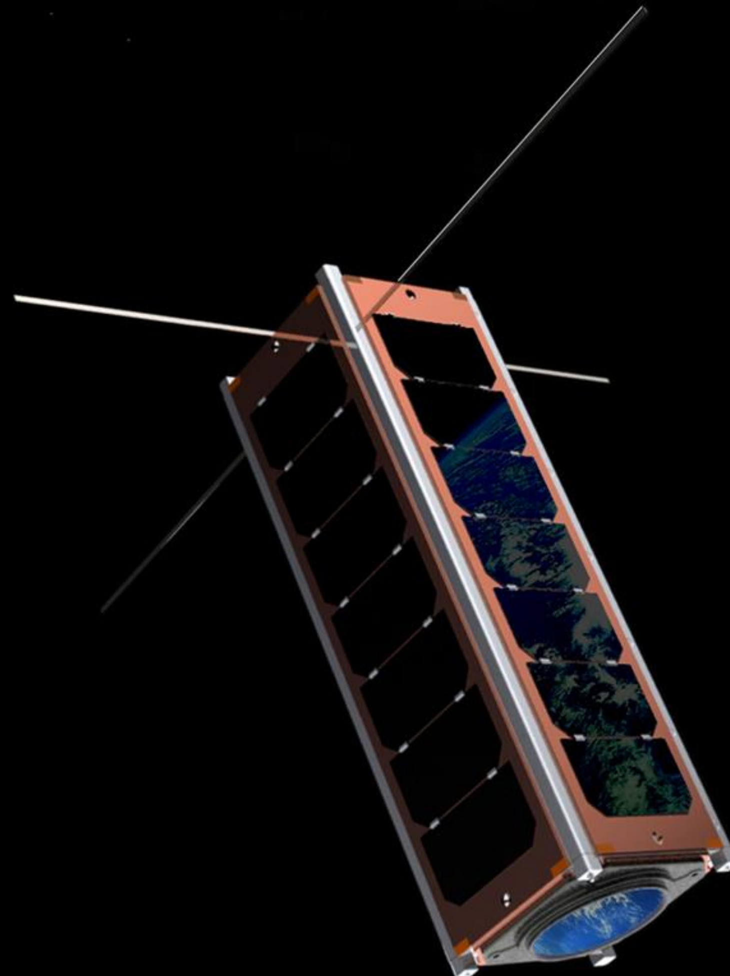

M802 ADCS Subsystems

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Course Content

- **Week 1:** Brief history & context: Background to the development of space, agencies, space history/policy, space economics, funding, future missions – case studies.
- **Week 2:** Introduction to space system design methodology: requirements, trade-off analysis, design specifications, system budgets. Introduction to space system architecture. Launch Vehicles.
- **Week 2:** Space and Spacecraft Environment: Radiation, vacuum, debris, spacecraft charging, material behaviour and outgassing.
- **Week 3:** Orbit Mechanics: celestial mechanics, orbits, trajectory design and spacecraft maneuvers
- **Weeks 4-10:** Spacecraft sub-systems design: Structure & configuration; Power, the power budget and solar array and battery sizing; Communications and the link budget; Attitude determination and control; Orbit determination and control; propulsion Thermal control.
- **Week 11:** Mission and payload types Spacecraft configuration: examples of configuration of spacecraft designed for various missions small satellite case study.
- **Week 12:** Assembly, Integration and Test processes; Launch campaign; Space mission operations. Space Project Management
- **Week 13**: Review, tutorial problems/exam mock up

Attitude Determination and Control Systems (ADCS)

