Attitude Determination And Control
AERO0025 – Satellite Engineering
OVERVIEW

• Introduction
• Rationale
• Attitude Requirements and Errors
• Governing Equations
• Attitude Parameterization
• Disturbance Torques
• Passive Control
• Active Control
• Going Further
• I am interested, what now?
INTRODUCTION

What is Attitude Determination and Control?
INTRODUCTION
What is attitude?

"The word around the office is that you have an attitude problem. ..."
What is attitude?

The orientation of a spacecraft in space is called its attitude.

The angular orientation of a body-fixed coordinate frame with respect to an external frame.
INTRODUCTION
Attitude Determination and Control

A rigid body satellite in space has 6 Degrees of Freedom

- Translation
  - Linear momentum
  - Motion of the center of mass
- Rotation
  - Angular momentum
  - Motion relative to the center of mass
INTRODUCTION
Attitude Determination and Control

AOCS: Attitude and Orbit and Control System

ADCS: Attitude Determination and Control System

- Translation
  - Linear momentum
  - Motion of the center of mass

- Rotation
  - Angular momentum
  - Motion relative to the center of mass
INTRODUCTION
Attitude Determination and Control

A rigid body satellite in space has 6 Degrees of Freedom

During this lecture, we will not care about the position of the spacecraft, but about its orientation.
INTRODUCTION
Side note about interdependency of attitude and orbit

For example

1. In LEO, the attitude will affect the atmospheric drag which will affect the orbit

2. The orbit determines the spacecraft position which determines both the atmospheric density and the magnetic field strength, which will, in turn, affect the attitude

But this dynamic coupling is often ignored, and the time history of the spacecraft position is assumed to be known and to be an input for ADCS

This coupling can even be used on purpose: (Limited) orbit control using differential drag.

Less drag

More drag
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• Attitude Requirements and Errors
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• Active Control
• Going Further
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RATIONALE

Why do we need Attitude Determination and Control?
Almost all spacecraft missions require attitude determination and control.

**Point Antennas**

- The antenna needs to be pointing towards the ground station.
- Higher accuracy → More data

**Point Solar Panels**

- The solar panels need to be pointing towards the Sun.
- Higher accuracy → More power

**Point Payloads**

- The payload needs to be pointing towards the point of interest.
- Higher accuracy → Better results
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• Rationale
• Attitude Requirements and Errors
• Governing Equations
• Attitude Parameterization
• Disturbance Torques
• Passive Control
• Active Control
• Going Further
• I am interested, what now?
ATTITUDE REQUIREMENTS AND ERRORS

What pointing requirements do missions have and how do we define them?
ATTITUDE REQUIREMENTS AND ERRORS
Mission attitude requirements

- RF/Communication
- Earth Science
- Reconnaissance
- Astronomy

Pointing accuracy [deg]

Year

100-180kg
10-100kg
-10kg

Nano/pico satellites
QbsX
TUDelft
CXBN

SSTL
Proba

Myriade
SSTL-150
Astro200
SSTL-150
Myriade
Astro200

TUB

Can-X2
JAXA
Falconsat
SSTL-100
JAXA
MIT

Proba

DRL

ATTITUDE REQUIREMENTS AND ERRORS

Communications

Wide beam low gain
Requirement: Several degrees

Narrow beam high gain
Requirement: \( \leq 1 \) degree
Envisat
- Accuracy: 0.01 deg
- Altitude: 800 km
- Ground Sample Distance: 140m

CubeSat EO
- Accuracy: 0.1 deg
- Altitude: 400 km
- Ground Sample Distance: 700m
One arc second is $1/3600^{\text{th}}$ of a degree.

- If a cake is split in pieces with an angle of 60 degrees, you have six pieces.
- If a cake is split in pieces with an angle of one arc second, you have 1,296,000 pieces.

Every inhabitant of the province of Liege could get a piece.

- **Accuracy**: 0.1 arc second

If you would give Hubble a laser pointer, it could hit the center of an archery target from 250 km away.
ATTITUDE REQUIREMENTS AND ERRORS

Pointing accuracy state of the art
ATTITUDE REQUIREMENTS AND ERRORS

Pointing Errors

Absolute Pointing Error (APE) → Accuracy
Dark black line
The actual pointing error at a given time t.

Windowed mean over period T
Horizontal black line
The mean of the pointing error over a period T.

Relative Pointing Error (RPE) → Jitter
The pointing error relative to the windowed mean.

Illustration of the different performance metrics as seen within one window with length $T$ [1]

ATTITUDE REQUIREMENTS AND ERRORS
Pointing Errors Example: Disaster Monitoring

Absolute Pointing Error (APE) \(\rightarrow\) Accuracy
The APE needs to be low enough, otherwise we are not looking at the correct scene.

- **High APE**
- **Low APE**

Relative Pointing Error (RPE) \(\rightarrow\) Jitter
The RPE during the observation time (e.g. 1 second) needs to be low enough to get a stable picture.

- **High RPE**
- **Low RPE**
OVERVIEW

• Introduction
• Rationale
• Attitude Requirements and Errors
• Governing Equations
• Attitude Parameterization
• Disturbance Torques
• Passive Control
• Active Control
• Going Further
• I am interested, what now?
GOVERNING EQUATIONS

What equations describe the spacecraft attitude?
GOVERNING EQUATIONS

Key Concept: Angular Momentum

Fundamental quantity in rotational dynamics.

Moment of the linear momentum about a defined origin

\[
\vec{H} = \sum_{i=1}^{n} \vec{H}_i = \sum_{i=1}^{n} \vec{r}_i \times m_i \vec{v}_i
\]

The angular momentum of a particle referred to an inertially fixed point is only changed if the forces on it have a moment M about this fixed point.

\[
\frac{d\vec{H}}{dt} = \vec{T}
\]
In the absence of an external torque, the angular momentum is preserved.

