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NEOTωIST - An Asteroid Impactor Mission Featuring Sub-spacecraft for Enhanced Mission Capability

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Abstract

Near Earth Object (NEO) deflection for the purpose of planetary defense has become increasingly recognized as a valid and valuable endeavour. NEOTωIST stands for Near-Earth Object Transfer of angular momentum (ω) Spin Test. This describes a demonstration mission intended to develop the capabilities required to execute an effective kinetic impactor NEO deflection mission. The chosen measurement technique and employment of small sub-spacecraft for observation purposes represent a novel approach to achieving the main goals of such a demonstration mission. The approach promises comparatively low cost and features capabilities that are unique and valuable for an operational deflection mission. Most standard deflection demonstration missions propose to quantify momentum transfer from the impactor spacecraft to the target object by measuring a change in its heliocentric orbit. The change is typically so small that it must be performed via radio-science from a second observer spacecraft which rendezvous with the NEO prior to impact. In our case the NEO is struck off-center which changes its spin rate. This rate change, which can be measured from Earth via light curve measurements, allows quantification of the transferred momentum. Using this measurement method the need for an observer spacecraft for the purpose of NEO orbit measurement is eliminated. The second function of the observer spacecraft is the close-up observation of the impact event for improvement of impact effectiveness modelling. The NEOTωIST mission achieves this observation by deploying several small sub-spacecraft from the main impactor spacecraft shortly before impact. These sub-spacecraft allow observation of the impact event from multiple vantage points some of which are unique because their destruction is accepted. At least one sub-spacecraft trajectory is planned such that survival is guaranteed, which enables it to receive observation data from the other spacecraft for delayed transmission to Earth. We present the overall mission concept as well as preliminary design work on the key technical challenges, in particular those associated with the highly dynamic operation of the small sub-spacecraft that are a key feature of the NEOTωIST mission.

Keywords: NEO, deflection, Kinetic Impactor, planetary defence, sub-spacecraft, small satellite

Abbreviations

CH	Chaser
DQ	Don Quijote
FBM	Flyby Module
HGA	High-Gain Antenna
ISL	Inter-satellite Link
KI	Kinetic Impactor
LPF	Lisa Pathfinder
MGA	Medium-Gain Antenna
MO	Mission Oportunity
MO	Mission Option
NEO	Near Earth Object
RF	Radio Frequency
Rx	Receive
S/A	Solar Array
STR	Star Tracker
Tx	Transmit

1. Introduction

NEOTωIST stands for Near-Earth Object Transfer of angular momentum Spin Test. This describes a demonstration mission intended to develop and validate the capabilities required to execute an effective kinetic impactor NEO deflection mission.

Near Earth Object (NEO) deflection for the purpose of planetary defence has become increasingly recognized as a valid and valuable endeavour. Small impact events like the Chelyabinsk air blast from an

18m object in 2013 serve as a reminder that that over a sufficiently long time horizon Earth will be faced with a real NEO impact threat. Ongoing efforts to create a comprehensive catalogue of potentially hazardous objects increasingly put us in the position to identify serious threats with a lead time of years. Threats with long lead times (years) and moderate size (< hundreds of meters) are prime candidates for deflection by means of Kinetic Impactor (KI), i.e. changing the target trajectory by impacting it with a spacecraft.

Although there is little doubt about the principle feasibility of this approach, preparation for a real deflection mission requires the development and validation of the associated technology. Significant functionalities to be developed include the GNC to hit the target accurately, as well as means to verify an effective impact. Both are addressed by the proposed NEOTωIST mission concept.

Further, the models used to predict the impact dynamics need to be improved and validated. Currently, the observational data needed to do this is not available for hypervelocity impacts of relevant size. The relevance of accurate impact modelling is that it allows quantification of the momentum change imparted on the target object, i.e. deflection effectiveness. This in turn is needed to define spacecraft mass, geometry, and impact velocity such that sufficient alteration of the NEO trajectory is ensured. The uncertainty in the amount of moment transfer achieved stems primarily from different hypotheses about the momentum of the material ejected at impact, which typically augments the momentum transfer to the target object. The amount of additional imparted momentum achieved by the ejecta with respect to the momentum of the Kinetic impactor is characterised by the beta factor

$$\beta = \frac{p_{KI} + p_{Ej}}{p_{KI}}$$

where p_{KI} is the momentum carried by the Impactor and p_{Ej} is the momentum of the ejecta in opposite direction of the impact velocity. For the reasons explained the predictions about the β-factor are an essential parameter for deflection mission design. The NEOTωIST mission concept is conceived to measure the β-factor and to perform additional observations which allow to validate and constrain models of the impact dynamics.

Next to the technical challenges of the mission, the mission concept must respect programmatic constraints. The likelihood of implementation will depend on cost and implementation flexibility.

The top-level mission objectives and constraints of the proposed mission can be summarized as shown in the tables below.

Table 1. NEOTωIST Mission Objectives

#	Objective	Derived functionality
O1	Technology demonstration Kinetic Impactor	Impact target NEO with a spacecraft in hypervelocity regime with sufficient accuracy to ensure momentum transfer
O2	Technology demonstration of an observer spacecraft for impact verification	Demonstrate observation, from a flyby vehicle, of the impact event with sufficient quality to verify that the impact took place as required for deflection
O3	Deflection validation	Measure target NEO orbit or rotation before and after impact to prove transfer of (angular) momentum.
O4	β determinat. / quantification of momentum transfer augmentation from ejecta	Quantify the magnitude of momentum carried by the ejecta
O5	Observational data to validate/ improve impact modelling	Measurement of the dynamics and effects of the impact event Note: Observables and accuracy is subject to selection based on utility/cost assessment.

