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Attitude Determination and Control System Implementation for 3-Axis-Stabilized Nanosatellites

School of Electrical Engineering

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SÄHKÖTEKNIIKAN KORKEAKOULU

ABSTRACT OF THE MASTER'S THESIS

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In this thesis, a procedure is created for Attitude Determination and Control System (ADCS) implementation for 3-axis stabilized nanosatellites. The procedure is modified from the European Cooperation for Space Standardization ECSS to achieve a straightforward procedure suitable for a student satellite project. The resulting implementation procedure is described in detail consisting of requirements specification, system selection, procurement, verification and operations.
The Aalto-1 student satellite project is used as an example case to demonstrate the functionality of the procedure. The importance of a thorough requirements specification for the whole project became evident during the Aalto-1 case to achieve a smooth implementation.

Keywords: Nanosatellite, Attitude Determination and Control System, ADCS, Aalto-1, CubeSat, Process

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Työssä luodaan menettelytapa 3-akselistabiloitujen nanosatelliittien asennonsäätöjärjestelmän toteutukselle. Menettelytapa perustuu eurooppalaisessa avaruusteollisuuden yleisesti käytössä olevaan ohjeistukseen, jonka perusteella on laadittu suoraviivainen opiskelijasatelliittiprojektiin soveltuva prosessi. Työn tuloksena syntynyt prosessi kuvataan yksityiskohtaisesti koostuen vaatimusmäärittelystä, järjestelmän valinnasta, hankinnasta, laadunvarmennuksesta ja operoinnista.
Aalto-1 opiskelijasatelliittiprojektia käytetään esimerkkitapauksena osoittamaan menettelytavan toimivuus. Projektinlaajuisen perusteellisen vaatimusmäärittelyn tärkeys tuli ilmeiseksi Aalto-1 projektin tapauksessa, jotta sulava toteutus on mahdollista saavuttaa.

Avainsanat: Nanosatelliitti, Asennonsäätöjärjestelmä, Aalto-1, CubeSat, Prosessi

Preface

This Master's Thesis is a result of my studies at Aalto University School of Electrical Engineering and my ongoing participation in the Aalto-1 student satellite project. I would like to show my gratitude to several people who made the writing of this thesis possible.

First of all to my family, who have given me their full support during the entire course of my studies.

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To the whole Aalto-1 team, who I have enjoyed working with to realize the first Finnish satellite.

And last but truly not the least, to our Project Coordinator Jaan Praks, my thesis supervisor Prof. Martti Hallikainen, the project's Steering Group and Science Team and the Multidisciplinary Institute of Digitalisation and Energy (MIDE), for providing this opportunity for both the Aalto-1 project and this thesis.

Otaniemi, 22.12.2012

Tuomas Tikka

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List of symbols and abbreviations

Symbols

a	Semimajor axis
B	Magnetic field strength
e	Eccentricity
H	Angular momentum
h	Satellite altitude
I	Moment of inertia or Current
i	Inclination
H	Angular momentum
N	Number of turns of wire
M	Mean anomaly
m	Mass or dipole moment
R_E	Radius of the Earth
S	Area of wire loop
T	Torque
t	Time
v	Orbital velocity
α	Angle from Nadir direction
$\dot{\alpha}$	Slew rate
β	Angle between the vector from the Earth's center of mass to the target location and the vector from the target location to the satellite
Ω	Longitude of ascending node
ω	Angular velocity or Argument of periapsis
μ	Earth's gravitational parameter
ϕ	Geocentric semi-angle

Abbreviations

1U	One Unit CubeSat
2U	Two Unit CubeSat
3U	Three Unit CubeSat
ACS	Attitude Control System
ADS	Attitude Determination System
ADCS	Attitude Determination and Control System
CAD	Computer Aided Design
COM	Communication System
COTS	Commercial Of The Shelf
CSK	CubeSat Kit
ECEF	Earth-Centered, Earth-Fixed coordinate system
ECI	Earth Centered Inertial coordinate system
ECSS	European Cooperation for Space Standardization
EHS	Earth Horizon Sensor
EM	Engineering Model or Electrical Model
EMC	Electromagnetic Compatibility

EMI	Electromagnetic Interference
EPB	Electro-static Plasma Brake
EPS	Electrical Power System
EQM	Engineering and Qualification Model
ESA	European Space Agency
ESD	Electrostatic Discharge
FDIR	Failure Detection Isolation and Recovery
FPI	Fabry-Perot interferometer
FM	Flight Model
FMI	Finnish Meteorological Institute
FMECA	Failure Mode Effects and Criticality Analysis
FPGA	Field Programmable Gate Array
GND	Ground Segment
GPS	Global Positioning System / Navigation system
IMU	Inertial Measurement Unit
ITAR	International Traffic and Arms Regulations
LEO	Low Earth Orbit
MEC	Structure
MEMS	Micro-Electro-Mechanical System
MM	Mechanical Model
OBC	Onboard Computer
OBS	Onboard Software
P-POD	Poly Picosatellite Orbital Deployer
PA/QA	Product Assurance / Quality Assurance
PCB	Printed Circuit Board
PFM	Proto-Flight Model
RADMON	Radiation Monitor
SEE	Single Event Effect
SPEC	Spectrometer
TBC	To Be Confirmed
TBD	To Be Defined
TDE	Total Dose Effect
TRR	Test Readiness Review
VTT	Technical Research Center of Finland

1 Introduction

In the year 2010 Aalto University started a project whose target is to build Finland's first satellite; Aalto-1. This nanosatellite is designed and built by the students of the university.

The project received immediate interest from the space industry in Finland and acquired three technology demonstrator instruments as payload for the satellite. To meet the requirements of this payload, the satellite requires a sophisticated Attitude Determination and Control System (ADCS).

ADCS implementation is a complex and a costly part of a nanosatellite project requiring accurate attitude control. It is also a highly critical system for the mission: its operation must be flawless to ensure mission success.

1.1 Objectives

The main objective of this thesis is to develop a procedure for attaining an ADCS suitable for nanosatellites. In order to establish the procedure details, treatment of the following topics is carried out:

1. Requirements specification
2. System selection
3. Procurement
4. Integration
5. Verification
6. Operations

1.2 Scope

The amount of information required to thoroughly understand all steps in ADCS implementation procedure is broad. This thesis reviews solutions and gathers the main features into a single document, which can be used as learning material and a starting point for further research.

Small satellite missions are becoming more demanding and many of them require sophisticated attitude determination and control. Thus, this thesis concentrates on attaining modern 3-axis stabilization systems for nanosatellite use. Particular attention is paid to the procurement side of the implementation as the design of these systems during the timescales of a typical nanosatellite project is unlikely.

There are also high quality theses considering the actual design of ADCSs suitable for nanosatellites, e.g. Jensen & Vinther (2010).

1.3 Methods

The main method for the study is literature research and analysis. Books from the field of spacecraft systems are used to compile the necessary background information. A definitive authority in space projects is the European Cooperation for Space Standardization ECSS (2012) initiative. ECSS aims to develop a coherent, single set of user-friendly standards for use in all European space activities. These recommended practices are used and adapted for the needs of a student satellite project. The use of standard practices provides a solid system engineering approach for collaboration with the payload teams and subcontractors and also trains students the practices of the industry. One should also note that the use of ECSS is mandatory in all European Space Agency's projects. Documentation from the Aalto-1 project is also used in the case study of Aalto-1 ADCS implementation.

In the requirement specification for the Aalto-1 ADCS, suitable parameters are determined by calculations done in MATLAB (2012). Also, Satellite Toolkit's (STK) (2012) attitude plug-in and Solid Edge (2012) ST3 CAD software allow an easy method to visualize and verify requirements, but the use of these programs is not presented in this thesis.

1.4 Structure of the Thesis

Section 2 presents background information from space technology and satellite projects in general. This information helps in understanding the ADCS implementation procedure presented in Section 3. On purpose, the procedure is presented in its final chronological order instead of following the actual, occasionally rather convoluted development work, to emphasize its purpose as learning material. Section 4 presents the Aalto-1 project and its ADCS implementation using the procedure from the previous section. In section 5, the thesis is concluded by presenting the lessons learned and future ADCS task in the Aalto-1 ADCS implementation. Section 6 presents the references.

2 Background

Space control engineering is considered a multi-disciplinary field that requires insight into, at least, mechanics, dynamics, the space environment and its effects, digital and analogue electronics, control theory, computer systems and networks, software engineering, operations and many more. This chapter gathers relevant background information, which helps to understand the ADCS implementation procedure for nanosatellites.

2.1 Small Satellites

Large satellites have dominated the space industry in the past decades. However, reducing budgets and new advances in technology have increased attention to the capabilities and advantages of small satellites. (Fortescue, et al., 2003)

Small satellites are generally constructed rapidly and at relatively low cost. Especially, maximizing the use of Commercial-Off-The-Shelf (COTS) technologies is desirable. The mission objectives are carefully traded against cost to reach sufficient performance to achieve the required outcome. The risks are often mitigated by, for example following methods:

- Investing in thorough software development and testing.
- Keeping the interfaces simple.
- Minimizing the number of moving parts.
- Using previously flown designs and components in essential systems.
- Using realistic safety margins.
- Ensuring that systems are capable of independent operation.
- Using carefully selected high volume components where possible, instead of using truly space-qualified components.
- Using a layered, failure resilient system architecture.
- Ensuring a thorough burn-in prior to flight.

Small satellite projects are well suitable for universities and private companies, which have small teams working in close proximity with good communication. Appropriate documentation and 'best practice' processes and procedures carefully selected from industry are also a crucial aspect for a successful project. In space technology, the European Cooperation for Space Standardization (ECSS) provides a comprehensive document package of recommended practices. These practices may be further simplified for the purposes of a small satellite project.

Small satellite projects' may differ from conventional projects also in model philosophy. Along the progress of the project, a satellite project may employ different models (e.g. mechanical, electrical, engineering, qualification and flight). In a small satellite project, the timescales and budgets do not generally allow the use of all models. One less expensive and time-saving solution is to use an engineering model in the design phase of the mission and a proto-flight model for all qualification and operating purposes.

Satellites can be classified according to their mass and cost as seen in Table 1. A nanosatellite is generally considered to have a mass between 1.0 kg and 10.0 kg, and picosatellites between 0.1 kg and 1.0 kg. Table 1 shows also the typical project costs for each class of satellites.

Table 1: Classification of spacecraft by mass and cost (Fortescue, et al., 2003).

Class	Mass (kg)	Cost (M€)
Conventional large satellite	>1000	>100
Conventional small satellite	500-1000	25-100
Minisatellite	100-500	7-25
Microsatellite	10-100	1-7
Nanosatellite	1-10	0.1-1
Picosatellite	<1	<0.1

CubeSat standard was introduced by California Polytechnic State University and Stanford University in 1999. According to a study conducted by Bouwmeester and Guo (2010), it has boosted the number of developed pico- and nanosatellites enormously, especially amongst universities. About half of the launched Pico- and nanosatellites are built with an educational objective. The CubeSat standard (2009) defines among many other things the external dimensions and weight limits. A standard CubeSat, called a one unit (1U) CubeSat, is a 10-cm cube which weighs up to 1.33 kg. The size of the CubeSats can be increased in 1U increments to create 2U, 3U or even bigger CubeSats. All satellites built according to the CubeSat standard can be launched with a single adapter to the launcher, the Poly Picosatellite Orbital Deployer (P-POD). This allows low cost launches together with larger satellites without endangering the main payload, launch vehicle or other CubeSats. Also, relatively low cost subsystems bought from the market using the standard CubeSat Kit bus can be easily integrated to the satellite.

Technology demonstration is the most common objective of pico- and nanosatellites. Operational use such as scientific measurements or radio communications is also popular, but often very limited compared with larger satellites. Bouwmeester and Guo (2010) also noticed that most subsystem technologies used are rather advanced except for attitude control systems and performance characteristics of subsystems that depend on attitude control. Only about 40 % of all launched pico- and nanosatellites use active attitude control.

2.2 ECSS Standards

The European Cooperation for Space Standardization (ECSS) is an initiative established to develop a coherent, single set of user-friendly standards for use in all European space activities. This thesis follows these standards and applies them to meet the purposes of a student satellite project. The use of standards trains students the normal practices in the industry and also helps in cooperation with the project's partners. Furthermore, the use of ECSS is mandatory in European Space Agency's (ESA) missions: other space agencies employ similar requirements to ensure proper system engineering, too.

ECSS standards provide a comprehensive document package of recommended practices and procedures for a space project. The standards are organized into three branches; Space Engineering, Space Project Management and Space Product and Quality Assurance (PA/QA). Space projects differ greatly from each other due to, for example, varying mission objectives, organizations, budgets etc. and should use practices which benefit their project the most. The at times complex and particular practices of the ECSS standards show that they are created for the purposes of large-scale undertakings with long project timescales, large organizations and big budgets. The recommended practices are simply not feasible in a small satellite project and thus these standards should be studied to determine which practices are mandatory for a particular project in order to succeed.

In this thesis, the ECSS standards are analyzed and evaluated considering the implementation of an ADCS for the purposes of a student nanosatellite mission. A thorough presentation of the original practices is out of the scope of this thesis. The derived practices are presented in Section 3 and then used in Section 4. The derivation uses the following main criteria for the new practices:

- Limit the amount of simultaneous tasks, because the implementation is done by a small number of students with limited time resources.

- Limit the amount of documentation to be done. Even in small projects there is a risk of serious miscommunication, however, and there are documents that are used to minimize this risk e.g. interface documentation.

- Limit the amount of unnecessary analysis when a real test will provide the same information. Early prototypes and laboratory tests can remove at least some of the demand for detailed analyses.

2.3 Schedule

A schedule (ECSS, 2009a) of a satellite project is presented in this chapter. The understanding of the whole project's schedule is crucial to be able to time the ADCS implementation tasks accordingly. The schedule does not define each phase's durations as they depend highly on the project. Also, the duration of a small satellite development is generally much shorter than in bigger satellites, typically only two to four years, but incorporates the same development structure. This structure is shown in Figure 1.

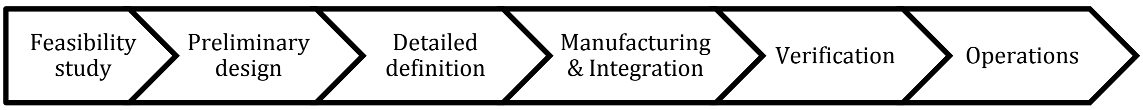

Figure 1: Satellite project schedule (ECSS, 2009a).

A feasibility study is a short evaluation to determine if the planned project and its mission are realistic. The design itself is divided to two phases, preliminary and detailed. In the preliminary design, the objective is to design a satellite that fulfills all the set requirements without going to too much detail. This approach allows more freedom in the design process before moving on to the detailed definition. Often several models are built to evaluate critical design details. Once the design has been finalized, manufacturing and integration of the final flight unit begins. Even though the verification phase itself is technically the final phase before launch, some tests and especially analysis should be performed already before integration. After the satellite and its systems have been fully verified and accepted by the launch provider, the operations can be started.

Formal project reviews are typically conducted before moving from one phase to the next. Their purpose is to provide a comprehensive assessment of the project status against targets and requirements. Often, an external review authority, usually consisting of experienced professionals, goes through the projects documentation, evaluates its feasibility and highlights potential problem and risk areas.

2.4 Models

Several models of the satellite (Fortescue, et al., 2003) are usually built during a satellite project. Due to low budgets and short timescales in a small satellite project, limiting the number of different models is unavoidable. This chapter presents the models and options for combining them. The different models are presented in Figure 2.

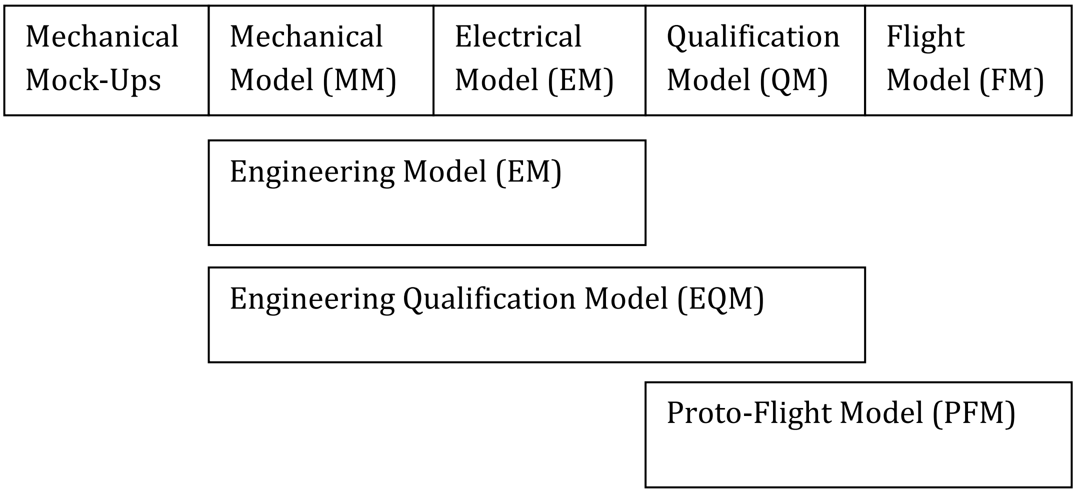

Figure 2: Models in satellite projects (Fortescue, et al., 2003).

Mechanical mock-ups are usually the first models to be built. They are used to visualize the satellites structure and placement of subsystems. The final Mechanical Model (MM) can be built only after the final dimensions, mass distribution, pointing requirements and the structure design is known. The Electrical Model (EM) is used to analyze and test the operation of the system. It does not have to look anything like the actual satellite and some subsystems may be even replaced by simulation software. These last two models may be combined to form a single model; an Engineering Model (EM). It has the mechanical functionality, or at least the same dimensions of the mechanical model as well as the operational functionality of the electrical model.

