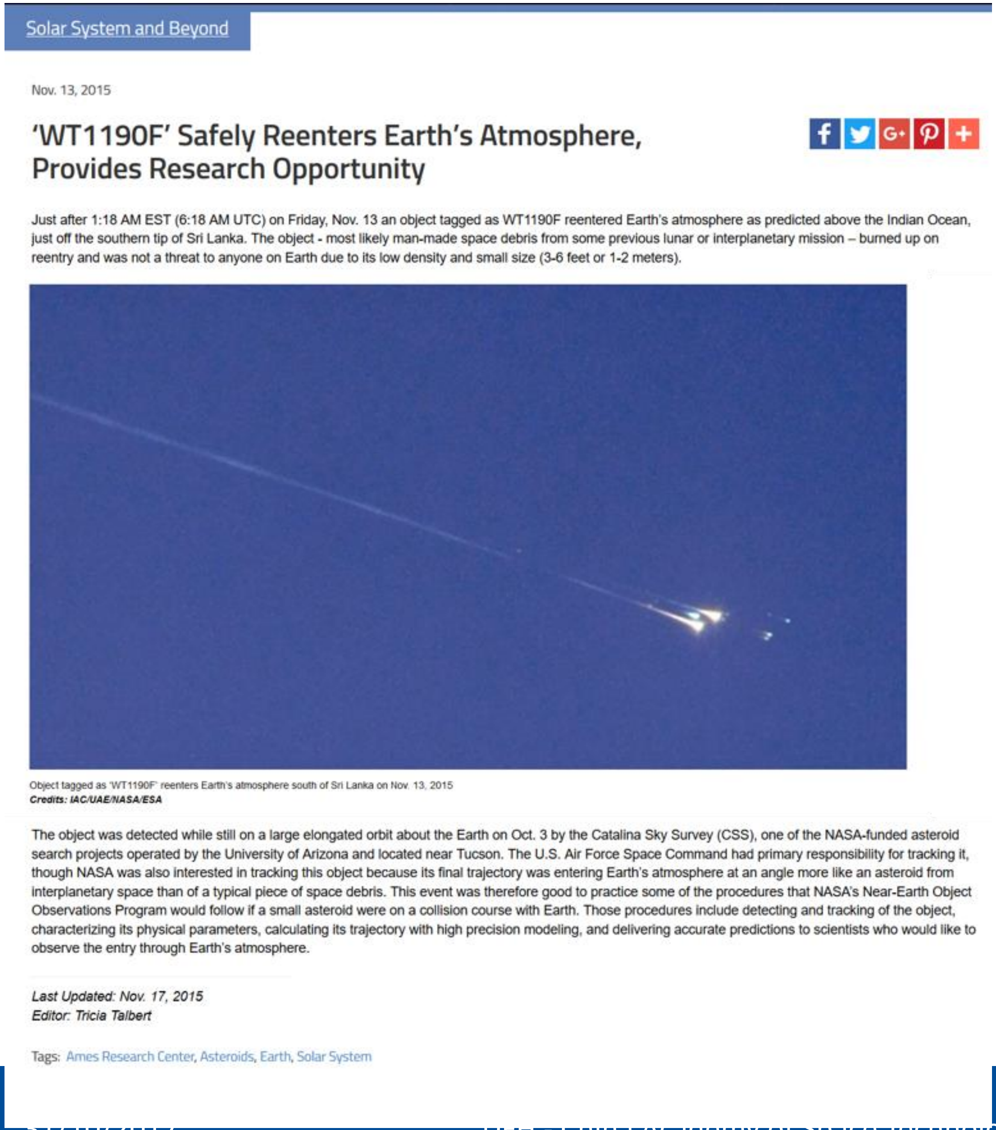
Forward contamination, contamination of celestial bodies other than the Earth by terrestrial life forms in the course of spaceflight missions

⬇

- Ensure that the impact probability of spacecraft and upper stages with planets and moons over 50-100 years is below the **required threshold** with a give **confidence level**.
- Compliance with requirements should be verified for
    - The nominal trajectory
    - Considering on-board failures
    - Considering uncertainties on orbit injection and s/c parameters

Introduction

Nov. 13, 2015: “WT1190F Safely Re-enters Earth’s Atmosphere”



Solar System and Beyond

Nov. 13, 2015

‘WT1190F’ Safely Reenters Earth’s Atmosphere, Provides Research Opportunity

Just after 1:18 AM EST (6:18 AM UTC) on Friday, Nov. 13 an object tagged as WT1190F reentered Earth’s atmosphere as predicted above the Indian Ocean, just off the southern tip of Sri Lanka. The object - most likely man-made space debris from some previous lunar or interplanetary mission – burned up on reentry and was not a threat to anyone on Earth due to its low density and small size (3-6 feet or 1-2 meters).

Object tagged as ‘WT1190F’ reenters Earth’s atmosphere south of Sri Lanka on Nov. 13, 2015
Credits: IAC/UAE/NASA/ESA

The object was detected while still on a large elongated orbit about the Earth on Oct. 3 by the Catalina Sky Survey (CSS), one of the NASA-funded asteroid search projects operated by the University of Arizona and located near Tucson. The U.S. Air Force Space Command had primary responsibility for tracking it, though NASA was also interested in tracking this object because its final trajectory was entering Earth’s atmosphere at an angle more like an asteroid from interplanetary space than of a typical piece of space debris. This event was therefore good to practice some of the procedures that NASA’s Near-Earth Object Observations Program would follow if a small asteroid were on a collision course with Earth. Those procedures include detecting and tracking of the object, characterizing its physical parameters, calculating its trajectory with high precision modeling, and delivering accurate predictions to scientists who would like to observe the entry through Earth’s atmosphere.

Last Updated: Nov. 17, 2015
Editor: Tricia Talbert

Tags: Ames Research Center, Asteroids, Earth, Solar System

- https://www.nasa.gov/feature/wt1190f-safely-reenters-earth-s-atmosphere-provides-research-opportunity

Design of heliocentric disposal for Libration Point Orbit missions

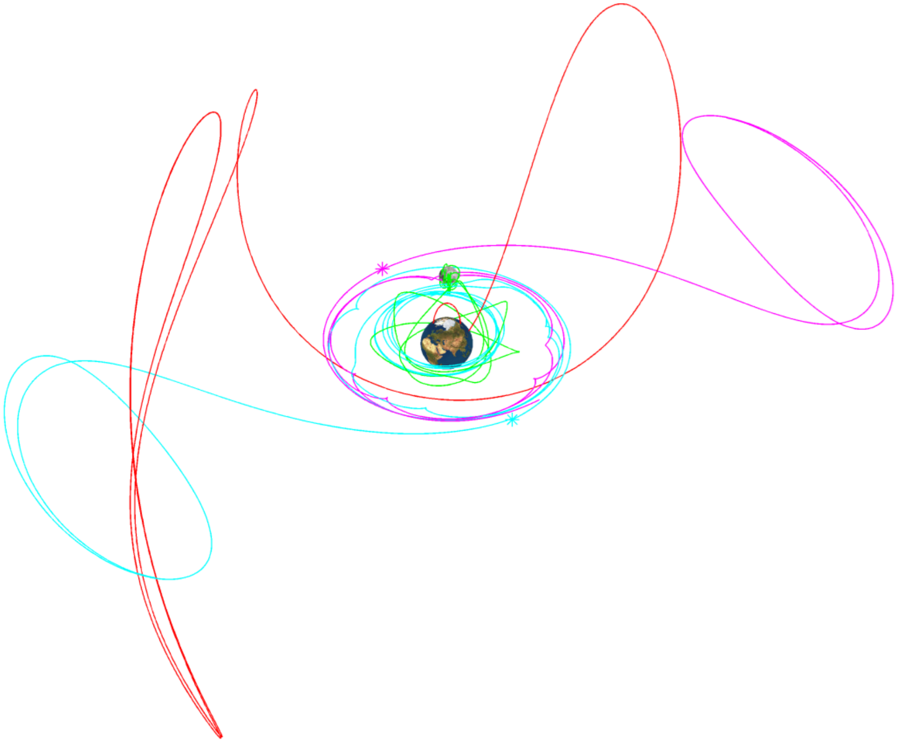
PLANETARY PROTECTION

Introduction

End-of-life disposal of Libration Point Orbit missions

- Libration Point Orbits (LPO) are often selected for astrophysics and solar terrestrial missions
- It is a critical aspect to clear these regions at the end-of-mission

  - Highly perturbed environment
  - Spacecraft with large dimensions
  - Uncontrolled s/c on manifold trajectories can re-enter to Earth or cross the protected regions
  - End-of-life trajectory can enhance science return and operational knowledge base



Gaia mission end-of-life disposal

Design and analysis

Design

- Heliocentric disposal in a n-dynamical model
- Robust disposal that prevent the return of the spacecraft to Earth for the following 100 years at least

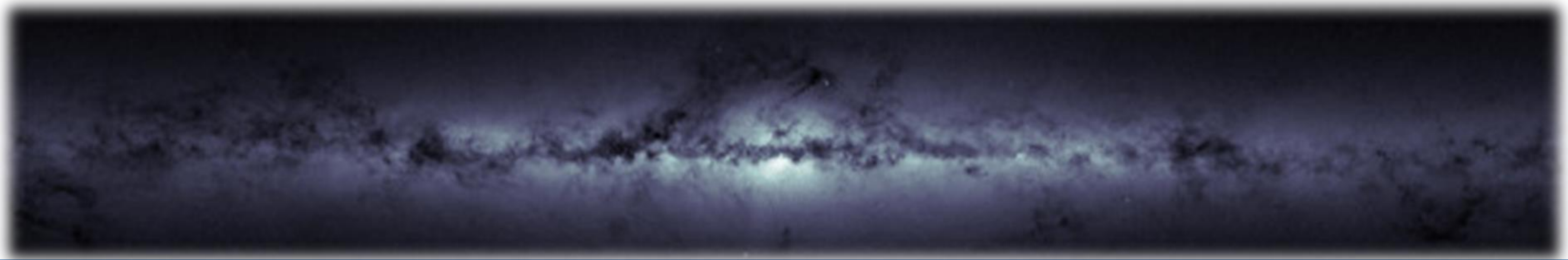
Analysis

- Effect of the Earth’s eccentricity and other perturbations on the disposal strategy
- Behaviour of the close approaches between spacecraft and Earth once the disposal manoeuvre is given

The Gaia mission

Mission datasheet

Launch date:	19 December 2013
Mission end:	Nominal mission end after 5 years (2018)
Launch vehicle:	Soyuz-Fregat
Launch mass:	2030 kg (710 kg of payload, a 920 kg service module, 400 kg of propellant)
Mission phase:	Completed first year of science observations
Orbit:	Lissajous-type orbit around L_2
Instruments:	Astro (2 identical telescopes and imaging system); BP/RP (Blue and Red Photometers) and RVS (Radial-Velocity Spectrometer)
Partnerships:	Gaia is a fully European mission designed, built and operated by ESA. The Gaia Data Processing and Analysis Consortium will process the raw data to be published in the largest stellar catalogue ever made.

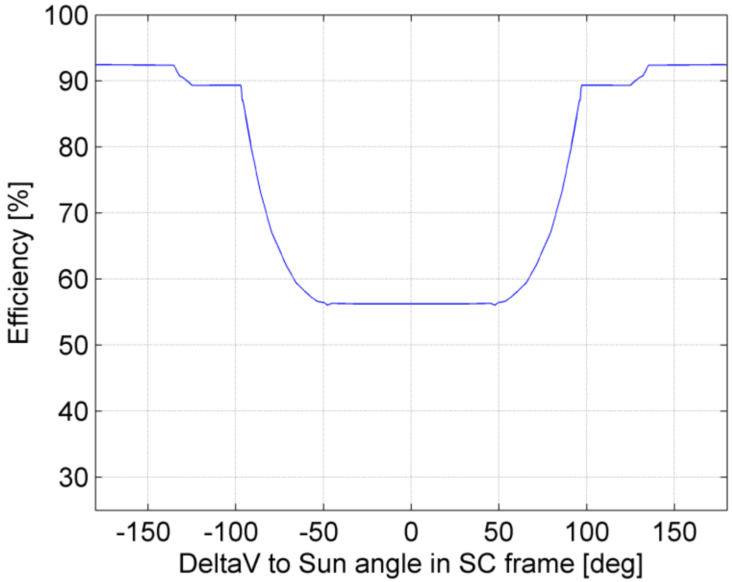


The Gaia mission

Gaia disposal constraints and requirements

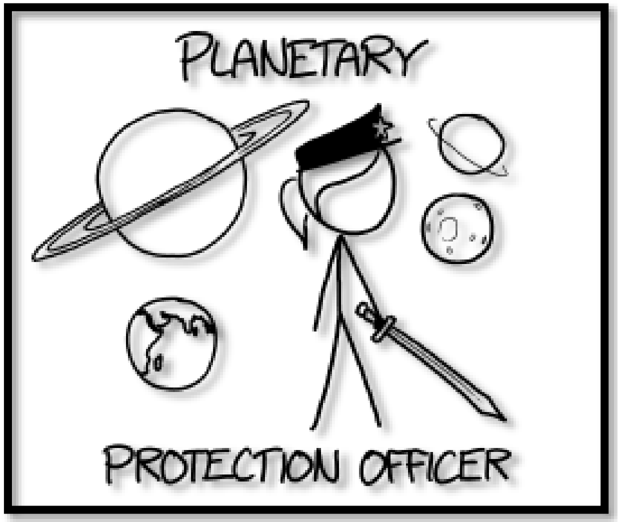
Constraints

- Time window: 01/07/2019 - 31/12/2020
- Operational: maximum 6 months between two disposal manoeuvers
- Thrust efficiency from European Space Agency



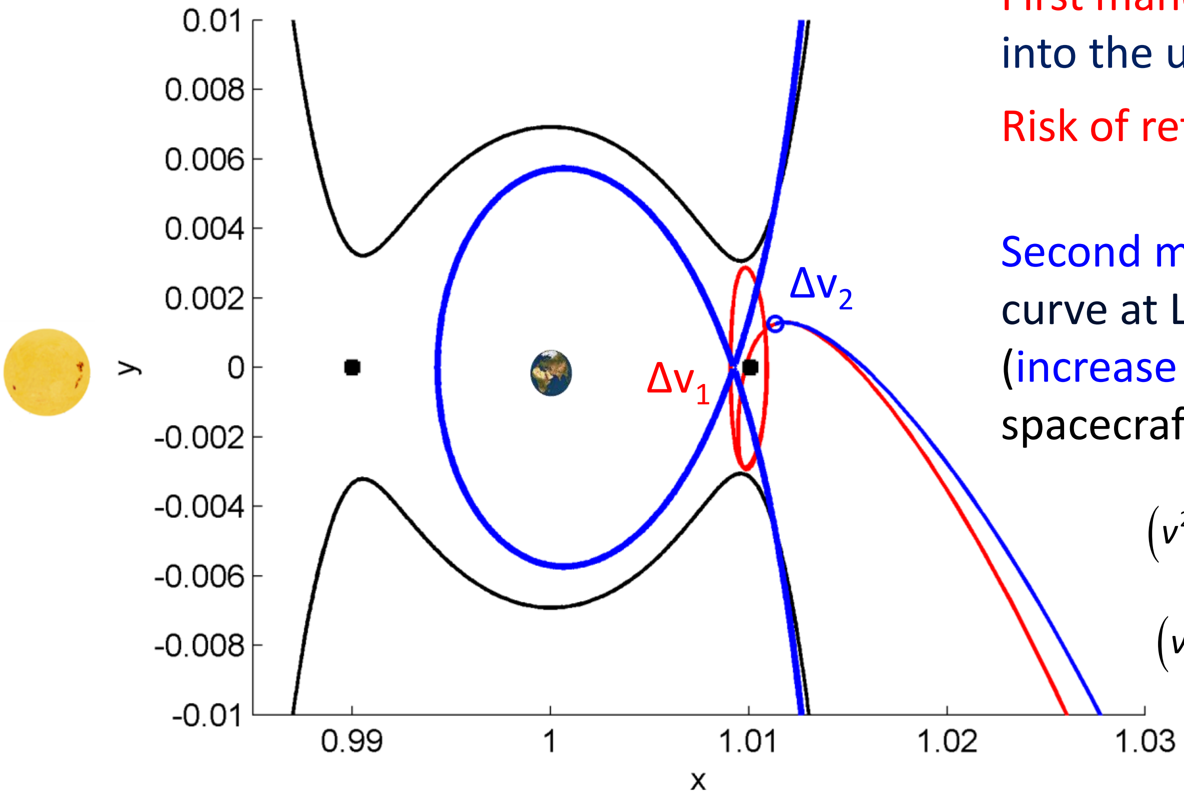
Requirement

The spacecraft should not return to the Earth for at least 100 years after the disposal epoch



