A SunCam online continuing education course

Spacecraft Subsystems Part 1 – Fundamentals of Attitude Control

by

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1. Introduction

Spacecraft are man-made machines which are capable of operating in space. An orbiting spacecraft is normally referred to as a satellite, although it is manmade (aka "artificial") as opposed to a natural satellite like our moon. ***Attitude*** (in aerospace engineering) generally means the orientation of a plane or a spacecraft with respect to a reference. For a plane, this reference is generally the earth's horizon. For a spacecraft, common references used are to itself (spacecraft's body), inertial space (with spacecraft at center), or its orbit. ***Control*** (in engineering) generally means the ability to maintain a desired system state. For example, a home heating system continuously measures your house temperature. When the temperature falls below the desired setting, the furnace is turned on (or other actuator), thus maintaining your desired temperature. The fundamentals of the home heating system also apply to spacecraft attitude control. Spacecraft attitude control refers to the ability of a spacecraft to maintain or change its orientation (attitude) to the desired position with respect to one or multiple references. One amazing spacecraft (also a satellite since orbiting the earth), the Hubble drifts over earth after its release on May 19, 2009 by the crew of the Space Shuttle Atlantis (see FIGURE 1.1).

Figure 1.1: Spacecraft Hubble
Final Release Over Earth (2009)
[Reprint from source: NASA/STScI, http://hubblesite.org/gallery/spacecraft/28/, 2010]

Spacecraft attitude is perturbed (changed) by either environmental or spacecraft disturbances; forces which can be measured in newton meters (N·m). Some environmental disturbances include aerodynamic, solar radiation, magnetic, and gravity gradient. Examples of disturbances caused by the spacecraft itself include vibration, propulsion thruster (used for orbital adjustments, not attitude), thermal snap (flexing), and mass distribution (e.g. sloshing of fuel, astronaut motion). The three common earth orbiting satellite types are geosynchronous orbit (GEO), highly elliptical orbit (HEO), and low earth orbit (LEO) (see FIGURE 1.2).

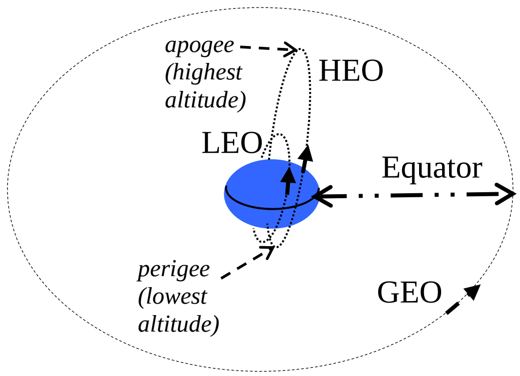


Figure 1.2: Common Earth Orbit Types
(no scale)

GEO satellites, which orbit about the equator at some small angle (inclination angle), have an altitude of ≈ 23,000 miles above earth. At this altitude, satellites have the same period of rotation as the earth, appearing fixed relative to earth (i.e. for fixed satellite dishes on earth, there is no need to change azimuth or elevation angles for tracking the satellite). Because of this, these satellites are most commonly used for communications purposes (e.g. television). Mission requirements generally dictate both the orbital type and the attitude pointing accuracy required for a spacecraft.

A spacecraft is typically subdivided into two major parts, the payload and the bus. Where the mission can be defined as the purpose of the spacecraft and is usually identified as the payload part of the spacecraft (e.g. scientific instruments, communications). The attitude control subsystem and other subsystems (e.g. thermal control) are part of the bus. With the primary goal of achieving a successful mission, most bus design constraints focus on maximizing the effectiveness of its payload. The attitude pointing accuracy required for a mission can be specified in degrees (within a range of degrees) or smaller (e.g. to the arc minute or arc second) where:

1 minute of arc (1′) = 1/60th of 1°
1 second of arc (1″) = 1/60th of 1′

Attitude control plays a critical role in spacecraft mission success. The following lists some of the spacecraft early flights (1958-2013) which encountered attitude problems (anomalies): Explorer I, Gemini VIII, Apollo 13, ANIK-B, ERBS, Polar BEAR, TDRS-1, NOAA-10, HST, Anik E-2, Clementine, Lewis, WIRE, Mars Climate Orbiter, FUSE, IMAGE, Landsat 5, GOES-13, and Kepler. In the final section, after we explore the fundamentals of attitude control, we will take a closer look at these spacecraft anomalies. In order to better understand determination and control methods, it is important to know the basic concepts of spacecraft attitude *Reference Frames*, *Representations*, and *Dynamics*.

Reference Frames

In order to determine spacecraft attitude, we need to first define its reference frame (i.e. its attitude compared to what?). A reference frame (aka coordinate system), is generally taken to be a set of three unit vectors that are mutually perpendicular (orthogonal or 90º) to one another. These vectors are unit vectors; which mean their lengths are unity (one). A reference frame (F_x) can be expressed using a triad of unit vectors; where letter x is replaced with the reference frame in use (e.g. o-orbital, b-body, i-inertial) and subscripts 1, 2, and 3 represent each unit vector; where $F_o = \{o_1, o_2, o_3\}$, $F_b = \{b_1, b_2, b_3\}$, and $F_i = \{i_1, i_2, i_3\}$ – the three common reference frames (see FIGURE 1.3).

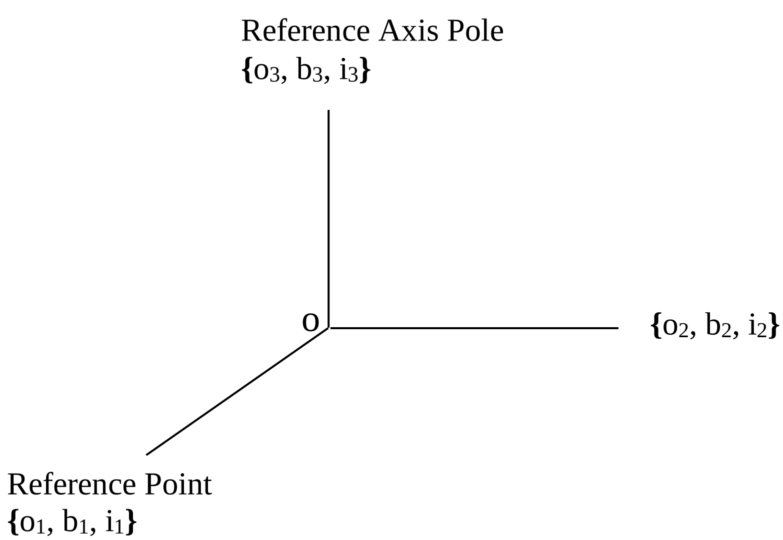


Figure 1.3: Reference Frames

The following table summarizes and further defines these three reference frames. For example, the orbital reference frame, has the coordinate system name of roll/pitch/yaw, is fixed with respect to its orbit, with its reference axis pole (yaw, o_3) nadir pointing towards earth, and has its reference point towards its velocity vector (roll, o_1).

Table 1.1: Reference Frames

Reference Frame	**Coordinate System Name**	**Fixed w/respect to**	**Reference Axis Pole** {o_3, b_3, i_3}	**Reference Point** {o_1, b_1, i_1}
orbital	roll/pitch/yaw* (aka RPY)	orbit	yaw axis; nadir pointing (i.e. bottom of spacecraft to earth, with zenith or top to space)	roll axis toward velocity vector
body	spacecraft fixed	spacecraft	spacecraft axis toward nadir	spacecraft axis toward velocity vector
inertial	celestial (spacecraft-centered)	inertial space	North celestial pole (1 degree from the bright star Polaris (aka North star)	vernal equinox**

* Most similar to the reference frame used by an airplane.
** The vernal equinox is defined as the reference point on the celestial sphere where the plane of the earth's orbit around the sun crosses the equator from south to north.

Representations

An attitude representation of a spacecraft is defined as its orientation with respect to a reference frame. The attitude of a spacecraft can be represented in three-axis or single-axis. Three-axis attitude of a spacecraft may best be visualized; where we take the orbital reference frame derived from FIGURE 1.3 and add a spacecraft with axes u, v, w (see FIGURE 1.4). With each axis of the spacecraft slightly skewed (i.e. small attitude error), it should be clearer that its three-axis attitude is defined as the orientation of the spacecraft axes u, v, w with respect to the orbital reference frame $F_o = \{o_1, o_2, o_3\}$.

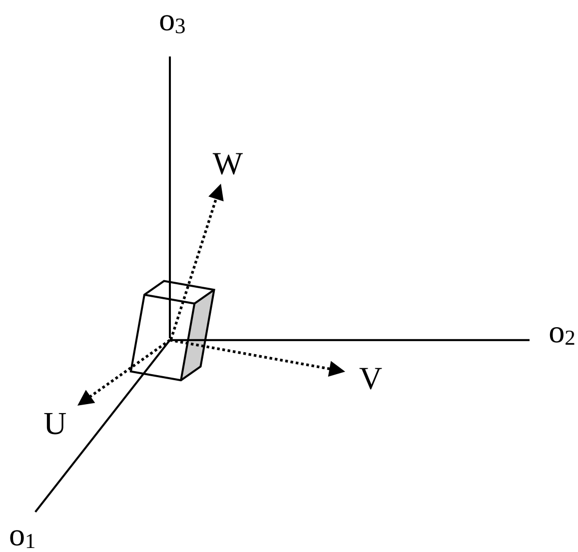


Figure 1.4: Three-Axis Attitude

A fundamental way to describe (parameterize) this attitude is using a Direction Cosine Matrix (DCM). The general representation of this in any reference frame is:

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

Where A is a 3 x 3 attitude matrix (aka rotation or transformation matrix) and subscripts 1, 2, 3 are the reference frame axes (e.g. o_1, o_2, o_3 respectively). Each element in the matrix is the cosine of the angle between a spacecraft unit vector and a reference axis (e.g. parameter u_1 is the cosine of the angle between vector u and reference axis o_1); and for a "perfect" attitude (i.e. no errors), the DCM is:

$$A = \begin{bmatrix} 1 & 0 & 0 \\ 0 & 1 & 0 \\ 0 & 0 & 1 \end{bmatrix}$$

...where the cosine of 0° is 1 and the cosine of 90° is 0

The following table summarizes the DCM, and defines three other approaches to quantify three-axis spacecraft attitude. This is generally the reason why attitude control is considered the most complicated of all the spacecraft subsystems; all four parameterizations can be used to describe the attitude depending on the application – and not to mention, you also need to know which reference frame you are using.

Table 1.2: Three-Axis Attitude Representations

Parameterization	Common Notation	Some Applications
Direction Cosine Matrix	$\begin{bmatrix} \alpha_{11} & \alpha_{12} & \alpha_{13} \\ \alpha_{21} & \alpha_{22} & \alpha_{23} \\ \alpha_{31} & \alpha_{32} & \alpha_{33} \end{bmatrix}$ *Note: This is identical to the previous matrix A; same quantities, just different notation used.*	Analysis (e.g. to transform vectors from one reference frame to another)
Euler Angles (RPY) Where Euler is pronounced "Oiler"; and the angles are defined by rotations about three orthogonal axes (e.g. o_1, o_2, o_3)	θ, ψ, φ	• Analytic studies • Input/Output • Onboard attitude control
Euler Angle/Axis Defined by Euler's Theorem: *The most general displacement of a rigid body with one point fixed is a rotation about some axis.* Note: For this course, rigid body = spacecraft	Φ $\begin{bmatrix} a_1 \\ a_2 \\ a_3 \end{bmatrix}$	Commanding maneuvers
Quaternion This is a parameterization of the DCM in terms of Euler symmetric parameters (i.e. 9 values reduced to 4)	$\begin{bmatrix} q_1 \\ q_2 \\ q_3 \\ q_4 \end{bmatrix}$	Onboard inertial navigation

If we take the inertial reference frame, derived from FIGURE 1.3 and add a celestial sphere with a spacecraft at the center; a single axis (Z) of a spacecraft can be projected to the surface of the spacecraft centered celestial sphere as shown by ***** (see FIGURE 1.5). It is from this point (*) we can determine the attitude parameters – declination (δ) and right ascension (α) angles.
The declination (elevation) is measured from the celestial equator 0 to +90º (North celestial pole) and 0 to –90º (South celestial pole). The right ascension is measured from the vernal equinox 0º to 360º going CCW (counter-clock wise) along the celestial equator back to the vernal equinox. For our example in the figure below:

Step 1) For declination – project from the intersection point (*) on the celestial sphere to the celestial equator, and mark the intersection with an **X**; so our $\delta \approx 75°$.

