# Attitude Control of the UASat

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# Document overview

This document describes the design of the control system of the UASat. Except design information, sections are included that describe theoretical aspects. These sections are ment for new GNC members.

# Contents

# Chapter 1

# Requirements

The purpose of the Guidance, Navigation and Control (GNC) subsystem is to provide the satellite with the proper orientation and stability to point its instruments and solar panels.

The pointing requirements can be divided in four control modes, one for each science experiment and one for pointing of the solar cells.

1. The main science mission is the detection and imaging of lightning and sprites. The satellite needs to be oriented such that the science instruments looks at the limb of the Earth. The attitude needs to be stabilized with respect to Earth with an pointing accuracy of about $1^\circ$

2. The secondary science mission is the UBV photometry experiment. In order to count the light coming from a star the satellite needs to be pointed such that the light falls on the photo detector in the science instrument. This requires that the attitude is stabilized with respect to the stars with and accuracy of about $1^\circ$about each axis.

3. The laser communication up-link experiments requires that the science instrument, which houses the laser detector, tracks the ground station in Tucson. In other words, the satellite needs to track an desired attitude with respect to Earth. As presently understood, and accuracy of about $1^\circ$will suffice. A high slew rate is necessary as the satellite passes over Tucson in only about five minutes. This results in a slew rate requirement of $180^\circ/(5 \cdot 60) = 0.6^\circ/s$.

4. The power budget of the satellite is very limited as the solar panel area is small. On the sun side of the orbit, the satellite needs to oriented such that the power generation is maximized, i.e. the sun rays incidence perpendicular on the solar panels. This requires tracking of the sun with an accuracy better than $8^\circ$to keep the power generation within 1% of the maximum.

| Description | Specification |
|---|---|
| Type of control | 3 axis Earth and inertial pointing |
| Slew rates | $< 1°$/second |
| Pointing accuracy | |
| - Earth pointing | 1° |
| - Inertial pointing | 1° |
| - Sun tracking | 8° |
| Others | The satellite will not be able to make changes in orbital parameters |

Table 1.1: ACS Specifications

In order to make orbital changes for station keeping some kind of thrusters would be needed. This would increase the complexity of the GNC subsystem highly. Further, strict safety requirements for thruster from NASA would apply. To keep matter simple, the satellite will not be able to make changes in orbital parameters. The specifications of the Attitude Control System (ACS) are summarized in Table 1.1.

# Chapter 2

# Design

## 2.1 Selection of control scheme

There are basically three types of attitude stabilization [5]:

1. *Spin stabilization* — Spin stabilization utilizes that the angular momentum is conserved. A momentum wheel is spun up and the gyroscopic stiffness will stabilize the attitude of the satellite with respect to the stars. The main science mission is lightning/sprite detection and imaging which requires attitude stabilization with respect to the Earth. This control type is obvious not suited.

2. *Gravity gradient control* — Gravity gradient control requires the structure of the satellite to be designed in such a way that the gravity gradient torque is the largest environmental torque on the system, which is usually implemented in the form of a long boom. The gravity gradient torque will in that case stabilize the attitude with respect to Earth. This control scheme is not suited because gravity gradient (1) only allows pointing accuracy of $\approx 5^\circ$, (2) does not support maximum power generation or inertial pointing and(3) using a long boom will also increase the aerodynamic drag torque drastically.

3. *Three axis control* — Three axes control utilizes reaction wheels to control the attitude in all three axis.

Three axis control is the only control scheme that will allow both Earth and inertial based pointing to support the primary and secondary science missions. Therefore, 3-axis stabilization will be used.

