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A CUBESAT ASTEROID MISSION: DESIGN STUDY AND TRADE-OFFS

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There is considerable interest in expanding the applicability of cubesat spacecraft into lightweight, low cost missions beyond Low Earth Orbit. A conceptual design was done for a 6-U cubesat for a technology demonstration to demonstrate use of electric propulsion systems on a small satellite platform. The candidate objective was a mission to be launched on the SLS test launch EM-1 to visit a Near-Earth asteroid. Both asteroid fly-by and asteroid rendezvous missions were analyzed. Propulsion systems analyzed included cold-gas thruster systems, Hall and ion thrusters, incorporating either Xenon or Iodine propellant, and an electrospray thruster. The mission takes advantage of the ability of the SLS launch to place it into an initial trajectory of C3=0. Targeting asteroids that fly close to earth minimizes the propulsion required for fly-by/rendezvous. Due to mass constraints, high specific impulse is required, and volume constraints mean the propellant density was also of great importance to the ability to achieve the required ΔV. This improves the relative usefulness of the electrospray salt, with higher propellant density. In order to minimize high pressure tanks and volatiles, the salt electrospray and iodine ion propulsion systems were the optimum designs for the fly-by and rendezvous missions respectively combined with a thruster gimbal and wheel system For the candidate fly-by mission, with a mission ΔV of about 400 m/s, the mission objectives could be accomplished with a 800s electrospray propulsion system, incorporating a propellant-less cathode and a bellows salt tank. This propulsion system is planned for demonstration on 2015 LEO and 2016 GEO DARPA flights. For the rendezvous mission, at a ΔV of 2000 m/s, the mission could be accomplished with a 50W miniature ion propulsion system running iodine propellant. This propulsion system is not yet demonstrated in space. The conceptual design shows that an asteroid mission is possible using a cubesat platform with high-efficiency electric propulsion.

I. INTRODUCTION

There is considerable interest in expanding the applicability of cubesat spacecraft into lightweight, low cost missions beyond Low Earth Orbit. A conceptual design was done for a 6-U cubesat for a technology demonstration to demonstrate use of electric propulsion systems on a small satellite platform.

An opportunity was proposed that the upcoming initial test flight of the Space Launch System ("SLS"), shown in figure 1, would be an available launch to inject a cubesat into a trajectory escaping Earth orbit. This mission, originally scheduled for December 2017, demonstrates the vehicle by putting the Orion capsule into a free-return trajectory that goes around the moon and then returns to Earth [1]. In addition to the Orion test, however, the mission is baselined to carry a number of cubesats, marking the first injection of cubesats into trajectories beyond Earth orbit.

The candidate mission was to test electric propulsion systems for cubesats, with the objective of not merely demonstrating the operation of the electric propulsion, but using it to do a mission.

The candidate objectives analyzed were a lunar mission, and a mission to visit a Near-Earth asteroid. Both asteroid fly-by and asteroid rendezvous missions were analyzed.

A candidate mission for the lunar mission was developed, in which the spacecraft is placed into a polar lunar orbit. Spiraling in to a circular lunar orbit using electric propulsion requires a total velocity change (ΔV) of 1.9 km/s, although the ΔV could be reduced to about half of this by switching to a highly eccentric orbit with with periapsis over the lunar pole. The science objective of the mission was to use a visible/Near IR laser reflectometer to probe the reflectance spectrum of the interiors of permanently-shadowed lunar craters. The science goal of this mission would be to determine whether the water believed to be present in these craters is in the form of highly-reflective ice, or in another form such as hydrated minerals.

The candidate asteroid mission analyzed was to perform a fly-by of a Near-Earth asteroid, with a science package consisting of cameras to image the surface, as well as a gravity package to measure the mass of the asteroid, allowing calculation of the density. An additional payload of a visible/NIR spectrometer to allow measurement of the asteroid type and surface composition was not baselined on this payload, but could be added as an additional science instrument.

The ΔV required for this mission turned out to be significantly less than the lunar orbiter. For a more ambitious goal, if the ΔV is available for a rendezvous mission, a lidar was added, allowing precision measurement of the shape and topography of the asteroid.

