Trajectory optimization for deflection of asteroid 2024 PDC25 using genetic algorithms and departure via lunar swing-bys

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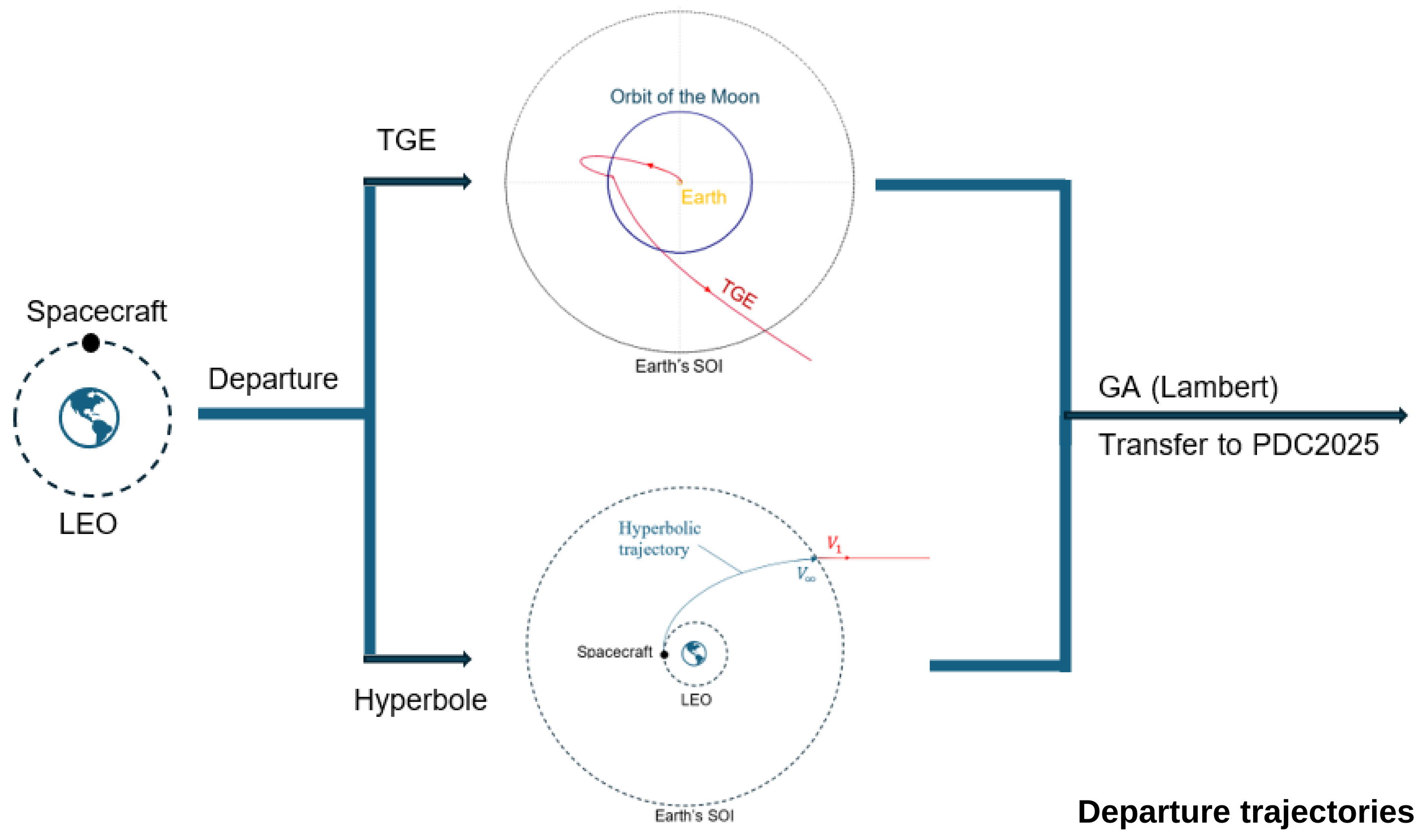
IAA-PDC25-01-135