What happens to the angular momentum when the spacecraft deploys its solar panels?
**GOVERNING EQUATIONS**

Time Differentiation in a Rotating Frame

\[ \vec{r}_i = \vec{R} + \vec{\rho}_i \]

\[ \vec{v}_i = \frac{d\vec{r}_i}{dt} = \frac{d\vec{R}}{dt} + \left( \frac{d\vec{\rho}_i}{dt} \right)_{body} + \vec{\omega} \times \vec{\rho}_i \]

\[ \vec{a}_i = \frac{d^2\vec{R}}{dt^2} + \left( \frac{d^2\vec{\rho}_i}{dt^2} \right)_{body} + 2\vec{\omega} \times \left( \frac{d\vec{\rho}_i}{dt} \right)_{body} + \frac{d\vec{\omega}}{dt} \times \vec{\rho}_i + \vec{\omega} \times (\vec{\omega} \times \vec{\rho}_i) \]
**GOVERNING EQUATIONS**

**Total Angular Momentum**

\[
\vec{H}_t = \sum_{i=1}^{n} \vec{H}_i = \sum_{i=1}^{n} \vec{r}_i \times m_i \vec{v}_i
\]

\[
\vec{r}_i = \vec{R} + \vec{\rho}_i
\]

\[
\vec{v}_i = \frac{d\vec{r}_i}{dt} = \frac{d\vec{R}}{dt} + \vec{\omega} \times \vec{\rho}_i
\]

Rotating frame at the center of mass

\[
\vec{H}_t = \sum_{i=1}^{n} m_i \vec{R} \times \frac{d\vec{R}}{dt} + \sum_{i=1}^{n} m_i \vec{\rho}_i \times (\vec{\omega} \times \vec{\rho}_i)
\]

\[
\vec{l} = \sum_{i=1}^{n} m_i \left( \rho_i^2 \vec{E}_3 - \vec{\rho}_i \vec{\rho}_i \right)
\]

\[
\vec{H}_t = \sum_{i=1}^{n} m_i \vec{R} \times \vec{\dot{V}} + \vec{\dot{l}} \vec{\omega}
\]
The angular momentum is equivalent to the linear momentum for rotational dynamics \((m \rightarrow I, v \rightarrow \omega)\)

For a rigid spacecraft, the spin angular momentum can be decoupled from the orbital angular momentum.

Choose principal axes of inertia! In practice, asymmetries and misalignments lead to coupling, which leads to unwanted disturbances to be removed by the control system.
**GOVERNING EQUATIONS**

External Torques

A force applied to the spacecraft produces a torque about the center of mass:

\[ \overrightarrow{T_i} = \overrightarrow{\rho_i} \times \overrightarrow{F_i} \]

Fundamental equation for attitude dynamics

\[ \overrightarrow{T_{total}} = \overrightarrow{T} = \sum_{i=1}^{n} \overrightarrow{\rho_i} \times \overrightarrow{F_i} = \sum_{i=1}^{n} \overrightarrow{\rho_i} \times m_i \frac{d^2 \overrightarrow{r_i}}{dt^2} \]

\[ \overrightarrow{H_{s/c}} = \overrightarrow{I} \overrightarrow{\omega} \]

\[ \overrightarrow{T} = \frac{d \overrightarrow{H}}{dt} = \left( \frac{d \overrightarrow{H}_{s/c}}{dt} \right)_{body} + \overrightarrow{\omega} \times \overrightarrow{H_{s/c}} = \overrightarrow{I} \left( \frac{d \overrightarrow{\omega}}{dt} \right)_{body} + \overrightarrow{\omega} \times \overrightarrow{I} \overrightarrow{\omega} \]

Newton’s second law for rotating rigid bodies
GOVERNING EQUATIONS
Equation Governing Attitude Dynamics

\[ \mathbf{T} \iff \mathbf{\dot{\omega}} \]

\[ \mathbf{T} = \mathbf{I} \left( \frac{d\mathbf{\omega}}{dt} \right)_{\text{body}} + \mathbf{\omega} \times \mathbf{I} \mathbf{\dot{\omega}} \]

Principal axes

Gyroscopic coupling

\[ \dot{H}_x = I_x \dot{\omega}_x = T_x + (I_y - I_z)\omega_y\omega_z \]

\[ \dot{H}_y = I_y \dot{\omega}_y = T_y + (I_z - I_x)\omega_x\omega_z \]

\[ \dot{H}_z = I_z \dot{\omega}_z = T_z + (I_x - I_y)\omega_x\omega_y \]

Nonlinear equations with no general solution \( \rightarrow \) computer simulations
GOVERNING EQUATIONS
Simplification

For quick calculations, we may choose to simplify and make our life easier:

\[ T_x \approx I_x \dot{\omega}_x = I_x \alpha_x \]
\[ T_y \approx I_y \dot{\omega}_y = I_y \alpha_y \]
\[ T_z \approx I_z \dot{\omega}_z = I_z \alpha_z \]

If the gyroscopic effect becomes more important, this gets less accurate!
GOVERNING EQUATIONS
Spinning Rigid Spacecraft

\[ \omega_x, \omega_y \ll \omega_z = \Omega \]

\[ I_x \dot{\omega}_x = T_x + (I_y - I_z)\omega_y \omega_z \]
\[ I_y \dot{\omega}_y = T_y + (I_z - I_x)\omega_x \omega_z \]
\[ I_z \dot{\omega}_z = T_z + (I_x - I_y)\omega_x \omega_y \]

Free response

\[ \dot{\omega}_x = \frac{(I_y - I_z)}{I_x} \Omega \omega_y \]
\[ \dot{\omega}_y = \frac{(I_z - I_x)}{I_y} \Omega \omega_x \]
\[ \dot{\omega}_z = 0 \]

Nutation frequency

\[ \omega_n^2 = \frac{(I_z - I_y)(I_z - I_x)}{I_x I_y} \Omega^2 \]
What happens if the spin axis inertia $I_z$ is intermediate between $I_x$ and $I_y$?

The nutation frequency is imaginary. Any perturbing torque will result in the growth of the nutation angle until the body is spinning about either the maximum or minimum inertia axis, depending on initial condition.

A rigid body can rotate about its extreme inertia axis, but not the intermediate axis.
What happens if the spin axis inertia $I_z$ is intermediate between $I_x$ and $I_y$?

$I_1 > I_2 > I_3$
OVERVIEW

• Introduction
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• Passive Control
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• Going Further
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ATTITUDE PARAMETERIZATION

How do we describe the attitude of a spacecraft?
ATTITUDE PARAMETERIZATION

Introduction

The attitude – or orientation – of the spacecraft needs to be described in an unambiguous way. There are several ways of doing this:

• Direction Cosine Matrix
• Euler Angles
• Quaternion
• Others that will not be described here.