Because of the lower required ΔV of the fly-by asteroid mission, with the possibility of a more ambitious "stretch" goal with higher ΔV, the asteroid mission with was chosen for further analysis [2], although using the SLS to launch a lunar polar cubesat is a mission being developed by others [3].

The mission was given the name DAVID, standing for "Diminutive Asteroid Visitor using Ion Drive."

COMPASS Study

Once the mission, science & technology demonstration goals were established, the next step was to utilize the COMPASS team to analyze the mission and provide a vehicle conceptual design.

The COMPASS team, standing for *COllaborative Modeling for Parametric Assessment of Space Systems,* is a multidisciplinary concurrent engineering team whose primary purpose is to perform integrated vehicle systems analysis and provide conceptual designs and trades for both Exploration and Space Science Missions. The team was formally established in 2006.

Figure 1: Artist's conception of the SLS launch vehicle

The COMPASS study is an iterative process balancing the mission design and spacecraft capabilities. The objective of the COMAPSS study is:

- Determine if the mission goals are realistic and can be achieved within the constraints of a 6U cubesat and SLS launch
- Select target asteroid (based on determined achievable ΔV)
- Provide a conceptual design and configuration of the spacecraft
- Provide 30% mass margin on all systems
- Establish the mission Concept of Operations (CONOPS)
- Provide designs for each major system of the spacecraft (Power, Propulsion, Communications, Thermal, Avionics, Structures, Command & Data Handling)

Once completed, the COMPASS study establishes the concepts feasibility and limitations.

ASTEROID SELECTION

The single most critical issue in designing the mission was selecting a suitable asteroid. Because only a limited ΔV can be achieved within the constraints of a cubesat, the mission strategy is to chose as a target a Near Earth Object (NEO) which makes a close pass to the Earth in the timeframe of interest, and use the spacecraft propulsion system to place the spacecraft in a position such that the asteroid's trajectory brings it past the spacecraft. Within this constraint, it is desirable to find an object with a low velocity relative to the Earth, in order to make the fly-by at a low enough relative velocity for meaningful measurements. The availability of Near Earth Objects with Earth approaches at low velocities brings up the possibility of a more ambitious mission, a rendezvous mission, in which the spacecraft becomes co-orbital with the object, allowing significantly better science return.

It was necessary to find an object which would make an approach at a close distance to the Earth. 12,874 close approaches by Near Earth Objects were analyzed. The list then narrowed down to objects making nearest approach between 2019 (>1 year after launch date) and 2021.

The asteroid 2001 GP2 was chosen as a target:

- Closest pass: Oct 3, 2020
- Closest approach 0.5-4.3 time Lunar distance
    - (~100,000 to 1 million miles)
    - Further observations will decrease uncertainty
- V relative: 2.37 km/sec

Targeting asteroids that fly close to earth minimizes the propulsion required for fly-by/rendezvous. The ΔV required is still significant for a cubesat: ~400 m/s for the fly-by, with the fly-by of 2001 GP2 occurring in 2020, and ~2000 m/s for the rendezvous mission.

Many fly-by opportunities exist, but not all have such low relative velocity to Earth. EM-1 places the vehicle on a trajectory C3 of 0.2 km^2/s^2 (using a 15 m/s burn and a lunar fly-by). The analysis here assumed the mission is ejected from the carrier during the translunar coast, and that the on-board EP system adjusts the lunar fly-by trajectory to take the C3 down to 0).

SPACECRAFT

Mission Overview

The spacecraft design was based on a 6-unit ("6U") cubesat platform.

Figure 2 shows the design of the fly-by spacecraft.

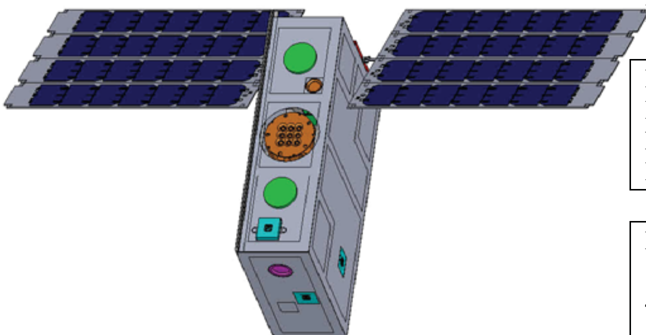

Figure 2: rendering of the DAVID spacecraft.