Introduction

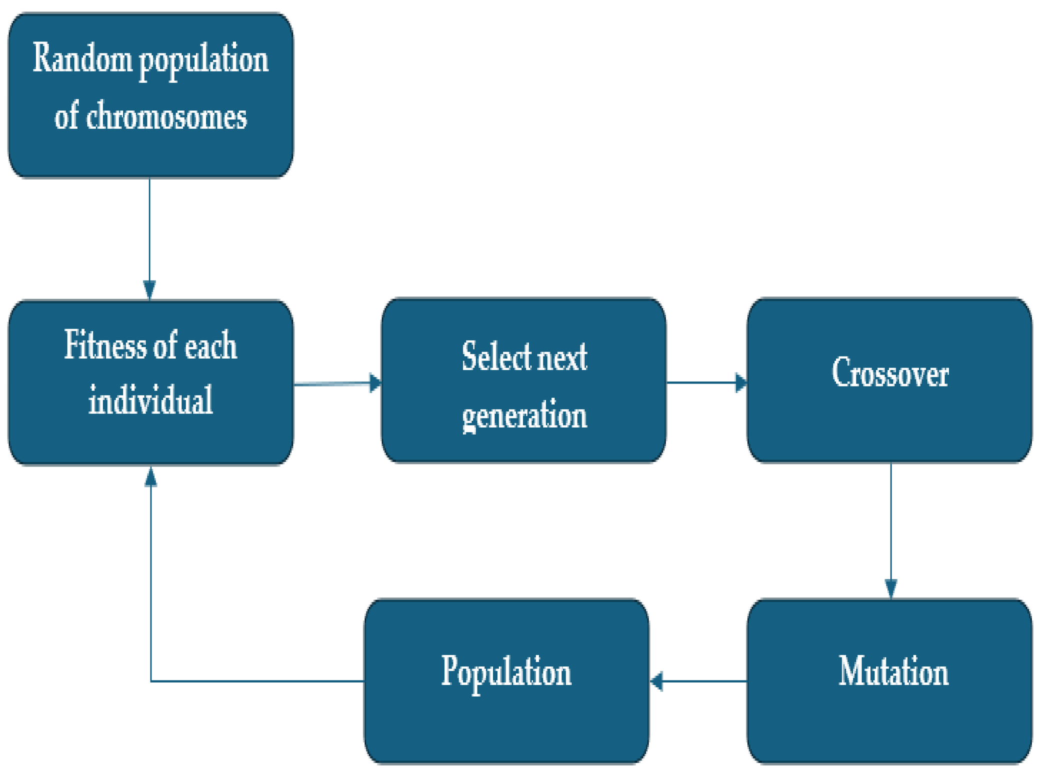
- Mission planning to "2024 PDC2025".
- Analysis of different departures from the Earth:
  - Hyperbolic departure (patched conics approximation).
  - Trajectory G of Escape (TGE, a low-cost escape trajectory, in which a lunar swing-by is performed) [1,2].
- Employment of Genetic Algorithms (GA) [3,4] to optimize the increment of velocity (ΔV) required to reach the asteroid:
  - Multiple launch dates evaluated ("Launch Scenarios").
  - Lambert's Problem solution in the fitness function.
- NASA/JPL NEO Deflection App for the evaluation of the asteroid deflection [5].



Departure trajectories

Methodology

- **System dynamics**
  - While within Earth's Sphere Of Influence (SOI):
    - Hyperbolic departure: Restricted Two-Body Earth-Spacecraft Problem (R2BP Earth-Spacecraft),
    - TGE departure: Restricted Four-Body Sun-Earth-Moon-Spacecraft Problem (R4BP)
  - After exiting Earth's SOI:
    - Restricted Two-Body Sun-Spacecraft Problem (R2BP Sun-Spacecraft)
- **Genetic Algorithms**
  - Random variables:
    - Hyperbolic departure: time of flight (Δt) and launch date
    - TGE departure: time of flight



Genetic Algorithm flowchart

$$\ddot{\boldsymbol{R}}_i = \sum_{\substack{j=1 \\ j\neq i}}^{4} \frac{\mu_j}{R_{ji}^3}(\boldsymbol{R}_j - \boldsymbol{R}_i), \qquad \mu_4 \approx 0$$

