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UNIVERSITY

School of Engineering and Digital Sciences

Bachelor of Engineering in
Mechanical and Aerospace Engineering

**Design of a Prototype of a Space Tug with an
Aerodynamic Brake for Space Debris Removal
(Final Capstone Project Report)**

by

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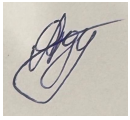
Declaration

We, Assylzat Ashirbayeva, Balzhan Abdikhanova, Altynbek Sadu, Abylai Ashken, hereby declare that this report, entitled “Design of a Prototype of a Space Tug with an Aerodynamic Brake for Space Debris Removal” is the result of our own project work except for quotations and citations which have been duly acknowledged. We also declare that it has not been previously or concurrently submitted for any other degree at Nazarbayev University or elsewhere.

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Abstract

LEO's growing accumulation of Space Debris threatens satellites and future missions. Conventional propulsion-based deorbiting methods are generally impractical for CubeSats due to size, weight, and energy considerations; therefore, alternative passive means of removing debris need to be investigated. This project aimed to create a CubeSat-size space tug with a novel Origami-based aerodynamic drag system for tether-assisted debris removal. The purpose was to demonstrate a compact, lightweight, mechanically feasible design that enhances drag and attitude stabilization during the demonstration of a mock debris towing experiment.

The methodology incorporated mathematical modelling, rarefied flow aerodynamic analysis, control system design, and prototype manufacturing. The use of mathematical modelling to establish the orbital and attitude dynamics of the tug and the orbital debris, coupled with free molecular aerodynamic forces used to establish drag coefficients and moment coefficients of the tug, allowed for predictions concerning tug-debris interaction, and guided the assessments of stability, deorbiting performance, and the creation of a preliminary control strategy for controlling attitude and maintaining tension on the tether.

The prototype was manufactured using Computer-Aided Design (CAD), 3D Printed Components, and a Tendon Driven Panel Deployment Mechanism with magnetic locks, and the test were successful in demonstrating the successful deployment of panels, Origami sail deployment, and locking of the sails, and tethered mock debris separation, which confirmed the mechanical design. Results of testing showed a significant increase in drag area while remaining compact and suitable for nano satellites. In addition, the prototype serves as a foundation for future Tether Based Orbital Debris Removal demonstration.

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1. Abstract

LEO's growing accumulation of Space Debris threatens satellites and future missions. Conventional propulsion-based deorbiting methods are generally impractical for CubeSats due to size, weight, and energy considerations; therefore, alternative passive means of removing debris need to be investigated. This project aimed to create a CubeSat-size space tug with a novel Origami-based aerodynamic drag system for tether-assisted debris removal. The purpose was to demonstrate a compact, lightweight, mechanically feasible design that enhances drag and attitude stabilization during the demonstration of a mock debris towing experiment.

The methodology incorporated mathematical modelling, rarefied flow aerodynamic analysis, control system design, and prototype manufacturing. The use of mathematical modelling to establish the orbital and attitude dynamics of the tug and the orbital debris, coupled with free molecular aerodynamic forces used to establish drag coefficients and moment coefficients of the tug, allowed for predictions concerning tug-debris interaction, and guided the assessments of stability, deorbiting performance, and the creation of a preliminary control strategy for controlling attitude and maintaining tension on the tether.

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2. Introduction

2.1 Space Debris Problem

The Earth's orbit is becoming more and more littered with debris, discarded rocket parts, and satellites at their end of life. According to the ESA Space Environment 2025 [5], even currently working satellites risk collision with floating debris due to how crowded some orbits have become over the years. Recorded total object mass and quantity for this year are approximately 6700 tons and 23000 units in Low Earth Orbit (LEO) and 13000 tons and 39000 units in Geocentric Earth Orbit (GEO) [5], with an increasing upward trend. Moreover, debris fragmentation increases the number of debris objects enough to potentially cause a chain reaction of collisions to occur, known as the "Kessler Syndrome." There are 2633 cataloged objects as the result of fragmentation of the rocket bodies and payload launched into LEO in 2024, suggesting that the number of objects grows even when no new satellites are launched into space. The increasing amount of debris in the orbits risks damaging satellites and rockets for future missions, rendering those orbits unusable and requiring a more active debris removal process. A variety of debris removal methods have been developed, each with different advantages and challenges, but they have to be chosen primarily based on feasibility of implementation, safety of operation, and potential of reduced fragmentation.

Along with designing reliable debris removal methods, however, one of the key challenges is designing propulsion systems for satellites to ensure long operation time and stability. Propulsion systems that require little to no fuel are of particular interest since satellites at their end of life become part of the rubble floating in the orbit. Moreover, fuel-based propulsion systems increase weight due to stored fuel, which is a potential challenge for small satellites, which have weight constraints to remain operational.

This report provides a basic overview of different debris removal and propulsion methods, then focuses on the chosen methods and required technology for the satellite prototype design. Then, this report will cover the main aim and objectives of the project, the description of the deployment stages of the space debris removal experiment, methodology, discussion of the results achieved until this point, and future work.

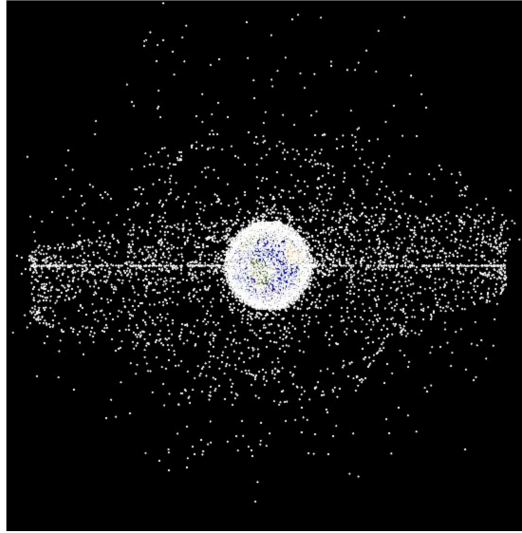


Figure 2.1: Artist's impression of space debris in Low Earth Orbit (LEO) and Geocentric Earth Orbit (GEO) [1]

2.2 Literature Review

2.2.1 Active Space Debris Removal Methods

In order to tackle with the issue of space junk, various Active Debris Removal (ADR) technologies has been developed. These techniques are mainly categorized into contact-based, contactless methods and self-eating satellites [1], as depicted in Figure 2.2 .Each of them has their own advantages, operational challenges, and applicability depending on the size of the debris, orbital attitude, and complexity of the mission.

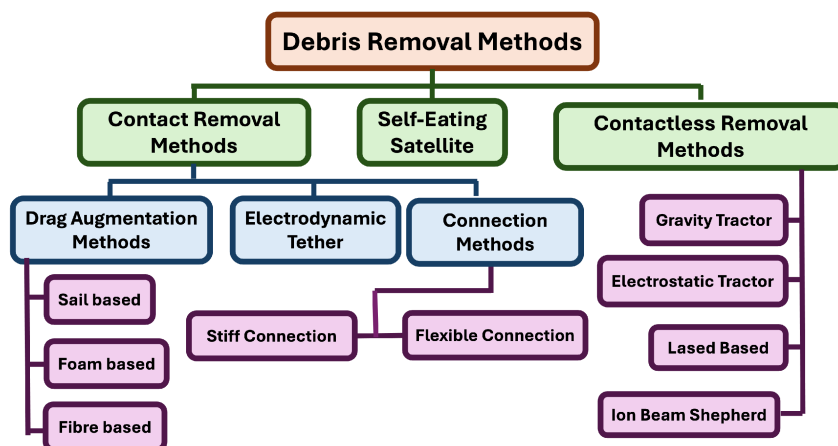


Figure 2.2: Active Space Debris Removal Methods.

Contact-based removal methods involve direct physical interaction with space debris, which is typically achieved by using robotic manipulators, nets, harpoons, magnetic mechanisms, or tether-assisted towing, as shown in Figure 2.3. They offer a high level of

control over the captured object, making them particularly suitable for large debris, such as defunct satellites and discarded rocket bodies. Nevertheless, using these kind of method, rigid capture systems can be risky and technically demanding because debris objects often move and rotate in an unpredictable way. Also, they lack docking interfaces and have unknown mass properties. These uncertainties make the rendezvous and capture phase one of the most complex and dangerous parts of missions with contact-based removal methods [2].

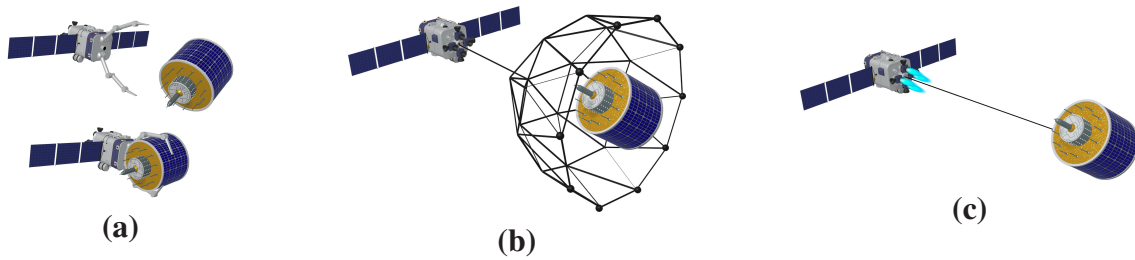


Figure 2.3: Contact-based space debris removal methods [2]: (a) Robotic manipulator. (b) Capturing with net. (c) Tethered towing.

Contactless methods, in contrast, do not require direct physical interaction with debris. These include laser ablation, ion beam shepherding, electrostatic methods and increasing the drag area of the debris object as shown in Figure 2.4. Instead of attaching to the debris, these systems apply forces externally from afar to accelerate the object and as a result gradually alter its orbit or increasing the cross-sectional area of the debris. They are especially useful for objects that are highly rotating, fragile or too small to safely capture. For example, laser ablation systems can apply photon pressure or induce material valorization to slowly change the debris trajectory. However, these systems demand high energy, long operational times and precise targeting capabilities, which make them expensive and difficult to scale [2].

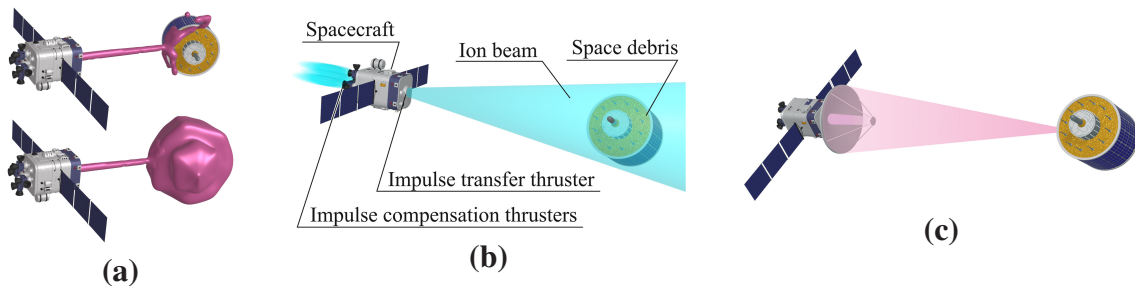


Figure 2.4: Contactless space debris removal methods [2]:(a) Foam based. (b)Ion beam shepherd. (c)Laser based.

2.2.1.1 Chosen Method: Tether Towing

In accordance with the MSHE grant topic, the tether-based method is chosen for this project. Tether-based methods' main advantage is the provided safety of the main

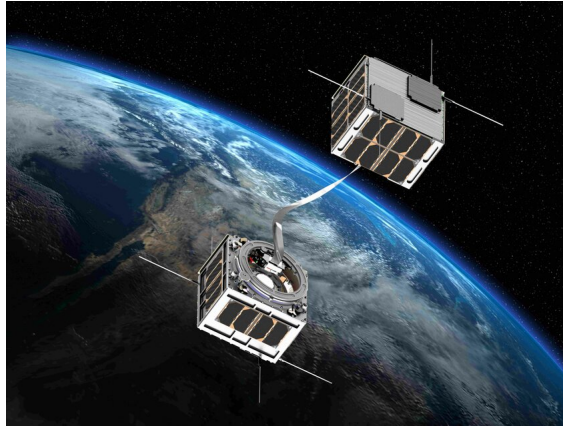


Figure 2.5: Rendering of Initial Tether Deployment Phase [3]

spacecraft and reduced need for propellant consumption during the debris removal process since they can extend to several hundred meters [1] without risking collision with target debris compared to other contact-based methods. Their light weight and flexibility also allow them to be installed into small and large satellites alike. However, for successful operation, tethers must be deployed with stability, or else they risk entanglement, resulting in mission failure. Once the target debris is captured, it can be towed and deorbited into the atmosphere to burn.

Tethers can be used in conjunction with different capturing mechanisms, including grippers, harpoons, and nets. Grippers have potential for capturing and handling different types of debris geometries but require fine attitude control over the tether and the gripper. Harpoons have high target precision but risk debris fragmentation and deflection of spinning or durable materials. Nets can capture multiple debris of different shapes and sizes even if they move [1], but said debris may slip out or get entangled in the net.

When it comes to tether deployment structures, there are different stowage, actuation, restraint, and location approaches with their own pros and cons. Both motors and springs can work as actuators, as an example [6]. Motors provide more control over deployment, as they often work to push and pull, which could fix potential deployment issues. However, they require power. Springs, meanwhile, are simpler in implementation and don't require power but don't provide as much control, and the compression has to be adjusted to control the spring force. Similarly, heat wires and metal wires can be used for restraint. Heat wires are lightweight and easy to cut, but they are soft and may release smoke when cut. Metal wires are stronger but require some power for pyro-cutters to cut, which could also result in the shock load capable of harm.

