Abstract—In this report, parts of the systems engineering of a spacecraft are presented. In 2014 the Royal Institute of Technology KTH initiated a new space technology and research platform, the KTH Space Centre. The first student project at KTH Space Centre was the MIST student satellite with the scope of the system design and construction of a real satellite due for launch in 2017.

As part of the MIST project this bachelor thesis covers the mission analysis and parts of the system design. The system design is confined to the orbit lifetime and attitude perturbation analysis of the spacecraft and a complete analysis of the solar array illumination for the power system.

The orbit lifetime was simulated in two different software, AGI’s Satellite Toolkit (STK), and NASA’s Debris Assessment Software (DAS). An approximate lifetime could be calculated from the two separate programs.

The perturbation analysis was done by deriving parts of the governing equations of disturbance torques in the orbit. The analysis is confined to aerodynamic, magnetic dipole moment and gravity gradient torques. From these calculations the actuation strength needed by the attitude control system was determined.

The solar array configuration analysis was done by looking at the power generation and illumination of the solar panels in all of the possible flight attitudes.

Following a computation of the power requirements a power budget was created and in conjunction with the complete configuration analysis of the solar array the optimal flight attitude was determined.

The results of the system design analyses are compiled into a baseline design specification. The report is concluded with a discussion of the future work in the project and the lessons learned from the systems design process.

I. INTRODUCTION

SPACE is not as far away as it used to be. The global market for satellite services has been growing uninterruptedly for 16 years [1] resulting in a profitable market where the pressure for innovation is high. The CubeSat [2] is a relatively new type of spacecraft in the industry and with increasing amounts of applications the number of orbiting CubeSats has increased with over 600% in the last 10 years [3]. With the majority of operable CubeSats being university projects, it shows that the complexity of designing and building functional spacecraft is diminishing.

At the Royal Institute of Technology in Stockholm, Sweden, two CubeSat missions are currently in development. The first is the Smart Explorer for Advanced Missions (SEAM) project [4], in which the project scope is to develop and operate a small spacecraft with precision attitude determination and high-bandwidth telemetry for science qualified measurements of the Earth’s magnetic field. The project started at the KTH Space and Plasma Physics department in 2013 and is scheduled for three years. Together with the MiniTurte Student saTellite project (MIST), which is the second CubeSat project currently in development at KTH, many valuable insights in the design and development process has been gathered as both projects are of the 3U CubeSat spacecraft type. The SEAM project is built mainly on subsystems from the Danish company GOMspace [5] and throughout the MIST project much has been learned from their collaboration.

The content of this bachelor thesis and the goal of the MIST project team was to write the design specifications for the MIST satellite such that a following student team could initiate the procurement and development of the subsystems. In the final stages of the project the complete flight model of the satellite will be constructed and launched into space. If successful it will be a great accomplishment for the students involved and for KTH Space Centre and it would show that even space is within our reach.

II. PROJECT OVERVIEW

The scope of the MIST project is divided into two main sections, an educational scope and a scientific scope. The educational scope of the project is to engage about 10 students each semester under a period of two educational years to design and assist in the procurement of the necessary subsystems required to build a satellite equipped to meet the payload specific requirements, i.e. the scientific scope.

The scientific scope of the project was set by a committee at KTH Space Centre. Seven experiments where given a chance to participate in the project. Each experiment provided a payload application document consisting of estimates on the shape, size, mass and power requirement of their experiment. Together the payload application documents generated the outline of the mission requirements.

Another main task for the project was to design and build a satellite mock-up for a new science exhibition at Tekniska Museet in Stockholm, Sweden. In conjunction with this exhibition the satellite would be designed to carry a camera that would provide the exhibition with pictures from space once the satellite is in orbit. In the following subsections a brief summary of each experiment is presented. Through the experiments requirements the project design specifications could formulated.

1) CUBES: The Institute of Particle- and Astrophysics at KTH are conducting research on the cosmic background radiation environment of the low Earth atmosphere and as a part of their research they want to test a new type of detection technology. This detection method is less sensitive to magnetic fields and operate at low voltages, the basic research of this technology will provide insights to future missions.
2) **Piezo LEGS**: PiezoMotor AB is a company from Uppsala, Sweden. The company has developed a high precision instrument called the LEGS linear motor based on the piezoelectric effect. The scope of the experiment is to further the knowledge of the product’s endurance in the hard vacuum and radiation environment of space.

3) **MoreBac**: The European Space research and TEchnology Centre (ESTEC) are conducting research in life support units, Scilife’s MoreBac experiment is an experimental design for such a life support system. The purpose of MoreBac is to design and implement a closed regenerative life support system consisting of freeze dried microorganisms. These microorganisms are to be revived and their metabolic processes measured. If successful the microorganisms could constitute a foundation for organic life support units in future space missions.

4) **CubeProp**: NanoSpace AB is another company from Uppsala, Sweden. The company is part of the Swedish Space Corporation group (SSC) and specializes in Micro Electric Mechanical Systems (MEMS). In this project NanoSpace will provide their Cubesat Propulsion unit, abbreviated CubeProp, using proportional valves to release compressed butane in order to produce momentum. Their scope is to test the CubeProps delta-V capabilities, i.e. testing the thrust strength, and the potential application in conducting attitude orientation maneuvers.

5) **RATEX-J**: The Institute of Space Physics in Kiruna, Sweden, have constructed the RATEX-J, an abbreviation for RARadiation Test EXperiment for JUICE, to test scientific equipment for the JUpiter ICy moon Explorer, JUICE. On the MIST satellite they will test the radiation mitigation properties, and the anti-coincidence algorithms of the technology.

6) **SEUDet**: The Department of Electronic Systems at KTH have developed a way to trap the errors caused by Single Event Upsets (SEU) in electronic equipment. With this technology it is possible to isolate and repair the damage caused by the SEU during run-time by moving operations to undamaged units in the system. Due to the harsh radiation environment in space, causing a lot of SEU, the technology can be useful for future spacecraft projects and it is therefore tested on MIST.

7) **SICIS**: The Department of Integrated Devices and Circuits at KTH have developed a Silicon Carbide (SiC) semiconductor especially suited for harsh environments. The semiconductor has demonstrated the ability to operate at temperatures up to 500°C. The technology may be used as a general electronics building block for equipment operating under extreme conditions in future space missions, for example missions to Venus. On MIST a first test on the performance in space will be done.

8) **Camera**: Regarding the pictures for the exhibition at Tekniska Museet the only design constraints gained was the pointing requirement, i.e. to take pictures of Earth. Which camera system to use was not decided and the MIST team became responsible for choosing and implementing the camera. A survey of known contractors was performed and from this it was found that the Danish company GOMSpace had a pre-designed camera system as a Commercial off the Shelf (COTS) product ready to be integrated into a CubeSat. Norrköping Visualiseringcenter and the Scientific Visualization Group from the Department of Science and Technology at Linköping University in Sweden was invited to handle the software of the imaging. They proposed the use of their image compression and reconstruction software algorithms on top of the GOMSpace camera hardware.

### III. Project Requirements

From the project scopes the mission requirements could be formulated. These requirements constitute the first set of boundary conditions for the system design analysis. Together with the payload specific requirements these boundary conditions form the complete set of subsystem requirements.

#### A. CubeSat requirements

Any CubeSat needs to be constructed within the framework of the CubeSat design regulations set by the California Polytechnic State University [2]. These regulations are a set of mechanical, electrical and operational specifications that every CubeSat type spacecraft needs to meet. These specifications guarantee that the requirements of the CubeSat deployment system, the P-POD [2] are met. The P-POD is a dispenser that launches the satellite into orbit from the launch vehicle as part of the rocket’s secondary payload. As this deployment system is attached to the launch vehicle, the contents of the dispenser may pose unnecessary risk to all payloads if not properly designed.

The National Aeronautic and Space Administration, NASA, a provider of launch vehicles, have additional requirements which put limits on the outgassing of particles from the materials [6] for all payloads flying on their spacecraft. In order to keep as many launch opportunities as possible available to the MIST project, the NASA regulations on materials requirements should be followed throughout the project.

#### B. Mission Requirements

From a feasibility study done by the project manager, Sven Grahn, two reference design orbits were considered. Both are Sun Synchronous Orbits (SSO) at altitudes of approximately 640 km, denoted reference orbit one and two, where the second orbit is the reference orbit of the SEAM project.

The initial mission requirement is a design lifetime of one year but from an engineering point of view the spacecraft will have good chances of operating much longer. However, for a design lifetime longer than one year one has to take into consideration the effects of the prolonged exposure to radiation of the electronics, the number of discharge cycles of the batteries and the weathering of the solar panels. A longer lifetime would therefore greatly increase the requirements and complexity across all subsystems.

It was taken as an initial design specification that the project would use magnetic actuation for the attitude control and determination system (ACDS), which is accomplished by the use of magnetic torquer coils, i.e. magnetorquers. The motivations to use magnetorquers is that they are small and would not require additional space. The MIST spacecraft is small and if the control authority provided by the magnetorquers is
sufficient for the ACDS, it would remove the need for large and expensive reaction wheels.

In accordance with the Inter-Agency Space Debris Coordination Committee (IADC), which has laid out regulations on spacecraft lifetimes to mitigate the generation of space debris, the spacecraft must not remain in orbit for longer than 25 years following the end of operation.