Step 2) For right ascension – project from the vernal equinox to the intersection point **X** on the celestial equator (from step 1) going CCW (counter-clock wise) along the celestial equator; so our $\alpha \approx 45°$.

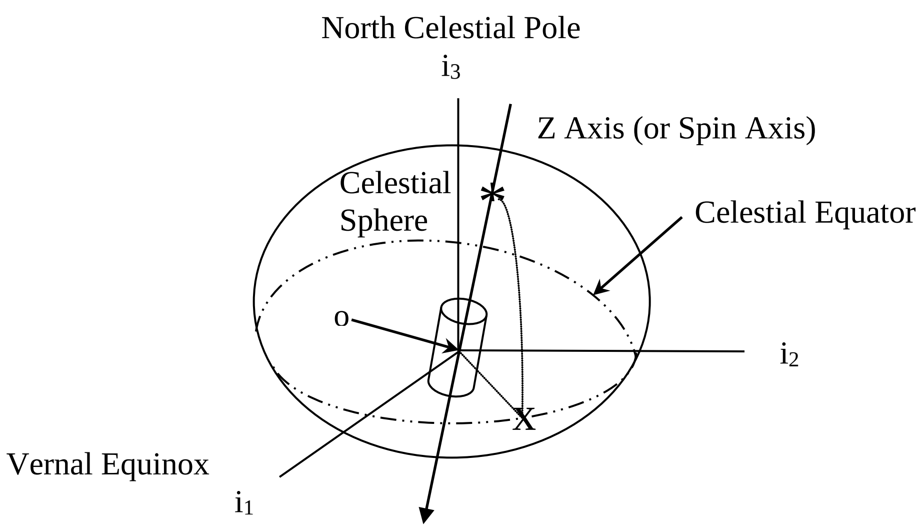


Figure 1.5: Single-Axis Attitude
Right Ascension/Declination

We can also represent single-axis attitude with a unit vector from the origin (o) along the Z axis to the point (*****) on the celestial sphere; defined by three direction cosines measured from the vector to each reference axis $\{i_1, i_2, i_3\}$.

Dynamics

Thus far, our attitude representations have been fixed in time (a snapshot). We now introduce the variable of time and how to model the attitude with respect to time. Dynamics is the study of moving objects, which is subdivided into kinematics and kinetics (see FIGURE 1.6). For translational dynamics, force affects velocity (kinetics) and velocity affects position (kinematics). For rotational dynamics, moment of force affects angular velocity (kinetics) and angular velocity affects orientation (kinematics). Kinematics is the study of a body's motion independent of the forces that bring about that motion. Whereas kinetics is the study of motion and the forces that cause motion.

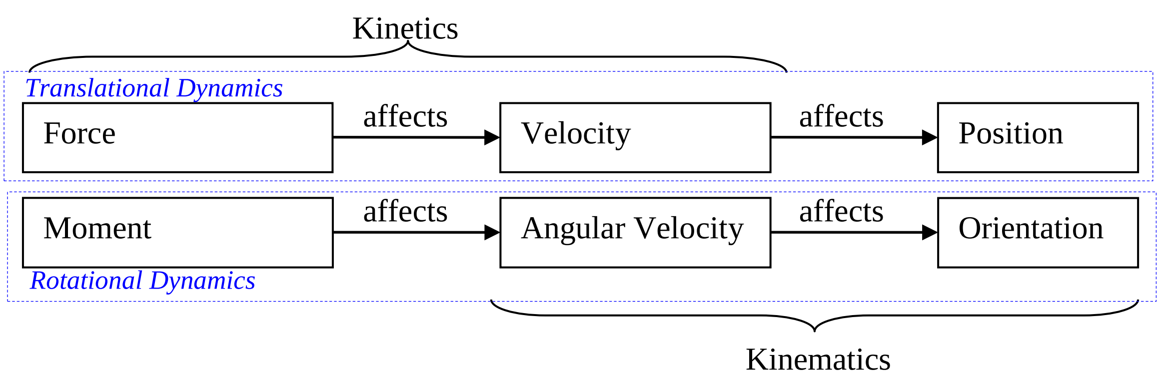


Figure 1.6: Kinetics vs. Kinematics

Generally speaking, the motion of a spacecraft can be analogous to the motion of a ball kicked or thrown; if the ball has no rotation – it has translational dynamics; however, if the ball is also rotating – it has both translational and rotational dynamics. For spacecraft attitude, we are primarily concerned with rotational dynamics – angular velocity and angular momentum.

<u>Angular Velocity (aka Rate)</u> is the first key component we need to characterize the spacecraft dynamics. In order to quantify the rotational motion of a spacecraft, we need to know its angular velocity. The average angular velocity (ω), of a rotating spacecraft is the ratio of the angular displacement ($d\theta$) to the time interval (dt) it takes the object to rotate through the angle $d\theta$, or $\boldsymbol{\omega = d\theta/dt}$; where units of measure are typically in radians or degrees per second (rad/s or deg/s). Revolutions per minute (rpm) can also be used to represent very high angular velocities.

The moment of inertia (I) quantifies the rotational inertia of a spacecraft in metric units of kilogram square meters ($kg \cdot m^2$); this describes how difficult it is to induce an angular rotation about a given axis. For a spacecraft, the following equation summarizes all moments of inertia of an object with one quantity:

$$I = \begin{bmatrix} I_{11} & I_{12} & I_{13} \\ I_{21} & I_{22} & I_{23} \\ I_{31} & I_{32} & I_{33} \end{bmatrix}$$

The previous form of I can be difficult to interpret and visualize; therefore, we can find the principle moment of inertia (I_{prin}) to reduce our analysis to three moments of inertia as represented by the following equation:

$$I_{prin} = \begin{bmatrix} I_1 & 0 & 0 \\ 0 & I_2 & 0 \\ 0 & 0 & I_3 \end{bmatrix}$$

The scalars I_1, I_2, and I_3 are the principal moments of inertia which represent the moments with respect to the principal axes. The principal axes are the three axes about which the spacecraft mass is symmetrically distributed. We can use this simplified approach to describe the moment of inertia for a spacecraft.

Angular Momentum is the other key component we need in order to characterize the spacecraft dynamics. The angular momentum (L) of a rotating spacecraft is the product of the moment of inertia (I) and the angular velocity (ω) or **L = I ω**. The angular momentum (L) is the measure of the extent to which an object will continue to rotate about some fixed reference point unless acted upon by an external force (torque); where the units of measure can be in the equivalent metric units of joule seconds (J·s), newton meter seconds (N·m·s), or kilogram square meters per second ($kg \cdot m^2/s$).

2. Determination Methods

For attitude control, a spacecraft requires two measurement states for input to its controller – angular positions and angular rates (see FIGURE 2.1). Onboard measurements from spacecraft sensors make attitude determination possible. This information can be determined using one or more methods. Earth horizon sensors provide pitch and roll angular positions. Rate and rate integrating (RI) sensors can provide angular velocities (rates) and angular positions respectively. Sun, star, and earth magnetic sensors provide reference vectors in order to determine angular positions.

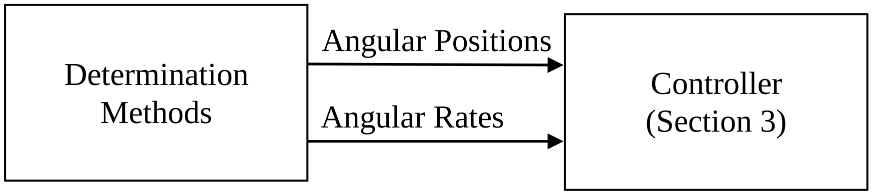


Figure 2.1: Determination Method Outputs

A few key characteristics of the sun have made sun sensors a very popular device used in spacecraft attitude control subsystems since the beginning of space exploration. The first key characteristic of the sun is that its magnitude (brightness) remains unaffected by the brightness of any other planet or star. The second important characteristic is that its small angular radius of 0.267° at 1AU (Astronomical Unit) as viewed from the earth is nearly constant for earth orbiting satellites; this allows them to view the sun as a point source (see FIGURE 2.2). One limitation of the sun sensor is that its reference source, the sun, is not always available (i.e. in view for the entire satellite orbital path).

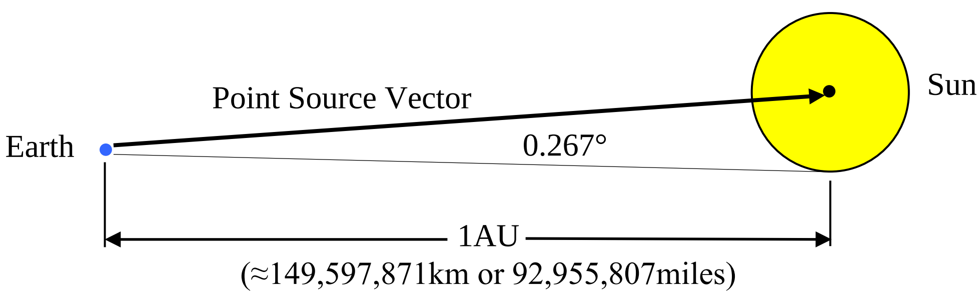


Figure 2.2: Sun as Point Source

Stars are so far away that a figure such as previous FIGURE 2.2 would be of no value. Star distances from earth are measured in light years (ly) – where a unit of length of 1 light year is just under ten trillion kilometers. With these extreme distances, stars can be characterized as fixed bodies in inertial space making them a true point source reference with virtually no (0.00°) angular radius as viewed from our solar system; thus allowing for very high accuracy sensing in the arc second range or better. Star sensors locate and track (follow) stars by using the star's apparent magnitude (brightness) and spectra (wavelength); where apparent magnitude (m_v) is how bright the stars appear to us. It is important to note that this magnitude scale is not linear, it is logarithmic; where a change of 5 magnitudes is defined as a change of exactly 100 times in brightness. The magnitudes in the visible spectra (m_v) of the ten brightest stars are summarized in the following table in order from the brightest to the dimmest.

Table 2.1: Ten Brightest Stars (*Bold Italics*)

Star/Object	**Magnitude (m_v)**	**Distance from Earth (Approximate)**
Sun[1]	-26.8	150 million km
Sirius	-1.49	8.6 ly
Canopus	-0.72	74 ly
Rigil Kentaurus	-0.27	4.3 ly
Arcturus	-0.04	34 ly
Vega	0.03	25 ly
Capella	0.08	41 ly
Rigel	0.12	≈ 1400 ly
Procyon	0.38	11.4 ly
Achernar	0.46	69 ly
Betelgeuse	0.50	≈ 1400 ly
Polaris (aka North star)[2]	2 (±0.05)	430 ly
star x[3]	6.5	varies ly
1. Shown here to provide scale (comparison) – for reference only. Note: The sun is not a star in this context (i.e. star sensors do not seek out and track the sun).		
2. The North star is used for navigation on earth and is not among the brightest; shown here to provide scale – for reference only.		
3. X can be any star with a magnitude of about 6.5 – which is about the weakest (dimmest) magnitude which is still visible to the human eye. Added to provide scale – for reference only.		

To put light years in perspective, if you were to find the brightest star Sirius in the night sky, the light you are seeing was originated by the star 8.6 years ago. One light year is the distance light travels in one year (or just under ten trillion kilometers as previously mentioned). Star sensors have the capability to track one or more stars simultaneously. With this capability, these sensors alone can be used in the vector algorithms presented in this section to determine the spacecraft's attitude.

Like sun sensors, earth magnetic (aka magnetometers) have also been a popular space flight device. These devices are commonly known as a three-axis magnetometer (TAM) because they consist of three orthogonal magnets. The earth's magnetic field strength becomes weaker with altitude; thus, limiting this sensors effectiveness for earth orbiting satellites to an altitude of about 1000km or lower. A primary advantage of these sensors is that the earth's magnetic field is always present; therefore, magnetometers provide continuous measurement capabilities. In contrast, the sun and star may not always be in the spacecraft's field of view (FOV).