2.2.2 Propulsion Methods

Traditionally, a lot of spacecraft rely on chemical or electrical propulsion systems for orbital maneuvers and de-orbiting, because it is easier to control the thrust. Chemical

thrusters generate high thrust by burning propellant, while electric ones produce low thrust but very high specific impulse by accelerating charged particles.

Chemical systems like hydrazine, AF-M315F, cold-gas dominate the market and are the most flight-proven propulsion system for small spacecraft [7]. In order to operate it needs pressurized tanks, feed lines, valves and the propellant itself. Therefore the hardware accounts for a large fraction of the satellite's mass, which often exceed to be 20% of the total [7].

Electric thrusters like electro spray, ion thrusters provide large velocity with only a few grams of propellant, but the power-processing units, high-voltage electronics, thermal-control hardware and associated structural support weights tens of kilograms [7].

Consequently, these propulsion systems require large amount propellant mass, which as a result will make the satellite heavier. Therefore, these propulsion methods are impractical for CubeSats and small satellites, which have strict constraints on size, mass and energy capacity. As a result, alternative passive disposal methods have become increasingly attractive for these smaller platforms.

2.2.2.1 Chosen Method: Drag Augmentation

One such passive approach is a drag augmentation method, which aims to enhance the atmospheric drag by deliberately increasing the spacecraft's surface area [8]. It uses lightweight deployable structure designs, such as drag sails or inflatable devices to operate atmospheric particles in low Earth orbit (LEO) without propellant. The increased area-to-mass ratio significantly lowers the ballistic coefficient, allowing natural drag to slow the object and gradually reduce its orbital altitude until reentry occurs [8].

Moreover, in recent years drag sails have demonstrated its feasibility with several missions. One of them is Icarus-1 drag sail, developed at Cranfield University, and was flown on the TechDemoSat-1 mission, where it successfully deployed in 2019 and demonstrated a measurable increase in drag, leading to faster orbital decay. Later, the Icarus-3 mission on the Carbonite-1 micro-satellite achieved similar success in 2018 by significantly increasing the projected area and drag, confirming that such systems can work reliably in real space conditions. These missions underscore the potential of drag augmentation systems as a low-cost, efficient and scalable debris mitigation method [9].

It is essential to avoid complex mechanisms in order to reduce failure risk and increase the likelihood of successful end-of-life deployment [9]. Therefore, to build an effective drag augmentation system, several key components are required. These include a deployable drag surface, such as a sail or membrane, a deployment mechanism, a support structure, like booms or frames, and a stowage system that allows the device to remain compact during launch. Lightweight and flexible materials, typically thin polymer films, are selected to maximize surface area while keeping mass low. Structural elements

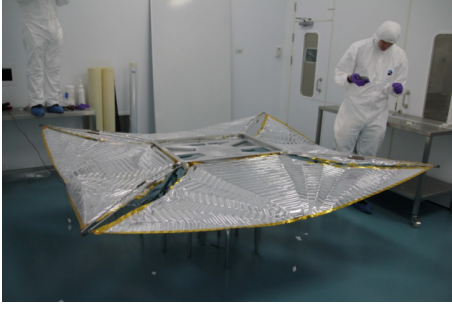


Figure 2.6: Icarus-1 in the deployed state

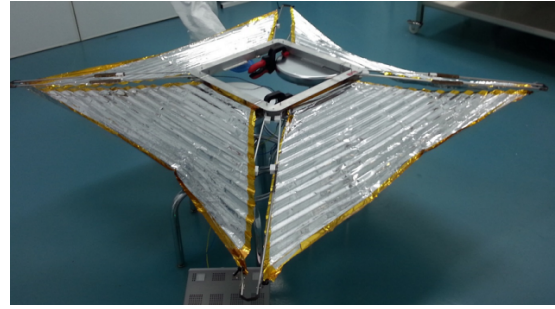


Figure 2.7: Icarus-3 in the deployed state

such as booms are used to tension and stabilize the membrane once deployed, ensuring predictable aerodynamic behavior [9].

2.2.3 Origami Technology in Space

Origami technology has inspired many deployable structures that satisfy lightweight, compact, and small-payload design requirements for today's satellites [4]. This has led to particular developments in CubeSat applications, with high packing efficiency yet large deployable surface areas. Translating various origami folding techniques to CubeSats has enabled shape-oriented kinematics, self-actuated structures with predictable motions, and scalable, packable designs [10]. Origami structures, due to their increased surface area, additionally serve as a drag-augmentation method.

Among the large variety of origami patterns, the Miura origami pattern is a widely studied one with a large database of modified parameters. The Miura pattern has many variations; however, it usually presents repeating parallelogram units with linked valley and mountain folds. In *Rubio* (2023), Miura-ori is emphasized to be highly valued in solar arrays, reflectarray antennas, and stowable panel applications. The pattern is predictable, and many studies on the pattern have allowed engineers to change key parameters such as thickness, panel dimensions, and volume efficiency.

2.2.3.1 Volume Efficient Miura-Ori

Altering parallelogram units in the Miura origami's shape allows the desired configuration to form in its stowed state. One such example is the development of the 5 by 3 Volume-Efficient Miura Origami (VEMO) pattern, which configures a cube in its stowed state and accounts for thickness [4]. The development of VEMO originates from the limitations of traditional Miura-ori structures that rely on the Offset Panel Hinge Shift (OPHS) technique [4]. OPHS, designed to thicken the structure, is limited at the minor ends of the Miura array, where significant protrusions emerge due to the geometry of the unit cell

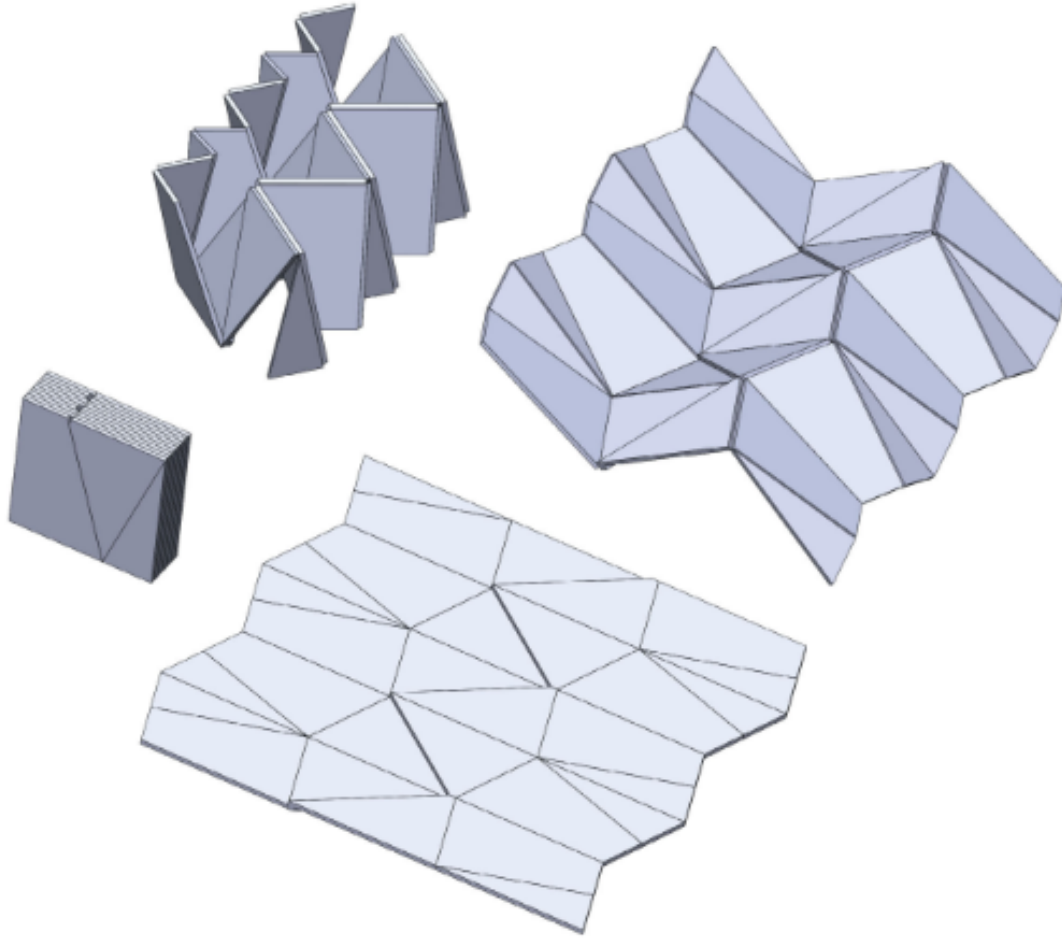


Figure 2.8: Volume Efficient Miura-Ori Pattern [4].

shaped like isosceles triangles. The unused protrusions and matching triangular gaps lead to wasted volume in the stowed configuration [4].

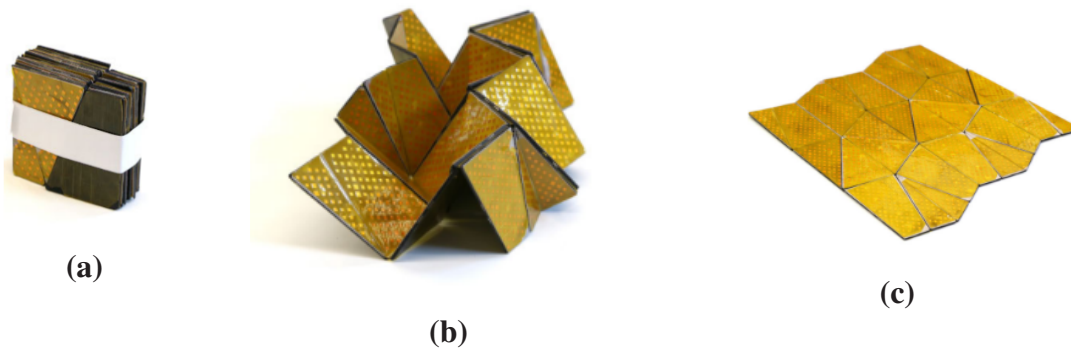


Figure 2.9: The assembled VEMO prototype [4]: (a) Stowed VEMO. (b)Unfolding VEMO. (c)Unfolded.

The VEMO array was optimized to maximize surface area and packing efficiency by folding each triangular protrusion into its adjacent triangular gap. In Figure 1.8, it can be

observed that the ends of the split array fit neatly into a cuboid shape. VEMO ensures that the pattern lies in the same plane when deployed, eliminating unwanted offsets. This innovative configuration of the Miura-ori preserves the original kinematics and allows the dimensions of the panel to be altered without concern for thickness.

2.2.3.2 Magnetic Locks

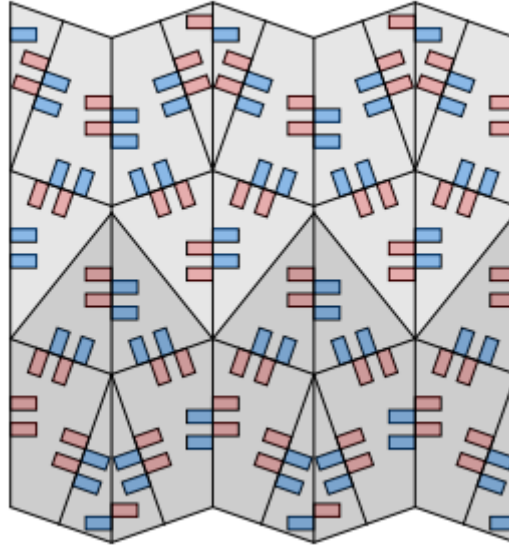


Figure 2.10: VEMO Polarized magnets' orientation [11].

Apart from its folding mechanism and stowed state, there is another consideration for origami structures, as they need to adhere to their deployed state by some unfolding mechanism. To figure out the unfolding mechanism of the VEMO, several methods were considered to control and lock the movement [10].

The deployment system uses small neodymium magnets embedded inside each unit cell structure of the origami. The magnetic locking mechanism both unfolds the structure and locks it in its final shape, forcing the origami structure into its deployed state. Each unit cell has hinges that contain a pair of magnets arranged in a Magnetic Longitudinal Offset (MaLO) configuration [12]. The working schematics and placement of MaLO hinges can be observed in Figure 1.11, where two magnets sit side by side in the same plane with a small lateral offset. Deployment is driven by the stored magnetic energy inside the MaLO hinge. Each hinges contains strategically placed neodymium magnets as described in Figure 1.10, and such placement creates a repulsive force when the origami structure is fully folded. In the closed state, the magnets in adjacent panels are pushed apart, creating a high-energy stowed configuration. Releasing the tether in Figure 1.9 causes the repulsive forces to rotate the hinges, pushing the panels toward the open, low-energy state. This

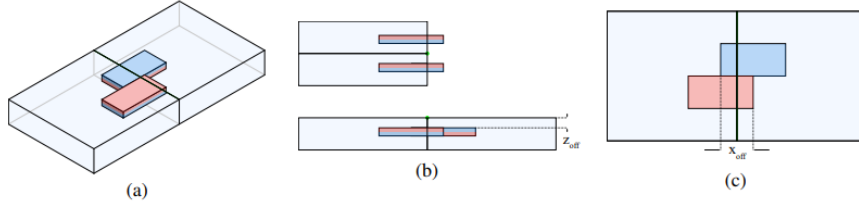


Figure 2.11: MaLO hinges working mechanism [11].

unfolding mechanism, due to the repulsive forces of the magnets, is the main actuation of the structure.