IV. MISSION ANALYSIS

Using the orbital elements found in detail in the reference orbit applicable document [7], the reference orbit was modeled using AGI software’s Systems ToolKit (STK) [8]. In STK a simple model of the orbit was developed and using the built-in report generating function the average orbital period was determined to approximately 98 minutes during which 63 minutes is spent in sunlight and 35 minutes is spent in eclipse. The model was equipped with a ground station located at KTH and with this it was possible to compute the communications access times between the satellite and KTH. An orbital period of 98 minutes yields approximately 15 revolutions around Earth per 24 hours, out of which there are 4 good passes over KTH where each passage last on average 10-12 minutes.

In conjunction with STK and the Debris Assessment Software (DAS) from NASA [9], an orbit lifetime analysis was done for the spacecraft in accordance to the 25 year upper bound set for any spacecraft in orbit with an altitude between 160 and 2000 km also called a Low Earth Orbit (LEO) [10].

Following the orbit and lifetime analysis the process of gathering system specifications from the project payloads by writing a set of Interface Control Documents (ICD), was initiated. The purpose of these documents is to contain the physical interfaces connecting the payload to the satellite and specifying in detail the mechanical, electrical and data requirements of the payload. Further, the document contains a complete description of the experiment duty cycle. Combined these ICD generate the payload specific mission requirements. In a successive version of the ICD the MIST team supplies the specifications on the available structural mounting points, material requirements, pre-flight testing philosophy and the on-board computer data handling protocol.

In the design process the subsystem supplier GOMspace [5] was contacted as part of an initial feasibility study of the preliminary design so far. From the discussion with GOMspace during Match 2015 it was gathered that magnetically actuated satellites are difficult to control in the direction parallel to the magnetic field lines since the magnetic actuation is proportional to the cross product of the direction of the magnetic field and velocity vector [11]. In the case with MIST that means that in the roll direction, or rotation about the \( \hat{V} \) direction as shown in Fig. 1, the actuation strength is weaker in a wide zone north and south of the equator where the magnetic field vector lies very close to the spacecraft velocity vector. When the actuation is weaker a larger variation of the true attitude about this axis is expected, something that had to be taken into account when choosing the baseline flight attitude.

A. Attitude and Lifetime Analysis

Due to the geometry of the satellite there are three possible spacecraft attitudes, they are shown in Fig. 1.

Several of the experiments flying on MIST require to be pointed in a fixed direction in the orbital frame, some pointing in the \( \hat{U} \) direction towards Earth and some pointing in directions orthogonal to the \( \hat{U} \) direction. With these pointing requirements it is not possible to use a spinning attitude for flight stabilization. In the following attitude perturbation analysis, the three attitudes are considered separately and the results are gathered in Table III.

For the lifetime analysis the values from DAS and STK’s automatic calculations of orbit lifetimes was compared. DAS uses the orbital parameters found in the MIST reference orbit document [7], and the cross-section-to-mass ratio of the spacecraft. In the worst case for MIST the ratio is 0.0225, occurring when flying in the Tower or the Rolling pin attitude with solar panel wings faced in the direction of flight resulting in the maximum possible drag area. In the case when the satellite flies in the Arrow attitude with the wings perpendicular to the direction of flight the ratio is 0.0025 thus constituting the minimum drag area. The resulting lifetimes are compared to the values automatically generated in the STK model and the results can be found in Table I. From STK plots were generated on how the orbit decays over time, the results are found in Fig. 2 and 3.

<table>
<thead>
<tr>
<th>TABLE I: ORBIT LIFETIME ANALYSIS</th>
</tr>
</thead>
<tbody>
<tr>
<td>Attitude:</td>
</tr>
<tr>
<td>Debris Assessment Software</td>
</tr>
<tr>
<td>Satellite ToolKit</td>
</tr>
</tbody>
</table>
drag determined by the structural geometry and \( V \) is the mean cross-section area of the spacecraft. Density of the atmosphere at the altitude of flight, \( A \) is the power generation is found in Section V, and a brief discussion of the satellite after it is launched into orbit from the dispenser.

is not addressed in this thesis is the process of de-tumbling dipole perturbation and budget.

B. Attitude Perturbation Analysis

Since the structure geometry of the satellite was known, the first part in the attitude perturbation analysis was to determine the aerodynamic drag. The aerodynamic drag is proportional to the cross-sectional area orthogonal to the direction of flight. Following the aerodynamic perturbation the governing equations of the gravity gradient torque is derived and the section ends with a method used to compute the magnetic dipole perturbation and budget.

A general topic in the attitude perturbation analysis which is not addressed in this thesis is the process of de-tumbling the satellite after it is launched into orbit from the dispenser. An overview of the consequences of tumbling with respect to power generation is found in Section V, and a brief discussion of the de-tumbling sequence is found in the discussion Section VI.

1) Aerodynamic Torque: Even at an altitude of 640km there is still residual atmosphere. The constant drag through the atmosphere is what will cause the satellite orbit to eventually decay.

\[
F = \frac{1}{2} C_d \rho V^2 A \tag{1}
\]

In the equation of the aerodynamic drag (1) [12], \( \rho \) is the density of the atmosphere at the altitude of flight, \( A \) is the cross-section area of the spacecraft. \( C_d \) is the coefficient of drag determined by the structural geometry and \( V \) is the mean orbital velocity. For an orbit with a small eccentricity (\( e \ll 1 \% \)) like ours, the mean orbital velocity \( V \) is given by:

\[
V = \sqrt{\frac{GM_{Earth}}{a}}
\]

\( G \) is the gravitational constant \( \approx 6.673 \times 10^{-11} \ N \cdot m^2 / kg^2 \)

\( a \) is the orbit average radius \( \approx 7020 \ km \)

\( M_{Earth} \approx 5.972 \times 10^{24} \ kg \)

Thus the mean orbital velocity \( V \approx 7.5 \times 10^3 \ \frac{m}{s} \). Again looking at equation (1) the coefficient of drag, \( C_d \), was determined to have a value of \( \approx 2 \) in a study by Li Qiao et.al [13]. The density of the atmosphere is assumed at a yearly average value of \( 4.95 \times 10^{-11} \ kg/m^3 \) [14]. The cross-sectional drag area of the satellite is 0.01 m\(^2\) in the best case with the Arrow attitude and 0.09 m\(^3\) in the worst case with the Tower or Rolling pin attitude. The result can be found in Table III.

2) Gravity Gradient Torque: Next the gravity gradient torque is calculated. This perturbation arises from the inverse square dependency of Newtonian gravity, where the mass of the satellite closer to Earth will feel a stronger gravitational pull. The Tower configuration is a gravity gradient stable attitude in the sense that any inhomogeneous mass distribution will act as the keel on a boat to stabilize the satellite.

In deriving the governing equation of the gravity gradient torque the differential form of Newtonian gravity is used:

\[
dF = -\mu \frac{\hat{r}}{r^3} dm \tag{2}
\]

Where \( \mu = GM_{Earth}, R \) is the distance vector from the center of mass (CM) of Earth to the satellite CM, \( \rho \) is the distance vector from the satellite CM to any mass in the satellite such that \( r = \rho + R \) is the total distance from \( CM_{Earth} \) to a differential mass dm.

This mass causes a moment about the CM of the satellite. Using \( \hat{r} = \frac{R}{|R|} \):

\[
dM = \rho \times dF = -\rho \times \mu \frac{(R + \rho)}{r^3} \ dm \tag{3}
\]

\[
dM = -\rho \times \mu \frac{R}{r^3} \ dm \tag{4}
\]

Now looking at \( r^{-3} \) and as \( \rho \ll R \):

\[
r^{-2} = R^2(1 + \frac{(\rho \cdot \rho)}{R^2} + 2 \frac{R \cdot \rho}{R^2}) \tag{5}
\]

\[
\frac{1}{r^3} = \frac{1}{R^3}(1 + \frac{(\rho \cdot \rho)}{R^2} + 2 \frac{R \cdot \rho}{R^2}) \tag{6}
\]

Expanding equation (6) with a generalization of the binomial theorem, the multinomial theorem [15] and neglecting terms of second order and higher:

\[
\frac{1}{r^3} = \frac{1}{R^3}(1 - 3 \frac{R \cdot \rho}{R^2}) \tag{7}
\]

Now it is possible to integrate equation (4):

\[
M = -\mu \int_M \left(1 - 3 \frac{R \cdot \rho}{R^2}\right) \rho \times R \ dm \tag{8}
\]
If it is assumed that CM \( \approx \) geometrical center then there is no moment at the CM, and:

\[
\int_M \rho \times \mathbf{R} \, dm = 0
\]

Thus the expression for the gravity gradient torque are left:

\[
M = -3 \frac{\mu}{R^3} \int_M \left( \frac{\mathbf{R} \cdot \rho}{R^2} \right) \rho \times \mathbf{R} \, dm
\]

Looking at the cross product in equation (9) it is found that for an arbitrary direction \( \mathbf{j} \):

\[
M_j = -3 \frac{\mu}{R^3} \int_M \left( \frac{\mathbf{R} \cdot \rho}{R^2} \right)(\rho \times \mathbf{R}) \cdot \mathbf{j} \, dm
\]

\[
M_z = -3 \frac{\mu}{R^3} \int_M C_R(\rho_z\rho_x - \rho_x\rho_y) \, dm
\]

In equation (11) the \( z \)-component of the cross product and the coefficient \( C_R \) is the \( z \)-component of \( \frac{\mathbf{R} \times \rho}{R^2} \), following equation (11) and the definition of inertia:

\[
I_{ij} \equiv -\int_m \rho_i \rho_j \, dm
\]

The final result is that the gravity gradient torque about any major axis is proportional to the difference between the moments of inertia. At this stage in the project it is not possible to do any measurements of the inertial properties of the satellite. However, if the analysis is continued under the assumption of homogeneous mass distribution, standard inertial properties of known geometries such as the cuboid can be used. The resulting torque was calculated using the standard inertia of the cuboid and the the coefficients \( C_R \) as derived in the Attitude Control and Determination booklet by Dr. Andrew Ketsdsever [16]. If the deviation angle of the satellite frame to the orbital frame is assumed to be 45 degrees (worst case) and using the inertial properties of the cuboid the resulting gravity gradient torque can be calculated using MATLAB. The MATLAB code is found in the appendix. Magnetic moment is measured in \( \frac{Nm}{m} \), as such the magnetic moment can be converted to mechanical torque by scaling the result with the value of the average magnetic field strength at an altitude of 640 km. The results can be found in Table III.