Each of these has its advantages and disadvantages and are used for different reasons.
ATTITUDE PARAMETERIZATION
Direction Cosine Matrix

Assume an orthogonal, right handed triad of unit vectors. (E.g. the body frame of a spacecraft.)

Specifying the components of these unit vectors in the coordinate frame 1-2-3 fixes the orientation completely.

\[ A = \begin{bmatrix} u_1 & u_2 & u_3 \\ v_1 & v_2 & v_3 \\ w_1 & w_2 & w_3 \end{bmatrix} \]
ATTITUDE PARAMETERIZATION
Direction Cosine Matrix

\[
A = \begin{bmatrix}
u_1 & u_2 & u_3 \\
v_1 & v_2 & v_3 \\
w_1 & w_2 & w_3
\end{bmatrix}
\]

Each of the elements of the matrix corresponds with the cosine of the angle between a body unit vector and a reference axis. E.g. \( u_1 \) is the cosine of the angle between \( u \) and the \( 1 \)-axis.

A is therefore often called the Direction Cosine Matrix.

The Direction Cosine Matrix maps vectors from the reference frame to the body frame.

\[
\begin{bmatrix}
a_u \\
a_v \\
a_w
\end{bmatrix} = \begin{bmatrix}
u_1 & u_2 & u_3 \\
v_1 & v_2 & v_3 \\
w_1 & w_2 & w_3
\end{bmatrix}\begin{bmatrix}
a_1 \\
a_2 \\
a_3
\end{bmatrix}
\]
Swiss mathematician, physicist, astronomer, logician and engineer, who published around 800 pages per year.

He produced, on average, one mathematical paper every week in the year 1775, while being almost blind.
ATTITUDE PARAMETERIZATION

Euler Angles

To define the Euler Angles, consider four orthogonal, right handed triads of unit vectors:

$$
\begin{bmatrix}
  x & y & z \\
  x' & y' & z' \\
  x'' & y'' & z'' \\
  u & v & w
\end{bmatrix}
$$

$$[x \ y \ z]$$ is parallel to the reference frame axes.
To define the **Euler Angles**, consider four orthogonal, right handed triads of unit vectors:

\[
\begin{bmatrix}
    x & y & z \\
    x' & y' & z' \\
    x'' & y'' & z'' \\
    u & v & w
\end{bmatrix}
\]

\([x \ y \ z]\) is parallel to the reference frame axes. 
\([x' \ y' \ z']\) is rotated as opposed to \([x \ y \ z]\) with an angle \(\Phi\) around an axis \(i\) (let’s assume the z-axis).
ATTITUDE PARAMETERIZATION

Euler Angles

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\]

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\([x' \ y' \ z']\) is rotated as opposed to \([x \ y \ z]\) with an angle \(\Phi\) around an axis \(i\) (let's assume the \(z\)-axis).

\([x'' \ y'' \ z'']\) is rotated as opposed to \([x' \ y' \ z']\) with an angle \(\Theta\) around an axis \(j\) (let’s assume the \(x\)-axis).

\(j\) cannot be equal to \(i\), otherwise we are just rotating further or back from the previous step.
ATTITUDE PARAMETERIZATION
Euler Angles

To define the Euler Angles, consider four orthogonal, right handed triads of unit vectors:

\[
\begin{bmatrix}
x & y & z \\
x' & y' & z' \\
x'' & y'' & z'' \\
u & v & w \\
\end{bmatrix}
\]

\([x \ y \ z]\) is parallel to the reference frame axes.
\([x' \ y' \ z']\) is rotated as opposed to \([x \ y \ z]\) with an angle \(\Phi\) around an axis \(i\) (let’s assume the \(z\)-axis).
\([x'' \ y'' \ z'']\) is rotated as opposed to \([x' \ y' \ z']\) with an angle \(\theta\) around an axis \(j\) (let’s assume the \(x'\)-axis).
\([u \ v \ w]\) is rotated as opposed to \([x'' \ y'' \ z'']\) with an angle \(\psi\) around an axis \(k\) (let’s assume the \(z''\)-axis).

\(k\) cannot be equal to \(j\), otherwise we are just rotating further or back from the previous step.
ATTITUDE PARAMETERIZATION
Euler Angles

The three rotation angles (\(\Phi\), \(\theta\), \(\psi\)), together with the axis order (3-1-3 in this case), define the attitude.

There are 12 different axis sequences:

313, 212, 121, 323, 232, 131, 312, 213, 123, 321, 231, 132

It is of course possible to convert Euler Angles to a Direction Cosine Matrix.

e.g. for a 313 sequence

\[
A_{313}(\Phi, \theta, \psi) =
\]

\[
A_3(\Phi)A_1(\theta)A_3(\psi) =
\]

\[
\begin{bmatrix}
\cos \psi \cos \phi - \cos \theta \sin \psi \sin \phi & \cos \psi \sin \phi + \cos \theta \sin \psi \cos \phi & \sin \theta \sin \psi \\
- \sin \psi \cos \phi - \cos \theta \cos \psi \sin \phi & - \sin \psi \sin \phi + \cos \theta \cos \psi \cos \phi & \sin \theta \cos \psi \\
\sin \theta \sin \phi & - \sin \theta \cos \phi & \cos \theta
\end{bmatrix}
\]
Sir William Rowan Hamilton knew that complex numbers could be interpreted as points in a plane, and he was looking for a way to do the same for points in three-dimensional space.

As he walked along the towpath of the Royal Canal with his wife, the concepts behind quaternions were taking shape in his mind. When the answer dawned on him, Hamilton could not resist the urge to carve the formula for the quaternions. [1]
Sir William Rowan Hamilton knew that complex numbers could be interpreted as points in a plane, and he was looking for a way to do the same for points in three-dimensional space.

