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Scalable Multi-Spacecraft Hybrid Electromagnetic Strategies for Non-Contact Asteroid Deflection

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Abstract

Non-contact electromagnetic methods offer promising alternatives to conventional kinetic or explosive techniques for planetary defense. Simulations targeting asteroid 99942 Apophis and the smaller 2025 HX demonstrate that gravity tractors and ion-beam shepherds deliver the greatest cumulative velocity change over three years, respectively. Hybrid schemes offer marginal gains over their primary modality. At the same time, multi-spacecraft analyses reveal that even a ten-vehicle fleet struggles to achieve the 0.14 m/s deflection threshold within mission timelines for Apophis. Despite lower Technology Readiness Levels (TRL), the continuous, reusable operation and low fragmentation risk of electromagnetic strategies make them compelling complements to existing planetary defense techniques.

Keywords: non-contact methods, electromagnetic deflection, multi-spacecraft, planetary defense, Near-Earth Objects (NEOs)

1. Introduction

Asteroids—remnants of the early solar system—are rocky bodies orbiting the Sun; any such body with perihelion < 1.3 AU is a Near-Earth Asteroid (NEA) [1], and those with diameters > 140 m and perihelion within 0.05 AU of Earth's orbit are classified as Potentially Hazardous Asteroids (PHAs) [2]. These objects and near-Earth comets are collectively known as Near-Earth Objects (NEOs). Although small compared to planets, NEAs can inflict regional to continental-scale damage depending on the impact velocity and size [3]. Crater records on Mercury, Jupiter's moons, and elsewhere attest to a long history of impacts. Accurate orbit determination is critical to assess the PHAs collision risk [2]. Recent advances in detection and tracking technologies have significantly increased the number of identified NEAs and their orbits. However, predicting long-term trajectories concerning potential Earth impact remains challenging. Early detection and precise tracking of NEOs are crucial for effective planetary defense [4].

Various mitigation strategies have been proposed, including kinetic impactors, gravity tractors, and nuclear explosions, each with advantages and limitations. Kinetic impactors collide with the asteroid to alter its trajectory. Gravity tractors (GTs) use the gravitational pull of a spacecraft placed near an asteroid to shift its orbit slowly over time, requiring substantial mass and lead times. Nuclear explosions fragment the asteroid into smaller pieces that fly off in different directions and are small enough to burn up in the Earth's atmosphere. While these methods provide valuable options, they each pose challenges.

Traditional mitigation strategies each have trade-offs:

- **Kinetic Impactors:** Simple, high-impulse Δv via direct collision, but risk fragmentation and require large Δv budgets [4, 5].

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- **Gravity Tractors:** Continuous, contactless tug using mutual gravity—very precise but low thrust and long lead times [6].
- **Nuclear Explosions:** Extreme impulse for large bodies, yet high political/risk profile, and uncertain fragment outcome [7].

NASA's DART mission (2022) validated kinetic impactors by imparting $\Delta v \approx 2.7$ mm/s on Dimorphos, but impulse-only techniques can struggle with large or late-detected NEOs [4, 5, 7]. Non-contact electromagnetic methods offer complementary pathways with unique scaling and risk profiles:

- **Electrostatic Deflection:** Coulomb forces scale as $1/r^2$, enabling fine, continuous Δv without contact and minimal fragmentation risk [8].
- **Magnetostatic Deflection:** Push-pull coupling with onboard fields on conductive asteroids—force scales as $1/r^4$, suited to metallic bodies [9].
- **Lorentz Force Deflection:** A charged body moving through the ambient interplanetary magnetic field experiences a force providing continuous, non-contact deflection over extended trajectories.

These three electromagnetic mechanisms can be combined or deployed in swarms to tailor the force magnitude depending on the range and target composition. This paper systematically models and simulates all three, alongside gravity tractors and ion-beam shepherds, to evaluate:

- Case studies on 99942 Apophis and 2025 HX for mass-scaling effects;
- Hybrid force combinations for enhanced Δv;
- Multi-spacecraft fleet architectures for mission-duration trade-offs.

The paper is organized as follows: Section 2 reviews traditional and electromagnetic deflection models; Section 3 presents the mathematical force models; Section 4 describes our numerical simulation framework; Section 5 presents results on Δv accumulation, station-keeping, and scaling; Section 6 discusses feasibility and technology readiness level (TRL); and Section 7 concludes with key findings and future research directions.

