Acta Astronautica ∎ (∎∎∎∎) ∎∎∎–∎∎∎

Contents lists available at ScienceDirect

Acta Astronautica

journal homepage: www.elsevier.com/locate/actaastro

Closed loop terminal guidance navigation for a kinetic impactor spacecraft

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ARTICLE INFO

Article history:
Received 30 October 2013
Received in revised form
11 February 2014
Accepted 25 February 2014

Keywords:
Kinetic impact
Asteroid deflection
Autonomous navigation
Closed loop guidance

ABSTRACT

A kinetic impactor spacecraft is a viable method to deflect an asteroid which poses a threat to the Earth. The technology to perform such a deflection has been demonstrated by the Deep Impact (DI) mission, which successfully collided with comet Tempel 1 in July 2005 using an onboard autonomous navigation system, called AutoNav, for the terminal phase of the mission. In this paper, we evaluate the ability of AutoNav to impact a wider range of scenarios that a deflection mission could encounter, varying parameters such as the approach velocity, phase angle, size of the asteroid, and the attitude determination accuracy. In particular, we evaluated the capability of AutoNav to impact 100–300 m size asteroids at speeds between 7.5 and 20 km/s at various phase angles. Using realistic Monte Carlo simulations, we tabulated the probability of success of the deflection as a function of these parameters and find the highest sensitivity to be due to the spacecraft attitude determination error. In addition, we also specifically analyzed the impact probability for a proposed mission (called ISIS) which would send an impactor to the asteroid 1999RQ36. We conclude with some recommendations for future work.

1. Introduction

On July 4, 2005, the Deep Impact (DI) spacecraft successfully impacted the comet Tempel 1 using a 400 kg impactor while the mother ship flew by the comet at a distance of 500 km and captured images of the impact. This event marked the first hypervelocity impact of a small solar system body, and, although not its primary goal, the mission demonstrated that such an impact could be accomplished with present day technologies, and on a relatively modest budget. Beyond its science benefits, the mission is notable because the same technology could be used to someday save the Earth from a devastating impact from a Near Earth Asteroid (NEA).

The potential threat from NEAs has been documented extensively [1] and cannot be understated. There are several options for mitigating this threat, including destroying the asteroid using a nuclear device, or altering its trajectory via gravity tractoring, impacting with another spacecraft (kinetic energy deflection), or a combination of any of these methods. The kinetic energy deflection technique is the most straightforward and easiest to implement using currently available technologies. The technique relies on both the direct momentum transfer from the impact, plus whatever is added by the momentum of the resulting ejecta. The momentum enhancement factor, however, is unknown, and it would be advantageous to narrow down the ranges via a deflection experiment. Ideally, this experiment would have two components: an impacting spacecraft and a second spacecraft in the proximity of the asteroid which measures the deflection (see Ref. [2] for a discussion on one proposed experiment). The main difficulty is the terminal guidance navigation; precisely hitting the target at high velocities where there is

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http://dx.doi.org/10.1016/j.actaastro.2014.02.024
0094-5765 © 2014 Published by Elsevier Ltd. on behalf of IAA.

little time to react and must be done largely autonomously onboard the spacecraft is a challenge. However, DI has demonstrated that this is feasible, and the next step is to show how the point solution used to successfully impact Tempel 1 can be generalized to cover the range of scenarios for an asteroid deflection.

DIs impact was made possible by the onboard closed loop autonomous navigation system (called AutoNav). The filter settings and sequence of events performed by AutoNav to achieve the impact were determined through simulations to maximize the probability of impact for this particular spacecraft and scenario. Some of the critical parameters were the size of the comet (roughly 6 km in diameter), approach velocity (10.5 km/s), and the approach phase angle (62 deg). The parameters for a NEA deflection could be considerably different. The range of possible sizes could be as low as 100 m in diameter or less, the approach velocity could range from below 10 km/s to as high as 20 km/s, and the approach phase could range from near 0 deg to near 180 deg (i.e., fully lit or no solar illumination). Thus, before a deflection mission is undertaken, it is important to understand how the terminal guidance will perform under a range of conditions.