A Qualification Model (QM) is used to verify the satellite's design and the flight model is used only for acceptance tests and the mission operations. Some missions have more than one Flight Model (FM) i.e. flight spares which can be used as back-up or for tests/troubleshooting after the launch. The qualification model may be combined with the engineering model to form an Engineering Qualification Model (EQM) or with the flight model to form a Proto-Flight Model (PFM). The use of a proto-flight model causes excess stress for the flight model, but decreases the total amount of testing to be done in the development as the qualification and acceptance tests can be combined. Combining different models is very common in nanosatellite projects, where the development times are generally very short and budgets are tight. This practice is also frequently used in larger space projects to reduce costs and development time when the associated risks are considered acceptable.

2.5 Environment

Satellite's and its subsystems' environment is not only space but also the environment in manufacturing and launch. The satellite and its subsystems must be able to withstand these environments, and thus they need to be considered during the design, manufacturing and verification. Environmental aspects (Fortescue, et al., 2003) that need to be taken into consideration in different parts of the project are presented in the following.

Manufacturing phase:

Environmental conditions must be taken into consideration already during manufacture. Wrong conditions may lead to deterioration or even failures. The satellite manufacturing and system integration are thus carried out in appropriate clean room environments. Also, the conditions at storing and transportation must be taken to account. For example the following environmental aspects must be considered:

- Cleanliness
- Humidity
- Temperature
- Electrostatic Discharges (ESD)

Launch phase:

The launch is often considered as the most environmentally challenging phase for the satellite. The satellite and its subsystems are tested and verified before launch by appropriate vibration and thermal tests to ensure its durability in these conditions. The CubeSat Standard defines test requirements to all CubeSats to be launched with the P-POD adapter. In addition, launchers may pose certain requirements for the satellite. In addition to the same aspects as during manufacturing, at least the following environmental aspects must be considered:

- Structural vibrations
- Acoustic vibrations
- High levels of acceleration
- Changing thermal environment
- Rapidly declining ambient pressure

Operational phase:

The space environment can be particularly harmful to COTS devices, which are generally not designed for use in space. Many commercial components contain plastic materials, which may outgas under vacuum. Usually COTS parts are also only rated to operate at temperatures between 0 and +70 °C. Thus, special care to thermal design and testing must be taken. The parts may also be particularly susceptible to the effects of ionizing radiation, which must be considered in the design in order for it to cope with total dose effects (TDEs) and single-event effects (SEEs). This can be achieved with proper electrical design, mechanical shielding and using parts that are radiation tolerant or hardened. In addition to hardware, appropriate software design is often required for robust control. This approach can be extended to programmable logic circuitry as well e.g. Field Programmable Gate Array (FPGA) components.

The space environment causes also disturbance torques to the satellite. Such torques can be caused by Sun's radiation pressure, the satellite's electrical currents interacting with the Earth's magnetic field, gravity gradient and atmospheric drag at lower altitudes. An ADCS must be able to counter these disturbance torques, to achieve accurate and stable operation, but they may also be utilized for control purposes. For example, the following environmental aspects must be considered:

- Solar radiation
- Cosmic radiation
- Magnetic field
- Atmospheric drag
- Gravity gradient
- Changing thermal environment
- Meteoroids
- Out-gassing
- Electromagnetic compatibility/interference (EMC/EMI)

2.6 Coordinate Systems and Orbits

The movement of a satellite can be represented as rotations between different coordinate systems and using the orbital elements. A coordinate system, or reference frame, is a set of three orthogonal basis vectors defining a grid of three-dimensional space. The following four coordinate systems (Leppinen, 2011) presented in Figure 3 are generally used in attitude control calculations.

1. Earth-Centered, Earth-Fixed coordinate system (ECEF)

   The origin of this coordinate system is located at Earth's center of mass. The vector $\boldsymbol{x}_{\mathrm{ECEF}}$ is defined as the unit vector in the equatorial plane from Earth's center of mass to the Prime Meridian. The vector $\boldsymbol{z}_{\mathrm{ECEF}}$ is the unit vector from the center of Earth to the geographic North Pole. And, $\boldsymbol{y}_{\mathrm{ECEF}}$ is defined according to the right hand rule. This coordinate system rotates together with the Earth.

2. Earth Centered Inertial coordinate system (ECI)

   This reference frame has its origin also at Earth's center of mass. The vector $\boldsymbol{x}_{\mathrm{ECI}}$ is defined as the unit vector pointing from Earth's center of mass to the vernal point and does not rotate. The vernal point is the location where Sun's and Earth's equatorial planes intersect around March 21. Due to the precession of Earth's rotational axis, the point varies in time. Thus, also the used time shall be defined. As in EFEC coordinate system, the vector $\boldsymbol{z}_{\mathrm{ECI}}$ is the unit vector from the Earth's center of mass to the North Pole and the unit vector $\boldsymbol{y}_{\mathrm{ECI}}$ is defined according to the right hand rule.

3. Satellite Orbital Reference coordinate system

   The origin of this coordinate system is located in the center of mass of the satellite. The vector $\boldsymbol{x}_{\mathrm{ORBIT}}$ is defined as the unit vector in the direction of the component of the velocity vector that is orthogonal to the radius vector. The vector $\boldsymbol{z}_{\mathrm{ORBIT}}$ is the unit vector which points to Earth's center of mass and $\boldsymbol{y}_{\mathrm{ORBIT}}$ is defined according to the right hand rule.

4. Satellite Fixed Body coordinate system

   As in the previous coordinate system the origin of this coordinate system is located at the center of mass of the satellite. The vectors can be defined arbitrarily and rotate together with the satellite. Usually, they are selected to point to the direction of the satellite's instruments for convenience.

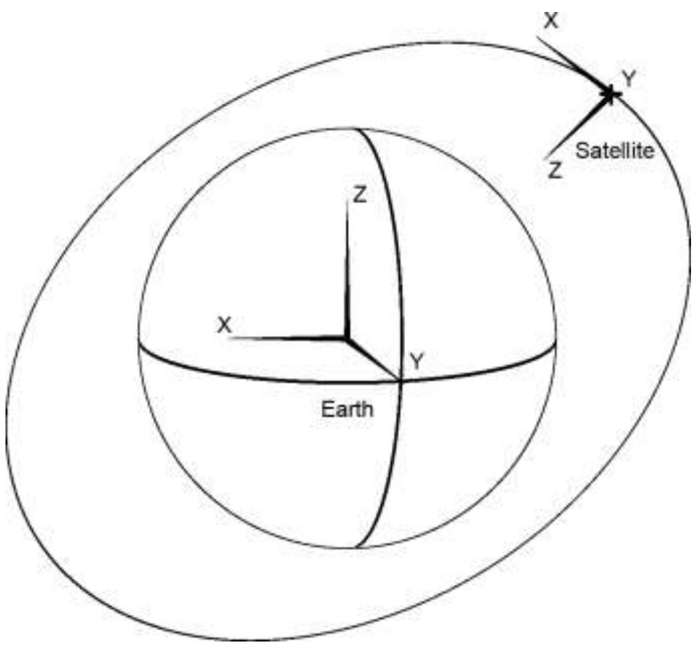

Figure 3: Earth-Centered, Earth-Fixed coordinate system and Satellite Orbital Reference coordinate system.

The orbit of a satellite can be defined using the following six orbital elements (Fortescue, et al., 2003) shown in Figure 4:

1. Eccentricity (e) determines the type of the conic section. It is a circle when e = 0, an ellipse when 0 < e < 1, a parabola when e = 1 and a hyperbola when e > 1.

2. Semimajor axis (a) is the average distance between the bodies.

3. Inclination (i) is the vertical tilt of the orbit with respect to the reference frame's vernal point (ECI).

4. Longitude of ascending node (Ω) is the horizontal orientation of the ascending node, i.e. where the orbit passes upward the reference frame, with respect to the reference frame's vernal point (ECI).

5. Argument of periapsis (ω) is the angle measured from the ascending node to the semimajor axis.

6. Mean anomaly (M) is the position of the orbiting body along the orbit at a specific time (epoch). This element may be replaced by true anomaly ν, which is the angle between the argument of periapsis and the orbiting body.

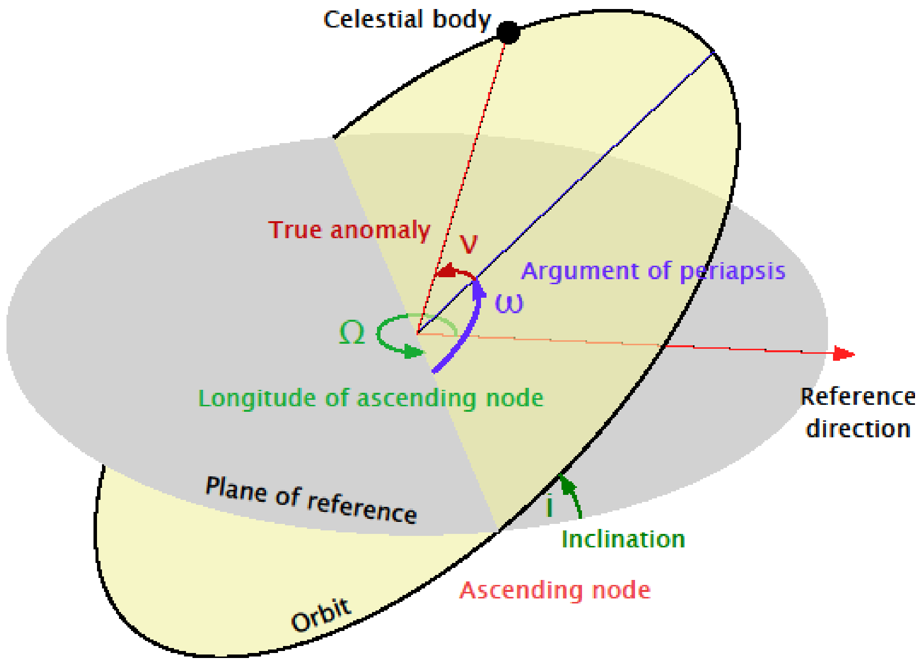

Figure 4: Orbital elements (Snyder, 2007).

Nanosatellites use generally quite circular (e ≈ 1) low altitude orbits, called Low Earth Orbit (LEO), commonly between 500 - 900 km altitudes. The advantages of using these orbits include; low cost launch, low radiation exposure and short revisit times, i.e. the time between successive overpasses. The desired and achievable inclinations depend heavily on the type of mission operations, ground station locations and possible launch options. An orbit with inclination close to 90 degrees is called a polar orbit. For optical remote sensing, a highly preferred orbit is a sun synchronous polar orbit. Its orbital plane rotates approximately one degree per day eastwards to keep pace with the Earth's rotation around the Sun. Thus, the satellite will pass the same areas at approximately the same time every day, thus having very similar lighting conditions. Also, scheduling daily mission operations becomes easier.

2.7 Dynamics

The motion (Fortescue, et al., 2003) of a satellite can be divided into:

1. The motion of the center-of-mass *C*, and
2. The motion relative to the center-of–mass.

Trajectory dynamics provides the rules governing the motion of the center-of-mass relative to some inertially fixed frame of reference. Attitude dynamics, on the other hand uses the center-of-mass as a reference point.

The attitude of the satellite can be determined as a rotation between the Satellite Fixed Body coordinate system and the Satellite Orbital Reference coordinate system using Euler angles, rotation matrices or quaternions. Euler angles are used in this thesis as they are a common and straightforward method to represent attitude. All possible attitudes of a satellite can be achieved as a sequence of three separate rotations (α, β and γ) about the coordinate axes shown in Figure 5. These rotations must be performed in this exact order to end up in the correct position.

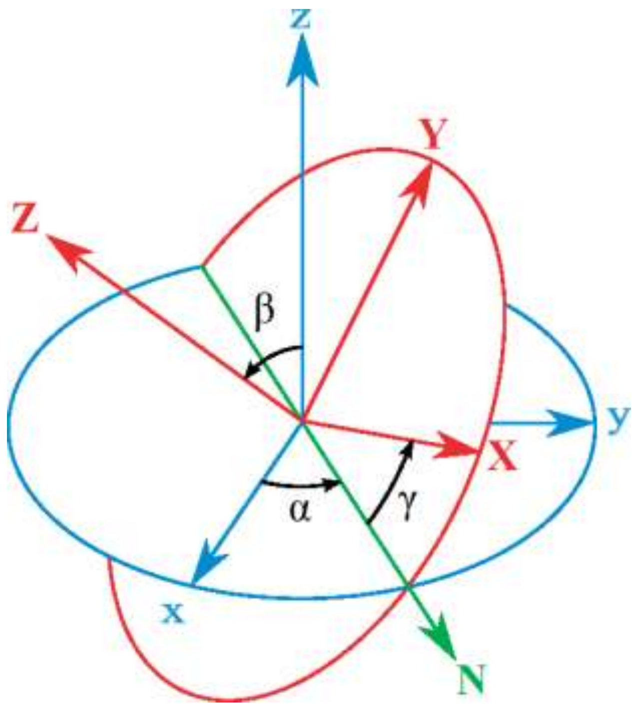

Figure 5: Euler angles (Brits, 2008).

The orientation of various coordinate systems may change with respect to each other. The rate of change can be described with angular velocity ω. It relates closely to angular momentum $\boldsymbol{H_c}$ of a single rigid body referred to its center-of-mass C. The angular momentum (Fortescue, et al., 2003) can be expressed as

$$\boldsymbol{H_C} = [I_C]\omega, \tag{1}$$

where I_C is the inertia matrix and ω is the angular velocity relative to an inertial frame of reference. In general, *$[I_C]$* can be expressed as

$$[I_C] = \begin{bmatrix} I_{xx} & -I_{xy} & -I_{zx} \\ -I_{xy} & I_{yy} & -I_{yz} \\ -I_{zx} & -I_{yz} & I_{zz} \end{bmatrix}, \tag{2}$$

where I_{xx}, I_{yy} and I_{zz} are the principal moments of inertia. They describe an object's tendency to resist changes in rotation. I_{xy}, I_{yz} and I_{zx} are the products of inertia, broadly representing a measure of the lack of mass symmetry, leading to cross-coupled behavior. The moment of inertia about the x-axis is, for example,

$$I_{xx} = \int (y^2 + z^2) dm, \tag{3}$$

where the integral extends over the whole mass distribution. And the product of inertia associated with the x-axis is

$$\mathrm{I_{yz}} = \int \mathrm{yz\, dm}. \tag{4}$$

The angular momentum of a satellite can be changed in two ways:

1. By applying an external torque

$$\mathbf{T} = \frac{\mathrm{d}}{\mathrm{dt}}\mathbf{H} \tag{5}$$

2. By ejecting some particles whose momenta have moments about the reference point, for example, by using a thruster.

Internal torques will not change the total momentum. Satellites will always be under naturally occurring external disturbance torques and the mean level will therefore cause a progressive build-up of the angular momentum over the lifetime of the satellite. A stabilized satellite needs to have external torquers to remove this build-up, called momentum dumping. For a 3-axis stabilized satellite with small angular velocities and roughly symmetrical mass distribution, the responses about the principal axes are largely uncoupled and can be approximated (Fortescue, et al., 2003) to

$$I_{xx}\dot{\omega}_x = T_x, \qquad I_{yy}\dot{\omega}_y = T_y, \qquad I_{zz}\dot{\omega}_z = T_z. \tag{6}$$

2.8 Attitude Determination and Control Methods

Satellites can be divided into different categories according to their stabilization type: three-axis stabilized with and without momentum bias, spin-stabilized, hybrid and non-stabilized. This thesis focuses on three-axis stabilization without momentum bias as it is used increasingly in nanosatellites requiring accurate control. Information about the other stabilization categories can be found, for example, in Fortescue, et al. (2003).

For three-axis stabilization, a dedicated satellite subsystem called an ADCS is used. The ADCS can be perceived as two different systems, Attitude Determination System (ADS) and Attitude Control System (ACS). The ADS is used in learning the satellite’s present attitude. This information is then used as input for ACS that rotates the satellite into a desired attitude.

The block diagram (Fortescue, et al., 2003) of an ADCS working principle is presented in Figure 6.

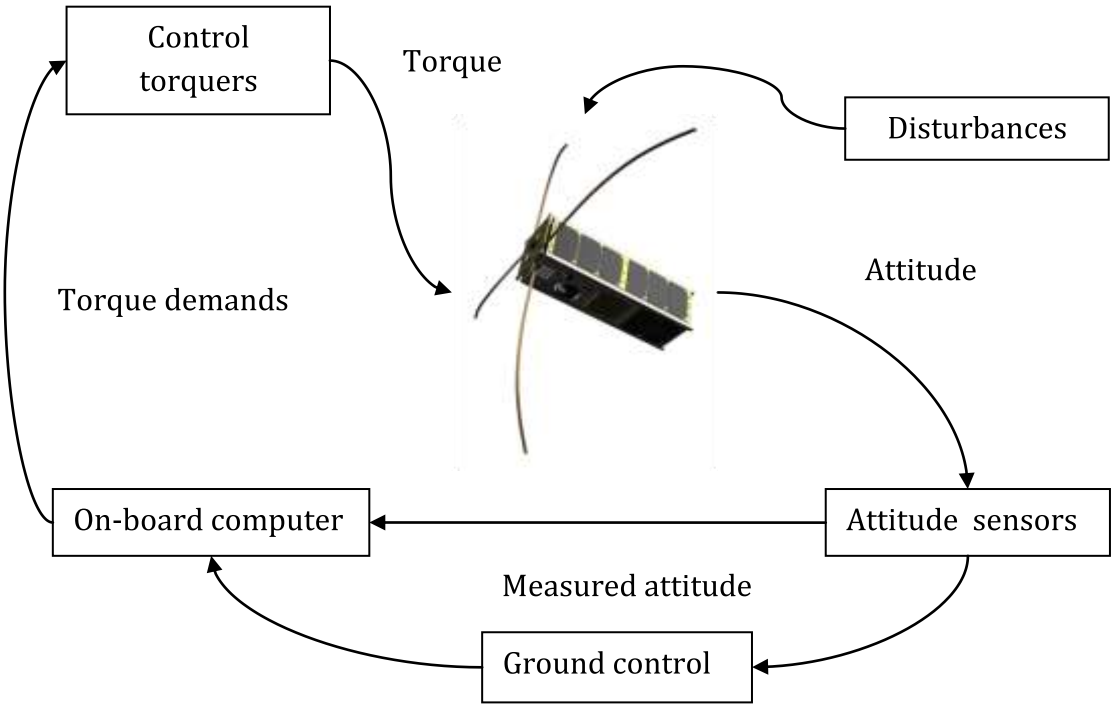

Figure 6: ADCS block diagram (Fortescue, et al., 2003).