Table 2. NEOTωIST Mission Constraints

#	Constraint	Consequence
C1	Mission cost	Mission design and launcher selection shall be compatible with reduced budget compared to conventional deflection demonstration mission
C2	Cost & partnering flexibility	Mission concept shall allow tailoring according to budget, and offer opportunity for partnering
C3	Flexibility of implement. timeline	Multiple mission opportunities shall exist over an extended period of time to be compatible with different programme timelines

Most deflection demonstration missions of reference quantify momentum transfer from the impactor spacecraft to the target object (β quantification) by measuring a change in its heliocentric orbit. The change is typically so small that it must be performed via radio-science from a second Observer spacecraft which rendezvous with the NEO prior to impact. The Observer also characterises in the NEO in order to estimate mass and provide targeting information for the KI. This observer may also observe the dynamics of the impact and ejecta cloud. A critical issue in this case is finding a vantage point which allows good observation but also results in a low probably of being struck by ejecta debris. While this mission concept satisfies the objectives of a deflection demonstration mission superbly, the fact that it requires two full-fledged interplanetary spacecraft including their launch makes it potentially costly. Further, the need to rendezvous with the target is demanding in terms of delta-V and constrains target and trajectory selection as compared to an impactor-only mission like NEOTωIST.

2. NEOTωIST measurement concept and mission architecture

The mission concept is largely driven by Objectives 4 and 5 (see previous section), with the other objectives being covered by the chosen design solution.

Achievement of objective 4 means determination of the magnitude of the ejecta vector caused by the KI impact. Standard deflection demonstration missions quantify momentum transfer from the impactor spacecraft to the target object by measuring a change in its heliocentric orbit. In the case of the proposed NEOTωIST concept the NEO is struck off-center which changes its spin rate (illustration in Fig. 1). The induced spin rate change can be measured accurately by Earth-based telescopes via light curve measurements, constituting a highly robust retrieval method. This is further described in [1].

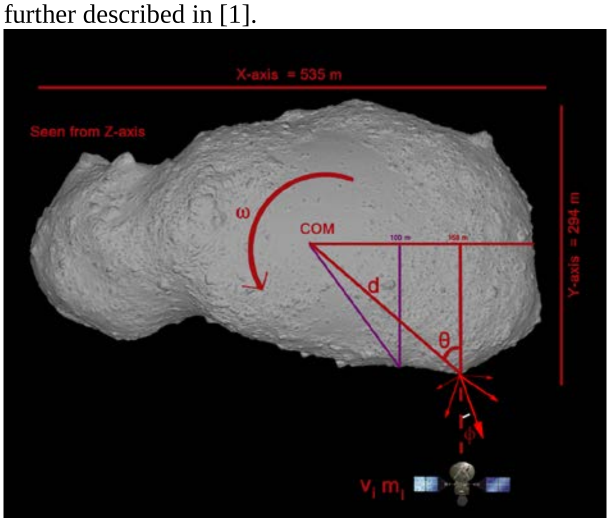


Fig. 1. Iimpact geometry and target object (Itokawa)

Relating the observed spin rate change to the momentum of the ejecta requires additional information about the mass properties of the object and the impact event geometry. The observation concept and mission architecture are largely defined the need to glean this information.

The information about NEO properties required for targeting and data interpretation is largely available through the fact that NEOTωIST is aimed at a known object. This object is the Itokawa asteroid (shown in Fig. 1), which has been thoroughly characterised by the Japanese Hayabusa mission. A further advantage of Itokawa is its elongated shape which makes a clean off-center strike easier and causes clearly observable brightness fluctuations as it rotates.

Information about the impact geometry is substantially improved over what is known a-prior by observations from small sub-spacecraft which are deployed from the KI prior to impact. These sub-spacecraft also observe impact dynamics relevant to model validation (O5). The resulting mission architecture is illustrated in Fig. 2.

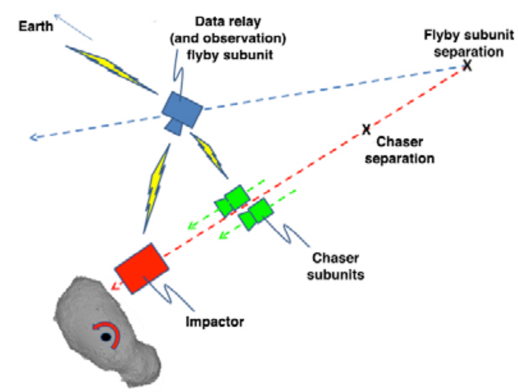


Fig. 2. Overview of mission concept

The baseline architecture consists of:

The Kinetic Impactor, which performs all functions associated with the interplanetary transfer, performs terminal homing on the target, and acts as a carrier for the sub-spacecraft.

A Fly-by Module (FBM) which is released onto a safe flyby course prior to impact. The FBM performs observations of the impact from different positions along a trajectory roughly parallel to the impactor trajectory, including a view from 90° with respect to the impact velocity vector. This perspective yields more information about the geometry and dynamics of the ejecta cloud than for instance a view along the impact trajectory. After deployment, the FBM also functions as a data buffering and re-transmission node for the other vehicles of the constellation.

One or two Chasers which are released from the Impactor follow it along its terminal trajectory, to be destroyed either by impacting the NEO or debris from the impact. This vantage point potentially provides an additional view of the ejecta cloud along its central axis. However, the unique observation opportunity that the Chasers offer is that of observing the impact crater, which the FBM cannot do because of obscuration of the crater by ejecta at the time when it may be geometrically possible. To have a chance of observing the impact crater, the Chaser must follow the Impactor with a sufficient delay (10s of seconds) to allow dispersion of most of the ejecta for a clear view on the crater. Crater characterisation is useful because it can constrain the assumptions about total volume of ejected material.

Table 3. Information return by Mission Option & means of observation

Info type	MO-1	MO-2	**MO-3 BL**
Itokawa mass properties	Hayabusa mission	Hayabusa mission	Hayabusa mission
Spin rate change	Earth-based obs.	Earth-based obs.	Earth-based obs.
Impact location	Impactor Nav-camera or targeting accuracy knowledge	Impactor Nav-camera & FBM observation	Impactor Nav-camera & FBM observation & Crater imaging
Ejecta moment-um vector direction	No obs.; modelling based on Asteroid geometry	FBM observation of ejecta geometry	FBM observation of ejecta geometry
Ejecta mass	No obs.	FBM observation; optical density of ejecta cloud	FBM observation ; Crater volume est. from Chaser observation
Impact physics	No obs.	Ejecta geometry & expansion rate, density distribution; Radiometry of blast	Ejecta geometry & expansion rate, density distribution from two vantage points and times; Radiometry of blast

The baseline mission scenario consists of all elements mentioned above (Mission Option 3). However, different implementation options with different levels of information quality are possible and can be chosen to match budget availability (C2). Other possibilities include the Impactor + Flyby Module (Mission Option 2) or only the Impactor (Mission Option 1). Table 3 shows what information is available and how it is obtained for the different Mission Options (MO). This provides an idea of what the de-scoping options mean in terms of mission return.