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As he walked along the towpath of the Royal Canal with his wife, the concepts behind quaternions were taking shape in his mind. When the answer dawned on him, Hamilton could not resist the urge to carve the formula for the quaternions. [1]
Any rotation or sequence of rotations of a rigid body or coordinate system about a fixed point is equivalent to a single rotation by a given angle $\theta$ about a fixed axis (called the Euler axis) that runs through the fixed point.[1]

$$q = e^{\frac{\theta}{2}(u_xi + u_yj + u_zk)} = \cos \frac{\theta}{2} + (u_xi + u_yj + u_zk) \sin \frac{\theta}{2}$$

$$q = \begin{bmatrix} q_w \\ q_x \\ q_y \\ q_z \end{bmatrix}$$

Scalar part

Vector part

A quaternion can again be converted to a Direction Cosine Matrix.

\[
A(q) = \begin{bmatrix}
q_x^2 - q_y^2 - q_z^2 + q_w^2 & 2(q_x q_y + q_z q_w) & 2(q_x q_z - q_y q_w) \\
2(q_z q_y - q_x q_w) & -q_x^2 + q_y^2 - q_z^2 + q_w^2 & 2(q_y q_z + q_x q_w) \\
2(q_x q_z + q_y q_w) & 2(q_y q_z - q_x q_w) & -q_x^2 - q_y^2 + q_z^2 + q_w^2
\end{bmatrix}
\]

There are no trigonometric functions in this equation.
## ATTITUDE PARAMETERIZATION

### Overview

<table>
<thead>
<tr>
<th>Parameterization</th>
<th>Advantages</th>
<th>Disadvantages</th>
<th>Common Applications</th>
</tr>
</thead>
</table>
| Direction Cosine Matrix | • No Singularities  
• No trigonometric functions  
• Convenient product rule for successive rotations | • Six redundant parameters | In analysis, to transform vectors from one reference frame to another |
| Euler Angles          | • No redundant parameters  
• Physical interpretation is clear in some cases | • Trigonometric functions  
• Singularity possible  
• No convenient product rule for successive rotations | Analytical studies and as input/output for human manipulation |
| Quaternion            | • No singularities  
• No trigonometric functions  
• Convenient product rule for successive rotations | • One redundant parameter  
• No obvious physical interpretation | • Onboard the spacecraft, in the control laws. |
OVERVIEW

• Introduction
• Rationale
• Attitude Requirements and Errors
• Governing Equations
• Attitude Parameterization
• Disturbance Torques
• Passive Control
• Active Control
• Going Further
• I am interested, what now?
DISTURBANCE TORQUES

Which torques are disturbing the spacecraft attitude?
DISTURBANCE TORQUES
Different Sources

External
- Aerodynamic Drag
- Gravity Gradient
- Solar Radiation Pressure
- Magnetic
- Jettisoned parts
- Venting (relief valves)

Internal
- Solar panel deployment
- Instrument motion
- Liquid sloshing
- Reaction wheel vibrations
- Etc.

Affect the total angular momentum
Total angular momentum conserved but influence spacecraft orientation
The drag force produces a disturbance torque on the spacecraft due to any offset that exists between the aerodynamic center of pressure and the center of mass.

\[ \vec{T}_{AD} = \vec{r}_{cp} \times \vec{F}_a \]

Center of Pressure vector relative to the Center of Mass

Frontal Area

\[ \vec{F}_a = \frac{1}{2} \rho V^2 S C_D \frac{\vec{V}}{V} \]

~2 for free-molecular flow

Center of Mass

Center of Pressure
Above 200km altitude, the mean free path is significantly greater than the dimensions of most space vehicles

- Aerodynamics must be based upon free molecular flow
- Heat exchange solely due to radiation (no convection)

<table>
<thead>
<tr>
<th>Altitude (km)</th>
<th>( \lambda_0 ) (m)</th>
<th>Altitude (km)</th>
<th>( \lambda_0 ) (m)</th>
</tr>
</thead>
<tbody>
<tr>
<td>100</td>
<td>0.142</td>
<td>300</td>
<td>2.6 \times 10^3</td>
</tr>
<tr>
<td>120</td>
<td>3.31</td>
<td>400</td>
<td>16 \times 10^3</td>
</tr>
<tr>
<td>140</td>
<td>18</td>
<td>500</td>
<td>77 \times 10^3</td>
</tr>
<tr>
<td>160</td>
<td>53</td>
<td>600</td>
<td>280 \times 10^3</td>
</tr>
<tr>
<td>180</td>
<td>120</td>
<td>700</td>
<td>730 \times 10^3</td>
</tr>
<tr>
<td>200</td>
<td>240</td>
<td>800</td>
<td>1400 \times 10^3</td>
</tr>
</tbody>
</table>
July 15, 2000: a strong solar flare heated the Earth’s atmosphere, increasing the air density to a value 100 times greater than that for which its ADCS had been designed to cope. The magnetorquers were unable to compensate and the satellite was lost.
DISTURBANCE TORQUES
Aerodynamic Drag Torque: Example

\[ S = 5 \, m^2 \]
\[ C_D = 2 \]
\[ h = 400 \, km \to \rho = 4 \times 10^{-12} \frac{kg}{m^3} \]
\[ r_{CP} = 0.01m \]

\[ T = 1.2 \times 10^{-5} \text{Nm} \]

For \( I = 1000 \, kgm^2 \)

\[ T = \frac{dH}{dt} = I \frac{d\omega}{dt} = I \frac{d^2 \omega}{dt^2} \]
\[ \theta(t) = \frac{T}{I} t^2 = 1.2 \times 10^{-8} s^{-2} t^2 \]

Drift of 57deg after 2 orbits (around 10,000 s) !!! Unacceptable.
Gravitational fields decrease with distance from the center of the planet → a spacecraft in orbit experiences a stronger attraction on its “lower” side than its “upper” side.

\[ \vec{T}_{GG} \]

\[ \vec{F}_{gravity1} \]

\[ \vec{F}_{gravity2} \]

\[ \Delta r \]

\[ \vec{r}_{pos} \]
DISTURBANCE TORQUES
Gravity Gradient Torque

\[ d\vec{F} = -\mu \frac{\hat{\rho} + \hat{r}}{|\hat{\rho} + \hat{r}|^3} dm \]

