Miniaturizing Docking and Undocking through DockSat

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***Abstract*—This project involves a 2U and 1U CubeSat that are launched together. The CubeSats are initially attached using a novel docking mechanism. The CubeSats will separate after being released from the deployer followed by rendezvous and re-docking maneuvers with one another. Multiple undocking and re-docking demonstrations will be attempted, with increasing separation distance between the satellites at each iteration.**

Docking demonstrations will commence once the joined satellites have deployed from the CubeSat deployer, and both chaser and target have been fully commissioned. Both the 1U target and the 2U chaser will have a docking interface. An undocking and re-docking experiment will involve an initial satellite release by the docking adapters. Initial relative velocity will be imparted by a combination of spring force and electromagnets. An electric thruster on the 2U satellite will bring the satellites closer together, while visual-based pose estimation will provide feedback for the control system. The 1U satellite will maintain a stable attitude while the rendezvous and proximity operations are taking place. The final close approach and docking will be assisted by electromagnets built into the docking system on each satellite.

The projects primary goal is to demonstrate critical technologies for reconfigurable satellites and in-orbit servicing. The technologies that will be demonstrated include vision-based pose estimation and navigation, modular and dynamic reconfigurable spacecraft, and trajectory planning with electric thrusters. In addition, the project has the objective of establishing a sustainable CubeSat program at Stellenbosch University, through which post-graduate students can gain experience in satellite design and integration.

In this paper, we include the conceptual design of the mission including the definition of the major subsystems, mass budget as well as simulations of the undock and re-dock demonstration to show the mission feasibility. Three components required for the mission have been identified that require the most additional research and development. The docking mechanism is designed to be androgynous with servo-actuated latches and a vision system for multiple separations and docking procedures. Electromagnets are also added to the mechanism and the behavior is modeled for initial separation and final close-proximity control. Additionally, a practical statistical model of the proposed electric thruster is constructed and used in simulation to obtain an expectation of the chasers trajectory tracking performance.

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1. INTRODUCTION

The ability for miniature nanosatellites to dock has many applications. These may include missions which require reconfigurable spacecrafts, in-orbit servicing and deorbiting of satellites at end-of-life. The current trend in satellites is towards smaller size and larger number of satellites, as is evident from current and future planned constellations. Reducing the size requirements of a docking ability to a nanosatellite form factor will unlock the various docking applications for nanosatellites constellations.

There are several challenges that need to be addressed to miniaturize the required components and functionality. These include creating a compact mechanical interface and simplifying the pose estimation and close-proximity control so that it can be executed on a low-power processing platform. The class of thrusters that are available to nanosatellites also have a significant implication. The CubeSat and nanosatellite form factor does not allow for large pressure vessels and CubeSat launch constraints usually do not permit pressurized vessels. Electric thrusters are available for nanosatellites, but these have low thrust specifications which poses a control challenge to rendezvous and proximity operations. Additionally, the accurate control of a nanosatellite's attitude while using propulsion is a great challenge.

In this paper we introduce the *DockSat* mission which aims to demonstrate the ability to dock and undock using nanosatellites. The primary goal of the project is to demonstrate key technologies for reconfigurable satellites and in-orbit servicing. In addition, the project has the objective of establishing a sustainable CubeSat program at Stellenbosch University, through which post-graduate students can gain experience in satellite design and integration. This paper will focus on (1) the high-level mission and satellite system design, (2) the docking interface including mechanical, pose estimation and electromagnet designs, and (3) a statistical model of the electric thruster with initial simulation of trajectory tracking performance.

Related Work

The docking of satellites is a common occurrence in larger spacecrafts and is used extensively in resupplying the International Space Station (ISS). These docking mechanisms are exceptionally large and have a number of additional requirements that enable humans to move from one spacecraft to the other.

There have been some attempts made to enable smaller vehicles to dock. Most notably is the CubeSat Proximity Operations Demonstration (CPOD) mission [1]. It entails two 3U CubeSats that make use of cold-gas thrusters for maneuvering. The relative navigation is done with several sensors that include differential GPS (Global Position System), inter-satellite link, fiducial LED's and two infrared and visible imagers. The docking mechanism is based on mechanical latches.

The Autonomous Assembly of a Reconfigurable Space Telescope (AAREST) mission is a good example of a specific application that is enabled by the docking of smaller vehicles [2]. It entails multiple small mirror-satellites to dock and create a single large spacecraft. Each mirror-craft increases the effective aperture of the telescope. It consists of smaller 3U CubeSat sized mirror-satellites docking to a larger main satellite. It uses micro-resistojet thrusters for orbit control and the docking mechanism consists of matching protruding conic adapters with permanent magnets. Electromagnets are then used to separate the spacecraft. Relative position feedback is envisioned using a combination LIDAR/Camera sensor. A similar concept mission is proposed by [3] as part of the Rendezvous and docking Autonomous CubeSats Experiment (RACE) mission.