Vector Algorithms

The vector algorithms discussed in this section require two references vectors as input. Using previous FIGURE 1.4, we add two reference vectors R_1 and R_2 (see FIGURE 2.3).

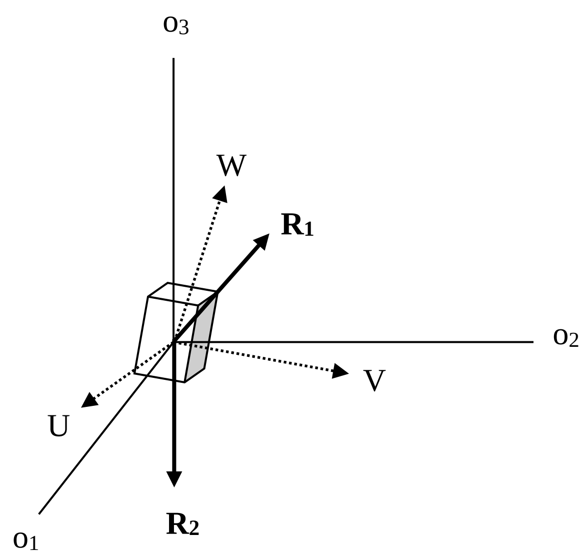


Figure 2.3: Reference Vectors

The sun and star sensors measure two angles which are relative to the z-axis called the optical axis (see FIGURE 2.4). Angle α is projected onto the X_s-Z_s plane and angle β is projected onto the Y_s-Z_s plane. Together, these sensor angle measurements provide the sun or star vector (s) in the sensor's coordinate frame.

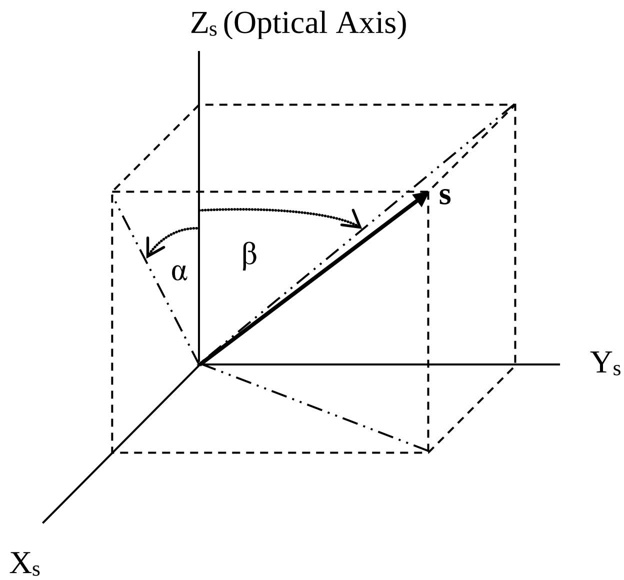


Figure 2.4: Sensor Coordinate System – Sun or Star

Similarly, earth magnetic sensors measure angular components of the earth magnetic field vector (m) and can be conceptualized in the form of direction cosines (see FIGURE 2.5). Where each reference axis (X_s, Y_s, Z_s) is aligned with and represents one magnet in a three-axis magnetometer configuration. Each magnet measures the earth magnetic field vector position relative to itself. Together, these measured angles (α, β, γ) provide the earth magnetic field vector in the sensor's coordinate frame. Compared to the sun and star sensors, there is no FOV constraint (optical blockage); therefore, this sensor type can continuously measure the earth's magnetic field vector in any direction about the origin.

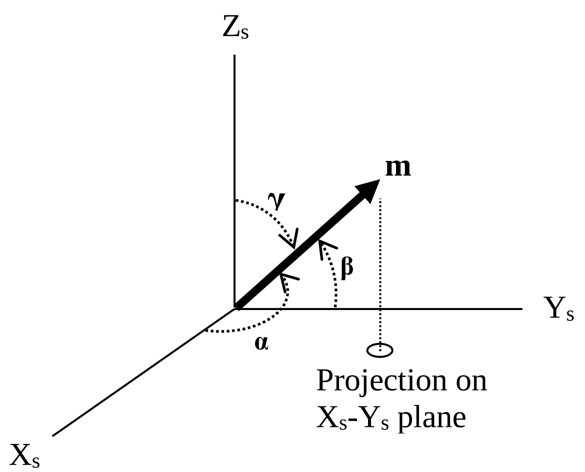


Figure 2.5: Sensor Coordinate System – Earth Magnetic

In order to use the sensor measurements for the vector algorithms, we need the reference vector measurements with respect to the spacecraft body frame (r_b) – where $\mathbf{r_b} = \mathbf{R^{bs}}\ \mathbf{r_s}$. The sensors will measure the reference vector with respected to their own frame (r_s); therefore, you need to use the rotation matrix R^{bs} to convert the sensor measurement to the spacecraft body frame. The value of R^{bs} depends on the spacecraft designers and will be a function of the sensor location and the sensor orientation with respect to the spacecraft body.

Once we have at least two reference vectors (r) measured in the body frame of reference (r_b), their theoretical values need to be computed by mathematical models in the inertial frame of reference (r_i) (see FIGURE 2.6). These vectors are then inputs to an estimation algorithm (triad, q, or QUEST) from which its output represents the spacecraft attitude at the measured time.

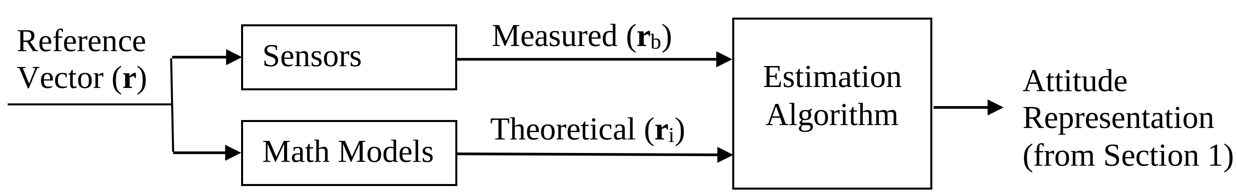


Figure 2.6: Attitude Estimation Algorithm

One of the first and most basic estimation algorithms is the triad. The triad algorithm uses the minimum set of data available (aka deterministic) to estimate the attitude; so is therefore an attitude approximation. In contrast, the q estimation algorithm is an exact statistical algorithm which uses all information available (aka optimal) from two or more reference observations. Another accurate and more efficient algorithm is the QUEST (QUaternion ESTimator). Where the q algorithm requires the calculations of all (four) attitude parameter sets to determine the largest of the four and is therefore numerically intensive; the QUEST algorithm approximates the largest of the four (i.e. does not compute the other three) and is therefore an efficient approximation of the q algorithm.

Sensor Methods

Some sensors have the unique ability to determine attitude quantities. The earth horizon sensor can determine the attitude pitch and roll parameters. Rate sensors can determine attitude velocities (rates) and rate integrating (RI) sensors can determine attitude positions.

There are two types of earth horizon sensors – static and scanning. Both sensor types detect the infrared (heat) spectrum of the earth – not visible to us. The sensor does not "see" the visible light spectrum as we would as seen in FIGURE 1.1. The sensor utilizes a key characteristic of the intensity profile in the infrared spectrum. In this infrared spectrum, in particular 14 to 16μm, the earth horizon is defined in a homogeneous profile where the intensity is more uniform; hence, simplifying the ability to detect the earth's horizon as "seen" by a satellite. This characteristic of the earth's infrared spectrum, along with its continuous presence (i.e. it exists day and night), make these sensors an ideal attitude determination method for earth orbiting satellites. The key difference between horizon sensors and other sensor types, is the ability of the scanning type horizon sensors to determine the pitch and roll independently, without the need for another type of sensor. The static sensor type determines the earth geocenter; and therefore the spacecraft nadir (lowest point or bottom). With the static type, the sensor can be used as a conventional reference sensor (e.g. sun) to provide a vector to the earth's geocenter relative to the spacecraft nadir.

As previously discussed, the scanning type horizon sensor has the unique ability to determine the pitch and roll (Euler) attitude angles independent of any other sensor or algorithm. This sensor scans (illuminates) a cone on the earth's horizon (see FIGURE 2.7); where horizon crossing 1 (HC1) is the LOS (loss of signal) or into the earth and horizon crossing 2 (HC2) is the AOS (arrival of signal) or out of the earth. These crossings are used to determine the phase angles (δ_1 and δ_2) with respect to the vertical reference. A roll will cause the vertical reference to slide (move) up or down the vertical – thus changing the magnitude of both phase angles. A pitch will rotate the vertical right or left – thus causing the magnitudes of the phase angles to be different.

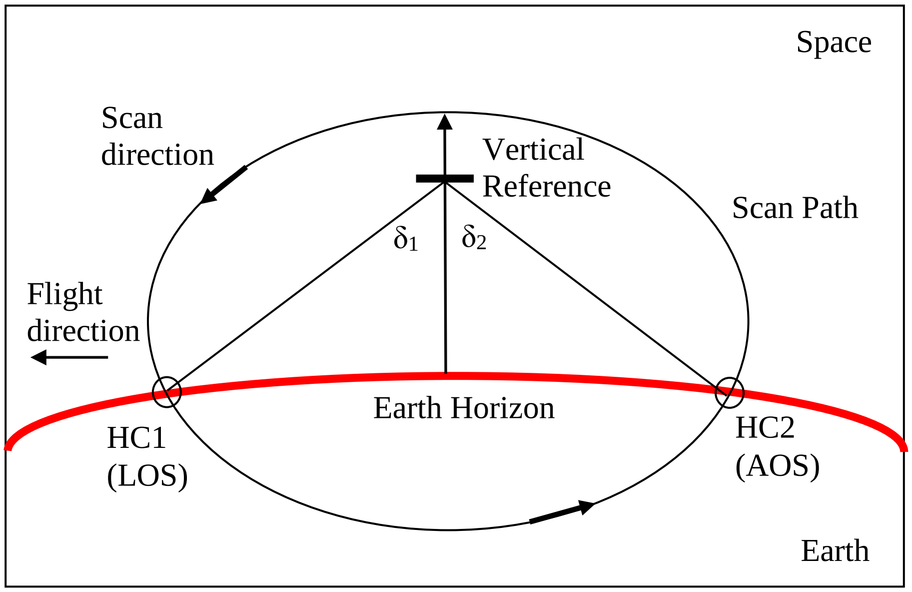


Figure 2.7: Earth Scanning Horizon Sensor
(right side view as seen by satellite)

We can find the roll (Euler) angle using the following equation: $\boldsymbol{\theta = [(\delta_2 - \delta_1) / C_1] - E_o}$ ...where $\delta_1 - \delta_2$ is the phase difference (E), E_o is a function of the altitude and is the normalizing variable necessary to determine roll from the phase difference, and C_1 is a constant. The pitch (Euler) angle can also be found in a similar manner using the following equation: $\boldsymbol{\psi = [-(\delta_2 + \delta_1) / 2C_2] + [(90° + C_3) / C_2] + C_r\,\theta}$...where C_2 C_3 and C_r are constants.

Rate and rate integrating sensors (aka gyroscopes) use mechanical, optical, or electrical principles to measure angular velocities (rates) and angular positions (rate integrating). These angular components can be directly equated to Euler angles. Unlike the other sensors, rate and rate integrating sensors have no object to reference (e.g. sun or star); therefore, they must be calibrated with an initial attitude of 0° relative to their own (sensor) reference frame. Subsequently, angular rotations will be proportional to the spacecraft angular rotations and directly translate into attitude angular position and velocity error data. These angular rates can also be measured using other sensors (e.g. sun and star) and algorithms using the time change of the angular positions; however, these are not always available for use as previously mentioned. Consequently, the capability to provide continuous measurements is a major advantage of rate and rate integrating sensors (i.e. do not have to rely on the sun or stars to be in the spacecraft FOV).

