ADCS Review

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• Attitude Determination and Control Subsystem (ADCS) Function
• Impact of Mission Requirements and Other Subsystems on ADCS
• ADCS Design Process
• Spacecraft Coordinate Systems
• Spacecraft Dynamic & Kinematical Equations
• Key Design Parameters
• Typical Environmental Disturbance
• Assignment – Gravity Gradient Stabilization
The ADCS stabilizes the spacecraft and orients it in desired directions during the mission despite the external disturbance torques acting on it:

- To stabilize spacecraft after launcher separation
- To point solar array to the Sun
- To point payload (camera, antenna, and scientific instrument etc.) to desired direction
- To perform spacecraft attitude maneuver for orbit maneuver and payloads operation

This requires that the spacecraft determine its attitude, using sensors, and control it, using actuators.
Impact of Mission Requirements and Other Subsystems on ADCS

**Thermal**
- Special thermal maneuvers required?

**Command & Data Handling**
- Data processing capability

**Propulsion**
- 1. Thruster size
- 2. Propellant load

**Communications**
- Antenna pointing accuracy?

**Mission**
- 1. Earth-Pointing or inertial-Pointing?
- 2. Control during ΔV burn?
- 3. Separate payload platform?
- 4. Accuracy/Stability needs?
- 5. Slewing requirement?
- 6. Orbit?
- 7. Autonomy?
- 8. Mission life?
- 9. On-board navigation data required?

**Power**
- 1. ADCS load
- 2. Special regulation

**Structures**
- 1. Center of mass constraints
- 2. Inertia constraints
- 3. Flexibility constraints
- 4. Thruster location
- 5. Sensor mounting

**ADCS Trades**
- 1. Spinner vs. Active vs. Passive
- 2. Attitude determination (On-orbit or Ground)
- 3. Sensor selection
- 4. Actuator selection
- 5. Computational architecture

1. Data processing capability
2. Mission load
3. Special regulation
4. Power
5. Structures
6. ADCS Trades
7. Mission
8. Command & Data Handling
9. Propulsion
10. Communications
# ADCS Design Process

<table>
<thead>
<tr>
<th>Step</th>
<th>Inputs</th>
<th>Outputs</th>
</tr>
</thead>
<tbody>
<tr>
<td>1a) Define control mode</td>
<td>• Mission requirements • Mission profile • Type of insertion for launch vehicle</td>
<td>• List of different control mode during mission • Requirement and constraints</td>
</tr>
<tr>
<td>1b) Define or derive system-level requirements by control mode</td>
<td>• Mission requirements • Mission profile • Type of insertion for launch vehicle</td>
<td></td>
</tr>
<tr>
<td></td>
<td>• List of different control mode during mission</td>
<td></td>
</tr>
<tr>
<td></td>
<td>• Requirement and constraints</td>
<td></td>
</tr>
<tr>
<td>2) Select type of spacecraft control by attitude control mode</td>
<td>• Payload, thermal &amp; power needs • Orbit, pointing direction • Disturbance environment</td>
<td>• Method for stabilizing and control: three-axis control, spinning, or gravity gradient</td>
</tr>
<tr>
<td>3) Quantify disturbance environment</td>
<td>• Spacecraft geometry • Orbit/Solar/Magnetic models • Mission profile</td>
<td>• Environmental disturbance: forces/torques</td>
</tr>
<tr>
<td>4) Select &amp; size ADCS hardware</td>
<td>• Spacecraft geometry • Pointing accuracy • Orbit conditions • Mission requirements • Slew rate</td>
<td>• Sensor suite: Earth, Sun, inertial, or other sensing devices • Actuators: reaction wheels, thrusters, or magnetic torquer • Data processing requirements</td>
</tr>
<tr>
<td>5) Define determination and control Algorithms (Design/Analysis/Simulation)</td>
<td>• All of above</td>
<td>• Algorithms, parameters, and logic for each determination and control mode</td>
</tr>
<tr>
<td>6) Iterate and document</td>
<td>• All of above</td>
<td>• Refines requirements and design • Subsystem specification</td>
</tr>
</tbody>
</table>
## Typical ADCS Requirements