Heliocentric disposal

Design in circular restricted three body problem



First manoeuvre to inject the spacecraft into the unstable trajectory leaving the LPO

Risk of returning to Earth through L_2

Second manoeuvre to close the Hill's curve at L_2: Δv to decrease the energy (increase Jacobi constant) tangent to spacecraft velocity in the rotating frame

$$\left(v^2\right)^- = x^2 + y^2 + 2\left(\left(1-\beta\right)\frac{1-\mu}{r_S} + \frac{\mu}{r_E}\right) - J$$

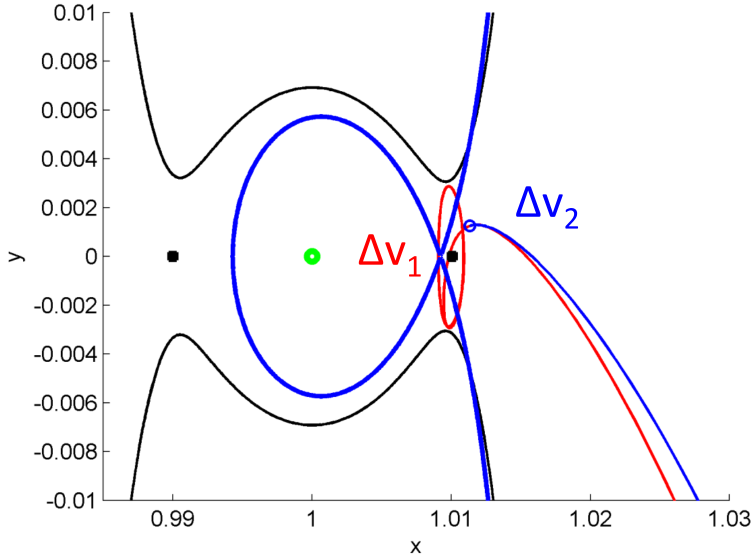
$$\left(v^2\right)^+ = x^2 + y^2 + 2\left(\left(1-\beta\right)\frac{1-\mu}{r_S} + \frac{\mu}{r_E}\right) - J_{L_2}$$

$$\boxed{\Delta v_{\text{closure @ } J_2} = v^- - v^+}$$

$$\Delta v_{\text{closure @ } FR} = v^- = \sqrt{\dot{x}^2 + \dot{y}^2 + \dot{z}^2}$$

Heliocentric disposal

Design in high-fidelity dynamics model



Manoeuvre 1

- M_1 ($\mathbf{\Delta v_1}$, $\boldsymbol{\alpha_1}$, $\boldsymbol{\beta_1}$)
- t_1

Manoeuvre 2

- M_2 (Δv_2, α_2, β_2) @ t_2
- **Δt** = t_2-t_1

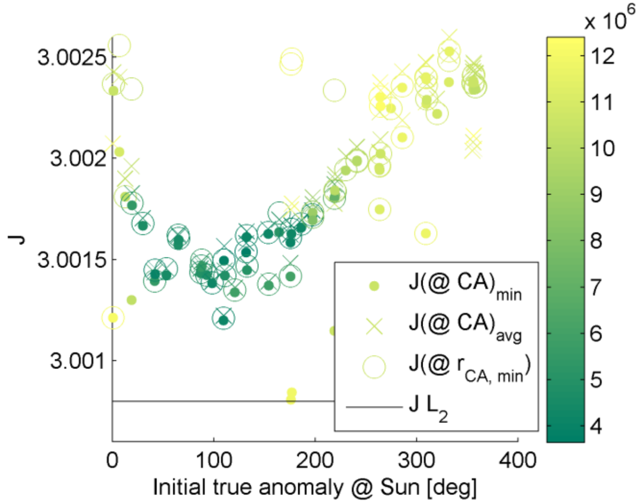
- $\boldsymbol{\alpha_2}$ and $\boldsymbol{\beta_2}$ are **fixed** to π and 0 with respect to the velocity in synodic frame
- The second manoeuvre is performed using all the remaining available Δv
- Find solutions where the **minimum J** in the monitored points is **higher than J at L_2**. **Monitor** only the points at the **close approach** with the Earth
- Disposal **starting time** treated with a **grid search**

Heliocentric disposal

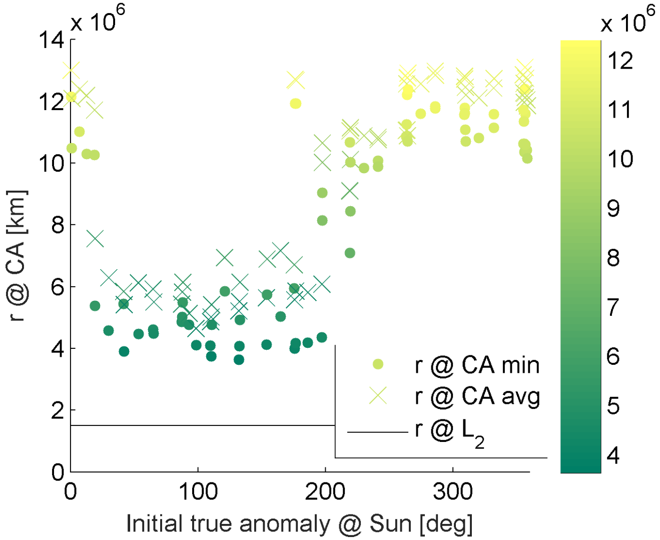
Results

- Each point is a fully optimised solution
- **True anomaly of Earth + Moon @ Sun** when leaving LPO drives effectiveness of the disposal:

  Higher minimum distance from Earth attained for solutions that leave the LPO when Earth + Moon is within [180, 360].



a) Optimisation with $\Delta t_{\mathrm{EOL}} = 30$ years



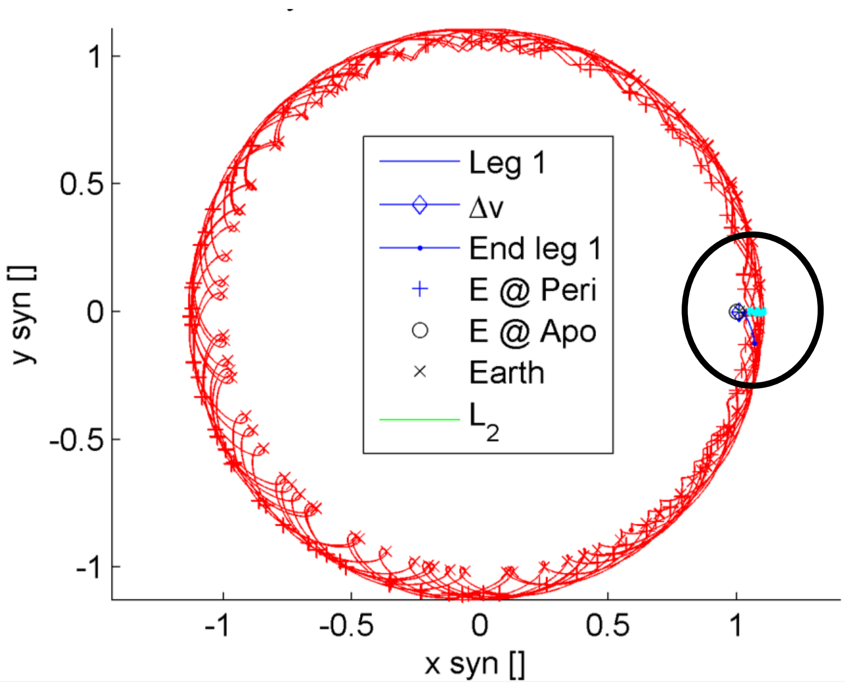
a) Optimisation with $\Delta t_{\mathrm{EOL}} = 30$ years

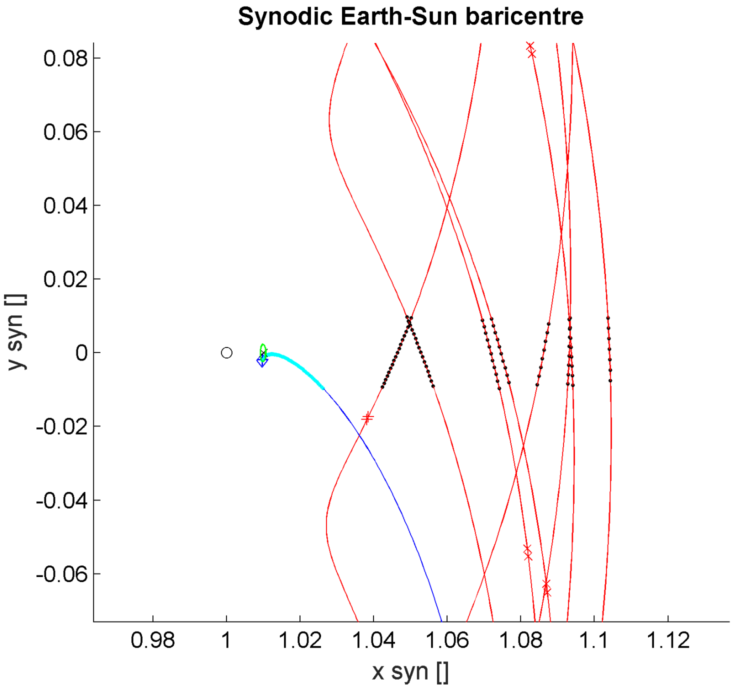
Heliocentric disposal

Selected solution

Robust solution

- Gaia always outside the Hill's curves with respect to the L_2 point

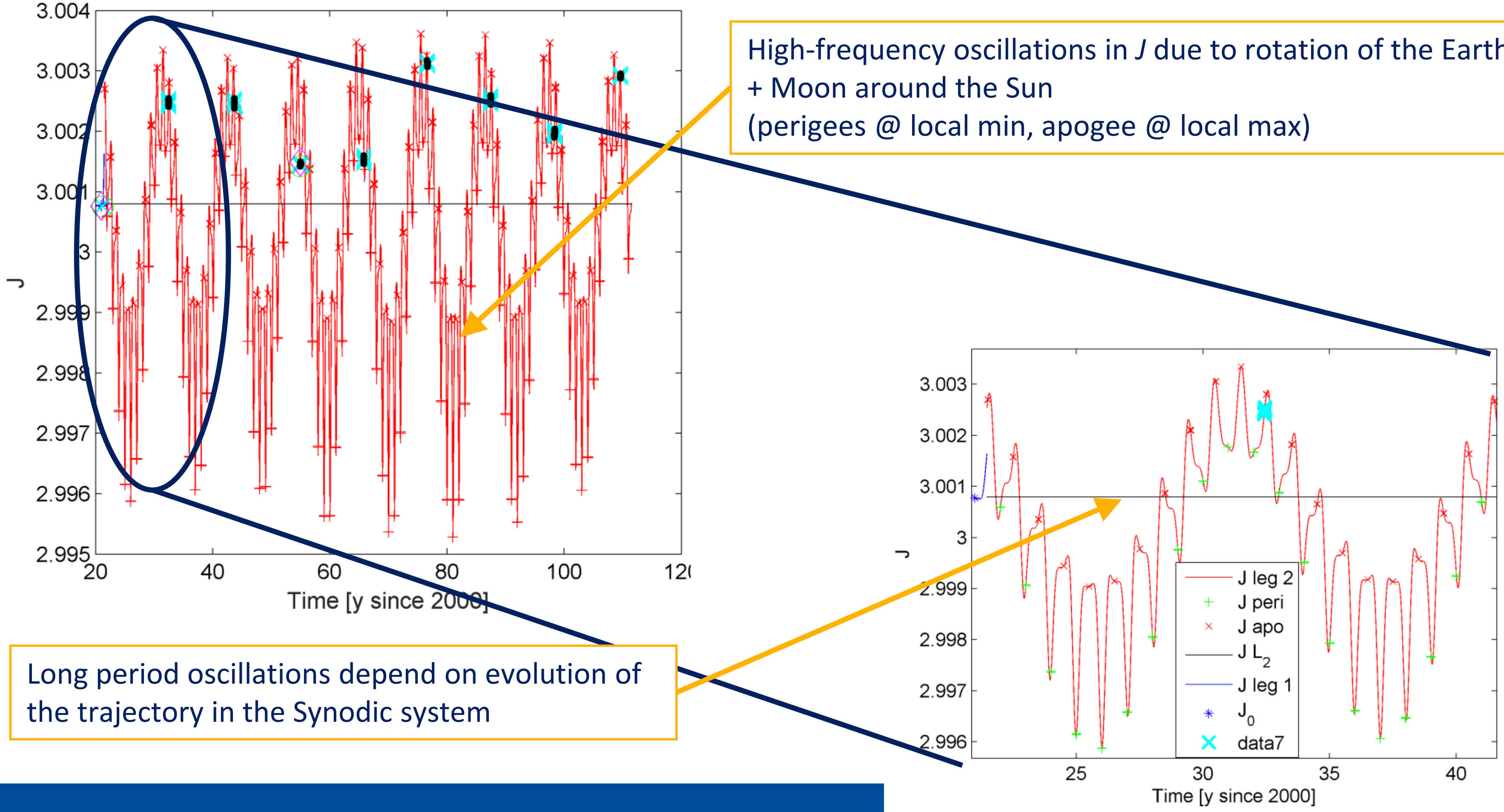




Heliocentric disposal

Selected solution

Design parameters $\mathbf{y} = \left[0.3561 \quad 37.74\text{ deg} \quad -0.16\text{ deg} \quad 173.38\text{ days}\right]$



High-frequency oscillations in *J* due to rotation of the Earth + Moon around the Sun (perigees @ local min, apogee @ local max)

Long period oscillations depend on evolution of the trajectory in the Synodic system

Heliocentric disposal

Selected solution

Sun-centred inertial system

- Disposal manoeuvres moves the spacecraft on an orbit which is far the Earth orbit.
  Leg 1 = transfer orbit,
  Leg 2 = “final” orbit
- Proposed optimisation is like Maximising Minimum Orbit Interception Distance