### C. Magnetic Dipole Moment

The final part of the attitude perturbation analysis was to determine the magnetic dipole moment the satellite experiences while orbiting through the Earth’s magnetic field. The attitude perturbation torque from the satellites magnetic dipole moment is given by:

\[
\tau = \bar{m} \times \bar{B}
\]

Where the perturbation torque \( \tau \) is calculated in the worst case when the direction of the dipole moment, \( \bar{m} \) is orthogonal to the magnetic field, \( \bar{B} \). Thus given the dipole moment the contribution to the attitude perturbation can be calculated.

A magnetic dipole budget was calculated using parts of the SEAM projects magnetic dipole budget [17] for the structural components of the MIST satellite in conjunction with the dipole moment of the MIST payload stack which had to be calculated separately since the MIST and SEAM satellites carry different scientific payloads. A crude estimate of the magnetic dipole moment was done by approximating each payload as a small disturbing magnetorquer, i.e. if the dipole moment would be caused by a current-loop.

The magnetic dipole moment [18] is thus defined as:

\[
\bar{m} \equiv I \int \bar{d}a = IA
\]

Where \( I \) is the current passing through a loop enclosing the area \( A \). From equation (12) we can calculate the magnetic moment of the payloads given the current passing through a loop with the payload area. If we assume a worst case current loop along the sides of the circuit board, the loop encloses an area of 0.001 \( m^2 \), since the structural limit is 10x10 cm\(^2\). An estimate of the current passing through the loop is given by:

\[
I = \frac{P}{U}
\]

The total power consumption, \( P_{tot} \approx 12W \) is given by summing the payload power requirements from Table VI, Section VI-A. Further, the GOMSpace P31us battery board, a candidate battery pack which is used in this exercise, provides output connectors for 3.3V and 5.0V aside from the raw battery voltage. Assuming that all the experiments are running on full power and if further it is assumed that the power is split over the two output voltage connectors we get the total current output as:

\[
I_{tot} = I_{3.3V} + I_{5.0V} \approx 3A
\]

With an estimate on the current the dipole moment could be calculated given the total area of the payload stack. So if the current loops are 0.001m\(^2\) and there are 8 payloads, the combined area is 0.008m\(^2\).

Thus we can calculate the total magnetic dipole moment as:

\[
m_{dip} = 3 \times 0.008 \, Am^2 = 24 \, mAm^2
\]

This is a very high estimate for the magnetic dipole moment of the experiments and consequently the entire satellite. Therefore other approaches to this problem were also examined. The allotment method in Table II could be compared to measurements of the magnetic dipole moment on actual 3U Cubesats. From a technical memo from GOMSpace to the MIST project in March 2015 it was gathered that: “... have just completed measurements of GOMX-3 at ESA MCF and the magnetic dipole moment was measured to 30mAm\(^2\). This is all concentrated in the battery pack, other parts we have measured to below 2 mAm\(^2\) ...” The SAFT battery cell (MP174565) was specifically selected for the SEAM project to minimize the total magnetic dipole moment, which is achieved partly by the battery packs non-magnetic aluminum casing. From the correspondence with GOMSpace in March 2015 and at the SEAM projects critical design review on 12 May 2015
GOMspace indicated that they estimated a magnetic dipole moment of < 2 mAm$^2$ for the SEAM battery pack. Combining the data from GOMX-3 and the estimate of the magnetic dipole moments from the SAFT cell the worst case magnetic dipole moment could be as low as 4 mAm$^2$.

Further, there was no a priori reason to assume that the experiments needs to be given a higher magnetic dipole moment allotment than for example the Nanomind boards, i.e. about 0.85 mAm$^2$ per board due to the fact that some of the MIST experiments are much smaller than a 10x10 cm$^2$ motherboard. However, since some of the payloads are from non space-specific companies, there is neither any reason to assume they have designed towards magnetic cleanliness. As such, some components might have residual magnetization which could induce a significant magnetic dipole moment. The final payload dipole allotment was set as a compromise between the high upper bound given by equation (13) and equating the magnetic dipole moment of each payload with a Nanomind board, weighted towards the Nanomind board approximation. This compromise allows for some tolerance with regards to eventual magnetization of the payload components. With this estimate the dipole allotment for the payload stack is calculated to 7.5 mAm$^2$ and in conjunction with the SEAM allotments from Table II the worst case magnetic dipole moment for the MIST satellite was calculated as 18 mAm$^2$. The resulting magnetic attitude perturbation torque is shown in Table III.

### TABLE II
**MIST Dipole Moment Allotment**

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Dipole Allotment [mAm$^2$]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Electrical Power System</td>
<td>0.71</td>
</tr>
<tr>
<td>2x fold out solar panels</td>
<td>0.71</td>
</tr>
<tr>
<td>8x 1U solar panel sides</td>
<td>1.84</td>
</tr>
<tr>
<td>GOMspace P310us battery pack</td>
<td>2</td>
</tr>
<tr>
<td>Communication Unit</td>
<td>0.40</td>
</tr>
<tr>
<td>Antenna array</td>
<td>0.40</td>
</tr>
<tr>
<td>Antenna board</td>
<td>0.43</td>
</tr>
<tr>
<td>Structure</td>
<td>3.52</td>
</tr>
<tr>
<td>3U CubeSat structure</td>
<td>3.52</td>
</tr>
<tr>
<td>Cables and Wires</td>
<td>1.78</td>
</tr>
<tr>
<td>OBC</td>
<td>0.43</td>
</tr>
<tr>
<td>Nanomind motherboard</td>
<td>0.43</td>
</tr>
<tr>
<td>Nanomind daughterboard</td>
<td>0.40</td>
</tr>
<tr>
<td>Net magnetic dipole moment</td>
<td>11.5</td>
</tr>
</tbody>
</table>

In summary the residual magnetic dipole moment of the satellite could be expected to lie in the range 4-20 mAm$^2$. As shown in Table III the actuation of the GOMspace torquer [19] is within an order of magnitude of the magnetic dipole moment of the satellite as calculated in the perturbation analysis. As such it was chosen to design toward the ISIS magnetorquer [20].

### TABLE III
**ATTITUDE PERTURBATION AND CONTROL TORQUES**

<table>
<thead>
<tr>
<th>Attitude</th>
<th>Arrow [Nm]</th>
<th>Tower [Nm]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aerodynamic</td>
<td>6.95$\times$10$^{-9}$</td>
<td>2.22$\times$10$^{-8}$</td>
</tr>
<tr>
<td>Gravity Gradient</td>
<td>5.42$\times$10$^{-8}$</td>
<td>5.42$\times$10$^{-8}$</td>
</tr>
<tr>
<td>Magnetic</td>
<td>0.57$\times$10$^{-6}$</td>
<td>-</td>
</tr>
<tr>
<td>GOMspace</td>
<td>1.14$\times$10$^{-6}$</td>
<td>-</td>
</tr>
<tr>
<td>ISIS</td>
<td>6$\times$10$^{-6}$</td>
<td>-</td>
</tr>
</tbody>
</table>

### D. Measuring Magnetic Moment

Piezo Motor proposed to use a magnetic sensor with unknown dipole moment for their payload Piezo LEGS. To allow the sensor to fly on MIST the dipole moment of the magnet had to be investigated and show that it would not affect the dipole budget too much. From a guide in the booklet "A Pedestrians Approach to Magnetic Cleanliness" [21], two independent measurements was set up and conducted on the magnet. In the first measurement an Astastic Magnetometer (AM) was used and set up as described in the guide with the following Fig. 4.

The deviation distance $R_s$ was measured using a standard reference magnet with a dipole moment of 2 mAm$^2$. Then the direction of the Piezo magnet with the strongest field was identified, i.e. the direction which gave the largest $R_s$, and the distance from the AM where the magnet gave the same result as the reference magnet was investigated. From this the magnetic dipole moment of the Piezo magnet, $m_t$ was calculated:

$$m_t = m_s \left( \frac{R_t}{R_s} \right)^4$$

Following the measurement using the AM a digital magnetometer shielded with a Mu-metal cover was used. The digital magnetometer calculated $m_t = 25$ mAm$^2$, while using the AM and calculating $m_t$ from $R_t = 30$ cm and $R_s = 16$ cm gave the result $m_t = 24$ mAm$^2$. With both the calculated and measured dipole moment of the magnet it is clear that the dipole moment is around the same order of magnitude as the allotment of the payloads and the GOMspace torquer moment [5]. It was concluded that this component must be excluded from the payload in order to assure the control authority of the ACDS.

### V. SPACECRAFT DESIGN

The following section is a presentation of the methods used in the solar array configuration design to determine the optimal flight attitude. The section is concluded with the power budget.