Rate sensor operation may be best visualized using Euler angles ─ roll(θ), pitch(ψ), and yaw(φ); where changes in the clockwise (CW) direction are positive (+) and changes in the counter clockwise (CCW) direction are negative (-). We can characterize the output of rate sensors using the following equation: **Angular Velocity (Rate) =** $\boldsymbol{\Delta_{(\theta\psi\varphi)} / \Delta t}$ **...**where $\Delta_{(\theta\psi\varphi)}$ is the change in angular position (roll, pitch, or yaw) and Δt is the change in time in seconds (s). We can then find the attitude using the following equation: $\mathbf{Attitude}_{\Delta t(\theta\psi\varphi)} = \mathbf{Attitude}_{i(\theta\psi\varphi)} + [\Delta_{(\theta\psi\varphi)}]$ ...where $\text{Attitude}_{i(\theta\psi\varphi)}$ is our initial attitude (roll, pitch, or yaw) and $\text{Attitude}_{\Delta t(\theta\psi\varphi)}$ is our new attitude (roll, pitch, or yaw) after time of Δt.

Rate integrating sensors provide an output of change in the Euler angular positions ($\Delta_{(\theta\psi\varphi)}$); therefore, are "rate integrating", since the integral of velocity is position. We can characterize the output of these sensors using the following equation: $\boldsymbol{\Delta_{(\theta\psi\varphi)} = (\text{Rate})\,(\Delta t)}$.

Kalman Filter

The Kalman filter is an algorithm which contains a set of mathematical equations that provides an efficient computational recursive estimation of the state of a process; where the state (for our application) refers to the estimated attitude solution of a spacecraft. This algorithm executes a continuous series of predictions and corrections; therefore, the operation of a Kalman filter algorithm can be described as a form of feedback with two distinct stages – prediction and correction (see FIGURE 2.8).

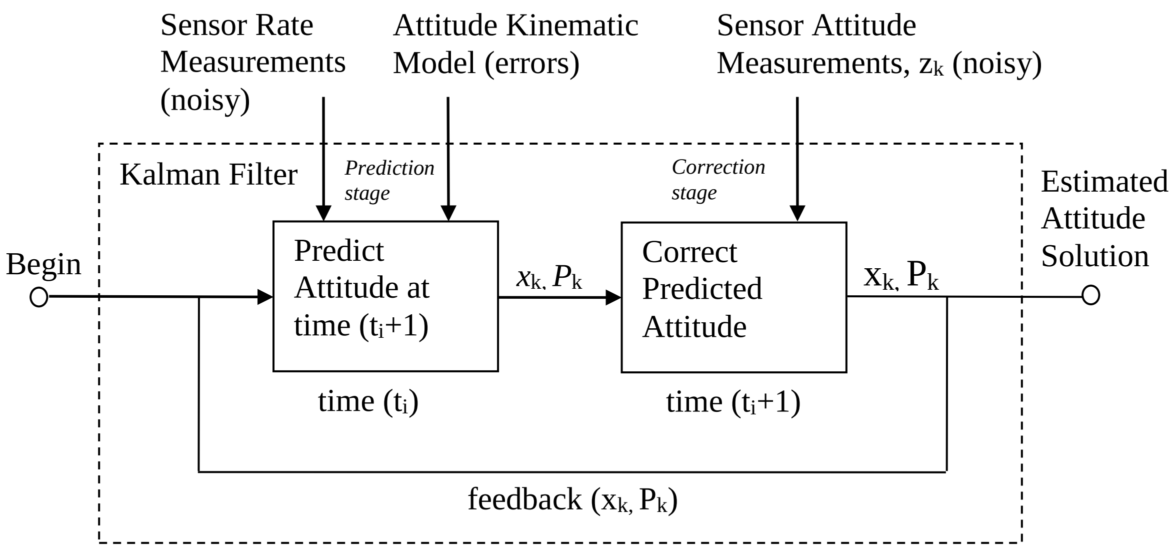


Figure 2.8: Kalman Filter

The prediction stage predicts the state and error at time (t_i+1) using the sensor rate measurements (typically gyros) and the attitude kinematic mathematical model as stage inputs. These two inputs contain measurement noise and modeling errors respectively, together called process noise. This stage also uses feedback from the correction stage as inputs. The prediction stage executes the following primary algorithmic steps at time (t_i):

(1) Predict the state ahead for time (t_i+1); output is x_k.
(2) Predict the error covariance ahead for time (t_i+1); output is P_k.

Outputs x_k and P_k are then inputs to the Correction stage.

The correction stage uses the outputs from the prediction stage along with noisy sensor attitude measurements as inputs to correct the predictions resulting in a more optimal estimated attitude solution. The correction stage executes the following primary algorithmic steps at time (t_i+1):

(1) Compute the Kalman gain, K_k.
(2) Correct the predicted state x_k using measurement z_k and K_k; output is x_k.
(3) Correct the predicted error covariance P_k using K_k; output is P_k.

Outputs x_k and P_k are fed back to the prediction stage and also represent your estimated attitude solution at time (t_i+1).

The Kalman filter statistically combines the attitude kinematic model with the sensor attitude measurements while filtering out noise to provide a more accurate estimated attitude solution than individual modeling or sensor measurements could provide alone.

Single-Axis Measurements

Estimating the attitude of a single axis allows you to determine the attitude of the spin axis for a body spin stabilized spacecraft (section 3) or for any axis of a three-axis stabilized spacecraft. To demonstrate, we start with the single-axis attitude from FIGURE 1.5; add attitude vector (A) and two reference vectors (E and S). The two known reference vectors with their origin at the spacecraft's center are required to determine the spacecraft attitude vector (see FIGURE 2.9). From this, a spherical triangle (EAS) can be carved out on our spacecraft-centered celestial sphere with arc lengths: η (A-E), ψ (E-S), β (A-S); and rotation angle Φ (EAS).
We introduce the following two basic methods which may be used for single-axis attitude determination:

- Arc-Length measurements
- Arc-Length and Rotation Angle measurements

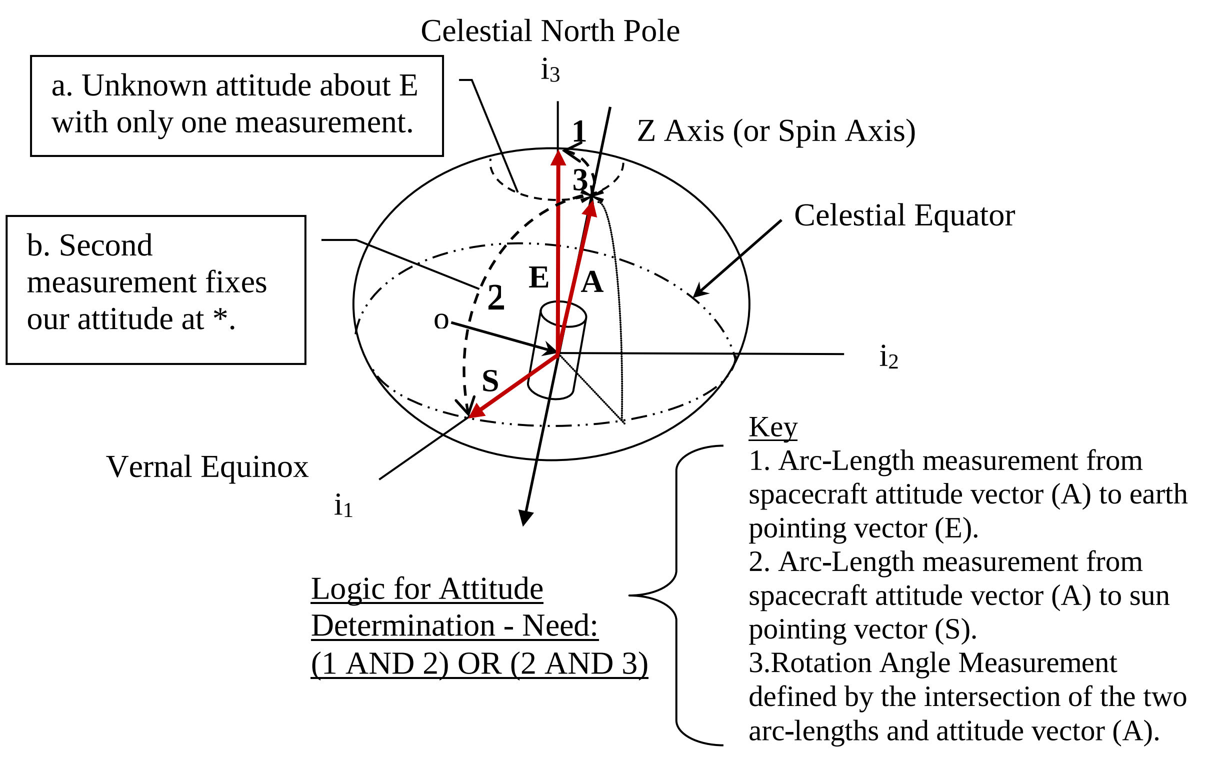


Figure 2.9: Single-Axis Attitude Measurements

This first method for determining the attitude of a single spacecraft axis requires two arc-length measurements – one from the spacecraft attitude vector of interest to each reference vector. For simplicity, our known sun (S) and earth (E) vectors are aligned along the i_1 and i_3 inertial axis respectively. Why a minimum of two measurements are required? If we only have one arc-length measurement (e.g. η, A-E); our attitude vector (A) can lie at any point along the circle (see a.) about the earth vector (E). If we introduce our second arc-length measurement to sun vector (S), we can now stop (or fix) the attitude vector (A) uncertainty about (E) at the point on the spacecraft celestial sphere (*) equal to our second arc-length measurement (see b.).

For our second method, the arc length β (A-S) and the rotation angle Φ (EAS) values are measured. The rotation angle measurement is then used to find the unknown arc-length η (A-E); hence, we have our second arc length needed to fix the attitude vector (A) using the same concept as in the first method.

3. Control Methods

We are now ready to work our final piece of the puzzle – *spacecraft attitude* __________? If we know a spacecraft has a bad attitude (pun intended), what can we do about it? This final piece, *control*, generally refers to the desire to impact a system in a good way; as opposed to "uncontrolled" or a bad way. For example, if your home thermostat is broken during winter time; the temperature will drop "uncontrolled" to a colder temperature than you may desire. To prevent this, a home heating control system is needed to continuously measure temperature and apply (add) heat to maintain the desired temperature (see FIGURE 3.1).

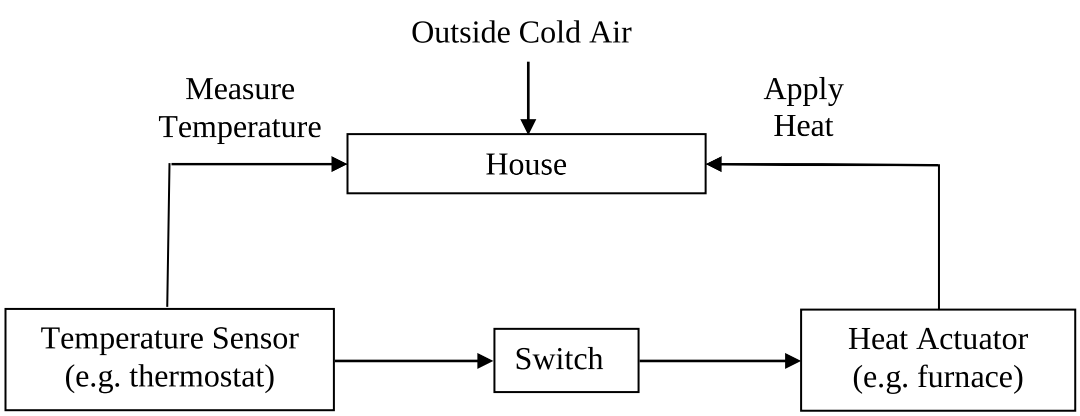


Figure 3.1: Home Heating Control System Functional Diagram

This basic control concept can also be applied to the attitude control subsystem (ACS), where not all sensors or actuators may be present on any one spacecraft (see FIGURE 3.2); where those chosen are largely based on mission pointing requirements for accuracy. For example, thrusters and reaction wheels can achieve high attitude accuracy (e.g. arc second); whereas, momentum wheels and gravity gradient boom and mass would provide a low attitude accuracy capability (e.g. degrees).

Recall that attitude error measurements can be provided from sun, star, earth magnetic, earth horizon, and rate or RI sensors. The control logic or attitude control electronics (ACE) includes the hardware and software which compute the attitude angular positions and velocities (if needed) from sensor inputs and output the control command(s) necessary for attitude control of the spacecraft. Control concepts can be characterized as active or passive.