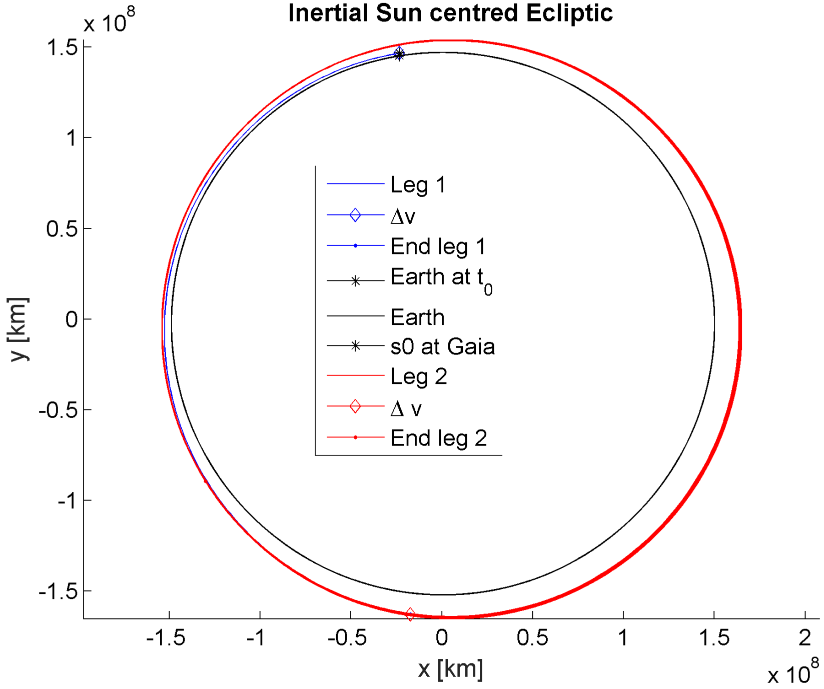
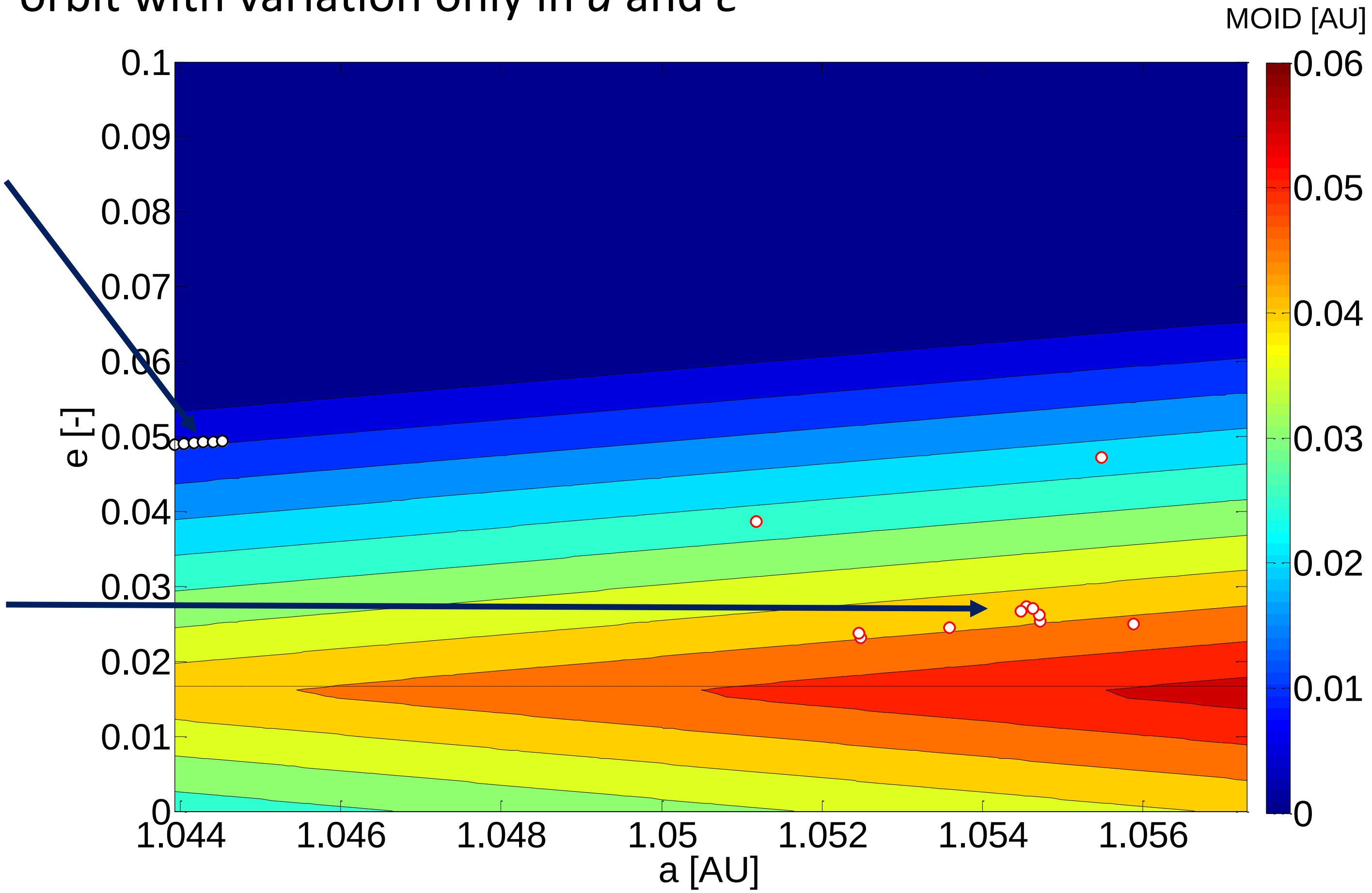


Figure 1. Gaia heliocentric disposal on 7669 MJD2000: trajectory in the Sun-centred ecliptic inertial system.

MOID variation with (*a*,*e*)

Variation of the MOID between the orbit of the Earth around the Sun and an aligned/coplanar orbit with variation only in *a* and *e*

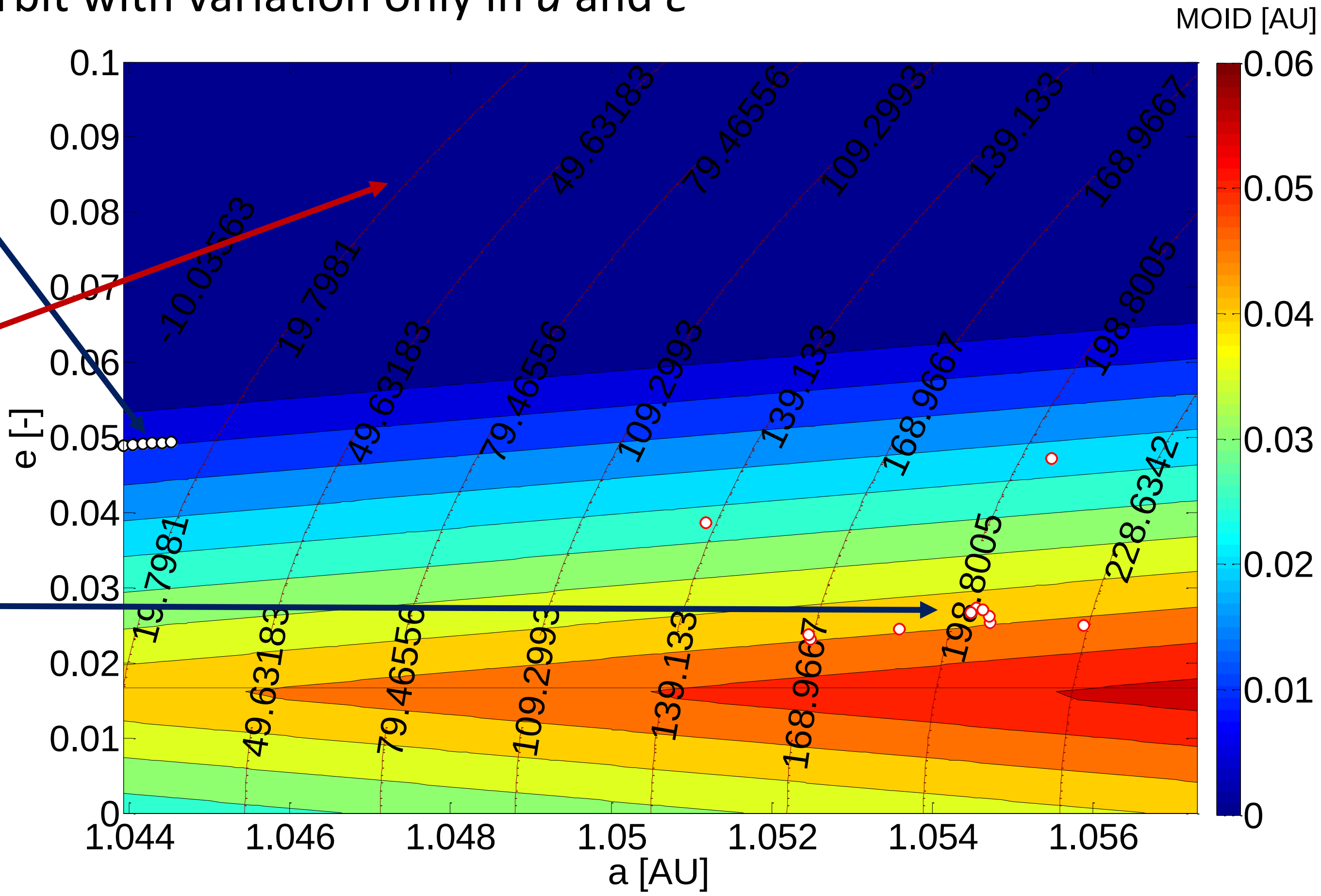


Initial GAIA orbit

Final orbits after optimisation

MOID variation with (*a*,*e*)

Variation of the MOID between the orbit of the Earth around the Sun and an aligned/coplanar orbit with variation only in *a* and *e*



Initial GAIA orbit

Required Δv to reach the new final orbit.

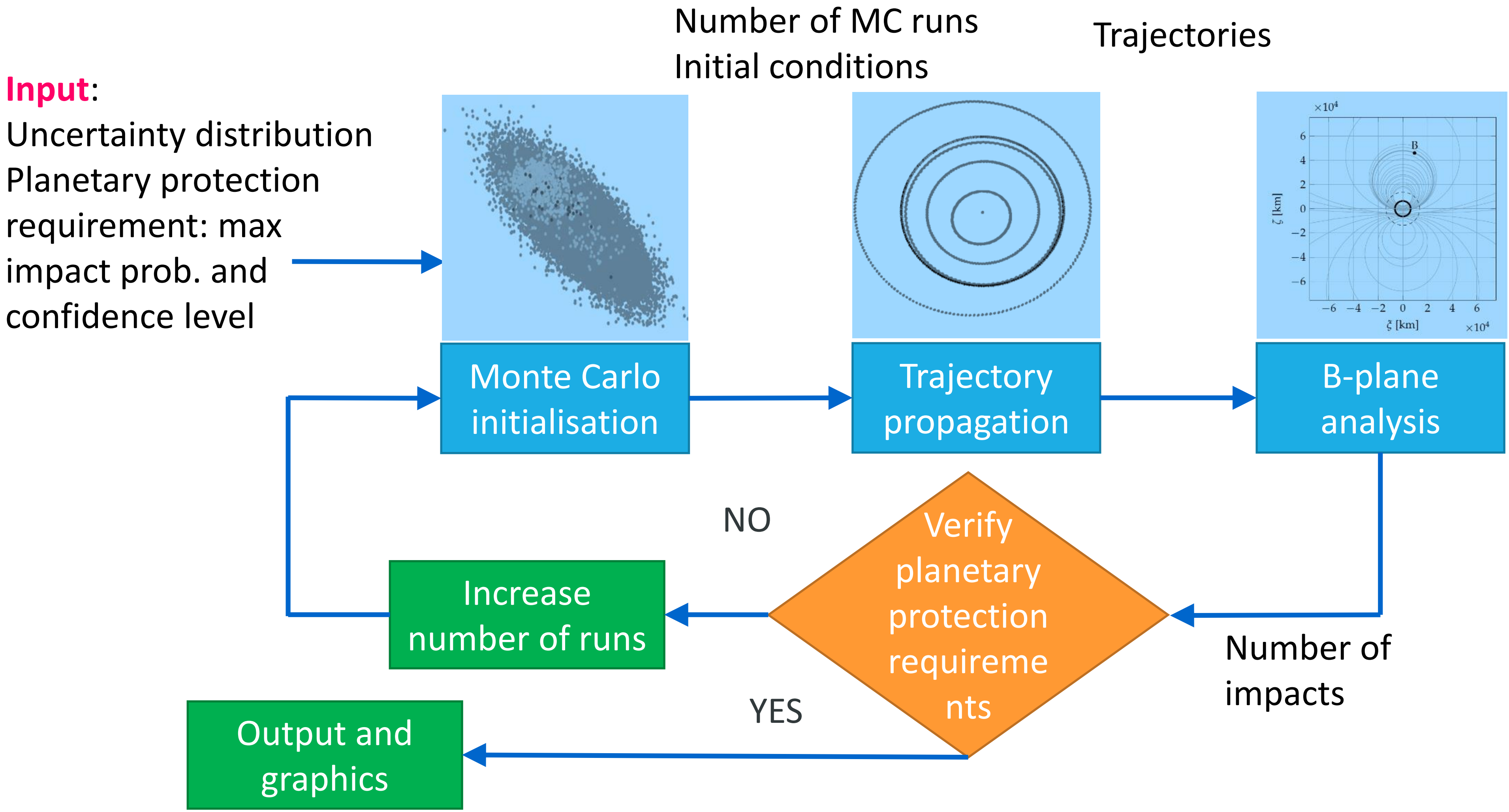
Final orbits after optimisation

Verification with the SNAPPshot suite

PLANETARY PROTECTION

SNAPPshot

Suite for Numerical Analysis of Planetary Protection



Input:
Uncertainty distribution
Planetary protection requirement: max impact prob. and confidence level

Number of MC runs
Initial conditions

Trajectories

Monte Carlo initialisation

Trajectory propagation

B-plane analysis

Number of impacts

Verify planetary protection requirements

NO

Increase number of runs

YES

Output and graphics

Monte Carlo initialisation

Defining the number of runs

- The output of the Monte Carlo (MC) run is treated as a binomial variable, with the two binary states impact/no impact

$$\boxed{X \sim \mathrm{B(n,p)}}$$

X= number of impacts
B = Bernoulli distribution
n = number of independent trials
p = probability of impact in each trial

- Estimate the **confidence interval** with a normal distribution with mean μ=np and variance $\sigma = np(1-p)$

$$p = (\hat{p}, c)$$

$\widehat{p}$= probability of success estimated from the statistical sample (i.e. $\hat{p} = n_I/n$)
n_I= number of impacts
c = confidence level

➢ Lawrence D. Brown, 2001

Monte Carlo initialisation

Defining the number of runs

- **Wilson's confidence interval** preferred: define the interval looking at the value of *p* that would put $\hat{p}$ at the **extremes** of the confidence interval

$$\boxed{p \leq \left(\frac{\hat{p} + \frac{z^2}{2n} + z^2 \sqrt{\frac{\hat{p}(1-\hat{p})}{n} + \frac{z^2}{4n^2}}}{1 + \frac{z^2}{n}} \right)}$$

$\hat{p} =$ probability of success estimated from the statistical sample (i.e. $\hat{p} = n_I/n$)

$z = \propto$ quantile from a standard normal distribution

Aim: estimating the **minimum number of MC runs** (*n*) required to verify the compliance with the planetary protection requirements: verify the maximum level of **impact probability** (*p*), with a level of **confidence** (α)

Input: **impact probability** (*p*), **confidence** (α)

minimum number of MC runs (*n*)

- Wilson (1927), Jehn (2015), Wallace (2015)

Application of the tool

Effects of initial uncertainty distribution

Dispersion of the initial condition:

- Launcher inaccuracy

Input: **6 x 6 Covariance matrix** describing the dispersion of the escape velocity and position of injection

- Failure of the propulsion system

Input: random failure time within an **interval**

- Uncertainty on spacecraft parameters (e.g. unknown area-to-mass ratio)

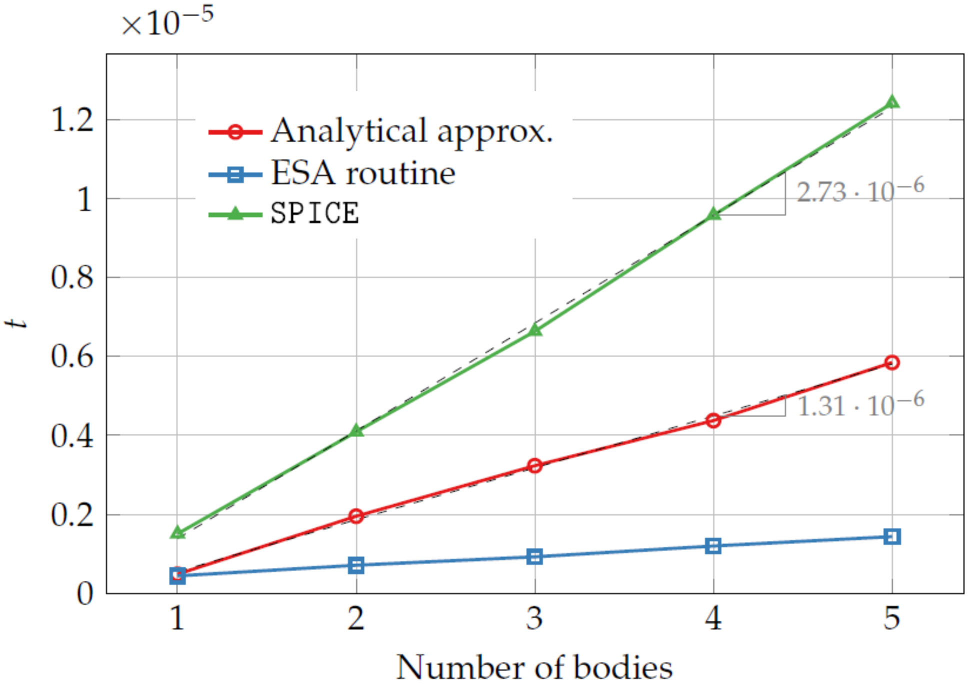
Input: Distribution can be selected (e.g., uniform, triangular) and known values