#### A. Solar Panel Configuration Constraints

When building satellites the power consumption does not decrease in direct proportion to the mass or volume of the satellite. This is due to the fact that most subsystems used on large satellites are also necessary for smaller satellites. This requirement places great demands on the power systems efficiency.

As MIST is a small satellite, it has a limited external area to mount solar panels on and therefore a limited capacity for
power generation. One way to increase the size of the solar panel array is to have deployable solar panels that fold out when the satellite is launched into orbit.

1) Mounting of deployable solar panels: The strict regulations from the CubeSat launch vehicles impose requirements on how to mount the deployable wings with solar panels [2]. When folded they have to be thin enough to keep the dimensions of the satellite within the given restrictions of the P-POD. After an investigation of deployable solar panels available on the market it was concluded that the even though there are twice foldable wings, which would yield even more surface area, the wings most likely to be used on MIST are to be folded once. This is mainly due to the increased structural complexity of having twice foldable wings. Both sides of the wings can have solar panels, making them able to generate power on both sides of the satellite.

There are two ways to fold the wings to the body of MIST. Either the hinges are placed on the long side or on the short side of the wing, in other words they are mounted on the sides or on the top of the satellite. The antenna array that probably will be used is most likely going to be placed at one of the top sides of the structure and the antenna mounting plate have hinges placed on all four edges. Therefore the hinges for the wings are not to be placed there since such a design would complicate the deployment of both wings and antenna. To avoid the need of buying a custom designed antenna or solar array mount it was decided that the hinges for the wings are to be placed on the long side of the satellite as shown in Fig. 5.

With the choice of wings and structural mounting point a power generation analysis of this solar array configuration was conducted. A detailed presentation of the methods used in the analysis is described throughout the next sections.

2) Requirements from experiments: The following experiments have specific requirements on the outer structure and solar panel configuration:

- CubeProp - The experiment should be mounted at one end of the satellite and the bottom cannot be covered by solar panels since it would block the nozzles for propulsion.
- CUBES - The experiment cannot be covered by a solar panel or shadowed by a wing, it would also like to avoid the shadow from the antenna. Prefers to look in the \( \hat{U} \)-direction but can accept to look in any other direction as well.
- Ratex-J - The experiment cannot be covered by a solar panel or shadowed by a wing and would also like to avoid the shadow from the antenna. It needs to look in any direction in the orbital \( \hat{W} - \hat{V} \) plane.
- Camera - The camera should take pictures of the Earth and should therefore look in the \(-U\) direction. If possible it should not be placed next to a deployable solar panel since it may limit the field of view.

3) Requirements from the internal structure: The solar panels made for a Cubesat usually has the same size as the side of a 1U Cubesat. Therefore the surface of MIST can be described in terms of 1U side-panels when regarding the mounting of solar panels. The surface area of a 3U CubeSat
like MIST is equivalent to 14 side-panels with an additional 12 side-panels from the deployable wings.

One of the faces at the end of the satellite will be partly covered by the antenna since the antenna mounting plate is the size of 1U with a hole cut in the middle. The hole is designed to be big enough to fit a solar panel. Our CAD-model (developed by the projects structural engineer Gábor Felscuti) showed that the Ratex-J experiment could not fit within this window without changing the design of their experiment, a task that would involve too much work and thus was not considered as an alternative solution. On the other hand the CAD-model showed that both CUBES and the camera fit in the window of the antenna mounting plate at the same time.

The Ratex-J experiment needs a window size of at least one half 1U side-panel. Solar panels with the size of one half 1U side exist but it is difficult to place the experiment next to a verge between the solar panels from a structural point of view. Thus, it was given a window the size of 1U as a design choice in the calculations to make sure that it would be possible to construct.

B. Variation of the design

The preliminary design choice of deployable solar panels only specifies the frame of the satellite but not specifically which sides that are covered by solar panels. The frame has an area of 26 squares with the same size as the side panel of a 1U Cubesat, therefore it can at most carry 26 solar panels. In total three solar panels have to be removed following the previously described requirements to avoid obscuring the field of view of the camera and the experiments CubeProp, CUBES and Ratex-J. CUBES and the camera are placed next to each other, looking out through the same window to minimize the number of removed solar panels.

For each attitude the normal vector of the wings can be pointed in four different directions and for each direction the solar panels can be moved to different locations on the frame. The total number of possible configurations for each attitude is described below.

Rolling pin
Wings normal in the direction of \( \hat{U} \) or \(-\hat{U}\):
- 2 possible positions for CubeProp
- 0 possible positions for Ratex-J
- 1 possible position for the camera and CUBES

Wings normal in the direction of \( \hat{V} \) or \(-\hat{V}\):
- 2 possible positions for CubeProp
- 2 possible positions for Ratex-J
- 1 possible position for the camera and CUBES

The total number of possible configurations:
\[ 2 \times (2 \times 0 \times 1) + 2 \times (2 \times 2 \times 1) = 8 \quad (14) \]

A simulation of the illumination showed that the small end of the satellite pointing in the direction of \(-\hat{W}\) is never lit by the Sun. When choosing the position of CubeProp the side facing \(\hat{W}\) and \(-\hat{W}\) are the only alternatives since CubeProp has to be placed at one end of the satellite. Since the side facing \(-\hat{W}\) is never lit by the Sun the cases when CubeProp is placed there will generate more power than the cases when it is placed on the illuminated side. Therefore only one of the two possible ways of positioning the CubeProp is interesting and of the eight possible cases, only four were investigated.

The total number of investigated configurations:
\[ 2 \times (2 \times 0 \times 1) + 2 \times (2 \times 2 \times 1) = 4 \quad (15) \]

Arrow
Wings normal in the direction of \( u \) or \(-u\):
- 2 possible positions for CubeProp
- 0 possible positions for Ratex-J
- 1 possible position for the camera and CUBES

Wings normal in the direction of \( w \) or \(-w\):
- 2 possible positions for CubeProp
- 2 possible positions for Ratex-J
- 1 possible position for the camera and CUBES

The total number of possible configurations:
\[ 2 \times (2 \times 0 \times 1) + 2 \times (2 \times 2 \times 1) = 8 \quad (16) \]

Tower
Wings normal in the direction of \( w \) or \(-w\):
- 1 possible positions for CubeProp
- 2 possible positions for Ratex-J
- 1 possible position for the camera and CUBES

Wings normal in the direction of \( v \) or \(-v\):
- 1 possible positions for CubeProp
- 2 possible positions for Ratex-J
- 1 possible position for the camera and CUBES

The total number of possible configurations:
\[ 2 \times (1 \times 2 \times 1) + 2 \times (1 \times 2 \times 1) = 8 \quad (17) \]

In total, considering all three attitudes with all their configurations, 20 different designs were investigated.

C. Calculations

To calculate the power generated by a solar panel configuration the illumination of the satellite needs to be known. This was simulated with software available through the project manager Sven Grahn. The simulation generates illumination factors \( f \) for all sides of the satellite at any given time and orbit. The illumination factors are given as the ratio of incoming power to the maximum power, depending on the angle of incidence of the light to the solar panel. The maximum power is measured when the solar panel is perpendicular to the incidence light from the Sun. The power generation of the solar panel is known by the manufacturer and in the calculations the properties of the solar panels were retrieved from GOMSpace’s NanoPower P110 series solar panel [19]. The minimum value of the generated power (here
The MIST satellite was used to predict worst case conditions.

\[ p = 2.27 \text{ W} \]

1) Illumination factors: The illumination factor \( f \) consists of the illumination from the Sun \( f_s \) and the factor \( f_a \) corresponding to the contribution from the albedo effect. The albedo effect is the phenomenon when visible sunlight is reflected from the surface of Earth back to space. This radiation can contribute to the power generation from solar panels facing Earth. All vectors are expressed in the orbital frame \( \hat{u}, \hat{v}, \hat{w} \).

\[ f = f_s + f_a \quad (18) \]

The illumination factor \( f_s \), corresponding to the angle of incidence of light from the Sun, is calculated for each side of the satellite:

- \( f_s \) on the \(+u\) side
  \[ f_s \text{ on the } +u \text{ side} = \begin{cases} \hat{u} \cdot \hat{p} & \text{if } \hat{u} \cdot \hat{p} \geq 0 \\ 0 & \text{otherwise} \end{cases} \quad (19) \]

- \( f_s \) on the \(-u\) side
  \[ f_s \text{ on the } -u \text{ side} = \begin{cases} -\hat{u} \cdot \hat{p} & \text{if } -\hat{u} \cdot \hat{p} \geq 0 \\ 0 & \text{otherwise} \end{cases} \quad (20) \]

- \( f_s \) on the \(+v\) side
  \[ f_s \text{ on the } +v \text{ side} = \begin{cases} \hat{v} \cdot \hat{p} & \text{if } \hat{v} \cdot \hat{p} \geq 0 \\ 0 & \text{otherwise} \end{cases} \quad (21) \]

- \( f_s \) on the \(-v\) side
  \[ f_s \text{ on the } -v \text{ side} = \begin{cases} -\hat{v} \cdot \hat{p} & \text{if } -\hat{v} \cdot \hat{p} \geq 0 \\ 0 & \text{otherwise} \end{cases} \quad (22) \]

- \( f_s \) on the \(+w\) side
  \[ f_s \text{ on the } +w \text{ side} = \begin{cases} \hat{w} \cdot \hat{p} & \text{if } \hat{w} \cdot \hat{p} \geq 0 \\ 0 & \text{otherwise} \end{cases} \quad (23) \]

- \( f_s \) on the \(-w\) side
  \[ f_s \text{ on the } -w \text{ side} = \begin{cases} -\hat{w} \cdot \hat{p} & \text{if } -\hat{w} \cdot \hat{p} \geq 0 \\ 0 & \text{otherwise} \end{cases} \quad (24) \]

The illumination factors from the albedo effect, \( f_a \) are calculated as follows:

Vectors are expressed in the orbital frame \( \hat{u}, \hat{v}, \hat{w} \) as shown in relation to the Sun and Earth in Fig. 6.

\[ \vec{r} = (R_0 + h)\hat{u} = (R_0 + h, 0, 0) \quad (25) \]

\( R_0 = \text{Earth’s mean radius} = 6378,145 \text{ km} \)

\( h = \text{the altitude of the satellite} \)

From the satellite’s point of view as shown in Fig 7:

\[ \vec{b} = (R_0 \cos \theta, R_0 \sin \theta \cos \varphi, R_0 \sin \theta \sin \varphi) \quad (26) \]

\[ \hat{b} = (\cos \theta, \sin \theta \cos \varphi, \sin \theta \sin \varphi) \quad (27) \]

\[ \vec{r} + \vec{a} = \vec{b} \Rightarrow \vec{a} = \vec{b} - \vec{r} \quad (28) \]

Fig. 6. The position of the satellite relative to the Sun and Earth’s surface.