For passive control, no control response computation is necessary from attitude sensor inputs; instead, "free" energy (natural forces) are utilized. Passive control methods include using a gravity gradient boom with an end mass or the spin dynamics of the spacecraft body. In contrast, active control requires control torque commands based on sensor outputs; therefore, electrical power or fuel is required for the actuator. Active actuation (i.e. the control torque) can be applied by thrusters, reaction wheels, momentum wheels, magnetic torque rods, or control moment gyros. A spacecraft mission may require attitude control for either stabilization, maneuvers, or both.

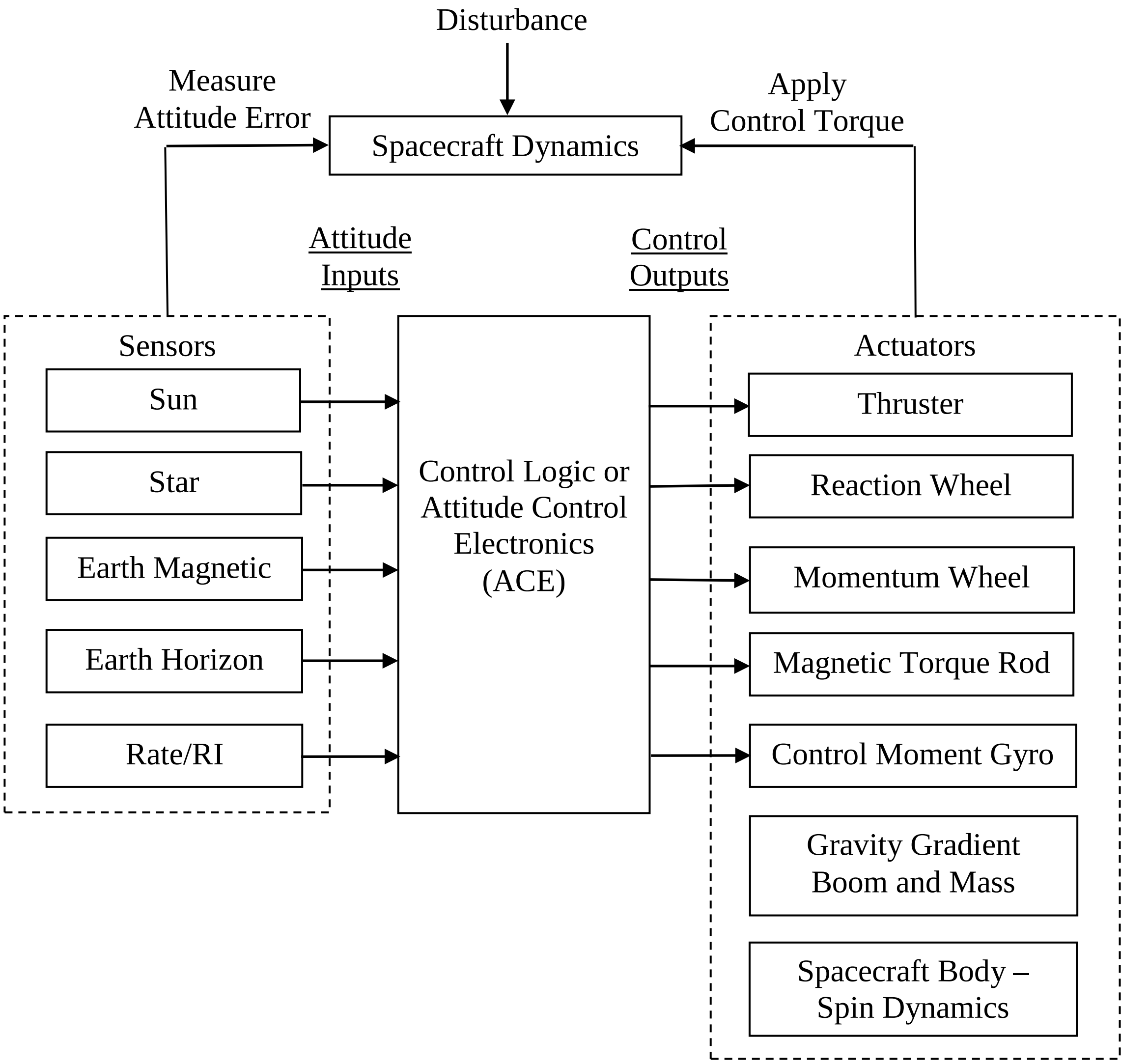


Figure 3.2: Attitude Control Subsystem (ACS) Functional Diagram

Stabilization refers to the spacecraft's ability to maintain a specific attitude; and generally stability refers to a system's response to a near equilibrium state. Equilibrium refers to the combination of input and state; where the following two concepts apply:

- Lagrange or "infinitesimal" – If a small deviation (change) from some equilibrium point remains bounded (within some range); then the motion is said to be Lagrange (or infinitesimally stable).

- Liapunov – requires that a system which begins close to equilibrium (origin) must remain close to itself after a perturbation (disturbance).

A maneuver refers to the ability of a spacecraft to change from one attitude to another attitude. For most spacecraft, maneuvers are necessary for initial attitude acquisition at the beginning of a mission. For some spacecraft, maneuvers are also required for payload repointing during the mission lifetime.

Before getting into the spacecraft attitude control methods used, we first use the mass-spring-damper to help further explain the control concepts (see FIGURE 3.3). This offers a good example of translational dynamics; where object (m) represents the spacecraft, spring (k) represents the actuator, damper (b) represents the system's ability to return to its original (desired) position (x_d) in the fastest time without overshoot, and the forcing function (f(t)) can represent the spacecraft disturbance forces.

Variables:
k – spring stiffness
m – mass of object
b – viscous damper coefficient
x(t) – position of object at time (t)
f(t) – forcing function at time (t)

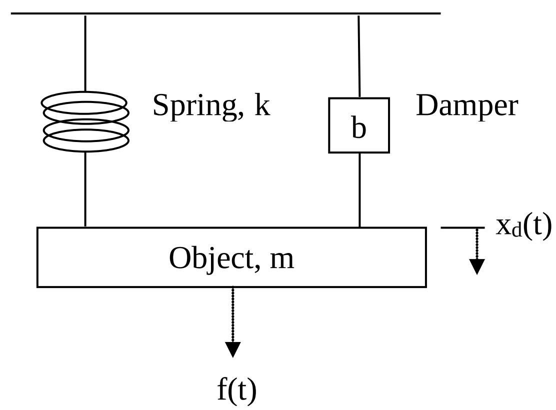
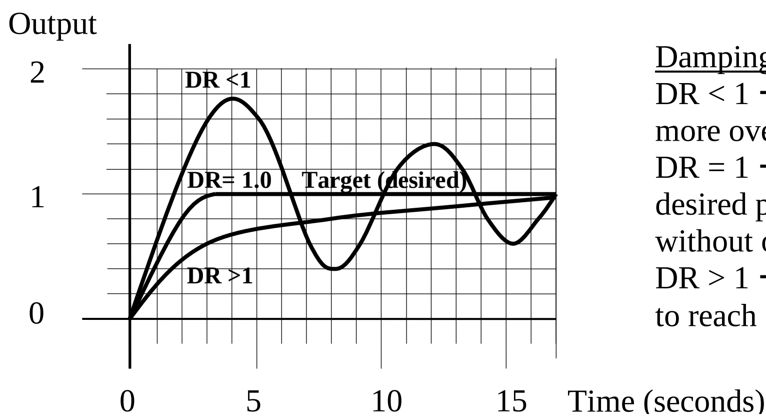


Figure 3.3: The Mass-Spring-Damper
Example: Translational Dynamics

The following equation of motion applies: **f(t) = ma + bv + kx**...where variable "a" is acceleration, "v" is velocity, and the damping ratio (DR) equation is: **DR = b/2√km**. It can be seen from the previous f(t) equation that the value of "b" affects the velocity term; therefore, the damper is used to control rate (velocity). Without this damper, the object (mass) could "bounce" uncontrolled. However, this damper can be used to slow motion to maintain (stabilize) the position of mass – depending on the value of "b". The damping ratio (DR) can be further explained using a graph; comparing the three states of the damping ratio function: under damped, critically damped, and over damped responses vs. time (see FIGURE 3.4).



Damping Ratios
DR < 1 → Under damped (faster), more overshoot beyond desired
DR = 1 → Critically damped, desired position in fastest time without overshoot
DR > 1 → Over damped, too slow to reach desired

Figure 3.4: Damping Ratios

The mass-spring-damper can also be represented (modeled) as a closed loop control system (see FIGURE 3.5). Actual position $x_a(t)$ gets fed back to the input; which means that the spring and damper responses are affected by a change in position (i.e. the difference between the desired and actual positions). This type of feedback is called unity feedback; where the feedback factor (multiplier) is one, since there is no component in the feedback loop. Hence, the feedback input to the controller is: $\mathbf{X_a(t) = (1)\, X_a(t)}$ or just $X_a(t)$.

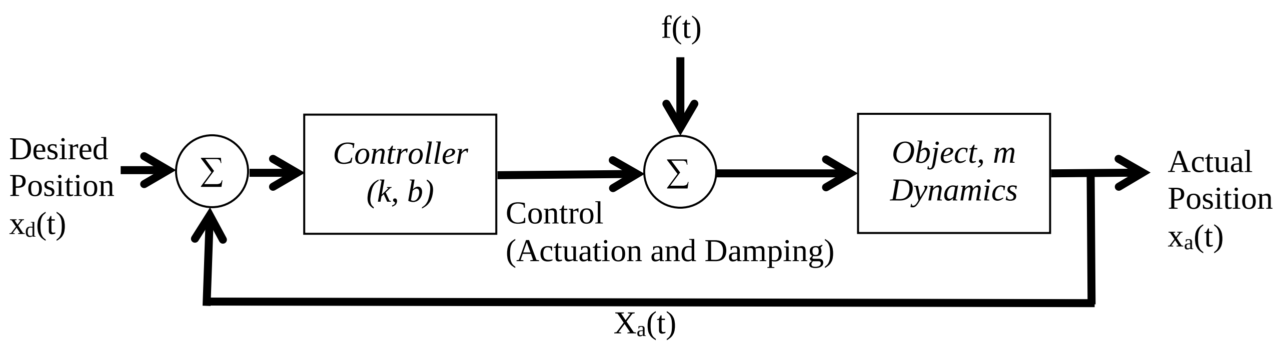


Figure 3.5: Mass-Spring-Damper Control System

The mass-spring-damper control concept can also be applied to spacecraft attitude control subsystems (ACS). Using the ACS components as shown in previous FIGURE 3.2, an onboard ACS is implemented to control the spacecraft dynamics (aka generally, "the plant") as either a closed or open loop control system.

For open loop control systems, the control law is calculated before activated; therefore, has no sensor input or output feedback loop (see FIGURE 3.6). The controller refers to the hardware and software which contains the control law used to generate control torque commands.

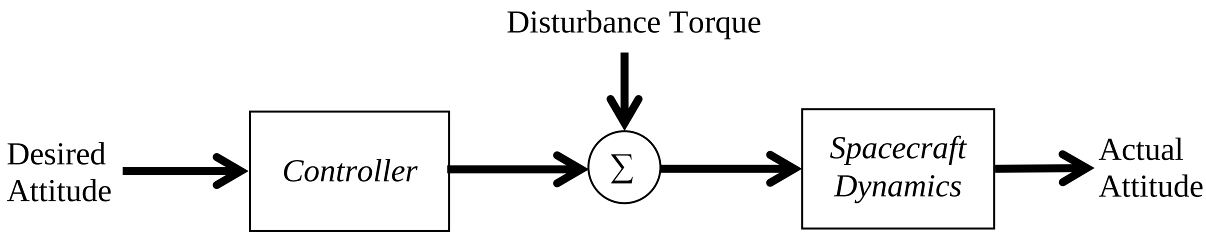


Figure 3.6: Open Loop Control System

For closed loop control systems, the response of the control law is determined real time by system state (see FIGURE 3.7). The sensor component measures the attitude spacecraft dynamics represented by the output (actual attitude) and provides input (feedback) which is compared to the reference (desired attitude). The attitude error, the difference between the desired and measured (with noise) attitude, is input to the controller. The controller computes the control torque command necessary for spacecraft attitude correction. The resulting actuator control torque (output from the controller) and disturbance torques (if present) are input to (affect) the spacecraft dynamics. This process is then repeated to provide continuous active control. This control system is referred to as a closed loop or feedback control system. Much better pointing accuracy can be achieved with an active closed loop control system than a passive open loop one can provide.