System Dinamics Equations (R4BP)

$$\ddot{\boldsymbol{R}}_4 = \frac{\mu_2}{R_{24}^3}(\boldsymbol{R}_4 - \boldsymbol{R}_2)$$

Spacecraft equations of motion (R2BP Earth-Spacecraft)

$$\ddot{\boldsymbol{R}}_4 = \frac{\mu_1}{R_{14}^3}(\boldsymbol{R}_4 - \boldsymbol{R}_1)$$

Spacecraft equations of motion (R2BP Sun-Spacecraft)

Results

Table 1: Results for the hyperbolic departures

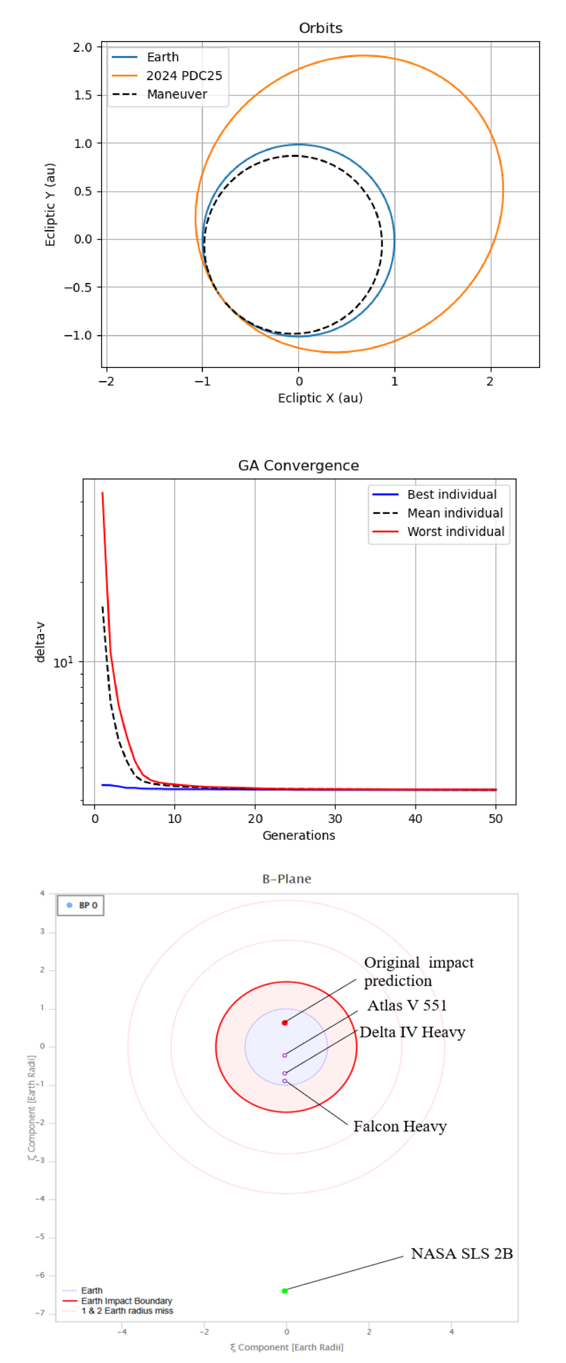
Launch Scenario (number of years before possible impact in 2041)	3 years	7 years	10 years	14 years
Launch Date (yyyy-mm-dd)	2038-04-26	2034-04-27	2031-04-27	2027-04-29
Interception Date (yyyy-mm-dd)	2039-03-11	2034-12-14	2032-10-31	2028-08-05
Time of Deflection (days)	775	2323	3097	4645
Transfer Time (days)	319	231	553	464
ΔV (km/s)	3.288	4.492	3.772	3.453
$\mathbf{V}_{\mathrm{rel}}$ (km/s)	[-3.952 4.564 -6.488]	[-6.668 7.826 -6.472]	[-1.124 0.838 -6.467]	[-1.836 1.878 -6.464]
V_{rel} (km/s)	8.863	12.149	6.618	6.977

Table 2: Deflection by launch vehicles for the different launch scenarios (hyperbolic departure)