The satellite's attitude changes as control and disturbance torques affect it. This can be measured with appropriate sensors which provide this information to the satellite's or ADCS's computer and also to the ground station. The computer calculates the current state, the torque demands and controls the torquers of the satellite to achieve the desired attitude. Usually, the computer also performs system monitoring for failure detection. In addition, the parameters of any used models needed for control may be adjusted based on the measurements.

2.8.1 Attitude Determination

The ADS measures the attitude of the satellite using a suite of sensors and sensor fusion to obtain an accurate measurement. This can be compared to a datum frame of reference e.g. Satellite Orbital Reference coordinate system and the angular departure from this datum can be defined. Selected determination methods (Fortescue, et al., 2003) for nanosatellites are presented in this chapter.

Magnetometers, e.g. SSBV (2011a), are very common sensors in nanosatellites' attitude determination. They are used together with magnetic torque rods or coils.

A system having these torquers requires knowledge of the Earth's magnetic field to operate as intended. Magnetometers measure the Earth's magnetic field vector local to the satellite. They might also be sensitive enough to pick up eddy currents from the satellite's other systems, so care must be taken to distinguish these two. Due to attitude ambiguity, magnetometers should be used in conjunction with other sensors, for example sun sensors.

Sun sensors, e.g. ISIS (2011a), provide the satellite with the azimuth and elevation of the sun vector, i.e. the direction to the Sun, giving two axis of attitude knowledge. To obtain full knowledge, the sensor must be used in conjunction with other sensors, for example the magnetometer. A sun sensor will not work while in eclipse. If only a crude knowledge of the Sun's direction is required, it can be determined from the solar panels' voltage variations.

Earth Horizon Sensors (EHS), e.g. SSBV (2011b), provide the satellite with attitude knowledge relative to the Earth. It measures infrared radiation coming from the direction of Earth's horizon so that the angle between the horizon and the sensor can be determined. They are generally more accurate than sun sensors.

Star trackers, e.g. BST (2011), determine the attitude of the satellite by taking a picture of stars and comparing it with known star positions. They are the most accurate attitude determination instrument, but generally very expensive and thus not usually used in nanosatellites.

Unlike the sensors presented so far, Inertial Measurement Units (IMUs), e.g. Analog Devices (2011), do not detect the absolute attitude of the satellite, but the changes in its rotation and acceleration. This is achieved, for example, with a gyroscope and an accelerometer. As the satellite rotates a gyroscope stays in place due to gyroscopic rigidity and the deflection can be measured. Nowadays, small and low-cost micro-electromechanical system (MEMS) gyroscopes and accelerometers are also available. IMUs are very effective when used together with sensors that measure absolute attitude. Because IMUs tend to drift over time, they should be calibrated with an absolute method at specified intervals.

2.8.2 Attitude Control

The main purpose of an ACS is to orientate the main structure of the satellite to the desired attitude with sufficient accuracy in the space environment. This can be accomplished with a broad range of methods. Selected methods (Fortescue, et al., 2003) for nanosatellite attitude control are presented in this chapter.

Magnetic torquing is a very common attitude control method for nanosatellites. It works simply by running a current through a coil of wire. This creates a magnetic field which interacts with Earth's magnetic field and generates a torque **T** that acts to bring the fields into alignment. Equation for the torque (Fortescue, et al., 2003) is

$$\mathbf{T} = \mathrm{NI}\mathbf{S} \times \mathbf{B}, \tag{7}$$

where N is the number of turns of wire, I the current. $\boldsymbol{S}$ the area of the wire loop and $\boldsymbol{B}$ is the local magnetic field vector. By reversing the current, also the direction of torque is reversed. Magnetic rods, e.g. ISIS (2011b), work the same way, but they have a ferromagnetic core to amplify the magnetic field created. The torque is typically small at reasonable currents and coil/rod sizes. Proper control also requires knowledge about the local magnetic field direction and strength. In equatorial orbit, 3-axis stabilization with only magnetic torque is not achievable, because the magnetic field lines point constantly to one direction. On the other hand, in polar orbits the magnetic field line directions are not always predictable near the poles which decreases the accuracy of this method. Nevertheless, magnetic torquers are widely used due to the simplicity and for being an external control method to achieve momentum dumping.

A nanosatellite can achieve very accurate control by using small reaction wheels, e.g. Maryland Aerospace (2011). They generate angular momentum by spinning a flywheel according to Equation (1). This does not change the satellite's total angular momentum as it does not interact with the environment and thus causes the satellite to start spinning to the opposite direction than the flywheel. Disturbance torques in the same direction will constantly add angular momentum to the satellite and the reaction wheel needs to counteract this by accelerating its spin. However, the wheels have maximum spin speeds and eventually they need to be de-spun. Therefore, an additional external method, such as the magnetic torque rods/coils, is required for momentum dumping.

3 Steps in ADCS Implementation

This section discusses the tasks of an ADCS implementation procedure, in Figure 7, created in this thesis. These tasks are divided into different steps or phases. Ideally, after requirement specification and selection phases, the phases are consecutive.

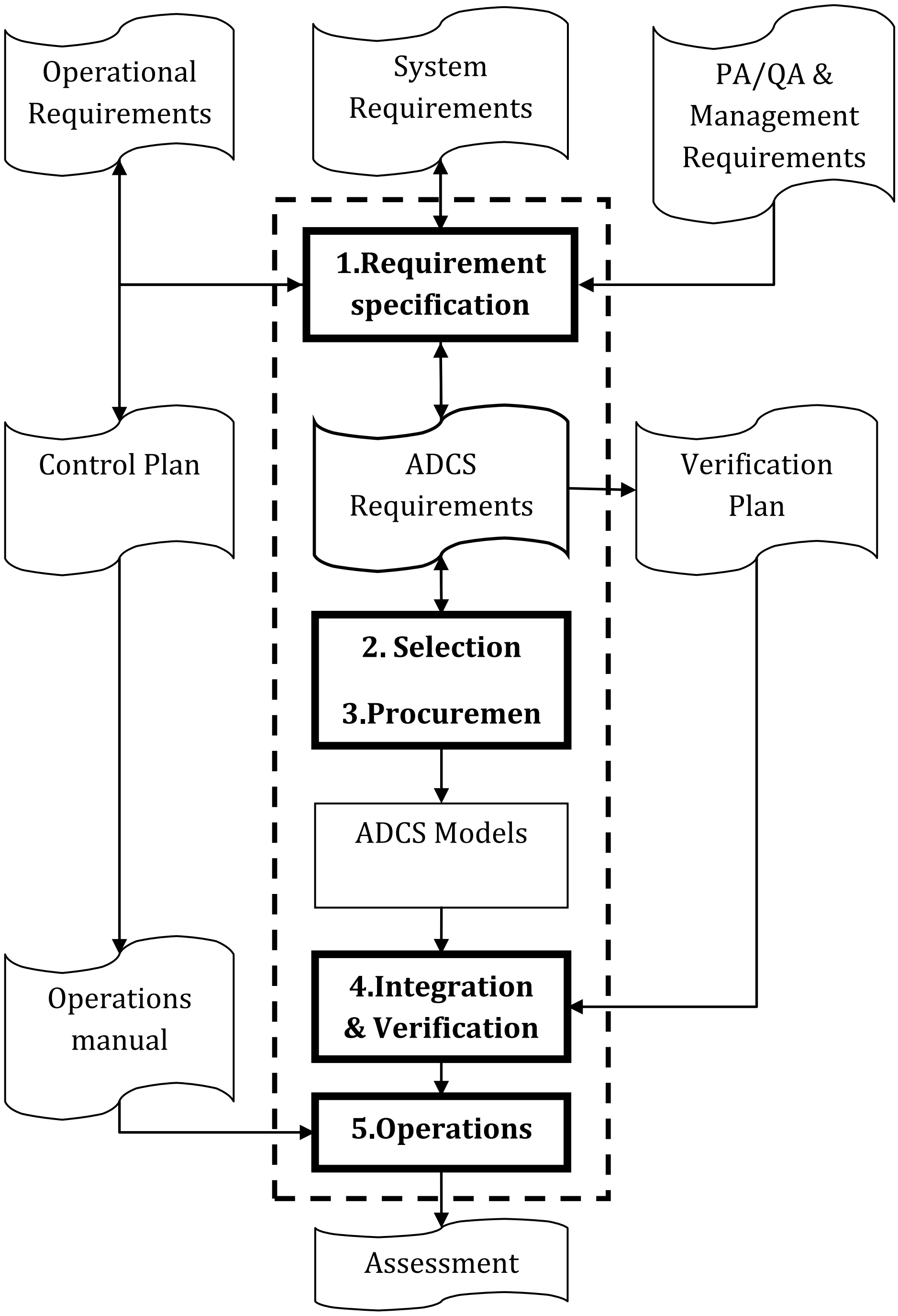

Figure 7: ADCS implementation procedure.

These phases, inside the dashed line in Figure 7, are derived from the phases in control engineering ECSS (2004) standard and are presented in detail in the following chapters. The procedure starts with the definition of the ADCS requirements. This is followed by system selection, procurement, integration & verification and operations. Figure 7 presents also the requirement documents, plans and models consisting of the ADCS itself and the supplier's documentation and also how they are used in each of these phases. An arrow towards the phase shows it is used as an input and outwards means it is an output. Two-way arrows show that the interaction is iterative and may affect both ways. Also, there are other relationships, which are considered as exceptions from the normal procedure and are thus not presented here. The inputs, outputs and tasks in each phase are also presented as tables in the beginning of each phase's chapter for convenience.

The schedule to perform these phases may differ from project to project, but a conceptual overview is presented in Figure 8, which also illustrates how the results of this thesis relate to the nominal space project flow shown in Figure 1.

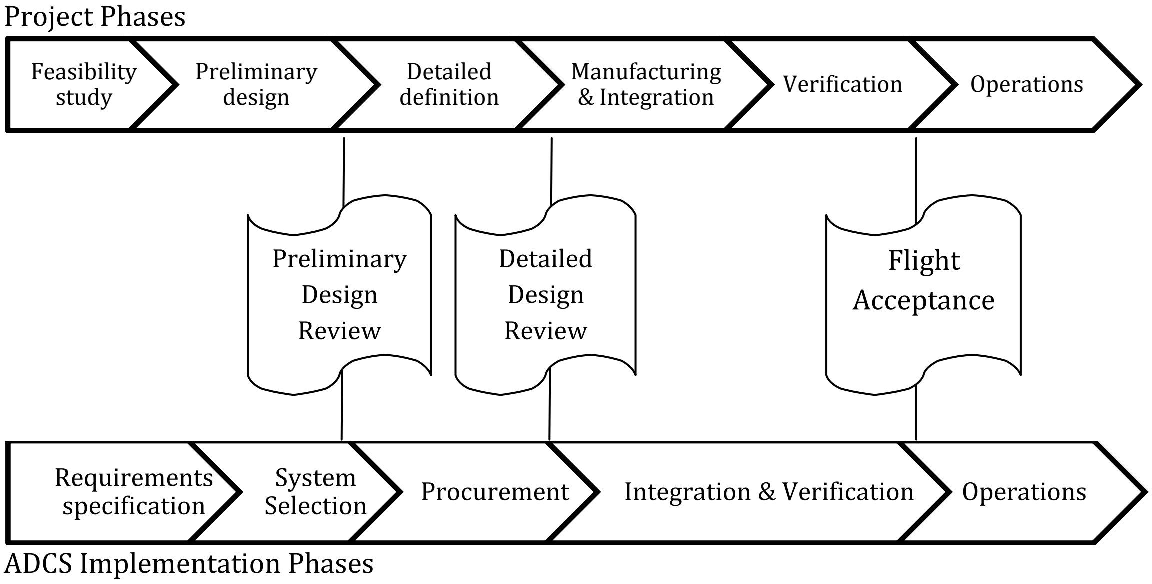

Figure 8: Project schedule in relation to ADCS implementation schedule.

The phases of a typical space project and the ADCS implementation differ slightly, but they have connecting factors. The first is that the procurement should not start before the preliminary design has been reviewed, as the design may change significantly if problems are noticed and a completely different ADCS solution may be required. The procurement should be done to schedule the ADCS delivery before the end of detailed definition to make sure the ADCS meets its requirements and it will not cause changes to the rest of the design. The final connecting factor is at the end of verification, where both the satellite and its ADCS are required to be accepted together for flight with flight acceptance level tests.

3.1 Requirements Specification

The first task in the ADCS implementation procedure is to define the requirements for the control system. A thorough requirements definition is necessary to reach sufficient performance to achieve the desired outcome. It is also generally the most time-consuming phase of the project and should be performed with great care as it affects the whole rest of the project as was shown in Figure 7. The identified and derived inputs and outputs as well as the related tasks in requirements specification are summarized in Table 2.

Table 2: Inputs, tasks and outputs of the requirements specification phase.

Input	Tasks	Output
- Operational Requirements - System Requirements - PA/QA & Management Requirements - Mission Plan	- Upper and lower level ADCS requirement specification - Preliminary Control Planning	- ADCS requirements - Preliminary Control Plan

The ADCS requirements specification process is shown in Figure 9. In the following, the details are elaborated.

Figure 9: Requirements specification process.

The requirements are specified by analyzing the system and operational requirements. Usually, management as well as product and quality assurance requirements also influence the ADCS requirements. The specified requirements need to be verified later, and thus also verification requirements must be defined. The outputs of this step are the ADCS requirements and the verification requirements as well as preliminary Mission Plan considering attitude control i.e. a Control Plan. The requirements from which the ADCS requirements are derived may change and new requirements may emerge changing the ADCS requirements. Thus, requirements engineering shall continue through the whole project with updates where necessary.

3.1.1 Requirement Hierarchy

The requirements in a satellite project are usually allocated to the upper and the lower level requirements, since the total number of requirements is generally very large and all details may not be clear before further analysis is carried out. The upper level requirements should set out what the satellite should do in broad terms. As requirements engineering is an iterative process, the requirements may change during the project. However, in an ideal case the upper level requirements do not need to be revised later. Upper level requirements for each subsystem should be defined very early in the project from the mission and system requirements. These requirements should define at least the satellite's orbit, the payload, objectives and interfaces to the system bus. The emphasis should be on functional and operational needs rather than implementation details. This approach creates a solid foundation for the lower level requirements.

The upper level requirements for the ADCS are derived from the following needs:

- The ADCS shall be able to withstand the operation environment.
- The ADCS shall meet the operational requirements for the mission.
- The ADCS shall be compatible with the system interfaces.
- The ADCS shall be compatible with the project's schedule and model philosophy.
- The ADCS shall be compatible with the project's Product Assurance (PA)/Quality Assurance (QA) philosophy.

These requirements should be specified early in the project and defined with sufficient accuracy to be able to select the preliminary stabilization method and design of the ADCS. This information is also needed for preparing an Invitation to Tender in case the ADCS is provided by a third party.

The lower level requirements are derived from the upper level requirements and other lower level requirements as shown in Figure 10.

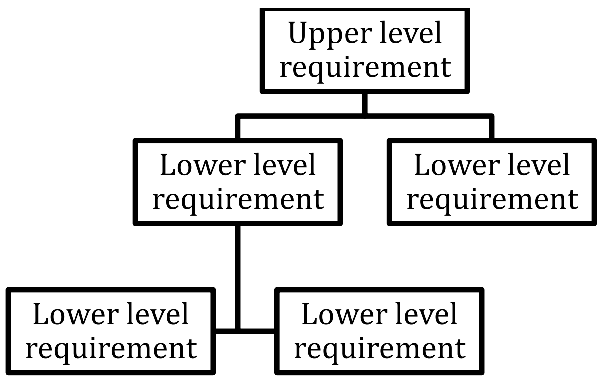

Figure 10: Requirement tree.

This type of requirements specification helps in finding all necessary requirements. The hierarchical structure clarifies the dependencies of all requirements and enables their systematic evaluation when changes are made. Some of the lower level requirements often concern the same functional needs such as attitude control accuracy. In these cases the strictest requirement should be used. A conscious effort should be made to minimize the number of requirements.

The lower level requirements may deal with details that cannot be determined in the beginning of the ADCS implementation, such as the system architecture or testing arrangements. Thus, all requirements cannot be specified in the beginning of the project, but should be documented and preferably given an estimate so that people are aware of them and can define them in more detail later. In space projects, the de facto standard is to use To Be Confirmed (TBC) or To Be Determined (TBD) to clarify intent when specifics cannot be provided.

3.1.2 Requirements derivation

Requirements in a space project can be allocated to groups in many ways. One allocation practice is presented in ECSS (2009c) standard. For simplicity, a more straightforward practice is used in this thesis. This is illustrated in Figure 11.