Propulsion Trade-off

Propulsion systems analyzed included cold-gas thruster systems, pulsed-plasma thruster, Hall and ion thrusters, incorporating either Xenon or Iodine propellant, and an electrospray thruster. An analysis of the propulsion system trade-off was presented in an earlier paper [2]. The mission takes advantage of the ability of the SLS launch to place it into an initial trajectory of C3=0.

Due to mass constraints, high specific impulse is required, and volume constraints mean the propellant density was also of great importance to the ability to achieve the required ΔV. This improves the relative usefulness of the electrospray salt, with higher propellant density. In order to minimize high pressure tanks and volatiles, the salt electrospray and iodine ion propulsion systems were the optimum designs for the fly-by and rendezvous missions respectively, combined with a thruster gimbal and wheel system.

Fly-by mission

For the candidate fly-by mission, with a mission ΔV of about 400 m/s, the mission objectives could be accomplished with a 10-watt PUC (Propulsion Unit for CubeSats) electrospray propulsion system [4-6] at a specific impulse of about 800 seconds, incorporating a propellant-less cathode and a bellows salt tank. This propulsion system is planned for demonstration on the 2015 mission to LEO [4] and a 2016 mission to GEO [5]. Figure 3 shows the demonstration unit for the 2015 LISA mission, incorporating four electrospray thrusters. (The DAVID mission would require only a single one of these thrusters).

Table 1 shows the assumptions used in the trajectory simulation. The distance to the sun does not vary significantly during the trajectory, and hence the power output from the solar array does not vary significantly during the mission, and the thruster is operated at a constant 9 watts input power during the 272 days of thruster operation.

Table 1: Parameters for trajectory analysis, 2001 GP2 Fly-by mission

Colloid (Electrospray) Thruster Parameters:
Power to thruster = 9W Isp = 800s Efficiency = 31% Duty Cycle = 90%
Trajectory Assumptions:
Fly-by of 2001 GP2 SLS Launch Date: 12/17/2017 4 days, 10 m/s to correct for worst-case SLS injection Spacecraft wet mass = 12 kg
Trajectory Details:
ΔV = 378 m/s Required Propellant mass = 0.56 kg Time of Flight = 877 days Total Thrusting Time = 272 days

Figure 3: Electrospray thrusters for the Laser Interferometer Space Antenna (LISA) Pathfinder

mission disturbance-reduction system demonstration [5].

Figure 4 shows the distance from Earth as a function of time after launch for the 2017 launch, with the duration of the thruster firing shown. The main propulsion burn occurs approximately nine months after launch. When the asteroid fly-by occurs, the spacecraft is slightly more than 0.2 Astronomical Units from the Earth.

To check the sensitivity of the trajectory calculation to a launch delay, a second trajectory optimization was done assuming a launch one year later, in December 2018. The required ΔV is slightly higher, 447 m/sec, and the amount of propellant used is increased by 0.1 kg, to 0.67 kg total. The total thrust duration increases to 330 days, but the time of flight decreases to 503 days. The result shows that with a delay of one year the mission is still achievable within the margin.

Systems

Figure 5 shows the notional schematic of the spacecraft systems.

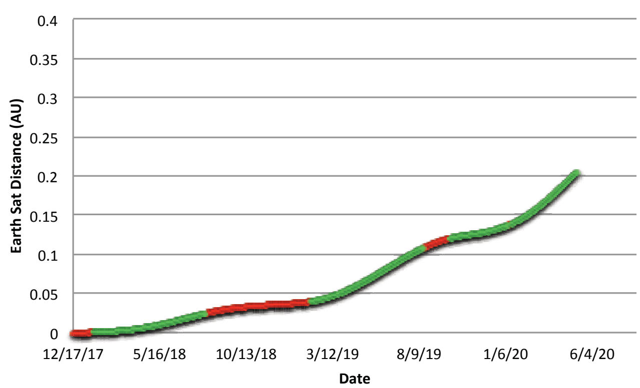

Figure 4: Trajectory of spacecraft for 2001 GP2 flyby, showing the Earth to spacecraft distance in AU as a function of time. The red portion of the curve is the time during which the thruster is firing.