2. Asteroid Deflection Methods Overview

2.1. Gravity Tractor

Lu and Love proposed a low-energy asteroid deflection concept using mutual gravitational force between a hovering spacecraft and a target asteroid as a towline [1, 6]. The key idea is to hover long enough around the target asteroid to affect its trajectory. To deflect an asteroid from an Earth impact trajectory, the mean change in velocity required is:

$$\Delta v = \frac{3.5 \times 10^{-2}}{t} \tag{1}$$

where t is the lead time in years [6]. Wie [1] proposed using multiple gravity tractors (MGTs) flying in halo orbits near a target asteroid. Additionally, the author presents a side-by-side comparison of thruster GTs and solar sail GTs. Subsequent research explored mass-optimized GT designs [10], integration with NASA's Asteroid Redirect Mission (ARM) architecture [11], and advanced propulsion strategies for maintaining precision station-keeping [12]. While GTs offer fine control and non-destructive interaction, their effectiveness diminishes for late-detected threats due to their slow momentum transfer rates. Developing heavier spacecraft weighing approximately 20 metric tons is not trivial and requires a multi-year effort.

2.2. Ion Beam Shepherd

The Ion Beam Shepherd (IBS or IB) concept, introduced by Bombardelli and Peláez [13], offers a non-contact alternative wherein a spacecraft directs a collimated ion beam onto the asteroid's surface, imparting continuous thrust without physical contact. IB systems can achieve significantly higher thrust levels than GTs, reducing required mission durations for modest Δv targets.

However, IB missions face technological challenges, including station-keeping, substantial power requirements, precise beam alignment, resilience to space radiation exposure, and micro-meteoroid impacts. Although promising, IB may not be suitable for late-stage asteroid deflection due to its slow response time; it is more suited for missions with longer lead times when gradual deflection is feasible.

2.3. Electromagnetic Deflection

Non-contact electromagnetic forces utilize electric and magnetic fields to exert a tug on an asteroid without physically touching it. Three main mechanisms are considered:

- Electrostatic deflection, via electrostatic tractor (ET), involves charging the spacecraft and the asteroid to generate a Coulomb force between the two bodies. Adequately controlled, this force enables precise momentum transfer without physical contact, making it attractive for structurally fragile asteroids.
- Magnetostatic deflection, via magnetic tractor (MT), leverages magnetic fields produced by current-carrying conductors onboard the spacecraft. These fields interact with the asteroid's spacecraft-borne conductors or intrinsic magnetic properties, enabling an additional mechanism for applying non-contact forces.
- Lorentz-force deflection, via Lorentz tractor (LT), requires a combination of charge and a magnetic field. If the asteroid carries a net charge q and moves with velocity $\mathbf{v}$ through an applied magnetic field $\mathbf{B}$, it experiences the Lorentz force, which imparts a continuous Δv orthogonal to the velocity and field directions.

3. Electromagnetic Force Modeling

The term "electromagnetic" encompasses all interactions that involve any combination of electric charge and electromagnetic fields. The electric charge could be moving, or the electric/magnetic field could be varying. This work examines three primary electromagnetic force mechanisms: electrostatic, magnetostatic, and electromagnetic forces resulting from magnetic moment interactions. Each mechanism is modeled to assess its contribution to the gradual alteration of asteroid trajectories.

The electrostatic force $\mathbf{F}_E$ between two charged bodies is modeled by the Coulomb law:

$$\mathbf{F}_E = k_e \frac{q_1 q_2}{r^2} \hat{\mathbf{r}} \tag{2}$$

where q_1 and q_2 are the spacecraft and asteroid charges, r is the separation distance, and k_e is Coulomb's constant. The magnetostatic force $\mathbf{F}_M$ between two magnetic dipoles $\mathbf{m}_1$ and $\mathbf{m}_2$ is given by [14]:

$$\mathbf{F}_M = \frac{3\mu_0}{4\pi r^4}\Big[(\mathbf{m}_1 \cdot \hat{\mathbf{r}})\,\mathbf{m}_2 + (\mathbf{m}_2 \cdot \hat{\mathbf{r}})\,\mathbf{m}_1 + (\mathbf{m}_1 \cdot \mathbf{m}_2)\,\hat{\mathbf{r}} - 5\,(\mathbf{m}_1 \cdot \hat{\mathbf{r}})(\mathbf{m}_2 \cdot \hat{\mathbf{r}})\,\hat{\mathbf{r}}\Big] \tag{3}$$