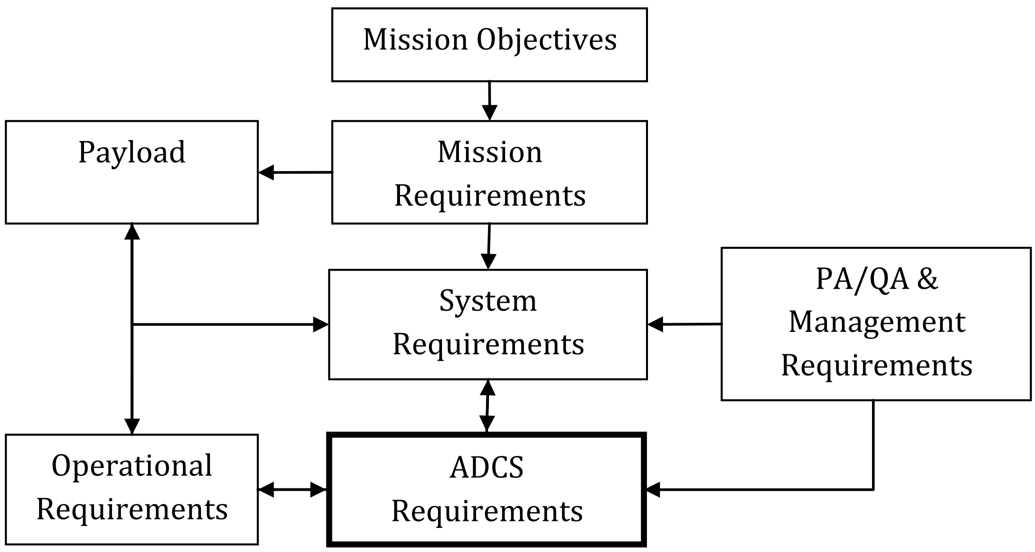

Figure 11: Requirement derivation.

The requirements specification in a satellite project starts by analyzing the mission objectives to generate mission requirements. From the mission requirements, it is possible to define requirements for the payload and the rest of the system. The ADCS requirements must consider constraints imposed by the satellite system (e.g. electrical power, mechanical configuration, thermal conditions) as well as operational requirements of the payload (e.g. pointing requirements, attitude knowledge) and the PA/QA & Management requirements (models, schedule) of the project. ADCS requirements may also generate requirements for other systems (e.g. mechanical stiffness, alignment, power) and operational requirements (e.g. orbit, operations), and is thus an iterative process. Therefore requirements engineering should be done in close collaboration between, at least, control engineers, system engineers and mission designers to determine requirements, which serve the whole satellite project in the best possible way. Every requirement should have a unique identifier, a description, reason for the requirement (reference) and also possibly a responsible person/group mentioned. The derivation of ADCS requirements from each of these requirement groups is presented in the following chapters. Also, the requirements for the Aalto-1 ADCS derived in this thesis are presented in Appendix A.

3.1.3 System Requirements

A satellite project should have system requirements prepared by the system engineers. They define common requirements for the satellite and may also define subsystem specific requirements. These requirements usually consider:

- Interfaces (mechanical, electrical, thermal, communication etc.)
- Dimensions
- Power consumption
- Mass
- Durability (radiation, thermal etc.)

These requirements should have sufficient margins, typically 20 % in the beginning of the project, to ensure they can be fulfilled. Often some details or aspects are missing in the original design plans. This will not be found out until the design has advanced to highlight specific needs leading to changes. For example, vibration tests may reveal that the mechanical structure needs additional support resulting in mass increase.

3.1.4 Operational Requirements

The operational requirements are resolved by analyzing the intended operation of the satellite. The payload instruments of the satellite may generate requirements for the ADCS. On the other hand, the satellite system constrains the operational performance of an ADCS. Thus, an iterative approach is often necessary. For a systematic approach in specifying operational requirements, it is beneficial to create a Control Plan. Space projects usually have a Mission Plan prepared by the mission designers. By analyzing the Mission Plan details, the required operations for the ADCS can be determined. The ADCS requirements derived from the operational requirements generally consider:

- Pointing requirements
    - Pointing accuracy and knowledge
    - Slew rates
- Operation modes

The Control Plan can be supplemented during the project with the schedule of the operations, attitude descriptions, operation modes, tasks, control commands and back-up plans. It can also be used in creating the Operations Manual for the mission operations. More information about creating a Control Plan is given in Chapter 3.5, and the preliminary Control Plan for the Aalto-1 mission is presented in Appendix B.

3.1.5 PA/QA & Management Requirements

A satellite project needs a dedicated set of PA/QA requirements prepared by the quality engineers to ensure the quality of the final flight unit. It defines among other things the verification, cleanliness and model philosophy requirements of the project. The satellite project management should also have determined the schedule for the project. The requirements for the ADCS set by the PA/QA and Management predominantly consider:

- Models
- Schedule
- Verification requirements
    - Tests
    - Analysis
- Documentation requirements
- Procurement requirements

3.1.6 Verification Requirements

It must be possible to verify all requirements during the course of the project. This is mostly performed during the verification phase of the project, but feasibility studies and analysis before that are recommended for early identification of possible problems. Verification must be defined for every requirement. A requirement, which is not verifiable, can be used only as a guideline and has no real value. The Verification Plan shall state:

- How each of the ADCS requirements should be verified.
- When each of the ADCS requirements should be verified.
- What the acceptance and qualification limits are.

Verification is presented in more detail in chapter 3.4 and an initial Verification Plan for the Aalto-1 ADCS developed in this thesis is presented in Appendix C.

3.2 Preliminary System Selection

The design of the ADCS can be selected according to the defined ADCS requirements and the preliminary Control Plan. The decision between in-house design and an outside procurement is desirable from very early on due to short timescales. Thus, the ADCS requirements do not have to be complete at this stage. The identified and derived inputs and output as well as the related tasks in preliminary system selection are summarized in Table 3. This chapter presents information required for the preliminary ADCS decisions.

Table 3: Inputs, tasks and output of the preliminary system selection phase.

Input	Tasks	Output
- Preliminary ADCS Requirements - Preliminary Control Plan	- Analysis of Feasible ADCSs - Analysis of design vs. procurement	- Preliminary ADCS Selection

A major decision is to select the type of stabilization. Different types of methods are presented in Figure 12.

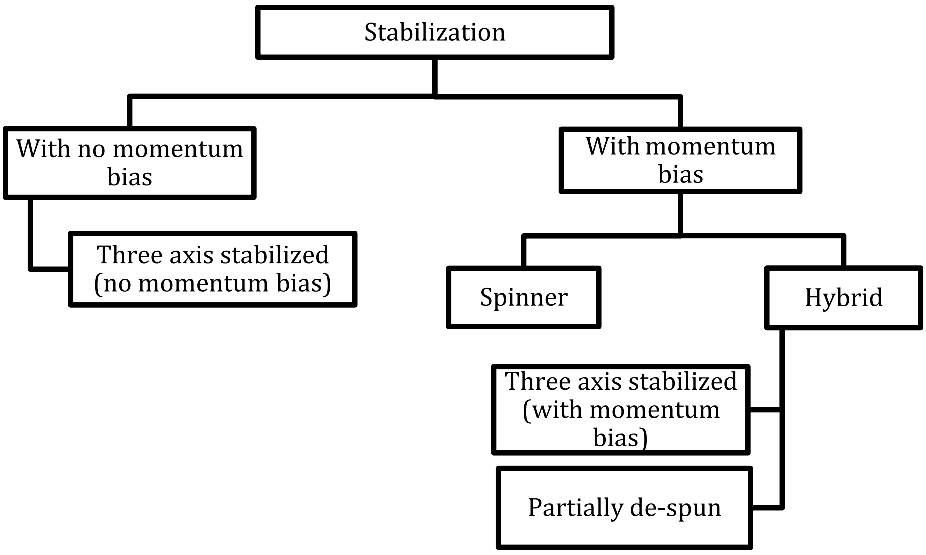

Figure 12: Satellite stabilization types (Fortescue, et al., 2003).

Stabilization can be achieved with or without momentum bias. By using momentum bias the satellite gains gyroscopic rigidity and thus resists disturbance torques well. Spinning the satellite is the easiest way to achieve this but allows pointing only to the direction of the spin axis. If the satellite needs to be able to point also to other directions, a momentum wheel can be used. Momentum bias, however, limits the maneuvering capability due to gyroscopic rigidity. Without momentum bias, the stabilization is generally harder to maintain, but allows more agile operations.

After the stabilization type has been selected, the preliminary configuration of the ADCS can be determined to meet the ADCS requirements. The ADCS must have at least one external torquer for momentum dumping and one absolute attitude sensor for calibration. Most commonly this is achieved by using magnet torquers and sun sensors. A greater accuracy, generally <1° for a 3U CubeSat, can be achieved by selecting internal torquers e.g. reaction wheels and accurate sensors, e.g. EHSs or a star tracker, to achieve better attitude knowledge. Attitude control and determination calculations may be done by the satellite on board computer or a specific computer embedded in the ADCS. Very different methods for attitude control and determination can result in similar performance. The final decision on what method to use must consider economical and experiential aspects in addition to the technical aspects. If the ADCS is procured, the supply in the market may play the biggest role in the selection. Some additional things to take into consideration in making the decision between the procurement and the design of the ADCS are for example:

- The complexity of the required design
- Expertise to design an ADCS
- Desire for knowhow in ADCS design
- Availability and cost of systems in the market vs. design cost
- Time limits and management
- Availability of test facilities

It is possible to design an ADCS consisting of basic components such as magnet torquers and Sun sensors within the time limits of a nanosatellite project, especially if the team has expertise in control systems engineering. High accuracy 3-axis-stabilized systems for nanosatellites are much more complex, in particular in software, and are seldom designed or even used. Nowadays, the supply in the market for highly accurate ADCSs for nanosatellites is, however, increasing and they are a viable option for universities and private companies requiring this type of a system for their mission. It also matches together with the modern day philosophy of using COTS components as much as possible. Thus, the procurement process of such systems is studied in more detail in this thesis. The output of this step is the preliminary selection of an ADCS type.

3.3 Procurement

This chapter presents a derived process to procure an ADCS. The identified and derived inputs and outputs as well as the related tasks for procurement are summarized in Table 4.

Table 4: Inputs, tasks and outputs of the procurement phase.

Input	Tasks	Output
- ADCS Requirements - Preliminary Control Plan - Preliminary ADCS Selection	- Survey - Quote Requests - Negotiations - Selection - Tracking - Co-operation with supplier	- ADCS Models - Documentation from the supplier

The procurement process (Artto, et al., 2006) shown in Figure 13 aims for the final selection of the ADCS to be procured and the delivery of the ADCS itself.

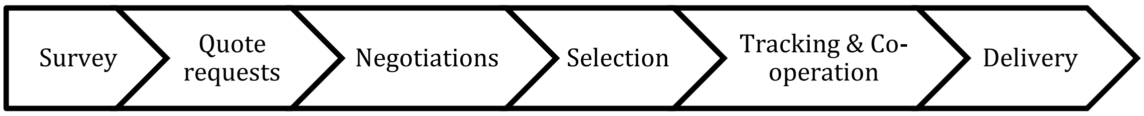

Figure 13: Procurement process (Artto, et al., 2006).

The process starts from surveying the market for different vendors selling ADCSs and requesting quotes from the suppliers, offering the most suitable solutions for the project. After this, the contract negotiations may begin. The contract negotiations should be scheduled so that the final decision of to procure a certain ADCS can be made after the preliminary design review, as it may change the design and so also the requirements for the satellite and the ADCS (ECSS, 2009a). After the ADCS has been selected, the procurement progress shall be tracked until the deliveries have been made. Cooperation with the supplier should continue still during the operations in form of mission support.

When requesting quotes, the suppliers should be asked to provide a comprehensive technical specification about their ADCS, the requirements set for the satellite, life cycle-support information, development schedule and pricing. Some organizations may require an Invitation To Tender to be published for a certain period to inform all potential suppliers and to allow them to send their quotes. This provides all suppliers a chance to offer their products for unbiased assessment. Within the European Union, there are legal requirements to follow this practice of open procurement. The invitation to tender should consist, at least, of the minimum requirements, the eligibility of suppliers, the selection criteria and delivery terms. The details should be always discussed together with the organization's procurement specialists who can provide legal advice where needed.

At least two alternative suppliers should be compared to create competition, and also to have an option for a possible back-up system to allow late replacement. The unsuitability of the selected solution may not be revealed before system-level tests. Also, development or manufacturing problems may occur at subcontractors.

The decision-making between possible systems can be eased by using trade-off tables. Selection criteria can be weighted differently according to the importance in the project. Artto, et al. (2006) groups the criteria to four different factors; the economic point of view, the credibility of the supplier, the technical solution and the feasibility. An example of a nanosatellite ADCS selection trade-off table is presented in Chapter 5.4.

The sales negotiations aim for creating a binding contract between the buyer and the supplier. An additional contract may not be even needed, if the buyer accepts the original offer. A contract should be unambiguous and mutually binding, but also flexible, so that it may be changed when plans are revisited or conditions change. The most important content of a contract is the scope, responsibilities, risks and pricing. An example of a contract's contents for a nanosatellite project is presented in the following:

1. Parties and contacts
2. Contract's target, scope and schedule
3. Technical description
4. Delay and warranty terms
5. Parties' rights and obligations
6. Pricing, billing and payment terms
7. Confidentiality
8. Terms of change
9. Dismantling and cancellation policies
10. Signatures

From management point of view, it is convenient to use milestones to track the development and manufacturing. The milestones may be, for example, hardware development, qualification, manufacturing, acceptance tests, software development and documentation development. The schedule should not end to the delivery, but also a mission support period should be determined. The schedule is convenient to have as an appendix in the contract: from the legal and management point of view, the appendixes can be updated with little difficulty compared with a full legal process of signing a contract.

The technical description should have a comprehensive explanation of the system, especially the interfaces, possible models (e.g. EM, QM, and FM), testing and what documentation will be provided. A product tree view helps at perceiving the system's configuration. Also, if the ADCS generates any requirements for the satellite or the project, it should be mentioned clearly. These requirements may be, for example, cleanliness requirements and distribution of mass limits.

The payments can be performed in many ways. One recommendable way is to divide the total payment sum to smaller sums to be paid when certain payment milestones, clearly identified in the schedule and with listed deliverables, have been completed. In this case the dismantling and cancellation policies must be carefully considered and explicitly determined in the contract. Also, delays and ahead of time deliveries may be fined and rewarded.

During tracking, the supplier should be requested to provide situation updates about the progress. Contacts should be appointed from both parties who should be reachable if needed in timescales defined in the contract's obligations. All details of the integration, especially the software, are very unlikely decided before the contract. Thus, regular progress meetings are recommended as well. Also, for example, the requirements for the ADCS or to the satellite may still change during the procurement phase and may impact the designs. In case the procurement is noticed to be delayed or other problems occur, a deadline to switch to the back-up solution should be determined. The date should be early enough before the launch, that a new contract, delivery, sufficient modifications and verification can be made in due time. However, the exact date and switch decision shall be discussed with the project's management.

3.4 Integration & Verification

In this chapter, a derived process for integration and verification is presented. The identified and derived inputs and output as well as the related tasks in integration and verification are summarized in Table 5.

Table 5: Inputs, tasks and output of the integration and verification phase.

Input	Tasks	Output
- ADCS - Verification Requirements - Documentation from the supplier	- Tests & Analysis - FMECA/FDIR - Integration	- Verified ADCS

ADCS's integration and verification to the satellite shall be carried out according to specified interface, configuration and verification requirements. The overall objective of the verification is to demonstrate, through a dedicated process, that the deliverable product meets the specified requirements. This may be divided to further objectives (ECSS, 2008a):

- Demonstrate the qualification of design and performance, as meeting the specified requirements at the specified levels.

- Ensure that the product is in agreement with the qualified design, is free from workmanship defects and acceptable for use.
- Confirm product integrity and performance at particular steps of the project life cycle (e.g. launch, commissioning, and mission events).
- Confirm that the overall system (including tools, procedures and resources) is able to fulfill the mission requirements.

The straightforward process for verification is presented in Figure 14:

Figure 14: Verification process.

Every requirement specified must be verified. Thus verification planning can start as soon as the ADCS requirements have been specified. When the ADCS arrives from the supplier, a receiving inspection should be conducted. The verification is conducted according to procedures specified in the Verification Plan, which also define the criteria for qualification and acceptance. There are four types of verification methods:

- actual tests
- design reviews
- inspections
- analysis of mathematical models

Actual tests shall be used where-ever possible and done using test procedures usually identified already in the Verification Plan. It should be noted that verification tools in test procedures might need to be verified also. Design reviews are done to parts that only operate once and cannot be tested. Analysis may be used before actual testing already during the design phase and for requirements, whose testing would be unreasonably difficult or impossible.

Analysis should be performed already before the verification phase of the implementation procedure. One widely used analysis tool in aerospace engineering is called Failure Mode, Effects, and Criticality Analysis (FMECA) specified in ECSS (2009b) standard. The designer of the ADCS should use this analysis method during the whole design process to find out critical parts for reliable operation of the system. The reliability of these parts may be then increased by using more reliable parts, adding redundancy or by using a new type of design. The designer should discuss the FMECA results with system engineers so that the reliability may be increased by design or possible back-up systems and the operations can be planned in case of system failures. Systems which are determined to be the most critical are named as critical items (ECSS, 2008b). The design, testing and operations of them can then be followed with great care. A critical item may be named also on the basis of previous projects or experience in the industry. Whereas FMECA tries to find critical parts in the system, Failure Detection, Isolation and Recovery (FDIR) presents the procedures in failure situations (ECSS, 2009d). A FDIR analysis should be made at least for the most probable failure situations and the FDIR ability should be implemented to the ADCS computer. The procedures can be either manual or automatic to detect, isolate and to try recover from in failure situations. The FDIR philosophy should be a conscious effort in the design process, so that all mission critical system failures are detectable and their effects can be mitigated or even repaired. As FDIR aims to prevent the propagation of failures, FMECA uses the FDIR analyses of all subsystems to determine the critical items. Especially identifying Single-Point Failures, or the cases where the failure propagation cannot be prevented contributes to understanding the potential outcomes.