Another demonstrator by [4] performed practical tests of small satellite docking on a 2D air-bearing table. The experiment was focused on the satellite docking mechanism and control. It used electromagnets for close-proximity tests and an internal reaction wheel for alignment. The docking mechanism consists of gripping fingers for final mechanical attachment. Their practical experiments showed that accurate control can be achieved with electromagnets, although these tests were not performed in a flight configuration.

Beyond these few projects and missions there are a number of other docking mechanisms that have been proposed for small satellites [5], [6], [7]. Many of these distribute some designs with important performance parameters such as alignment requirements for the mating mechanisms, the required approach speed, suitability to multiple docking and undocking maneuvers and the fact whether the mechanism is androgynous.

The *DockSat* mission aims to demonstrate multiple docking and undocking of nanosatellites and thus continue the miniaturization of the systems required for rendezvous and docking missions. In this paper we introduce the overall system design of the *DockSat* mission in Section 2. Section 3 focuses on the current design of the docking mechanism which contains all the required functionality for multiple docking and undocking maneuvers. The practical expectation of the rendezvous is analyzed within simulation in Section 4. Finally, some conclusions and future work are defined in Section 5.

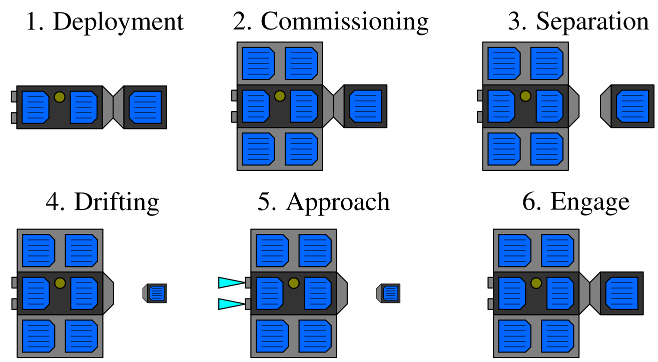

Figure 1. The missions concept diagram of *DockSat*, indicating the all the major functions and mission phases.

2. SYSTEM DESIGN

The main objective of the *DockSat* mission is to demonstrate docking and undocking of nanosatellites. This will enable many future satellite missions, as described in Section 1. The satellite mission will need two separate satellites: the *Target* and the *Chaser*. These two satellites are mechanically fixed with a docking mechanism and launched together as a single spacecraft. After the combined spacecraft has been commissioned, the satellites are undocked, and the two satellites will drift away from each other. At a certain predefined time, when the two spacecrafts are a certain distance away from each other, the docking maneuver will be initialized. The *Chaser* will make use of an optical system for pose and relative position estimation, and electric thruster to maneuver itself closer to the target to finally reattached again. This undocking and docking cycle may continue, increasing the initial relative distance before the re-docking phase is reinitialized. Figure 1 shows the mission concept diagram.

Unlike some of the other and previous rendezvous and docking missions, *DockSat* will:

- attempt to optimize the docking sub-system as much as possible, by reducing volume and implementing pose estimation on a low-power platform.
- attempt to perform the rendezvous with an electric thruster which has the potential for smaller volume requirements than conventional approaches.

In order for the spacecraft to fulfill these functions and requirements, the system design can be broken into two parts, i.e. the two satellite segments.

Chaser Satellite

The *Chaser* is a 2U CubeSat that should be able to find the *Target* and move towards it until docking can occur. It should be a complete satellite that contains all the required subsystems. The subsystems with their key specifications are defined below as:

- Electrical Power System (EPS):
  - $20\,\mathrm{Wh}$, supplying $5\,\mathrm{W}$ continuously to thruster
  - Switched power channels for all other sub-systems
  - $2\times$ deployable and $2\times$ fixed 2U solar panels
- Attitude Determination and Control System:

- Full 3-axis attitude control
  - 3-axis reaction wheels and magnetorquers
  - Sun sensor, 3-axis magnetometer, 3-axis rate sensors
- Ground-to-satellite Communications:
  - VHF/UHF up and downlink
  - 9600 baud
  - Telemetry and control
  - File downloads (log files and camera images/video)
- Thruster:
  - Electric, plasma thruster
  - $50\,\mu\text{N}$ thrust (for $5\,\text{W}$ input power)
- Inter-satellite Communications:
  - Telemetry and commands between *Chaser* and *Target*
  - Low power, short-range wireless communication
- Flight Computer (OBC):
  - Interface with all sub-systems on the satellite
  - Process camera images and closed-loop navigation
  - Log telemetry
  - Store camera frames and videos for download
- Camera:
  - Pose estimation for docking approach control system
  - Record images and video of docking demonstration
  - $70.8°$ Horizontal Field-of-View, $55.6°$ Vertical Field-of-View
  - 640 x 480 pixel resolution
- Docking Interface:
  - Androgenous docking mechanism
  - Servo for locking
  - Electromagnets for close-proximity and initial repulsion

The satellite design will mainly focus around the requirements of the electric propulsion. The electric thruster that is considered is the plasma thruster from Hypernova Space Technologies[2]. It is a vacuum arc type thruster with a nominal thrust level of $50\,\mu\text{N}$ at $5\,\text{W}$. Additional specification are listed in Table 1.