Fig. 7. A surface element of Earth as seen from the satellite.

Fig. 8. The reflected sunlight from a surface element on Earth illuminating the satellite.

Equations 25, 26 and 28 give:

\[ \vec{a} = (R_0 \cos \theta - R_0 - h, R_0 \sin \theta \cos \varphi, R_0 \sin \theta \sin \varphi) \quad (29) \]
The reflected radiance \( (Wm^{-2}sr^{-1}) \) at the Earth:
\[
L = \frac{E_0 \rho \cos \alpha_1}{\pi}
\]
\( \rho = \text{Earth’s albedo} \approx 0.3; \)

The reflected radiant intensity \( (Wsr^{-1}) \):
\[
I = \frac{E_0 \rho \cos \alpha_1}{\pi} dS
\]
\( dS = R_0^2 \sin \theta d\theta d\phi \)

The irradiance \( (Wm^{-2}) \) at the point corresponding to the position of the satellite:
\[
E_1 = \frac{E_0 \rho \cos \alpha_1 \cos \alpha_2 \cos \alpha_3}{\pi a^2} dS
\]

The irradiance \( (Wm^{-2}) \) on the appropriate side of the satellite:
\[
E_2 = \frac{E_0 \rho \cos \alpha_1 \cos \alpha_2 \cos \alpha_3}{\pi a^2} dS
\]

The power \( (W) \) generated by a solar panel:
\[
P_s = \int_D E_2 dS = \int_D \frac{E_0 \rho \cos \alpha_1 \cos \alpha_2 \cos \alpha_3}{\pi a^2} dS
\]
\( P_s \approx \sum_i E_{2i} \eta dA_s \) \( i \) for all sides of the satellite in one minute intervals during one orbit of 98 minutes. First the simulation of \( f_s \) runs and creates a matrix \( F_s \) with 6 columns and 98 rows, one column for each side of the satellite and the value of \( f_s \) on that side for each minute of the simulation. An equivalent matrix \( F_a \) with \( f_a \) is also created. For each design of the satellite this simulation is done for one orbit each month during a year in both reference orbit one and two.

\[
F_s = (\bar{F}_{a,u}, \bar{F}_{a,-u}, \bar{F}_{a,v}, \bar{F}_{a,-v}, \bar{F}_{a,w}, \bar{F}_{a,-w})
\]

\[
F_a = (\bar{F}_{a,u}, \bar{F}_{a,-u}, \bar{F}_{a,v}, \bar{F}_{a,-v}, \bar{F}_{a,w}, \bar{F}_{a,-w})
\]

The total amount of generated power \( P \) is computed by multiplying the matrix \( (\bar{F}_s + \bar{F}_a) \) with a vector \( \bar{n} \), containing the number of solar panels on each side, and the scalar \( p \) for the power generation of one solar panel. In other words \( P \) is given as the power generated by one solar panel multiplied by the amount of solar panels on each side scaled with the illumination factor on each panel.

\[
P = p (\bar{F}_s + \bar{F}_a) \bar{n}
\]

However, this formula does not take into account that some solar panels will be shadowed by the deployable wings. When the sunlight hits the side of the satellite opposite to the side where the wings are mounted it illuminates three different surfaces consisting of the satellite main body and the back side of both wings. When this side is illuminated the closest surface, the one on the main body, will never be shadowed while the surfaces on the wings may be shadowed by the body depending on the angle of incidence of the light. Now if the satellite is rotated 90 degrees, still showing you the same side as in Fig. 9 and 10, the shadows on the two surfaces on the back side of the wings will change assuming the Sun remains fixed. In other words, looking at the bottom of the wings, the orientation in that plane has to be taken into account when calculating the shadowing. With 3 surfaces and 2 cases of orientation (rotated 90 degrees), there are 6 scenarios in total for this side of the satellite.
The 6 scenarios are indexed as following:

- The first character is a letter denoting the normal vector (with sign if negative) of the side, same for all 6 scenarios e.g. the letter U in Fig. 9.
- The second character is a number indexing the three surfaces in the direction of the wings. The direction of the wings is defined as the direction of the positive vector perpendicular to both the normal vector of the surfaces and the verge between them. I.e. increasing in the W direction in Fig. 9.
- The third character is a letter denoting the direction the wings.

When part of solar panel is shadowed the performance can be significantly reduced due to both less exposure to sunlight and the fact that shaded solar cells require energy to become conductive [22]. How much energy that is lost from shading depends on the design of the solar panel and the circuits connecting the solar cells. Since it is not decided which solar panels are to be used on MIST it is difficult to predict the impact shading will have on the generated power. It is assumed that a partially shadowed solar panel will not contribute with any power generation since this constitutes the worst possible case and simplifies the calculations.

As previously mentioned the matrix \( F_s \) consist of one vector for each side of the satellite, every vector containing the solar illumination factor \( f_s \) for that side every minute of one revolution around Earth. An expanded matrix \( F_{se} \) is now constructed from \( F_s \) by, for every vector in \( F_s \), creating 6 new vectors corresponding to the 6 scenarios of each side.

Let \( X \) denote an arbitrary side of the satellite (u, -u, v, -v, w or -w) with surfaces in the plane with vectors \( \vec{y} \) and \( \vec{z} \). The 6 scenarios for this side is:

- \( X1y \)
- \( X2y \)
- \( X3y \)
- \( X1z \)
- \( X2z \)
- \( X3z \)

The vector for each scenario is calculated from the column vectors in \( F_s \):

\[
X1y_i = \begin{cases} 
F_{xi} & \text{if } F_{yi} > 0 \\
0 & \text{otherwise} 
\end{cases} 
\]  
(51)

\[
X2y_i = F_{xi} 
\]  
(52)

\[
X3y_i = \begin{cases} 
F_{xi} & \text{if } F_{zi} > 0 \\
0 & \text{otherwise} 
\end{cases} 
\]  
(53)

\[
X1z_i = \begin{cases} 
F_{xi} & \text{if } F_{zi} > 0 \\
0 & \text{otherwise} 
\end{cases} 
\]  
(54)

\[
X2z_i = F_{xi} 
\]  
(55)

\[
X3z_i = \begin{cases} 
F_{xi} & \text{if } F_{zi} > 0 \\
0 & \text{otherwise} 
\end{cases} 
\]  
(56)

When this calculation is done for all sides of the satellite an expanded matrix, \( F_{se} \), with 36 vectors has been created from \( F_s \):

\[
F_{s,u} \Rightarrow \{ F_{s,U1v}, F_{s,U2v}, F_{s,U3v}, F_{s,U1w}, F_{s,U2w} and F_{s,U3w} \} 
\]  
(57)

\[
F_{s,-u} \Rightarrow \{ F_{s,-U1v}, F_{s,-U2v}, F_{s,-U3v}, F_{s,-U1w}, F_{s,-U2w} and F_{s,-U3w} \} 
\]  
(58)
P1. MIST SATELLITE

F with regards to the shadows but to complete the calculations for direction to the source of light at a fixed time since this is the method used.

account when calculating the illumination for each design, that can be shadowed with 0.5 gives us the same result and the wings are oriented in the field of view and limitation of the visible part of the field of view is halved (equivalent to a quarter of a sphere). Since their field of view is limited by the body of the satellite to only 90 degrees side, the field of view for the two lower surfaces at the side may also happen, causing the estimated illumination to be higher than in reality. It is assumed that the over- and underestimates in the scaling of f_a are equal and expected to compensate for one another.

The new matrix F_s can be used to calculate the illumination with regards to the shadows but to complete the calculations by including the albedo effect, F_a has to be modified as well.

Unlike in F_s, the elements in F_a do not have a fixed direction to the source of light at a fixed time since f_a is given by an integral over the visible surface of Earth. Every element in F_a is therefore the sum of the intensity from light in all visible directions at that time and consequently shadows can occur in many directions.

The shadows from the albedo effect cannot be deduced given only the information in F_s. Instead of calculating the shadow from the wings the illumination factors for the surfaces are scaled in relation with the limitation in field of view the wings impose. Initially every side of the satellite has a 180 degrees field of view that can be illustrated by half a sphere. When observing the 6 scenarios previously described for each side, the field of view for the two lower surfaces at the side is limited by the body of the satellite to only 90 degrees (equivalent to a quarter of a sphere). Since their field of view is halved f_a is assumed to be halved for those surfaces.