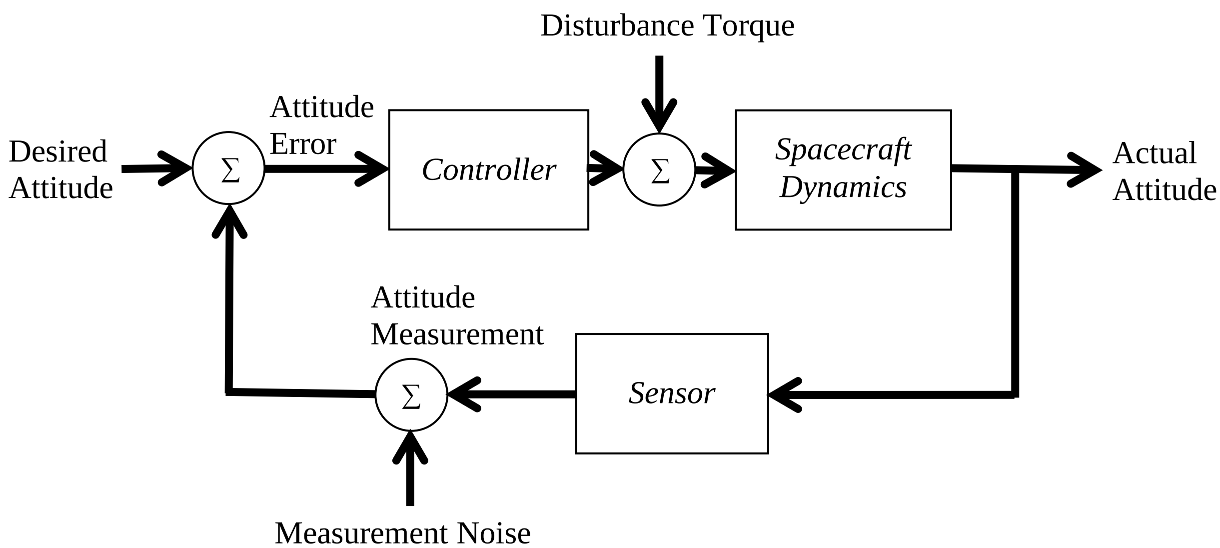


Figure 3.7: Closed Loop Control System

Body Spin

Body spin stabilized spacecraft refers to the method of control where a spacecraft spins about a single axis; keeping this axis in a specific orientation (attitude). This method was the first used in space flight – for communications satellites. With this method, large angular momentum is produced which gives the spacecraft the ability to resist external disturbance torques; where the spacecraft's angular momentum can be defined by the following equation: $\mathbf{H} = \mathbf{I_s}\,\boldsymbol{\omega}$ ... where the variable definitions are:

- I_s – spacecraft moment of inertia ($kg{\cdot}m^2$)
- ω – spacecraft angular velocity (rad/s)
- H – spacecraft angular momentum ($J{\cdot}s$)

This control method is an open loop passive control system, where the spacecraft angular momentum (spin dynamics) provides resistance (i.e. gyroscopic stiffness) to disturbance torques. The desired attitude is achieved first using other control systems before spin dynamics begins. The attitude will be maintained until a "big enough" external moment (disturbance) is applied about its center of mass and perpendicular to its spin axis. Many spacecraft employ the body spin method at some time during its mission (e.g. for its transfer/initial orbit or final orbit); even though it may not be the primary attitude control method used.

A spacecraft will spin about its major axis, which is its lowest energy state and maximum moment of inertia axis (see FIGURE 3.8). Lowest energy state requires the least amount of energy to keep spinning. The spin axis moment of inertia (I_s) should be a minimum of 10% greater than the moment of inertia about the transverse axis (I_t); and therefore, is typically cylinder or drum shape.

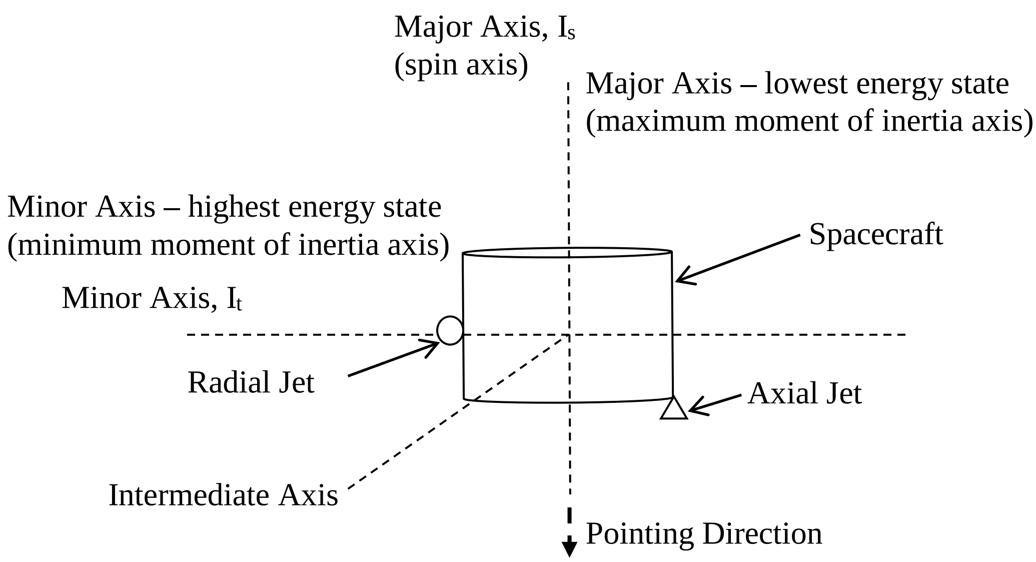


Figure 3.8: Body Spin Control Method

Axial jets parallel to the spin axis are used for attitude maneuvers or corrections. Radial jets can be used to initiate a spacecraft body to spin (rotate). Due to inertia, the spacecraft will continue to spin unless an opposing torque is applied to slow the spin. Typical spin rates are 50 to 100 rpm. Radial jets can then be used to increase or decrease the spin rate.

Another option to decrease or slow the spin rate of a spacecraft is to use yo-yo despin, where two masses are deployed (see FIGURE 3.9). In phase I, the masses are released and are tangent relative to the spacecraft. In phase II, the masses become radial relative to the spacecraft which decelerates its rotation (e.g. like a figure skater extending their arms). After the despin maneuver is completed, the yo-yo masses are typically released from the spacecraft.

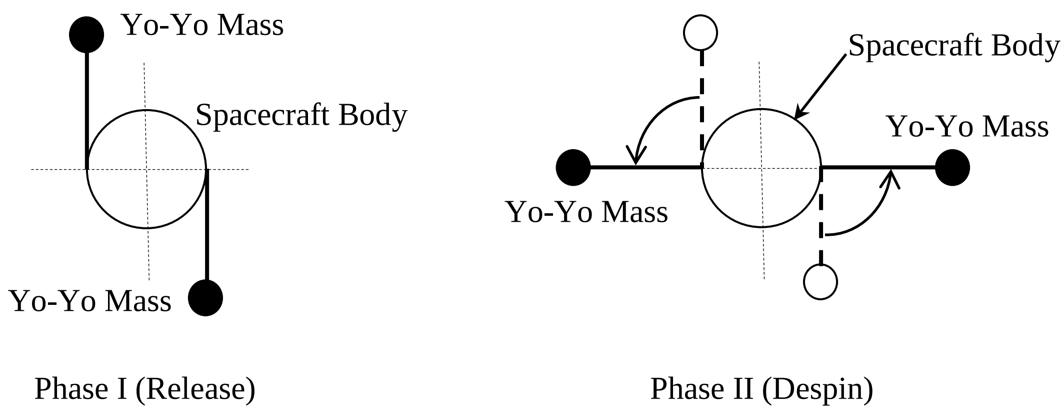


Figure 3.9: Yo-Yo Despin

Pure spin stabilization, since open loop, cannot maintain spacecraft attitude indefinitely due to drift caused by environmental torques. Even small torque over time will cause attitude errors. Therefore, in order to maintain attitude using the body spin method, you will also need to control precession and nutation.

Given time, the spacecraft spin axis (I_s) will precess to an undesired position (or pointing direction). Precession is defined as the rotation of the body spin axis relative to the desired spin axis. To correct for pointing error due to precession, periodic thrusting of the axial jet is required. This precession control, a closed loop control system, is necessary for initial (aka injection) attitude and subsequently attitude corrections.

The body spin stabilized spacecraft also needs a means to control nutation. Nutation occurs when the spin axis becomes off center and rotates about the desired spin axis by some nutation angle (θ); thus, causing a right circular cone pattern of rotation about the origin – an unstable "wobble" condition (see FIGURE 3.10). This condition is caused by the angular momentum vector (h) perpendicular to the spin axis. Passive or active control systems can be employed for nutation control to keep the spin axis as close as possible to desired.

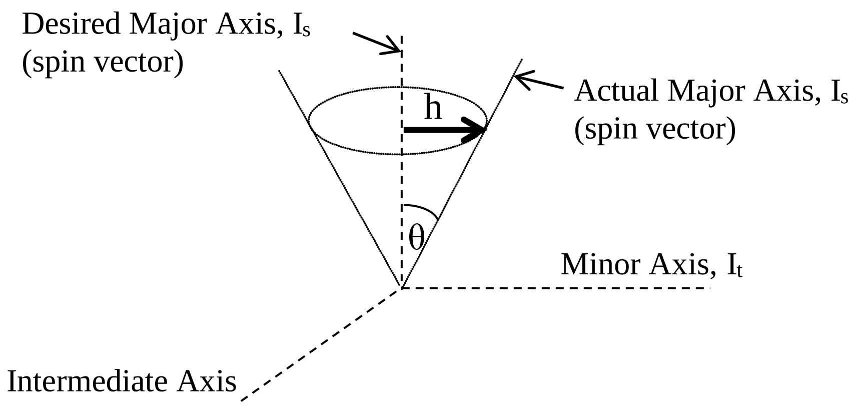


Figure 3.10: Spacecraft Nutation (Wobble)

Gravity Gradient

Another early employed control method was gravity gradient. This method utilizes the differences in the earth's gravity gradients (i.e. gravitational forces) as a function of altitude by changing the inertia tensor of the spacecraft. The inertia tensor of a spacecraft can be increased by extending a long boom where a typical length might be 8m. The gravity gradient torques acting on the spacecraft are increased by placing a mass at the end of the boom to counteract the forces acting on the spacecraft mass in order to keep the spacecraft earth pointing (i.e. control attitude). Gravity gradient stabilization is a passive closed loop control system (like the mass-spring-damper example) with no sensor feedback in the loop; therefore, the restoring torque generated by the controller is based on error size and will be proportional to the attitude error.

Gravity gradient control of spacecraft is limited by altitude (see FIGURE 3.11); where to be effective, its altitude needs to be much lower than geosynchronous (GEO) and above 600km. The boom and two masses (m_1 and m_2) produce the inertias required to maintain nadir (z-axis) pointing toward earth.