3. Mission opportunities and transfer analysis

The analysis for potential mission opportunities has been performed assuming a launch on VEGA, which is selected as a low-cost European launch option. It places stringent mass constraints on the mission. As shown in Table 4, good Mission opportunities exist at regular intervals of approximately 3 years. Transfer duration is on the order of 2-3 years in most cases, and communications distances at time of impact are manageable at below 1 AU, sometimes significantly less.

Table 4. Mission opportunities and parameters

Arrival year	2024	2027	2030	2033	2036
Flight time (days)	581	942	589	95	503
Earth V_∞ (km/s)	2.07	2.39	2.77	2.18	1.62
Arrival V_∞ (km/s)	8.01	8.48	8.89	8.98	8.93
Payload mass * (w/o LPF, kg)	424	401	372	416	447
Sun-Itokawa-impactor angle at impact (deg)	30	24	21	12	13
Earth-Itokawa distance at impact (AU)	0.95	0.68	0.39	0.09	0.25
Earth-Itokawa-impactor angle at impact (deg)	90	90	90	74	42

* Note that the payload mass refers to the mass delivered excluding the propulsion module, i.e. mass available for Impactor Mission Module, FBM, & Chaser

The 2030 mission opportunity is marginal in terms of delivered payload mass. The mass situation for this and all other mission opportunities will be somewhat relaxed once the upgraded VEGA-C launcher goes into service.

4. Sub-spacecraft deployment and observation strategy

The primary observation target of the FBM during the flyby is the ejecta cone that develops after the impact. The high velocity, energies and observation geometries makes this a highly dynamic and challenging imaging task. A suitable image tracking strategy has to be selected that is able to cope with errors in the flyby geometry that in turn has to be robust enough to cope with the uncertainties related to the ejecta cone.

Most of the operations take place in the last 64 hours of the mission (16h for terminal navigation + 48h for post-impact data downlink), with the observation core campaign that lasts only 10 seconds during the most critical part of the flyby.

All this aspect regarding the FBM, its flyby geometry and interactions with other spacecraft will be treated in the following section.

4.1 Flyby geometry

The flyby geometry, shown in Fig. 3 can be simplified by a right triangle with vertices the impact point (A), the point at offset distance from the impact point (B) and the separation point (C). In a first order approximation it has been assumed that the normal of the surface at impact point has the same direction of the triangle side AC, whose length define the release distance d_r. This also means that the angle $\hat{A}$ is by definition assumed to be equal to 90°. The main parameters that drive the flyby geometry are the offset distance (d_{off}) represented by the triangle side AB and the release distance.

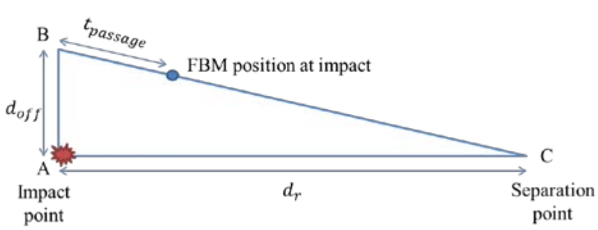


Fig. 3. Illustration of flyby geometry

To design the FBM subsystems a strategy has been developed that involves the development of a numerical model of the flyby event alongside with a simplified model of the ejecta cone. This has been performed in order to find a way into the design of the key spacecraft subsystems, which are highly dependent on the geometry chosen.

The ejecta cone has been modelled in a first order approximation assuming a linear expansion behaviour. The uncertainties in the cone geometry, that are also subjects of the observations to be made, appear in the half cone aperture angle δ and in the characteristic velocity of the expelled material V_{char}, which in turn depends on the impact velocity. To allow the FBM to sense the ejecta cone from different points of view in order to determine the geometry of ejecta cone it is important that the point of view vary fast enough in relation to the ejecta dynamics. For assessing this the ejecta has been considered visible from a brightness point of view for a period of 10 s after the impact. The matching of the flyby geometry and the ejecta dynamics is essentially driven by the choice of the passage time ($t_{passage}$) that is the time interval between the impact and the passage of the FBM at point B. On the other hand the offset distance drives how fast the point of view changes during the observation phase.

To find appropriate values for these parameters an extensive test campaign has been performed and a good set of d_{off} and $t_{passage}$ has been chosen such as to fulfil the described considerations.

A reference scenario has been selected. Table 5 sums up the main parameters the flyby numerical model of this scenario.

Table 5. Flyby model parameters

Parameter	Value
d_{off}	69.5 km
d_r	28836 km
V_{IM}	8.01 km/s
V_{FBM}	8.0056 km/s
$t_{passage}$	2 s
ΔV_N	4.45 m/s
ΔV_S	19.29 m/s
δ	40÷50 °
V_{char}	0.01÷0.1 V_{IM}
t_{obs} (core)	10 s
t_{obs} (extended)	10 s
FOV	8.35°
# pixel	2048 x 2048

*N=Alongtrack, S=Crosstrack (with respect to an impactor-centred reference system)

To link together flyby event, ejecta cone and camera parameters it is necessary to define a pivoting point or aiming point the camera should follow during the observation phase. This is a crucial step since the definition of such point influences many key subsystem parameters. In the previous phases of the project such point has been chosen to be the impact area on the asteroid, however such definition doesn't take into consideration the ejecta dynamic and may end up in sizing a bigger FOV than required. For this reason the pivoting point has been defined in a more dynamic fashion as that point that is halfway between the top of the ejecta cloud and the base, on the cone line that is closer to the FBM during the flyby.

Note that such definition of pivoting point proved to work out well for the spacecraft design, but might not represent the absolute optimal solution. Its optimization is left for further study in next phases of the project.