<table>
<thead>
<tr>
<th>Determination</th>
<th>Definition</th>
<th>Examples/Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Accuracy</strong></td>
<td>How well a vehicle’s orientation with respect to an absolute reference is known</td>
<td>0.25 deg, 3-sigma, all axes; may be real-time or post-processed on the ground</td>
</tr>
<tr>
<td><strong>Range</strong></td>
<td>Range of angular motion over which accuracy must be met</td>
<td>Any attitude within 30 deg of nadir</td>
</tr>
<tr>
<td><strong>Control</strong></td>
<td>Definition</td>
<td>Examples/Comments</td>
</tr>
<tr>
<td><strong>Accuracy</strong></td>
<td>How well the vehicle attitude can be controlled with respect to a commanded direction</td>
<td>0.25 deg, 3-sigma, including determination and control errors, may be taken with respect to an inertial or Earth-fixed reference</td>
</tr>
<tr>
<td><strong>Range</strong></td>
<td>Range of angular motion over which control performance must be met</td>
<td>All attitude, within 50 deg of nadir, within 20 deg of Sun</td>
</tr>
<tr>
<td><strong>Jitter</strong></td>
<td>A specified angle bound or angular rate limit on short-term, high-frequency motion</td>
<td>0.1 deg /min, 1deg/s, 1 to 20 Hz; usually specified to keep spacecraft motion from blurring sensor data</td>
</tr>
<tr>
<td><strong>Drift</strong></td>
<td>A limit on slow, low-frequency vehicle motion. Usually expressed as angle/time</td>
<td>1 deg/hr, 5deg max. Used when vehicle may drift off target with infrequent resets</td>
</tr>
<tr>
<td><strong>Setting Time</strong></td>
<td>Specifies allowed time to recover from maneuvers or upsets</td>
<td>2 deg max motion; may be used to limit overshoot or nutation</td>
</tr>
</tbody>
</table>
Attitude Control Type

- **Passive Control Techniques:**
  - **Gravity-gradient control** uses the inertial properties of a spacecraft to keep it pointed toward the Earth. This relies on the fact that an elongated object in the gravity field tends to align its longitudinal axis through the Earth’s center.
  - **Passive magnetic control** uses permanent magnets on board the spacecraft to force alignment along the Earth’s magnetic field. This is most effective in near-equatorial orbits where the field orientation stays almost constant.
  - **Spin stabilization** is a passive control technique in which the entire spacecraft rotates so that its angular momentum vector remains approximately fixed in inertial space. Spin-stabilized spacecraft employ the gyroscopic stability to passively resist disturbance torques about two axes.
Three-axis Active Control Technique:

- Spacecraft stabilized in three axes are more common today than those using passive control.
- It can be stable and accurate but also more expensive, complex, and potentially less reliable.
- Broadly, these systems take two forms: one uses momentum bias by placing a momentum wheel along the pitch axis; the other is called zero momentum with a reaction wheel on each axis. Either option usually needs thrusters or magnetic torquers for wheel momentum unloading.
# ADCS Control Methods and their Capabilities

<table>
<thead>
<tr>
<th>Type</th>
<th>Pointing Options</th>
<th>Attitude Maneuverability</th>
<th>Typical Accuracy</th>
<th>Lifetime Limit</th>
</tr>
</thead>
<tbody>
<tr>
<td>Gravity Gradient</td>
<td>Earth local vertical only</td>
<td>Very limited</td>
<td>+/- 5°</td>
<td>No limit</td>
</tr>
<tr>
<td>Passive Magnetic</td>
<td>Earth local vertical only</td>
<td>Very limited</td>
<td>+/- 5°</td>
<td>No limit</td>
</tr>
<tr>
<td>Spin Stabilization</td>
<td>Inertially fixed (any direction)</td>
<td>High propellant usage to move stiff momentum vector</td>
<td>+/- 0.1° ~ +/- 1.0° (proportional to spin rate)</td>
<td>Propellant</td>
</tr>
<tr>
<td>Pitch Momentum Bias</td>
<td>Best suited for Earth local vertical pointing</td>
<td>Momentum vector of the bias wheel prefers to stay normal to orbit plane</td>
<td>+/- 0.1° ~ +/- 1.0°</td>
<td>Propellant (if applies) Life of sensor and wheel bearing</td>
</tr>
<tr>
<td>Zero Momentum Bias (Thruster)</td>
<td>No constraints</td>
<td>No constraints High rates possible</td>
<td>+/- 0.1° ~ +/- 5.0°</td>
<td>Propellant</td>
</tr>
<tr>
<td>Zero Momentum Bias (Three wheels)</td>
<td>No constraints</td>
<td>No constraints</td>
<td>+/- 0.001° ~ +/- 1.0°</td>
<td>Life of sensor and wheel bearing</td>
</tr>
</tbody>
</table>
Control Example: Active Attitude Control System