Trajectory propagation

Dynamical model

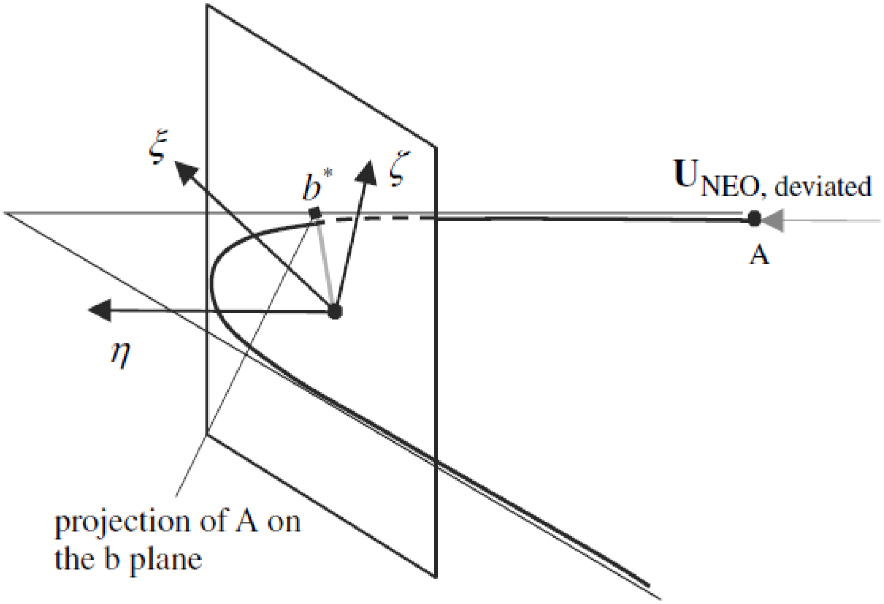
- Cartesian coordinates centred in the Solar System Barycentre J2000
- Dynamics of n planets and solar radiation pressure with cannonball model
- Ephemerides
  - Analytical ephemerides
  - ESA routine based on DE422
  - NASA SPICE
- Normalisation in dimensionless variables

$$\hat{L} = \frac{L}{AU} \qquad \hat{t} = \frac{t}{2\pi\sqrt{AU^3/\mu_{SUN}}}$$



Planet close-approach representation

B-plane definition



Plane **orthogonal** to the object **planetocentric velocity** when the object enters the planet’s sphere of influence

- η-axis: parallel to the planetocentric velocity
- ζ-axis: parallel to the projection on the b-plane of the planet velocity, but in the opposite direction
- ξ-axis: to complete a positively oriented reference system

- Intersection of the **incoming asymptote** and the b-plane: $\boldsymbol{b}^* =$ impact parameter
- $\eta = 0$ on the b-plane identifies a **fly-by**

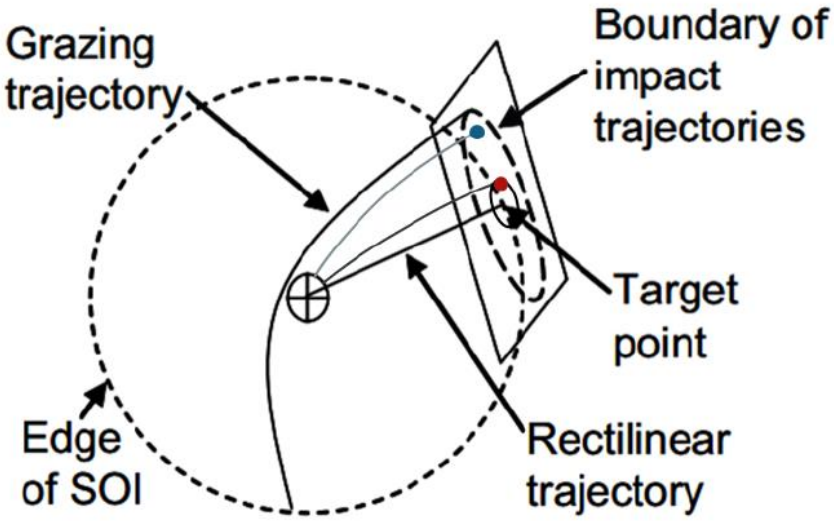
- (Öpik, 1976)
- Vasile and Colombo, 2008

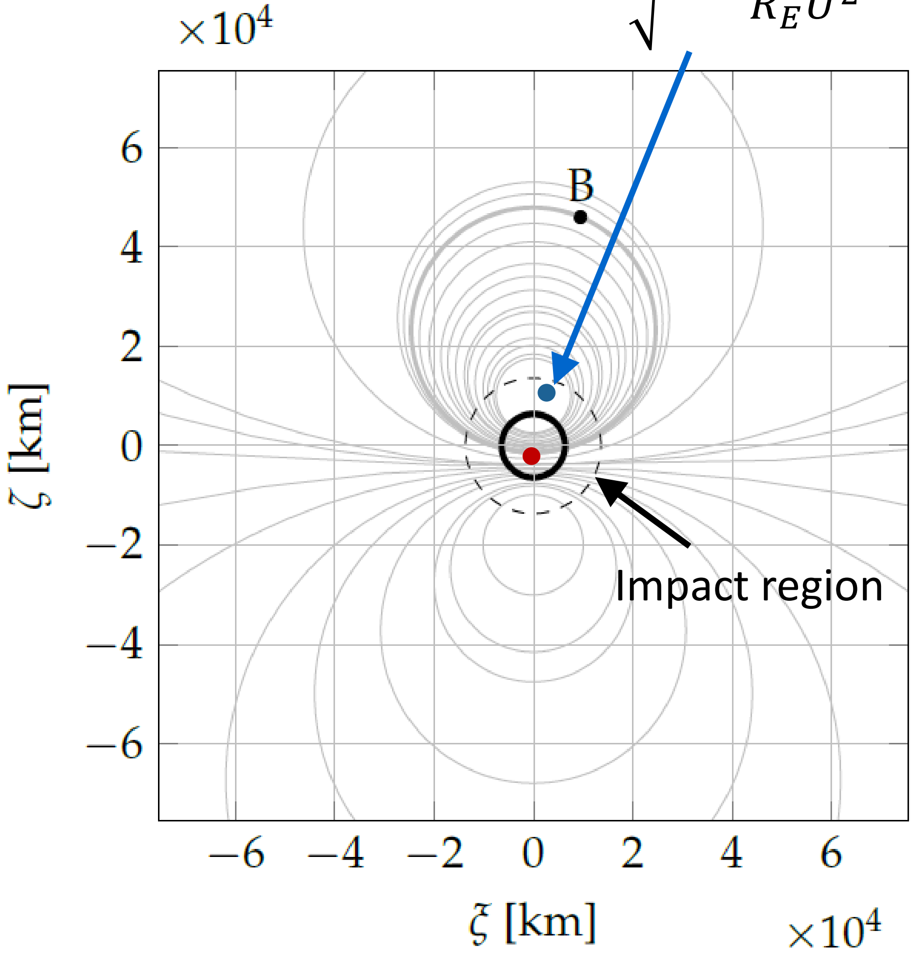
Planet close-approach representation

State characterisation

- Impact
- Gravitational focussing

$$R_{GF} = R_E \sqrt{1 + \frac{2\mu_E}{R_E U^2}}$$





Planet close-approach representation

State characterisation

- Resonance:

Circle on the b-plane $\xi^2 + \zeta^2 - 2D\zeta + D^2 = R^2$

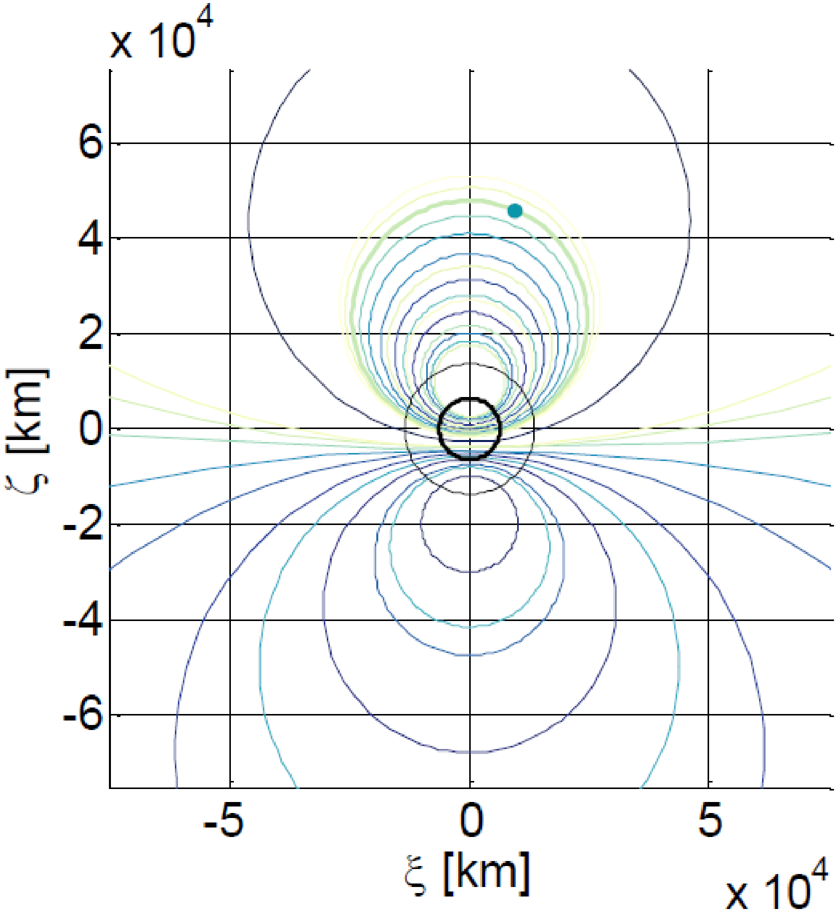
Requirement: Tisserand criterion < 3

➢ Valsecchi et al. (2003)

For a given close encounter, the **post-encounter semi-major axis** is computed. The resulting period is compared to the ones of **possible** resonances.

The **severity** measured by the value of *k* (planet's period repetitions): the lowest, the most critical.
Resonance selection: closest resonance or resonance with the lowest *k* (and below the period threshold)

Resonance plotted according to their k value: dark low k, light low k

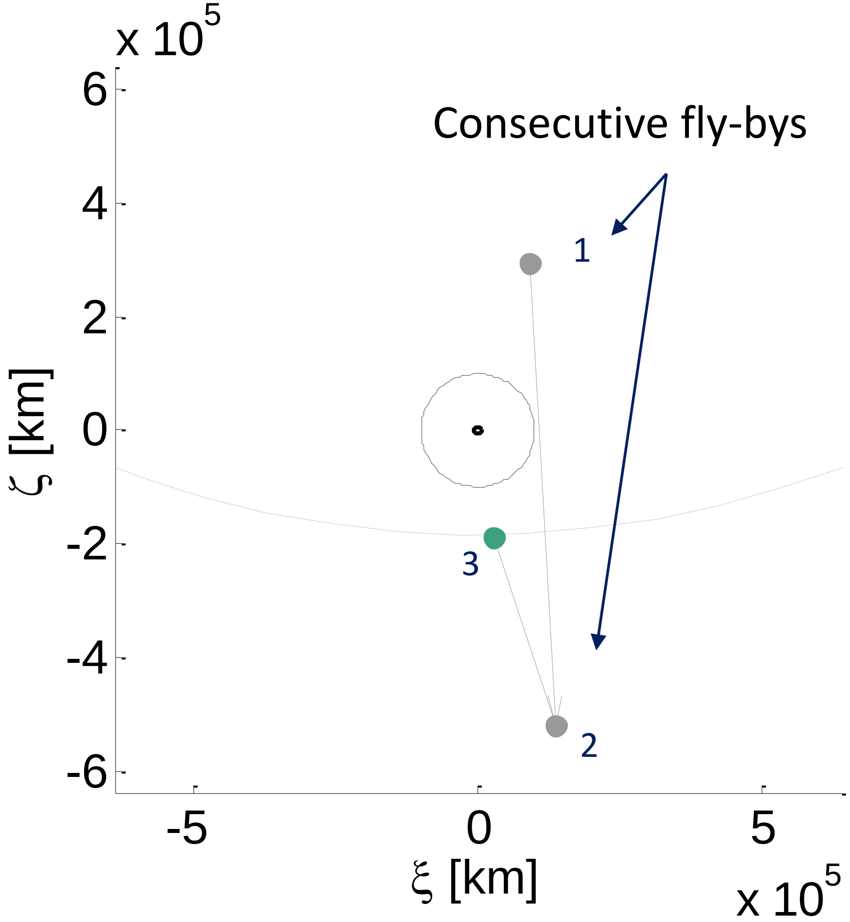


B-plane analysis

Close-encounter sorting

- When **multiple fly-bys** are recorded, for the Monte Carlo analysis only one state should be selected to characterise the trajectory. Two implemented options:
  - first encounter
  - worst encounter.
- Multiple encounters are **sorted** sorting = identify the most critical ones (e.g. impact with Earth > resonance with Mars
  - Distance-driven: worst case is the one with the minimum distance from the Earth
  - State-driven: impact > resonance > simple close approach Earth > Mars > Venus

Evolution of one GAIA Fregat trajectory on the Earth's b-plane for 100 years of propagation

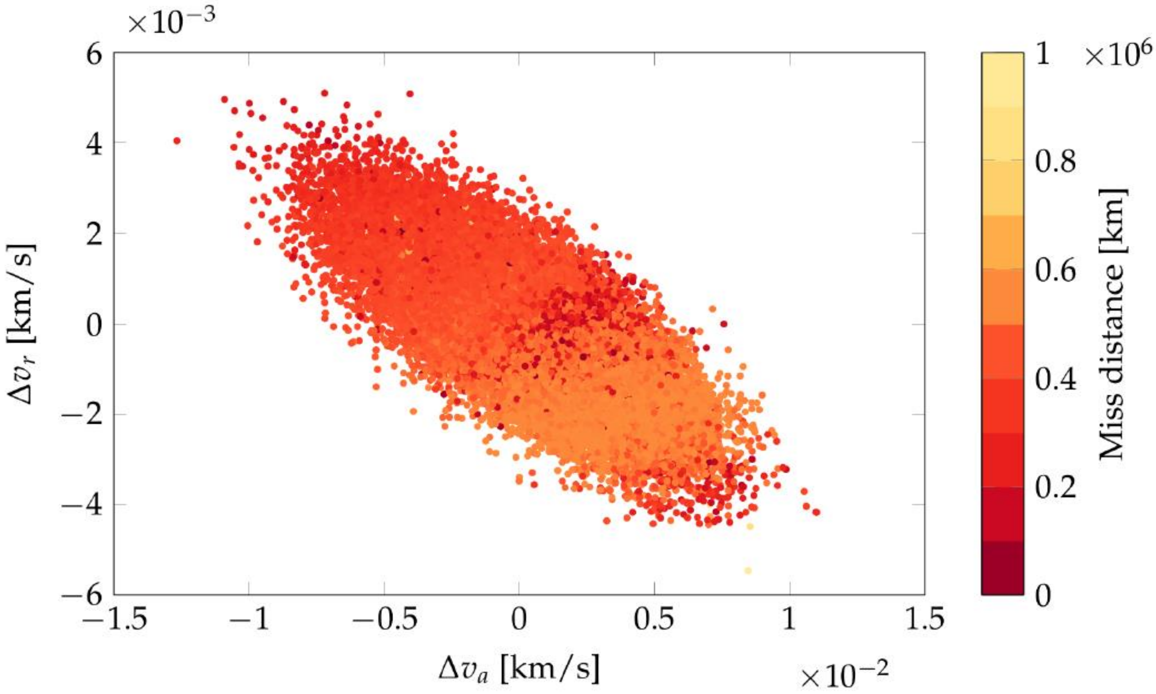


Results

Effect of launcher dispersion: Ariane launcher of BepiColombo

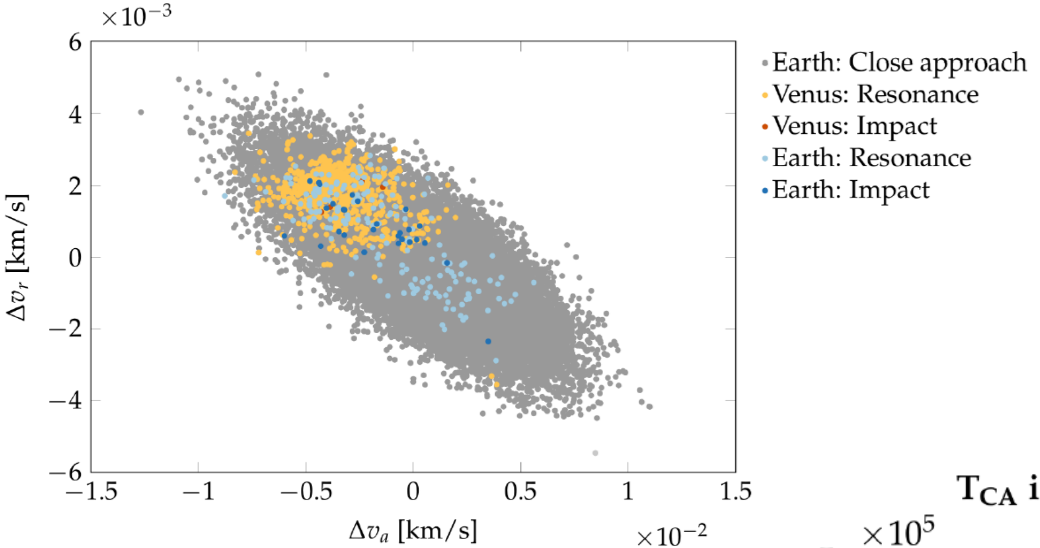
- Uncertainty: **state dispersion (covariance matrix)** and **area-to-mass** ratio distribution (triangular distribution)
- Propagation: time 100 years, RK8(7)

