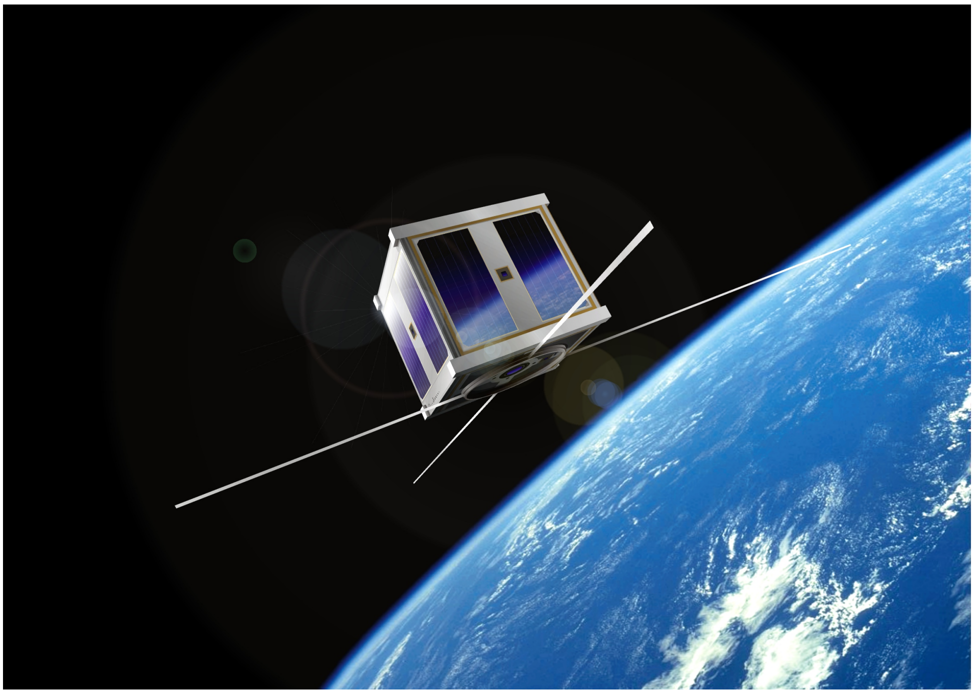
Satellite Formation Flying for Pico Satellite



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Synopsis

Abstract goes here

Preface

This report has been written by group 931 on third semester in Control and Automation on Aalborg University. References made before a full stop regards the sentence and reference after full stop regards the paragraph. Quotes are inside quotations marks and in cursive. Attached to report is a zip file with:

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Report by:

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Contents

1 | Introduction

[1] some intro [...]

1.1 Problem statement

Design and implement a controller for attitude and position of several pico-satellites in orbit.

1.2 Use-case

Denmark has a small island called Greenland, where the Danish Government needs to monitor it. One method is to have a constellation of six satellites going around the orbit. The idea is that whenever the satellites are located in the northern hemisphere, both of them will point down and look towards Greenland. The surveillance might contain taking pictures and measurements. After the surveillance, the concept is to change the attitude and having a control of the distance between them as they are in orbit.

The task the satellite has to perform is acquiring data by flying around Greenland.

This gives two main objectives for the satellite:

- Studying the orbital dynamic model by looking at one satellite neighbours
- Attitude and orbit determination using momentum wheels.

2 | System Description

The overall idea of the project is to consider more than one satellites flying in formation., with a certain distance in between and with the purpose of maintaining that distance by exchanging information. As a proof of concept, an AAU-CubeSat will be used, by choosing six AAU-CubeSat that orbit the Earth like is shown in *figure 2.1*. Therefore, a control system is developed, where the six satellites are nodes and they represent periods. Each satellite can only communicat with his two neighbour. In this project, all CubeSat's will be assumed identical, where each satellite needs to fulfill a few requirements stated in *chapter 3*. Moreover, a full-scale implementation of the system will not be possible, therefore, the whole system will be simulated using MATLAB and Simulink.