Launch providers specify the mandatory tests (ECSS, 2002) for launch. These are usually vibration and thermal vacuum tests, which may depend on the type of the launcher. For these types of requirements, there are two test stress levels; qualification and acceptance. The acceptance tests are used to determine that the system works in its intended operational environment and to ensure that there are no major flaws. The qualification levels are higher and aim to find all possible problems and weak spots in the design, but may damage the system. For this reason, a separate qualification model is generally used to spare the flight model from excess stress. In small satellite projects, these two models are often combined to reduce costs. In a nanosatellite mission, it is very typical that the launch provider (and so also the launcher to be used) is decided quite late in the project. Thus, different options for launch and their test levels should be surveyed already early in the project so that the system can be designed to withstand these levels.

Testing should be conducted at both system and subsystem levels. Subsystem level testing ensures that the system meets its requirements in independent operation. It also saves time as major problems may be addressed before all systems are ready for integration. All testing results shall be documented. If the system is procured and some tests are done by the supplier, they shall provide the testing documentation. Receiving inspection and a set of functional tests shall be conducted when the ADCS is received from the supplier to ensure that all required items are received, the ADCS is free of transportation damages and works as intended. The supplier should provide all the required equipment and procedure manuals needed to conduct these tests.

System level testing can be done after the ADCS has been integrated to the satellite and a Test Readiness Review (TRR) has been conducted. The TRR gives a formal go-ahead for testing. It determines that the system is build according to the final design and the test procedures and facilities are approved. The system level tests are usually similar to sub-system level, but also additional tests, considering e.g. EMC and operations, may be performed. EMC testing is especially important for a satellite using magnet torquers/coils. A dipole moment may be generated by the satellite's electronics, which may hinder magnetic control (NASA, 1969). A test sequence, showing recommended tests and the order they should be performed can be found from ECSS (2008) standards. The functional tests should be performed at least three times during the whole project: before delivery, before qualification tests and after qualification tests.

After the system has passed all verification requirements, it is ready for pre-launch stage verification. It demonstrates that the satellite is properly configured for launch activities and early operations. These tests may be carried out by the launch provider. Commissioning is performed in orbit before the satellite starts its operational use. It ensures that no degradation has occurred during the launch, and also calibrates and tests the system under its operational conditions. Some experimental operations may be left out of the commissioning, if they are considered to risk the mission too early in the operations.

3.5 Operations

This chapter presents the operations phase of the ADCS implementation procedure. The identified and derived inputs and output as well as the related tasks in integration and verification are summarized in Table 6.

Table 6: Inputs, tasks and output of the operations phase.

Input	Tasks	Output
- Verified ADCS - Verified Satellite - Operations Manual	- Mission Operations - Evaluation	- Assessment Results

Operations shall be carried out according to an Operations Manual. Control engineers should input to the Operations Manual to ensure its feasibility. It should have troubleshooting procedures and back-up operations ready in case of problems in the ADCS or other systems. This way, valuable mission time can be saved.

The control engineers shall create and maintain an attitude Control Plan, which defines the control operations during the extent of the mission. A preliminary Control Plan should be prepared already during the requirements specification process to determine operations; including all required operation modes and the system's required performance. The operation modes should be designed together with the supplier of the ADCS to ensure compatibility with the Control Plan. In the design, command levels and the division of actions between software, hardware and human operations should be considered. The operation modes may be either low level modes with only simple manual commands or high level with complex operations. In small satellite projects, the software in loop approach is generally avoided to limit the complexity of the software, but may be required for some operations. The operation modes should be designed so that an unexpected loss of communication between the ground station and the satellite does not cause failure propagation. The tasks of the ADCS, e.g. providing telemetry data, and the durations of each operation shall also be determined in the Control Plan.

Typical operations of a 3-axis stabilized nanosatellite mission are presented in Figure 15.

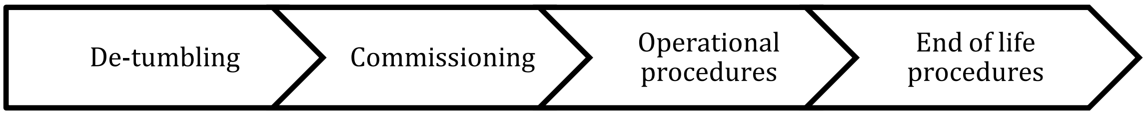

Figure 15: Mission operations phases.

After the separation from the launch vehicle the satellite's attitude is unknown. The mission of a 3-axis-stabilized satellite typically starts with de-tumbling the satellite. The ADCS shall have an autonomous operation mode that stabilizes the satellite and preferably points it to a direction where it can gather maximum energy with its solar cells and contact a ground station. After the satellite has been contacted, it and the ADCS will go through a commissioning phase where required control procedures are verified to work as intended. Specific operational procedures of each subsystem start after their operation has been verified in commissioning. End of life procedures shall also be considered; e.g. low power operation modes, de-orbiting etc.

The mission success should be assessed during and after the operations. A part of it is analyzing the operation of the ADCS. The idea is to evaluate how well the ADCS has fulfilled the requirements set for it. The evaluation results may generate valuable information, which can be utilized already during the on-going mission or in future design and implementation tasks.

4 Use case: Aalto-1

In this chapter, Aalto University's student satellite project Aalto-1 is used as an example case to demonstrate the ADCS implementation procedure. The Aalto-1 satellite, in Figure 16, and its intended operations are presented in this chapter.

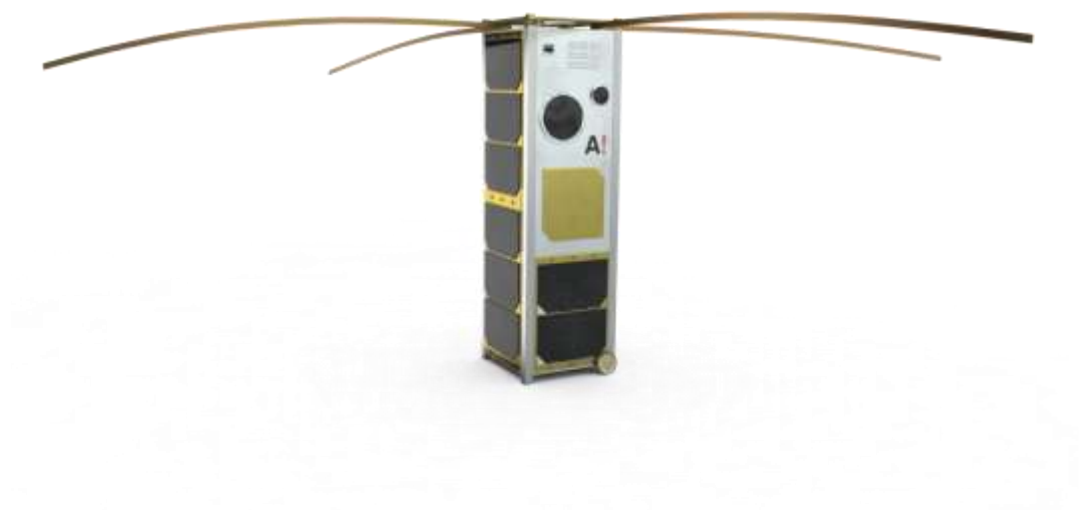

Figure 16: Artist's vision of the Aalto-1 satellite (Courtesy of the Aalto-1 Team).

Aalto-1 has various and demanding attitude control requirements and thus requires an advanced 3-axis stabilization ADCS. In this thesis, these control requirements are defined and a suitable ADCS is selected according to them. The procurement process for the ADCS is demonstrated and also verification methods and operations are studied. The objective of this Section is to present the work that has been done so far for the ADCS implementation and to present future tasks necessary to perform the implementation procedure successfully.

4.1 Background

The Aalto-1 satellite is an Earth Observing multi-payload nanosatellite, designed according to CubeSat 3U satellite specifications. The main specifications (Aalto-1, 2011a) of Aalto-1 are given in Table 7.

Table 7: Aalto-1 satellite main specifications (Aalto-1, 2011a).

Parameter	Value
Mass	4 kg
Dimensions	340x100x100 mm
Payloads	Imaging Spectrometer
	Radiation Monitor
	Electrostatic Plasma Brake
Orbit	Polar orbit between 600-800 km
Attitude Stabilization	3-axis attitude determination and control
Power	Solar powered, max 8W
Communication	VHF-UHF telemetry and command, amateur radio compatible
	S-band data transfer
Design	CubeSat 3U
Launch	2013

The schedule (Aalto-1, 2011a) of the development is shown in Figure 17.

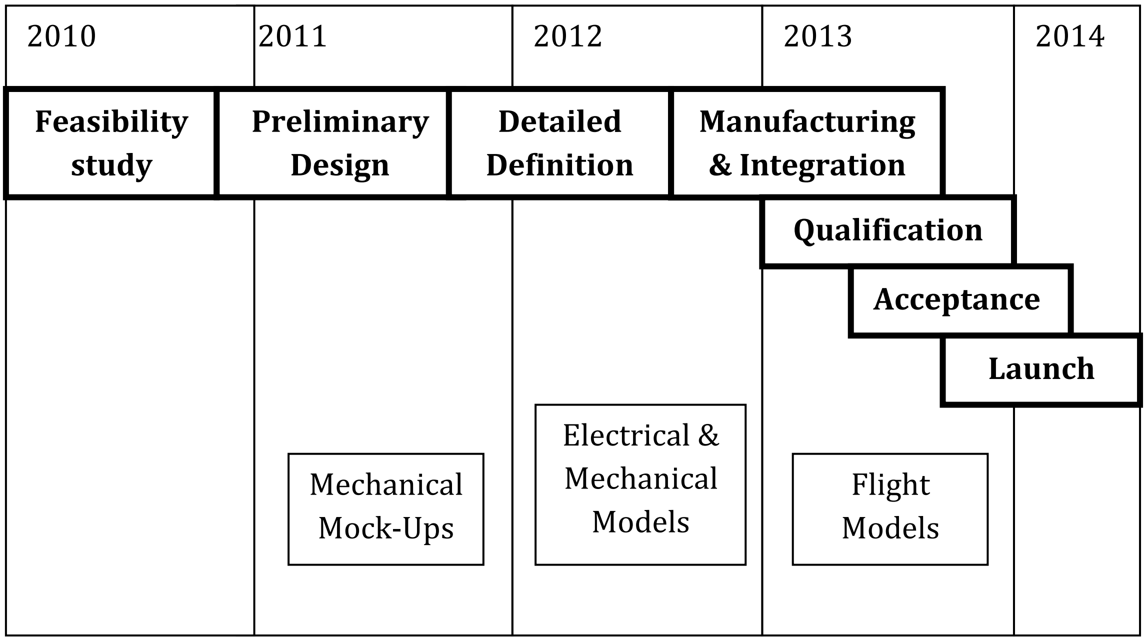

Figure 17: Aalto-1 project roadmap (Aalto-1, 2011a).

The development is divided into the following phases: Feasibility study, preliminary design, detailed definition, manufacturing & integration, qualification, acceptance and launch. After every phase, a written review is made to sum up the work done and to ensure that all parties' tasks are on schedule. Figure 17 also shows which models are going to be manufactured and estimated need dates.

4.1.1 System

The Aalto-1 satellite's system bus (Aalto-1, 2011b) is constructed for the purposes of the payload. It consists of the following sub-systems:

- Onboard Computer (OBC)
- Onboard Software (OBS)
- Electrical Power System (EPS)
- Communications System (COM)
- Attitude Determination and Control System (ADCS)
- Navigation (GPS)
- Structure (MEC)
- Ground Segment (GND)

The interfaces of the ADCS with these other subsystems are summarized in Table 8, where CSK means CubeSat Kit and RS485 is the used communication protocol.

Table 8: Interfaces of the Aalto-1 ADCS.

	OBC	**OBS**	**EPS**	**GPS**	**MEC**	**COM**	**GND**	**SPEC**	**RADMON**	**EPB**
ADCS	CSK	RS485	+5V	TBD	CSK	Through the On Board Computer				

It should be noted, that the ADCS may have to provide telemetry through the on board computer to systems, which it is not directly connected to, such as telemetry to control the payload's operations. Due to the incompleteness of the OBC and its software, a more detailed presentation of these operations is left out.

4.1.2 Dynamics

The moments of inertia (Young and Freedman, 2004) of the Aalto-1 satellite can be calculated with Equation (8). The mass is assumed to be the maximum allowed 4 kg and the mass distribution is approximated to be constant. The lightweight antennas are also omitted from the calculations. The moments of inertia are thus the same as for a simple cuboid

$$I_{cc} = \frac{1}{12} m(a^2 + b^2)\,, \tag{8}$$

where m is the mass of the whole satellite and a, b and c are the major axes and their dimensions (340x100x100 mm). The calculated values are shown in Table 9. More accurate values can be later obtained from CAD (Computer Aided Design) models, when the weight distribution of other subsystems comes clearer. The Aalto-1 satellite has two axes of maximum moment inertia and one with minimum moment of inertia.

Table 9: Moments of inertia of the Aalto-1 satellite.

I_{zz}	$33.3 \cdot 10^{-3}$ kg m^2
I_{xx}	$33.3 \cdot 10^{-3}$ kg m^2
I_{yy}	$6.66 \cdot 10^{-3}$ kg m^2

4.1.3 Payload

The Aalto-1 satellite has three technology demonstrator payload instruments:

- SPEC is a highly miniaturized adjustable imaging spectrometer developed by the Technical Research Center of Finland (VTT). The spectrometer is based on a tunable Fabry-Perot interferometer (FPI), which is either a piezo-actuated or a micro-electromechanical MEMS FPI. It is able to record 2D spatial images at one to three selected wavelength bands simultaneously. Its mission goals are successful imaging of a target area in Finland during the growing season and a successful scientific spectral imaging campaign during the mission life time. (Aalto-1, 2011c)

- RADMON is a radiation monitor developed by the University of Helsinki and the University of Turku. It is capable of detecting the fluxes of electrons and protons as a function of time and energy at energies that pose a threat to spacecraft operations. Accessible observations are inner radiation belt protons, outer belt electrons, solar energetic particle events and galactic cosmic rays. As the mission is likely to occur during the solar maximum activity, the experiment will provide valuable data on outer-belt dynamics in response to geomagnetic activity driven by solar eruptions. (Aalto-1, 2011d)

- EPB is an electro-static plasma brake developed by the Finnish Meteorological Institute (FMI). It is a simple de-orbiting device, based on an electric solar wind sail idea (Janhunen and Sandroos, 2007). A charged wire, or tether, in plasma will experience Coulomb drag from the plasma whenever the plasma is moving with respect to the tether. This fact can be utilized for efficient interplanetary spacecraft propulsion (electric solar wind sail) as well as for braking down, i.e. de-orbiting, LEO satellites. Aalto-1 will deploy a 10-100 m tether to test this proposition and possibly de-orbit the satellite at the end of the mission. (Aalto-1, 2011e)

The positions of the payloads, subsystems and the satellite fixed body coordinate axes are shown in Figure 18 (Aalto-1, 2011f).

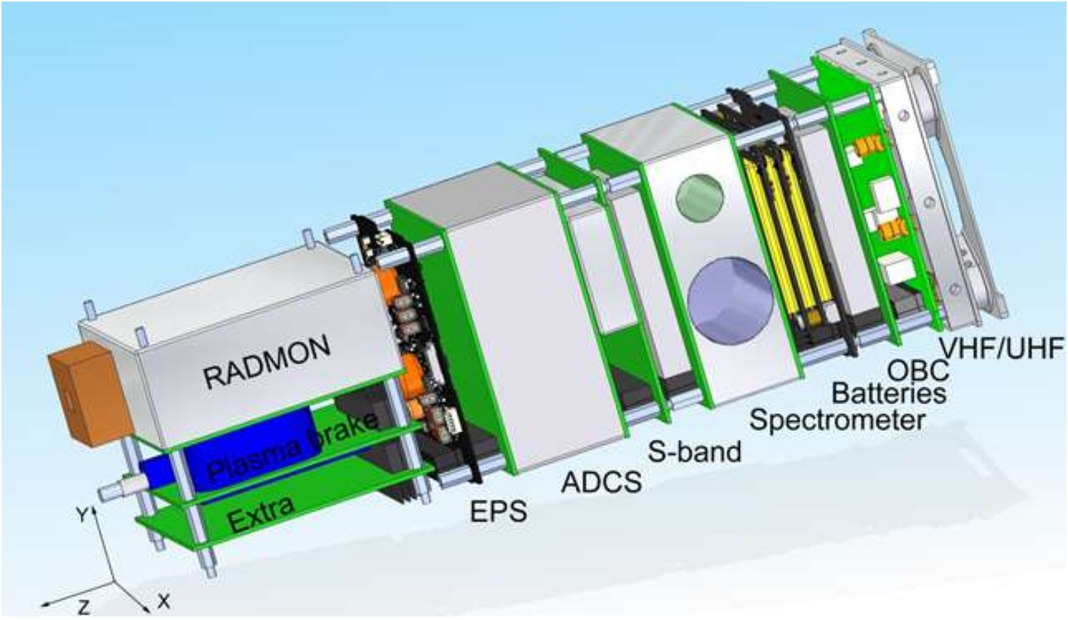

Figure 18: Aalto-1 satellite payload, subsystems and coordinate axes (Aalto-1, 2011f).

4.1.4 Mission

The mission of Aalto-1 is to successfully use and to space qualify the payload instruments. It consists of the following main mission goals:

- Successful imaging of Earth's surface done by the spectrometer from space.
- Successful measurement of the LEO environment done by the radiation monitor.
- Successful deployment and measurement of the braking force experienced by the de-orbiting device in LEO.