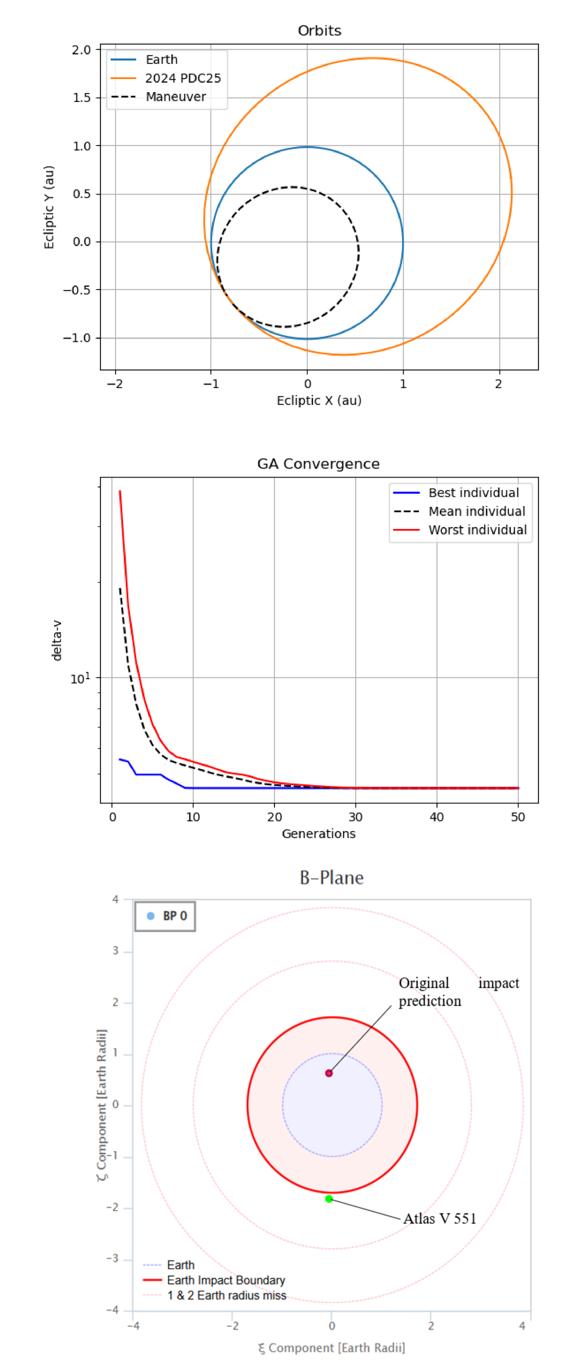
Launch Scenario	3 years	7 years	10 years	14 years
Launch Vehicle	NASA SLS 2B	Atlas V 551	Atlas V 551	Atlas V 551
Mass of the interceptor (kg)	44.6×10^3	3.16×10^3	4.96×10^3	5.56×10^3