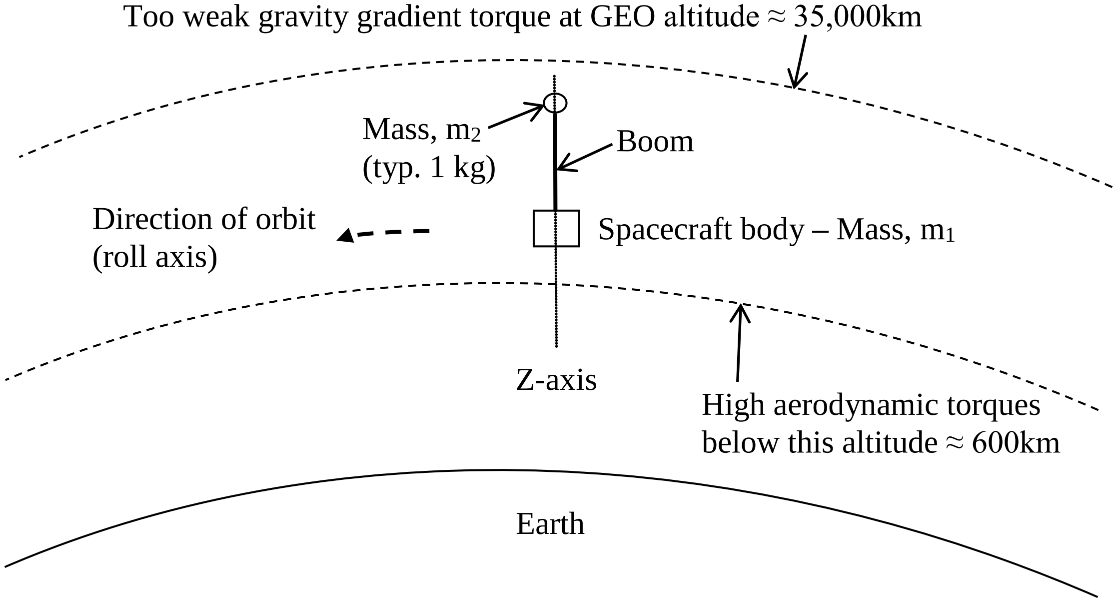


Figure 3.11: Gravity Gradient Control Method

Suppose a gravity gradient controlled satellite (in the orbital reference frame) is tilted about the pitch axis by θ degrees; how will the mass forces (F) interact (see FIGURE 3.12)? The two masses are displaced by boom of length (L), where the force interaction between the two masses cause the spacecraft to act like a pendulum. The difference in forces between the two masses cause a stabilizing (restoring) torque; returning the spacecraft (both masses) to the desired local vertical axis position ($\theta = 0°$). This position is an equilibrium state of a rigid pendulum; hence, also for a satellite. Finally, for stability, the following principle moment inertia (I_{prin}) inequality constraint needs to be met: $\mathbf{I_1 > I_2 > I_3}$; where boom length affects the scalar I_3.

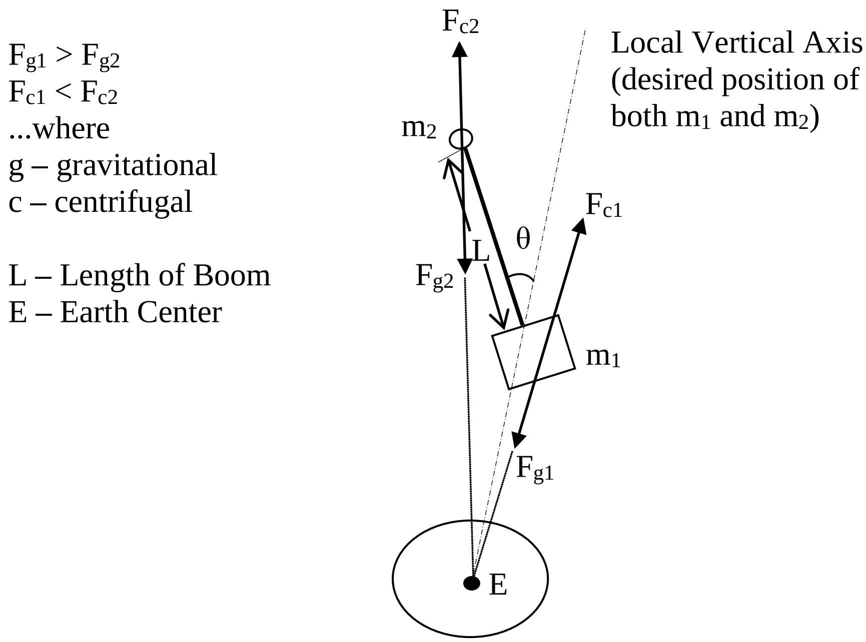


Figure 3.12: Gravity Gradient Forces

Gravity gradient stabilization also needs damping to reduce the energy (magnitude) of oscillations (libration) motion about the point of equilibrium (local vertical axis). One type of damping is to employ three orthogonal electromagnets in the end mass (m_2) to interact with the earth's magnetic field. Although passive, a primary advantage, there are also disadvantages to the gravity gradient control method.

A major disadvantage is that you also need another control method (e.g. momentum bias) employed with the gravity gradient method for the following reasons:

- Prior to boom deployment, for gravity gradient stabilization to begin – initial attitude needs to be achieved
- Most spacecraft are nearly symmetric ($I_1 \approx I_2$) which would violate the previously mentioned constraint for I_{prin}; hence, gravity gradient stability for all three axes is difficult

Moreover, bending or deformation (aka thermal flexing) of the boom can cause destabilizing torques which reduce pointing accuracy and can lead to attitude inversion (i.e. the satellite is upside down). Therefore, before employing this method, risk should be considered and boom thermal properties chosen to minimize flexing. The thermal flexing modes which need mitigation are:

- Dynamic – snapping of boom as satellite crosses from earth shadow into sunlight
- Static – steady state offset of the boom on which the sun shines (i.e. thermal difference between cold side and hot side)

Magnetic

Magnetic actuators (aka torque rods) are effective at lower altitudes where the earth's magnetic field is strongest. Recall from section 2 that the earth's magnetic field intensity is higher at lower altitudes; therefore, the magnetic control method is most effective for low earth orbit (LEO) satellites.

The interaction (vector cross product) between the earth's magnetic field (B_e) and the magnetic moment (M_s) generated within a spacecraft (from the torque rods) produce a mechanical torque (T_B) acting on the spacecraft as represented by the following equation: $\mathbf{T_B = B_e \times M_s}$...where the variable definitions are:

- B_e – earth's magnetic flux density in weber's per square meters (Wb/m^2)
- M_s – applied magnetic moment generated internal to spacecraft in ampere square meter ($A \cdot m^2$)
- T_B – applied magnetic torque acting on spacecraft in newton meters ($N \cdot m$)

Using previous FIGURE 1.4, if we add three torque rods aligned along each axis it is possible to provide magnetic control torques (T_B) for all three axes of a spacecraft (see FIGURE 3.13). This method is a closed loop active control system; where the controller calculates the control torque (M_s) needed for attitude correction based on a control law and feedback inputs of actual attitude from sensors. Torque rod size is dependent on and directly proportional to spacecraft mass and generally range from pencil size to about 2.5m.

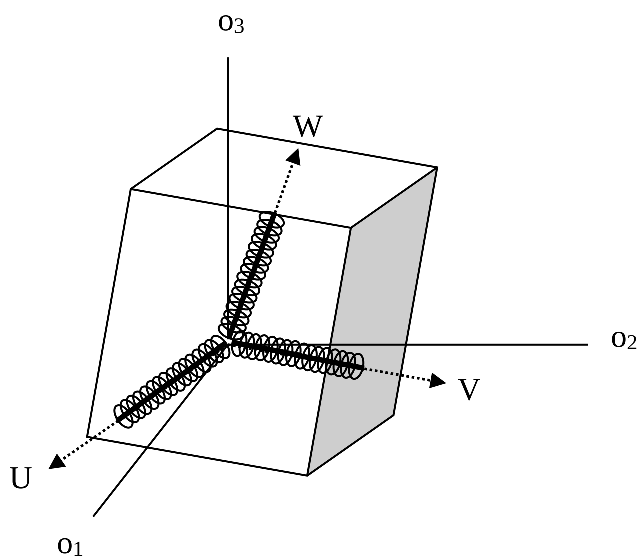


Figure 3.13: Magnetic – Torque Rod Configuration

Mass Expulsion

Spacecraft attitude control can also be achieved by mass expulsion; where a mass (e.g. hydrazine gas) is expelled (directed outward) from a spacecraft thrust (aka jet) nozzle. Mass expulsion can provide very large and rapid changes in spacecraft angular velocity. However, since the fuel used is nonrenewable (i.e. after the tank is empty you are out of gas), using this method can directly impact the expected mission life of a spacecraft. Another major design constraint which affects mission duration is battery design life ─ an electrical power subsystem constraint.

For three-axis control, a minimum of six thrusters are needed; two for each axis, one to provide positive (+) error correction and the other to provide negative (–) error correction. Using previous FIGURE 1.4, we add six thruster nozzles (see FIGURE 3.14). For maximum effectiveness (maximum torque), each thrust nozzle should be placed a maximum distance from the spacecraft body center of mass. In practice, multiple thruster nozzles may be configured on each corner of the spacecraft body, at right angles (orthogonal) to one another; hence, achieving design efficiency and redundancy.

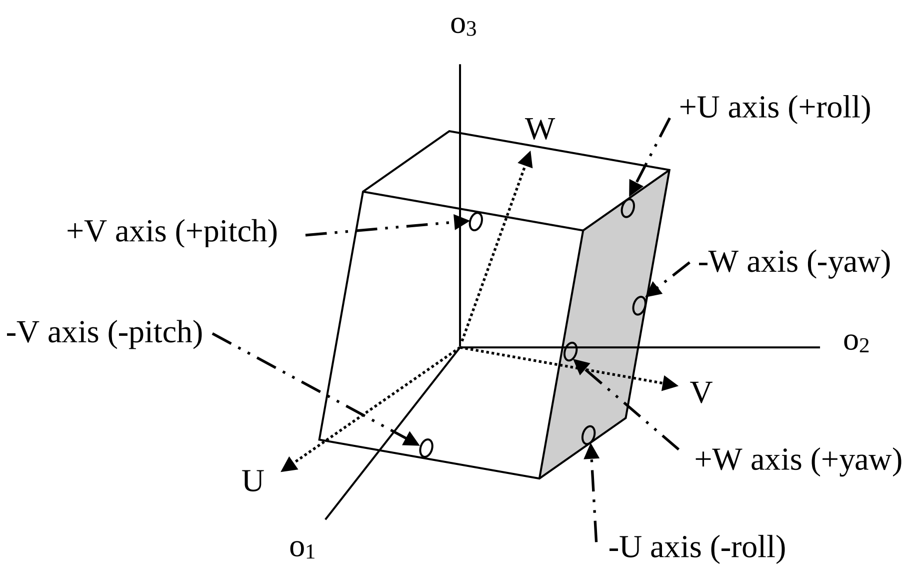


Figure 3.14: Mass Expulsion – Thrust Nozzle Configuration

These thrusters will correct the attitude error to align the spacecraft u,v,w axes with the orbital axes o_1, o_2, o_3 respectively. Where the control torque (M) generated about the spacecraft center of mass is equal to the vector cross product of the distance from the spacecraft body center of mass to the thrust nozzle in the spacecraft body frame (r) and the thrust vector (F): **M = r x F**... where the variable definitions are:

- **r** – distance from spacecraft origin to thrust nozzle in meters (m)
- **F** – applied thruster torque external (expelled from nozzle) to spacecraft in newtons (N)
- **M** – applied control torque acting on spacecraft in newton meters (N·m)

Typical thrust torque values (F) range from 0.05N to 22N and will be directly proportional to spacecraft mass, with a pulse duration of 10ms to several seconds. This method is a closed loop active control system; where the controller calculates the control torque (F) and pulse duration needed for attitude correction based on a control law and feedback inputs of actual attitude from sensors.

Momentum Transfer

Angular momentum can be transferred to the spacecraft using reaction wheels to correct attitude errors. In order to provide three-axis control – three wheels are needed at minimum. In practice, four reaction wheels are often employed in a pyramid configuration to share the workload and also provide redundancy (i.e. a spare wheel). These wheels are zero momentum bias; this means their typical non-active state is at rest (angular velocity = 0 and therefore angular momentum = 0). Using previous FIGURE 1.4, we add three reaction wheels centered about each spacecraft axis (see FIGURE 3.15). These reaction wheels will correct the attitude error to align the spacecraft u, v, w axes with the orbital axes o_1, o_2, o_3 respectively.