Gravitational constant
(for Earth: \(3.986 \times 10^{14} \text{ m}^3/\text{s}^2\))

\[ d\vec{T} = \hat{\rho} \times d\vec{F} \rightarrow \vec{T} = \int \hat{\rho} \times \left(-\mu \frac{\hat{\rho} + \hat{r}}{|\hat{\rho} + \hat{r}|^3}\right) dm \]
DISTURBANCE TORQUES
Gravity Gradient Torque

\[ \hat{\rho} \ll \hat{r} \Rightarrow \frac{1}{|\hat{\rho} + \hat{r}|^3} \approx r^{-3} \left[ 1 - 3 \frac{\hat{r}}{r^2} \hat{\rho} \right] \]

\[ \vec{T} = \int \hat{\rho} \times \left[ \frac{-\mu}{r^3} \left( 1 - 3 \frac{\hat{r}}{r^2} \hat{\rho} \right) \hat{r} \right] dm \]
\[ \int \hat{\rho} \, dm = 0, \hat{r} = \frac{\vec{r}}{r} \]

\[ \vec{T} = \frac{3\mu}{r^3} \left( \int \hat{\rho} \cdot \hat{r} \, dm \right) \hat{r} \times \hat{r} \]

\[ \vec{T} = \frac{3\mu}{r^3} \hat{r} \times \vec{I} \hat{r} \]

Perpendicular to Earth-spacecraft vector
Further derivation leads to the following equation we can use to calculate the gravity gradient torque around one axis:

\[
\vec{T}_{GG} = \frac{3\mu}{2r^3} |I_z - I_y| \sin(2\theta)
\]

- \( \mu \) is the orbit radius (m)
- \( \theta \) is the maximum deviation of the Z-axis from local vertical in radians
- \( I_z \) and \( I_y \) are moments of inertia about z and y axes in kgm².

Altitude = 700km
Inertia moment difference = 30 kgm²
\( \Theta = 1 \text{ deg} \)

\[
\vec{T}_{GG} = \frac{3 \times 3.986 \times 10^{14}}{2 \times (7,078 \times 10^6)^3} 30 \sin(2\text{deg}) = 1.8 \times 10^{-6} \text{Nm}
\]

Solar radiation pressure produces a disturbance torque on the spacecraft, which depends upon the distance from the sun. It is independent of spacecraft position and velocity and is perpendicular to the sun line.

Solar radiation is NOT related to solar wind, which is a continuous stream of particles emanating from the sun. The momentum flux in the solar wind is small compared with that due to solar radiation.
DISTURBANCE TORQUES
Solar Radiation Pressure Torque

\[ \vec{T}_{SP} = \vec{r}_{CP} \times \vec{F}_s \]

Projected area normal to sun vector

\[ \vec{F}_s = (1 + K) \rho_s A_p \]

Surface reflectivity

1360 \( W/m^2 \)

\( \frac{3 \times 10^8 \text{m/s}}{3 \times 10^8 \text{m/s}} \)

Vector from the center of mass to the center of solar pressure

\[ A_p = 5 m^2, K = 0.5, r = 0.1 m \]

\[ \vec{T}_{SP} = 3.5 \times 10^{-6} Nm \]
The magnetic field generated by a spacecraft interacts with the local field from the Earth.

\[
\vec{T}_{MA} = \vec{M} \times \vec{B}
\]

- **Spacecraft magnetic dipole moment** (current loops + residual magnetic moment). E.g., residual moment in the magnetorquers of PROBA 2 = 0.03 Am²
- **Earth magnetic field vector in spacecraft coordinates** (proportional to 1/r³ with r the distance of the center of Earth to the spacecraft.)
- **Current loop through solar panels**

How can we reduce the "disturbance" magnetic moment generated by the current loops in the solar panels?
The magnetic field generated by a spacecraft interacts with the local field from the Earth.

\[ \vec{T}_{MA} = \vec{M} \times \vec{B} \]

Spacecraft magnetic dipole moment (current loops + residual magnetic moment). E.g., residual moment in the magnetorquers of PROBA 2 = 0.03 Am²

Earth magnetic field vector in spacecraft coordinates (proportional to $1/r^3$ with $r$ the distance of the center of Earth to the spacecraft.)

Current through the loop

\[ M = nIS \]

Number of turns Surface area of the loop

Counteracting current loop through solar panels
\[ \vec{T}_{MA} = \vec{M} \times \vec{B} \]

\[ B \text{ @ 700km altitude} = \frac{2M_{\text{Earth}}}{R^3} = \frac{2 \times 7.96 \times 10^{15}}{(6378 + 700)^3} \approx 4.5 \times 10^{-5}\text{Tesla} \]

\[ M \text{ (Small spacecraft)} \approx 0.1\text{Am}^2 \]

\[ \vec{T}_{MA} = 4.5 \times 10^{-6}\text{Nm} \]
The Earth’s Magnetic Field

The Earth’s North Magnetic Pole is, in fact, a south pole. (North poles on compasses point towards it.) Notice that the compass needle in the picture has the white (south) tip pointing north, and the field line arrows point from south to north.

Larger versions of this image are available: contact peter.reid@ed.ac.uk

Peter Reid, 2007
Total magnetic field intensity at Earth’s surface in Gauss (=0.0001 Tesla) in 1965
## Disturbance Torques

### Overview

The effect of disturbance torques generally decreases when the orbit is higher.

<table>
<thead>
<tr>
<th>Disturbance</th>
<th>Dependence on the distance to the Earth</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aerodynamic</td>
<td>( e^{-ar} ) ( \text{Strong dependence on solar activity and also day/night} )</td>
</tr>
<tr>
<td>Gravity Gradient</td>
<td>( \frac{1}{r^3} )</td>
</tr>
<tr>
<td>Solar Radiation Pressure</td>
<td>Independent ( \text{Almost no dependence on solar activity but care for eclipse!} )</td>
</tr>
<tr>
<td>Magnetic</td>
<td>( \frac{1}{r^3} )</td>
</tr>
</tbody>
</table>
Solar radiation pressure can be the primary source of disturbance torque. The lifetime of a GEO satellite is often controlled by the mass budget available for stationkeeping and attitude control fuel. Designers must avoid center-of-mass to center-of-pressure offsets.

Different disturbance torques must be considered.

The aerodynamic torque is dominant.
OVERVIEW

- Introduction
- Rationale
- Attitude Requirements and Errors
- Governing Equations
- Attitude Parameterization
- Disturbance Torques
- Passive Control
- Active Control
- Going Further
- I am interested, what now?
PASSIVE CONTROL

How can the spacecraft attitude be controlled* passively?

* Kept stable
INTRODUCTION

Rationale

Attitude Requirements and Errors

Governing Equations

Attitude Parameterization

Disturbance Torques

Passive Control

Active Control

Going Further

I am interested, what now?
PASSIVE CONTROL
Principle

The disturbance torques can be used to our advantage to keep the spacecraft stable.