General Structure of a Satellite Attitude Determination and Control Subsystem
## Typical ADCS Sensors

<table>
<thead>
<tr>
<th>Sensors</th>
<th>Typical Performance</th>
<th>Weight (kg)</th>
<th>Power (Watt)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Gyro</td>
<td>Drift rate: 0.003°/hr ~ 1°/hr</td>
<td>3 ~ 25</td>
<td>10 ~ 200</td>
</tr>
<tr>
<td>Sun sensor</td>
<td>Accuracy: 0.005° ~ 3°</td>
<td>0.5 ~ 2</td>
<td>0 ~ 3</td>
</tr>
<tr>
<td>Star sensor</td>
<td>Accuracy: 0.0003° ~ 0.01°</td>
<td>3 ~ 7</td>
<td>5 ~ 20</td>
</tr>
<tr>
<td>Horizon sensor</td>
<td>Accuracy: 0.1° ~ 1°</td>
<td>2 ~ 5</td>
<td>0.3 ~ 10</td>
</tr>
<tr>
<td>Magnetometer</td>
<td>Accuracy: 0.5° ~ 3°</td>
<td>0.6 ~ 1.2</td>
<td>0 ~ 1</td>
</tr>
</tbody>
</table>
## Typical ADCS Actuators

<table>
<thead>
<tr>
<th>Actuators</th>
<th>Typical Performance</th>
<th>Weight (kg)</th>
<th>Power (Watt)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrusters</td>
<td></td>
<td>Variable</td>
<td>N/A</td>
</tr>
<tr>
<td>• Hydrazine</td>
<td>0.5 ~ 9000 N*</td>
<td>Variable</td>
<td>N/A</td>
</tr>
<tr>
<td>• Cold gas</td>
<td>&lt; 5</td>
<td>Variable</td>
<td>N/A</td>
</tr>
<tr>
<td>Reaction Wheel</td>
<td>0.4 ~ 400 Nms (momentum) 0.01 ~ 1 Nm (torque)</td>
<td>2 ~ 20</td>
<td>10 ~ 110</td>
</tr>
<tr>
<td>Control moment gyro</td>
<td>25 ~ 500 Nm (torque)</td>
<td>&gt; 40</td>
<td>90 ~ 150</td>
</tr>
<tr>
<td>Magnetic torquer</td>
<td>1 ~ 4000 Am²</td>
<td>0.4 ~ 50</td>
<td>0.6 ~ 16</td>
</tr>
</tbody>
</table>

* Multiply by moment arm (typical 1 or 2 m) to get torque
Spacecraft Coordinate Systems

- Spacecraft Body Coordinate System

1. Spacecraft (ROCSAT-2) Coordinate System

2. Euler Angle Definition
Spacecraft Coordinate Systems (Cont.)
- Earth Centered Inertial (ECI) Coordinate System

$Z_{ECI}$: the rotation axis of the Earth
ECI is a inertial fixed coordinate system
Spacecraft Coordinate Systems (Cont.)
- Local Vertical Local Horizontal (LVLH) Coordinate System