After the deployment the hinges are positioned behind the antenna's ground plane to not interfere with the CubeSat's overall performance.

2.2.4 Tendons

In large space structures, such as space stations, antennas, solar arrays or deployable frameworks, structural flexibility is a major challenge. As a result of their size and low stiffness, these structures are prone to vibrations and dynamic oscillations and this can degrade performance and complicate attitude control. *Murotsu et al.* (1987) introduced tendon control systems as an effective approach to control and stabilize such flexible space structures.

They refer to tendons as tensioned cables or wires attached to strategic points in a structure, similar to how biological tendons transmit forces, these artificial tendons are used to apply controlled tensile forces that influence the shape, stiffness and dynamic behavior of the structure. Tendon control systems are lightweight, simple, and well adapted for applications in space, where mass restriction is a key requirement [13], because they can only carry tensile loads without adding additional load and actuation capacity of the body.

Such a principle can be applied to actuate and control the panel deployment mechanism, avoiding abrupt movement that could induce structural vibrations or destabilize the spacecraft. By applying tension in a controlled manner, the tendon system allows the panels to open progressively, reducing dynamic loads and minimizing the risk of mechanical shock during deployment.

Tendons can also serve to actively control the effective drag area. It can be done by adjusting tendon lengths or tensions, so that the angle of the panels relative to the airflow can be increased or decreased. This makes it possible to regulate the aerodynamic drag experienced by the system, allowing the CubeSat to change its speed and adapt de-orbiting rate depending on missions requirements, orbital conditions.

2.3 Aim & Objectives

Aim: Build a prototype of a space tug with an origami-based deployable aerodynamic brake.

Objectives:

1. Select layout of the satellite (deployable, inflatable, or combined).
2. Design the satellite prototype.
3. Develop mathematical model of the orbital and attitude motion.
4. Design a deployment control system and attitude control system.
5. Manufacture and assemble the prototype.

3. Deployment Stages of Space Debris Removal Experiment

Our prototype space tug deployment consists of six main stages: initial stage, mock debris separation, tail panel deployment, tail panel latching, origami deployment, and the towing stage. At the initial stage, the four panels fully cover both the space tug and the mock debris, bound together by heat-cut wires and the long tether in the tether storage box within the tug. At this stage, the spring between the tug and the mock debris is compressed, storing potential elastic energy necessary to launch the mock debris.

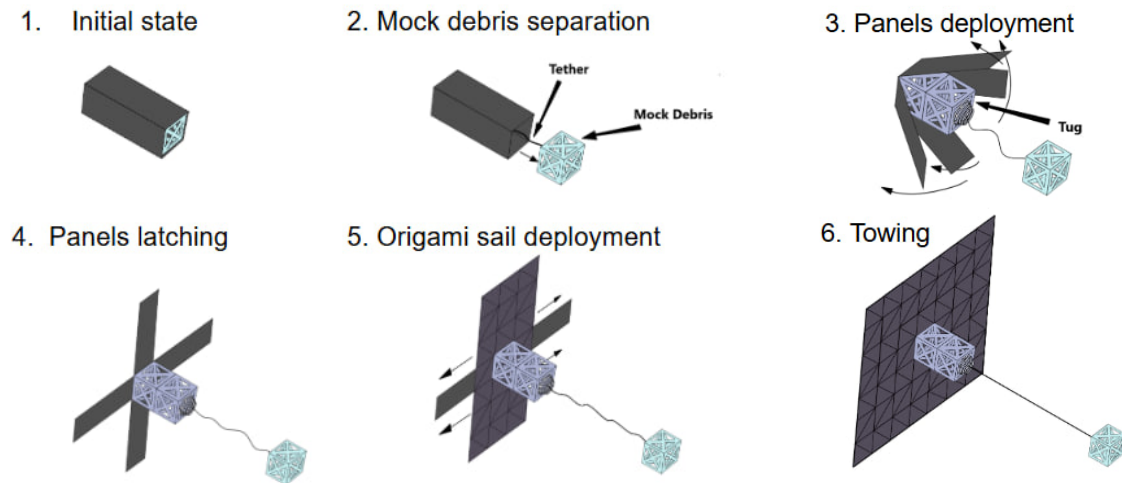


Figure 3.1: Deployment Stages of Space Debris Removal Experiment

At the next stage, the heat wires are cut, releasing stored elastic energy that propels the mock debris away from the tug. The mock debris momentum pulls the tether gradually out of the storage box until the maximum separation distance is reached, at which point towing is achieved. At the tail panel deployment stage, input power is transmitted to the worm gear motor, which is then transmitted to the tendon actuators through bevel gears. As tendons move the panels, the worm gear eventually stops, which provides the self-locking feature necessary to keep panels stable. Then, the Miuri origami is deployed and unfolded by the repulsive force resulting from pairs of magnets with the same polarity. Magnets are placed on the origami such that each magnet is adjacent to another magnet of the opposite polarity when unfolded, keeping the origami rigid and stable with the resulting attractive force. Once the origami reaches full deployment and the tug reaches full towing, the space

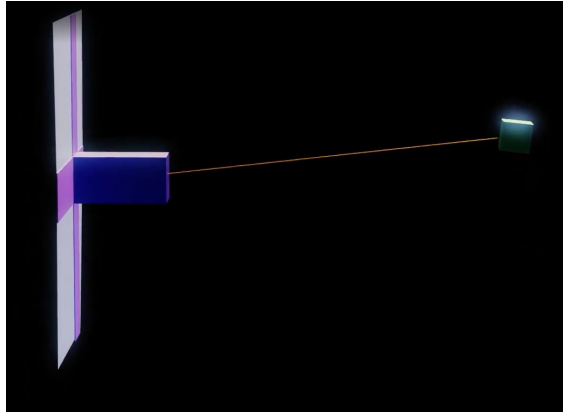


Figure 3.2: Aruzhan Bakyt, Aruzhan Temreshova (2025). Towing Stage of 3U CubeSat Deployment in Space.

debris removal demonstration begins. The tether is pulled back into its storage box by the motor in the tug, pulling the mock debris along. Figure 3.2 shows the towing stage of the satellite as part of the MSHE project.

4. Methodology

In this work, we developed a methodology based on 4 main components: mathematical modeling, aerodynamic analysis, control system development, and manufacturing prototypes. Each of the components ensures that at the end of the project, when built, the tug will have the characteristics of being dynamically stable, mechanically feasible, and realistically implementable during the Low Earth Orbit (LEO) debris removal mission. The entire project progressed through the development of theoretical models to build prototypes, continuously checking that every stage and component meet both physical constraints and mission goals.

1. Modelling Orbital and Attitude Dynamics.

Initially, we need to develop an accurate mathematical model of how the tug and debris would interact when linked via a tether. Due to the complexity of the system with many bodies interacting, we rely on a combination of both Newtonian and Lagrangian mechanics. Orbital mechanics are described well using Newton's laws since they describe how forces act upon a body, such as gravity, atmospheric drag and the tension in the tether. Therefore, these equations allow us to monitor how an object evolves over time and estimate how long it takes for an object to deorbit under the drag force created by the sail.

However, the best way to solve an attitude motion problem is through the Lagrangian method. The deployed structure has irregular geometric properties, and different areas of the spacecraft are affected by aerodynamic forces, so Lagrangian mechanics allow us to represent rotations with generalized coordinates and include all of the aerodynamic torque, gravity gradient torque, and internal actuation effects in the equations of motion, as well as conduct a simulation of how the system reacts under realistic scenarios (i.e., how the tow behaves at the moment the sail is deployed, if any oscillations will grow or diminish, and if the system will remain at an angle while the tow is held), which then tends to lead you to the design of the deployment sequence and control strategy for the deployed structure.

2. Aerodynamic Analysis in Rarefield Flow.

The operation of the system occurs within a low-earth orbit (LEO) atmosphere; therefore, one cannot treat the atmosphere as a continuous fluid, as the density of a LEO atmosphere is relatively low and gas molecules interact with one another as individuals (i.e., they collate to make up the large-scale structure). For that reason, we chose to utilize

rarefied-flow (free-molecular) aerodynamics in our analysis rather than the normal drag equation approach.

The first step was to approximate the shape of the sail and panel configurations upon deployment and determine the number of particles reflecting off of those surfaces (based on both diffuse and specular reflection models) over a broad selection of angles of attack to determine drag coefficients, C_d , for both geometric shapes defined by their sail and panel arcs. This informed us of the amount of atmospheric drag that a deployed sail could create and how effectively it would decelerate the space vehicle.

We also calculated the aerodynamic moment coefficient C_m , which is important for determining the magnitude and direction of stabilizing or destabilizing torque on the payload or spacecraft. If the aerodynamic moment is greater than that that can be balanced out by the payload, the tug could begin to spin uncontrollably or slacken the tether. Thus, these aerodynamic characteristics were applied as the base assumption for both the control system and mechanical designs.

3. Control System Design

Based upon the aerodynamic coefficients and a dynamic model, a control strategy has been developed for the tug. The strategy's primary objective is not to carry out high-performance manoeuvres, but rather to ensure continuous and stable performance during the towing of debris with the deployment of a sail.

The control system will be designed to complete three key tasks:

1) Stabilizing Attitude During Towing. The aerodynamics sail will create a natural stabilizing effect, but external disturbances (e.g., Differences in density and gravity) can still create perturbations to the system. Control laws will be developed to eliminate oscillations from external disturbances while maintaining a proper angular alignment of the tug with respect to the payload or debris.

2) Deployment Safety. The mass distribution and aero-area of the panels and the origami sail can change rapidly during deployment and create unexpected torques. Therefore, the controller must ensure that the system does not drift or roll over during this transition.

3) Tether Tensions. The tether must remain taught and not whip, wrap, or lose effectiveness, which is why the controller directs the motion along the tether axis and minimizes the likelihood of creating slack.

To create our controller's design, we will utilize a basic state-space representation of the system created from the dynamics. Subsequently, we will complete stability analyses to demonstrate that the closed-loop system remained stable when perturbed with expected disturbances.

4. Prototype Design, Manufacturing and Assembly

After completing our theoretical work, we started the design phase for the physical prototype for testing whether our deployment mechanism and overall geometry would function correctly in practice. This stage began with detailed CAD models of all subsystems: CubeSat frame, tail panels, worm gear reducers, bevel gearing pairs, tendons, magnetic locks, and Miura-ori elements.

After validating the designs, we will use 3D printing to make the components according to the approved design. Using this technology allow us to quickly alter the component size based on tolerance and weight and therefore have rigid, lightweight components created. The origami element magnetic pockets, gears, and tendon components were also 3D printed to ensure they worked correctly.

Once constructed, we will assemble the entire prototype together. During the assembly, we will observe how both the folding and unfolding sequences operate, how the tendons align within each subsystem, how reliable the magnetic locking mechanisms are, and whether all the panels deployed synchronously with the same timing. This test will confirm that the folding mechanism works as planned, and that the origami sail geometry is consistent with what we had simulated.

5. Results and Discussion

5.1 Key Dimensions

Table 5.1 summarizes key dimensions of major prototype satellite components.

Component	Length (cm)	Width (cm)	Thickness (cm)
Tug	20	10	10
Mock Debris	10	10	10
Tail Panels	30	10	
Origami Sail	70	70	

Table 5.1: Key Dimensions of Satellite Components

5.2 Panel Deployment Mechanism

The panel deployment mechanism shown in Figure 5.1 is a system that controls the opening and closing of the panels after the CubeSat launch. As it is shown in Figure 5.2, the mechanism consists of tendons, stepper motor and set of spools, which hold the tendon to control the process.

Initial idea was to use the motor with worm gear, which has a self-locking characteristic, meaning that once the rotation stops, the motor will hold the position of the shaft, so that it will not rotate the opposite direction. Nevertheless, the way new actuation system utilizes a stepper motor to drive the panel deployment mechanism. Unlike worm gear system, the stepper motor provides precise motion control by doing discrete angular steps, allowing for the accurate control of panel deployment without complex feedback system. Moreover, the holding torque of the stepper motor enables to maintain the shaft position, reducing the need for a separate locking mechanism. Next, the rotational motion of the stepper motor is transmitted through the shaft with spools. These spools are connected to the panels of the CubeSat via another set of spools, which direct the tendons. As the motor rotates, the tendons are gradually unwound, resulting in a controlled and smooth deployment of the panels. This approach is used in order to ensure smooth opening motion of the panels rather than abrupt, minimizing disturbances and preventing destabilization of the space tug. Furthermore, the panels' angular position can be controlled by adjusting the number

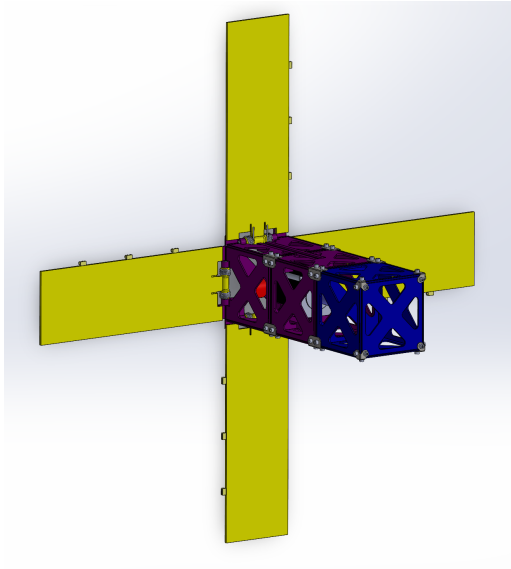


Figure 5.1: 3D model of panel deployment mechanism.