In this paper, we expand the experience base of using AutoNav for terminal guidance in an asteroid deflection mission and parameterize the probability of achieving a successful impact on a sample set of deflection missions. We first determined the ranges of parameters for deflection missions through example scenarios from the literature in order to define the scope of conditions that AutoNav needs to handle. This is combined with information on several important spacecraft hardware considerations that affect AutoNav performance and their interaction with the mission parameters. Finally, Monte Carlo simulations of AutoNav terminal guidance were performed on selected scenarios, varying all relevant parameters, to obtain statistics on probabilities of impact. In addition, a specific case, the ISIS mission, was analyzed. This mission is a proposal to impact the asteroid 1999RQ36 in early 2021 while the OSIRIS-REx mission is already there, such that the latter can observe the impact and measure the subsequent deflection. The simulations are run using high fidelity models which describe the spacecraft trajectory, spacecraft attitude errors, and ephemeris errors of the target body. The simulations include: (1) generation of realistic images using triaxial ellipsoid shape model and the parameters of the camera, (2) determination of the orbit using a batch least-squares filter, and (3) maneuver targeting to achieve impact conditions.

2. Deep space navigation

Before we describe the AutoNav system used for terminal guidance, we provide here a brief overview of deep space navigation in general. The first step in any deep space mission is to design the reference trajectory to achieve the target conditions, which, in this case, is to impact a candidate set of asteroids or a particular one. The techniques to design these trajectories, optimizing parameters of interest such as fuel expenditure or time-of-flight, are covered elsewhere (see Ref. [3] for an example) and not in the scope of this paper. However, the ensemble set of trajectories found in Ref. [3] was used to define the scope of the problem and will be referred to later.

Once a particular set of trajectories is found, the next step is to analyze them for its flyability from a navigation perspective. This includes performing linear covariance analyses to determine the navigation delivery performance and statistics on maneuver ΔV requirements for the mission. The covariance analysis involves using a realistic tracking schedule to simulate tracking data and fitting the data in a least-squares process to determine the orbit. The tracking data includes two-way Doppler and range, plus optical data (images of the target object taken with an onboard camera). The least-squares fitting provides error covariances which statistically describe how accurate the spacecraft can be navigated. This also provides inputs to simulate the maneuvers required to achieve the target; this is done via Monte Carlo simulations which sample the orbit determination errors to design the maneuvers. The end result describes the amount of fuel needed to deliver the vehicle to the target, or in our case, to the start of the autonomous phase. When the mission is actually flown, real tracking data replaces the simulated ones, and the best fit orbit is used to design the maneuvers. If all has gone according to plan, the delivery errors to the target and the fuel required will be within the statistics of the pre-flight analysis. We will not cover the details of navigation for the launch, cruise, and target approach mission phases; for a general overview deep space navigation techniques, see Refs. [4,5] show an example of its application for a comet flyby mission which closely resembles that of an asteroid impactor.

For this paper, we are concerned with the impactor's terminal guidance navigation, which we define as the phase beginning roughly 2–3 h prior to impact. At this stage, ground navigation techniques are impractical due to the round-trip light time and the onboard navigation system, AutoNav, must be used. Details of the AutoNav system and how it was used on various missions can be found in Refs. [6–8]; here we will briefly describe the orbit determination filter as used by AutoNav in a subsequent section. First we describe some other considerations necessary to completely describe the problem set up.

3. Asteroid ephemeris

In order to successfully hit an asteroid with a spacecraft, two critical pieces of information are needed. The first is knowledge of the spacecraft's trajectory as described above. The second is knowledge of the target asteroid's orbit. Preliminary estimates of the latter are provided through ground-based observations of the asteroid, primarily from optical telescopes, but also in a limited number of cases from radar bounces off the asteroid [2]. The accuracy of this method is dependent on many factors including the density, quality, resolution, and geometry of the observations, the orbital characteristics of the asteroid, and the length of time from the last observation to the time of spacecraft approach and impact. In general, however, we can say that the accuracy of the orbit from ground-based observations as the spacecraft approaches will be in the tens of km range.