## 2.2 Disturbance torques

Disturbance torques acting on the spacecraft are due to [6]: gravity gradient, solar pressure, the Earth magnetic field and aerodynamic drag. Detailed deriva-

| Disturbance Source | Estimated Torque (Nm) |
|---|---|
| Gravity Gradient | $8.1 \cdot 10^{-6}$ |
| Solar Pressure | $1.8 \cdot 10^{-7}$ |
| Magnetic Field | TBD |
| Aerodynamic Drag | $7.2 \cdot 10^{-5}$ |

Table 2.1: Estimates of disturbance torques for LEO (400km, 51.6°)

tions of the models used to compute those torques are given in [3]. Upper bounds on those torques for the satellite are computed using the assumptions that (1) the satellite will fly at 400km altitude and 51.6°inclination, (2) the satellite has the shape of an cylinder with an diameter and height of 0.5m, (3) the total mass of 68kg is uniformly distributed and (4) the satellite will be launched around 2001 which is an solar maximum year. The estimates are given in Table 2.1.

Due to the relative high atmospheric density at 400km, the aerodynamic drag is the dominant disturbance torque. The disturbance torque due to the Earth magnetic field depends on the residual magnetic dipole moment of the satellite. At this design stage the dipole moment is not known but this torque can be significant. The satellite should be designed such that magnetic dipole moment is small enough.

## 2.3 System overview

*block diagram needs to be added*

Five sensors will be used for attitude determination: (1) sun sensor, (2) magnetometer, (3) GPS, (4) horizon sensor and (5) micro mechanical gyros. The sun sensor determines the location of the sun with respect to the spacecraft. This sensor will be used in the maximum power generation mode. The combination of the magnetometer and GPS determines the attitude of the satellite with respect to Earth. The horizon sensor will complement these sensor to obtain a higher precision when the telescope is pointed at the limb of the Earth.

Two types of actuators will be utilized for reorientation of the satellite: reaction wheels and magneto torquers. Four reaction wheels will be used for fine motion control while three magnetic torquers provide momentum dumping.

## 2.4 Actuators

### 2.4.1 Using reaction wheels and magneto torquers

This section gives a basic theoretical background for the use of reaction wheels and torque rods for attitude control. Readers familiar with these methods can skip this section.

In essence, a reaction wheel is an electric motor with a heavy disk attached to its axis. By applying a current to the motor, a torque can be generated by changing the angular velocity of the disk. According to Newton's third law, a reaction torque with the same magnitude will act on the satellite in opposite direction. This torque can be used to reorient the spacecraft when a pointing error is detected.

A magnetic torquer or torque rod is basically an electromagnet that generates a magnetic dipole moment that interacts with the Earth's magnetic field to produce a torque on the spacecraft. The generated torque depends on the local intensity of the Earth's magnetic field, magnetic dipole moment and the angle between the dipole moment and the Earth's B-field. Also this torque can be used for reorienting the satellite.

### Dynamics

The dynamics of the satellite will give a better feel of how the satellite can be controlled. Those equations will be derived now. The law of conservation of angular momentum states that, with respect to an inertial frame, the change with time of the angular momentum of a system equals the torques applied to that system[2]:

$$\frac{d}{dt}\mathbf{H}^o = \boldsymbol{\tau}^o$$

where $\mathbf{H}^o$ is the angular momentum of the satellite and $\boldsymbol{\tau}^o$ are the *external* torques acting on the satellite. The superscript $o$ denotes that the vectors are with respect to an interial frame.

The external torque is the sum of the torque due to the interaction of the magnetic fields of the Earth and the magneto-torquers, $\boldsymbol{\tau}^o_{mt}$, and the torque due to environmental effects, $\boldsymbol{\tau}^o_{env}$, like aerodynamic drag. Furthermore, the angular momentum of the satellite is the sum of the angular momentum, $\mathbf{H}^o_{ss}$, of the satellite structure — the angular momentum of the satellite excluding the angular momentum due to the spinning rotors of the reaction wheels — and the angular momentum, $\mathbf{H}^o_{rw}$, of the spinning rotors of the reaction wheels:

$$\frac{d}{dt}(\mathbf{H}^o_{ss} + \mathbf{H}^o_{rw}) = \boldsymbol{\tau}^o_{mt} + \boldsymbol{\tau}^o_{env} \tag{2.1}$$