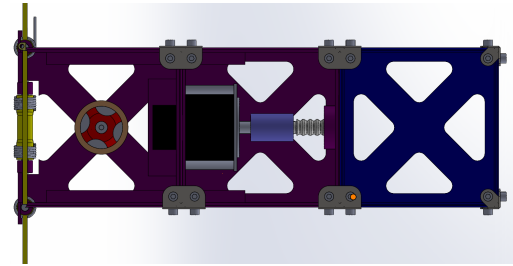


Figure 5.2: Close up view of inside of panel deployment mechanism.

of motor steps. It allows to actively regulate the exposed surface area of the CubeSat, consequently, control over the aerodynamic drag force.

5.3 Drag Augmentation System



Figure 5.3: Unfolded state of origami structure .



Figure 5.4: Folded state of origami structure

5.3.1 Muiri Origami Structure

Based on our thorough research, our project chose to use an origami structure as the main drag-augmentation mechanism. To build a feasible prototype, the instructions from [11] were applied. The VEMO structure was designed by 3D printing with overall dimensions of $15 \times 15 \text{ cm}^2$ in its unfolded state (Figure 4.3) and $5 \times 5 \times 5 \text{ cm}^3$ in its cubic stowed state (Figure 4.4). Building this model deepened our understanding of foldable mechanisms and allowed us to adjust the dimensions of the origami structure to achieve our desired augmented drag value. The stowed configuration of our chosen origami design would be used to compactly store it within one of the 1U CubeSat units.

5.3.2 Magnetic Locks

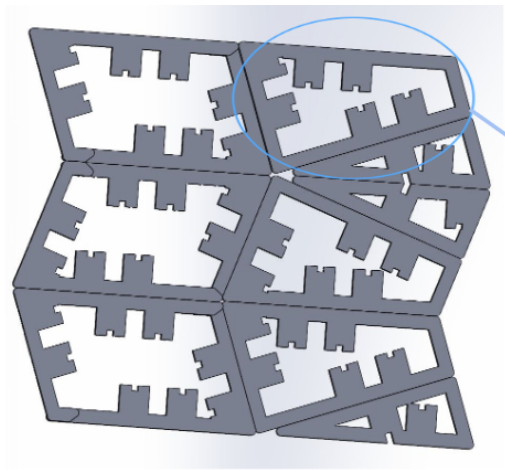


Figure 5.5: Assembly of the Origami Structure with pockets for magnets .

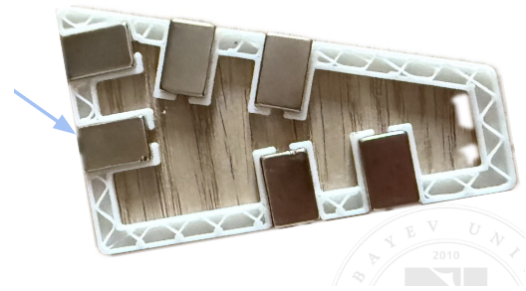


Figure 5.6: 3D printed trapezoid origami element with magnets

Accounting for the unfolding mechanism, which was chosen to be magnetic locks [11], the previously built prototype was modified to include specially designed pockets for magnets, as shown in Figure 4.5. The pockets account for the size of the neodymium magnets purchased from the local market, which have dimensions of $15 \times 10 \times 3 \text{ mm}^3$. The magnet dimensions may be altered in the future for flexibility with the origami structure's parameters such as thickness, volume, and face orientation. Figure 4.6 shows the printed prototype of one of the unit cells with magnets embedded inside. The 3D-printed trapezoid origami confirmed our objectives for magnet placement and demonstrated the working unfolding mechanism. Overall, the prototype is still in the testing stage, and further printing work is scheduled. After several iterations of printing, it was evident that proposed VEMO design needs following adjustments. Particularly in the central section that connects the top and bottom parts. As illustrated in Figure 4.7, the elements are currently solid rather

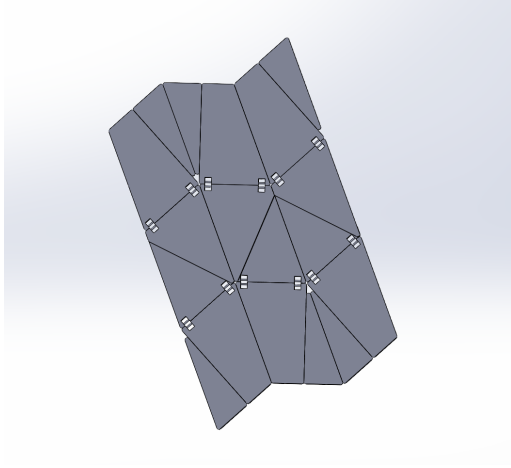


Figure 5.7: Assembly of the Origami Structure with pockets for magnets .

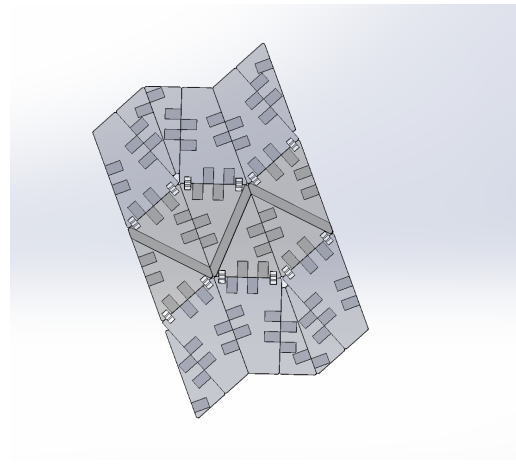


Figure 5.8: 3D printed trapezoid origami element with magnets

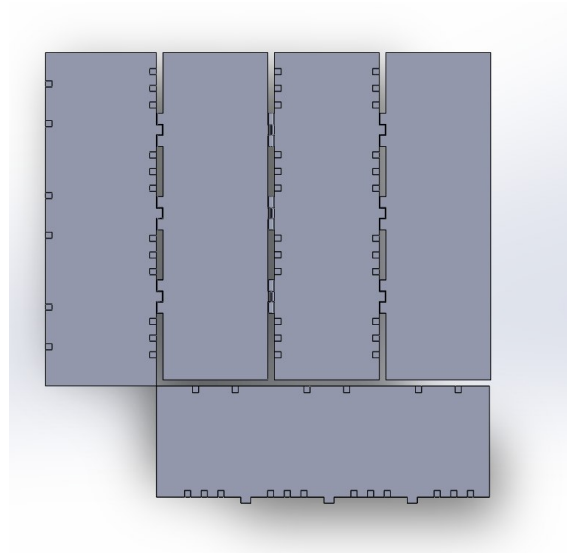


Figure 5.9: Origami structure

than hollow, which deviates from the original design. Additionally, another important modification involves incorporating hinges between the central connecting elements of the structure, allowing for improved flexibility and functionality.

The initial VEMO design had some minor limitations, especially when it came to fitting and storing it inside CubeSat structure. Because of these issues, a redesign was necessary, and a new origami-based configuration was chosen. This new design approach helps improve how compact the system is, while also making it easier to integrate into the CubeSat and ensuring a more reliable deployment. The revised origami structure is composed of several panels that function as deployable and structural parts. The design has a layout akin to a standard CubeSat panel, as seen in Figure 4.9. Three foldable panels are joined to a main panel to form the origami structure. The origami structure is $94.5 \times$

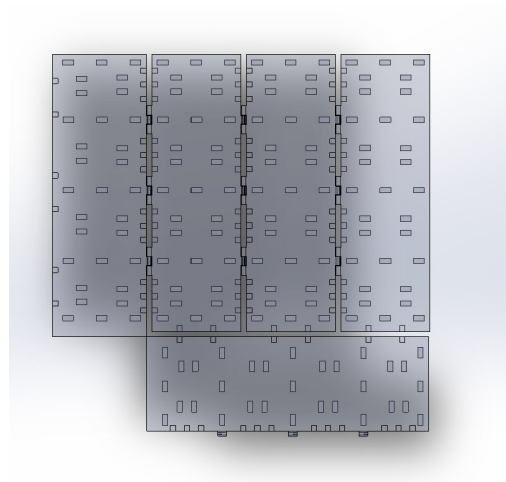


Figure 5.10: Origami structure transparent

294 × 2 mm, and the main panel measures 100 × 295 × 3 mm. The structure can fit the CubeSat correctly while retaining its strength and functionality thanks to this slight size discrepancy. Door-style hinges are used to join the origami structure to the main panel. These hinges allow rotation up to 180 degrees, which is important for achieving the desired forward deployment. This implies that the system can fully stretch outward in the desired direction once it is deployed. For space applications where dependability is crucial, the use of these hinges makes the design straightforward and efficient. Another important part of the design is the placement of the magnets, as shown in Figure 4.10. The system uses neodymium magnets of N52 grade with custom dimensions of 11 × 5 × 1 mm³. These magnets are placed based on an optimized version of the original VEMO design. Their purpose is to create an opening force that helps the origami panels unfold. This happens because the folded panels are in an unstable position, and the magnetic force helps push them toward their deployed state. In addition, the magnets work on the MalDO[11]hinge principle. They not only help with the deployment force but also act as additional magnetic hinges, assisting in guiding and stabilizing the motion during deployment. This very same magnets make the deployment of the origami structure in the horizontal "lay down" position as well, when the gravitational forces and mass of the origami structure work against the opening of the magnets. Additionally once the origami structure finishes its opening in the forward direction the origami structure is designed to accommodate another additional magnets at the bottom of its structure. This was done in order to secure the origami panels in both XY direction by locking it with the another panel based on the same principle. This reduces the need for extra mechanical parts and makes the overall design more efficient. Overall, the new origami configuration improves the performance of the system while keeping the design relatively simple and practical for CubeSat applications.

5.4 Aerodynamic Calculation

In order to test the 3U CubeSats's aerodynamic characteristics, the drag coefficient's and moment coefficient's dependency on angle of attack was analyzed. It was done by analyzing three configurations of the CubeSat, namely folded, panels deployed and fully deployed states as it shown in Figure 5.11.

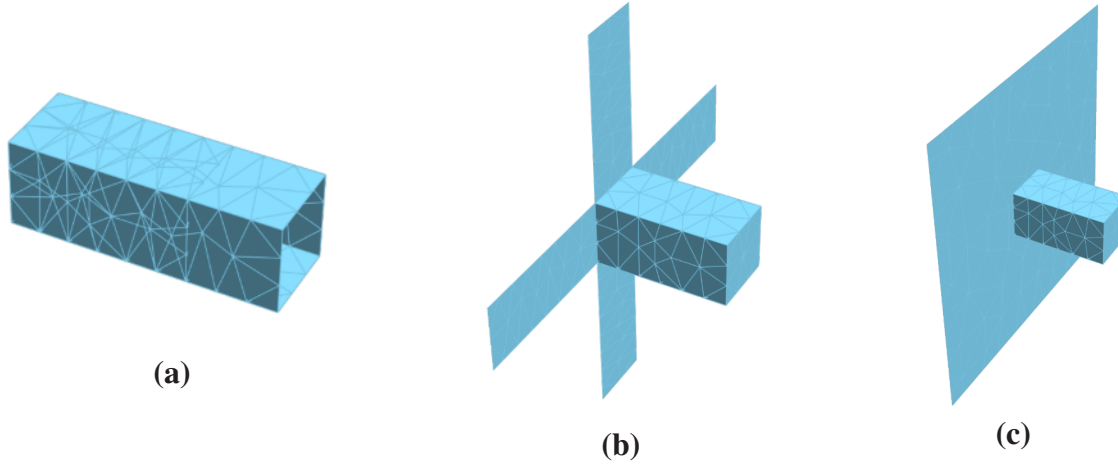


Figure 5.11: CubeSat configurations:(a) Initial state. (b) Panels deployed state. (c) Fully deployed state.

The graph representing the change in drag force with change in angle of attack is given in Figure 5.12. It can be seen that the origami sail largely increases the drag coefficient as the drag force and area are directly proportional to each other, which proves that origami sail can be used to control the drag force exerted on the CubeSat.

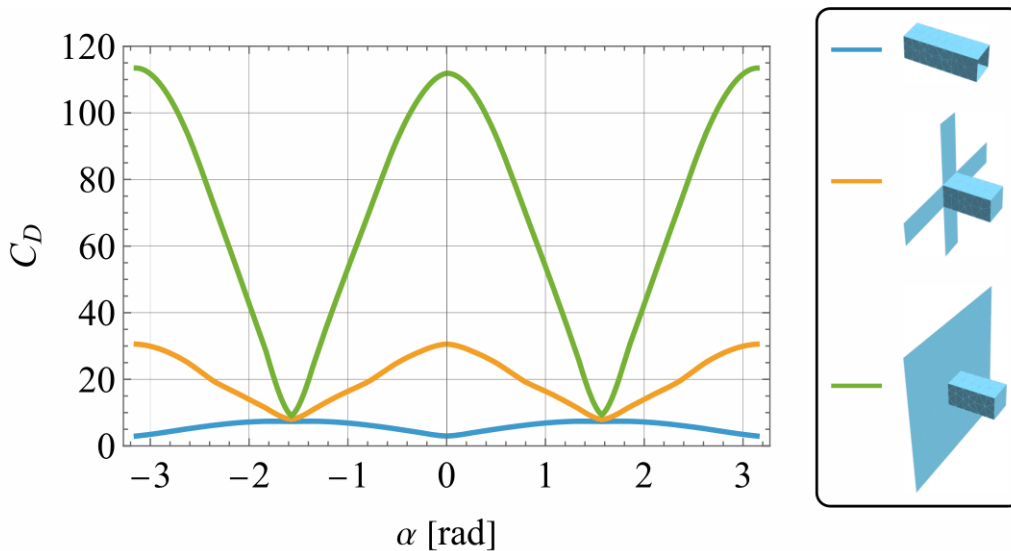


Figure 5.12: Angle of Attack vs. Drag Coefficient.