Figure 2.1: desc.

About AAU-CubeSat

The AAU-CubeSat shown in *figure 2.2* is a pico-satellite developed by Stanford University, but assembled at Aalborg University by students and used mainly for Low Earth Orbit (LEO) tests.

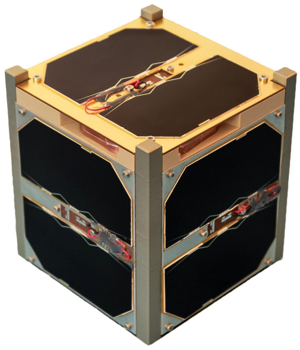

Figure 2.2: descrip

The pico-satellite is designed for LEO, therefore a few constraints are imposed. The CubeSat is limited in size and weight. The dimensions of the satellite are $10cm \times 10cm \times 10cm$, while the weight around 1 kg. [1]

In order place the CubeSat on the orbit, a deployment system is used, called P-POD [2] This system uses the force of a spring to launch the satellite into space. The satellite will be placed inside the launch rocket as payload. By using this system, an important advantage is reducing the cost of the launch.

AAU-CubeSat actuators

The selection of attitude control components is important in order to meet the performance requirements. For this project, three magnetorquers and three momentum wheels have been chosen as actuators. Initially, using only three momentum wheels has been considered, but the downside of using only momentum wheels is that some amount of momentum can be stored in the wheel, which will imply having a way to take back all that momentum and use it. Therefore, there are two ways to release that torque, one is to use magnetorquers and the second to use thrusters. [3]

Magnetorquers are wire coils which generate an electromagnetic field. The field interacts with the Earth magnetic field and a torque is generated for stabilizing the satellite. An important aspect of the magnetorquer is when the momentum wheel reaches a maximum speed and can no longer produce the torque (this is referred as wheel saturation'), so a magnetorquer is used to extract the momentum from the wheel.

[1] FiXme Note: ref
[2] FiXme Note: ref
[3] FiXme Note: ref

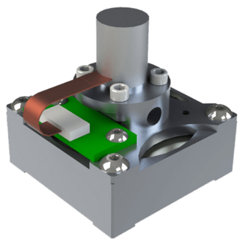

Figure 2.3: Example of a reaction wheel for CubeSats

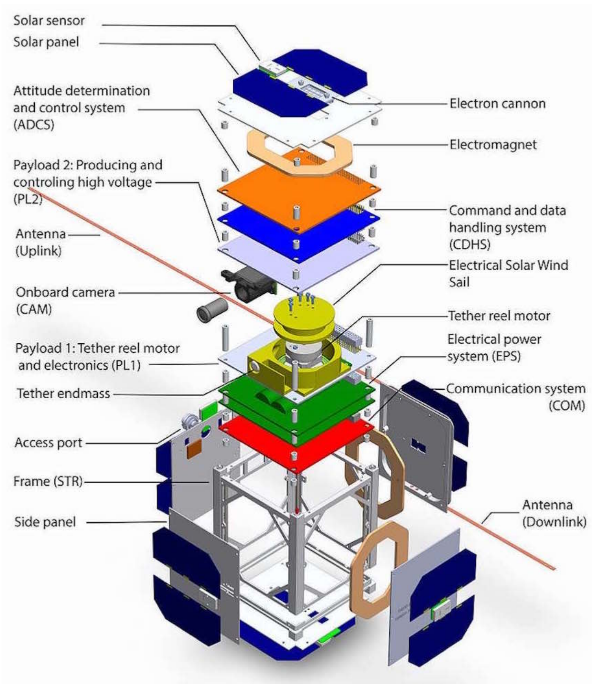

Figure 2.4: Cubesat 3D view

Momentum wheels shown in *figure 2.3* strength is that no information is needed about the magnetic field in order to control the CubeSat torque. These wheels are capable to store the momentum needed for maneuvering or pointing.