Depending on the orientation of the satellite the limitation of the field of view and limitation of the visible part of Earth are not necessarily the same. It is not the same when the wings are oriented in the u direction, in this case the upper wing (the surface indexed as number 3) can only see the sky while the lower wing (the surface indexed as number 1) can still see the same area of Earth as that side before the mounting of wings. In this case the upper wing should be scaled with 0 and the lower wing should be scaled with 1. On the other hand, using the model when both sides are scaled with 0.5 gives us the same result since \(1f_a + f_a + 0f_a = 0.5f_a + 1f_a + 0.5f_a\). Since scaling all sides that can be shadowed with 0.5 gives us the same result and minimizes the number of properties that has to be taken into account when calculating the illumination for each design, this is the method used.

Surfaces indexed as 1 or 3 = 0.5
Surfaces indexed as 2 = 1

Scaling factors for f_a and each surface:

\[
\begin{align*}
\{ U1v, U3v, U1w, U3w, -U1v, -U3v, -U1w, -U3w, V1u, V3u, V1w, V3w, -V1u, -V3u, -V1w, -V3w, W1u, W3u, W1v, W3v, -W1u, -W3u, -W1v, -W3v \} &= 0.5
\end{align*}
\]

The matrix F_a in expanded in the same manner as F_s, every row vector becoming 6 new vectors corresponding to the scenarios of that side of the satellite. The vectors in the new expanded matrix F_s are scaled as described above.

Note: f_a = 0 is always true for the u side of the satellite since it can never see the Earth.

In the mathematical model that is used for calculating f_a, Earth’s albedo is approximated with the average of 30% [23]. That means every surface element of Earth is assumed to reflect the same amount of light even though some areas (for example clouds, oceans and glaciers) actually reflect more light than others. Even if f_a is calculated with the average albedo it still takes to account the fact that some parts of the visible Earth will rest in shadow where no light is reflected.

When f_a is scaled with the size of the visual field from that side of the satellite, this fact is neglected. When the visual field from a surface is halved the illumination factor is assumed to be halved even if it in reality may be unchanged. That may happen if the Earth’s surface that is obscured by the wings is in shadow and thus the wings impose no loss in illumination from the albedo effect in that case. The opposite alternative when the obscured surface is the only source of light may also happen, causing the estimated illumination to be higher than in reality. It is assumed that the over- and underestimates in the scaling of f_a are equal and expected to compensate for one another.

2) Solar panel configuration vector: Now the only thing missing to complete the calculation of illumination with respect to shadowing, is the vector describing the solar panel configuration. First, every side of the satellite is described in a satellite fixed coordinate system (\(\hat{x}, \hat{y}, \hat{z}\) which is translated into the orbital fixed coordinate system (\(\hat{a}, \hat{v}, \hat{w}\)).

The vector in the satellite coordinate system contains 8 elements, one for each surface of the satellite as described by Fig. 11. The elements of the vector are the number of
solar panels the size of 1U on that side of the satellite. For example, if the Sun is shining from above in Fig 11, then the +X element in the vector would have the value 9 assuming that the whole area of the side is covered with solar panels.

When visualizing the satellite in orbit the sides of the satellites have to be translated to the 36 different surfaces theoretically possible for the illumination factors adjusted for shadowing. Those 36 surfaces correspond to the Sun hitting the 3 surfaces on the bottom of the wings no matter which side of the satellite that is investigated, i.e. the bottom of the wings points in all directions simultaneously. In reality, only one side of the satellite will be the backside of the wings and in total it has no more than 8 different surfaces, the other 28 elements will be zeros.

The following rules define the translation of the vector in the satellite fixed coordinate system (\(\vec{x}, \vec{y}, \vec{z}\)) to a vector in the orbital fixed coordinate system (\(\vec{u}, \vec{v}, \vec{w}\)):

Note: Elements in the vector for the orbital fixed coordinate system is named X1y, X2y, X3y, X1z, X2z and X3z where X denote an arbitrary side of the satellite with surfaces in the plane with vectors \(\vec{y}\) and \(\vec{z}\).

- The relation between the sides of the satellite has to be conserved since the form of the satellite should remain the same.
- The translation of the normal vector for the sides are done with respect to the flight configuration that is investigated, the side that should be in front is translated to the \(\vec{V}\)-direction etc.
- Z, -Z, X and -X2 in the satellite frame are indexed as number 2 in the orbital frame since they cannot be shadowed by the wings.
- -X1 and -X3 in the satellite frame are indexed at 1 and 3 in the orbital frame, not necessary the same number as in the satellite frame since the numbers can alter depending on the flight configuration.
- Y and -Y in the satellite frame are indexed as 1 or 3 in the orbital frame. This is due to the fact that the shadow will fall on the surface in the same way as if it was the backside of one wing and the real wing corresponds to an imagined main body of the satellite.

The vector in the orbital frame created with regards to these rules in named \(\vec{n}_o\). Equation 50 can now be expressed as:

\[
P = p(F_{se} + F_{ae})\vec{n}_o
\]  
\[\text{(63)}\]

\(F_{se}\) is the matrix with \(f_s\) expanded to cover shadowing

\(F_{ae}\) is the matrix with \(f_a\) expanded to cover shadowing

\(\vec{n}_o\) is the solar panel configuration vector

\(p\) is the generated power per solar panel = 2.27 W

In other words, given \(p\), \(F_s\) and \(F_a\) it is possible to compute the generated power by constructing \(\vec{n}_o\) and transforming the matrices to \(F_{se}\) and \(F_{ae}\). The result is a vector describing the power output in watts every minute in the orbit described by the input \(F_s\) and \(F_a\). The elements in the output vector is summed and divided by 60 to give the number of watt-hours that is generated during one revolution in the specified orbit.

3) Comparison: For each of the 20 different variations on the design the number of generated watt-hours was computed for one revolution around Earth each month during one year in both reference orbit 1 and 2. The maximum and minimum value for the generated power during the whole year was calculated for each design and reference orbit. The minimum value shows the worst case in terms of generated power for that design and since the satellite has to be able to work under those conditions it is important to have a design with a minimum value that is as high as possible.

It is not decided which orbit the satellite will be launched into, thus the minimum and maximum value for both reference orbits was compared for every design. The lowest value is the worst case for that design. The design with the highest value as worst case was selected as the optimal design for each attitude and they were compared to each other.

4) Tumbling: When the satellite is launched from the rocket it tumbles and can spin in any direction until it is stabilized by the attitude control system. In this state the generated power is distributed only to the crucial subsystems since the available power is difficult to predict. The three optimal designs was analyzed to learn approximately how much power that will be available during tumbling.

A design in the satellite fixed coordinate system is translated to an attitude (Rolling Pin, Arrow or Tower) that can be rotated around the z-axis 90° resulting in 4 different positions. There are therefore 3 possible attitudes with 4 positions, thus the design can be translated into 12 different positions in total. The generated power is computed for the design in all 12 positions for one orbit each month of the year for both reference orbit 1 and 2. This procedure is done for the designs considered to be the most optimal, i.e. the one for each attitude with the highest minimum power generation.

For each design the power generation is known in all twelve positions during one orbit each month. When calculating the minimum generated power during tumbling the minimum value from all positions is chosen from each month, assuming the satellite will spin between the worst cases. When the maximum generated power during tumbling is calculated, this value is chosen instead, assuming the satellite will spin between the most optimal positions. Looking at the whole year all twelve positions every month are taken into account.

The use of twelve different positions is only an approximation, as when the satellite is tumbling it can also fly in positions in between those cases. For example it can rotate only 45° around the z-may axis instead of 90°. Some of these cases may generate less power and some of them generate more power than the values given from the calculation. Thus this approximation results in an interval of probable power generation, and not strictly the best or worst case.

D. Results

In Table IV the results from the power generation analysis as shown in Fig. 12 are presented. From a power generation perspective the Tower or Rolling pin attitudes are preferable.
During tumbling it is shown in Fig. 13 that there is only a marginal difference in the power generation between the attitudes and as such no additional design constraints were generated. The generated power during tumbling are shown in Table V.

### Table IV

<table>
<thead>
<tr>
<th></th>
<th>Rolling Pin</th>
<th>Arrow</th>
<th>Tower</th>
</tr>
</thead>
<tbody>
<tr>
<td>Orbit 1</td>
<td>15,10</td>
<td>11,04</td>
<td>11,58</td>
</tr>
<tr>
<td>Orbit 2</td>
<td>14,57</td>
<td>13,20</td>
<td>13,09</td>
</tr>
<tr>
<td>Min</td>
<td>14,77</td>
<td>8,99</td>
<td>12,20</td>
</tr>
</tbody>
</table>

### Table V

<table>
<thead>
<tr>
<th></th>
<th>Rolling Pin</th>
<th>Arrow</th>
<th>Tower</th>
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<tbody>
<tr>
<td>Year average</td>
<td>12,52</td>
<td>11,91</td>
<td>11,86</td>
</tr>
<tr>
<td>Max</td>
<td>15,17</td>
<td>14,99</td>
<td></td>
</tr>
<tr>
<td>Min</td>
<td>6,57</td>
<td>5,86</td>
<td>5,59</td>
</tr>
</tbody>
</table>

Fig. 12. Plot showing how the generated watt-hours for one revolution around Earth varies over the year. The colors represent the three optimal designs, one for every attitude, and are shown for both reference orbits.

Fig. 13. Plot showing how many watt-hours can be generated as minimum, average and maximum during tumbling. The colors represent the three optimal designs, one for every attitude, and their performance is investigated over a year.