Table 1. High-level specifications of Hypernova's NanoThruster A

Item	Specification
Thrust	$5 - 10\,\mu\text{N/W}$
	Nominal thrust $50\,\mu\text{N}$ ($5\,\text{W}$)
Power	$5\,\text{W}$ nominal
Specific impulse	$500\,\text{s}$
Volume	0.35 U
Mass	$600\,\text{g}$
Fuel mass	$3\,\text{g}$

The electric thruster will need to be pointed in the correct direction, thus a complete 3-axis stabilized Attitude Determination and Control System (ADCS) will be required. It will need reaction wheels in all three axes for fine control of the thrust vector and absorbing any disturbance torques created by the electric thruster. The exact requirements of the ADCS performance is mainly determined by the effect of incorrect pointed thrust in the approach and the misalignment error that the final docking mechanism can withstand.

The Electrical Power System (EPS) will need to supply enough power for the $5\,\text{W}$ electric thruster to operate. This would require attention to the battery capacity and power generation capabilities of the *Chaser*. Two deployable solar panels are proposed to ensure adequate power generation.

An initial mechanical layout of the spacecraft with all the required subsystems can be seen in Figure 2, along with the initial mass budget in Table 2.

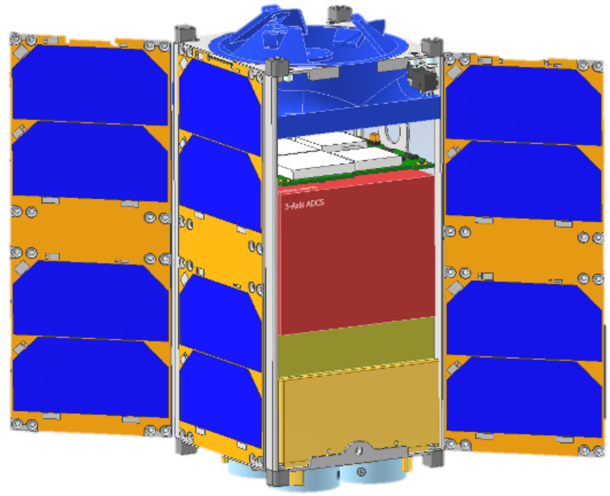

Figure 2. Mechanical layout of *Chaser* spacecraft.

Table 2. Initial mass budget for the *Chaser* spacecraft.

Component	Mass	Margin
2U Chassis	$216\,\text{g}$	$5\,\%$
EPS + Battery	$273\,\text{g}$	$5\,\%$
Comms + Antenna	$184\,\text{g}$	$5\,\%$
ADCS	$554\,\text{g}$	$5\,\%$
Thrusters	$600\,\text{g}$	$20\,\%$
Docking Subsystem	$300\,\text{g}$	$20\,\%$
OBC	$60\,\text{g}$	$5\,\%$
Solar Panels	$330\,\text{g}$	$10\,\%$
Subtotal	$2517\,\text{g}$	

Target Satellite

The *Target* is a 1U CubeSat form factor that should be a complete operational satellite. All subsystems are required, including communication, power and attitude control, although an important distinction is that the *Target* will communicate to the *Chaser* satellite via an inter-satellite link, and not directly to the ground station. The subsystems with their key specifications are defined below as:

- Electrical Power System (EPS):
  - $10\,\text{Wh}$ with 1U body-mounted solar panels only
- Attitude Determination and Control System:
  - 3-axis attitude stabilisation
  - Single momentum wheel and three axis magnetorquers
  - 3-axis magnetometer, 3-axis rate sensors
- Inter-satellite Communications:
  - Telemetry and commands between *Chaser* and *Target*
  - Low power, short-range wireless communication
- LED markers:
  - High-power LEDs, arranged in specific pattern
  - Aids *Chaser* satellite to find range and pose of *Target*
- Docking Interface:
  - Androgynous docking mechanism
  - Servo for locking
  - Electromagnets for close approach and initial repulsion

The capabilities of *Target* can be reduced, when compared to the *Chaser*. Ideally the attitude of the *Target* spacecraft

[2]more information at `https://www.hypernovaspace.com`

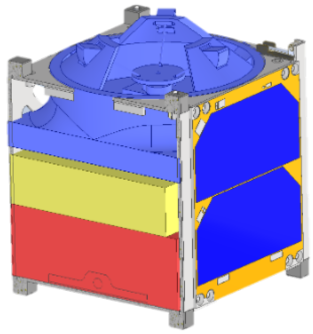

Figure 3. Mechanical layout of *Target* spacecraft.

should be actively controlled and also communicated to the *Chaser* spacecraft. This will greatly simplify the approach and docking phases of the mission. This does however, necessitate a compact and complete ADCS, which should include a momentum wheel to bias the spacecraft's angular momentum and thus effect a stabilized satellite. To simplify the communication system on the smaller spacecraft only an inter-satellite link is proposed, which would reduce the over all volume and power requirements of this system. This however does place a risk of losing the *Target* if the distance between the spacecrafts would increase beyond the design threshold.