The graph in Figure 5.13 shows the change in the moment of the CubeSat as the angle of attack alters. It can be seen that CubeSat exerts no torque when angle of attack is equal

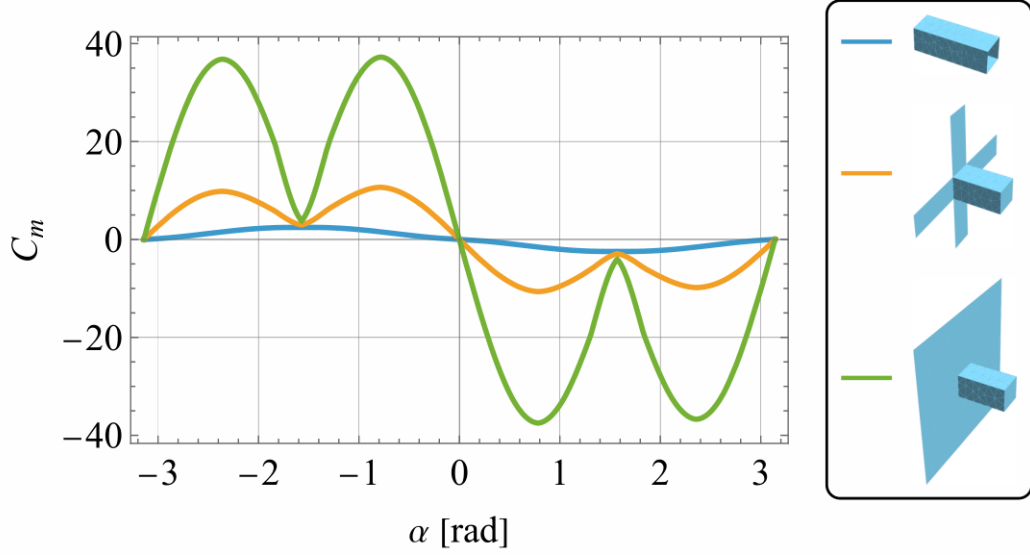


Figure 5.13: Angle of Attack vs. Moment Coefficient.

to zero. Nevertheless, when the angle of attack is negative and external force is applied to the bottom of the CubeSat, it exerts positive torque to stabilize itself and comeback to its initial state. Therefore, CubeSat always tries to stabilize itself without any other external assistance.

5.5 Components selection calculation

Table 5.2: Initial Condition

Variable Name	Value	Unit
θ_{min}	0	rad
θ_{max}	1.571	rad
θ_{fp}	2.007	rad
m_p	0.3	kg
m_o	0.25	kg
m_{fw}	0.1	kg
r_{fw}	0.05	m

5.5.1 Torsion spring

As it was stated, the torsion spring is used for the panel deployment mechanism. The main parameter necessary to choose the right torsion spring is spring constant k . In order to find it, the torque required to deploy the panels was estimated by calculating the gravitational torque acting on the panels about the hinge axis. It was done by taking into account the weight of the panel and origami structure acting at their center of mass and the

perpendicular distance from the hinge. Therefore, the gravity torque is expressed in the following way:

$$\tau_g = (W_p + W_m) \frac{L}{2} \quad (5.1)$$

where, τ is the maximum gravity torque, W_p is the weight of the panel, W_m is the weight of the origami, and L is the panel's length.

The value of the τ was used to approximately estimate the spring constant k using the equation below.

$$k = \frac{\tau}{\theta_f} \quad (5.2)$$

where, θ_f is the free position angle of the torsion spring.

Typically, CubeSat panels are deployed to an angle of 90° relative to the satellite body because this configuration provides an optimal balance between power generation, structural simplicity, and reliability. Nevertheless, it is necessary to choose torsion spring, which deflect more than needed angle in order to maintain positive torque during the operation. Therefore, θ in Eq. (5.2) was taken to be 115° . As a result, the approximate value of the spring constant is found to be:

$$k = \frac{\tau}{\theta} = \frac{0.10521}{115^\circ} = 0.00092$$

$$k = 0.00092 \text{ Nm/deg}$$

5.5.2 Stepper Motor

According to the design, in order to prevent sudden and abrupt motion of the panels during opening process, tendon controlled by the stepper motor will be used. The main idea is to unwind the tendons, which are directly connected to the panels via sets of spools, in a controlled manner. Therefore, the torque from the torsion spring and gravity torque has to be taken into account. The torsion spring torque was found the following way:

$$T_s[\theta] = k(\theta_f - \theta) \quad (5.3)$$

where, T_s is the torsion spring torque depending on the θ and θ is the instantaneous deployment angle.

Illustration of their dependency is given below as a graph of $T_s[\theta]$ vs θ :

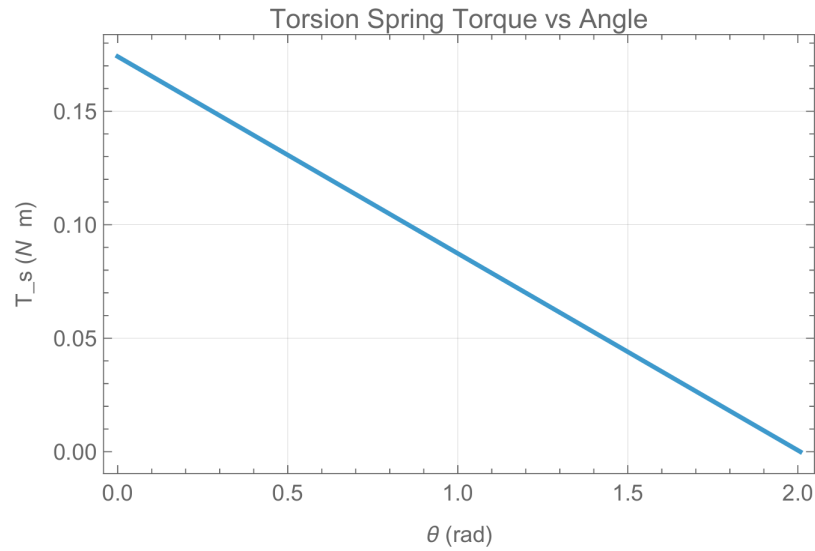


Figure 5.14: Angle of Deployment vs. Torsion Spring Torque.

In order to find the dependency of the gravity torque acting on the panels, the distance to the rotation point was expressed in terms of θ .

$$T_g = (W_p + W_m) \frac{L \cos(\theta)}{2} \quad (5.4)$$

Then, the graph of $T_g[\theta]$ vs θ was plotted:

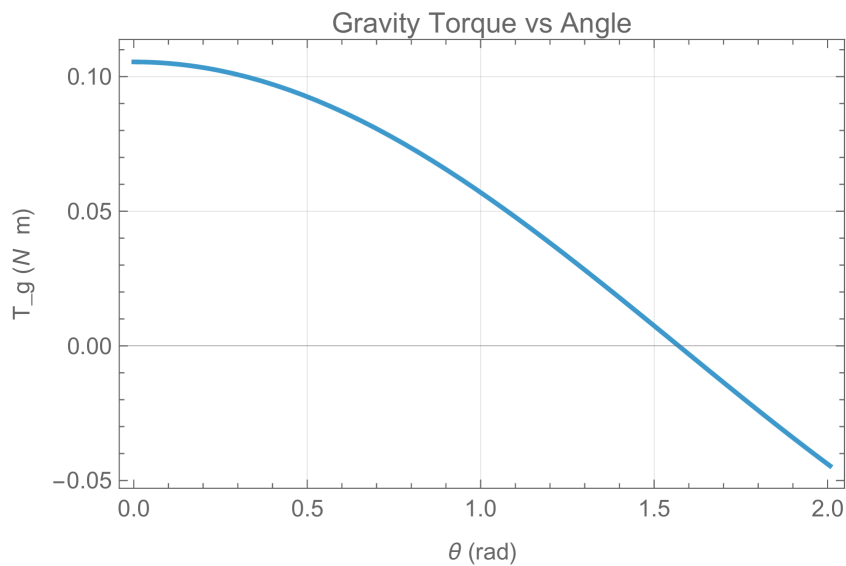


Figure 5.15: Angle of Deployment vs. Gravity Torque.

The torques acting on the panels differ depending on the panels' placement, namely top, bottom and side. After developing the equations for the calculation of the torques involved, torque acting on each panel can be calculated.

Top panel

5. Results and Discussion

As the gravity torque assist to pull the panel down, the torque equation for the top panel is derived to be like this:

$$T_{tp} = T_s - T_g = 0.0867458(2.007 - \theta) - 0.105458\cos(\theta) \quad (5.5)$$

where T_t is the torque acting on the top panel.

As it can be seen, the maximum torque occurs when θ is equal to 0, which is found to be $T_t = 0.069$

Bottom panel

In contrast with top panel, the gravity torque adds up to the torsion torque acting on the panel. Therefore, the torque equation for the bottom panel is given below:

$$T_{bp} = T_s + T_g = 0.0867458(2.007 - \theta) + 0.105458\cos(\theta) \quad (5.6)$$

where T_b is the torque acting on the bottom panel.

Similar to the top panel, the maximum torque occurs when θ is equal to 0, which is found to be $T_b = 0.279$

Side panel

Only torsion torque acts on the side panel, thus the side panel equation is given below:

$$T_{sp} = T_s = 0.0867458(2.007 - \theta) \quad (5.7)$$

where T_{sp} is the torque acting on the side panel.

The maximum torque occurs when θ is equal to 0, which is found to be $T_{sp} = 0.174$

The torque necessary from the stepper motor is found in the following way:

$$\tau_n = F_{\text{total}} \cdot r_{\text{spool}} \quad (5.8)$$

where F_{total} is the total force acting on the spool and the r_{spool} is the radius of the spool.

Each acting force was found using the following equation:

$$F = \frac{\tau_t(\theta)}{r(\theta)} \quad (5.9)$$

where $\tau(\theta)$ is the torque acting on the panel and $r(\theta)$ is the distance from axis of rotation to the application point.

The $r(\theta)$ is found by using the schematic below:

According to it, the $r[\theta]$ was found to be:

$$r(\theta) = \frac{0.0006 \sin(\theta)}{\sqrt{0.0013 - 0.0012 \cos(\theta)}} \quad (5.10)$$

Therefore, combining the Eq.(5.5), Eq.(5.6) and Eq.(5.7) with Eq. (5.10), each acting maximum forces were found.

$$F_{\text{top}} = 11.8 \text{ N} \quad F_{\text{bottom}} = 48.0 \text{ N} \quad F_{\text{side}} = 29.9 \text{ N}$$

As a result,

$$F_{\text{total}} = F_{\text{bottom}} + F_{\text{top}} + 2F_{\text{side}} = 119.7 \text{ N}$$

Now, the torque required from the stepper motor is calculated to be:

$$\tau_n = F_{\text{total}} \cdot r_{\text{spool}} = 119.7 * 0.005 = 0.59 \text{ Nm}$$

5.5.3 Brushless Motor

A brushless motor is used to drive a flywheel inside a CubeSat as a part of an attitude control system, commonly known as a reaction wheel. In order to do so, the torque from the flywheel τ_{FW} has to counteract the torque from the satellite itself τ_S .

$$\tau_{FW} = \tau_S \quad (5.11)$$

The torque from the flywheel is equal to the multiplication of the moment of inertia I_{FW} of the flywheel and its angular acceleration α_{FW} .

$$\tau_{FW} = I_{FW} * \alpha_{FW} \quad (5.12)$$

The torque from the CubeSat is equal to the multiplication of the moment of inertia I_S of the flywheel and its angular acceleration α_S .

$$\tau_S = I_S * \alpha_S \quad (5.13)$$

Since the angular acceleration of the CubeSat during the disturbance and control phases was not provided analytically, a numerical approach by plotting its graph was used to estimate its maximum value. In calculations Time T , which means stabilization time of the CubeSat body, was taken to be 10 seconds. Calculations were done by using the software Wolfram Mathematica. Below are the resultant graphs, which codes are given in Appendix.