where μ_0 is the permeability of free space. The Lorentz or electromagnetic force is proportional to the external magnetic field $\mathbf{B}$ and charge:

$$\mathbf{F}_L = q(\mathbf{E} + \mathbf{v} \times \mathbf{B}) \tag{4}$$

Each force model scales differently with distance: the electrostatic force scales as $1/r^2$, the magnetostatic force as $1/r^4$, and the Lorentz force typically as $1/r^3$. The mutual forces are incorporated into the heliocentric equations of motion for the asteroid and spacecraft:

$$\ddot{\mathbf{r}}_A = -\frac{\mu}{|\mathbf{r}_A|^3}\mathbf{r}_A + \frac{\mathbf{F}_{S\to A}}{m_A} \tag{5}$$

$$\ddot{\mathbf{r}}_S = -\frac{\mu}{|\mathbf{r}_S|^3}\mathbf{r}_S + \frac{\mathbf{F}_{A\to S}}{m_S} \tag{6}$$

where $\mathbf{F}_{S\to A}$ represents the selected interaction force (electrostatic, magnetostatic, or electromagnetic) and $\mathbf{F}_{A\to S} = -\mathbf{F}_{S\to A}$ by Newton's third law. These models form the basis for the subsequent numerical simulation of asteroid deflection using different non-contact force techniques.

4. Numerical Simulation Framework

The force models described in the previous section are incorporated into a numerical simulation framework to assess the cumulative velocity change imparted to an asteroid over extended engagement periods. The simulations evaluate each non-contact deflection method individually and in hybrid combinations, using realistic asteroid and spacecraft parameters.

4.1. Purpose and Overview

Due to the nonlinear and coupled nature of spacecraft-asteroid interactions under electromagnetic forces, numerical simulation is essential to evaluate the cumulative effects of different deflection strategies. Simulations enable the assessment of gradual momentum transfer over multi-year engagement periods, scalability through multi-spacecraft deployment, and hybrid deflection architectures. The following framework was developed to systematically model and propagate the asteroid and spacecraft trajectories under different force regimes.

4.2. Custom Simulation Architecture

The simulation framework uses an object-oriented structure to modularly represent spacecraft dynamics, asteroid dynamics, force interactions, and control systems. Core orbit propagation capabilities are provided by the open-source Kete astrodynamics toolkit [15], which is extended with custom modules developed in this work.

Custom force models representing electrostatic, magnetostatic, and electromagnetic interactions are implemented and integrated into the simulation environment. Force model classes are designed to allow seamless switching between different deflection methods, enabling side-by-side evaluation of individual and hybrid strategies.

The simulation architecture also includes independent station-keeping controllers, decoupled from force application logic, supporting flexible testing of proximity operations and engagement stability. This modular design approach allows for future additions, such as environmental perturbations or swarm coordination algorithms.

The computational framework was developed following best practices in software engineering, with modularization, extensibility, and reproducibility as primary design goals.

4.3. Station-Keeping Control

The interaction between the spacecraft and the asteroid generates reaction forces on the spacecraft, causing a drift from the desired standoff distance. To maintain proximity, a simple Proportional-Derivative (PD) controller is implemented:

$$\mathbf{u}_{\text{sc}} = K_p(\mathbf{r}_{\text{cmd}} - \mathbf{r}) + K_d(\dot{\mathbf{r}}_{\text{cmd}} - \dot{\mathbf{r}}) \tag{7}$$

where $\mathbf{r}_{\text{cmd}}$ is the commanded separation vector, $\mathbf{r}$ is the current separation, $\dot{\mathbf{r}}_{\text{cmd}}$ is the commanded velocity vector, $\dot{\mathbf{r}}$ is the current velocity, and K_p, K_d are tuned gains. The controller ensures stabilization with minimal overshoot during the initiation of engagement.