An initial mechanical layout of the spacecraft with all the required subsystems can be seen in Figure 3, along with the initial mass budget in Table 3. The high-level system architecture of the complete system is shown in Figure 4.

Table 3. Initial mass budget for the *Target* spacecraft.

Component	Mass	Margin
1U Chassis	113 g	5 %
EPS + Battery	210 g	5 %
Comms + Antenna	70 g	5 %
ADCS	345 g	5 %
Docking Subsystem	180 g	20 %
OBC	60 g	5 %
Solar Panels	220 g	10 %
Subtotal	1198 g	

3. DOCKING MECHANISM

When the spacecrafts are in close proximity, they should be able to maneuver towards one another in order to dock and later undock. To allow for the *Chaser* and *Target* spacecraft to dock and undock, a suitable adapter design is needed. This section deals with the design of this adapter and the integration of its different parts.

Specifications

The design of the docking mechanism was approached with the mission goals of *DockSat* in mind. The most important criteria that were identified as being mission critical, are the ability to dock and undock multiple times, the need for a suitable optical pose estimation system to be integrated with the mechanism, and the initial contact conditions as described in Table 4.

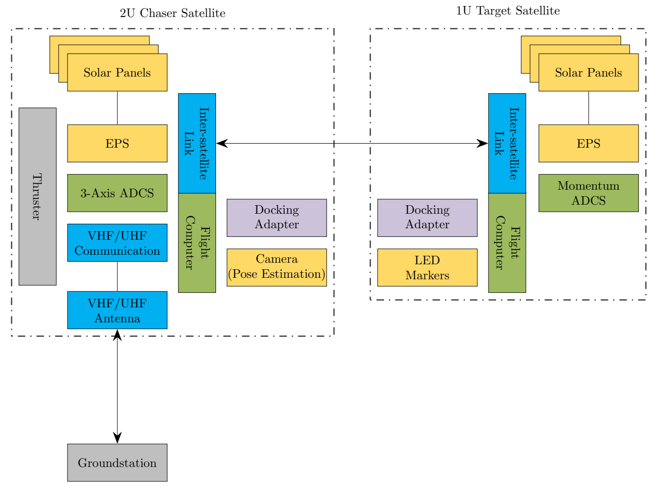

Figure 4. The high-level system architecture of the *Chaser* and *Target* spacecrafts.

Table 4. Identified specifications for the docking mechanism.

Initial Contact Condition	Limiting Value
Lateral Misalignment	15 mm
Pitch/Yaw Misalignment	5°
Roll Misalignment	5°
Pitch/Yaw rate	12 °/min
Roll rate	12 °/min

Adapters Base Design

As a starting point, the International Docking System Standard (IDSS) [8] mechanism was adapted and miniaturized, as to fit on a $10\,\mathrm{cm}^2$ square. This mechanism design consists of two circular androgynous adapters with 3 inward facing guide petals on each. These petals guide the two mechanisms together, accounting for misalignment. On each of the petals is a latch that gets pressed down by a latching petal on the interior of the docking adapter. These latches spring back out behind the latching petals, when the adapters are fully inserted. This holds the adapters together.

The base of the adapter was designed first. The outer diameter was chosen as 100 mm. This is so that the mechanism fits on the face of a 1U satellite. The guide petals on the mechanism were designed with an angle of 40°. This is less steep than the 45° as set out in the IDSS. This saves some volume, while still being steep enough to conform to our specifications. To ensure that the system worked within our specified lateral misalignment, the guiding petals had to protrude into the center by at least 15 mm. To do this, a height of 15 mm for the petals was chosen. This leads to a protrusion of approximately 17.876 mm. The guide petals were designed with an angle of 60° at their base and 36° at their ends. The difference in these angles is enough to allow for a roll misalignment of up to 12°. The adapter's shape can be seen on the top of Figures 2 and 3.

Latch Mechanism Design

To facilitate docking, the latches should be able to spring back after being pressed down. They should also have the ability to pull the latches back and lock them in place when docked by the action of a motor. This was achieved by attaching the

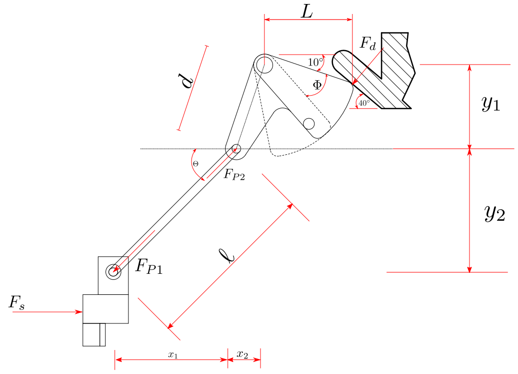

Figure 5. Cross section of the latch mechanism within the docking adapter.

latches via a freely rotating pin to a slider moving linearly. This slider can then be made to move with either a rotating spindle or springs. A cross section of this mechanism can be seen in Figure 5. The force F_S indicates the force imposed by the spring on the slider. Figure 6 shows the front view of the prototype *Chaser* satellite docking adapter, including the mechanism. The spindle design is shown in Figure 7. This spindle allows the slider to slide freely while docking is taking place. The spindle can move the sliders to pull the latches back when undocking, or it can lock the sliders in place while docked.