Passive control is not really control, in the sense that we can command the spacecraft to take a certain attitude. It rather allows to keep the spacecraft stable.
PASSIVE CONTROL
Aerodynamic stabilization

At Low Earth Orbits, the aerodynamic drag can be high enough to align the most aerodynamic side of the spacecraft with the velocity vector.

e.g. The Belgian CubeSat QARMAN will rely on its aerodynamic properties to maintain a stable attitude during re-entry.
Gravity Gradient stabilization uses the inertial properties of the spacecraft to align the spacecraft longitudinal axis to the Earth’s center. This does not influence the yaw angle of the spacecraft.

Long booms with weights are used to achieve the desired inertias and to have sufficient stabilization.
Solar Radiation Pressure stabilization is not as common.

When the second of four reaction wheels of Kepler failed, Kepler mission and Ball Aerospace engineers developed an innovative way of recovering pointing stability by maneuvering the spacecraft so that the solar pressure is evenly distributed across the surfaces of the spacecraft.
Magnetic stabilization uses permanent magnets on board of the spacecraft to align it with the Earth’s magnetic field.

This is most effected in near-equatorial orbits where the field orientation stays almost constant for an Earth-pointing spacecraft.

The intrinsic gyroscopic stiffness of a spinning body is used to maintain its orientation in inertial space.

Unlike the other types of stabilization, we maintain here the orientation in an inertial space (conservation of angular momentum).

The advantage of spinning a spacecraft is that the thermal environment improves (more equal exposure for components). For scanning payloads, a spin can be even required.

A real (therefore flexible) body can spin stably only about the axis of maximum moment of inertia.

→ The vehicle must be a “wheel” rather than a “pencil”.

Meteosat
Spin stabilized with nutation damper
**PASSIVE CONTROL**

Spin Stabilization

**NO SPIN**

\[ T = I \dot{\theta} \]

Constant disturbing torque, along one axis

\[ \theta = \frac{1}{2} \frac{T}{I} t^2 \]

Small disturbance torque can lead to large deviation angle. Quadratic growth

**SPIN STABILIZED**

Constant disturbing torque, Perpendicular to the angular momentum

\[ \vec{T} = \frac{\Delta \vec{H}}{\Delta t}, \theta \approx \frac{\Delta H}{H} = \frac{T \Delta t}{H} \]

Linear growth inversely proportional to H and hence to the angular velocity (Faster spin rate \(\rightarrow\) Less effect of disturbances)
ACTIVE CONTROL

How can the spacecraft attitude be controlled actively?
With active attitude control, we estimate the spacecraft attitude and control actuators to actively change it to a desired attitude.
With active attitude control, we estimate the spacecraft attitude and control actuators to actively change it to a desired attitude.
The gyroscope is a special case among the sensors, in the sense that it does not measure the absolute attitude of the spacecraft, but rather the change in attitude (the rotational rate).

A gyroscope can therefore only propagate the attitude.

A set of three orthogonal gyroscopes measures the three components of the spacecraft angular velocity.

**MEMS gyroscopes:** Cheap, but low accuracy

**Laser ring gyroscopes:** Expensive, but high accuracy
ACTIVE CONTROL
Sun Sensor

The sun sensors measures the position of the sun, relative to the spacecraft.

If we know the vector towards the sun in the body frame (measured by the sun sensor) and we know the vector towards the sun in the inertial frame (calculated based on the known spacecraft position and the time), we have obtained knowledge about the spacecraft attitude.

→ However: One vector is not enough to know the attitude! (The spacecraft can still be rotated around the known vector) → A second vector is needed
The digital output of this sensor allows to determine the plane in which the sun lies.

Two such detectors mounted orthogonally can determine the spacecraft to Sun vector in the body frame.
ACTIVE CONTROL
Sun Sensor

<table>
<thead>
<tr>
<th>Specification</th>
<th>Details</th>
</tr>
</thead>
<tbody>
<tr>
<td>Field of View</td>
<td>120° x 120°</td>
</tr>
<tr>
<td>Output</td>
<td>2-axis, 16 bit data word per axis, through CAN interface</td>
</tr>
<tr>
<td>Accuracy</td>
<td>0.02° (95% probability)</td>
</tr>
<tr>
<td>Noise Equivalent Angle</td>
<td>0.01°</td>
</tr>
<tr>
<td>Resolution</td>
<td>0.01°</td>
</tr>
<tr>
<td>Data processing</td>
<td>In AOCS system</td>
</tr>
<tr>
<td>Operating temperature</td>
<td>-50°C to +80°C</td>
</tr>
<tr>
<td>Mass</td>
<td>250g</td>
</tr>
<tr>
<td>Power consumption</td>
<td>&lt;1W</td>
</tr>
<tr>
<td>Supply Voltage</td>
<td>5 Volts DC (±0.25 VDC, 135 mV ripple)</td>
</tr>
</tbody>
</table>
ACTIVE CONTROL
Magnetometer

The magnetometer measures the direction of the magnetic field, relative to the spacecraft.

With three orthogonal magnetometers, the magnetic field vector can be measured.

If we know the vector along the magnetic field in the body frame (measured by the magnetometer) and we know the vector along the magnetic field in the inertial frame (calculated based on the known spacecraft position and a magnetic field model), we have obtained knowledge about the spacecraft attitude.

One vector is not enough to know the attitude!
The Earth sensor measures the direction towards the Earth, relative to the spacecraft.

If we know the vector towards the Earth in the body frame (measured by the Earth sensor) and we know the vector towards the Earth in the inertial frame (calculated based on the known spacecraft position), we have obtained knowledge about the spacecraft attitude.

One vector is not enough to know the attitude!
Earth sensors can determine the vector towards the center of the Earth by:

1. Taking an image of the Earth and performing image processing.
2. Sensing the horizon of the Earth (typically in infrared).
   - Works in eclipse
ACTIVE CONTROL
Star Tracker

The star tracker determines the attitude based on an image of the stars.

A star tracker can fully determine the spacecraft attitude (if there are at least two stars in the image).

Using the stars to determine your orientation and the direction in which you should travel has been done for centuries.
ACTIVE CONTROL
Star Tracker

How a star tracker works can be simply explained with an example that everyone knows.

Let’s say we want to know where North is on a starry night.