### E. Power budget

In orbit the available power on the satellite has to be distributed to the experiments with regards to a power budget decided in advance. The purpose of the power budget is to make sure that the power consumption does not exceed the amount of power that the battery is capable of delivering. The available power depends on how much the solar panels are able to generate, the efficiency of regulators and the batteries depth of discharge. The power consumption in eclipse is especially important since the solar array will not generate any power and thus the battery must provide all of the power directly.

The battery’s depth of discharge should not exceed 30% since the total number of discharge cycles decreases rapidly with the depth of discharge as shown in Fig. 14 based on a study made by SAFT [24]. The study is relevant for the project as the typical battery-pack available for CubeSats are Li-Ion batteries.

Following the requirements analysis and the specifications from the payload ICD, a first version of the power budget was written. The average power consumption of each payload is calculated from the specified experiment duty cycle per day and the approximated power consumption while active.

Presented in Table VI is the first version of the power budget, in which the estimated subsystem requirements are based on the SEAM projects power budget. The power generated from the Tower configuration is based on the power generated in the worst case of the attitude from Table IV.
VI. DISCUSSION

A. Baseline Design

Summarizing the spacecraft design analysis and the conclusions drawn from the results, it was decided that the flight attitude would be the Tower configuration. This decision was motivated by the pointing requirements of the experiments as the Tower configuration is the only attitude that simultaneously allows the camera to point in the direction without the wings appearing in the pictures, allows Ratex-J to point in the $\vec{V} - \vec{W}$ plane without shadowing the field of view and accommodates the requirements on the field of view from the CUBES payload.

The solar illumination analysis shows that both the Tower and Rolling pin attitude can be used while the Arrow configuration should be avoided due to large differences between reference orbit one and two. Rolling pin generates slightly more power then the Tower attitude does both in flight position and during tumbling but the difference is not crucial in the choice of flight attitude. The optimal solar panel configuration when flying in the tower attitude is flying with the wings in front of the satellite, i.e. with the normal vector of the wings in the direction of flight.

In the mission analysis it was shown that while flying in the Tower or Rolling pin configuration with foldout solar panels the natural lifetime of the orbit is within the limit of 25 years, even considering the discrepancy between STK and DAS. Flying in the Arrow configuration would, as opposed to the Tower or Rolling pin configuration, require us to manually initiate a de-orbit algorithm to terminate the mission since it otherwise survives for longer than 25 years.

Following the conclusions from the power budget and solar array analysis, it was decided that foldout solar panels are essential in order to meet the power requirements of the experiments. As shown in the power budget, the generated power is not sufficient to run all the experiments every orbit. This can be solved by dividing the experiments into groups and making the other experiments wait on their turn while one group of experiments is running.

B. Further Work

The results of the initial perturbation analysis was very promising as it shows that at least two of the three sources of attitude disturbance are negligible compared to the actuation strength of the magnetic torquers. However, the dipole moment calculation and the measurement of the Piezo magnet showed that the dipole moment of the payloads needs to be carefully measured for the flight model. One workaround of the problem is to use a second set of magnetic torquier coils to counter the dipole moment of the structure and subsystems. Thus providing the ACDS coils a "magnetically clean" structure to control. This design is already in use on the SEAM satellite project. Even though such a system might create a magnetically clean environment for the ACDS, it would require space and power.

Further, from the results of measuring the dipole moment of the Piezo Motors component it will be increasingly important to measure the magnetic moment of any component not designed specifically to have a low magnetic dipole moment. Even though it was chosen to construct the magnetic dipole budget with some tolerance with regards to residual magnetization the exercise in Section IV-C showed that there is no easy way to deduce the true magnetic dipole moment of the components unless the dipole moment is thoroughly measured.

From a study conducted on the applications of Princeton Satellite Systems’ CubeSat Toolbox [25] it was concluded that the software could be used for modeling of the communication link systems power requirements. This would be done by creating a model of the link coverage and using their build in radio frequency (RF) amplitude loss calculation tools [26]. Using the RF loss in conjunction with their bit error probability calculator one could produce the power margins needed by the antenna array to assure appropriate data transfer. Which would provide input for a more thorough power requirements analysis.

Using STK one could set up a model of the orbit using the orbital elements of the reference orbit. In this orbit model one could upload a CAD model of the satellite with the

Table VI

<table>
<thead>
<tr>
<th>Experiment</th>
<th>Power req.[W]</th>
<th>Typical &quot;ON&quot; time (min)</th>
<th>Wh/day in eclipse</th>
<th>Wh/day in eclipse</th>
</tr>
</thead>
<tbody>
<tr>
<td>CUBES</td>
<td>1</td>
<td>98</td>
<td>8.2</td>
<td>2.9</td>
</tr>
<tr>
<td>Piezo LEGS</td>
<td>0.55</td>
<td>63</td>
<td>8.5</td>
<td>0.6</td>
</tr>
<tr>
<td>MoreBac</td>
<td>1.2</td>
<td>98</td>
<td>28.9</td>
<td>10.3</td>
</tr>
<tr>
<td>CubeProp</td>
<td>3.2</td>
<td>10</td>
<td>1.8</td>
<td>0</td>
</tr>
<tr>
<td>Ratex-J</td>
<td>2.7</td>
<td>30</td>
<td>19.9</td>
<td>7.1</td>
</tr>
<tr>
<td>SEUDet</td>
<td>1.2</td>
<td>98</td>
<td>28.9</td>
<td>10.3</td>
</tr>
<tr>
<td>SiCIS</td>
<td>0.4</td>
<td>98</td>
<td>9.6</td>
<td>3.4</td>
</tr>
<tr>
<td>Camera</td>
<td>0.6</td>
<td>30</td>
<td>0.6</td>
<td>0</td>
</tr>
<tr>
<td>Subsystems (SEAM)</td>
<td>1.8</td>
<td>98</td>
<td>43</td>
<td>15</td>
</tr>
<tr>
<td>Transmitter &quot;ON&quot;</td>
<td>2.2</td>
<td>10</td>
<td>4</td>
<td>2</td>
</tr>
<tr>
<td>Total requirement</td>
<td>-</td>
<td>-</td>
<td>153.4</td>
<td>51.1</td>
</tr>
<tr>
<td>With 30% margin</td>
<td>-</td>
<td>-</td>
<td>199.6</td>
<td>66.5</td>
</tr>
<tr>
<td>Power generated (Tower)</td>
<td>-</td>
<td>-</td>
<td>191</td>
<td>0</td>
</tr>
</tbody>
</table>
baseline design geometry of the solar array and with the built in perturbation toolkit generate reports on the attitude variation throughout the orbit. Specifically interesting is looking at the deviation of the satellite frame from the orbital frame’s Sun centered nadir vector, i.e. the vector from the satellite towards the center of the Sun. The deviation would give a good estimate of the change in cosine factor for the inclination of the Sun to the solar array and the deviation of the camera from the Earth centered nadir vector. This analysis would be of value to a refined power generation analysis taking this attitude perturbation into account. A similar model could be created by the CubeSat toolbox and the results could be compared to assess design margins on the ACDS. The CubeSat toolbox has a built in attitude perturbation tool [27] which provides a complete set of disturbance torques which could be used for comparison with the results in III.

Further work on the perturbation analysis would be in the analysis of de-tumbling. For example analyze the power requirements of the de-tumbling process when the satellite is launched from the P-POD and determine if it will be necessary to wait until the satellite comes out of eclipse to initiate the de-tumbling sequence. When out of eclipse the satellite can use the Sun sensor to determine the attitude and the solar panels can generate power.

Using the CubeSat toolbox it would be possible to create a crude propulsion model of the NanoSpace CubeProp. where the calculations done in the software takes as input the characteristics of the mass flow propulsion unit and calculated the thrusting force and generated torque [28].

The next steps in the power analysis would be in the sizing of the power system. Given the baseline design of the solar array one must look into the different battery packs available. The subsystem designer GOMspace has a guide on how to compose the power harness given different solar panel configurations [29]. In conjunction with the battery pack sizing a complete power conditioning analysis should be conducted including the measurements of inrush currents. If there are significant inrush currents, limiters or negative temperature coefficient (NTC) thermistors could be considered as a general design philosophy to assist in the power conditioning of the satellite. The way a NTC works is by having an initially very high resistance, but as the internal temperature increases the resistance decreases [30].

Considering the very tight margin of the total power consumption and the power generated from the solar array, it will be very important for the next student teams to carefully plan the activation cycles of the experiments and heaters. The activation cycles have to assure adequate power quality from the battery throughout the mission lifetime since the battery depth of discharge rapidly reduces the total amount of discharge cycles as shown in Fig. 14. The different power cycles, each with their group of experiments, have to be controlled by the on board computer that should be able to switch between the modes when necessary. An emergency power mode should also be designed where the power is used only by the critical subsystems and the heating of temperature sensitive experiments.

Further, considering the GOMspace P31us battery packs 3.3V and 5.0V connectors an analysis on the experiments and payloads connector load needs to be conducted. In such a study the relative load on the two connectors during different activation cycles should be evaluated. Could it affect the general power condition if all the experiments require an input voltage of 3.3V or if more than two experiments require the raw battery voltage during the same activation cycle? Both are questions that were raised during this semester but are handed over to the following student team.

Concerning the structure and Fig. 5 it will become increasingly important for the team to acquire fit-check models of each experiment. As seen in Fig. 5 no cables are depicted and one experiment is still a placeholder box. The dimensions of the outer structure is a hard constraint that is impossible to remove and even though it looks feasible in this CAD model the amount of cables and heaters are not shown which might take a lot more space than predicted.