AAU-CubeSat sensors

The CubeSat can sustain itself using solar pannels [ref in fig 2.4] with in the middle a sun sensor, which provide a vector equal to the direction of the sun and also a magnetometer that gives a vector of the Earth's magnetic field. Whether the Earth's magnetic field is measured, or the sun vector, the objective is to use these sensors to deliver vector solutions for determining the satellite's pointing and rotation rates.

Magnetometer is a sensor used for attitude control, which measure the direction and intensity of the magnetic field. The atittude is determined from the magnetometer by comparing the measure magnetic field with a referance field.

Sun sensor is used for delivering a vector of measurements from the Sun. (ref to the fig 2.4)

Pointing accuracy

The required pointing accuracy when acquiring a photo is based on the a height from the picture is taken, in this case around 700 km above the Earth surface is going to cover approximately ?? km.

3 | Requirements

Based on the use-case introduced and the available system a set of requirements are formulated.

System requirements

1. **The constellation shall be able to maintain a given distance**

   Measure the position of the satellites and using the drag force to control the velocity

2. **The satellite should be able to turn twords target**

   Measure the current attitude of the satellites and to control it in order to achieve the specify orientation

4 | Modelling

// maybe a short intro about the whole modelling part, like what we are doing in the kinematics part and dynamics, the overall puropose

———-

In order to control the average distance between two or more satellites in orbit, a mathematical description of the governing equations should be derived. Since precious work have been made in previous projects, and all the measurements are available, in-depth analysis it is deemed not necessary.

// maybe a different structure like:
- Coordinate systems: Reference Coordinate Systems and Satellite Coordinate Frame
- Kinematics
- Dynamcs
- Disturbace model

4.1 Orbital Model

4.2 Equations of Motion

This chapter provides a description of the dynamic and kinematic equations of motion which constitute the basis for further analysis and description of the forces and/or disturbances, which may affect a rigid body within Low Earth Orbit(LEO).

Dynamic Model

In order to describe the behavior of the satellite a dynamic model based on reaction wheels and by using Euler's equation of motion has been derived. Euler's equation of motion describing the rotation of a rigid body is given by: [1]

$$\dot{L} = N_{tot} - \omega \times L \tag{4.1}$$

where N_{tot} represents all the external torques caused from the actuator and the disturbances, ω is the angular velocity of the satellite and L is the total angular momentum of the satellite and the reaction wheels, given by:

$$L = I_s \omega + h_{tot} \tag{4.2}$$

where h_{tot} is the vector of the angular momentum of the wheels $[h_1 h_2 h_3]^T$, all seen in the satellites coordinate system and I_s is the inertia matrix of the satellite. Inserting the

[1] FiXme Note: ref

equation *equation (4.2)* into *equation (4.1)* we obtain

$$\frac{d}{dt}(I_s\omega) + \dot{h}_{(tot} = N_{tot} - \omega \mathrm{x}(I_s\omega + h_{tot}) \tag{4.3}$$

For three reaction wheels attached at the body coordinate system which are the axis roll, pitch and yaw, three equations shall be derived. The derivation of the three equations of motion along with the diagonal inertia matrix can be found in the appedix A ([2]rite in Appedix). For the ease of notation, the cross product can be written as matrix operation using the $S()$ representing the skew symmetric matrix. Solving for $\dot{\omega}$ the dynamic equation can be written as

$$\dot{\omega} = -I_s^{-1}S(\omega)I_s^{-1}\omega - I_s^{-1}S(\omega)h_{tot} - I_s^{-1}\dot{h}_(tot) + I_s^{-1}N_{tot} \tag{4.4}$$

The rate of change in angular momentum $\dot{h_{tot}}$ can be absorbed from the controller. This can be written as:

$$\dot{h}_{(tot)} = -Nc \tag{4.5}$$

where the negative sign denotes the absorbed momentum. The total torque from external disturbances can be written as N_{dis}. Rearranging, equation *equation (4.4)* now reads

$$\dot{\omega}(t) = -I_s^{-1}S(\omega)I_s\omega(t) - I_s^{-1}S(\omega)h_{tot} + I_s^{-1}N_c(t) + I_s^{-1}N_{dis}(t) \tag{4.6}$$

which constitute the dynamics of the satellite with 3 reaction wheels. At the final equation *equation (4.6)* is shown the time dependency of the variables.