For representation purposes the ejecta cloud as seen from the spacecraft point of view in 3 different moments of time is represented in Fig. 4. The green area represents the half part of the ejecta cone that is closer to the FBM while the red part is the other half that is further away. The white object is the target asteroid Itokawa. It is possible to note from these pictures how the choice of the flyby geometry along with a proper timing of the events influences the point of view of the observation campaign.

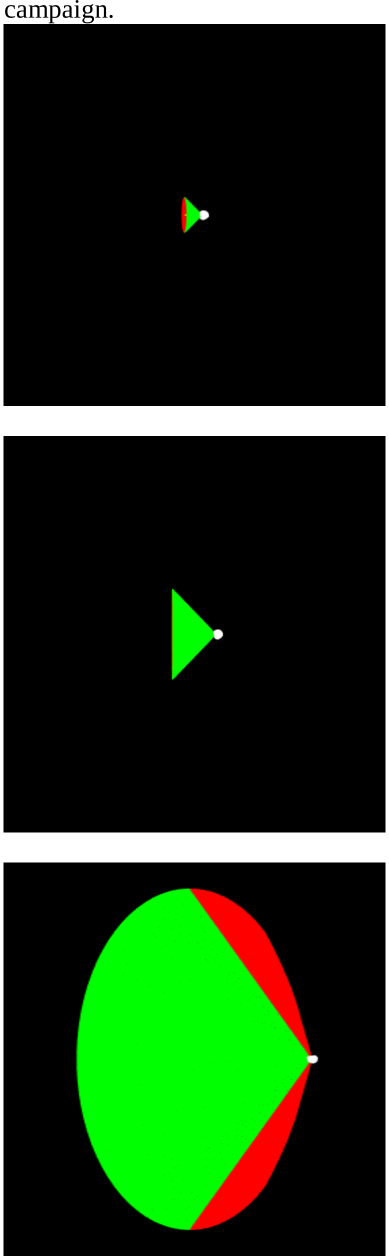


Fig. 4. Ejecta cloud and Itokawa seen from FBM at 0.8s, 2s and 10s after impact

In particular it is possible to note that in the reference scenario it would be possible to observe the cone from a change of perspective of about 58°, without excessive stresses on the turning manoeuvre or on the possible resolution as is shown later on.

It is worth pointing out that the sensor FOV has been chosen in relation to the ejecta cone in such a way that the latter is always in the FOV of the sensor during the whole observation phase.

The observation phase has been dived into two sections, a core section that last between 0 s and 10 s (where 0s represent the impact time) and an extended section that is divided in 5 seconds before impact and other 5 seconds after end of the core phase.

Fig. 5 shows the changes of the minimum FOV to use depending on the offset distance chosen and on the uncertainties on the ejecta previously discussed. The lower family of curves represents the FOV for a characteristic velocity (ejecta expansion velocity) equal to 1% of the impact velocity while the upper one represents the case where the characteristic velocity is 10% of the impact velocity. It is visible that uncertainties on V_{char} have a strong influence in the minimum FOV while uncertainties on δ have a smaller but not negligible influence. As it is possible to see from Fig. 5 increasing the offset distance has the effect of decreasing the required FOV and thus increases the size of the optics to use. Note that however the minimum FOV it is driven by different factors whether a close or far flyby is considered. In the first case the minimum FOV is driven by geometric consideration upon the change of point of view of the FBM during the observation phase. On the other hand in the second case the minimum FOV is mainly determined by the rapid expansion of the ejecta cone seen from an almost fixed point of view.

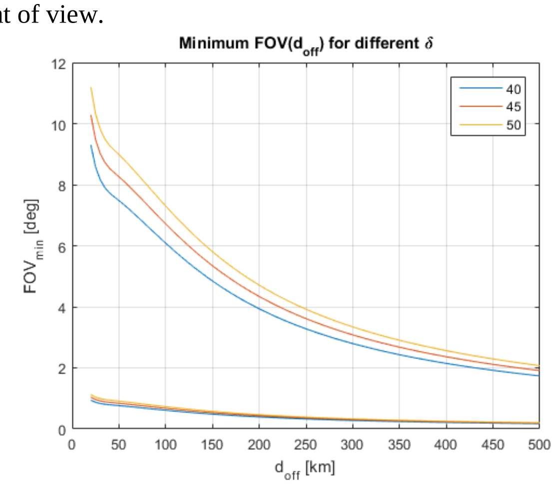


Fig. 5. Minimum FOV as a function of offset distance, tested for different ejecta uncertainties

As previously pointed out a strong dependency on the ejecta cone parameter uncertainties has been generally observed for all the parameters of the numerical model. The worst case scenario in terms of ejecta uncertainties has been considered in the reference scenario. The result is that a FOV of at least 8.35° is necessary to observe the ejecta cone, when assuming it has a wide angle and fast expansion velocity.

Another fundamental result of the numerical model that drives the design of the spacecraft has been given by the necessary manoeuvre to rotate the camera FOV around the pivoting point. Defining the azimuth angle as the angle formed by the vector from the FBM to the pivoting point and the trajectory line it is possible to investigate the slewing performance of the required for flyby observation. Fig. 6 Show the azimuth profile and azimuth rate needed to follow up the ejecta cone during its expansion during the flyby in the reference scenario. The profile is shown only for a short time interval after impact (that occurs at 0 s) to highlight the magnitude of the maximum azimuth rate. Such outcome from the numerical model translates into a fundamental system requirement to take into consideration for the spacecraft design.

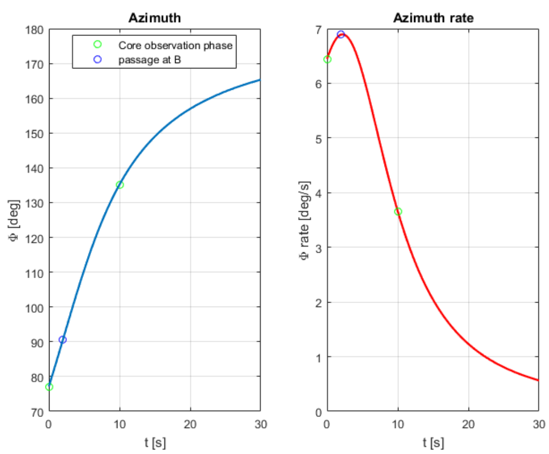


Fig. 6. Azimuth profile and azimuth rate

Fig. 5 shows the change of a pixel size projected from the FBM to the pivoting point for flyby scenarios with different offset distances. It is possible to see how generally performing the flyby further from the impact area has the disadvantage of increasing the minimum pixel size but has at the same time the non-negligible advantage to decrease the variation of the resolution over a longer period of time. On the other hand a closer flyby would result in a better minimum resolution but only for a limited amount of time.