Angular momentum can be expressed as the product of the moment of inertia and the rotational speed, assuming four reaction wheels,

$$\frac{d}{dt}(\mathbf{I}^o_{ss}\boldsymbol{\omega}^o_{ss} + \sum_{i=1}^{4} I_{rw_i}\omega_{rw_i}\mathbf{e}^o_{rw_i}) = \boldsymbol{\tau}^o_{mt} + \boldsymbol{\tau}^o_{env}, \tag{2.2}$$

where $\mathbf{I}^o_{ss}$ is the moment of inertia of the satellite, $\boldsymbol{\omega}^o_{ss}$ the angular velocity of the satellite, $I_{rw_i}$ the moment of inertia of the rotor of reaction wheel $i$ around its spin axis, $\omega_{rw_i}$ the angular velocity of rotor of reaction wheel $i$ and $\mathbf{e}_{rw_i}$ the unit vector in the direction of the spin axis of reaction wheel $i$ according to the right hand rule for positive $\omega_{rw_i}$.

It will prove to be more convenient to switch to a spacecraft fixed frame. This rotating frame will be denoted by a superscript $s$. Denoting the orthonormal transformation matrix from the spacecraft frame to the inertial frame by $\mathbf{R}^{so}$ the relationship between the moments of inertia of the satellite structure in both frames is [2]:

$$\mathbf{I}^o_{ss} = \mathbf{R}^{so}\mathbf{I}^s_{ss}\mathbf{R}^{so\,T}, \tag{2.3}$$

where the superscript $T$ denotes the transpose of the matrix.

Using the transformation matrix and the similarity transform 2.3 equation 2.2 can be written in terms of the spacecraft fixed frame:

$$\frac{d}{dt}(\mathbf{R}^{so}\mathbf{I}^s_{ss}\mathbf{R}^{so\,T}\mathbf{R}^{so}\boldsymbol{\omega}^s_{ss} + \sum_{i=1}^{4} I_{rw_i}\omega_{rw_i}\mathbf{R}^{so}\mathbf{e}^s_{rw_i}) = \mathbf{R}^{so}\boldsymbol{\tau}^s_{mt} + \mathbf{R}^{so}\boldsymbol{\tau}^s_{env},$$

Using that the transformation matrix is orthonormal, $\mathbf{R}^{so\,T}\mathbf{R}^{so} = \mathbf{E}_3$, and that $\mathbf{I}^s_{ss}, I_{rw_i}$, and $\mathbf{e}^s_{rw_i}$ are constant, the equation can rewritten as:

$$\begin{aligned}\dot{\mathbf{R}}^{so}\mathbf{I}^s_{ss}\boldsymbol{\omega}^s_{ss} + \mathbf{R}^{so}\mathbf{I}^s_{ss}\dot{\boldsymbol{\omega}}^s_{ss} + \sum_{i=1}^{4}(I_{rw_i}\dot{\omega}_{rw_i}\mathbf{R}^{so}\mathbf{e}^s_{rw_i} + I_{rw_i}\omega_{rw_i}\dot{\mathbf{R}}^{so}\mathbf{e}^s_{rw_i})&\\ = \mathbf{R}^{so}\boldsymbol{\tau}^s_{mt} + \mathbf{R}^{so}\boldsymbol{\tau}^s_{env},&\end{aligned}$$

Expressing the change of rate of the orientation as [4] $\dot{\mathbf{R}}^{so} = \mathbf{R}^{so}\tilde{\boldsymbol{\omega}}^s_{ss}$ and pre-multiplying both sides by $\mathbf{R}^{so\,T}$ obtain:

$$\tilde{\boldsymbol{\omega}}^s_{ss}\mathbf{I}^s_{ss}\boldsymbol{\omega}^s_{ss} + \mathbf{I}^s_{ss}\dot{\boldsymbol{\omega}}^s_{ss} + \sum_{i=1}^{4}(I_{rw_i}\dot{\omega}_{rw_i}\mathbf{e}^s_{rw_i} + I_{rw_i}\omega_{rw_i}\tilde{\boldsymbol{\omega}}^s_{ss}\mathbf{e}^s_{rw_i}) = \boldsymbol{\tau}^s_{mt} + \boldsymbol{\tau}^s_{env},$$

where $\tilde{\mathbf{a}}$ denotes the matrix denoting the cross product with $\mathbf{a}$:

$$\tilde{\mathbf{a}} = \begin{bmatrix} 0 & -a_3 & a_2 \\ a_3 & 0 & -a_1 \\ -a_2 & a_1 & 0 \end{bmatrix}$$