The secondary mission goals are:

- Successful de-orbiting of the satellite by the plasma brake.
- Successful measurement campaign from space completed by the spectrometer.
- Successful measurement campaign from space completed by the radiation monitor.

The mission of the Aalto-1 satellite is divided into phases (Aalto-1, 2011f). The overall mission lifetime goal is two years. These phases are presented from the ADCS point of view in the following:

1. Launch and Orbit phase starts from the satellite's integration to the P-POD and lasts to the separation from the P-POD, when it has reached the orbit. While integrated, the satellite has power off. The duration of this phase is assumed not to be longer than 24 hours.

2. Power up & De-tumbling phase starts from the satellite's separation from the P-POD. After separation the power turns on and the system is booted after 30 minutes. After the boot, the system does a full system check, puts the communication on beacon mode and starts de-tumbling into normal flight mode. This phase lasts until planned flight characteristics are achieved and the satellite is contacted.

3. Validation phase starts after successful de-tumbling and contact and lasts until all systems are validated.

4. In Observation Experiment phase, the spectrometer and the radiation monitor perform measurements. The main target for the spectrometer is Finland. The radiation monitor prefers pointing to east or west direction. The duration of this phase is six months to one year.

5. Electrostatic Plasma Brake Preparation phase start when the measurement campaign for the spectrometer ends. For the deployment of the Electrostatic Plasma Brake, the satellite needs to be spun. The spin creates centrifugal force, which keeps the tether straight and prevents it from getting tangled. The satellite is planned to be spun about the Earth's local vertical. This phase lasts until the required spin rate has been achieved. Measurements with SPEC and RADMON are most likely not performed in this phase.

6. Tether Deployment Phase starts after the required spin rate for tether deployment has been achieved. The mass of the tether slows down the spin. The attitude of the tether and the satellite can be adjusted by charging the tether or by using the ADCS. Measurements with SPEC and RADMON are not performed in this phase.

7. Stabilization and Drag Measurement phase starts when the tether is fully deployed. The drag force caused by the charged tether lowers the spin rate, which can be measured with the ADCS. The effect on orbit can be measured with the navigation sub-system. Measurements with SPEC and RADMON are not performed in this phase.

8. Braking Phase starts when the drag measurements have been completed and the satellite's spin has slowed down. The optimal scenario is that the tether will end up pointing downwards due to the gravity gradient force and the braking force. In this configuration the satellite slowly de-orbits into the Earth's atmosphere.

4.2 Requirement Specification

In this thesis, the Aalto-1 ADCS requirements are derived mainly from the mission, system, PA/QA and management requirements. The requirements presented in the following chapters are based on current best estimates received from payload teams and System, Mission and PA/QA & Management documentations or derived by analysis. Their purpose is not to define the requirements with best possible accuracy, but to instruct the supplier and to ensure all necessary needs are taken into consideration. They can be considered upper level requirements and need to be later supplemented with lower level requirements for more accurate definitions once the project's other requirements come clearer. An ADCS Control Plan is derived by analyzing the overall Mission Plan and can be utilized for determining the operational requirements for the Aalto-1 ADCS. The Control Plan for the Aalto-1 mission is presented in Appendix B.

In the Aalto-1 project, the satellite bus is designed around the payload's needs. These needs are, however, quite unclear due to simultaneous development of the payload instruments by the partners. This generates uncertainty, especially to the operational requirements. In a CubeSat project it is also unsure, what kind of launcher/orbit will be available for the launch, which makes the mission requirements inexact. For example, the available power may change for different orbits. Also, the system requirements for each subsystem are defined generously in the Aalto-1 project: A subsystem shall be compatible with the CubeSat Kit standard and the use of power, mass and volume are allocated between the sub-systems as the designs advance. These uncertainties in the requirements set by the mission, system and payload makes requirements specification for the subsystems difficult and also uncertain. The situation can be improved by incorporating margins to the mass, volume and power budgets as well as performance requirements. Insufficient use of margins may create huge problems, if subsystems do not absolutely meet the requirements set for them in the beginning, which is very common.

By analyzing the Preliminary Mission Design & Operations (Aalto-1, 2011g), it is clear that the satellite has four control objectives:

- The spectrometer has to point close to nadir direction.
- The S-band antenna has to point close to nadir direction.
- The Radiation monitor has to point close to east or west direction.
- The Electro-static Plasma Brake has to be spun for deployment.

A set of requirements shall be defined, to ensure reaching these objectives with sufficient performance and reliability. By analyzing the Preliminary Mission & System Requirements (Aalto-1, 2011b), the ADCS requirements caused by the mission requirements can be determined:

M1. The ADCS shall be able to withstand the launch environment
M2. The ADCS shall be able to withstand the LEO environment for at least 2 years.

M3. The ADCS shall be able to control the pointing of two instruments on orthogonal axes simultaneously. (3-axis stabilization)

The first two requirements, M1 and M2, define the limits to the durability of the design. M1 cannot be defined more accurately until the decision on what launcher to use is made. M2 instructs the supplier about the duration and levels of radiation and other environmental stresses, which the ADCS has to withstand. Estimates for the launch and orbit stresses can be found from, for example, ECSS standards. The exact stresses can be defined later and supplemented as lower level requirements. M3 is derived directly from the control objectives and defines the required stabilization method.

Preliminary Mission & System Requirements define also:

S1. The mechanical interface shall be CubeSat Kit compatible.
S2. The electrical interface shall be CubeSat Kit compatible.
S3. The electrical interface shall use RS485, I2C or LVDS.
S4. The CubeSat Kit Connector shall go through the ADCS.
S5. The ADCS's dimensions shall be less than 95 mm · 95 mm · 57 mm.
S6. The ADCS's mass shall be less than 500g.
S7. The ADCS's nominal power consumption shall be less than 1.5 W and peak power consumption less than 2W.

The requirements S1, S2 and S4 originate from the requirement that the subsystems shall be CubeSat Kit compatible. S3 comes from the allowed interface protocols, which are implemented to the OBC. Currently, the LVDS interface is planned for communicating with the payload. Having a dedicated communication line such as RS485 between the OBC and the ADCS would prevent a propagation of a failure in the payload communication block. Requirements S5-S7 are determined by analyzing possible ADCSs from the market and allocating sufficient amount of volume, mass and power from the total budgets. The given values are without margins, as they are later added by the system engineer in budget calculations.

Also, the Product Assurance Plan (Aalto-1, 2011h) and the Management Plan define requirements:

P1. A mechanical model and an electrical model or alternatively an engineering model shall be delivered before June 2012.
P2. A flight model shall be delivered before January 2013.

The given dates are estimated to give enough time to ensure thorough verification before flight. Additional requirements, analyzed from the Product Assurance Plan are presented in Chapter 4.5. Also, operational requirements are derived in the Chapter 4.6. A complete numbered list of all the requirements is presented in Appendix A. Each of the ADCS requirements specified must also be verified somehow. A method, schedule and approval criteria must be defined for every requirement. The Verification Plan for the Aalto-1 ADCS, which takes into account the upper level requirements derived earlier, is presented in Appendix C.

4.3 System selection

The analysis leading to a decision between in-house design or procurement and the preliminary ADCS type selection for the Aalto-1 project is presented in this chapter. The requirements presented in the last chapter determine that the satellite has to be stable in respect with two different axes, which prevents using spin stabilization. 3-axis stabilization is the only method to meet these requirements at once. A simple 3-axis stabilization control systems using only magnet torquers and magnetometers can barely reach one degree accuracy. On a polar orbit, the accuracy may be even less as the magnetic field line orientations close to the poles may vary and thus the magnetometer accuracy decreases. For these reasons, more advanced systems need to considered, especially as the main payload instrument spectrometer's target imaging area is quite close to the North Pole. The required system should thus use better sensors than just sun sensors and magnetometers, as well as internal torquers in addition to magnet torquers and also complex control algorithms.

Due to the complexity of the required ADCS, a decision between design by the university or procurement from a supplier must be carefully assessed. The following topics presented in Chapter 3.2 are discussed:

- The complexity of the required design:

  The design of the required ADCS may come very complex. The integration of the system components of the ADCS is plausible but the software would require implementation of complex algorithms for target tracking and spin maneuvers.

- Expertise to design an ADCS:

  Aalto University has no experience in designing attitude determination and control systems for satellites. The main subject of study in the Department of Radio Science and Engineering (and previously Laboratory of Space Technology) in Aalto University is remote sensing. Rovers are studied in the Department of Automation and Systems Technology and aircraft control systems from the dynamics side are studied in the Department of Applied Mechanics (and previously Laboratory of Aircraft Engineering). The design of an ADCS does not really fit to any of these areas and should be carried out in collaboration between these departments. At the moment, designing an ADCS would be of interest for the Department of Automation and Systems Technology but there is neither previous experience nor suitable test facilities. Also, the project lacks people with strong programming experience for the software implementation.

- Desire for knowhow in ADCS design:

  As mentioned before, the design of ADCS type systems is not one of the current strong points in Aalto University. The design of an ADCS for satellite projects is not seen as a useful task in the long run. There may be synergies with other automation control projects, which need to be considered in future.

- Availability and cost of systems in the market vs. design cost:

  The use of high accuracy ADCSs in CubeSats is uncommon. Commercial systems are currently being designed for the market and there are also a few already flown systems. The price for these systems is, however, generally much less than the cost of a design process.

- Time limits and management:

  The time limits in nanosatellite projects are generally very short. The Aalto-1 project has a design timeline of about four years. The design could be done in these time limits only if great investment of time and money would be made. Designing the ADCS can be considered also risky from both the system failure side and in case of delays.

- Availability of test facilities

  Aalto University does not currently have any testing equipment for the ADCS. The testing would require at least thermal vacuum chambers, vibration benches and a system to test the control operations. The procurement of the other systems would have to be a great investment and is not reasonable for the purposes of only one project.

The discussed topics show almost unanimously that procurement from a supplier is the preferred option. Even though the system is procured, the educational objectives of the project are fulfilled as the requirements specification, integration and operations require knowledge of the ADCS's design and operation. Also, the procurement increases understanding of how to procure systems and teaches and promotes international cooperation. The knowledge gathered from this project can be, however, used also in upcoming satellite projects and the design/procurement decision can be re-evaluated.

4.4 Procurement

Once the decision to procure the ADCS was made, the market was searched for potential systems. Three ADCS solutions were found from the market, which offer good enough performance with suitable dimensions and low power consumptions:

- MAI-400 from Maryland Aerospace (USA)
  - 10 cm · 10 cm · 5 cm
  - 3 reaction wheels
  - 3 torque rods
  - 3-axis magnetometer
  - 2-camera Earth Horizon Sensor
  - Coarse sun sensor
  - Computer with software

The MAI-400 offers sufficient performance (<1° pointing accuracy) and is from a company that has a good flight history for its ADCSs. On the downside it is under US International Traffic and Arms Regulations (ITAR), which may complicate the procurement and the whole satellite project management. For example, the names and tasks of all the students taking part in the project must be informed to the US authorities at all times and the launch site must be approved by them. The MAI-400 is designed to be integrated to the end of the CubeSat stack, which is not a preferred place for Aalto-1's configuration. Integration to the center part of the satellite is possible, but may require relocating the earth horizon sensors or replacing them with sun sensors, which lowers its accuracy.

- Satellite Services ADCS from Satellite Services ltd (Great Britain)
  - 9.1 cm · 9.1 cm · 3 cm
  - Momentum wheel
  - 3 torque rods
  - 3-axis magnetometer
  - 6 sun-sensors
  - 3-axis rate gyro
  - GPS receiver
  - Computer without software

The Satellite Services ADCS offers good estimated performance (<1° pointing accuracy) and has a GPS receiver, which is desired for the plasma brake experiment. The company has a prototype flying in the near future for attitude determination test purposes. A major downside is the need to purchase the control software separately.

- iADCS-100 from Berlin Space Technologies (Germany) and Syspa (Netherlands)
    - 9.5 cm · 9.0 cm · 5.7cm
    - 3 reaction wheels
    - 3 torque rods
    - 3-axis magnetometer
    - 3-axis rate gyro
    - 3-axis accelerometer
    - Star tracker
    - Computer with software

The iADCS-100 offers the best estimated performance (<0.02° pointing accuracy) provided mainly due to its star tracker. The system is currently under development and is thus quite a risky choice although their attitude control algorithms have been used in previous missions successfully. The companies designing the iADCS-100 also allow taking part in the development, which would support the educational objectives of the Aalto-1 project.

These three systems are compared in a trade-off Table 10 to help in evaluation for the best possible system. The table compares different attributes from the fields of the economic point of view, the credibility of the supplier, the technical solution, education and the feasibility. This thesis rates the different attributes from one (the worst) to three (the best) for each system and a result is calculated as an un-weighted average.

Table 10: Trade-off between possible systems.

	Technical Solution	Feasibility	Credibility	Education	Price	Average
MAI-400	2	2	3	1	3	2.2
Satellite Services ADCS	2	2	2	2	3	2.2
iADCS-100	3	3	2	3	3	2.8

The technical solution considers performance, dimensions and power consumption, which are reviewed to be best in iADCS-100. The feasibility considers how well the system fits to the planned mission, other system and its modifiability. iADCS-100 receives the highest rating also here as its interfaces and operation can be fully customized according to the system and mission requirements. The credibility considers how well the supplier can be trusted in supplying the system in due time and promised performance. The MAI-400 receives the highest rating here as it is a COTS system with good flight heritage. This, on the other hand gives it a low rating in education, whereas the iADCS-100 receives the highest rating because the system can be developed in cooperation with the supplier. Finally, the price does not affect the decision as it is approximately the same with all system. The iADCS-100 gets the best result and is selected as the primary choice for the attitude determination and control for Aalto-1. The MAI-400 is a somewhat similar system, which can be integrated quite easily in iADCS-100's place and in rapid timescales if problems with the iADCS-100 occur. It is thus selected as a possible back-up system. It should be noted, that it is under US international traffic and arms regulations (ITAR), which may hinder the project's cooperation with international parties and the timescale of procuring the system may be longer than anticipated due to the involvement of the US authorities. Thus, the progress of the iADCS-100's design must be monitored closely early on, as the decision to change the system must be made in due time.

Aalto University requires that an Invitation To Tender is published prior purchasing anything costing over 30 000 € or the direct purchase must be very well justified. A direct purchase may be possible, for example, if the system required is purpose build for the project and is for scientific use only. The invitation to tender consists of a minimum requirements specification, criterion for supplier eligibility, criterion for tender comparison, procurement procedures and contract term definitions. The minimum requirements specification is a crucial part of the invitation and shall state the model philosophy, schedule, testing and documentation requirements in addition to system and performance requirements. The criterion for tender comparison can be done in multiple ways: comparing only the price, only the quality factors or in combination. The quality factors fall generally into the attributes in Table 5, but have to be defined more accurately to ensure unbiased comparison. The invitation to tender ensures that the procurement is legal and also works as the first binding contract. If only the price is compared, the minimum requirements have to be very strict. This ensures that all systems reaching the requirements are viable options. This is a common approach when procuring highly specific systems. Even if the ADCS is purchased directly and an invitation to tender is not required, the buyer should make sure the supplier fulfills similar requirements and terms as in the invitation to tender. For this reason, the ADCS requirements should be well defined at this stage.

After the primary supplier for the ADCS is selected, the contract negotiations may begin. The contents of a good contract were presented in Chapter 3.3. The requirements need only to be discussed in more detail with the selected supplier in addition to legal/warranty terms.

In addition to the ADCS models itself, the supplier should provide ground support equipment necessary to transport, integrate and operate the ADCS (Requirement P-3) and documentation (Requirement P-4) such as:

- Interface Control Document ICD / Manual
- Test Specifications & Test Report
- Functional Test Report including a procedure to do functional tests at Aalto University for product initial test
- Copy of statement of origin and relevant export papers

4.5 Integration & Verification

The general verification philosophy in the Aalto-1 project is to test all subsystems first separately, then integrate them and do the flight qualification tests for the whole satellite after this. The designer of the ADCS is also required to perform FMECA and FDIR analysis for the ADCS to increase its reliability. The understanding of possible failures of major components and their effects on the mission is beneficial as they may raise questions and bring out critical items already early in the project. The planned testing is presented and a preliminary FMECA is performed in this chapter.

4.5.1 Tests

The Aalto-1 project's Product Assurance plan (Aalto-1, 2011h) defines the following tests to be performed:

- Functional tests
- EMC test
- Thermal vacuum bake-out
- Thermal vacuum cycling with qualification levels
- Mechanical tests with flight qualification levels
    - Vibration
    - Shock

As already mentioned, the Aalto-1 project is manufacturing a proto-flight model. This model has to be qualified using qualification test levels. To perform these tests, the satellite needs to be fully integrated with all systems. Performing tests to the systems may cause stress to them, thus excess testing should be avoided. The supplier may anyhow be asked to perform acceptance level tests (Requirement P-5) for the system to discover possible problems already before the delivery, and to reduce the possibility of failures in the qualification level testing. The thermal vacuum bake-out test is used to remove any moisture from the flight model and is not required for the acceptance level.

The functional tests can be started already with the electrical or engineering model to address possible problems early. The qualification test levels cannot be defined exactly before the launch vehicle has been selected. The Preliminary Verification Plan on how to conduct the verification of the ADCS requirements is presented in Appendix C. The verification schedule, procedure and criteria are defined for each requirement.

4.5.2 Analysis

The supplier of the ADCS is required to perform FMECA to understand the critical parts of the system for reliable operation (Requirement P-6). FDIR shall be done for all critical parts to ensure that detection, isolation and recovery can be performed, in case of failures (Requirement P-7). Automatic failure detection and isolation is desirable and should be implemented to the ADCS computer or the OBC. These analyses and procedures have not yet been completed and thus cannot be presented in detail in this thesis. However, a preliminary FMECA is performed and possible FDIR procedures are discussed in this chapter to present the analysis method, to find possible critical components and to launch the failure mitigation planning. In this thesis, the criticality classification (Catastrophic, Major, Minor) follows the ECSS (2009b) standard. The results have been discussed with the Aalto-1 team for additional input.