4.4. Assumptions and Simplifications

The following assumptions and simplifications are made to reduce computational complexity:

1. Physical & Geometric
   - Point mass bodies: Both asteroid and spacecraft are treated as point masses (no finite size, no attitude dynamics, no shadowing or mutual occultation).
   - Spherical symmetry and no asteroid spinning.
   - Fixed mass, charge, and magnetic dipole: no propellant depletion, no charge leakage or redistribution.
2. Force Model
   - Linear superposition: hybrid model sums accelerations from each method, i.e., no cross-coupling such as beam-plasma interactions, charge-magnet interactions beyond the Lorentz term.
   - Idealized fields: ignored plasma shielding, surface roughness, or field emission limits.
   - No external perturbations such as solar radiation pressure, Yarkovsky/YORP effect, outgassing, and gravitational harmonics.
3. Control & Guidance
   - The PD station-keeping controller perfectly maintains a fixed offset; no sensor noise, actuator saturation, time delays, or any nonlinearities in thruster pointing.

- Independent spacecraft: multiple spacecraft each fly their own PD loop without interacting with each other (no collision avoidance, no cross-interaction of their force fields).

4. Parameters

   - Masses come from assumed densities and diameters; any error feeds directly into the required deflection.
   - Constant thrust and I_{sp}: the ion beam thrust is held at 0.1 N with no throttling or power-budget variations over multi-year runs.

These simplifications focus the analysis on the first-order feasibility of electromagnetic deflection, with future work intended to incorporate more realistic environmental effects.

4.5. Limitations

The assumptions affect the simulation results as follows:

1. Beyond a Debye length, the ambient plasma shields charge, reducing the force and Δv estimates.
2. Beam divergence and partial capture can reduce effective thrust, leading to extended deflection times.
3. The magnetic coupling can be reduced because of field distortion by solar wind pressures, asteroid permeability, or eddy currents.
4. Sensor noise can drive offsets to smaller or larger distances, leading to oscillations.
5. The neglected perturbations can add accelerations comparable to some tractor forces.

Addressing these limitations in future work, through plasma-kinetic modeling, beam-capture experiments, high-fidelity field simulations, and full mission environment integration, will be critical for a flight-ready design.

5. Results and Analysis

The National Research Council considers 15,000 km as a safe miss as long as the orbit of NEO is well-determined [16]. With a 10-year warning time, the required $\Delta v_{\text{req}} \approx 0.04$ m/s. Considering a late-detected NEO, we aim to achieve an Earth-miss distance of approximately two Earth radii with a 3-year time of flight (t_{TOF}) and include an appropriate margin for modeling uncertainties:

$$\Delta v_{\text{req}} = \frac{R_{\text{Earth}} + R_{\text{safety}}}{t_{\text{TOF}}} \approx \frac{12,742 \times 10^3\ \text{m}}{9.46 \times 10^7\ \text{s}} \approx 0.135\,\text{m/s} \tag{8}$$

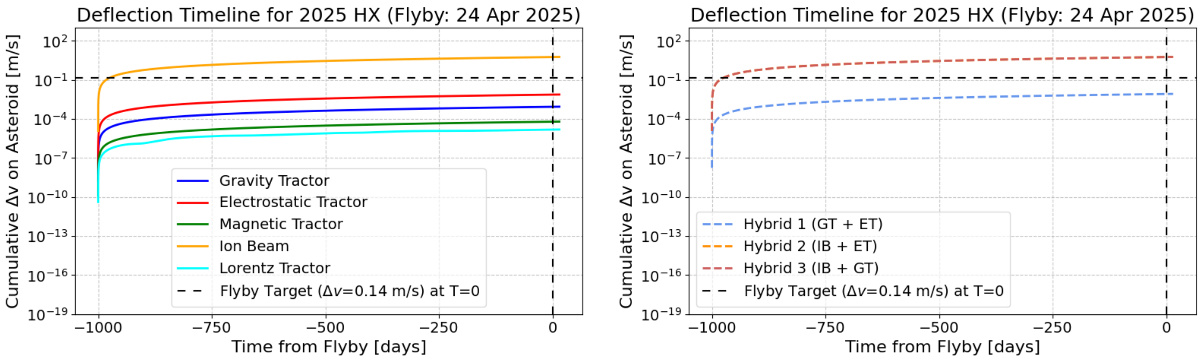
In our simulation, we use 0.14 m/s as the target velocity. Five primary and two hybrid deflection techniques are considered in this study, for which the parameters are listed in Table 1. The time step is 3 minutes, and the initial standoff distance, i.e., the separation between the asteroid and spacecraft, is 100 m. Two asteroids—2025 HX (smaller) and 99942 Apophis (larger)—are selected to evaluate the efficiency of the described deflection techniques.