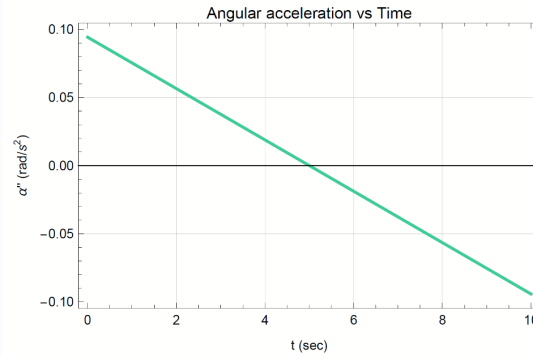


Figure 5.16: Angular Acceleration vs. Time.

$$\alpha'' = 0.09426 - 0.018852t \quad (5.14)$$

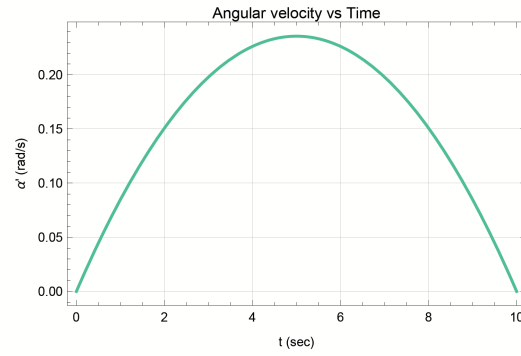


Figure 5.17: Angular Velocity vs. Time.

$$\alpha' = 0.09426t - 0.009426t^2 \quad (5.15)$$

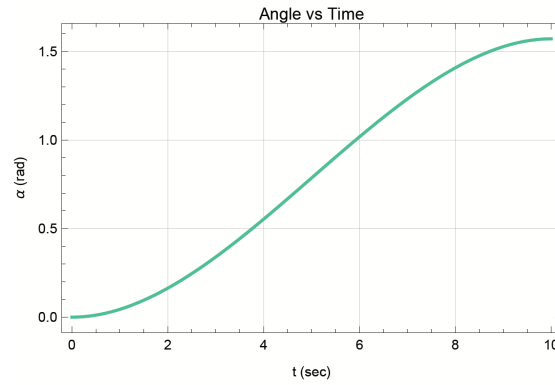


Figure 5.18: Angle vs. Time.

$$\alpha = 0.04713t^2 - 0.003142t^3 \quad (5.16)$$

Now, by using the Eq.5.11, Eq.5.12 and Eq. 5.13, the angular acceleration of the flywheel α_{FW} can be found.

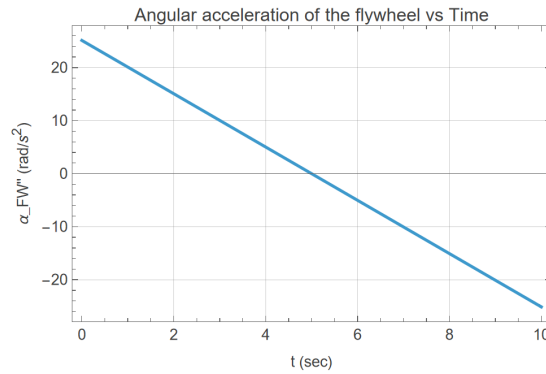


Figure 5.19: Angular acceleration of the flywheel vs. Time.

$$\alpha_{FW} = 266.56(0.09426 - 0.018852t) \quad (5.17)$$

As it can be seen, it is maximum when time is equal to 0. Now, substituting all the value into Eq/5.12, the torque necessary from the motor is found.

$$\tau_{FW} = I_{FW} * \alpha_{FW} = 0.000125 * 25.1259 = 3.14 \times 10^{-3} Nm$$

List of Components

Component	Specification	Manufacturer	Quantity
Magnets	Dimensions: 11×6×1 mm Tolerance: 0.05 mm	Chinese Henan Samson (Neodymium N52)	1000
Stepper Motor	2.8 A, speed: 200 RPM 56.90 mm × 77.47 mm	STEPPERONLINE Nema 23 CNC	1
Torsion Spring	Material: Stainless Steel Length: 173 mm Spring Index: 6.75	Local Supplier	16
Brushless Motor	A2208, 1800kV 27.2 mm × 36 mm	FEICHAO	1
Arduino Uno R3	ATmega328P, 5 V 68.6 mm × 53.4 mm	Arduino	1

Additional components: Filament, TB6600 Driver, Arduino set, bolts, cable, stopper, switch, sensor (encoder), springs

5.6 Assembly

As a result, final version of the 3D CAD model of the space tug is developed.

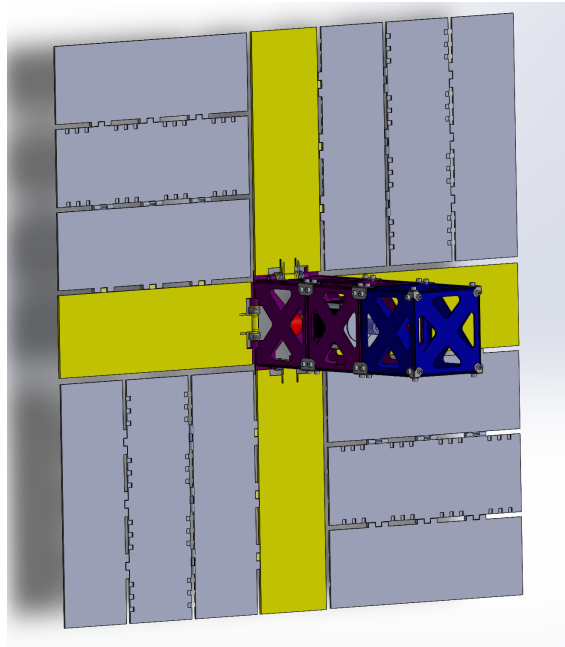


Figure 5.20: 3D Model of the CubeSat

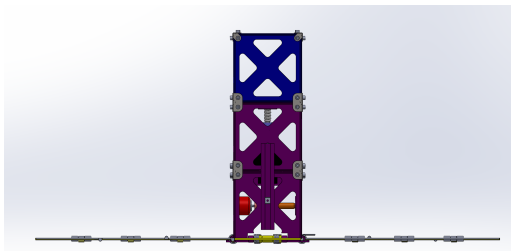


Figure 5.21: Side View of the CubeSat.

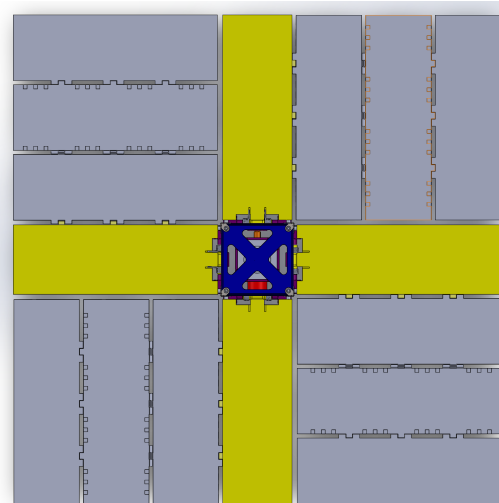


Figure 5.22: Top View of the CubeSat.

The connection between the components is presented in the technical drawing.

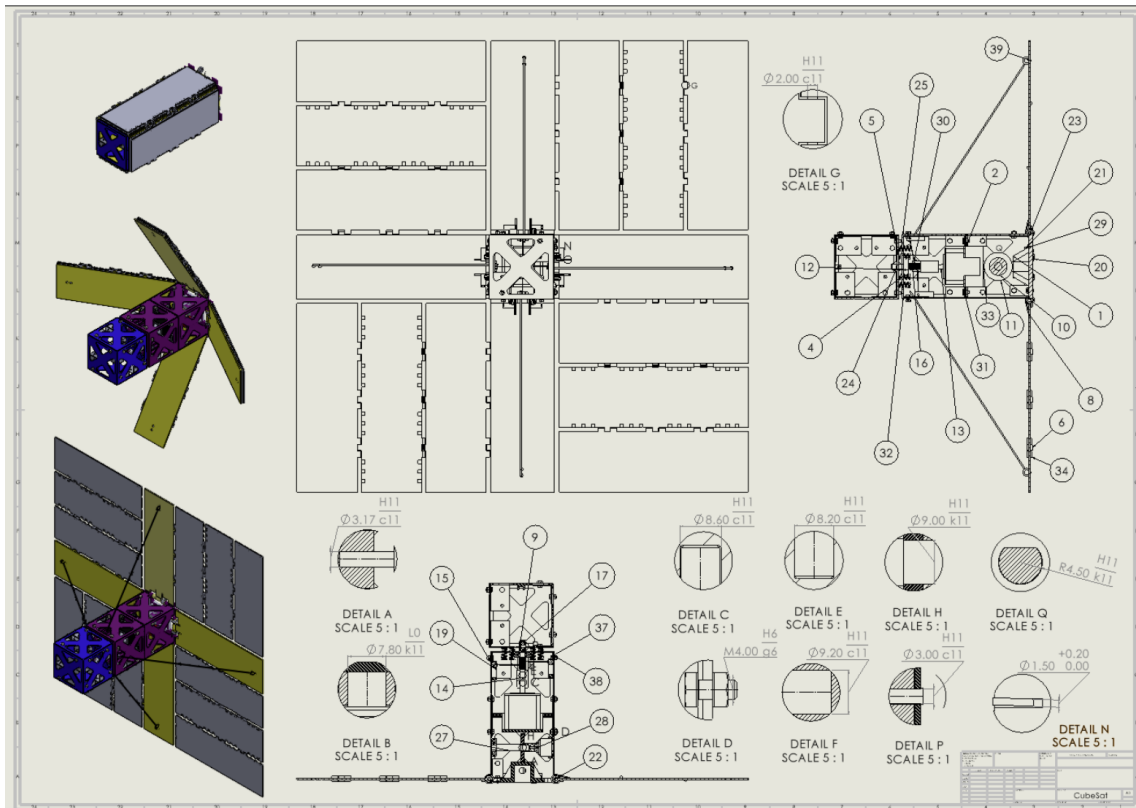


Figure 5.23: Technical Drawing and Bill of Materials of the CubeSat

5.7 Simulation

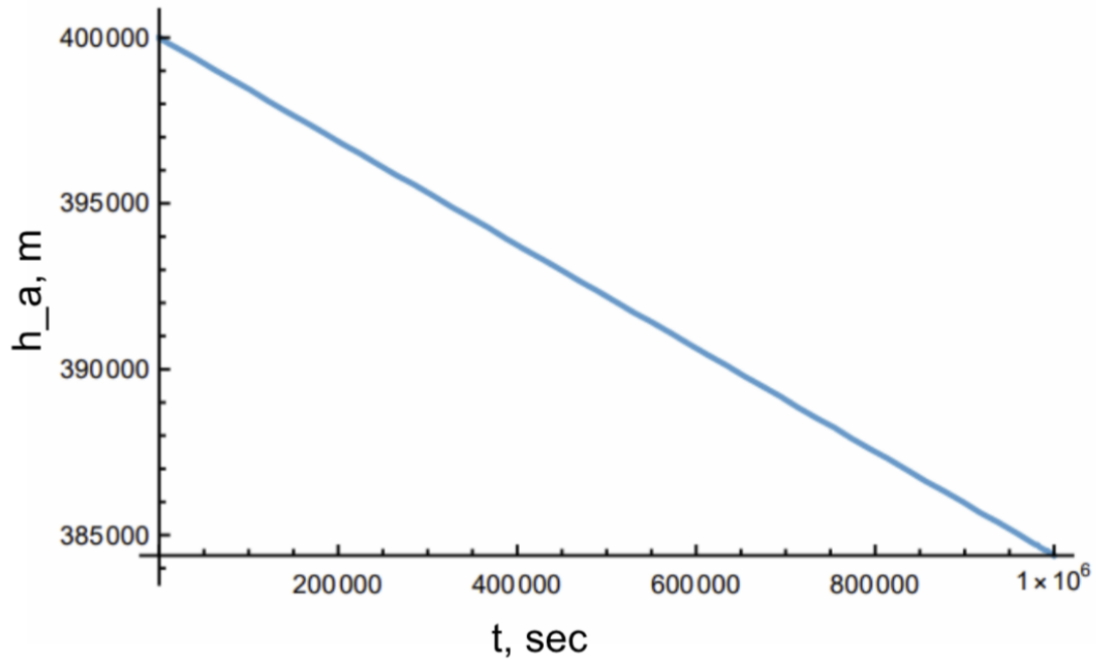


Figure 5.24: Altitude h_a vs. t

As a result of the simulation of the de-orbiting of the CubeSat with fully opened configuration, the graph of altitude vs time was plotted.

The plot shows a continuous decrease in orbital altitude over time, indicating effect of atmospheric drag on satellite. It demonstrates that the satellite is gradually losing its orbital energy due to drag force acting on the fully deployed panels and origami. As it can be seen, it takes approximately 2 weeks for a space tug to de-orbit, if its initial altitude was approximately 400 km. Thus, it shows that effective surface area can significantly enhance drag and be used as passive method for controlled deorbiting.

5.8 Discussion

The outcomes achieved through this investigation validate both the mechanical viability and functional potential of a CubeSat sized space tug with an origami style aerodynamic drag system for use as a means of eliminating space debris.

The first observation was that the panel deployment mechanism exhibited reliable and controlled motion characteristics because of the use of both a tendon driven system and a worm-gear motor to create smooth gradual motions as the panels deployed thereby precluding any violent motion left unrestrained that may have caused the CubeSat to

become unstable during the transition from the folded configuration to the deployed configuration; therefore confirming that this actuation method is appropriate for small satellite space missions, in particular with respect to stabilization of the CubeSat during deployment.

The second observation was that the origami based drag augmentation system demonstrated a capability of increasing aerodynamic drag while remaining compact when stowed. The aerodynamic analysis (refer to images Figures 4.10 and 4.11) concluded that once deployed, the aerodynamic drag coefficient of the system, when compared to its folded state, experienced significant increases in value which contributes directly towards fulfilling the goal of enhancing the efficiency of deorbiting satellites. In addition, the moment coefficient of the origami based drag augmentation device was measured to have an aerodynamic stability characteristic; specifically whereby the generated torque acts to return the CubeSat to an equilibrium orientation as a result of being subject to aerodynamic drag forces acting on the deployed flap system (i.e., flap drag). The use of magnets to create magnetic locks is another validation of the design concept. The magnets were able to provide both deployment and structural rigidity during the fully deployed configuration while eliminating much of the complexity of a mechanical locking solution. There were, however, some minor design issues noted as part of the prototype build including a lack of sufficient "hollowness" and the addition of hinges to the origami structure indicating a need to further optimize the geometry.