- Number of runs: 54114 (the minimum amount of runs required to prove that the object does not impact with a selected planet, with a confidence level of 99%)

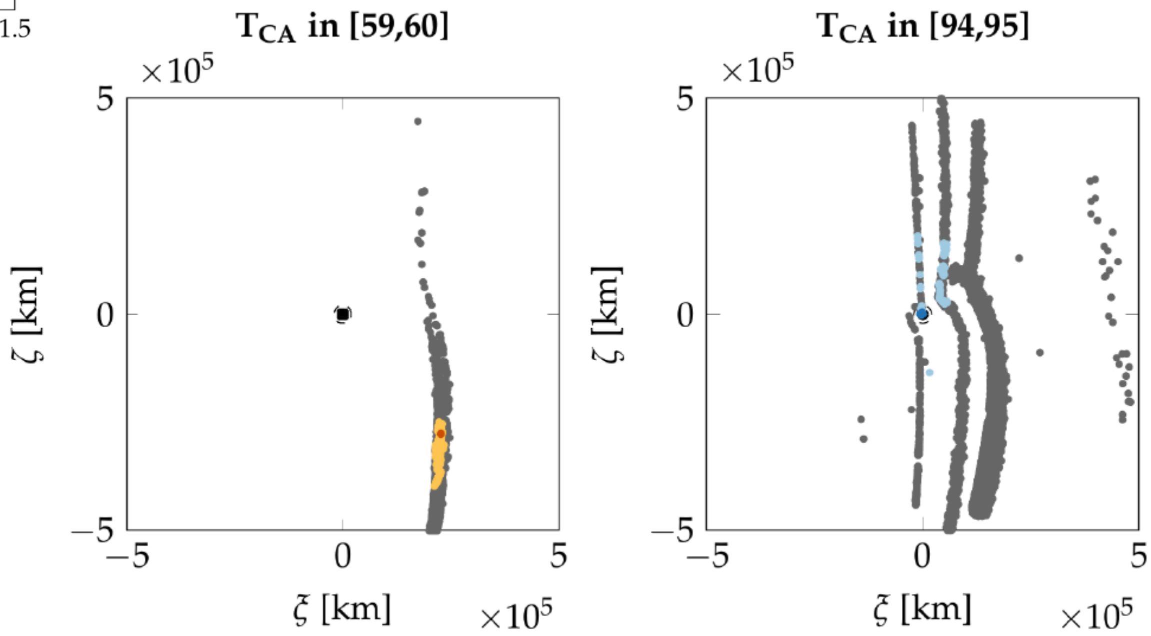


Results

Effect of launcher dispersion: Ariane launcher of BepiColombo

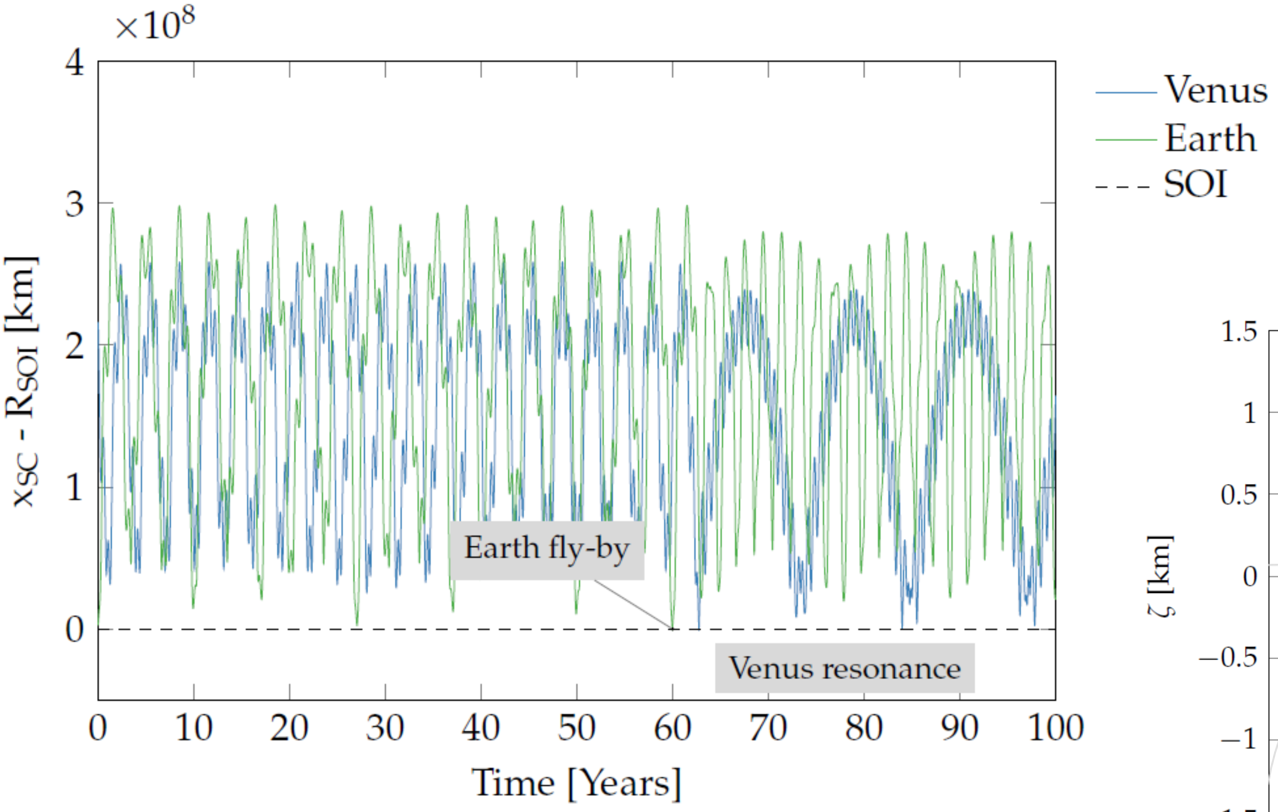


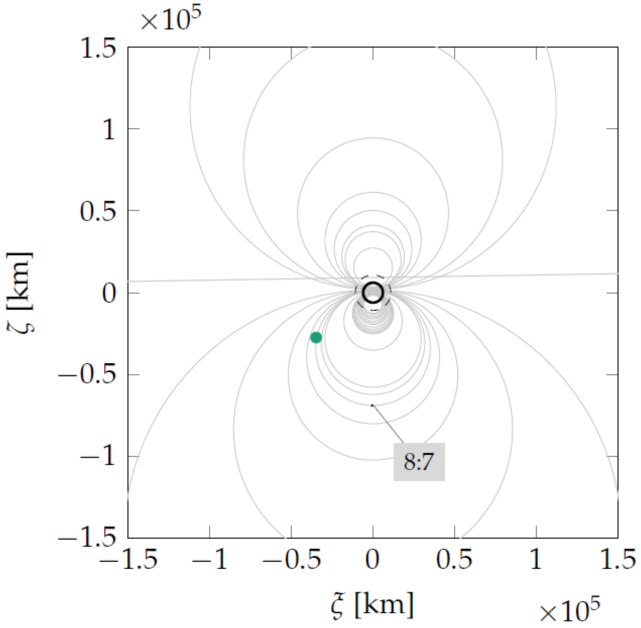
Number of impacts:
4 Venus, 28 Earth



Results

Effect of launcher dispersion: Ariane launcher of BepiColombo

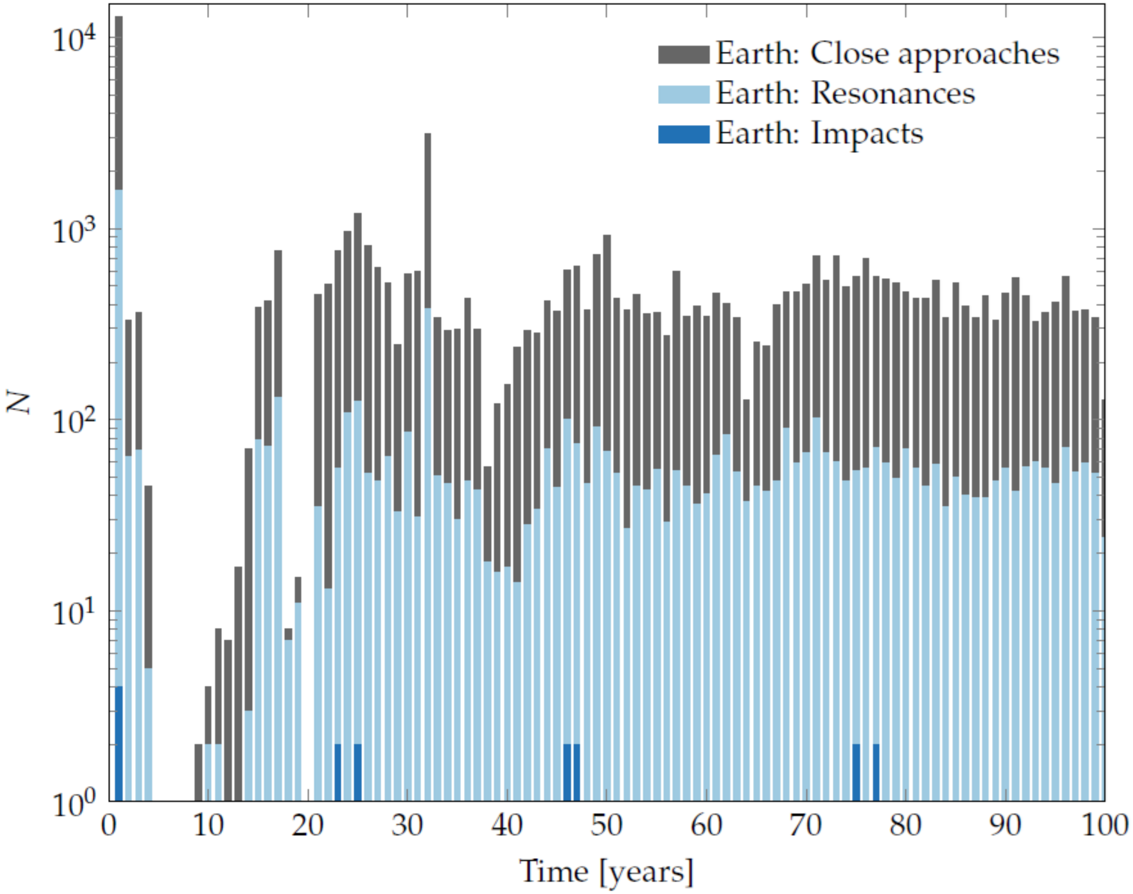




Results

Effect of failure of propulsion system: BepiColombo

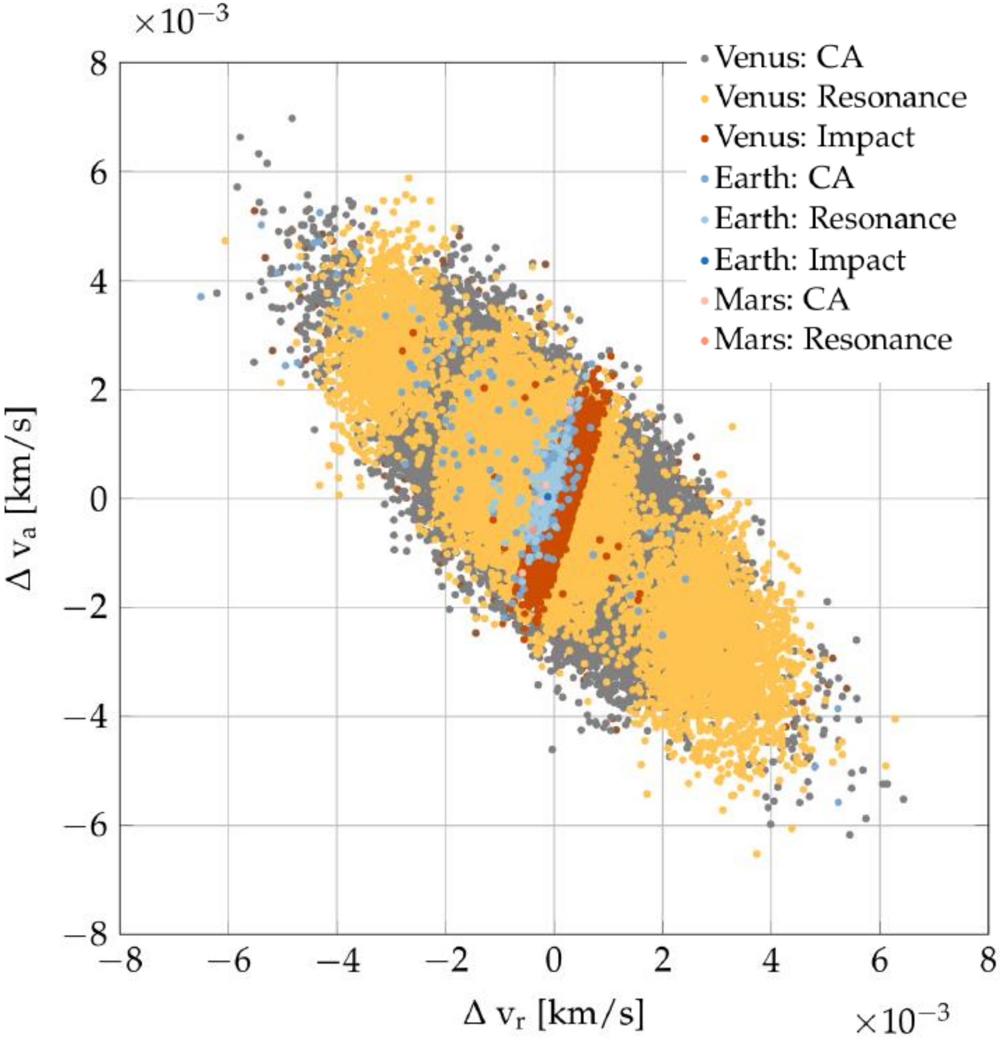
- Uncertainty: **state dispersion** following failure of propulsion system
- Propagation: time 100 years, RK8(7)
- Number of runs: 54114 (the minimum amount of runs required to prove that the object does not impact with Mars of 10^{-4}, with a confidence level of 99%)
- Number of impacts: **28 Earth**



Results

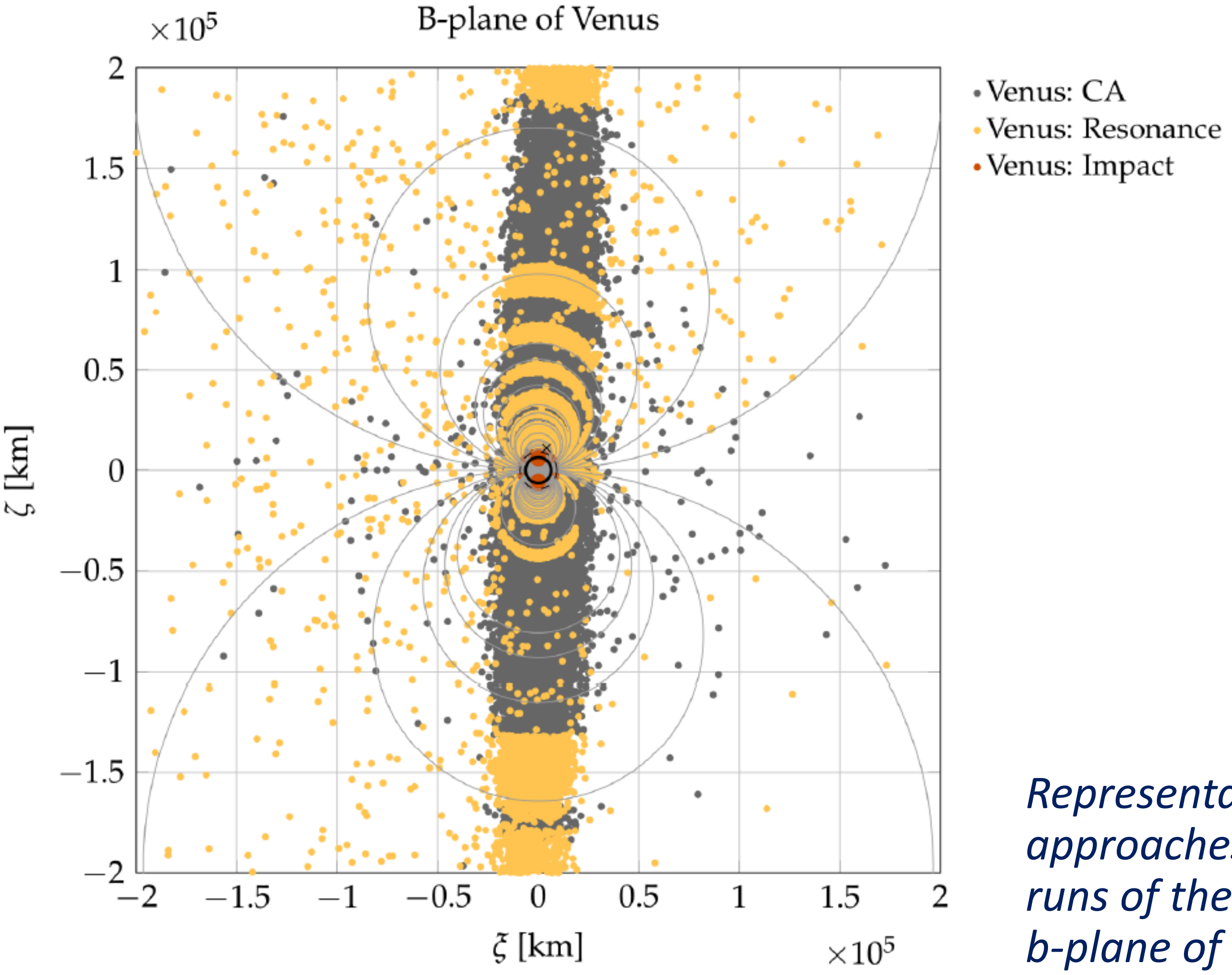
Effect of launcher dispersion: Solo launcher

- Uncertainty: **state dispersion (covariance matrix)**
- Propagation: time 100 years, RK8(7)
- Number of runs: 54114 (the minimum amount of runs required to prove that the object does not impact with a selected planet, with a confidence level of 99%)
- Number of impacts: **4 Earth**, **2348 Venus**



Results

Effect of launcher dispersion: Solo launcher



Representation of the worst close approaches for the 1000 Monte Carlo runs of the launcher of Solo on the b-plane of Venus.