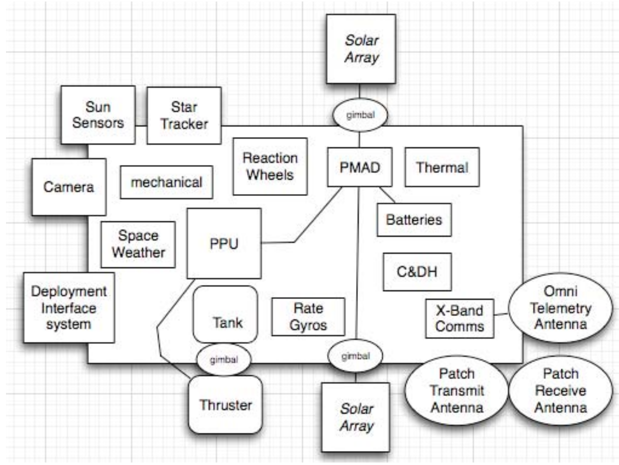

Figure 5: System schematic

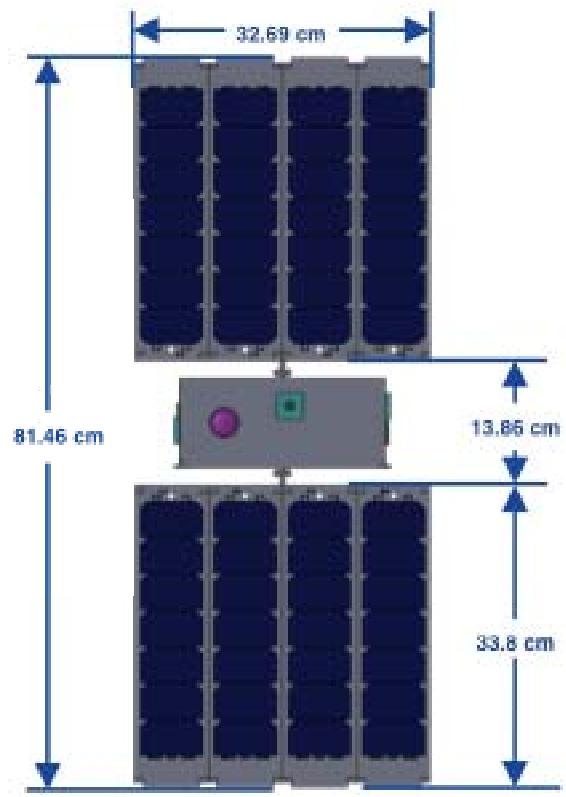

Figure 6: Dimensions of DAVID spacecraft.

Power: The mission requires 40 watts of solar array power. The array has two wings, with two folds to retract it into launch position against the side of the spacecraft. The solar arrays are mounted on a one-axis gimbal for sun tracking, with the second direction of sun tracking accomplished using spacecraft roll.

The array is based on HaWK array design [7], populated with triple-junction ZTJ solar cells, with efficiency of 29.5% at beginning of life (BOL) at 28°C)

. Each array consists of 7 series connected cells per panel, 4 panels/wing, 2 parallel strings/wing. When folded, the design is only two panels deep (two joined root panels, with one folded panel on each side). This array is scheduled to be will be flight demonstrated on the ORS 2 Satellite Demonstrator in November 2014. The arrays provide a combined total of 54 W BOL (at one 1 AU) at 16 V. At 1.027AU, end of life conditions, with power conversion losses, the power system provided 46W to the user. Note that based on the trajectory, some operation at distances slightly less than 1AU will provide about 3% more power; this contributes additional power margin, but was not assumed in the power analysis.

The array deployed area is 33.7 cm x 32.7 cm per wing. Figure 6 shows the dimensions of the deployed solar array for the fly-by spacecraft.

Attitude Determination and Control: COTS reaction wheels, using available rate-gyros, as well as a star-tracker and sun-sensors to determine attitude.

The thruster is mounted on a gimbal (+/- 5°), to account for thrust misalignment, center of gravity mis-alignment, and solar torques. The thruster can thus be used to de-saturate the momentum wheels, although desaturation task can also be achieved by use of solar torque produced by angling the solar arrays (during periods when maximum power is not required).