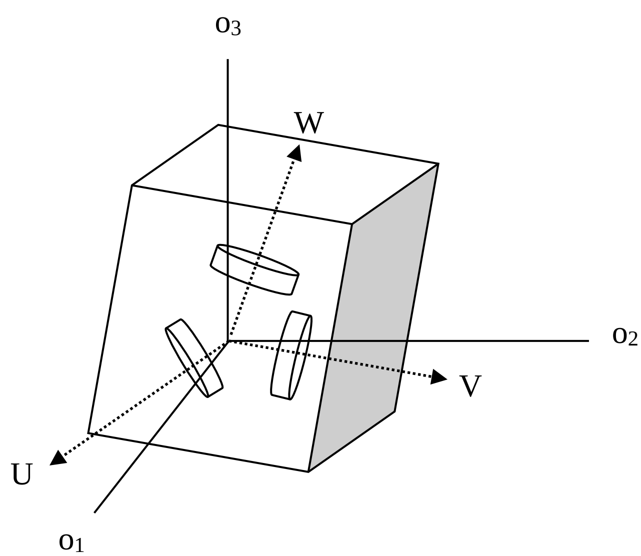


Figure 3.15: Momentum Transfer – Reaction Wheel Configuration

The following equation can be used to find the angular momentum generated by a reaction wheel: $\mathbf{h = I\,\omega}$... where the variable definitions are:

- I – moment of inertia of wheel ($kg \cdot m^2$)
- ω – angular velocity of wheel (rad/s)
- h – reaction wheel angular momentum ($J \cdot s$)

Reaction wheels rotate in either direction (CW or CCW). A clock-wise (CW) wheel rotation will rotate the spacecraft in the opposite direction (CCW) and a CCW wheel rotation will rotate the spacecraft in the opposite (CW) direction. Reaction wheels are limited in maximum angular momentum which they can store and therefore can saturate (i.e. can store no more momentum); therefore, if attitude disturbances exceed wheel capability, wheel saturation will likely occur. When saturated, reaction wheels need to remove their excess momentum using another actuator, typically magnetic torque rods or thrusters. In practice, reaction wheels can be useful in fine error correction (small error correction) then thrusters can be used for coarse error correction (large error correction). Control moment gyros (CMGs) are also used primarily when larger moments (i.e. for larger spacecraft) are needed; they operate on the same principle as reaction wheels. This method is a closed loop active control system; where the controller calculates the control torque (h) needed for attitude correction based on a control law and feedback inputs of actual attitude from sensors.

Momentum Bias

Momentum wheels (aka fly wheels) can operate onboard a spacecraft using a single wheel aligned along the pitch axis which is perpendicular to the orbital plane. These wheels rotate in one direction at a constant velocity (typically between 2000-6000 rpm) to provide gyroscopic stiffness for roll and yaw (passive control) and generate a control torque by varying wheel velocity (Δv) to provide pitch error corrections (active control), similar in concept to the body spin and momentum transfer methods respectively.

The delta velocity (Δv) creates a control torque to rotate the spacecraft in the opposite direction of the wheel acceleration where the control torque generated is proportional to the product of the flywheel moment of inertia (I) and its angular acceleration (α). If bias is CW, the wheel rotates at a constant velocity in the CW direction. An increase in velocity (+ acceleration) will cause the spacecraft to rotate CCW and a decrease in velocity (- acceleration) will cause a CW rotation. This method is a closed loop active control system; where the controller calculates the control torque needed for attitude correction based on a control law and feedback inputs of actual attitude from sensors. In contrast, the gyroscopic stability provided by the constant velocity (bias) of the wheel is open loop passive; same as in the body spin control method.

For gyroscopic stability, momentum wheels maintain minimum angular momentum (h) where the $\Delta v \approx 0$ (i.e. continuous rotation = bias momentum). The angular momentum (L) of a spacecraft with very large wheel angular momentum (h) is given by the following equation:
$\mathbf{L = I\omega + h}$ ... where the variable definitions are:

- h – angular momentum of wheel (J·s), where $h = I_w\omega_w$
- I – spacecraft moment of inertia ($kg \cdot m^2$)
- ω – spacecraft angular velocity (rad/s)
- L – spacecraft angular momentum (J·s)

For spin (gyroscopic) stabilization the following must be true: $|\mathbf{h}| >> |\mathbf{I\omega}|$. This generally means if the wheel (I_w) is not big enough and/or its constant velocity (ω_w) not fast enough; the external disturbance torques will be too much for the internally produced gyroscopic stability provided by the momentum wheel. Disturbance torques can also cause the spacecraft to precess (move) about the spin axis of the momentum wheel by some angle φ. However, if h is very large compared to the environmental disturbance torque; angle φ will remain very small.

In practice, the momentum bias method alone is not enough to control the spacecraft; therefore, another control method (e.g. gravity gradient or magnetic) is also necessary to maintain the required attitude. In addition, as previously mentioned, wheels saturate when they have too much momentum; hence, they need to transfer this excess momentum to the environment using either thrusters or magnetic torque rods.

4. Early Flight Anomalies

We conclude this course by reviewing some spacecraft attitude problems (anomalies) at the beginning of space flight from 1958 to 2013 in the following table – from earliest to latest. For the most part, these anomalies resulted in a total loss of attitude control and/or attitude control subsystem degradation; thus, impacting or interfering with the spacecraft mission. The causes which were due to the natural space environment (NSE) include: thermal environment, plasma, meteoroids, solar environment, ionizing radiation, and the geomagnetic field. For many of the anomalies, footnote numbers in brackets [ ] in the "Caused by" column provide more information which can be found after this table. As you review each anomaly, the significance of the previous sections should be clearer, and help reinforce what you have learned.

Table 4.1: Spacecraft Attitude Anomalies (1958-2013)

Spacecraft	**Year**	**Anomaly**	**Caused by**
Explorer I	1958	Loss of attitude stabilization about its intended spin axis	Engineering design error; chose minor axis as spin axis instead of major axis [1]
Gemini VIII	1966	Began rolling after docking with Agena Target Vehicle (ATV)	One Gemini capsule thruster stuck on; after undocking with the ATV, the spacecraft rolled up to one rev/s [2]
Apollo 13	1970	Sudden loss of attitude control	Oxygen tank explosion caused oxygen venting; which acted as uncontrolled mass expulsion [3]
ANIK-B	1986	Increased roll error	Roll and yaw axis torque rod malfunction (NSE: geomagnetic field) [4]
ERBS (Earth Radiation Budget	1986	X-gyro in the inertial reference unit (IRU-1)	Hardware failure – mechanical wear

Spacecraft	Year	Anomaly	Caused by
Satellite)		failed	
ERBS	1987	Command roll error	Human error; command error in procedure caused roll thruster to burn too long, continuously for 13 minutes before stopped by ground command [5]
Polar BEAR (Beacon Experiment and Auroral Research)	1987	As spacecraft entered first period of full sun orbit, the attitude became degraded substantially and then the spacecraft inverted	Probable cause, thermal bending of boom caused by the difference between cold side and hot side (NSE: thermal environment) [6]
TDRS-1 (Tracking and Data Relay Satellite)	1989	Temporary loss of attitude control	A probable single event upset (SEU) in the command processor electronics (NSE: ionizing radiation)
NOAA-10 (National Oceanic and Atmospheric Agency)	1989	Excessive x-axis gyro velocity after magnetic momentum unloading caused roll and yaw torque rods to switch to backup mode	Suspect caused by solar activity (NSE: solar environment)
HST (Hubble Space Telescope)	1990	Gyro operations affected	Low frequency vibration [7]
HST	1990	Random access memory (RAM) of the fine guidance electronics malfunctioned resulting in star acquisition failures	RAM bit-flips occurred when passing through the South Atlantic Anomaly (SAA); an area of high radiation (NSE: ionizing radiation)
HST	1993	Y-axis star tracker failed to acquire navigation stars for about five hours	Single event upset caused by the SAA (NSE: ionizing radiation) [8]

Spacecraft	**Year**	**Anomaly**	**Caused by**
Anik E-2	1994	Uncontrolled spin	Gyroscopic actuators (aka control moment gyros) failed due to electrostatic discharge (NSE: plasma) [9]
Clementine	1994	Loss of attitude control thrusters	Human error; a software sequencing error opened four of the twelve attitude control thruster valves, depleting all the fuel
Lewis	1997	Flat spin about non-sun pointing z-axis	Engineering design error; safe mode incorrectly pointed x-axis of inertia (intermediate/unstable) instead of z-axis (major/stable) toward sun [10]
WIRE (Wide-Field Infrared Explorer)	1999	High rate tumbling due to excessive hydrogen venting	Pyro electronics box malfunction due to a digital logic design error [11]
Mars Climate Orbiter	1999	Incorrect mars trajectory insertion resulted in loss of spacecraft	Human error; failure to use metric units in AMD file resulted in incorrect navigation Δv (delta velocity) after using thrusters for AMD which affected spacecraft trajectory [12]
FUSE (Far Ultraviolet Spectroscopic Explorer)	2001	One gyroscope sensor failed	Hardware failure – mechanical wear [13]
FUSE	2001	Two of the four reaction wheels failed	Hardware failure – mechanical wear [14]
IMAGE (Imager for Magnetopause to Aurora Global Exploration)	2005	Passive nutation damper failed	Liquid mercury in the damper tube could not move due to surface tension (liquid adhering to solid tube) [15]
Landsat 5	2009	Extreme gyro rates resulted in uncontrolled attitude rotations	Suspected from Perseid meteor shower (NSE: meteoroids) [16]
GOES-13 (Geostationary Operational Environmental	2013	Lost star tracking required to maintain earth pointing attitude	Apparent micrometeorite impact (NSE: meteoroids) [17]

Spacecraft	**Year**	**Anomaly**	**Caused by**
Satellite)			
Kepler	2013	Second reaction wheel failed; had total of four	Hardware failure – mechanical wear [18]

Note: This table is by no means intended to be all inclusive, but a sample of some spacecraft attitude anomalies to be representative of the time period.

[1] Spacecraft began to spin about its major axis – which it wants to do; however, by incorrect design, intent was to spin about minor axis. This was the first U.S. satellite: Length=203cm(80in), Diameter=15cm(6in), Weight=13.9kg(30.7lb).

[2] This was a manned NASA spacecraft; it rolled faster after undock due to Moment of Inertia change (decreased).

[3] Regained control after oxygen depleted from tank. The oxygen tank explosion was due to faulty electrical wiring.

[4] Control was maintained using thrusters as needed during the NSE disturbance event.

[5] Produced undesired roll rates of 2°/s (smaller about pitch and yaw). Attitude was corrected.

[6] A gravity gradient stabilized spacecraft consisting of a single boom with tip mass. It used momentum wheel spin/despin to provide enough torque to reinvert the spacecraft (i.e. get it back upright). This spacecraft was built from the Transit-O 17 navigational satellite; where it had been on display in the Smithsonian’s National Air & Space Museum for eight years.

[7] Thermally triggered by solar arrays due to transition from day to night and night to day.

[8] Star tracker functioned correctly after power cycle.

[9] Backup system failed to operate; used ground control commands to activate thrusters to restore and control attitude.

[10] During safe mode (a mode intended to protect the spacecraft during anomalies), the spacecraft transferred its spin from its x-axis (intermediate axis) to its z-axis (major axis); subsequently, causing a flat spin which resulted in the solar array pointing edge to sun instead of panel (flat) side. Unable to generate power, this anomaly resulted in total mission loss.

[11] Continued hydrogen venting of about two times (2x) the counter torque which the magnetic torque rods were capable of applying. Attitude was stabilized after the hydrogen was depleted; but without hydrogen, the spacecraft was unable to perform its mission.

[12] Angular momentum desaturation (AMD) used thrusters to dump excess angular momentum from the reaction wheels. The AMD file incorrectly used English units of pounds (force)-seconds (lbf-s) instead of metric units of Newton-seconds (N-s) for impulse (force x time). This resulted in orbit solutions to be low by a factor of 4.45 (1 pound force = 4.45 newtons).

[13] The spacecraft had six gyros (two for each axis); where loss of two for any axis would cause attitude problems.
[14] Used magnetic torque rods to provide actuator control in place of failed reaction wheels.
[15] A spin stabilized (controlled) spacecraft. After failure, used magnetic torque rod as active nutation damper.
[16] The spacecraft attitude was recovered within 24 hours of the anomaly.
[17] Recovered attitude and operational within about 3 weeks. Note: geostationary satellites have an inclination angle of 0°; but have the same altitude as geosynchronous satellites.
[18] With only two reaction wheels left; the spacecraft had limited pointing using thrusters (i.e. coarse pointing) and no fine pointing without three reaction wheels.

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