LVLH is not a inertial fixed coordinate system
Spacecraft Attitude Definition

- Spacecraft Attitude: the orientation of the body coordinate w.r.t. the ECI (or LVLH) coordinate system
- Euler angle representation:
  - $[\phi \theta \psi]$ : rotate $\psi$ angle around $Z$-axis, then rotate $\theta$ angle around $Y$-axis, finally $\phi$ angle around $X$-axis
  - The transition matrix:

$$R_{123}(\phi, \theta, \psi) =
\begin{bmatrix}
1 & 0 & 0 \\
0 & \cos \phi & \sin \phi \\
0 & -\sin \phi & \cos \phi
\end{bmatrix}
\begin{bmatrix}
\cos \theta & 0 & -\sin \theta \\
0 & 1 & 0 \\
\sin \theta & 0 & \cos \theta
\end{bmatrix}
\begin{bmatrix}
\cos \psi & \sin \psi & 0 \\
-\sin \psi & \cos \psi & 0 \\
0 & 0 & 1
\end{bmatrix}$$
Quaternion: rotation an angle ($\theta$) around arbitrary axis ($N$) of ECI coordinate system:

- $Q = \begin{bmatrix} n_1 \sin(\theta/2) & n_2 \sin(\theta/2) & n_3 \sin(\theta/2) & \cos(\theta/2) \end{bmatrix}$
- $N \equiv \begin{bmatrix} n_1 & n_2 & n_3 \end{bmatrix}$
Spacecraft Dynamic & Kinematical Equations

Dynamic Equation (3x1) to be stabilized:

- \( I \): Spacecraft moment of inertia
- \( N \): External torque (Thruster & Environment)
- \( H_w \): Wheel angular momentum
- \( w \): Spacecraft angular velocity

\[
I \frac{d\vec{w}}{dt} = \vec{N} - \frac{d\vec{H}_w}{dt} - \vec{w} \times (I\vec{w} + \vec{H}_w)
\]
Kinematical Equation (4x1):

- \( Q \): Spacecraft quaternion

\[
\dot{Q} = \frac{1}{2} \bar{\omega} \otimes Q
\]

\[
Q = [q_1, q_2, q_3, q_4]^T
\]

\[
\dot{Q} = \frac{1}{2} \begin{bmatrix}
0 & w_z & -w_y & w_x \\
-w_z & 0 & w_x & w_y \\
w_y & -w_x & 0 & w_z \\
-w_x & w_y & -w_z & 0
\end{bmatrix} [q_1, q_2, q_3, q_4]^T
\]
### Key Design Parameters

- **Control Accuracy on Hardware Selection and ADCS Design**

<table>
<thead>
<tr>
<th>Required Accuracy (3σ)</th>
<th>ADCS Design</th>
<th>Hardware Selection</th>
</tr>
</thead>
</table>
| > 5°                   | • Permits major cost saving  
                        • Permit gravity gradient stabilization | • No sensors required for gravity gradient stabilization |
| 1° ~ 5°                | • Gravity gradient not feasible  
                        • Spin stabilization feasible  
                        • Three-axis stabilization will work | • Sensor: sun sensor & horizon sensor  
                        • Actuator: Thruster dead band control or wheel control  
                        • Magnetic torquer (and magnetometer) useful |
| 0.1° ~ 1°              | • Three-axis stabilization an momentum-bias stabilization feasible | • Sensor: star tracker or horizon sensor and gyros  
                        • Actuator: reaction wheel  
                        • Magnetic torquers feasible on light spacecraft (magnetometer also required) |
| < 0.1°                 | • Three-axis stabilization is necessary | • Need star tracker and better class of gyros  
                        • Control laws and computational needs are more complex  
                        • Flexible body performance is very importance |
Typical Environmental Disturbance

• **Gravity-Gradient Torque:**
  – Any nonsymmetrical object of finite dimensions in orbit is subject to a gravitational torque because of the variation in the Earth’s gravitational force over the object.

• **Solar Radiation Torque:**
  – Radiation incident on a spacecraft’s surface produces a force which results in a torque about the spacecraft’s center of mass.

• **Aerodynamic Torque**
  – The interaction of the upper atmosphere with a spacecraft’s surface produces a torque about the center of mass.