Insights into planetary protection analysis and tool enhancement

SNAPPSHOT EXTENSION

Integration methods

Symplectic methods

- Planetary protection analysis involves long-term orbital propagations (up to 100 years)
- Numerical methods accumulate errors during the integration
  - This may cause the constants of motion (e.g. energy) to change in time, obtaining a bad estimate of the spacecraft state
- Alternative numerical approaches may be beneficial to the accuracy of the orbital propagation
  - **Symplectic schemes** ensure that the constants of motion are conserved exactly or have a variation bounded in time
  - Additional methods can **"force" the conservation of those quantities**

Integration methods

Symplectic schemes

Method	(Order, Stages)	Type	Time step	Property	Other
RK4	(4,4)	Explicit	Fixed step	**(3)**	+ projection method
RK45	(5/4,7)	Explicit	Variable step	**(3)**	+ projection method
RK8	(8,13)	Explicit	Fixed step	**(3)**	+ projection method + regularised step
RK78	(8/7,13)	Explicit	Variable step	**(3)**	+ projection method
GLRK4 **GLRK6** **GLRK8**	(4,2) (6,3) (8,4)	**Implicit**	Fixed step	**Symplectic (1)**	+ projection method + regularised step
RKN8	(8,26)	Explicit	Fixed step	**Symplectic (1)**	+ projection method + regularised step
RKN64	(6/4,6)	Explicit	Variable step	**Symplectic (1)**	+ projection method
SY4 **SY6** **SY8**	(4,4) (6,8) (8,16)	Explicit	Fixed step	**Symplectic, canonical (2)**	+ projection method + regularised step

Physical model update

Fly-by detection through Jacobian

Aim: use projection only in correspondence of a fly by to save computational time → **Fly by detection with Jacobian**

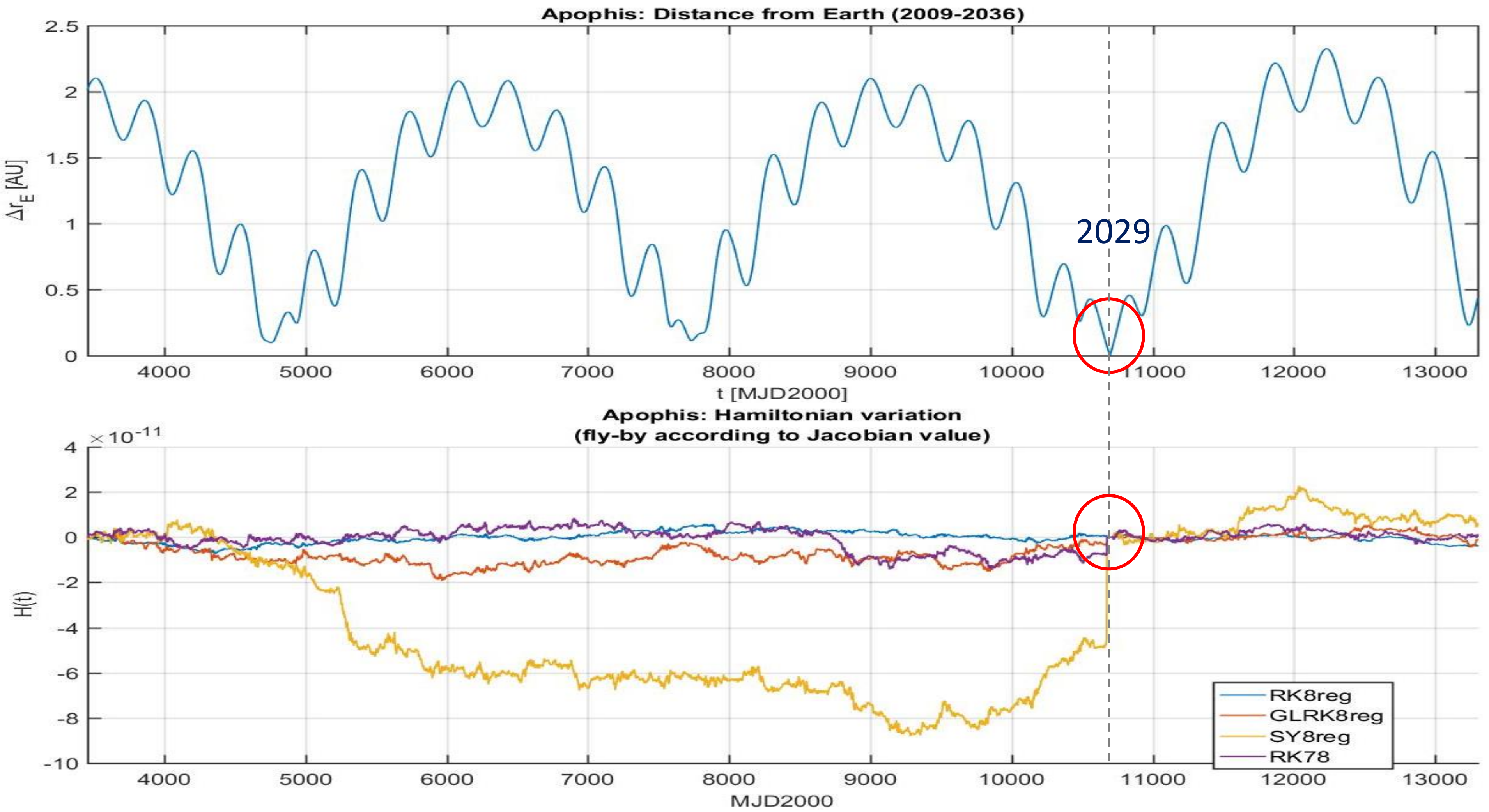
- Eigenvalues of the whole Jacobian: $\lambda^2 = eigs(GI) = eigs(G), \quad \Lambda = \max(\lambda)$
- Body alone contribution: $\lambda_j^2 = eigs(G_j), \quad \Lambda_\mathrm{j} = \max(\lambda_j) \qquad \lambda^2 \neq \sum(\lambda_j^2)$
- Value of planet contribution (grows approaching to the planet) $$\Lambda_\mathrm{j} = \frac{2\mu_j}{\left|\underline{r} - \underline{r}_j\right|^3}$$
- Time variation of planet contribution (grows approaching to the planet) $$\dot{\Lambda}_\mathrm{j} = 2\mu_j \frac{3(\underline{r} - \underline{r}_j)(\underline{v} - \underline{v}_j)}{\left|\underline{r} - \underline{r}_j\right|^5}$$
- Fly-by detection criteria (approximation)

Relative value w.r.t. main attractor: $\Lambda_j/\Lambda_{Sun} \geq Tol\ e.g.\ 10^{-1}$

Relative variation w.r.t. main attractor: $\dot{\Lambda}_\mathrm{j}/\dot{\Lambda}_\mathrm{Sun} \geq Tol\ e.g.\ 10^{-1}$

Physical model update

Fly-by detection through Jacobian



Advanced sampling techniques

Monte Carlo approach

- Verification that planetary protection requirements are satisfied implies a **large number of long-term orbital propagations** with standard Monte Carlo Simulations

- More efficient sampling methods may reduce the amount of propagations and the computational cost
- The **Line Sampling** method probes the impact region of the uncertainty domain by using lines instead of random points
  - This generally improves the estimation of impact probability and reduces the amount of random samples required

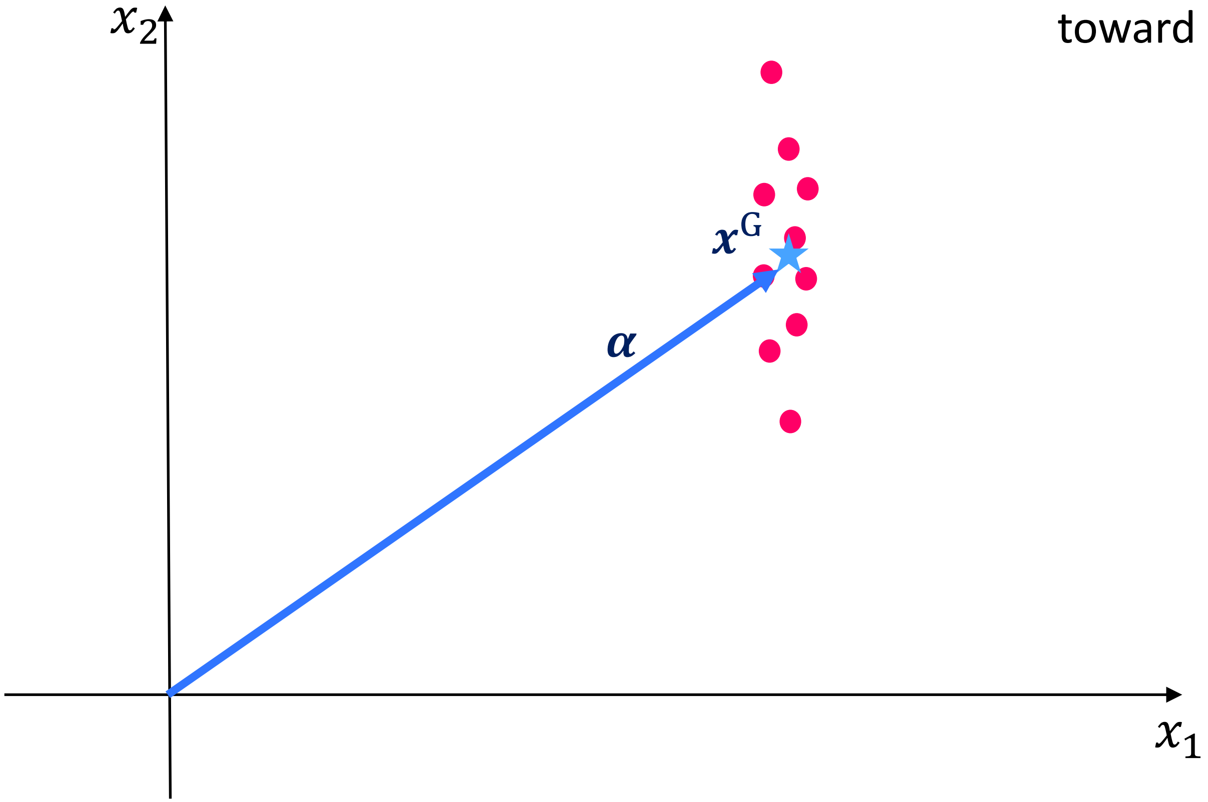
Sampling techniques

Line Sampling

1. **Determination of the "reference direction"**
   a. Impact region not known a priori

Sampling techniques

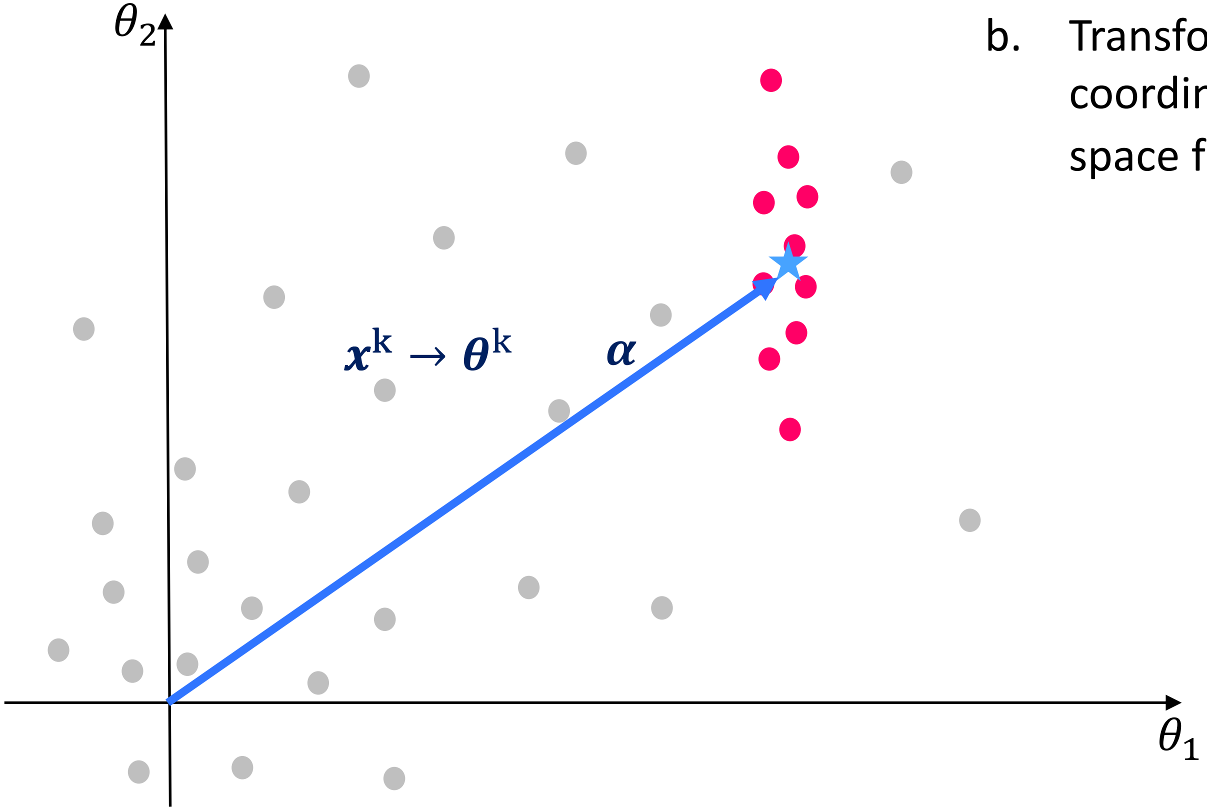
Line Sampling



1. **Determination of the "reference direction"**
   1. Impact region not known a priori
   2. Reference direction generally pointing toward impact region