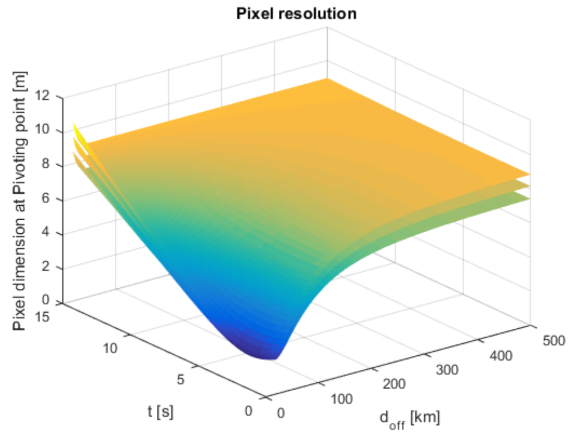


Fig. 7. Projected pixel size at pivoting point during the flyby for different offset distances and δ.

From all the previous results it is possible to understand why a reference scenario at 69.5 km has been considered a good compromise. This scenario allows a relatively fast change of the point of view (about 58°) during the 10 s core observation phase, with a good level of pixel size over time, a reasonable azimuth peak rate (~ 6.9 °/s) and an adequate FOV (8.35°) and size of the optics.

4.2 Tracking strategy

A tracking strategy for the flyby scenario requires certain flexibility to cope with the possible uncertainties and errors given by the real flyby geometry. In this sense the possibility to feed the system directly with a fixed azimuth profile as reference command to follow in an open loop fashion is not pleasing.

A more robust approach proposed for this mission is a tracking strategy made up by the combination of a geometry reconstruction phase (T1) and an open loop phase (T2).

4.2.1 Geometry reconstruction phase (T1)

During this phase observations about the real flyby geometry are collected, as the FBM approaches the target. These can be divided into two groups, the one collected by the FBM and the one taken by the KI and exchanged with the FBM.

From observations of the latter group it is possible to make use of the fact that the KI position error relative to the target constantly decreases between FBM separation and the precision impact. It is therefore possible to use the communication link between the two spacecraft to communicate to the FBM a history of the correction manoeuvres performed by the Impactor after separation. If the uncertainties of the ΔV at separation are carefully threated it is possible for the FBM to use this information to solve the flyby geometry.

In principle the communication link between the two spacecraft itself could be used as additional source of information that gives relative distance and velocity between the two (not currently baselined).

Another possibility is to use the on-board instrumentation of the FBM to perform visual navigation from the time when the illuminated asteroid becomes visible to the payload camera (possible at subpixel apparent size). Assuming that after separation the FBM attitude allows to have the asteroid target within the field of view, a constant inertial orientation makes it possible to see the asteroid at sub-pixel level moving around the sensor with a measurable velocity. Comparing such velocity with a predefined catalogue it is possible to obtain information about the real flyby geometry. As distance from the asteroid decrease the target asteroid will start to be imaged from tens to hundreds of pixels. At this point a more sophisticated algorithm such as a template matching technique could be used comparing the real illuminated shape of the asteroid with the one from a predefined catalogue. Information about the asteroid-pixel velocity, number and shape as viewed in the sensor can be therefore used to solve the real flyby geometry.

The techniques cited above can be used in combination or in redundancy depending on the performances and computational efforts required. The current baseline is trajectory reconstruction based on visual observations during approach.

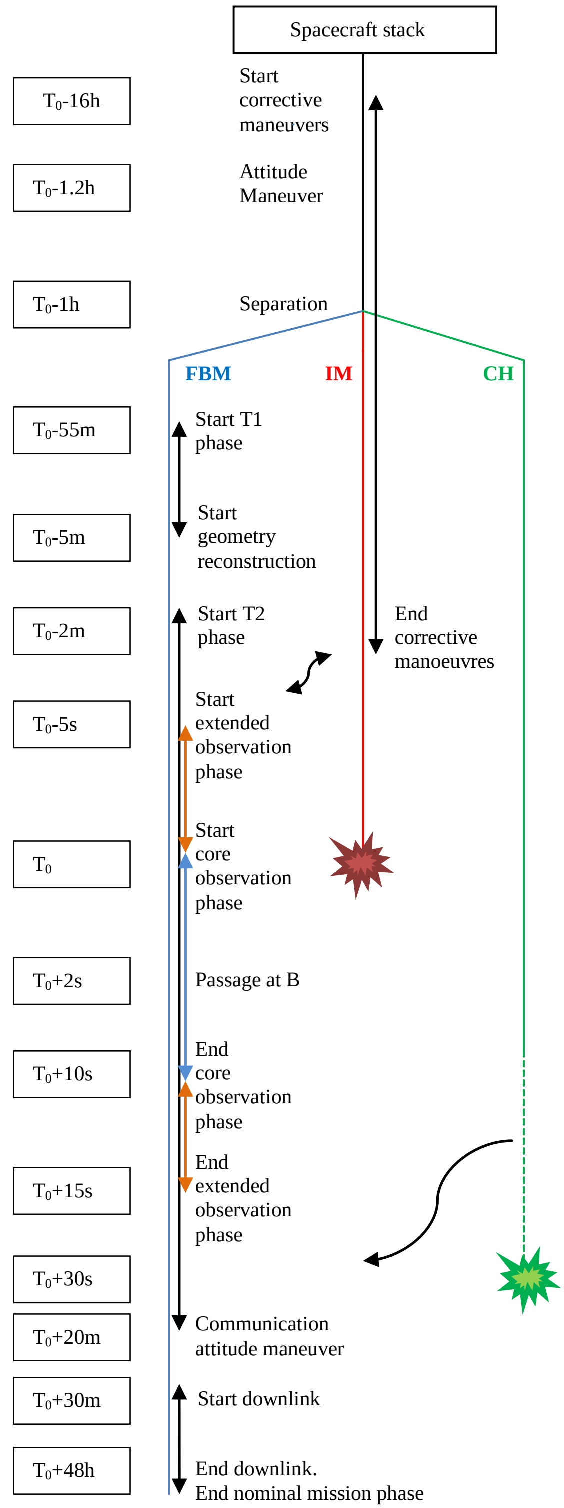
It is important to note that all of the techniques discussed above are not considered feasible for the phases of the flyby after the impact because of the loss of the KI and because of the unknown brightness of the impact and ejecta cone. This may temporarily "blind" the sensors or might require an additional complexity in the algorithm design, compromising the turning manoeuvre robustness during the most critical part of the flyby. For these reasons we choose to make use of an open loop tracking strategy for the phase lasting from immediately before the impact until the end of the observation.