• **Magnetic Disturbance Torque**
  – Magnetic disturbance torques result from the interaction between the spacecraft’s residual magnetic field and the geomagnetic field.
The interaction between the satellite inertia ellipsoid and the gravitational field can be used to provide geocentric orientation of the satellite.

Passive gravity gradient stabilization achieves this orientation for spacecraft without use of active control elements.

Primary advantages of gravity gradient stabilization are long life, low power requirements, and Earth or planet pointing attitude capability.

The disadvantages are low control torques, libration damper requirements, and imprecise pointing capability (e.g., 1 to 10° about all axes).
For a satellite with orbital angular velocity $\omega_0$ and principal moments of inertia $I_1$, $I_2$, and $I_3$ in a circular orbit, Euler’s equation of motion in terms of the spacecraft angular momentum vector, and the gravity gradient torque vector is

$$\vec{T}_g = \dot{\vec{h}} + \vec{\omega} \times \vec{h}$$

where

$$\vec{h} = \omega_1 I_1 \vec{e}_1 + \omega_2 I_2 \vec{e}_2 + \omega_3 I_3 \vec{e}_3$$

$$\vec{T}_g = T_1 \vec{e}_1 + T_2 \vec{e}_2 + T_3 \vec{e}_3$$
Coordinate axes for an orbiting satellite.
Refer to the figure, the orientation of the orbiting reference frame is related to the spacecraft principal axes, through three sequential rotations $\theta_1$, $\theta_2$, and $\theta_3$. The transformation matrix can be written as
\[
\begin{pmatrix}
\vec{e}_1 \\
\vec{e}_2 \\
\vec{e}_3
\end{pmatrix} = \begin{pmatrix}
c\theta_3 & s\theta_3 & 0 \\
-s\theta_3 & c\theta_3 & 0 \\
0 & 0 & 1
\end{pmatrix} \begin{pmatrix}
c\theta_2 & 0 & -s\theta_2 \\
0 & 1 & 0 \\
0 & c\theta_1 & s\theta_1
\end{pmatrix} \begin{pmatrix}
1 & 0 & 0 \\
0 & c\theta_1 & s\theta_1 \\
0 & -s\theta_1 & c\theta_1
\end{pmatrix} \begin{pmatrix}
\vec{E}_1 \\
\vec{E}_2 \\
\vec{E}_3
\end{pmatrix}
\]

\[
= \begin{pmatrix}
(c\theta_2 c\theta_3) & (c\theta_1 s\theta_3 + c\theta_3 s\theta_1 s\theta_2) & (s\theta_3 s\theta_1 - c\theta_3 s\theta_2 c\theta_1) \\
(-c\theta_2 s\theta_3) & (c\theta_1 c\theta_3 - s\theta_3 s\theta_1 s\theta_2) & (c\theta_3 s\theta_1 s\theta_2 c\theta_1) \\
(s\theta_2) & (-c\theta_2 s\theta_1) & (c\theta_2 c\theta_1)
\end{pmatrix} \begin{pmatrix}
\vec{E}_1 \\
\vec{E}_2 \\
\vec{E}_3
\end{pmatrix}
\]

\[
= \begin{pmatrix}
a_{11} & a_{12} & a_{13} \\
a_{21} & a_{22} & a_{23} \\
a_{31} & a_{32} & a_{33}
\end{pmatrix} \begin{pmatrix}
\vec{E}_1 \\
\vec{E}_2 \\
\vec{E}_3
\end{pmatrix}
\]
For small angular deviations, the matrix can be written as

\[
\begin{pmatrix}
\vec{e}_1 \\
\vec{e}_2 \\
\vec{e}_3 \\
\end{pmatrix} = \begin{pmatrix}
1 & \theta_3 & -\theta_2 \\
-\theta_3 & 1 & \theta_1 \\
\theta_2 & -\theta_1 & 1 \\
\end{pmatrix} \begin{pmatrix}
\vec{E}_1 \\
\vec{E}_2 \\
\vec{E}_3 \\
\end{pmatrix}
\]