Frames and Kinematic Model

This section will provide the orbit-attitude determination of the satellite using quaternion parameters. Since the differential Drag control method is based on the rotation of the satellite in order to achieve the effective cross-sectional area, a notation with respect the collaborating frames should be obtained.

Earth Centered Inertial frame(ECI)

In order to describe the orbit formation of the satellite, the ECI frame is used, since it can be seen as a non-accelerating frame. The z axis is pointing through the geographical north pole, the x axis is crossing from the point where the equatorial of the earth and the vernal equinox met and the y axis is the cross product of x and z creating a right-handed coordinate system.

Orbit Reference frame(ORF)

The orbit reference frame in Cartesian coordinates can be seen as a non-changing frame with respect the earth and the satellite. The z axis always pointing at the nadir point and it is parallel to the z_e axis o the inertial frame of the earth. The x_o axis, it is parallel to the orbit plane and y_o is the cross product of the x_o and z_o.

[2]FiXme Note: w

Satellite Controller frame(SCF)

In order to derive the kinematic equations, a controller reference frame should be specified. It is located in the center of mass of the satellite and it is defined such that the axis of higher inertia z_c pointing in the center of ECI and the x_c axis with the smallest inertia, pointing along with the orbit's x_o

4.3 Disturbances

Gravitational torque

An unbalanced satellite in orbit is subjected to a torque due to the gravitational torque. Assumed that the earth is a point mass and the satellite is a rigid body, the gravitational torque can be estimated. Each infinitesimal element of the satellite of mass dm_i is subjected to an infinitesimal force dF_i that can be calculated thanks to the Newton's law of universal gravitation.

$$dF_i = -G\frac{m_{earth}}{||R_i||^2}dm_i \cdot \frac{R_i}{||R_i||}$$

where G is the gravitational constant, m_{earth} is the mass of the earth and R_i^2 is the vector from the earth to the infinitesimal element of the satellite.
The moment of the gravitational force about the geometric center is calculated as the formula:

$$N_{gra} = \int_{sat} r_i \times dF_i$$

with r_i is the vector from the geometric center to the infinitesimal element. r_i can be written as the sum of the vector from the geometric vector to the mass center $r_{g,m}$ and the vector from the mass center to the elemement $r_{m,i}$. Therefore, the expression of the gravitational torque is simplified:

$$\begin{aligned} N_{gra} &= \int_{sat} r_{g,m} \times dF_i + \int_{sat} r_{m,i} \times dF_i \\ &= \int_{sat} r_{g,m} \times -G\frac{m_{earth}}{||R_i||^3}dm_i \cdot R_i + \int_{sat} r_{m,i} \times -G\frac{m_{earth}}{||R_i||^3}dm_i \cdot R_i \end{aligned}$$

We can assumed that $||r_{m,g}|| << ||R_i||$ and R_i can be supposed constant and equals to the vector from the center of the earth to the geometric center of the satellite $R_{e,g}$. Thus, The second term is null by definition of the mass center.

$$\Rightarrow N_{gra} = G\frac{m_{sat} \cdot m_{earth}}{||R_{e,g}||^3} \cdot (R_{e,g} \times r_{g,m})$$

The position of the center of mass was measured for the previous project and is eqals to [?;?;?] in the frame of the satellite. Therefore, $r_{g,m,i}$ can be expressed in the inertial frame as following:

$$[r_{g,m,i}; 0] = q_{i,s} \otimes [?; ?; ?.0] \otimes q_{i,s}*$$

where $q_{i,s}$ is the quaternion that represents the rotation of the satellite in the inertia frame and $\otimes$ is the quaternion multiplication. Thus, the moment of force can be calculated by this expression above.