Rearranging terms and using that $I_{rw_i}\dot{\omega}_{rw_i} = \tau_{rw_i}$ we can write:

$$\begin{aligned}\mathbf{I}^s_{ss}\dot{\boldsymbol{\omega}}^s_{ss} \quad = \quad & -\tilde{\boldsymbol{\omega}}^s_{ss}\mathbf{I}^s_{ss}\boldsymbol{\omega}^s_{ss} \\ & -\textstyle\sum_{i=1}^{4}(\tau_{rw_i}\mathbf{e}^s_{rw_i} + I_{rw_i}\omega_{rw_i}\tilde{\boldsymbol{\omega}}^s_{ss}\mathbf{e}^s_{rw_i}) \\ & +\boldsymbol{\tau}^s_{mt} + \boldsymbol{\tau}^s_{env}\end{aligned} \tag{2.4}$$

where $\tau_{rw_i}$ is the torque at the shaft of the reaction wheel. Note that $\tau_{rw_i}$ are *internal* torque's, these torque's do not change the total sum of angular momentum.

Equation 2.4 are the equations of motion of the satellite. The rate of change of rotation of the satellite, $\dot{\boldsymbol{\omega}}_{ss}$ can be changed by torques, $\tau_{rw_i}$ generated by the reaction wheels and by a torque, $\boldsymbol{\tau}^s_{mt}$, generated by the torque-rods. The environmental torque's, $\boldsymbol{\tau}^s_{env}$, act as an disturbance on the system — we cannot control the environmental torques.

#### Momentum dumping

*why do we need momentum dumping*

When the cumulative value of the applied torques is greater than zero, the reaction wheel will eventually reach its maximum rotational velocity (a condition referred to as saturation), preventing the application of any further torques. Saturation of a reaction wheel typical occurs when more torque is consistently applied in one direction than another over a long period of time. Other actuators (see below) are needed to slow down the reaction wheel by applying an external torque. This process is called de-saturation, momentum unloading or momentum dumping.

### 2.4.2 Torque and angular momentum requirements

#### Disturbance torques

#### Slewing maneuvers

### 2.4.3 Configuration requirements

Four reaction wheels will be flown for redundancy; one along each axis and the fourth reaction wheel placed skew with respect to the other three wheel. If one wheel fails the other three are still able to apply a torque in arbitrary direction. If two reaction wheels fail, the satellite will still be controllable with the remaining two reaction wheels. This can be seen easily as any arbitrary change in attitude can be represented by three consecutive rotations about two body-fixed axes, a fact, which is well known, to anyone who is familiar with Euler angles. A formal proof of the feasibility of this method is given by Walsh et al (1993) which includes constructive proofs.

### 2.4.4 Other requirements

It is possible to recapture electric energy that is used to accelerate the reaction wheels. Electric energy is converted to kinetic energy when a torque is applied to increase the speed of a reaction wheel. One method of decreasing the wheel's speed is to use the motor as an electric generator, converting rotational kinetic energy back into electric energy. Although this approach requires special steering amplifiers which are able to supply energy back to the power supply, the result is a very energy-efficient actuation system; only energy that is needed to overcome external disturbances and internal friction is lost.

### 2.4.5 Selection reaction Wheels

The smallest reaction wheels on the market are 4Nms which is 10 times bigger then required! Therefore the reaction wheels will be designed and build in house.

### 2.4.6 Selection magnetic Torquers

Magnetic torquers have no moving parts and are easy to construct. The magnetic torquers will be build in house by winding wire around a core. The magnetic torquers will be sized such that a stored momentum of 0.4Nms can be dumped in about 10 minutes.

## 2.5 Attitude sensors

### 2.5.1 Sun sensor for locating the sun

For the maximum power generation mode the location of the sun should be determined with an accuracy of 1-5 degrees. A in house build sensor can easily accomplish this.