In terms of component selection, the analytical calculations for the torsion springs, stepper motor torque, and reaction wheel system have all confirmed that the selected components are capable of meeting the operational requirements. The calculated motor and flywheel torque (0.59 Nm and 3.14×10^3 Nm respectively) also indicate the components selected will operate within an acceptable engineering limit, as such the system design is feasible/practical.

There are however limitations that should be noted. The aerodynamic analysis was performed using simplified rarefied flow assumptions, therefore additional factors such as atmospheric variability and outer torques may affect actual orbital conditions. Additionally, testing was performed in the lab environment and will not be able to accurately replicate micro-gravity or a vacuum as present in space.

The results suggest that the integrated drag augmentation, tether tow, and origami structures can be incorporated into a small CubeSat platform. The information gathered from this study is an excellent starting point for the future improvement of the structure, refinements to the control algorithms, and testing the cube satellites against more realistic environments.

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improvement of the structure, refinements to the control algorithms, and testing the cube satellites against more realistic environments.

- **Wire Diameter (d):** $d = 1.5 \text{ mm}$
- **External Diameter (D_1):** $D_1 = 14.4 \pm 0.36 \text{ mm}$
- **Lever Arm Length (r):** $r = 20.0 \text{ mm}$
- **Initial Angle (θ_0):** $\theta_0 = 115^\circ$
- **Number of Coils (n):** $n = 5.25$
- **Body Length (L_0):** $L_0 = 9.375 \pm 0.31 \text{ mm}$
- **Spring Mass (m):** $m = 0.0034 \text{ kg}$

6. Engineering Ethics

1. Safety first.

First and foremost, safety was taken into account when designing and testing this project. The deployable panels, the torsion springs and tether system all included protective features to help reduce possible hazards. Along with this, there were procedures in place for safe handling of the equipment during testing, and designated "keep out" zones on site to assist the team in remaining safe at all times.

2. Responsible space engineering mindset.

The project was created using a responsible approach toward space engineering by doing its best to develop a low risk of producing additional debris or loose parts. Priority was given to the safe deployment and end-of-life characteristics of the design so the project contributes to a sustainable environment in space.

3. Integrity in results.

Integrity in results was demonstrated through the accurate reporting of all calculations and presenting testing results honestly. No manipulation of test data or selective reporting was used thus allowing for the firm confidence that the results provide an accurate reflection of the performance of the system.

4. Competence.

Competence was achieved through the continuous peer reviews conducted with the supervisors and through following a systematic requirement-based approach to the system engineering process. All assumptions made, design decisions and limitations were thoroughly documented thus ensuring the work will be able to be duplicated and verified by other engineers.

7. Future Work and Calendar Plan

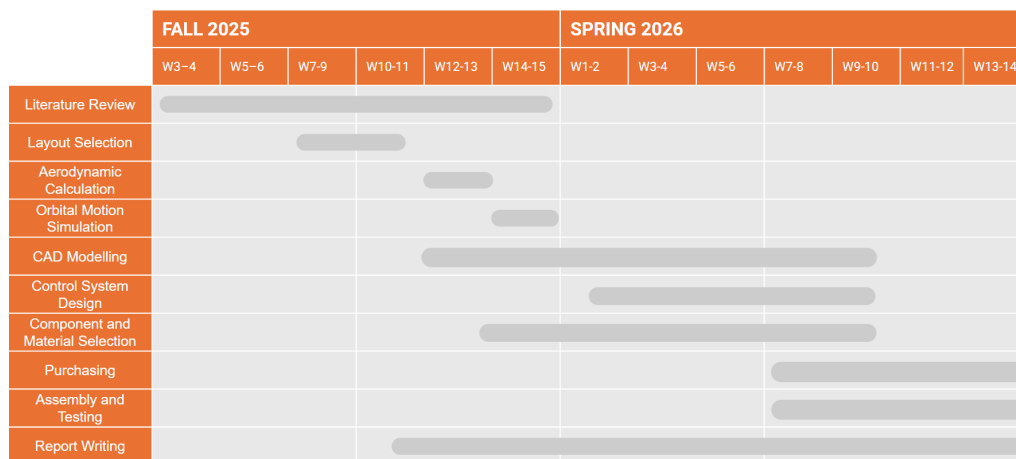


Figure 7.1: Calendar Plan

The timeline of this project stretched across two academic years: Fall (2025) and Spring (2026), and consisted of a step-by-step procedure that gave rise to the project through extensively developing analytical and conceptual methods, performed from a theoretical perspective to provide the necessary scientific foundation for the missions success, and from a practical perspective to evaluate how the concept would work in relation to design and feasibility for implementation.

To create the project's foundation during the Fall term (2025) the focus was on developing analytical and conceptual methods associated with the project through a published analysis of the literature was accomplished from Week 3 to Week 15 of this term, by conducting a literature search that created a theoretical foundation for the project. During this time frame, a layout selection was conducted from Weeks 5 - 9 to evaluate various design concepts that could potentially be utilized for a mission in space. Once the design concepts had been identified, analytical calculations were performed to establish the aerodynamic properties of the layouts from Weeks 10 - 11 in order to confirm that at least one layout could meet the necessary performance criteria to develop the CAD (computer-aided design) models. During this time frame, the final CAD models of all three designs of the spacecraft will be completed by the conclusion of the Fall term (Week 14 - 15).

During Spring 2026, the project was changed to the detailed design phase and the implementation phase of the project began. At the start of this semester, the CAD model of the spacecraft was completed and the control system design of the spacecraft was completed. The component selection, material selection, and manufacturability of the components for the spacecraft were performed in accordance with the requirements of the design and the limitations of the manufacturing process. As of this point (Weeks 13 - 14 of Spring 2026), the focus of the project has moved into the purchasing phase, assembly phase, testing phases, and preparation of the final report.

Most of the remaining effort for this project is related to assembly and validation of the prototype. During assembly, the components of the deployment mechanism, the origami-based drag structure, and the tether system will be merged to create an operational prototype. Once assembly is complete, performance testing of the deployment mechanism, structural integrity, and general operation of the completed system will occur under simulated conditions to verify the validity of the design and to support the conclusions from the project.

In general, however, the overall progress of the project is on track with respect to the planned timeline, and assembly and validation of the prototype is currently on-going.

8. Conclusion

This project has created the groundwork for a space tug built to the CubeSat scale using an origami-based aerodynamic drag system that will assist in the removal of space debris via tethering. The combination of drag augmentation through multiple methods (tether tow, deployed origami structures) has demonstrated that this type of system can fit into the limitations of small satellite platforms but also operate as an efficient functional system.

Aerodynamic analysis confirmed that the use of an origami sail as deployed provides a positive outcome for the drag coefficient and therefore an increased ability to deorbit. The moments produced during testing were also consistent with the aerodynamic stability of the system, allowing for controlled conditions during the retrieval process.

The mechanical design has been validated through the creation and testing of prototypes with a tendon-driven mechanism and magnetic locking configuration to deploy the sail and use of origami structures to provide efficient transition between stowed to deployed position. The results have verified that the proposed design is feasible, compatible for use on CubeSat scale and can be realized.

The calculation results for determining all required torques to deploy and maintain attitude control of the mechanism are within the current engineering limits providing practical use of the proposed system.

There are still many issues with current research. Data collected so far was based on simplified aerodynamic models and stationary test conditions that do not accurately reflect the actual conditions in space; therefore, further research will still be needed to develop specific structural design details, improve control algorithms, and validate system performance in realistic orbital test environments.

Overall, the proposed system has the potential to be a viable, low-cost, lightweight, scalable means to mitigate the hazards of orbital debris compared to traditional propulsion-based mechanisms. Developing the proposal will require more complete integration of the entire system, more advanced computer simulations, and experimental validation before it can be implemented into actual space operations.

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A. Appendix A. Technical Drawing