Table 3: Results for TGEs departures

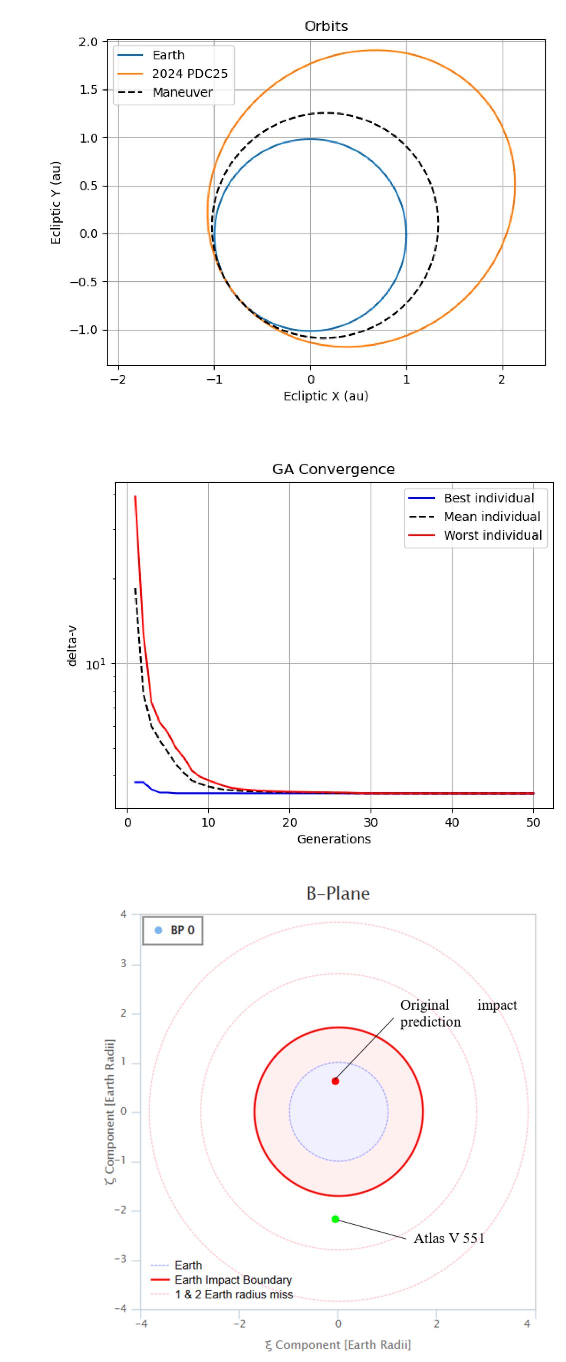
Launch Scenario (number of years before possible impact in 2041)	3 years	7 years	10 years	14 years
Launch Date (yyyy-mm-dd)	2038-04-16	2034-04-01	2031-04-16	2027-04-16
Departure Date (yyyy-mm-dd)	2038-05-22	2034-04-23	2031-05-23	2027-05-23
Interception Date (yyyy-mm-dd)	2039-03-11	2034-12-02	2032-10-31	2028-10-31
Time of Deflection (days)	775	2335	3097	4558
Transfer Time (days)	293	223	527	440
ΔV_1 (km/s)	3.159	3.160	3.161	3.160
ΔV_2 (km/s)	0.694	7.654	1.846	3.500
ΔV_t (km/s)	3.853	10.814	5.007	6.660
$\mathbf{V}_{\mathrm{rel}}$ (km/s)	[-4.242 5.258 -6.691]	[-6.668 7.826 -6.472]	[-1.390 0.935 -6.390]	[-1.212 2.656 -6.313]
V_{rel} (km/s)	9.509	11.172	6.606	6.955



3-Years scenario



7-Years scenario



10-Years scenario

Conclusions

- Hyperbolic departures:
  - Launch dates as late as 3 years before the impact with Earth were analyzed, and interception solutions with transfers time around 300 days and $\Delta V = 3.29$ km/s were found.
  - However, for the complete deflection of the asteroid, the launched mass is 10 times greater than for other scenarios.
  - The 3-years scenario is critical; the 7-years scenario appears to be a good alternative, since it has the shortest transfer time and requires one of the smallest masses for the interceptor.
- TGE departures:
  - The ΔV_2, Applied athe the exit of Earth's SOI and required to direct the spacecraft to the asteroid is large, which makes this alternative unfeasible.

References

[1] R. A. Broucke, Periodic orbits in the restricted three-body problem with earth-moon masses. technical report 32-1168, Jet Propulsion Laboratory, Cal. Tech (1968).

[2] R. S. Ribeiro, C. F. de Melo, A. F. Prado, trajectories derived from periodic orbits around the lagrangian point l1 and lunar swing-bys: Application in transfers to near-earth asteroids, Symmetry 14 (2022) 1132.

[3] C. Toglia, Optimization of orbital trajectories using genetic algorithms, Journal of Aerospace Engineering, Sciences and Applications 1 (2008).

[4] G. Neves, D. Santos, J. Formiga, R. Domingos, Orbital maneuvers for asteroids using genetic algorithm, Journal of Physics Conference Series 1365 (2019).

[5] NASA/JPL, NASA/JPL NEO Deflection App, 2025. Accessed on: March 29, 2025.

Acknowledgments

The authors wish to express their appreciation for the financial support provided by the National Council for Scientific and Technological Development (CNPq), project 141133/2021-0. This study was financed in part by the Coordination for the Improvement of Higher Education Personnel – Brazil (CAPES) – Finance Code 001.

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9th IAA Planetary Defense Conference 2025
Stellenbosch, Cape Town, South Africa
5-9 May 2025