Sampling techniques

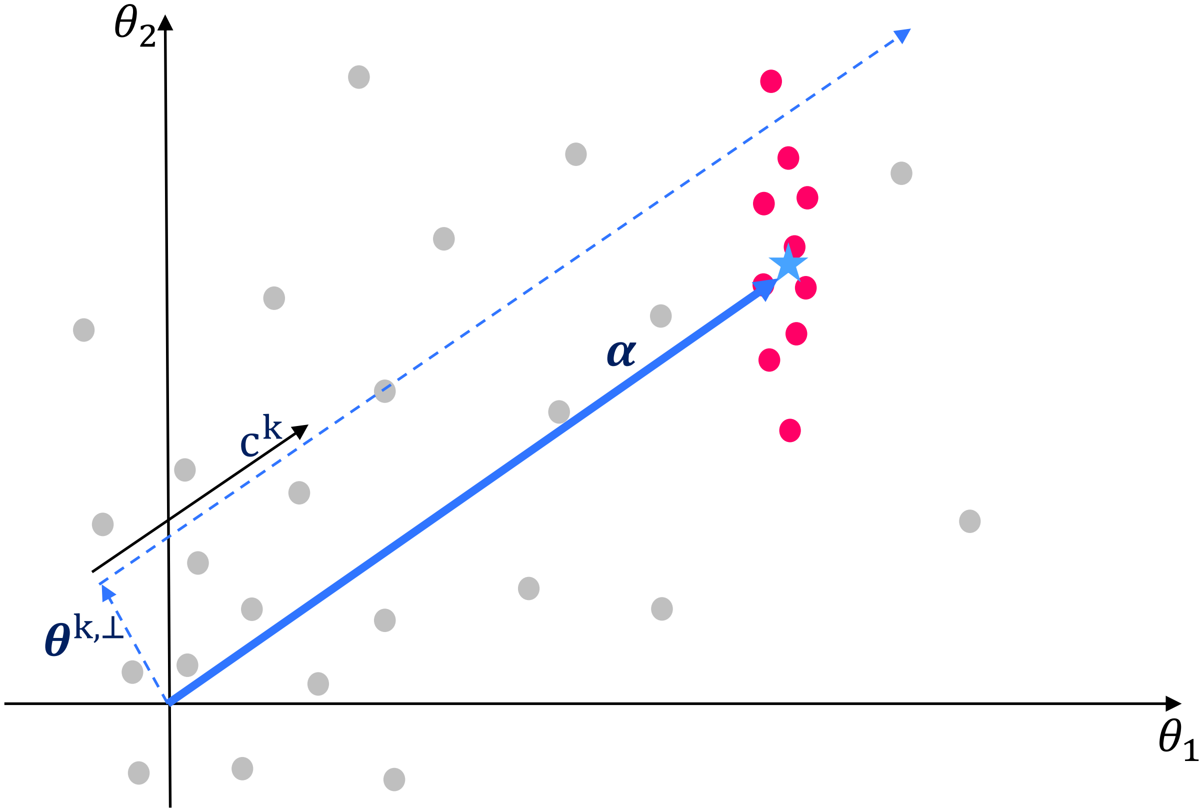
Line Sampling



2. **Mapping onto the standard normal space**
   a. Generation of random samples according to given distribution
   b. Transformation from physical coordinates into normalised standard space following $\Phi(\boldsymbol{\theta}^{\mathrm{k}}) = \mathrm{F}(\boldsymbol{x}^{\mathrm{k}})$

Sampling techniques

Line Sampling

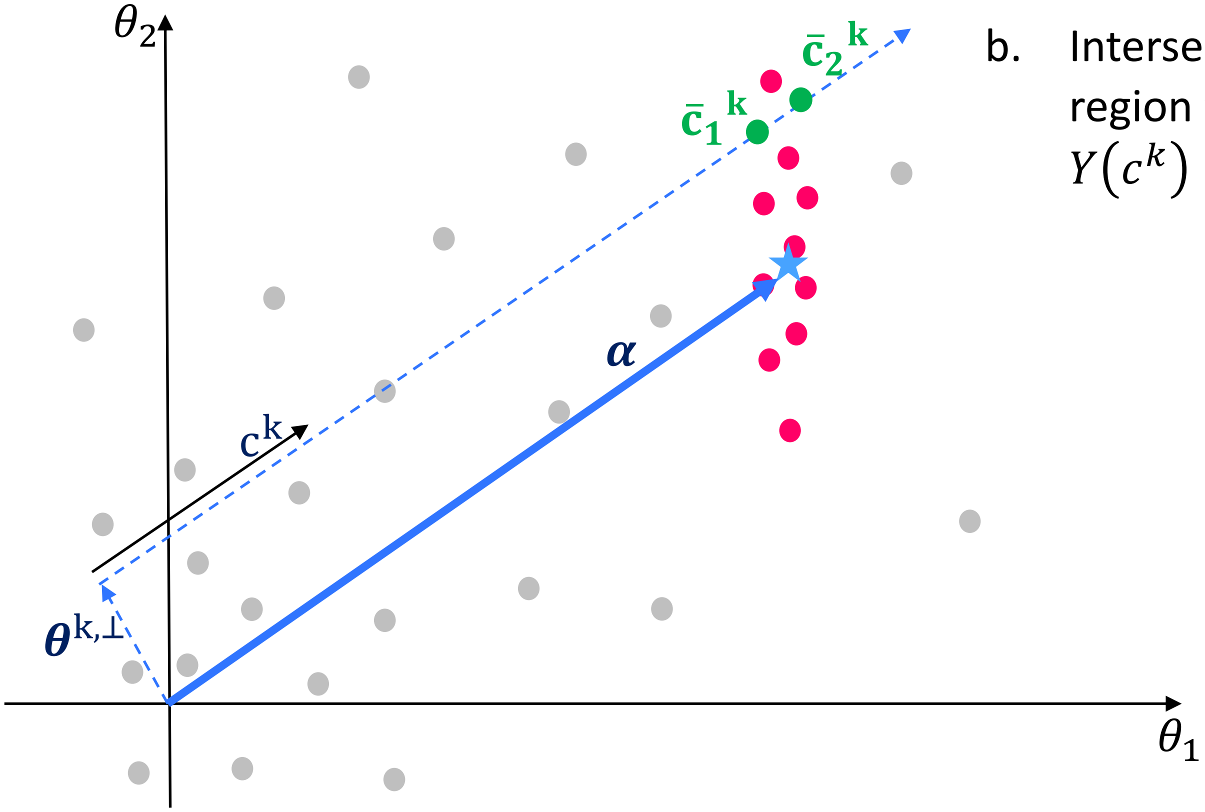


3. **Sampling along the lines**
   a. Lines defined in normalised space

$$\widetilde{\boldsymbol{\theta}}^k = c^k \boldsymbol{\alpha} + \boldsymbol{\theta}^{k,\perp}$$

Sampling techniques

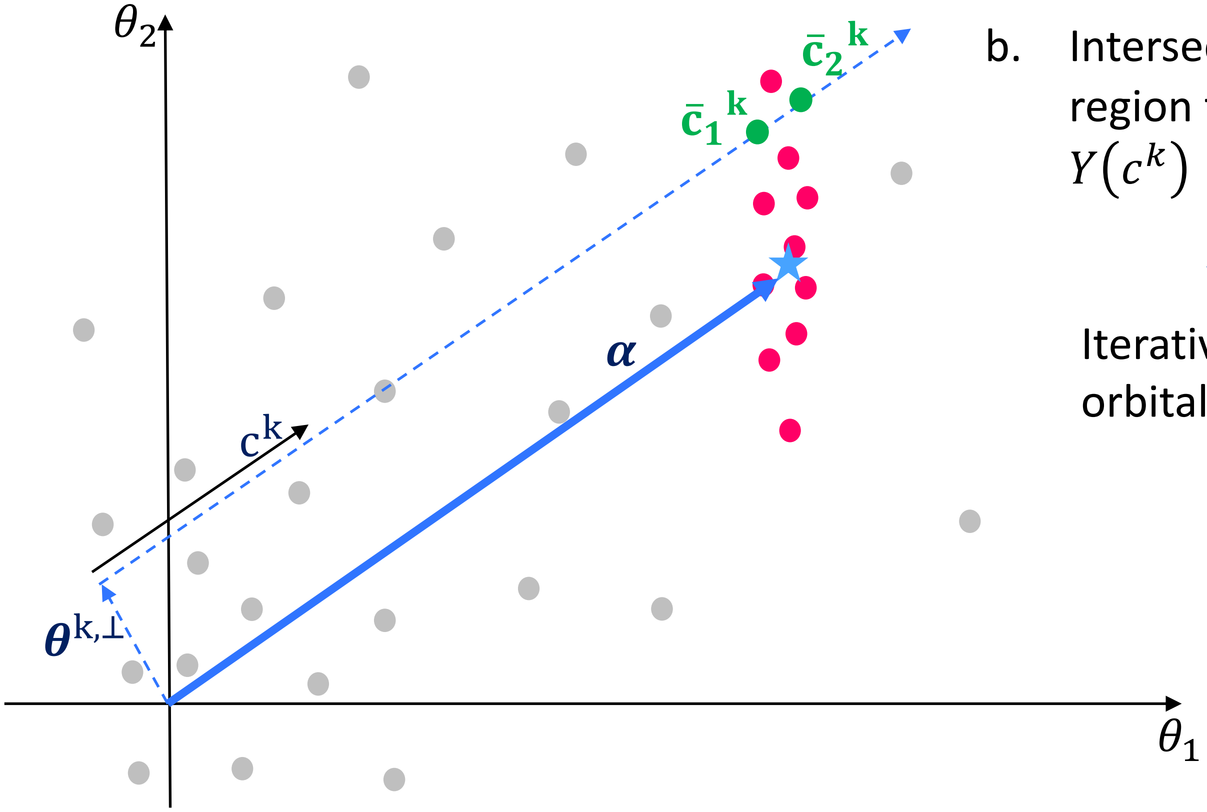
Line Sampling



3. **Sampling along the lines**
   a. Lines defined in normalised space
      $$\tilde{\theta}^k = c^k\alpha + \theta^{k,\perp}$$
   b. Intersections $\left(\bar{c}_1{}^k, \bar{c}_2{}^k\right)$ with impact region found where objective function $Y\left(c^k\right) = 0$

Sampling techniques

Line Sampling



3. Sampling along the lines

a. Lines defined in normalised space

$$\tilde{\theta}^k = c^k \alpha + \theta^{k,\perp}$$

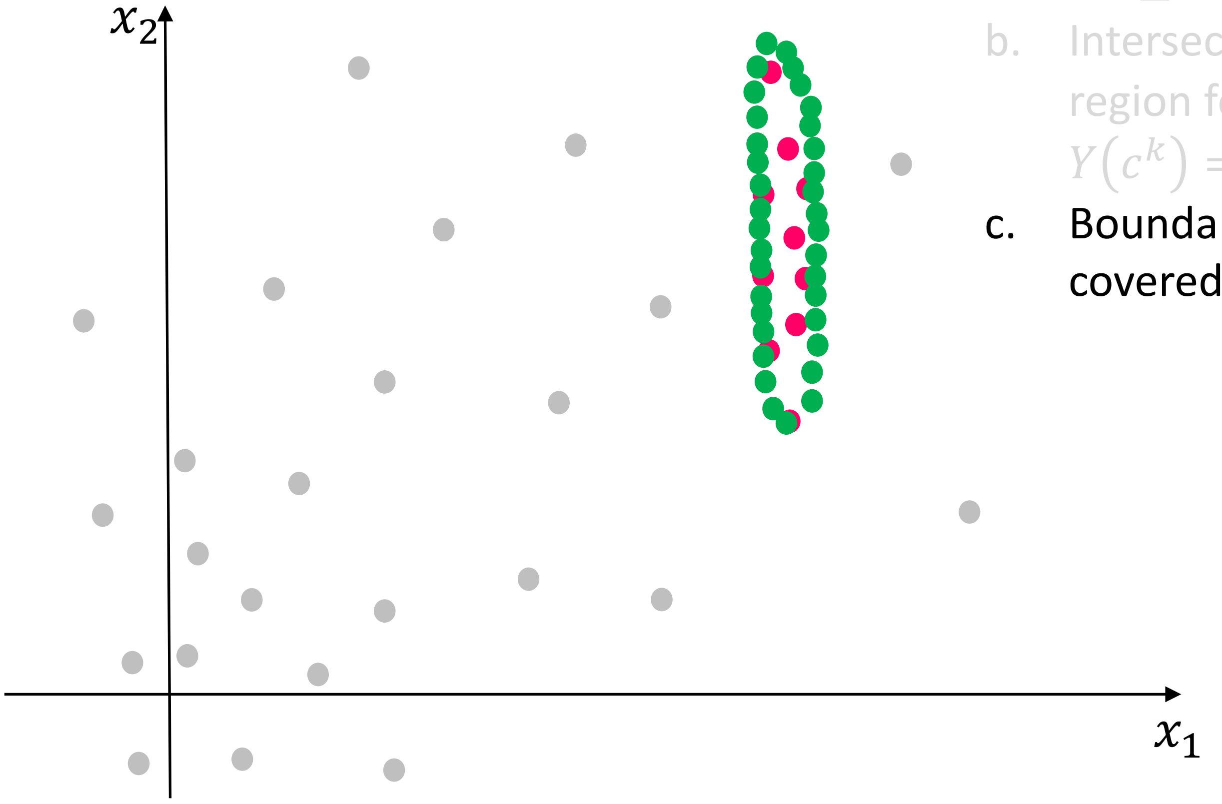
b. Intersections $(\bar{c}_1{}^k, \bar{c}_2{}^k)$ with impact region found where objective function $Y(c^k) = 0$

↓

Iterative procedure requires extra orbital propagations

Sampling techniques

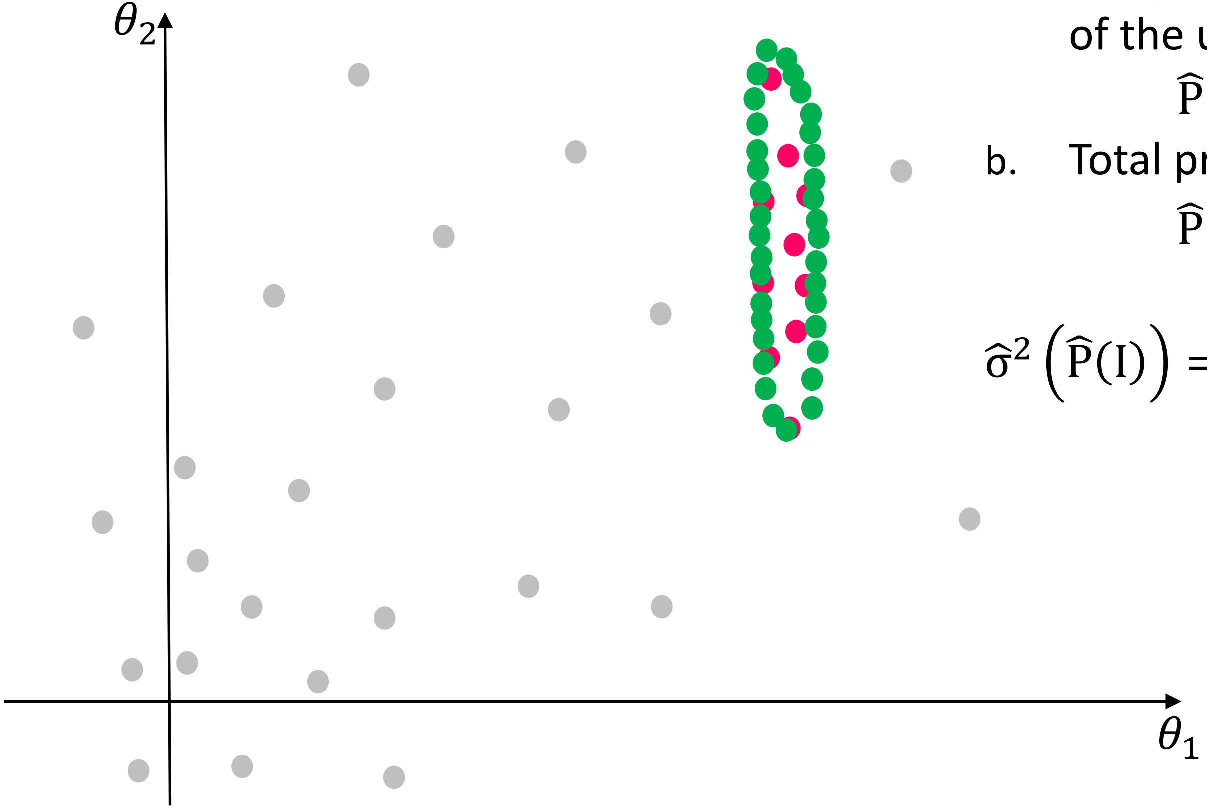
Line Sampling



3. **Sampling along the lines**
   - a. Lines defined in normalised space $$\underline{\tilde{\theta}}^k = c^k \underline{\alpha} + \underline{\theta}^{k,\perp}$$
   - b. Intersections $\left(\bar{\mathbf{c}}_1{}^{\mathbf{k}}, \bar{\mathbf{c}}_2{}^{\mathbf{k}}\right)$ with impact region found where objective function $Y\left(c^k\right) = 0$
   - c. Boundaries of the impact region are covered

Sampling techniques

Line Sampling



4. Estimation of impact probability

a. Partial probability estimates are computed along each line using the CDF of the unit gaussian:

$$\widehat{P}^k(I) = \Phi\left(\bar{c}_2{}^k\right) - \Phi\left(\bar{c}_1{}^k\right)$$