Failure: ADCS Computer
Effects: No attitude determination or control
- No pointing for spectrometer
- No pointing or attitude data for radiation monitor
- No measurement data or controlled spin for plasma brake
- No pointing for S-band antenna

Criticality: Catastrophic
Mitigation: A failure making the ADCS computer un-operational would have a catastrophic impact on the science mission of Aalto-1 as it would prevent proper use of every payload instrument. Redundancy in hardware or software should be carefully considered and comprehensive testing shall be conducted to ensure reliable operation. Also, the OBC should be able to reboot the ADCS computer.

Failure: ST-200 Star Tracker
Effect: Attitude determination accuracy lowered impacting also control accuracy (exact decrease TBD)
- Lowered pointing accuracy for spectrometer
- Lowered pointing accuracy and attitude data for radiation monitor
- Lowered pointing accuracy for S-band antenna

Criticality: Minor
Mitigation: The science mission of Aalto-1 would be affected, as the accuracy of science measurements for the spectrometer and radiation monitor is lowered. The attitude determination can, however, still be done using the magnetometer and the gyroscope. Redundancy could be increased by adding sun sensors to the ADCS or by implementing the ability use the solar panel voltages for attitude measurement. The plasma brake experiment does not suffer from lowered accuracy as much as the other payload measurements.

Failure: Reaction Wheels
Effect: Attitude control accuracy lowered (exact decrease TBD)

- Lowered pointing accuracy for spectrometer
- Lowered pointing accuracy for radiation monitor
- Lowered pointing accuracy for S-band antenna

Criticality: Minor
Mitigation: A reaction wheel failure would lower the overall pointing accuracy but measurement data would remain unchanged. The lowered accuracy would hinder mostly the spectrometer measurements. Attitude control can be still performed using the magnet torquers. The failure of a single wheel is the most common and the system shall be designed in such way that other wheels can be still operated.

Failure: Magnet Torquer
Effect: Momentum dumping capability lost causing loss of stability (per axis)

- Probable loss of attitude control after certain TBD period
- No controlled spin for the plasma brake experiment

Criticality: Major
Mitigation: Magnet torquer failure would mean losing the only external control method of the ADCS. This would cause losing momentum dumping capability per axis per magnet torquer. The reaction wheels would saturate after a certain period due to disturbance torques. An algorithm which could perform momentum dumping for all axes, using only one or two magnet torquers, could be considered to be implemented for the ADCS. Losing all magnet torquers would also prohibit performing a controlled spin for the plasma brake experiment. The failure in magnet torquers is, however, unlikely as they are considered very robust due to the simple design.

Failure: 3-axis Magnetometer
Effect: No measurement data from the magnetic field
Criticality: Minor
Mitigation: The effects of a magnetometer failure are currently pretty unclear. The magnetometer may be required at least for the spin rate calculations during plasma brake deployment. On stable operations, the system may use the gyro and the star tracker for attitude determination.

Failure: 3-axis MEMS Accelerometer
Effect: No acceleration data
Criticality: Minor
Mitigation: This would be the least harmful failure as the payload instruments do not require accelerometer data. It is used only as an extra sensor for the plasma brake experiment for drag force measurements.

Failure: 3-axis MEMS Gyro
Effect: Lowered angular velocity data causing lowered attitude determination accuracy impacting also control accuracy

- Lowered pointing accuracy for spectrometer
- Lowered pointing accuracy for radiation monitor
- Lowered pointing accuracy for S-band antenna
- Lowered accuracy angular velocity data for the plasma brake experiment

Criticality: Minor
Mitigation: Failure in the gyro would cause lowered attitude knowledge and control.

This preliminary failure analysis shows clearly that a single component failure would very unlikely cause a total mission failure due to redundancy by the use of multiple sensors and torquers. The ADCS computer failure is the only occurrence that would jeopardize the whole mission. It is thus selected as a critical component and great care must be taken in the design and testing of it, to ensure reliable operation. Also, reaction wheels can be considered unreliable due to their mechanical structure, but in case of failure they fortunately hinder the operations only slightly. Anyhow, the reaction wheels shall be tested thoroughly and the ADCS computer shall be able to detect their failures. Other failures that need to be detectable and isolated by the ADCS computer or the OBC include at least:

- Over-increased power consumption
- Conflicting measurements
- Extraordinary measurement data

After the FMECA has been completed and the most critical components have been defined, a deeper analysis about the effects as well as isolation, recovery and back-up plans shall be implemented for single and multiple component failure occurrences. . The usage of sensors and torquers for different operation modes may also change as the planned operations may change and cause different effects to the mission. At this moment, it is anyway clear that a fully manual diagnostics mode (Requirement O-1) is required for troubleshooting the satellite in case of failures.

4.6 Operations

The operation of the satellite from the ADCS's point of view is defined in the Control Plan. The control operations are organized to a chronological order. A specific attitude, operation mode, telemetry transfer, operating commands, power consumption and duration shall be specified for every operation phase. The phases in the operation of Aalto-1 are presented and analyzed in this chapter. Also, two calculations are presented to show how operational requirements can be determined and how the Control Plan can be developed. The analysis brings out results for the minimum performance, to be able to perform these operations, and thus helps in specifying the ADCS requirements. The Control Plan is implemented to a spreadsheet and is provided in Appendix B. The different phases of the plan, the maneuvers required in performing them and the transitions between the phases are analyzed in this chapter.

4.6.1 De-tumbling

The control operations begin with de-tumbling. The satellite has an unknown attitude after it has separated from the P-POD and the antennas have been deployed. The ADCS shall have a safe-mode (Requirement O-1), which it uses to stabilize the satellite and point it to the normal flight position, presented in Figure 19.

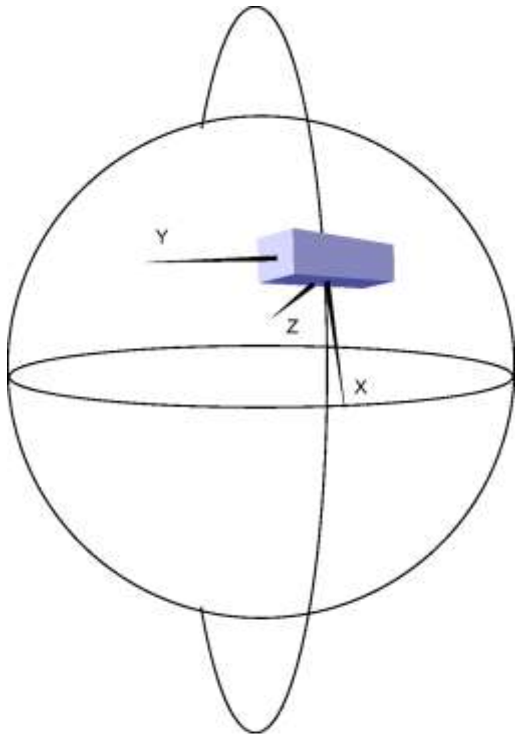

Figure 19: Normal flight position in Satellite Orbital Reference coordinate system.

In this position the satellite travels along the orbit (x-axis), points the radiation monitor instrument in the end of the satellite stack to east or west direction (±y-axis), and at the same time the spectrometer instrument to nadir direction (z-axis) for the duration of the whole orbit. The axes are orthogonal with each other and the satellite's sides which they pass through. For a circular orbit, this position can be maintained without changing the angular momentum of the satellite. In this stage, the ADCS shall send telemetry data to the ground station to able to verify that the satellite is indeed stabilized. The duration of this phase can be estimated by calculating the time to stabilize from maximum expected spin rates.

4.6.2 Commissioning

In this phase, the whole satellite is commissioned to verify its operation under space conditions and to calibrate the measurement instruments. All operation modes and commands to operate the ADCS should be tested. The testing of the spin operation mode, meant for the late mission plasma brake experiment should be carefully assessed. A possible failure caused by its testing at this point might endanger the rest of the mission. The satellite's attitude is maintained in normal flight position as it is also used in the actual measurements. This phase lasts until all instruments and subsystems have been tested and calibrated.

4.6.3 Radiation measurements

Radiation measurements are done continuously for the whole duration of the normal operations period of the mission. The power and data transfer budgets may however limit the measurements to shorter time periods. The radiation monitor shall point to east or west direction. This requires maintaining the normal flight position i.e. nadir pointing. The orbit of the satellite does not always go over the ground station or other area of interest and thus imaging with the spectrometer or the S-band communications may require tilting the satellite. This changes the radiation monitor direction. The radiation monitor measurements prefer maximum 10 degrees of movement from the east/west direction but larger deviation are also allowed (Aalto-1, 2011d). More important is that the deviation from the normal direction is known in one degree accuracy and it can be compared with the measurement data. This phase lasts for the whole duration of the normal operations.

4.6.4 Spectrometer imaging

The accuracy required for spectrometer imaging is calculated by the spectrometer team as it depends on the optics and exposure times. The minimum pointing accuracy and pointing knowledge has been defined to be 1 degree (Requirements O-2 and O-3) (Aalto-1, 2011c). Vibrations, i.e. jitter, inside these limits may also affect the measurements and should be defined (Requirement O-7). The spectrometer may take pictures in nadir (normal flight position) or in target tracking modes (Requirement O-1). In target tracking, the accuracy will be, most probably, higher and the exposure times can be longer, but the picture is not always necessarily taken from the same angle for repeated measurements. Pointing of the spectrometer to target, while passing over the area of interest requires rotating the satellite more than in normal flight. In the following, an equation for the required rotation, i.e. slew rate, for target tracking is derived for multiple orbit altitudes.

The satellite's spectrometer needs to change its pointing direction to follow a target on the surface of the Earth. The amount of change in direction is defined as α in Figure 20.

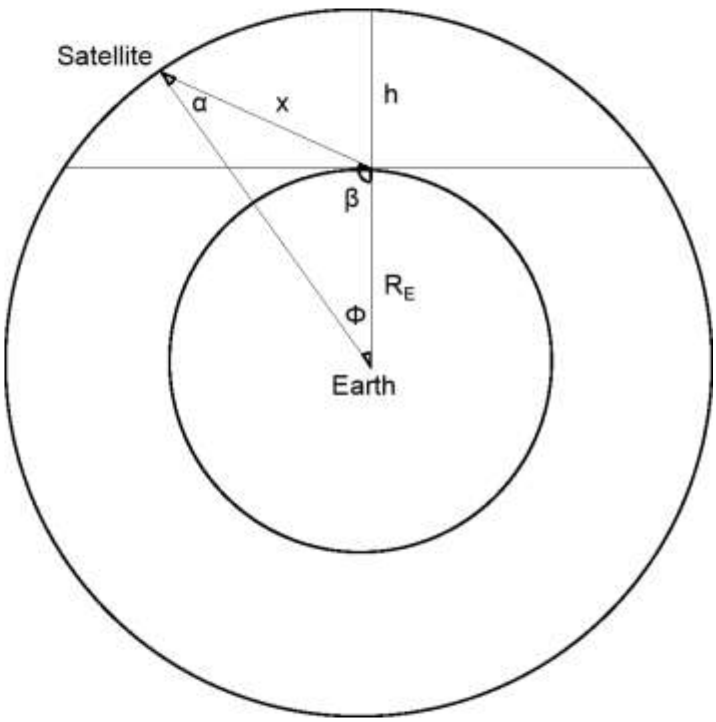

Figure 20: Geometry for satellite target tracking.

The rate of change of α, i.e. the slew rate, can be calculated by derivation. α changes in relation to the geocentric semi-angle ϕ. This can be defined starting from the law of sines (Alonso and Finn, 1992)

$$\frac{R_E}{\sin(\alpha)} = \frac{R_E+h}{\sin(\beta)} = \frac{x}{\sin(\phi)}, \tag{9}$$

where R_E is the radius of the Earth (6380 km), h is the altitude of the satellite. β is the angle between the vector from the Earth's center of mass to the target location and the vector from the target location to the satellite. x is the distance between the target location and the satellite. It is also known that

$$\beta = 180\,^{\circ} - (\phi + \alpha). \tag{10}$$

By combining Equations (9) and (10), the relation between α and ϕ can be solved to be

$$\alpha = \tan^{-1}\frac{\sin(\phi)}{\frac{R_E+h}{R_E} - \cos(\phi)}. \tag{11}$$

ϕ changes in relation to time (Fortescue, et al., 2003) as

$$\phi = \omega t = \frac{vt}{R_E+h} = \frac{\sqrt{\mu}t}{(R_E+h)^{3/2}}, \tag{12}$$

where ω is the angular velocity of the satellite's rotation around Earth's center of mass, v is the satellite's orbital velocity, t is the time and μ is the Earth's gravitational parameter $3.986 \cdot 10^{14}$ m^3/s^2.

The equation for the required slew rate $\dot{\alpha}$ for target tracking can be obtained by inserting Equation (12) to Equation (11), and deriving it in relation to time:

$$\dot{\alpha} = \frac{\frac{\omega * \cos(\omega t)}{\frac{R_E+h}{R_E} - \cos(\omega t)} - \frac{\omega * \sin^2(\omega t)}{(\frac{R_E+h}{R_E} - \cos(\omega t))^2}}{\frac{\sin^2(\omega t)}{(\frac{R_E+h}{R_E} - \cos(\omega t))^2} + 1}. \tag{13}$$

Figure 21, showing the required slew rate for multiple altitudes, is obtained by inserting Equation (13) to MATLAB. The source code used in this thesis is given in Appendix D. Figure 21 shows the first pass over the target after one full revolution around the Earth. The required slew rate is at maximum when the satellite passes over the target position. The maximums are at different time for each orbit as one revolution takes longer for the higher orbits. For typical CubeSat orbits 600-900 km, the required slew rate at the pass is about 0.7 °/s. If the satellite has initial rotation, for example for nadir pointing purposes, the required slew rate for target tracking will be the absolute value of the division. At this point, the requirement for the slew rate is selected to be 1.5 °/s, to incorporate a margin due to uncertain orbit altitude and required operations (Requirement O-4). The slope of the curve gives the angular acceleration required. For a 600 km orbit, the highest angular acceleration required is about 0.005 °/s². Due to uncertainties, the requirement for the minimum angular acceleration is selected to be 0.01 °/s² (Requirement 0-8).

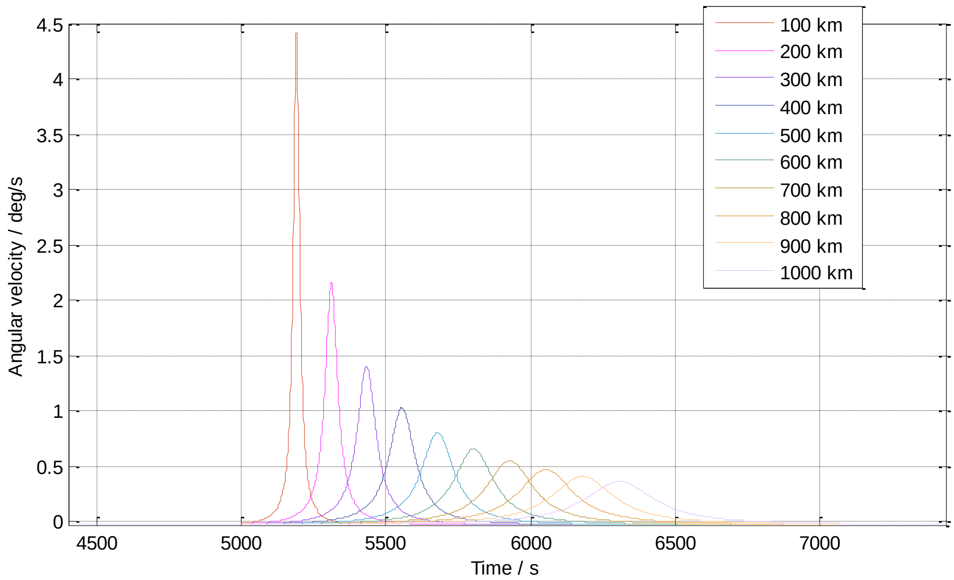

Figure 21: Required slew rate for multiple orbit altitudes.

4.6.5 S-Band communications

The S-band communications may be pointed similarly as the spectrometer. The accuracy required is currently estimated to be lower and thus nadir pointing can be used, at least, for close passes over the ground station. A longer S-band radio communications period may however be possible using the target tracking mode if required.

4.6.6 Plasma Brake deployment

After the measurement campaign of the spectrometer and the radiation monitor has ended (approximately after 6 months of measurements), the plasma brake experiment begins. For the deployment, the satellite must be spun at least 200 °/s in the Earth's equatorial plane, i.e. parallel to the Earth's rotation axis (Aalto-1, 2011e). The maximum spin speed error shall be less than 10% and the spin axis direction error shall be less than 10° (Requirement O-5). The spin axis of the satellite shall be the axis of maximum inertia to achieve long term stability.

The ADCS shall have a spin operation mode (Requirement O-1), which allows active attitude control during spinning. All sensors may not work properly in high spin speeds. Thus the operation mode must be designed to use only functional sensors (e.g. the magnetometer). Some sensors, i.e. the star tracker, may be unable to operate, at least with full accuracy, in high spin speeds resulting in lowered accuracy of attitude determination and control. The accuracy should be, however, sufficient as the accuracy in the plasma brake experiment is required to be much less than in spectrometer operations. The solar panel voltages may be also used as a coarse sun sensor, but requires solar panels or additional sun sensors to all sides of the satellite. The satellite's capability to recover from the spin to a stable state (Requirement O-6) must be also analyzed in the future. The deployment of the tether increases the satellite's moment of inertia and slows down the spin, and thus it may be necessary to accelerate the spin even during deployment. According to current calculations from the plasma brake team, the centrifugal force to the tether's end mass will maintain it straight during acceleration. In the end of the deployment and before measurements, the spin rate should be about 72 °/s.