Figure 1a shows that the asteroid 2025 HX accumulates velocity imparted by a planetary defense spacecraft (PDS). The MT and LT impart the least velocity to the asteroid. The GT imparts a velocity that is an order of magnitude higher, the ET by two orders of magnitude, and the IB by five orders of magnitude higher than the MT. This shows that the IB has the highest cumulative Δv. Figure 1b shows three hybrid methods—Hybrid 1 (GT + ET), Hybrid 2 (IB + ET), and Hybrid 3 (IB + GT). The Hybrid 1 method exhibits a higher velocity change than GT alone and is almost the same as for ET alone. The Hybrid 2 method exhibits similar performance to IB, as its force is approximately three orders of magnitude greater than the ET force. Thus, the overall cumulative velocity change is not significantly different from the IB alone. The Hybrid 3 method also matches performance with the Hybrid 2 method, as the IB force still dominates GT by four orders of magnitude.

Imparting velocity to the asteroid creates a reaction on the PDS, changing the separation between the two bodies. As the separation changes, the station-keeping feedback controller acts to restore the initial standoff distance. Figure 2 shows the transient response of the PDS, indicating that the

Table 1: Simulation parameters derived from the physical properties for the asteroid and spacecraft used by [13, 17, 18].

Parameter	2025 HX	99942 Apophis	PDS
Mass (kg)	1.5×10^6	4×10^{10}	1500
Charge (μC)	-12	-12	12
Magnetic Dipole (A·m^2)	28.7	28.7	6.23×10^6
K_p	-	-	0.0001
K_d	-	-	0.001
Charge-to-mass (C·kg^{-1})	-8×10^{-12}	-3×10^{-16}	8×10^{-9}
Dipole-to-mass (A·m^2·kg^{-1})	1.91×10^{-5}	7.17×10^{-10}	4.15×10^3



(a) **(b)**

Figure 1: Cumulative velocity change for asteroid 2025 HX under active deflection methods. (a) Individual methods include GT, ET, MT, IB, and LT. (b) Hybrid methods combine multiple force mechanisms. The horizontal dashed black line indicates the minimum Δv required to significantly alter the asteroid's trajectory during Earth flyby.

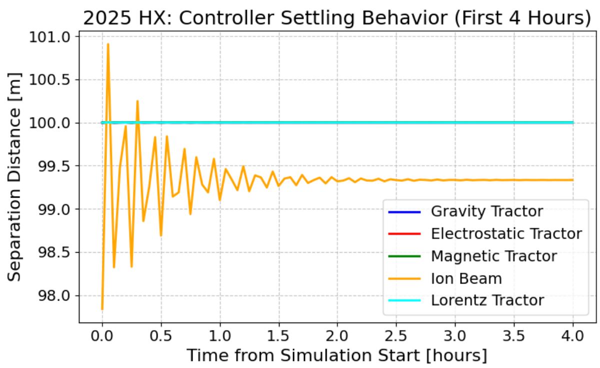
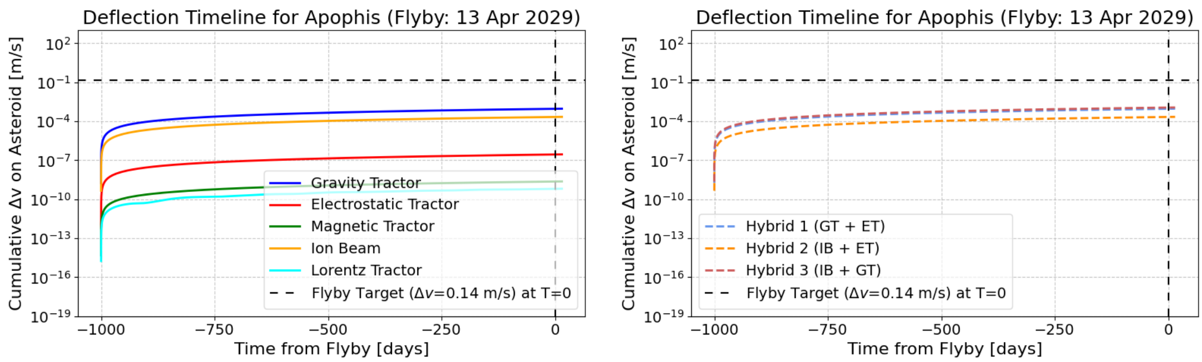


Figure 2: Station-keeping response of the PDS during the deflection of asteroid 2025 HX using primary deflection methods. The spacecraft initially maintains a standoff distance of 100 meters relative to the asteroid. The figure illustrates the spacecraft separation's transient behavior and settling time following initial deflection engagement. Controller gains are tuned for rapid stabilization with minimal overshoot under realistic deflection forces.