4.2.2 Open loop phase (T2)

It has been explained how during phase T1 observations from different sources can be collected. The idea is to make use of these observations to best fit a numerical flyby model in order to determine the real flyby geometry. Once the main flyby parameters as offset distance, passage time and release distance are determined it will be possible to determine the reference command to allow the sensor FOV to perform observation rotating its centre around the predetermined pivoting point.

4.3 Operations timeline

In the following section a possible timeline involving all the spacecraft's of the mission has been designed. T_0 is the impact time, which has been set at 0s.



5. Communications concept

A core feature of the NEOTωIST concept is its communications architecture. During the cruise of the spacecraft stack all TT&C is performed via a link to the Impactor. In the terminal phase shortly before and after impact, multiple commination links connect the different sub-spacecraft and Earth, as summarized in Table 6. The FBM functions as central data buffering and communications node for the constellation, with most of the payload data being downlinked to Earth from the FBM after the impact and flyby.

Table 6. Overview communications links

Link	Type	Purpose
Impactor <-> Earth	X-band, low rate	Telemetry and compressed images from targeting camera for redundancy
Impactor -> FBM	X-band, high rate	Telemetry & manoeuvre history for relative trajectory estimation between KI and FBM ; KI Nav-camera images for later downlink to Earth
Chaser ->FBM	X-band, very high rate	Transmission of chaser imagery in real time (transmission before Chaser destruction)
FBM <-> Earth	X-band, medium rate	Delayed downlink of telemetry and observation data from the terminal phase, for all sub-spacecraft

A summary of the data budget and link speeds (inter-satellite and payload data dump) is given in Table 7. Some of the inter-satellite (ISL) link speeds are challenging. Feasibility is supported by the fact that communications need to be maintained only for short periods, which supports the use of very high Tx power (energy availability and thermal design).

Table 7. Data budgets and link speeds

	FBM*	KI**	CH***
Spacecraft Data			
Frames/s	15 (5)	1	1
# Pixel	2048 x 2048	1024 x 1024	2048 x 2048
Obs time [s]	10 (+10)	40	30
Payload data[Mbit]	7200	250	720
Telemetry [Mbit]	200	1	3.6
Data [Mbit]	7400	251	723.6
Inter-spacecraft links			
Max range [km]	/	72	235
Band	X	X	X
Comm. Window [s]	/	20 + 40	30
Link data rate [Mbit/s]	50	6.25	24.1
Antenna type	HGA & MGA	MGA	MGA
Downlink performance			
Range [AU]	1		
Data rate [Kbps]	50		
Data[Mbit]	8374.6		
Downlink time [h]	46.53		

6. Design of impactor spacecraft and stack

The KI design is envisioned as an updated version of the Impactor design for the Don Quijote (DQ) mission. The concept of the DQ impactor is shown in Fig. 8. The spacecraft consists of a large propulsion module which was flown for the Lisa Pathfinder (LPF) Mission, and a small Mission Module which is mission-specific and performs other functions required for the mission (interplanetary transfer and terminal impact GNC).

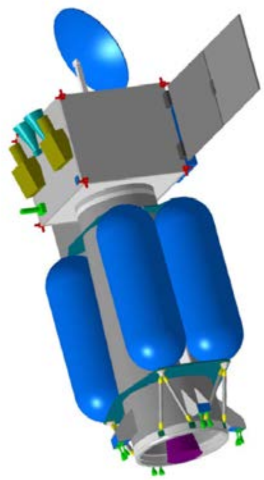


Fig. 8. Impactor concept, based on Don Quijote design

The only modification to the LPF propulsion module is the addition of manoeuvring thrusters needed for terminal GNC.

Main updates of the Mission Module consist of tailoring power and communications systems to the mission geometry. The communications system can be simplified since only low rate transmission of TT&C and some compressed images are required between the KI and Earth. Avionics, in particular power and data handling, are updated to save mass and power. Table Table 8 shows a top-level mass breakdown of the spacecraft stack.

Table 8.Top-level spacecraft stack mass budget

Vehicle stack mass synopsis		
	Dry [kg]	**Wet [kg]**
Fly-by Module	104.1	104.6
Chaser	25.7	26.2
Impactor Mission Module	263	263
Propulsion module (incl. resid.)	280	1420
Stack	673	**1814**
Stack at launch incl. adapter		**1929**
Delivered mass w/o prop. mod.		**394**
Min. impacting mass	**543**	

The masses shown contain maturity margin between 5% and 20%, as well as 20% system margin. For some mission opportunities (see section 3) the combined mass of Mission Module, FBM, and Chaser is marginal. The VEGA-C launcher upgrade will add margin in all cases.

7. Design of Flyby Module

The Flyby Module (FBM) functions as the main imaging platform for the impact event, and as a data buffering and communications node for the satellite constellation.