The duration of the spin operation from a steady state to 200 °/s must be calculated for the Control Plan. The spin is accelerated with the magnet torquers of the ADCS only in the equatorial plane, which means, only the x-component of the Earth's magnetic field strength (NASA, 1969) can be used:

$$\boldsymbol{B}_X = (\frac{R_E}{R_E+R})^3 g_1^0 sin\theta, \tag{14}$$

where R_E is the radius of the Earth, R is the radius from the center of the Earth to the satellite, θ is the geocentric colatitudes and g_1^0 is the harmonic coefficient:

$g_1^0 = 30401.2$ nT .

As can be seen from Equation (14), the magnetic field strength varies at different latitudes, and is highest near the poles for the x-component. For this reason, an average field strength is calculated with equation (15) using a 700 km orbit altitude.

$$\boldsymbol{B}_{avg} = \frac{\int_0^{\pi/2} \boldsymbol{B}_X}{\pi/2} = 1.744 \cdot 10^{-5}\, T. \tag{15}$$

The magnetic torque rods of the ADCS generate a dipole moment of approximately 0.2 Am^2, which causes them to rotate the satellite inside the Earth's magnetic field. The torque generated (Fortescue, et al., 2003) is proportional to the magnetic field strength according to equation (16).

$$\boldsymbol{T}_{B_X} = \boldsymbol{m} \cdot \boldsymbol{B}_{avg}, \tag{16}$$

where $\boldsymbol{m}$ is the dipole moment. In the equatorial plane the ADCS using 3-axis torque rods or coils is able to rotate the satellite when operated in turns. The dipole moment direction must be also rotated in relation to the magnetic field direction to keep the generated torque in the correct direction. The available dipole moment for the acceleration thus changes in different parts of the orbit. The average value of the dipole moment is

$$\boldsymbol{m}_y = \frac{\int_0^{\pi/2} \boldsymbol{m} \sin(\theta)}{\pi/2} = 0.127\ Am^2\ , \tag{17}$$

where $\boldsymbol{m}$ is the dipole moment of the ADCS 0.2 Am^2. A rotational acceleration (Fortescue, et al., 2003) about the z-axis, which can be achieved is

$$\dot{\omega_Z} = \frac{\boldsymbol{m}_y \cdot \boldsymbol{B}_{x_{avg}}}{I_{zz}} = 5.52 \cdot 10^{-5} 1/s^2. \tag{18}$$

The duration for the spin to reach 200 °/s is then

$$t = \frac{\omega_z}{\dot{\omega_Z}} = 63236\ s \approx 1054\ min \approx 17.6\ h\ . \tag{19}$$

Further analysis needs to be performed taking into account hysteresis and the ADCS's ability to switch voltages between the magnet torquers. The actual dipole moment, which can be utilized for the spinning will, most probably, be lower especially at higher spin rates, which will lengthen the duration. In reality, accelerating the spin may be limited to only a short part of the orbit, where the Earth's magnetic field lines are close to the equatorial plane direction. Spinning the satellite outside this region provides only little acceleration but the same power consumption as near the poles.

4.6.7 Plasma Brake measurement

The braking force of the plasma brake can be measured using the magnetometer, the gyroscope and possibly sun sensors in the ADCS's measurement mode (Requirement O-1). The plasma brake interacts with the ionosphere while it is on lowering the spin rate of the satellite and its orbit. The torquers of the ADCS should not be used during the measurements as it would interfere with the measurement data. Thus, the mode has to be changed to spin mode if additional acceleration between the measurements is required. In the end of the measurements, the plasma brake may be left to brake the satellite and it will reach a stable state, where both the braking force and the gravity gradient force pull the tether. In about two years after the braking has started, depending on the orbit and the braking force, the satellite should burn in the atmosphere.

5 Conclusions

In this thesis, a straightforward procedure for implementing an ADCS, shown in Fig. 7, was developed. This procedure provides a framework for methodical engineering of a highly mission critical system and a base for further research.

The requirements specification plays a significant role in the overall implementation procedure as it defines the desired outcome and guides through the rest of the implementation procedure. Thus, great care needs to be employed to define and to maintain the conformity of all requirements throughout the project. A thorough and timely requirements specification is especially important if the system is procured, as the requirements need to be unambiguously identified for the official tender invitations and contract negotiations. Accurate requirement specification can be challenging due to uncertainties in requirements concerning the launcher, orbit as well as payload details. Often, the payload is designed concurrently with the satellite system. Problems can be mitigated by documenting all requirements, even if they cannot be defined accurately immediately and by leaving sufficient margins. This provides an overview of the whole system and a more unified framework for system engineering. By following the requirements specification presented in this thesis, the requirements for an ADCS can be defined rapidly and comprehensively.

Space control engineering requires knowledge from many disciplines. This comes even more evident in a nanosatellite project, where single persons can be responsible for complete subsystems, such as the ADCS. This multidisciplinary character causes a vast amount of time to be spent on finding all the necessary information. In order to succeed in this multidisciplinary and rapid design environment, it is necessary to have documented procedures, in addition to the actual requirements and designs obligatory for design reviews. This allows new students joining the project, to get a quick overall view of the development process and use the procedures for actual work to meet the deadlines. High quality reference procedures can be applied to future tasks and projects as well.

The presented procedure can also be used if the ADCS is not procured, by replacing the procurement sub-process with a design process. Utilizing this framework in an ADCS design project allows a quick, design focused and reliable approach. This can be very beneficial especially in a nanosatellite project, where the available time is usually very limited.

5.1 Aalto-1 ADCS Use Case

At the time of writing, the Aalto-1 project has progressed to the start of detailed design. The ADCS implementation procedure described in this thesis allowed a methodical step-by-step identification of top-level requirements for attitude control of a nanosatellite. The derivation of upper level ADCS requirements (Appendix A), outlining control and verification plans (Appendixes B and C) highlight important design aspects affecting the whole Aalto-1 mission. For example, the Control Plan provides input for system engineering (operation modes) as well as software engineering (telemetry). Also, the outcome was used in preparing an Invitation To Tender for purchasing a complete system from a third party.

Some tasks in the integration, verification and operations for the Aalto-1 mission are not presented in this thesis. All subsystems are under development and more specific lower level requirements, e.g. for testing, cannot be derived until the design is sufficiently mature. Selected major tasks to be completed during the ongoing detailed design phase are listed here:

- ADCS-OBC software integration: Once the software of the ADCS and the OBC has advanced, the software integration can be started. The telecommands and telemetry used in operating the ADCS have not yet been defined. This is an essential part of the interface between the ADCS and OBC.

- Test procedures: The supplier of the ADCS will provide test procedures for functional tests. However, tests procedures, for instance, for the ADCS-OBC software integration, must be designed by the Aalto-1 team.

- Ground Station Control Software: The supplier of the ADCS does not provide the ground station software. In case a graphical user interface is considered useful, for example, for telemetry visualization, it needs to be designed and tested by the Aalto-1 team.

- Detailed analysis: More analysis needs to be carried out for specifying missing requirement details as the subsystem designs progress. Especially the effect of the space environment and EMC on the required mission operations is currently not sufficiently well understood.

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Appendix A Aalto-1 ADCS Requirements

M1. The ADCS shall be able to withstand the launch environment
M2. The ADCS shall be able to withstand the LEO environment for at least 2 years.
M3. The ADCS shall be able to control the pointing of two instruments on orthogonal axes simultaneously. (3-axis stabilization)

S1. The mechanical interface shall be CubeSat Kit compatible.
S2. The electrical interface shall be CubeSat Kit compatible.
S3. The electrical interface shall use RS485, I2C or LVDS. (TBC)
S4. The CubeSat Kit Connector shall go through the ADCS.
S5. The ADCS's dimensions shall be less than 95 mm · 95 mm · 57 mm.
S6. The ADCS's mass shall be less than 500 g.
S7. The ADCS's nominal power consumption shall be less than 1.5 W and peak power consumption less than 2 W.

P1. A mechanical model and an electrical model or alternatively an engineering model shall be delivered before June 2012.
P2. A flight model shall be delivered before January 2013.
P3. The supplier shall provide ground support equipment necessary to integrate and operate the ADCS.
P4. The supplier shall provide at least the following documentation:
   a. ICD / Manual
   b. Test Specifications & Test Report
   c. Functional Test Report including a procedure to do functional test at Aalto University for product initial test
   d. Copy of statement of origin
   e. Copy of relevant export papers
P5. The supplier shall conduct verification to present the ADCS meets all the requirements.
   a. Functional tests
   b. EMC tests
   c. Thermal vacuum cycling with acceptance levels
   d. Mechanical tests with acceptance levels
      i. Vibration
      ii. Shock
P6. The supplier shall provide failure mode, effects and criticality analysis (FMECA) results.
P7. The supplier shall provide failure detection, isolation and recovery (FDIR) information.

O1. The ADCS shall have at least the following autonomous operation modes:
   a. Safe mode: De-tumbling.
   b. Sun-pointing: Points the determined axis to the sun.
   c. Nadir pointing: Points the determined axis to nadir direction.
   d. Target tracking: Points the determined axis to a determined target.

e. Spin: Spins the satellite about a determined axis to a determined angular velocity.
f. Diagnostics: Only manual control.
g. Measurement: No attitude control. Sends telemetry data.

O2. The ADCS shall have 1 degree pointing accuracy or higher.
O3. The ADCS shall have 1 degree pointing knowledge or higher.
O4. The ADCS shall have 1.5 degree per second slew rate or higher.
O5. The ADCS shall be able to generate and measure 200 degrees per second angular velocity or lower about the axis of the satellite's maximum inertia with a maximum attitude error of 10 degrees in relation to Earth's spin axis.
O6. The ADCS shall be able to recover to a stabilized state from 200 degrees per second angular velocity or lower about the axis of the satellite's maximum inertia.
O7. The ADCS shall have jitter lower than TBD.
O8. The ADCS shall have 0.01 degrees per second^2 angular acceleration or higher.

Appendix B Aalto-1 ADCS Control Plan

Schedule	Operation	Attitude	ADCS Operation Mode	Tasks	Duration
Separation + 30min	De-tumbling	Unknown	Safe-Mode	Send telemetry 1/min TBC	Until contact
Validation phase	Comissioning	Normal flight	All except spin TBC	Send telemetry 1/min TBC	Until verified
Measurement phase	Measuring	Spectrometer to nadir/RadMon east or west	Nadir pointing (10 deg accuracy ok)	Collect attitude information TBC	6-12 months TBC
Sub: Measurement phase	Taking picture	Spectrometer to Nadir/target	Nadir pointing/Target tracking	Telemetry to OBC TBC	Once a day TBC
Sub: Measurement phase	S-Band comm.	S-Band to Nadir/target	Nadir pointing/Target tracking	Telemetry to OBC TBC	Once per orbit TBC
Plasma Brake experiment	Spin acc.	Spin about the Earth's spin axis 0 - 200 deg/s	Spin	Send telemetry 1/min TBC	~20 h TBC
Plasma Brake experiment	Measuring	Spin about the Earth's spin axis at 10-72 deg/s	Measuring	Send telemetry 1/min TBC, no control	TBD
Plasma Brake experiment	Spin acc.	Accelerate spin from TBD to TBD deg/s	Spin	Send telemetry 1/min TBC	< 20 h TBC
De-orbiting	Measuring	Unknown	Measuring	Send telemetry 1/min TBC, no control	TBD
In case of failure:	Diagnostics	No control	Diagnostics	All manual	TBD

Appendix C Aalto-1 ADCS Verification plan

Requirement	Deadline	Method	Criteria
M1. The ADCS shall be able to withstand the launch environment	Delivery	Acceptance tests	TBD
	Verification	Qualification tests	TBD
M2. The ADCS shall be able to withstand the LEO environment for at least 2 years.	Delivery	Acceptance tests	TBD
	Verification	Qualification tests	TBD
M3. The ADCS shall be able to control the pointing of two instruments on orthogonal axes simultaneously. (3-axis stabilization)	Contract	Analysis	Confirmed
	Verification	Inspection	Confirmed
S1. The mechanical interface shall be CubeSat Kit compatible.	Contract	Analysis	Confirmed
	Verification	Inspection	Confirmed
S2. The electrical interface shall be CubeSat Kit compatible.	Contract	Analysis	Confirmed
	Verification	Inspection	Confirmed
S3. The electrical interface shall use RS485, I2C or LVDS (TBC).	Contract	Analysis	Confirmed
	Verification	Functional Tests	Confirmed
S4. The CubeSat Kit Connector shall go through the ADCS.	Contract	Analysis	Confirmed
	Verification	Inspection	Confirmed
S5. The ADCS's dimensions shall be less than 95 mm * 95 mm * 57 mm.	Contract	Analysis	Confirmed
	Verification	Measurement	Confirmed
S6. The ADCS's mass shall be less than 500g.	Contract	Analysis	Confirmed
	Verification	Measurement	Confirmed
S7. The ADCS's nominal power consumption shall be less than 1.5 W and peak power consumption less than 2W.	Contract	Analysis	Confirmed
	Verification	Functional tests	Confirmed
P1. A mechanical model and an electrical model or alternatively an engineering model shall be delivered before June 2012.	Contract	Analysis	Confirmed
	Verification	Inspection	Confirmed
P2. A flight model shall be delivered before	Contract	Analysis	Confirmed

January 2013.			
	Verification	Inspection	Confirmed
P3. The supplier shall provide ground support equipment necessary to integrate and operate the ADCS.	Contract	Analysis	Confirmed
	Verification	Inspection	Confirmed
P4. The supplier shall provide at least the following documentation:	Contract	Analysis	Confirmed
	Verification	Inspection	Confirmed
a. ICD / Manual			
b. Test Specifications & Test Report			
c. Functional Test Report including a procedure to do functional test at Aalto University for product initial test			
d. Copy of statement of origin			
e. Copy of relevant export papers			
P5. The supplier shall conduct verification to present the ADCS meets all the requirements.	Contract	Analysis	Confirmed
	Delivery	Acceptance tests	TBD
	Verification	Qualification tests	TBD
a. Functional tests			
b. EMC tests			
c. Thermal vacuum cycling with acceptance levels			
d. Mechanical tests with acceptance levels			
i. Vibration			
ii. Shock			
P6. The supplier shall provide failure mode, effects and criticality analysis (FMECA) results.	Contract	Analysis	Confirmed
	Delivery	Analysis	Confirmed
	Verification	Inspection	Confirmed
P7. The supplier shall provide failure detection, isolation and recovery (FDIR) information.	Contract	Analysis	Confirmed
	Delivery	Analysis	Confirmed
	Verification	Inspection	Confirmed
O1. The ADCS shall have at least the following autonomous operation modes:	Contract	Analysis	Confirmed
	Delivery	Functional tests	Confirmed
	Verification	Functional tests	Confirmed
a. Safe mode: No autonomous operation.			
b. Sun-pointing: Points the determined axis to the sun.			

c. Nadir pointing: Points the determined axis to nadir direction.			
d. Target tracking: Points the determined axis to a determined target.			
e. Spin: Spins the satellite about a determined axis to a determined angular velocity.			
f. Diagnostics: Only manual control.			
g. Measurement: No attitude control. Sends telemetry data.			
O2. The ADCS shall have 1 degree pointing accuracy or higher.	Contract	Analysis	Confirmed
	Verification	Functional tests/Analysis	Confirmed
O3. The ADCS shall have 1 degree pointing knowledge or higher.	Contract	Analysis	Confirmed
	Verification	Functional tests/Analysis	Confirmed
O4. The ADCS shall have 1.5 degree per second slew rate or higher.	Contract	Analysis	Confirmed
	Verification	Functional tests/Analysis	Confirmed
O5. The ADCS shall be able to generate and measure 200 degrees per second angular velocity or lower about the axis of the satellite's maximum inertia with a maximum attitude error of 10 %.	Contract	Analysis	Confirmed
	Verification	Analysis	Confirmed
O6. The ADCS shall be able to recover to a stabilized state from 200 degrees per second angular velocity or lower about the axis of the satellite's maximum inertia.	Contract	Analysis	Confirmed
	Verification	Analysis	Confirmed
O7. The ADCS shall have less than TBD jitter.			
	Contract	Analysis	Confirmed
	Verification	Analysis	Confirmed
O8. The ADCS shall have 0.01 degrees per second^2 angular acceleration or higher.			
	Contract	Analysis	Confirmed
	Verification	Analysis	Confirmed

Appendix D Target tracking MATLAB code

```
Re = 6380000; % earth radius (m)
h = 300000; % orbit altitude (m)
myy = 3.986*10.^14; % Earth gravitational parameter
v = sqrt(myy/(Re+h)); % orbital velocity (m/s)
h = 100000; % 100 km altitude

% angular rate derivation and plotting
for index = h:100000:1000000 % altitudes 100 km - 1000 km
a = (Re+h)/Re;
omega_e = (sqrt(myy))./((Re+h).^(3/2));

p = (2*pi*(Re+h))/sqrt((myy)/((Re+h))); % one orbit time
t = linspace(0,p,10000); %time
t = t*2; % how many orbits calculated

% angular velocity function derivation gives:
i = (omega_e.*cos(omega_e.*t))./(a-cos(omega_e.*t));
o = (omega_e.*sin(omega_e.*t).^2)./(a-cos(omega_e.*t)).^2;
d = ((sin(omega_e.*t).^2)./(a-cos(omega_e.*t)).^2)+1;
omega_s = (i-o)./d;

figure(1);
plot(t, (omega_s*360)/(2*pi)) %angular rate
grid on;
hold on;
h = h + 100000; % 100 km addition
end
```