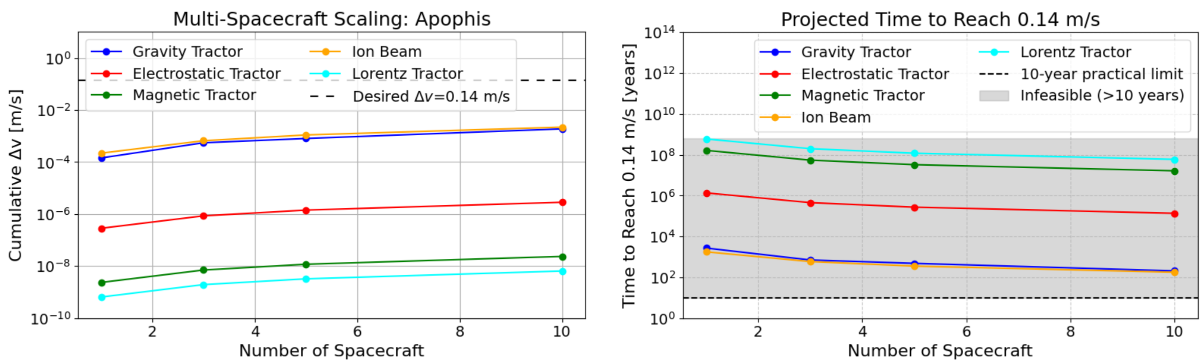
separation restoration for IB is not precisely at 100 m but settles slightly shy of 99.5 m. For other deflection techniques, the controller restores the exact standoff distance.

Using the same parameters for deflecting Apophis yields different results. Figure 3a shows that the MT and LT follow the same pattern as for asteroid 2025 HX but are reduced by five orders of magnitude. The ET imparts a higher velocity by three orders of magnitude, and the IB by six orders of magnitude than the MT. The GT outperforms the other methods for Apophis, albeit only slightly, compared to the IB. This is opposite to 2025 HX, since the mass of Apophis is much higher than that of the PDS, and a strong gravitational pull exists between the asteroid and the PDS. The PDS must compensate for this



(a) **(b)**

Figure 3: Cumulative velocity change for asteroid 99942 Apophis under active deflection methods. (a) Individual methods include GT, ET, MT, IB, and LT. (b) Hybrid deflection strategies combining electrostatic and magnetic effects to enhance cumulative Δv performance.



(a) **(b)**

Figure 4: Results for a multi-spacecraft mission to deflect asteroid Apophis. (a) Cumulative deflection velocity (Δv) achieved after a 3-year mission duration as a function of spacecraft fleet size. (b) Extrapolated time required to reach a target Δv of 0.14 m/s as a function of fleet size. Results are shown for multiple deflection methods, assuming continuous operation and ideal station-keeping. The dashed line indicates a 10-year operational feasibility limit. Shaded regions represent mission durations exceeding the practical timeline for planetary defense response.

Table 2: Technology Readiness Levels and Key References for Deflection Methods

Method	TRL	Justification & Key References
Gravity Tractor	3–4	Concept introduced by Lu and Love (2005) [6].
Electrostatics	2–3	Lab charging tests; no in-space asteroid demo [8, 19].
Magnetic Tractor	2–3	Push–pull magnetic coupling concepts; no lab or flight demos [9].
Ion-Beam Shepherd	2–5	Concept studies (Bombardelli and Peláez 2011, TRL 2–3); T7 and DST thruster breadboards at TRL 5 [13, 20].
Kinetic Impactor	9	NASA's DART mission demo (2022); β-factor measured [21, 22].

gravitational pull. The hybrid methods yield different results than 2025 HX; see Fig. 3b. The Hybrid 1 and Hybrid 3 methods impart similar velocity to Apophis, while the Hybrid 2 method falls short by an order of magnitude. Since the GT, followed by IB, imparts the highest velocity changes to Apophis, combining any other technique with them does not give noticeable changes in asteroid velocity. Using the same gains on the PDS controller as for the 2025 HX, the controller's performance is nearly identical for all methods, except for the GT, which settles at a distance of 97 m from the asteroid (plot not shown for clarity).