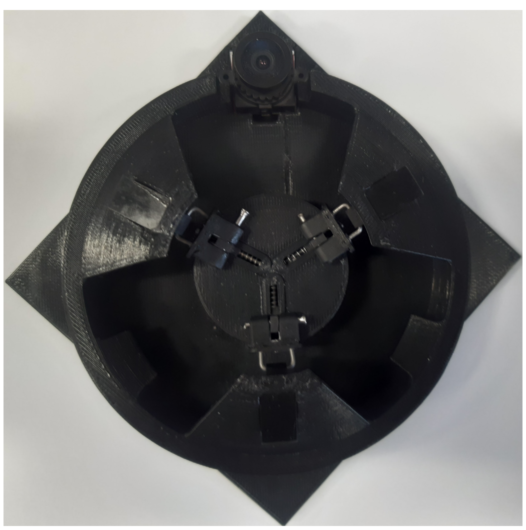

Figure 6. The front view of the 3D printed *Chaser* satellite adapter prototype.

Modeling

Initial modeling was done of the contact forces and dynamics of satellites and the internal mechanism of the docking adapter. An analysis of these results indicate that a relative velocity of approximately $0.044\,\mathrm{m/s}$ at contact is enough to overcome the forces of the springs. Figure 8 shows the total force exerted by the *Target* spacecraft on the *Chaser* spacecraft over time, when contact is made with a relative velocity of $0.05\,\mathrm{m/s}$. At the final point, the latches spring into place and no more forces are exerted on the *Chaser*. When the contact velocity is decreased to below $0.044\,\mathrm{m/s}$, the graph is symmetrical, since the same forces get applied when the *Chaser* bounces off as when it impacted.

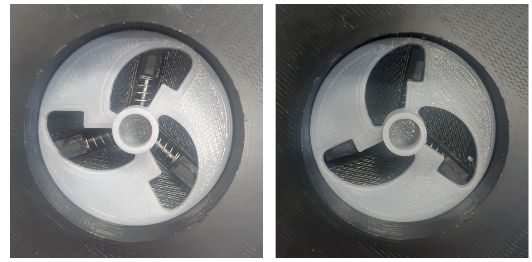

Figure 7. Bottom views of *Chaser* satellite prototype showing both unlocked (left) and locked (right) positions of the spindle.

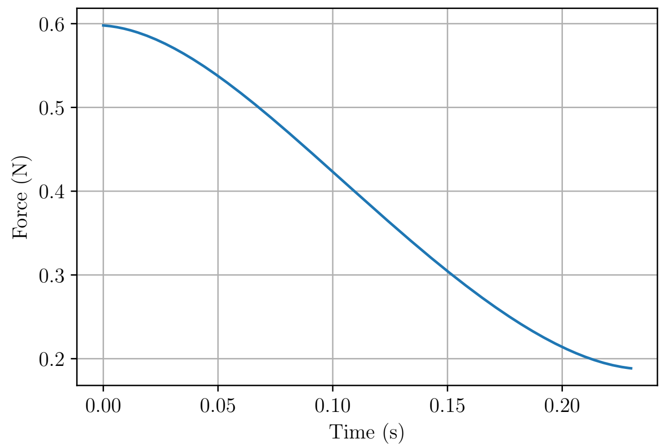

Figure 8. Force between the satellites with a relative contact velocity of $0.05\,\mathrm{m/s}$.

Pose Estimation

The visual pose estimation system was based on work done by [9] and [10]. The *Chaser* spacecraft is equipped with a camera, while the *Target* spacecraft has 5 LEDs arranged on the docking adapter to act as fiducial markers. 4 of these LEDs are put on the same plane, while one is placed out-of-plane on one of the guide petals. This allows for an analytical solution. Setting the colors of some of the LEDs to a unique color, allows us to identify which marker is being processed more easily. Three different colored LEDs are used, red for 3 of the in-plane LEDs, blue for one of the in-plane LEDs and green for the out-of-plane LED. Using an OV7725 sensor at 640x480 resolution and a lens with an Horizontal FOV of 70.8° and a Vertical FOV of 55.6° allows us to estimate the position with a maximum theoretical accuracy of approximately $2.2\,\mathrm{mm}$ at a distance $1\,\mathrm{m}$ away from the lens. Figure 9 shows a picture taken with the camera on the *Chaser* satellite's camera. Circles indicate the identified blobs.

Magnetic Control

In order to allow for accurate control of the satellites at close ranges, electromagnets are embedded into the docking adapter. This approach allows for generating forces at close distance. An electromagnet capable of accelerating the two satellites fast enough to dock, that is to say a relative velocity of $0.044\,\mathrm{m/s}$, from a distance of $1\,\mathrm{m}$ was set as an aim. The coil was designed to be wound around the adapter base and make use of an air core. The coil was designed with a rectangular cross-section of $2.5\,\mathrm{mm} \times 7\,\mathrm{mm}$ with the outer diameter being $100\,\mathrm{mm}$, as seen in Figure 10.