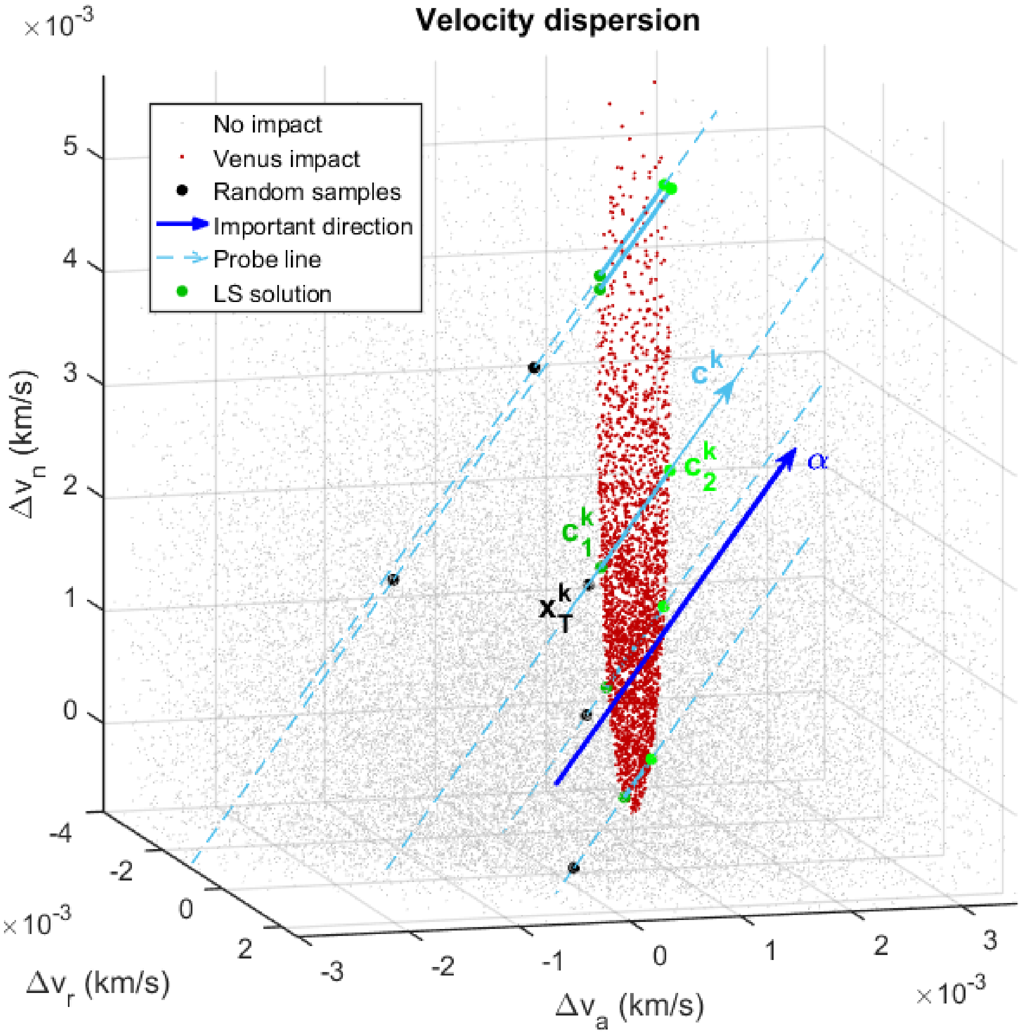
b. Total probability and variance

$$\widehat{P}(I) = \frac{1}{N_T}\sum_{k=1}^{N_T} \widehat{P}^k(I)$$

$$\hat{\sigma}^2\left(\widehat{P}(I)\right) = \frac{1}{N_T(N_T-1)}\sum_{k=1}^{N_T}\left(\widehat{P}^k(I) - \widehat{P}(I)\right)^2$$

Advanced sampling techniques

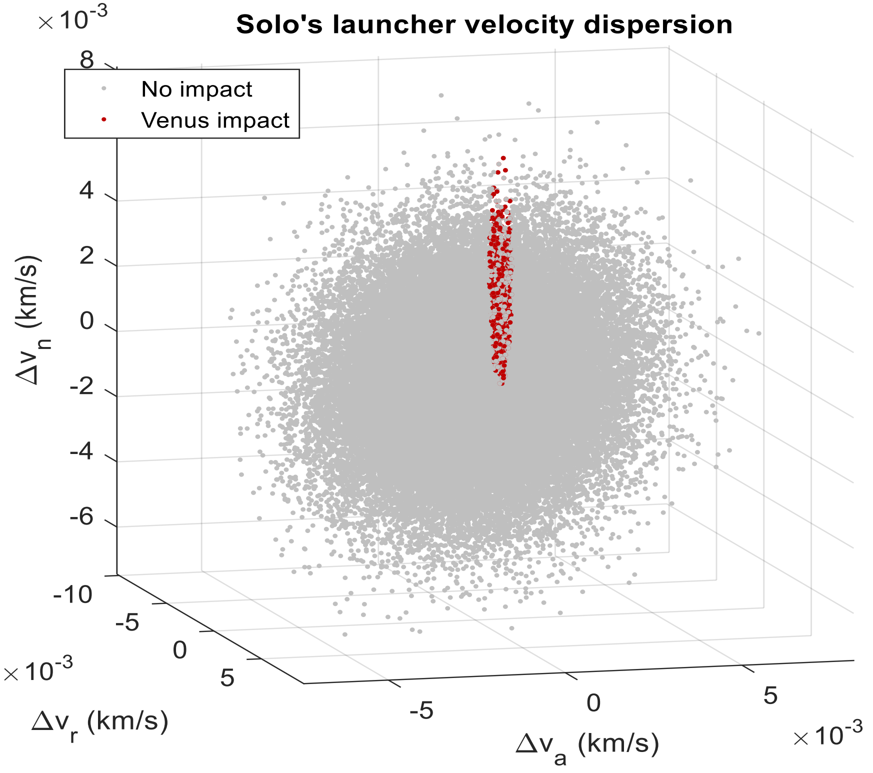
Line sampling



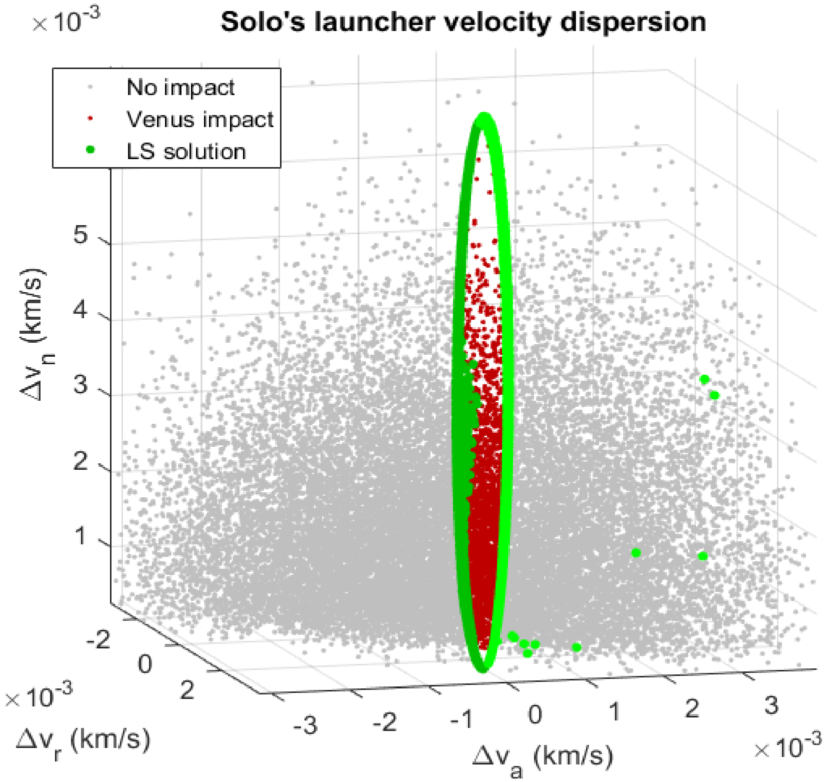
Example:
visualization of probe lines
crossing a generic impact region

Advanced sampling techniques

Results



Solution with **standard MCS**

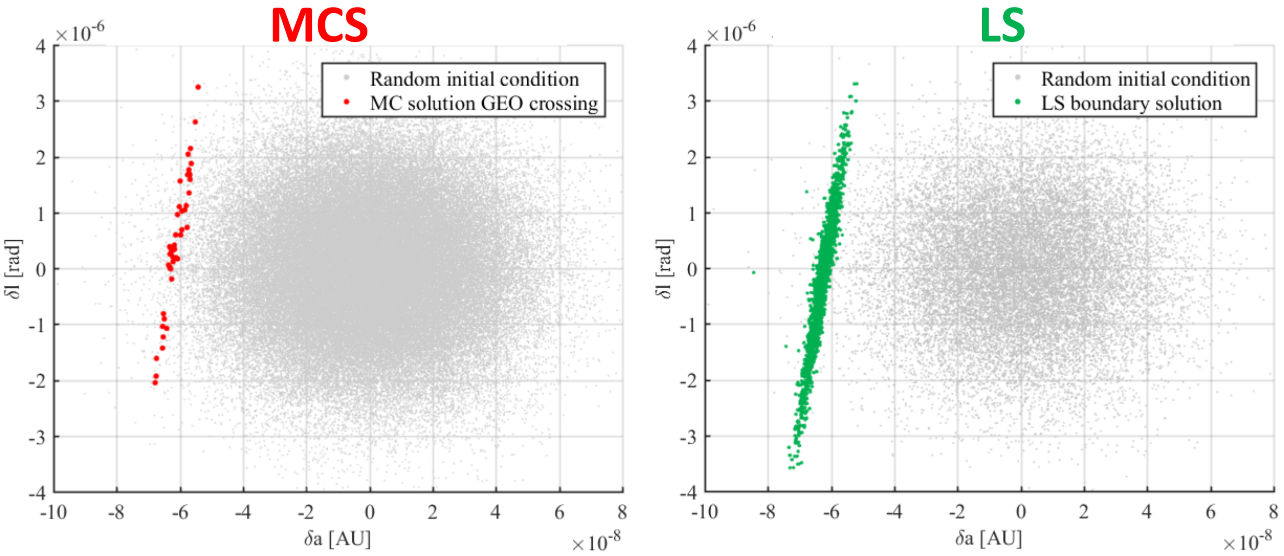


Boundaries of impact region computed with **LS**

Sampling techniques

Results: asteroid Apophis

Analysed event: expected return in 2036 (according to observations in 2009)[9]



Small expected probability
Distributed impact region

	$N_{Samples}$	N_{Prop}	$\hat{P}(I)$	$\hat{\sigma}$
MCS	1e6	1e6	5.00e-5	6.86e-6
LS	1e4	~1e5 ↓	5.38e-5	1.18e-6
	1e5	~1e6	5.32e-5	3.45e-7 ↓

Similar confidence level as MC

Similar number of orbital propagations as MC

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[9] http://newton.dm.unipi.it/neodys

* propagations performed with RK8(7) with relative tolerance 10^{-12}

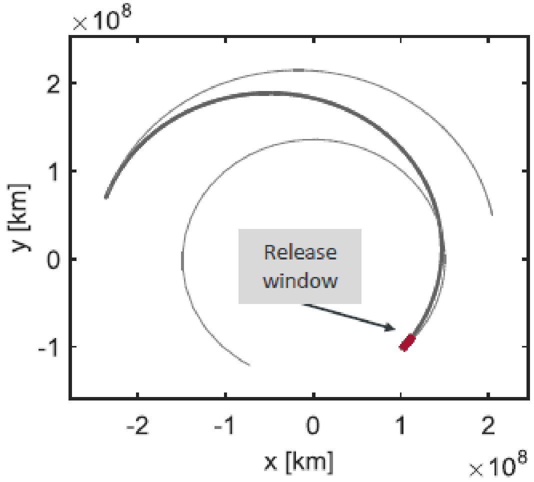
Conclusions

- **End-of-life disposal strategy should be planned** during mission design
    - End-of-life transfer enable **enhancing the scientific return** of the mission
    - However disposal cost is not only the **additional delta-v**, but also the **operations**.
- Gaia and Lisa Pathfinder mission disposal performed, robustness of the solution depends on true anomaly Earth-Moon around the Sun
- Planetary protection analysis needed for every interplanetary mission
    - SNAPPshot suite for planetary protection analysis
    - Full body dynamics
    - Efficient integration method needed
    - Representation in the b-plane
    - Advanced sampling methods
    - Study of close approaches

Conclusions

Future directions of research

- Propagation and fly-by characterisation
  - Symplectic integration techniques and projection methods
  - Fly-by detection through Jacobian computation and B-plane analysis
  - Semi-analytical techniques can be extended to describe fly-bys
- Simulation
  - Parallel programming
  - Machine learning
  - Solutions grows like a tree (branch and bound)
- Applications
  - Planetary moon missions e.g., JUICE mission
  - CubeSat swarms (different initial conditions for different s/c)
  - Robust trajectory optimisation
  - Launcher injection error



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- European Space Agency
    - Rüdiger Jehn, Jose Manuel Sánchez Pérez

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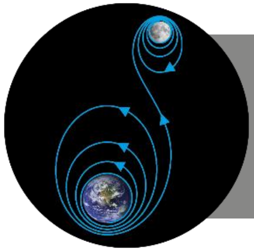
- Francesca Letizia, Jeroen Van den Eynde, Camilla Colombo, Rüdiger Jehn, Compliance Evaluation of Planetary Protection Requirements with SNAPPshot. Part 2: Application and Results, ASR, under review 2017
- Matteo Romano, Camilla Colombo, Jose Manuel Sánchez Pérez, Verification of planetary protection requirements with symplectic methods and Monte Carlo line sampling, International Astronautical Congress, IAC-17-C1.9.5

The COMPASS project

Services, technologies, science, space exploration

SPACE TRANSFER



Reach, control operational orbit

SPACE SITUATION AWARENESS

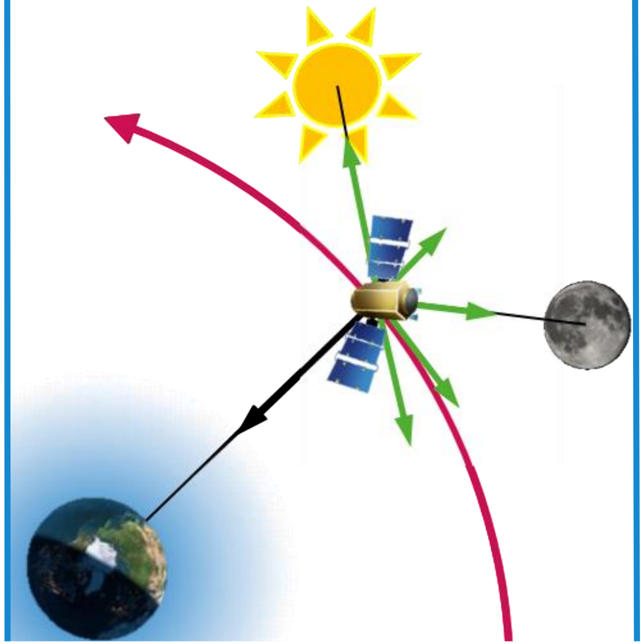
Asteroids. Planetary protection

Space debris

ORBIT PERTURBATIONS

Traditional approach: counteract perturbations

- Complex orbital dynamics
- Increase fuel requirements for orbit control



COMPASS

Novel approach: leverage perturbations

Reduce extremely high space mission costs

Create new opportunities for exploration and exploitation

Mitigate space debris

Develop novel techniques for orbit manoeuvring by surfing through orbit perturbations

Collaboration

COMPASS project

合作与机遇：

- 非欧盟地区学者
- 博士&博士后
- European Research Council and 中国国家自然科学基金

Collaboration:

- Non-European researcher
- PhD, Post Doc and researchers
- European Research Council and Chinese National Natural Science Foundation

For information see:

https://erc.europa.eu/sites/default/files/document/file/agreement_ERC_NSFC_zh.pdf

https://erc.europa.eu/sites/default/files/document/file/agreement_ERC_NSFC_en.pdf

http://www.nsfc.gov.cn/publish/portal0/tab87/info51450.htm

Moreover collaboration with many professors in the department:

PhD scholarships @PoliMi by Chinese State Scholarship Fund

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China Academy of Space Technology

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感谢观看
Thank you

Planetary protection: design and verification

Camilla Colombo, Politecnico di Milano
Email: camilla.colombo@polimi.it

DFH - China Academy of Space Technology, Beijing