The body components of the spacecraft angular velocity are

\[
\begin{pmatrix}
\omega_1 \\
\omega_2 \\
\omega_3 \\
\end{pmatrix} = \begin{pmatrix}
\dot{\theta}_1 + \omega_0 \theta_3 \\
\dot{\theta}_2 + \omega_0 \\
\dot{\theta}_3 - \omega_0 \theta_1 \\
\end{pmatrix}
\]
Upon substitution of the expressions, the components of the Euler equation become

\[
I_1 (\ddot{\theta}_1 + \omega_0 \dot{\theta}_3) + (\dot{\theta}_2 + \omega_0)(\ddot{\theta}_3 - \omega_0 \theta_1)(I_3 - I_2) = T_1
\]
\[
I_2 \ddot{\theta}_2 + (\dot{\theta}_1 + \omega_0 \theta_3)(\ddot{\theta}_3 - \omega_0 \theta_1)(I_1 - I_3) = T_2
\]
\[
I_3 (\ddot{\theta}_3 - \omega_0 \dot{\theta}_1) + (\dot{\theta}_1 + \omega_0 \theta_3)(\dot{\theta}_2 + \omega_0)(I_2 - I_1) = T_3
\]

for a circular orbit,

\[
T_1 = 3 \omega_0^2 (I_3 - I_2) a_{21} a_{31} \approx 0
\]
\[
T_2 = 3 \omega_0^2 (I_1 - I_3) a_{11} a_{31} \approx 3 \omega_0^2 (I_1 - I_3)
\]
\[
T_3 = 3 \omega_0^2 (I_2 - I_1) a_{11} a_{21} \approx 3 \omega_0^2 (I_2 - I_1)
\]
Then the linearized equations of motion become

\[ I_1 (\ddot{\theta}_1 + \omega_0 \dot{\theta}_3) + (I_2 - I_3)(\omega_0^2 \theta_1 - \omega_0 \dot{\theta}_3) = 0 \]

\[ I_2 \ddot{\theta}_2 + 3\omega_0^2 (I_3 - I_1) \theta_2 = 0 \]

\[ I_3 (\ddot{\theta}_3 - \omega_0 \dot{\theta}_1) + (I_2 - I_1)(4\omega_0^2 \theta_3 + \omega_0 \dot{\theta}_1) = 0 \]

The pitch (\( \theta_2 \)) equation, which is uncoupled from the roll (\( \theta_3 \)) and yaw (\( \theta_1 \)) equations, is given as

\[ \ddot{\theta}_2 + \frac{3\omega_0^2 (I_3 - I_1)}{I_2} \theta_2 = 0 \]
or

$$\ddot{\theta}_2 + \omega_2^2 \theta_2 = 0$$

where

$$\omega_2^2 = \frac{3\omega_0^2 (I_3 - I_1)}{I_2} = 3\omega_0^2 k_p$$

The pitch resonance occurs at $k_p = 1/3$. 
The state equation of the coupled motion is

\[
\begin{pmatrix}
\dot{x}_1 \\
\dot{x}_2 \\
\dot{x}_3 \\
\dot{x}_4
\end{pmatrix}
= \begin{pmatrix}
0 & 1 & 0 & 0 \\
-k_y \omega_0^2 & 0 & 0 & (k_y - 1) \omega_0 \\
0 & 0 & 0 & 1 \\
0 & (1 - k_r) \omega_0 & -4k_r \omega_0^2 & 0
\end{pmatrix}
\begin{pmatrix}
x_1 \\
x_2 \\
x_3 \\
x_4
\end{pmatrix}
\]

where \( x_1 = \theta_1, \ x_2 = \dot{\theta}_1, \ x_3 = \theta_3, \) and \( x_4 = \dot{\theta}_4 \)
Show that

- the pitch axis must be the minor or major principal axis for roll-yaw stability.
- pitch stability requires that roll inertia be greater than yaw inertia.
- only two orientations are permissible, when $I_2 > I_3 > I_1$ or $I_3 > I_1 > I_2$.

Design a damper to reduce the oscillatory motion of gravity gradient stabilized system.
References