Figure 9. LED fiducial markers as seen and identified from the *Chaser* camera.

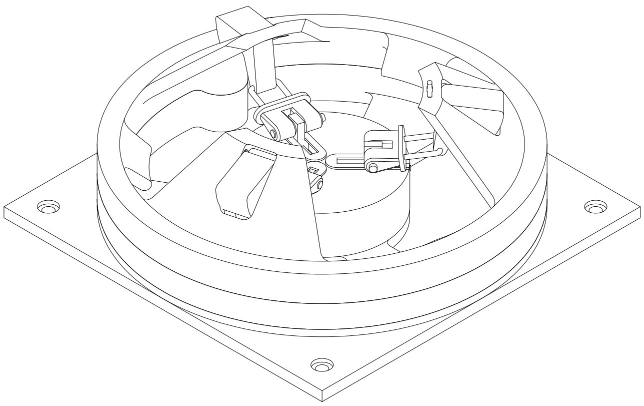

Figure 10. Isometric CAD drawing of the androgynous docking adapter with a coil wound around its body.

With a wire diameter of $0.254\,\mathrm{mm}$ (30 AWG) and assuming each wire fills a $0.54\,\mathrm{mm} \times 0.54\,\mathrm{mm}$ square, it should be possible to fit 271 windings on the groove in the body. To account for tolerances in the wire and non ideal winding, it was decided that the coil should comprise of 262 windings. Through simulation using the *Altair Flux* software, it was found that these coils could accelerate the satellites' relative velocity up to $0.14\,\mathrm{m/s}$ starting from rest $1\,\mathrm{m}$ apart if $860\,\mathrm{mA}$ current is used. Figure 11 shows the magnetic fields of the coils as rendered in *Altair Flux*.

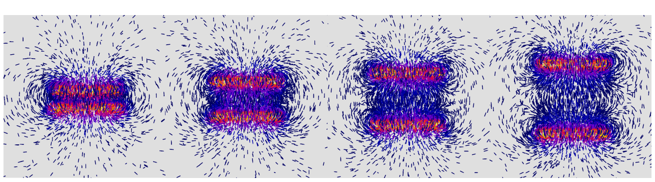

Figure 11. Magnetic fields with the centers of the coils (from left to right) $50\,\mathrm{mm}$, $100\,\mathrm{mm}$, $150\,\mathrm{mm}$ and $200\,\mathrm{mm}$ apart.

4. Rendezvous Analysis

As described in Section 2 the satellites will first separate and drift away from each other. Their initial relative velocities should be minimal. Therefore docking should be achievable through simple rendezvous maneuvers. This section will focus on the simulation analysis of the rendezvous.

Rendezvous Synopsis

The spacecrafts are initially docked. By inducing a repulsion force through the electromagnets, the satellites will undock and start separating from one another. After reaching the desired relative distance the electric thruster on the *Chaser* satellite will activate. The thruster will produce a force on the *Chaser* which is tangential to the direction of motion. The thruster will stay active until the chaser has reached a relative distance of $60\,\mathrm{cm}$ and a relative velocity of $7\,\mathrm{cm/s}$ with respect to the target satellite. At this point a visual feedback controller will gain priority and ensure a successful docking. This visual controller may use the electromagnets within the docking mechanism and feedback from the optical markers. Undocking and docking will be attempted multiple times each with an increase in relative distance between the *Chaser* and *Target* satellite. This is a simple rendezvous maneuver and its results and feasibility will be investigated.

To simplify this analysis, the ADCS will be assumed to have infinite bandwidth. This means all changes and corrections in attitude are accurately measured and any changes in attitude occur instantaneously.

Thruster Fluctuation

The electric thruster has multiple sources of uncertainty within the produced thrust vector that may result in the *Chaser* satellite to veer off of the originally planned trajectory. These irregularities include a variation in thrust magnitude, angle of thrust and point of thrust. The variation in thrust magnitude should not have a significant effect because the maneuver is not dependent on a precise single ∇V change at a single point but rather on a continuous reduction in the relative distance and velocity over time due to the low thrust levels. The angle offset may cause the *Chaser* to unnecessarily change its orbital plane and attitude. This will require more energy being spent on the reaction wheels to re-orientate the *Chaser*. The offset in the generated point of thrust is by far the most influential. This causes the thrust to be offset from the spacecraft's center of mass which may cause significant torques on the *Chaser*. This is prevented by storing angular momentum within the reaction wheels to prevent the thrust torque influencing the *Chaser*. For the initial analyses of the rendezvous the torque exerted by the offset in point of thrust is assumed to be fully absorbed or negated by the ADCS.