1. We search for the Big Dipper and can recognize it based on its famous shape. **Star Identification**

2. Based on this, we know where the North Star is. We know that if we look at it, we are looking North (knowledge on our orientation). **Star Tracking**

A star tracker follows the same principle, it recognizes stars in the image and matches it to their known position in the inertial frame. It does this for thousands of stars with high accuracy.
The baffle shields the star tracker from light that comes from outside of the field of view (e.g. from the Sun, Moon or Earth).

The detector actually takes the image. This used to be a CCD detector but more often, a CMOS is used.

The lens focuses the light and determines the focal length, how much light enters, etc.

The processing electronics determine the attitude based on the star image.
ACTIVE CONTROL
Star Tracker Software

Centroiding
Determine the centroids of stars

Lost-in-Space or Matching
Identify corresponding database stars

Tracking
Estimate the attitude

Image (body frame)

Database (inertial frame)

Hipparchos Database
### Sensor Overview

<table>
<thead>
<tr>
<th>Sensor</th>
<th>Performance (deg)</th>
<th>Availability</th>
</tr>
</thead>
<tbody>
<tr>
<td>Gyroscope</td>
<td>0.01/h (drift)</td>
<td>Continuous coverage</td>
</tr>
<tr>
<td>Sun Sensor</td>
<td>0.01-0.1 (angle that the sun subtends)</td>
<td>Outside of eclipse (intermittent use)</td>
</tr>
<tr>
<td>Magnetometer</td>
<td>1 (variability and uncertainty of the magnetic field)</td>
<td>Continuous coverage, but below 6000 km</td>
</tr>
<tr>
<td>Earth Sensor</td>
<td>0.02-0.03 (Earth oblateness + fuzziness of horizon)</td>
<td>Continuous coverage</td>
</tr>
<tr>
<td>Star Tracker</td>
<td>0.001</td>
<td>Not blocked by the Earth or Sun in the FOV (intermittent use)</td>
</tr>
</tbody>
</table>
The estimator fuses the information of different sensors and can use a model of the spacecraft to obtain an accurate estimate of the spacecraft attitude and rotational rate.

- Extended Kalman Filter
- Unscented Kalman Filter
- Particle Filter
- ....

Important and interesting Control Theory problems, but not the focus of this lesson.
The controller determines the action of the actuators. A spacecraft typically has a set of controllers for different situations (e.g. to detumble from fast rates, for fine control, etc.)

- Detumble Bdot controller
- PID Controller
- Model Predictive Control

Important and interesting Control Theory problems, but not the focus of this lesson.
The actuators change the orientation of the spacecraft
ACTIVE CONTROL
Actuators

Torques

External
- Thrusters
- Magnetorquers

Internal
- Reaction Wheel

Affect the total angular momentum

Necessary

Total angular momentum conserved but influence spacecraft orientation

Optional
ACTIVE CONTROL
Thrusters

- Common and effective means of providing spacecraft attitude control.
- Common on satellites intended to operate in relatively high orbit, where a magnetic field will not be available for angular momentum dumping.
- Potentially largest source of torque
- Usually a redundant set of thrusters
- Thrusters for orbit control may or may not be used as attitude thrusters as well
A magnetorquer is in essence an electrical coil that generates a magnetic moment $m$, that interacts with the Earth’s magnetic field to generate a torque.

This leads to a torque:

$$\vec{T} = \vec{m} \times \vec{B}$$

Magnetorquers do not allow for full 3-axis control: If the magnetic moment is aligned with the Earth’s magnetic field, no torque can be created.
Reaction wheels are spinning flywheels mounted on a central bearing whose rate of rotation can be adjusted by an electric motor.
They exchange momentum with the spacecraft by changing wheel speed but no influence on the total angular momentum.

If the reaction wheel spins up in one direction, the satellite spins up in the other direction.

\[
\mathbf{H}_{\text{sat}} + \mathbf{H}_{\text{wheel}} = \mathbf{0}
\]

\[
\mathbf{I}_{\text{sat}} \mathbf{\omega}_{\text{sat}} + \mathbf{I}_{\text{wheel}} \mathbf{\omega}_{\text{wheel}} = \mathbf{0}
\]

What do the torque, velocity and orientation profile of the satellite look like when performing a slew maneuver with a reaction wheel?

https://www.youtube.com/watch?v=-Cc-jGnlwCM
ACTIVE CONTROL
Reaction Wheels: Slew

Let’s say we want to perform a slew maneuver of 90 degrees around one axis.
ACTIVE CONTROL
Reaction Wheels: Slew

Let’s say we want to perform a slew maneuver of 90 degrees around one axis.
Let's say we want to perform a slew maneuver of 90 degrees around one axis.
Let's say we want to perform a slew maneuver of 90 degrees around one axis.

Momentum is borrowed from the wheels and then returned to the wheel. There is no net change of momentum of the wheel in this case.
External torques give rise to unwanted angular momentum. The control system applies control torques to the reaction wheels to leave the spacecraft angular momentum unchanged.

Example: when a clockwise disturbance torque is imposed on the spacecraft, the attitude control system holds attitude constant by rotating a reaction wheel counterclockwise.
When disturbing torques do not average out over one orbit, constant wheel speed increase is necessary to hold the spacecraft.

There is a risk to “saturate” the wheel; the wheel is spinning at its maximum rate and cannot counterbalance further disturbing torques.

The stored momentum needs to be cancelled; this process is called momentum dumping.

Thrusters or magnetorquers are used to hold the spacecraft stationary while reducing wheel speed.
Wheels are not operated near 0 rpm, because:
- Nonlinear wheel friction.
- Zero-crossing is a large factor in bearing wear
- Hard to control electronically

The rotational axis of a wheel is usually aligned with a vehicle control axis; the vehicle must carry one wheel per axis for full attitude control.