7.1 Design drivers for FBM

The main design drivers for the FBM are:

Table 9. Fly-by module design drivers and implications

Driver	Design implications
High quality imaging of impact event during high velocity pass	- Relatively powerfully optics - Fast actuation of image tracking - Precise and flexible image tracking GNC - Precision inertial stabilisation during imaging
Data buffering & transmission to Earth for entire constellation	- Robust high-rate ISL links - Powerful Tx capability for Earth downlink - Attitude control authority for duration required to dump payload data
Size and mass constraints imposed by overall mission budget	- Small satellite design approach - Low mass equipment selection - Critical approach to redundancy
Mission cost constraints	- See above - Use of existing equipment where possible
Deep space environment	- Radiation tolerance for some attractive LEO equipment to be checked

Next do these considerations the design of the FBM is influenced by the decision to not pursue a solution which is highly integrated with the primary mission vehicle, here the KI. This leads to a certain amount of redundancy between both vehicles, for instance in communications or power generation. The rationale for this is mainly programmatic with the following main reasons:

- The ad-on character of the FBM allows flexibility in the choice of whether to implement it or not
- Vehicle interfaces and design are simper (no cross feeding of power, less data & control interfaces, less configuration constraints).
- The developed solution can be used as a basis for add on capabilities on other mission scenarios (e.g. very low flybys of planetary objects for imaging or sampling)

7.2 Fly-by module payload

The main payload of the FBM consists of a panchromatic camera with a 2048 x 2048 sensor. Since the peak azimuth rate of the reference scenario is high (6.9 °/s), the use of a rotating pointing mirror instead of an attitude manoeuvre is considered a more appealing solution. Fig. 9 outlines the proposed solution for the FBM field of view tracking.

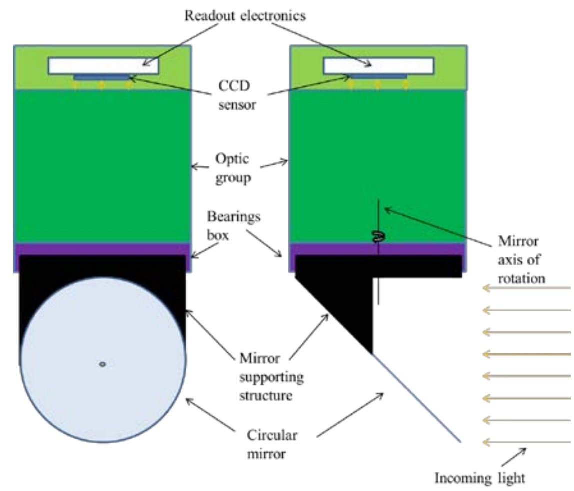


Fig. 9. Payload subsystem with external pointing mirror

The use of a pointing mirror has several advantages over an attitude manoeuvre. Less mass is rotated, which means that less power is needed to move the mirror. The level of vibration is smaller and that the angular position of the mirror can be controlled more easily. Given this design it is easier, stable and cheap to design a system where only an optical component of the payload is put in a rotational state and not the whole spacecraft. A solution with an external circular rotating mirror is therefore proposed for the FBM camera.

Next to the camera the FBM features a Radiometer which samples the brightness of the impact at high frequency (kHz). Since it is non-imaging, the optics and detector are compact and simple. Target pointing is integrated with the main camera, whereby pointing requirements are much less stringent. High-frequency radiometric measurements provide information about dynamics of the impact event during the first milliseconds.

7.3 *Outline FBM platform*

Table 10 gives an overview of the FBM specifications and equipment type.

Table 10. Flyby module specifications

Item	**Description**
Mass	Dry/ Wet: 104.1 / 104.6 kg
Power	DC: 100 W S/A size: 0.75 m² Battery: 150 Wh (usable)
Dimensions	Bus structure: 100 x 100 x 25 cm³
Payload	Medium angle camera Field of view: 8° Aperture: ~ 9 cm Detector: 2048 x 2048 pixel Max. resolution at 70 km: 5m Field of view pointing: pointing mirror Image targeting based on visual navigation using payload camera Non-imaging radiometer
Power and data handling	PROBA-NEXT avionics integrated power and data handling, >1 Gbit mass memory
Comms.	Earth-link: X-band, 1m HGA, 20 W RF, 50 kb/s Impactor link: X-band, MGA, Rx only, < 10 Mbit/s Chaser link: X-band, MGA, Rx only, ~ 20 Mbit/s
AOCS	3-axis stabilised 2 x STR (DTU Micro ASC) 2x Coarse sun sensors 3 x Micro-wheels, 0.42 Nms (SSTL) IMU (DTU, int. with STR electronics) RCS: 4 x cold gas thrusters for momentum management (0.5 kg of N_2)
Thermal	Heaters & radiators

Fig. 10 and Fig. 11 show the configuration concept of the FBM.

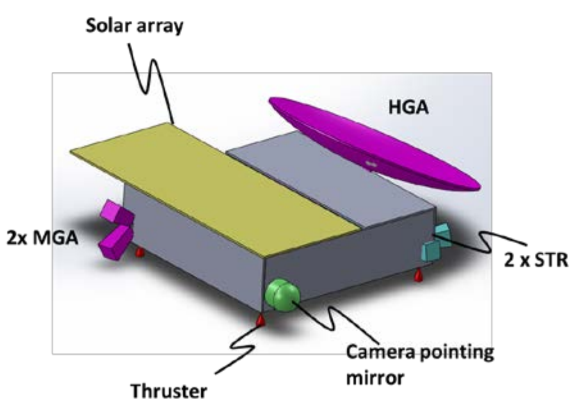


Fig. 10. Flyby module configuration

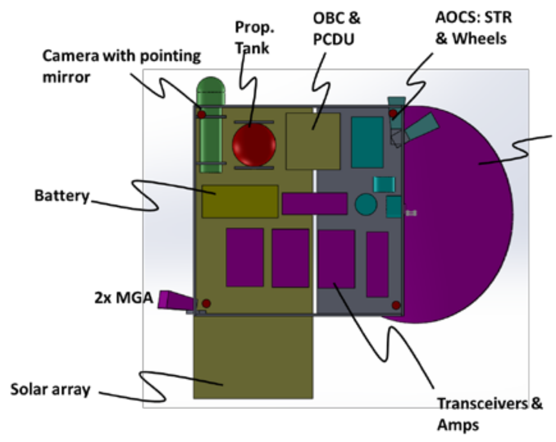


Fig. 11. Flyby module equipment configuration

The FBM configuration is roughly a box shape (dimensions 100 x 100 x 25 cm³), with the mounting and orientation of the equipment driven by the communications and imaging geometry. This geometry is illustrated in Fig. 12. Note that the angle between high gain antenna and solar array must be adjusted according to the specific mission opportunity. The FBM design allows implementing this via and antenna mounting that can be adjusted accordingly before launch.