Within the simulation the *Chaser* satellite will thrust in the same direction as its velocity vector. However, noise is added to the thrust magnitude by using a Gaussian distribution with a mean of $50\,\mu\mathrm{N}$ and 3σ set to 10% ($5\,\mu\mathrm{N}$) of the thrust. The offset in the angle of thrust consists of two components namely θ and ϕ as illustrated in Figure 12. Both are defined through a Gaussian distribution with a mean of 0° and a 3σ value of 3°. The offset in the generated point of thrust consists of r_y and r_z. Each is determined by a Gaussian distribution with a mean of 0 and a 3σ value of $1\,\mathrm{mm}$. The specification of this probabilistic thruster model is shown in Table 5.

Orbital Perturbations

The duration of the rendezvous operation will span a significant part of an orbit (or even multiple orbits when the separation distance is increased). As a result, the simulation includes orbital motion of the satellites. The position and velocity of the satellites are modeled by taking the following perturbing effects into account (in order of importance): the effect of the non-spherical earth, gravitational forces of the

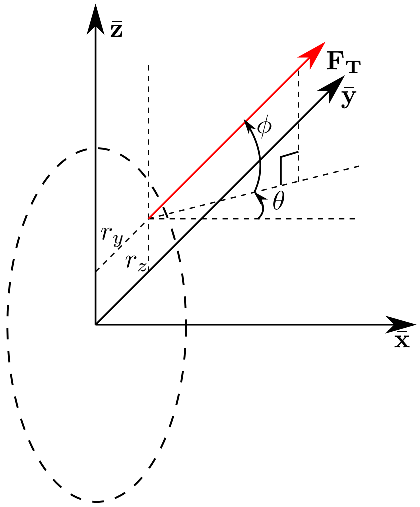

Figure 12. Definition of uncertainty parameters that determines the origin and direction of the required thrust force, $\mathbf{F_T}$.

Table 5. The specification of all the parameters within the probabilistic thruster model.

Thrust Irregularity	μ	3σ
F_T	50 µN	5 µN
θ	0	3°
ϕ	0	3°
r_y	0	1 mm
r_z	0	1 mm

sun and moon, and aerodynamic drag. The first two are modeled by applying the Joint Earth Gravity Model, namely JGM-3 [11].

Rendezvous Simulation

The satellites are initially docked and then separated by means of the electromagnets. This causes them to slowly drift away from each other. At a certain point, the thruster on the *Chaser* satellite will activate and thrust in the direction opposite to the direction of motion. Four scenarios are simulated. The first is where the two satellites are separated, and the electric thruster is not activated to highlight the natural motion of the spacecrafts. In the other three scenarios the thruster is constantly active after the satellites have reached a minimum relative distance of $2\,\mathrm{m}$, $4\,\mathrm{m}$ and $5\,\mathrm{m}$. The relative movement of the satellites are shown in Figure 13.

It is clear that the induced force from the electromagnet pushes the satellites away from each other. In the ideal case the satellites would at some point start moving closer to each other due to orbital dynamics. However, because of the difference in orbital perturbations and aerodynamic drag the satellites continue to slowly drift away from each other. Activating the electric thruster does momentarily reduce the drift rate but the satellites never get close enough for the visual feedback controller to gain priority. In Figure 13 the chaser and target satellites are separated through the electromagnet. The force of the electromagnet was directed such that the target satellite would be propelled in the direction of motion while the *Chaser* satellite would be propelled in the opposite direction. It is clear that this method of rendezvous does not work, and an alternative strategy is required.

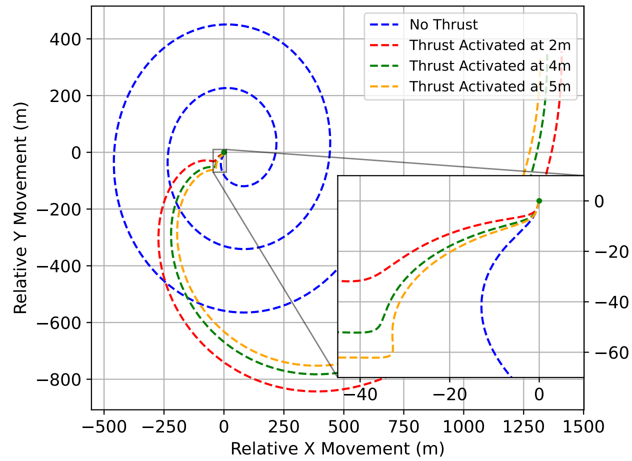

Figure 13. Simulated relative movement of satellites in each scenario.

Alternate Rendezvous

By changing the attitude of the docked satellites, the direction in which the initial separation force is applied can be manipulated. In this scenario the *Target* satellite is propelled at a 45° angle relative to the direction of motion while the *Chaser* satellite is propelled in the opposite direction. Scenarios like Figure 13 were simulated and are depicted in Figure 14.