As for the magnetic dipole moment, it will be increasingly important to measure the magnetic moment of the specific components and payloads when they are procured since the current dipole allotment does not take into consideration residual magnetization of hardware or other components. Further, the assumed down-scaling of the magnetic dipole moment of the payload stack might prove to restrictive, or to conservative which further strengthens the need for empirical measurements of the components actual magnetic dipole moments. In Fritz Prindahls booklet [21] there is a short list of common satellite components and their typical magnetic dipole moment measured using the AM method described in the booklet. However, as the Piezo magnet proved to have such a strong magnetic dipole moment, all components without a referenced dipole moment should be measured for the fit-check model. Since some components may have to be replaced if they have to much residual magnetization the components which are intended to go into the satellite should have their dipole moment measured prior to mounting.

As concluding remarks from the authors hope that the report presented through this bachelor thesis will constitute an insightful starting point for some of the future work of the next MIST student team.

ACKNOWLEDGEMENTS

We would like to express our outmost gratitude for the effort and guidance provided by our project supervisor Sven Grahn. Without his leadership and support throughout the project we would not have come this far nor learned this much.

REFERENCES

MATLAB code used in the attitude perturbation analysis

```matlab
close all; clear all; clc

%% Position (Bottom-Top)
% The hitherto calculated positions
% of centers of gravity for payloads
NSP = [98, 98, 30, 0, 0, -135, 300]; % [W, L, H, Xg, Yg, Zg, weight]
NSP1 = [98, 98, 10, 0, 0, -115, 100];
SEU = [98, 98, 20, 0, 0, -62.5, 300];
POCO = [96, 90, 30, 0, 0, -20, 90];
EPS = [90, 95, 15, 0, 0, 2.5, 250];
CAM = [96, 90, 10, 0, 0, 26, 166];
SEU = [98, 98, 20, 0, 0, -37.5, 60];
LEGS = [70, 30, 30, 0, 1, 87.5, 75];
ACDS = [96, 90, 15, 0, 0, 94.5, 195];
COMS = [90, 96, 15, 0, 0, 109.5, 85];
RTJX = [98, 98, 30, 0, 0, 132, 300];
ANT = [98, 98, 3, 0, 0, 148.5, 30];
PSC1 = [2, 2, 320, 49.5, 49.5, 0, 50];
PSC2 = [2, 2, 320, 49.5, -49.5, 0, 50];
PSC3 = [2, 2, 320, -49.5, 49.5, 0, 50];
PSC4 = [2, 2, 320, -49.5, -49.5, 0, 50];
PSB = [98, 98, 1, 0, 0, -150, 15];
PSB1 = [96, 98, 1, 0, 0, -5, 15];
PSB2 = [98, 98, 1, 0, 0, -5, 15];
PSB3 = [98, 98, 1, 0, 0, 93, 15];
PSB4 = [98, 98, 1, 0, 0, 93, 15];
SST = [100, 100, 1, 0, 0, 150, 15];
SSB = [100, 100, 1, 0, 0, -160, 20];
SSS1 = [100, 100, 1, 0, 0, 160, 20];
SSS2 = [100, 100, 1, 0, 0, 160, 20];
SSS3 = [100, 100, 1, 0, 0, 160, 20];
SSS4 = [100, 100, 1, 0, 0, 160, 20];
DSA1 = [100, 100, 50, -100, 0, 0, 78];
DSA2 = [100, 100, 50, -150, 0, 0, 78];
DSA3 = [100, 100, 50, -100, 0, 0, 78];
DSA4 = [100, 100, 50, -150, 0, 0, 78];

MIST = vertcat(ANT, RTJX, COMS, SEU, SiC, LEGS, OBC, CAM, EPS, POCO, MOBA, ACDS, ... 
NSP1, NSP2, PSC1, PSC2, PSC3, PSC4, PSB1, PST1, PSB2, PST2, PSB3, ... 
PST3, SSB, SST, SSS1, SSS2, SSSS3, SSS4, DSA1, DSA2, DSA3, DSA4);

%% Total Mass
% summing the masses of the payload,
M=0;
for i=1:34
    M=M+MIST(i,7);
end

%% Centroid
X=0;
Y=0;
Z=0;
for i=1:34
    X=X+MIST(i,4)*MIST(i,7);
    Y=Y+MIST(i,5)*MIST(i,7);
    Z=Z+MIST(i,6)*MIST(i,7);
end
X=X/M;
Y=Y/M;
Z=Z/M;

%% Moment of Inertia
% calculating the inertia tensor
MoI=zeros(3,34,3);
for i=1:34
    MoI(:,:,i)=[(1/12)*MIST(i,7)*(MIST(i,2)^2+MIST(i,3)^2)+MIST(i,7)*(MIST(i,5)^2+MIST(i,6)^2),
                0, 0;
                (1/4)*MIST(i,7)*MIST(i,1)^2, MIST(i,1)*MIST(i,2)^2+MIST(i,3)^2+MIST(i,4)^2+MIST(i,5)^2+MIST(i,6)^2, ... 
                (1/4)*MIST(i,7)*MIST(i,2)^2, (1/4)*MIST(i,7)*MIST(i,3)^2, (1/4)*MIST(i,7)*MIST(i,4)^2, (1/4)*MIST(i,7)*MIST(i,5)^2+MIST(i,6)^2);
end
```

MIST = vertcat(ANT, RTJX, COMS, SEU, SiC, LEGS, OBC, CAM, EPS, POCO, MOBA, ACDS, ... 
NSP1, NSP2, PSC1, PSC2, PSC3, PSC4, PSB1, PST1, PSB2, PST2, PSB3, ... 
PST3, SSB, SST, SSSS1, SSSS2, SSSS3, SSSS4, DSA1, DSA2, DSA3, DSA4);

% Total Mass
% summing the masses of the payload,
M=0;
for i=1:34
    M=M+MIST(i,7);
end

% Centroid
X=0;
Y=0;
Z=0;
for i=1:34
    X=X+MIST(i,4)*MIST(i,7);
    Y=Y+MIST(i,5)*MIST(i,7);
    Z=Z+MIST(i,6)*MIST(i,7);
end
X=X/M;
Y=Y/M;
Z=Z/M;

% Moment of Inertia
% calculating the inertia tensor
MoI=zeros(3,34,3);
for i=1:34
    MoI(:,:,i)=(1/12)*MIST(i,7)*(MIST(i,2)^2+MIST(i,3)^2+MIST(i,4)^2+MIST(i,5)^2+MIST(i,6)^2)+... 
                MIST(i,7)*MIST(i,1)^2+MIST(i,2)^2, MIST(i,3)^2+MIST(i,4)^2+MIST(i,5)^2+MIST(i,6)^2, ... 
                (1/4)*MIST(i,7)*MIST(i,2)^2, (1/4)*MIST(i,7)*MIST(i,3)^2, (1/4)*MIST(i,7)*MIST(i,4)^2, (1/4)*MIST(i,7)*MIST(i,5)^2+MIST(i,6)^2);
% Gravity Gradient Torque

ER=6.371E6;%Earth radius
r =0.640E6; %distance to satellite from mean sea level
GC=6.673E-11; %newtons gravitational constant
EM=5.972E24; %mass of the earth
mu=EM*GC; %shorthand

%calculating the gravity gradient torque
%from the difference in inertia, from equation (11), using the
%maximum amount of torque angle for certainty

% Arrow configuration
AGx=(3*mu/R^3)*(Izy-Izx);
AGy=(3*mu/R^3)*(Izx-Iyx);
AGz=(3*mu/R^3)*(Iyx-Izy);

% Tower configuration
TGx=(3*mu/R^3)*(Izx-Iyx);
Tgy=(3*mu/R^3)*(Izy-Iyx);
Tgz=(3*mu/R^3)*(Izx-Izy);

% Rolling Pin configuration
RGx=(3*mu/R^3)*(Iyx-Izy);
Rgy=(3*mu/R^3)*(Izy-Izx);
RGz=(3*mu/R^3)*(Izx-Iyx);

GGT=[AGx,AGy,AGz;
     TGx,Tgy,Tgz;
     RGx,Rgy,RGz]

%% Aerodynamic Torque
Cd=2.2; %structural coefficient +10% margin for certainty
Cp=2E-2;
h=7.64E3;
AA=1.0E-2;
TA=3.2E-2;
RA=3.2E-2;
V=sqrt(mu/R); %mean orbital velocity
% rhoLSA=7.33E-15;
% rhoMSA=1.04E-13;
% rhoHSA=7.23E-13;
 rho=4.89E-13; %density of the atmosphere at ~640km above mean sea level
RCp=sqrt(X^2+Y^2+Z^2)*10^-3; %mean distance from cm to center of pressure

% Arrow
AAx=(1/2)*Cd*rho*V^2*AA*Cp;
% Tower
TAx=(1/2)*Cd*rho*V^2*TA*Cp;
% Rolling Pin
RAx=(1/2)*Cd*rho*V^2*RA*Cp;
AT=[AAx,TAx,RAx]

%% Solar Pressure
RP=0.8;
IA=0.16;
MIST SATELLITE

SF=1367; %solar flux in W/m^-2
C=2.998E9; %speed of light

SPT=(SF/c)*IA*(1+RF)*Cp

%%% Magnetic Torquer Ability

m=0.20; %torque in Am^2
earthB=1.92E-5; %magnetic field from the earth (mean around the orbit)
legB =1.96E-6;
B=earthB-legB; %worst case
MT=m*B %magnetic torque converted to mechanical torque