Command and Data Handling: COTS boards (100 krad to minimize SEU)

Communications: X-band transceiver board, ~50 bps to DSN. The spacecraft takes data at a high rate, and then downlinks the data over the course of several weeks following the pass. The spacecraft may also serve as a testbed for advanced communications systems; in particular, the possible demonstration of an integrated solar array/phased array patch antenna will be analyzed.

Thermal: The spacecraft will use passive thermal control. Waste heat is conducted through structure, Batteries insulated/heated, thruster isolated from bus

Environmental: The deep space radiation environment is more benign than Earth orbital environment in terms of total dose, but is subject to transient events due to solar and galactic cosmic ray impacts, which can cause Single Event Upset (SEU) events. 20 krad parts were shown to sufficient for radiation resistance in deep space for 1-3 year mission, but we need to assess the impact of SEUs on systems. Shielding is an option

Mechanical: Al frame structure, designed for 30g loading.

Rendezvous mission

The rendezvous mission was more difficult. This could be achieved with a 50-watt RF-ion propulsion system, running at a specific impulse of 2400 seconds. To achieve the required propellant in the tank volume available, iodine propellant [8] was chosen; iodine-fueled ion engines have been demonstrated on Earth but not flown in space, and have an estimated technology readiness level (TRL) of 3-4.

The propulsion was based on BRFIT-3 [9]. The mission required a ~150 watt solar array. This array, slightly more than three times the size of the array for the fly-by mission, is a significant driver of the mission performance. The analysis here simply assumed the existence of such an array, although design and development of such a high-power array that is able to fold to fit within the 6U form factor is challenging.

The trajectory assumptions are shown in Table 2.

Table 2: Parameters for trajectory analysis, 2001-GP2 Rendezvous mission

3cm RF Ion Thruster Parameters:
Power to thruster = 50W Isp = 2200s Efficiency = 27% Duty Cycle = 90%
Trajectory Assumptions:
Rendezvous of 2001 GP2 SLS Launch Date: 12/17/2017 4 days, 10 m/s to correct for worst-case SLS injection Spacecraft Wet mass = 12 kg
Trajectory Details:
ΔV = 2088 m/s Required Propellant mass = 1.11 kg Time of Flight = 611 days Total Thrusting Time = 456 days

CONCLUSIONS

The conceptual design shows that an asteroid mission is possible using a cubesat platform with high-efficiency electric propulsion.

Lessons learned:

- Targeting an asteroid that flies close to Earth minimizes the propulsion required for flyby/rendezvous
- The required ΔVs are still significant for a 6U cubesat: ~400 m/s for a fly-by mission, ~2000 m/s for a rendezvous, for the case analyzed, a 2020 mission to 2001 GP2.
    - The required mission total thrust duration exceeds the tested lifetime of the thruster, and the thruster would have to be requalified for the longer thrust time.
- Many asteroid close approaches to Earth exist, but not all have such low relative velocity to Earth of ~2km/s. However, some are in the 3-4 km/s range, so might be reachable

- The EM-1 launch places the cubesat on a trajectory with hyperbolic excess energy (C3) of 0.2 km^2/s^2 (using a 15 m/s burn and a lunar flyby)
  - If this is in the correct direction it can reduce the required ΔV
  - But in the wrong direction it can cost >500 m/s
  - The analysis here assumed the on-board EP system tweaks the 4-day trans-lunar trajectory sufficiently to eliminate the excess velocity (C3 down to 0)
- The fly-by mission appears to be achievable with an electrospray thruster, a propulsion technology that are available today. Although this technology has not yet been demonstrated in space, test missions will have demonstrated operation before the launch date of this mission
- The rendezvous mission is more difficult, and requires a development of iodine fuel for the small ion propulsion system, as well as development of a higher power solar array.
- Cubesat propulsion systems are inherently less efficient that their larger satellite counterparts due to their small size
  - ~ 30% propulsion efficiency for 10-100 Watt devices seem about the best efficiency on available devices (although work on a higher efficiency electrospray thrusters is underway)
- Propellant density is of great importance to achieve large ΔVs within the volume allocated
  - A Xenon tank is a challenge to fit into a 6U for 2000 m/s ΔV.
  - In order to minimize high pressure tanks and volatiles, the salt electrospray and the iodine ion propulsion systems were optimal, combined with a thruster gimbal and wheel system.

ACKNOWLEDGEMENTS

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