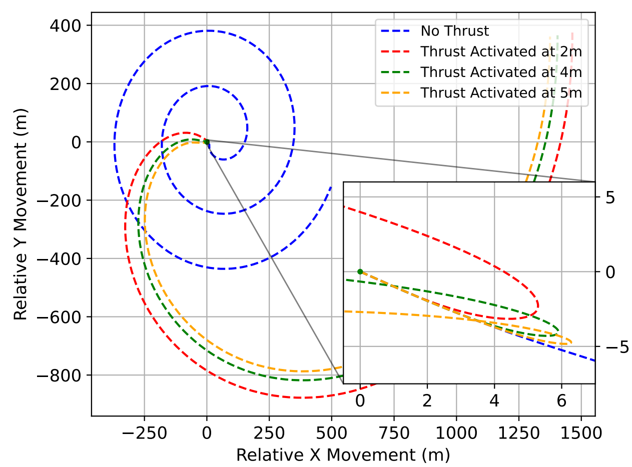

Figure 14. Simulated relative movement of satellites in alternative rendezvous method.

At first the satellites drifted away from each other. However, after the thruster is activated the relative distance between the satellites slowly starts to decrease. The relative distance never returns to zero meaning the *Chaser* satellite only does a slow fly-by past the target satellite. Each simulated scenario ($2\,\mathrm{m}$, $4\,\mathrm{m}$, $5\,\mathrm{m}$) has a different minimum relative distance for the fly-by namely $3.1\,\mathrm{m}$, $0.6\,\mathrm{m}$ and $2.6\,\mathrm{m}$ respectively. This shows that from the simulated scenarios only when the thruster starts firing at a relative distance of $4\,\mathrm{m}$ does the chaser satellite get within the initially specified minimum distance of $60\,\mathrm{cm}$ when the release angle is 45°.

Figure 13 and Figure 14 illustrate the ideal relative movement of the two satellites in their orbital plane (XY-plane). This is due to the forces which are normal to the orbital plane being much smaller than those within the orbital plane. The residual effect of the orbital perturbations and aerodynamic drag are predominantly visible in the orbital plane.

After implementing the electric thruster irregularities into the scenario where the thruster is activated at a relative distance of $4\,\mathrm{m}$ a Monte Carlo analysis is performed. Figure 15 depicts the nearest relative positions of the simulated trajectories. The mean distance is $0.61\,\mathrm{m}$ with a distribution due to the uncertainties applied in the thruster model. Thrusters that generates larger forces and more discrete behavior such as conventional cold gas thrusters may result in even larger variation in end conditions. The long duration of electric thruster activation has an average effect reducing the final uncertainty.

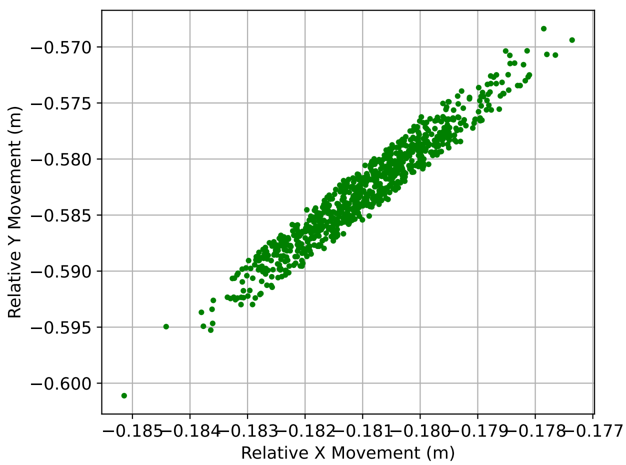

Figure 15. Monte Carlo simulation of nearest relative positions for the $4\,\mathrm{m}$ docking scenario.

The Figures 13, 14 and 15 show that these simple rendezvous maneuvers are not reliable, even at small separation distances. Various methods exist which can improve the rendezvous and ensure its success. Numerical optimization and planning techniques can be used to determine the required thrust and pointing angle for the *Chaser* spacecraft's trajectory to intersect that of the *Target*. It is also clear from the rendezvous analysis that the planned trajectory needs to be updated throughout its execution to absorb the uncertainties due to the orbital effects and effective thrust vector. Future work would entail the development of such a model-predictive control implementation.

5. Conclusion

In this paper the *DockSat* mission was introduced. The mission aims to demonstrate multiple undocking and docking maneuvers using nanosatellites. Enabling nanosatellites to perform rendezvous and docking missions will enable a number of mission applications such as in-space assembly. The system design of the 2U, *Chaser* and 1U, *Target* spacecraft were defined. The mission entails an initial separation whereby the spacecrafts will drift, and their relative distance will increase. The *Chaser* will then use its electric thruster to approach and finally dock. The docking mechanism design was introduced along with the integrated pose estimation system and electromagnetic actuation system. Prototypes of this mechanism have been constructed. The performance of the electric thruster is analyzed, and a practical expectation is obtained. The analyses show that a simple rendezvous solution with the electric thruster is not effective against natural orbital motion, orbital perturbations and thrust vector uncertainties. More extensive simulation and trajectory planning is required to improve the performance of the rendezvous. Development of the mission and its spacecrafts will continue with additional practical testing of the docking mechanism, integration of the final docking mechanism into the spacecrafts and the initial development of flight software in an engineering model.

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