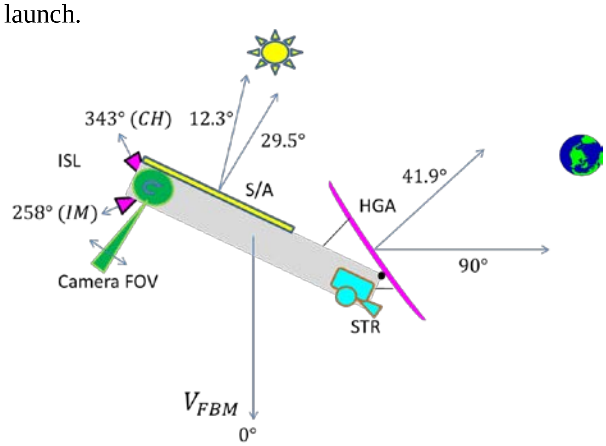


Fig. 12. FBM communications and imaging geometry during flyby and data downlink

The FBM attitude is inertialy fixed using the reaction wheels and star trackers during the flyby. Image pointing is performed by a pointing mirror as described above. The flyby attitude is adjusted after release from the Impactor in two steps:

Coarse orientation: Based on known approach trajectory, attitude is adjusted such that target NEO is in field of view.

Fine adjustment: based on visual tracking of target NEO, during approach attitude is adjusted to place target object at 0° elevation (center of field of view) during flyby.

After the flyby the FBM is oriented for optimal power generation and data dumping to Earth. The cold gas propulsion system is intended to provide additional momentum management capability during this phase.

8. Design of Chaser spacecraft

The Chaser functions as a secondary imaging platform with the primary goal of imaging the impact crater after the ejecta cloud has dispersed sufficiently to be transparent. In addition it may perform long range ejecta observations from a secondary vantage point. The significant temporal delay required to allow ejecta dispersion requires a relatively large along-track separation between Chaser and the Impactor, which in turn requires a relatively large delta-V after Chaser separation. Since the Chaser follows the Impactor without lateral offset, the default scenario is its destruction by collision with either the target NEO or ejecta debris.

8.1 Design drivers for Chaser

The main design drivers for the Chaser are:

Table 11. Chaser design drivers and implications

Driver	Design implications
Uncertainty in imaging geometry & small target	- Wide field of view & - Relatively high resolution (large number of pixels)
Real-time transmission of imaging data before vehicle destruction	- Robust very high-rate intersatellite link
Large delta-V for separation from Impactor	- High-Isp propulsion solution
Mission cost and mass constraints	- Small satellite design approach - Critical approach to redundancy - Use of existing equipment where possible - Simple AOCS approach
Deep space environment	- Radiation tolerance for some attractive LEO equipment to be checked

8.4 Outline Chaser design

Table 10 gives an overview of the Chaser specifications and equipment type.

Table 12. Chaser specifications

Item	**Description**
Mass	Dry/Wet: 25.7 / 26.2 kg
Power	DC: 40 W Battery: 60 Wh (usable)
Dimensions	20 x 30 x 10 cm³
Payload	Medium angle camera Field of view: 16° Aperture: still to be determined Detector: 2048 x 2048 pixel Max. resolution: 4m @ 30 km, 2m @ 15 km No active target tracking
Power & data	Cube-sat/ small sat equipment (details still to be selected)
Comms.	Link to FBM: X-band, MGA, TX only, ~ 20 Mbit/s
AOCS	Stabilization along velocity direction with single uncontrolled momentum wheel, spun up before ejection from Impactor
Propulsion	Hydrazine, single thruster in anti-velocity direction, stabilization with momentum wheel, dV capability 70 m/s
Thermal	Heaters & radiators, non-stationary design for terminal phase possible

Fig. 13 shows a preliminary sketch of a possible internal configuration of the Chaser and the relative position of the FBM and imaging target. A tentative shape goal is a 6U-Cubesat envelope.

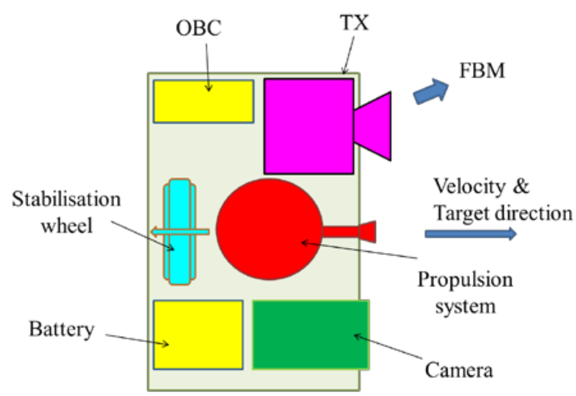


Fig. 13. Chaser configuration sketch

6. Conclusions

The NEOTωIST mission concept has been presented as a mission option which achieves the chore objectives of a Kinetic Impactor Demonstration Mission while offering the potential to implement the mission at a lower cost and in a more flexible manner than typical reference concepts. The concept achieves this by:

- One, adopting a new measurement principle for quantifying the achieved momentum transfer to the target NEO, i.e. measuring the spin state change rather than the heliocentric orbit change
- Two, replacing the large rendezvousing reconnaissance spacecraft of a typical demonstration mission with small sub-spacecraft deployed from the main spacecraft shortly before impact

The work presented shows a snap-shot of ongoing feasibility work. The preliminary conclusion is that there are technical challenges to be overcome, but no show stoppers have been identified. The mission utility to cost ratio is high, and the implementation timeline and scope are flexible. These are highly attractive features given the programmatic constraints that such a demonstration mission faces.

Future work is envisioned in particular on refinement of the payload design, detailed development of the combined AOCS image targeting concept, iterations on equipment selection and overall vehicle design, optimisation of communications and data management, as well as estimation of vehicle and mission cost.

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