Redundancy is usually desired, requiring four or more wheels, in a position oblique to all axes.

https://n-avionics.com/subsystems/cubesat-reaction-wheels-control-system-satbus-4rw/
# ACTIVE CONTROL

## Reaction Wheels: Some Remarks

<table>
<thead>
<tr>
<th>PERFORMANCE ITEM</th>
<th>UNIT</th>
<th>HR12</th>
<th>CAPABILITY HR14</th>
<th>HR16</th>
</tr>
</thead>
<tbody>
<tr>
<td>Momentum</td>
<td>N·m·s</td>
<td>12, 25, 50³</td>
<td>25, 50, 75</td>
<td>50, 75, 100'</td>
</tr>
<tr>
<td>Reaction torque</td>
<td>N·m</td>
<td></td>
<td>0.1 to 0.2³</td>
<td></td>
</tr>
<tr>
<td>Nominal</td>
<td>N·m</td>
<td></td>
<td>up to 0.4 (@3000 rpm)³</td>
<td></td>
</tr>
<tr>
<td>Extended</td>
<td>N·m</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Rotor Balance</td>
<td>g·cm</td>
<td>0.15, 0.24, 0.44</td>
<td>0.22, 0.35, 0.48</td>
<td>0.28, 0.38, 0.48</td>
</tr>
<tr>
<td>Static</td>
<td>g·cm²</td>
<td>2.2, 4.6, 9.1</td>
<td>4.6, 9.1, 13.7</td>
<td>7.7, 11.5, 15.4</td>
</tr>
<tr>
<td>Dynamic</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Peak Power</td>
<td>Watts</td>
<td>105, 195³</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Steady State (@ 6000 rpm)</td>
<td>Watts</td>
<td>&lt; 22 typical</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Bus Voltage Range</td>
<td>Volts</td>
<td>14 up to 80</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Wheel Speed</td>
<td>rpm</td>
<td>± 6000</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Mass</td>
<td>kg</td>
<td>6.0, 7.0, 9.5</td>
<td>7.5, 8.5, 10.6</td>
<td>9.0, 10.4, 12</td>
</tr>
<tr>
<td>Integrated Wheel Outline (Height x Width)</td>
<td>mm</td>
<td>159 x 316</td>
<td>159 x 366</td>
<td>178 x 418</td>
</tr>
<tr>
<td>Separate Electronics Outline</td>
<td>mm</td>
<td></td>
<td>WU H148X316D</td>
<td>WU H152X418D</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>WDE H60XW160XL230</td>
<td>WDE H60XW160XL230</td>
</tr>
<tr>
<td>Life</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Storage</td>
<td>Years</td>
<td>&gt; 5</td>
<td></td>
<td></td>
</tr>
<tr>
<td>On-orbit Operation</td>
<td>Years</td>
<td>&gt; 15</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Radiation Hardness Capability</td>
<td>Krad (Si)</td>
<td>&gt; 300</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Parts Screening</td>
<td></td>
<td>S</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Operational Temperature Range (Qual)</td>
<td>℃</td>
<td>-30 to 70</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Vibration</td>
<td>Grms</td>
<td>13.8</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Interface</td>
<td>NA</td>
<td></td>
<td>Analog/Digital</td>
<td></td>
</tr>
</tbody>
</table>
# ACTIVE CONTROL
## Actuators Overview

<table>
<thead>
<tr>
<th>Actuator</th>
<th>Performance (deg)</th>
<th>Availability</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrusters</td>
<td>0.05</td>
<td>Used in any environment, but needs propellant (limits lifetime)</td>
</tr>
<tr>
<td>Magnetorquers</td>
<td>1-5</td>
<td>Need a strong magnetic field (near Earth usage)</td>
</tr>
<tr>
<td>Reaction Wheels</td>
<td>0.005</td>
<td>Used in any environment, but need external torque device for momentum dumping.</td>
</tr>
<tr>
<td>Control Moment Gyros</td>
<td>0.005</td>
<td>Used in any environment</td>
</tr>
</tbody>
</table>
Examples of ADCS Configurations

Depending on the mission requirements, different ADCS configurations can be used.

The simplest configuration that delivers the required performance should be selected.

\[ > 5 \text{ deg} \]

If no control is needed and we just need to keep the spacecraft stable

\[ \rightarrow \text{Passive stabilization} \]

Otherwise: sun sensors + magnetometer and magnetorquers

\[ 1-5 \text{ deg} \]

Sun sensors + horizon sensors + magnetometers + gyroscope

Reaction wheels or magnetorquers (thrusters if spin stabilization is used)

\[ 0.1-1 \text{ deg} \]

Star Tracker + (Sun sensors + horizon sensors + magnetometers + gyroscope)

Reaction wheels with magnetorquers or thrusters

Flexible effects not modeled in the controller

\[ <0.1 \text{ deg} \]

Star Tracker + (Sun sensors + horizon sensors + magnetometers + gyroscope)

Reaction wheels with magnetorquers or thrusters

Flexible effects modeled in the controller

Possibly vibration-isolation payload platform
ACTIVE CONTROL
Examples of ADCS Configuration: Mars Express

8 attitude thrusters (10 N each)
4 reaction wheels (12 NMs)

Star trackers
Gyroscopes
Sun sensors
ACTIVE CONTROL
Designing the ADCS

**INPUTS**

- **Mission**: orbit, cost, lifetime, payload, pointing requirements, spacecraft type, etc.
- **Thermal control**
- **Power**
- **Communication**
- **Propulsion**

**OUTPUTS**

- **Propulsion**: Thruster type + amount of propellant
- **Power**: (ADCS power consumption)
- **Structure**: Center of mass + inertia constraints, flexibility, sensor and thruster locations

**Component selection**

**Configuration**

**Computational architecture**

**Hardware design**
ACTIVE CONTROL
When things go wrong

Another NASA space telescope shuts down in orbit
October 12, 2018 by Marcia Dunn

Aging reaction wheels
ACTIVE CONTROL
When things go wrong

Software error doomed Japanese Hitomi spacecraft
Space agency declares the astronomy satellite a loss.

Alexandra Witze
28 April 2016
OVERVIEW

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• Passive Control
• Active Control
• Going Further
• I am interested, what now?
GOING FURTHER

What lies beyond?
Main trends:

• Miniaturization: Small spacecraft are on the rise and the ADCS needs to scale with it.

• More accurate: There is no such thing as “good enough” for scientists. They always want higher accuracy.
Once we want to point a satellite with arc second range accuracy, we can no longer rely on the ADCS alone. (The sensors and actuators are too slow to compensate for high frequency errors)
Once we want to point a satellite with arc second range accuracy, we can no longer rely on the ADCS alone.

(The sensors and actuators are too slow to compensate for high frequency errors)
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I AM INTERESTED, WHAT NOW?

How can I find eternal glory in the field of attitude determination and control?
REFERENCES