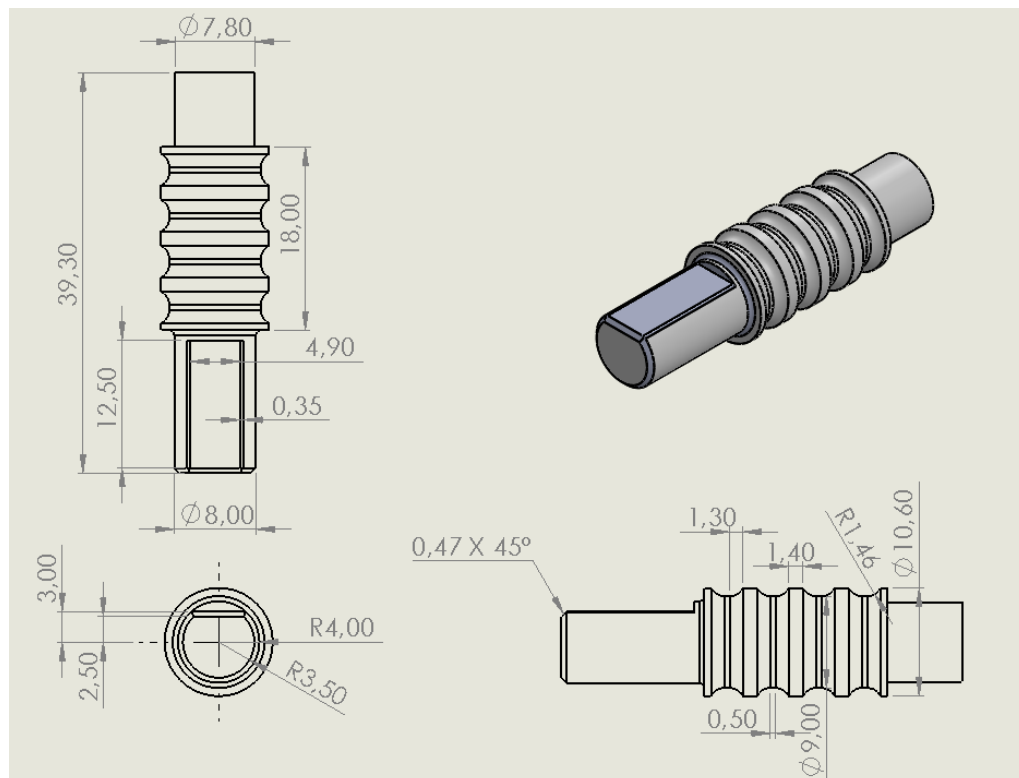


Figure A.1: Shaft and Spool for Stepper Motor

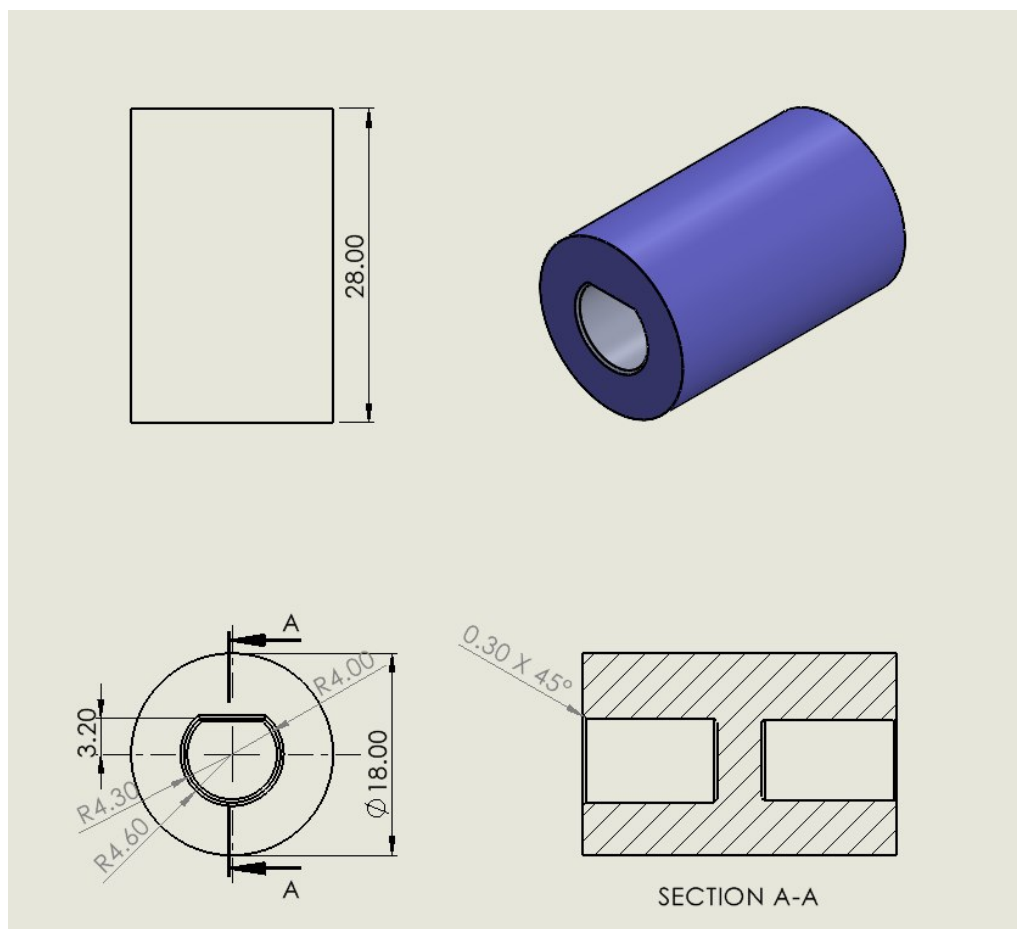


Figure A.2: Coupling for Stepper Motor

Appendix B. Source Code

1. Torsion spring and stepper motor selection

```
1 values1 = {
2     thetaMin -> 0 ,(*[rad] when panels are closed*)
3     thetaMax -> 1.571,(*[rad] when panels are opened*)
4     thetaF -> 2.007 ,(*[rad] free position of torsion spring*)
5     M -> 2.3,(*mass of the CubeSat body*)
6     m1 -> 0.3,(*[kg] panel mass*)
7     m2 -> 0.25,(*[kg] origami mass*)
8     g -> 9.81,(*[m/s^2] *)
9     L -> 0.03 (*[m] CubeSat length*),
10    W -> 0.01 (*[m]CubeSat width*),
11    omega -> Pi/10,(*[rad/s]*)
12    Wp -> 0.001 (*thickness of the panel*),
13    Fs -> 1.3,(*safety factor*)
14    rSpool -> 0.005(*[m]*),
15    U -> 0.01,
16    rgearsmall -> 0.02 (*[m]*),
17    T -> 20 (*[s]*),
18    rGear -> 0.03 (*[m]*),
19    mfw -> 0.1 (*[kg]*),
20    rfw -> 0.05 (*[m]*)};
21
22 (*Gravity torque*)
23 maxtorque = Fs*((m1 + m2)*g*L/2) /. values1 (*Nm*)
24 0.105212
25
26 kdeg = maxtorque/115
27 0.000914889
28
29 (*Rate k*)
30 kcalc = maxtorque/thetaF /. values1
31 0.0524226
32
```

```

33 (*Spring torque max*)
34 tauSpring = k*(thetaF - thetaMin) /. values1
35 0.174099
36
37 Ts = k (thetaF - theta) /. values1;
38 Plot[Ts, {theta, 0, thetaF /. values1}, Frame -> True,
39   FrameLabel -> {"\[Theta] (rad)",
40    "T_s (\!\!(\*SubscriptBox[\(N\), \(\ \)]\)\m)"}],
41   PlotLabel -> "Torsion Spring Torque vs Angle",
42   GridLines -> Automatic]
43
44 Tg = (m1*g*L/2 + m2*g*5*L/6)*Cos[theta] /. values1;
45 Plot[Tg, {theta, 0, thetaF /. values1}, Frame -> True,
46   FrameLabel -> {"\[Theta] (rad)",
47    "T_g (\!\!(\*SubscriptBox[\(N\), \(\ \)]\)\m)"}],
48   PlotLabel -> "Gravity Torque vs Angle", GridLines -> Automatic]
49
50 Top panel
51
52 Tctop = Ts - Tg
53 0.0867458 (2.007 - theta) - 0.105458 Cos[theta]
54
55 Bottom panel
56
57 Tcbottom = Ts + Tg
58 0.0867458 (2.007 - theta) + 0.105458 Cos[theta]
59
60 Side panel
61
62 Tcside = Ts
63 0.0867458 (2.007 - theta)
64
65 steps = {l -> Sqrt[(3 U)^2 + (2 U)^2 - 2*3 U*2 U*Cos[theta]],
66   Cos[alpha] -> (3 U*Sin[theta])/l}
67 {l -> Sqrt[13 U^2 - 12 U^2 Cos[theta]], Cos[alpha] -> (3 U Sin[theta])/l
68   }
69
70 r\[Theta] = 2 U*Cos[alpha] /. steps /. U -> 0.01
71 (0.0006 Sin[theta])/Sqrt[0.0013 - 0.0012 Cos[theta]]
72
73 Fbottom = M\[Theta]/r\[Theta] /. { M\[Theta] -> Tcbottommax}
74 465.927 Sqrt[0.0013 - 0.0012 Cos[theta]] Csc[theta]

```

```

74
75 Ftop = M\[Theta]/r\[Theta] /. { M\[Theta] -> Tctopmax}
76 114.402 Sqrt[0.0013 - 0.0012 Cos[theta]] Csc[theta]
77
78 Fside = M\[Theta]/r\[Theta] /. { M\[Theta] -> Tcsidemax}
79 290.165 Sqrt[0.0013 - 0.0012 Cos[theta]] Csc[theta]
80
81 Fsidemax = NMaximize[{Fside, 0.1 <= theta <= (thetaF /. values1)}, theta
    ][[1]]
82 29.9234
83
84 Ftopmax = NMaximize[{Ftop, 0.1 <= theta <= (thetaF /. values1)}, theta
    ][[1]]
85 11.7978
86
87 Fbottommax = NMaximize[{Fbottom, 0.1 <= theta <= (thetaF /. values1)},
    theta] [[1]]
88 48.0491
89
90 Ftotal = Fbottommax + Ftopmax + 2*Fsidemax
91 119.694
92
93 torques = Ftotal*rSpool /. values1
94 0.598469

```

2. Brushless motor selection

```

1 angularacc = -\[Beta]*t + angularaccmax
2 angularaccmax - t \[Beta]
3
4 omega = Integrate[angularacc, {t, 0, T} ]
5 angularaccmax T - (T^2 \[Beta])/2
6
7 \[Phi] = Integrate[omega, T] /. \[Beta] -> 2*angularaccmax/T(
    angularaccmax T^2)/6
8
9 solll = (Solve\[Phi] == thetaMax, angularaccmax)[[1]] // FullSimplify)
    /. values1
10 {angularaccmax -> 0.09426}
11
12 torqueRW = Iy*angularaccmax /. solll
13 0.09426 Iy
14

```

```

15 funIy = 0.00833 m;
16 torqueRW //. {Iy -> funIy, m -> 4, T -> 10}
17 0.00314074
18
19 angularaccc = -\[Beta]*t + angularacccmax /. solll /. values1 /. \[Beta]
    -> 2*(angularacccmax /. solll)/(T /. values1)
20 0.09426 - 0.018852 t
21
22 Plot[angularaccc, {t, 0, 10}, Frame -> True,
23   FrameLabel -> {"t (sec)",
24     "\[Alpha]" (rad/!\(\*SuperscriptBox[\(s\), \(\(2\)\)\])"}},
25   PlotLabel -> "Angular acceleration vs Time", GridLines -> Automatic]
26
27 angularvelocity = Integrate[angularaccc, t]
28
29 0.09426 t - 0.009426 t^2
30 Plot[angularvelocity, {t, 0, T /. values1}, Frame -> True,
31   FrameLabel -> {"t (sec)", "\[Alpha]' (rad/s)"},
32   PlotLabel -> "Angular velocity vs Time", GridLines -> Automatic]
33
34 angularvelocity = Integrate[angularaccc, t]
35 0.09426 t - 0.009426 t^2
36
37 Plot[angularvelocity, {t, 0, T /. values1}, Frame -> True,
38   FrameLabel -> {"t (sec)", "\[Alpha]' (rad/s)"},
39   PlotLabel -> "Angular velocity vs Time", GridLines -> Automatic]
40
41 Ifw = mfw*rfw^2/2 /. values1
42 0.000125
43
44 angularaccfw = funIy*angularaccc/Ifw /. m -> 4
45 266.56 (0.09426 - 0.018852 t)
46
47 Plot[angularaccfw, {t, 0, T /. values1}, Frame -> True,
48   FrameLabel -> {"t (sec)",
49     "\[Alpha]_FW" (rad/!\(\*SuperscriptBox[\(s\), \(\(2\)\)\])"}},
50   PlotLabel -> "Angular acceleration of the flywheel vs Time",
51   GridLines -> Automatic]
52
53 angularaccfw /. t -> 0
54 25.1259
55

```

```

56 rpmfw = angularaccfw*30/\[Pi] /. t -> 0
57 239.935
58
59 angularvelocityfw = Integrate[angularaccfw, t]
60 25.1259 t - 2.51259 t^2
61
62 Plot[angularvelocityfw, {t, 0, T /. values1}, Frame -> True,
63   FrameLabel -> {"t (sec)", "\[Omega]' (rad/s)"},
64   PlotLabel -> "Angular velocity vs Time", GridLines -> Automatic]
65
66 angularaccfw /. t -> 0
67 25.1259
68
69 m = Tm*angularvelocityfw /. t -> 5
70 0.197285

```

3. Simulation

```

1   In[2]:= inic = {mE -> 5.9722 10^24, (*mass of the Earth in kg*)
2   G -> 6.6743 10^-11, (*gravitational constant in m^3/kgs^2*)
3   rE -> 6.371 10^6,(*radius of the Earth in m*)
4   r0 -> rE + h,
5   \[Mu] -> mE G,
6   h -> 400 10^3,
7   \[Alpha] -> 6.7,
8   mA -> 2.6,
9   mB -> 1.3,
10  m0 -> mA mB/(mA + mB),
11  Ar -> 0.01,
12  \[Rho] -> 1.0*10^-12,      (*very low density at LEO*)
13  CDA -> 115,
14  CDB -> 2.2,
15  k -> 4000,
16  l -> 4.0};
17
18 eqn = {r''[t] == -(\[Mu]/r[t]^2) + r[t] v'[t]^2,
19   v''[t] == -((2 r'[t] v'[t])/r[t]) - (
20   Ar \[Rho] (CDA + CDB) ((r'[t])^2 + r[t]^2 v'[t]^2))/(
21   2 r[t] (mA + mB)), \[Theta]''[
22   t] == -((\[Alpha] \[Mu] Sin[\[Theta][t]])/r[t]^3) +
23   3/2 (\[Mu] Sin[2 \[Theta][t]])/r[t]^3 - (
24   2 \[Delta]'[t] (\[Theta]'[t] + v'[t]))/(1 + \[Delta][t]) + (
25   2 r'[t] v'[t])/r[t] + (

```

```

26      Ar \[Rho] (CDA + CDB) ((r'[t])^2 + (r'[t])^2 (v'[t])^2))/(
27      2 r[t] (mA + mB)),
28      \[Delta]''[
29      t] == -(k/m0) \[Delta][
30      t] + (\[Mu] Cos\[Theta][t]) \[Alpha] (1 + \[Delta][t])^2)/
31      r[t]^3 + (\[Mu] (3 Sin\[Theta][t]^2 - 1) (1 + \[Delta][t]))/
32      r[t]^3 + (1 + \[Delta][t]) (\[Theta]'[t] + v'[t])^2} //. inic;
33
34 Out[3]= {Derivative[2][r][t] == -(3.98603*10^14/r[t]^2) +
35      r[t] Derivative[1][v][t]^2,
36      Derivative[2][v][
37      t] == -(2 Derivative[1][r][t] Derivative[1][v][t])/r[t]) - (
38      1.50256*10^-13 (Derivative[1][r][t]^2 +
39      r[t]^2 Derivative[1][v][t]^2))/r[t],
40      Derivative[2][\[Theta]][
41      t] == -(2.67064*10^15 Sin\[Theta][t])/r[t]^3) + (
42      5.97904*10^14 Sin[2 \[Theta][t])/r[t]^3 + (
43      2 Derivative[1][r][t] Derivative[1][v][t])/r[t] + (
44      1.50256*10^-13 (Derivative[1][r][t]^2 +
45      Derivative[1][r][t]^2 Derivative[1][v][t]^2))/r[t] - (
46      2 Derivative[1][\[Delta]][
47      t] (Derivative[1][v][t] + Derivative[1][\[Theta]][t]))/(
48      1 + \[Delta][t]),
49      Derivative[2][\[Delta]][t] == -4615.38 \[Delta][t] + (
50      3.98603*10^14 (-1 + 3 Sin\[Theta][t]^2) (1 + \[Delta][t]))/
51      r[t]^3 + (2.67064*10^15 Cos\[Theta][t] (1 + \[Delta][t])^2)/
52      r[t]^3 + (1 + \[Delta][t]) (Derivative[1][v][t] +
53      Derivative[1][\[Theta]][t])^2}
54
55 In[4]:= (*\[Alpha]==(Subscript[A, r] \[Rho] r[t]^3(Subscript[C, \
56 Subscript[D, A]]Subscript[m, B]-Subscript[C, Subscript[D, \
57 B]]Subscript[m, \
58 A]))((r'[t])^2+r[t]^2(v'[t])^2))/(2 \[Mu] Subscript[m, A]Subscript[m, \
59 B])l(1+\[Delta][t]));*)
60
61 stopEvent =
62      WhenEvent[Evaluate[r[t] <= (rE //. inic) + 100 10^3],
63      "StopIntegration"];
64
65 Out[5]= WhenEvent[r[t] <= 6.471*10^6, "StopIntegration"]
66
67 ini = {r[0] == r0,

```

```

68     r'[0] == 0.0, \[Theta][0] == 10.    , \[Theta]'[0] ==
69     0.0, \[Delta][0] == -0.001, \[Delta]'[0] == 0.0, v[0] == 0.0,
70     v'[0] == Sqrt[\[Mu]/r0^3]} /. inic;
71
72 Out[6]= {r[0] == 6.771*10^6,
73   Derivative[1][r][0] == 0., \[Theta][0] == 0.174533,
74   Derivative[1][\[Theta]][0] == 0., \[Delta][0] == -0.001,
75   Derivative[1][\[Delta]][0] == 0., v[0] == 0.,
76   Derivative[1][v][0] == 0.00113316}
77
78 sol = NDSolve[
79   Join[eqn, ini, {stopEvent}], {r[t], r'[t], v[t],
80     v'[t], \[Theta][t], \[Theta]'[t], \[Delta][t], \[Delta]'[t]}, {t,
81     0, 10^7}][[1]];
82
83 Plot[Evaluate[r[t] - (rE /. inic) /. sol], {t, 0, 10^7}(*altitude*);
84
85 Plot[Evaluate[\[Theta][t]/Degree /. sol], {t, 0,
86   5000}(*tether oscillation*);

```