The effectiveness of a deflection method depends on the parameters such as charge, mass, and orbital elements. An important consideration is the ratio of the underlying parameters responsible for accelerating the asteroid, like the charge-to-mass ratio for ET and the magnetic dipole-to-mass ratio for MT. From Table 1, we see that the absolute charge-to-mass ratio for 2025 HX is higher than for Apophis by four orders of magnitude. Similarly, the magnetic dipole-to-mass ratio is approximately five orders higher for 2025 HX.

These results suggest that multiple spacecraft may be necessary to deflect heavier or larger asteroids. Scaling up the number of spacecraft for an asteroid deflection mission reduces the cost per spacecraft. Figure 4a shows the velocity scaled with the number of spacecraft, while Fig. 4b shows the time it would take to reach the desired Δv. As seen, the primary methods, even with a 10-spacecraft fleet, fall orders of magnitude short of the required Δv targets within realistic mission timescales (taken 10 years as the practical limit). Alternatively, a hybrid deflection technique combining two or more methods may prove more efficient if they are non-competing or counteracting. Cirelli et al. [23] explored a hybrid method (GT + MT), emphasizing the possibility of higher deflection—the choice of combining which ones would depend on the asteroid target's physical and orbital properties.

6. Feasibility and Technology Readiness

The TRLs of each technique are listed in Table 2; with these readiness levels, the physical and electrical properties of the target body become a critical driver in selecting the optimal deflection method:

1. **Conductivity & Electrostatics:** High bulk conductivity (typical of M- and some C-type asteroids) allows rapid charge redistribution and stronger Coulomb forces, making ETs most effective on metallic or carbonaceous bodies. Conversely, low-conductivity S-types (silicate-dominated) limit surface charge density—and thus achievable Δv—unless pretreated or seeded with conductive coatings.

2. **Magnetic Susceptibility & Magnetic Tractors:** Paramagnetic minerals (e.g. olivine in Sq-class Apophis) exhibit weak positive susceptibility, enabling a small induced dipole and thus a push–pull magnetic coupling force. Bodies with predominantly diamagnetic or non-magnetic regolith (e.g. C-types) will have near-zero response, so MTs are best matched to metal-rich M-types or, to a lesser extent, olivine-rich S-types, and are poorly suited to carbonaceous C-types.

3. **Surface Cohesion & Ion-Beam Shepherd:** Ion-beam momentum transfer depends on sputter yield and surface binding energy of the target material. Fine-grained, loosely consolidated regolith (characterized by high sputter yield and low surface binding energy) enables efficient impulse delivery. In contrast, consolidated rock or highly porous substrates can scatter the ion plume, reducing the efficacy of the neutralizer.

4. **Mass Density & Gravity Tractors:** Gravity tractors are insensitive to composition but scale linearly with mass. Denser, metal-rich bodies require proportionally higher station-keeping thrust; porous C-types can be offset with lower propellant budgets for the same Δv. The navigation and control requirements for maintaining the standoff distance remain unchanged.

5. **Mechanical Strength & Kinetic Impactors:** Momentum enhancement factor (β) depends on target porosity and tensile strength. Rugged, coherent silicate surfaces (low porosity) yield lower ejecta mass and smaller β, while fractured or regolith-covered bodies can produce higher β, amplifying impactor effectiveness.

7. Conclusion

This paper comprehensively evaluates electromagnetic asteroid deflection strategies, including electrostatic, magnetostatic, and electromagnetic interactions, alongside gravity tractor and ion-beam shepherd methods. Our results indicate that ion-beam shepherds outperform other non-contact approaches in velocity change effectiveness, with electrostatic methods showing significant promise. Hybrid combinations have similar performances to their dominating deflection method and are viable as scalable solutions for planetary defense missions. Although electromagnetic methods currently exhibit lower TRLs, their operational flexibility, reusability, and reduced fragmentation risk position them as valuable complements to mature deflection techniques. Future work will address practical considerations such as plasma interactions, charge leakage, spacecraft attitude dynamics, and robust multi-spacecraft coordination to transition these strategies from theoretical models to realistic, mission-ready solutions.

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