The TSL230 of Texas Instruments is a chip that integrates a photocell and a current to frequency converter. The frequency of the TTL square wave output is a function of the angle of incidence of the sunrays. By placing several chips under different angles and at different places one can combine their outputs to a set of equations that can be solved for the location of the sun. Obvious, there are bad and good ways to place the cells. Different configurations are under analysis. A final decision on where to place the cells on the satellite is not yet made. The advantage of the frequency output is the easy interface with the onboard computer.

The readings of the photocells will be disturbed by light reflected by the Earth. This is a possible source of error. At this point it is not clear to what extent this reflected light will degrade our attitude measurement.

### 2.5.2 Magnetometer and GPS for determination of attitude w.r.t. Earth

The magnetometer and GPS will be used to determine the attitude of the satellite relative to the Earth. The basic algorithm is straightforward. The GPS is used to obtain the position of the satellite. Using this position data a model of the Earth magnetic field, the International Geomagnetic Reference Field (IGRF) model, computes the Earth magnetic B-field. On the other hand, the magnetic B-field is also measured by the 3-axis magnetometer. The attitude can be solved from these two vectors and the time history.

A Kalman filter will be used to make an optimal estimate of the attitude using the GPS and magnetometer data as input. The accuracy is typical limited by the geomagnetic model and will be around 1 °. In case the GPS fails the Kalman filter will be used to estimate both the attitude and position [1]. Rate data from the gyros can be incorporate into this filter to obtain a better estimate.

As magnetometer we use the TAM-2 from MEDA which is a fully space rated magnetometer designed for satellites. As GPS we are planning to fly the SixVee CM2 from Trimble. This is a terrestrial board that is not space rated but is qualifies for the shock and temperature range. This board will also fly on the ASUsat. We hope to get useful flight data from their mission.

Micro mechanical gyros will be flown as these devices are small and low power. Both Draper laboratories and JPL are developing these devices. Hopefully the gyros will be donated by JPL in return of flight data.

## 2.6 Control modes

Computer hardware and software on board the satellite will decode the input signals from the sensors described above to determine the attitude of the satellite. This information is then used to compute the torques required to reorient the satellite.

After being deployed or after an error condition the satellite first needs to be reacquired. A B dot control law using the magneto torquers will detumble the satellite. Then all the subsystems will be tested. Errors will be reported to the DCH team. The attitude of the satellite will first be determined using the magnetometer and GPS data. After the satellite is reacquired the control system will go to the standby mode. From this mode the four main control modes can be entered:

1. *Maxium power generation mode:* For maximum power generation the sun sensor will be used to determine the location of the sun. This information is then used to orient the satellite such that the sunrays fall perpendicular on the solar cells. Gyros are used to let the satellite spin about the z-axis to prevent that the satellite will get to hot on one side.

2. *Inertial pointing:* *What to do here? we don't have a startracker anymore*

3. *Sprite detection:* In the lightning experiment mode the magnetometer and GPS data will be used to point the science instrument at the limb of the Earth. The horizon sensor will be used to obtain more precise attitude information. Desired viewing directions are commanded from the DCH subsystem.

4. *Laser experiment:* In the laser experiment mode the ground station in Tucson will be tracked. The magnetometer and gyros will be used to track the ground station. The horizon sensor or sun sensor cannot be used in this mode.

*Momentum dumping* will be utilized during all modes. The torque rods are controlled to keep the average angular momentum at an desired value.

## 2.7 Low level control

# Chapter 3

# Various lists

# Chapter 4

# Interface requirements

1. *Placement reaction wheels* — The four reaction wheels need to be placed such that the four axis of rotation are not co-linear.

2. *Placement torque rods* — The torque rods need to be ....

# Chapter 5

# Current status

# Chapter 6

# Test plan

# Chapter 7

# Concerns and open issues

- Update of this document!
- Check the ACS specifications with the science and laser team.
- Recompute disturbance torques and document derivations.
- The magnetic field of the Earth may be disturbed so badly in and around the satellite that accurate readings with the magneto meter are no possible. Some disturbances can be estimated with Kalman filtering.

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