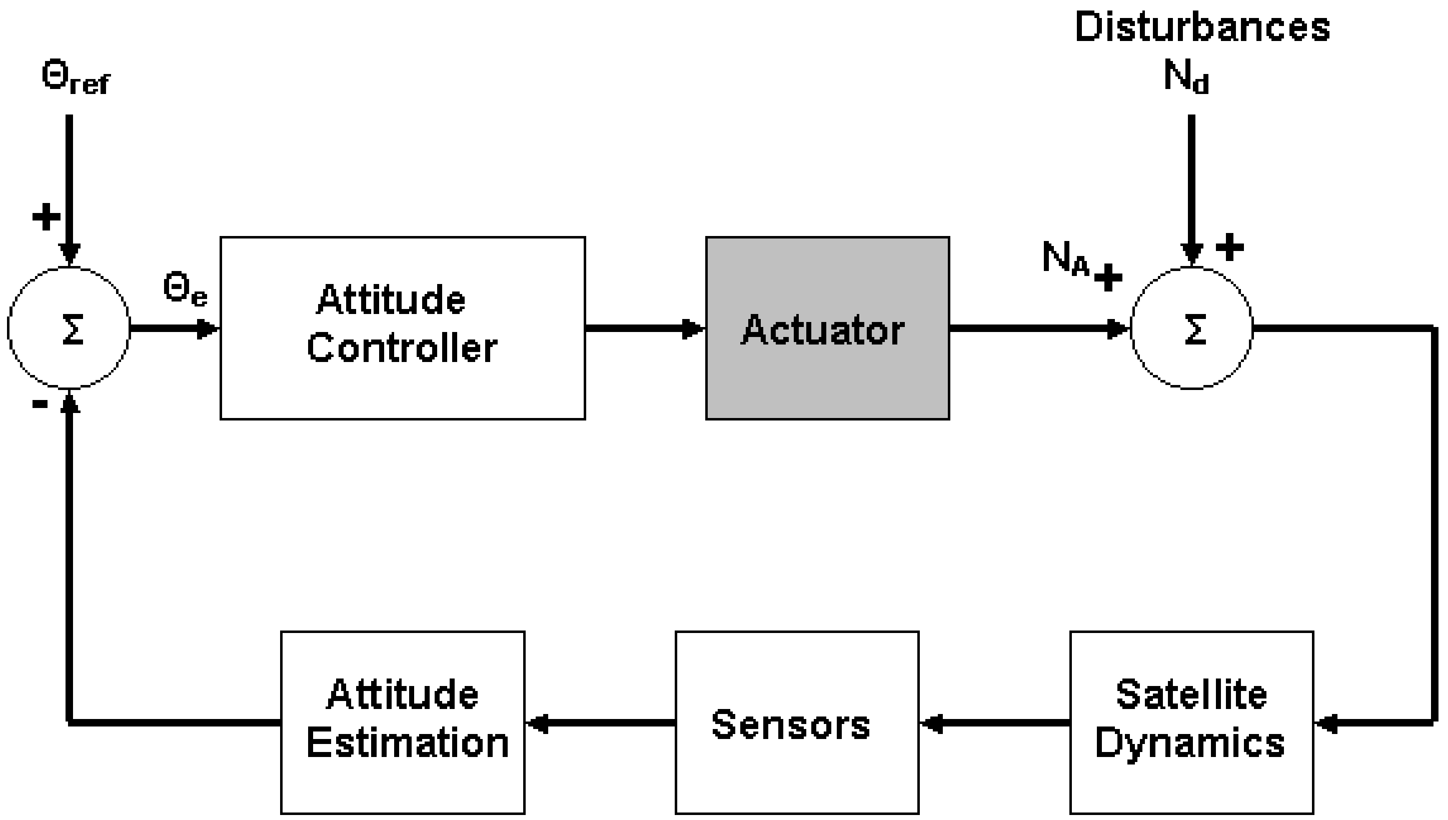
- Key Subsystem of Spacecraft
- Important for stability and pointing
- Think of a person taking a picture with a camera (blurry pictures if not stable)
- ADCS Systems are complicated:
  – Fusion of software/hardware components
  – Need electronics/aerospace/controls principles
- Brief overview of ADCS components:
  – Stabilisation
  – Sensors for attitude determination
  – Actuators

Introduction to ADCS

- **Attitude Control Systems (ACS): Orientation of a spacecraft in a particular direction (pointing) during a mission, despite external disturbances**
- Need to know s/c attitude (Determination) and then point using an actuator (Control) to desired target

- An ADCS system takes sensor readings from sensors in order to provide rate and position estimates of the spacecraft with respect to its centre of gravity (mass). Then, in a closed loop system a command is implemented to use actuators to point the spacecraft to a particular direction of interest or in order to cancel an external disturbance. The closed loop system performs these tasks in 'feedback' type of way until the attitude error is made to go to zero and the manoeuvre is completed.

ADCS Block Diagram

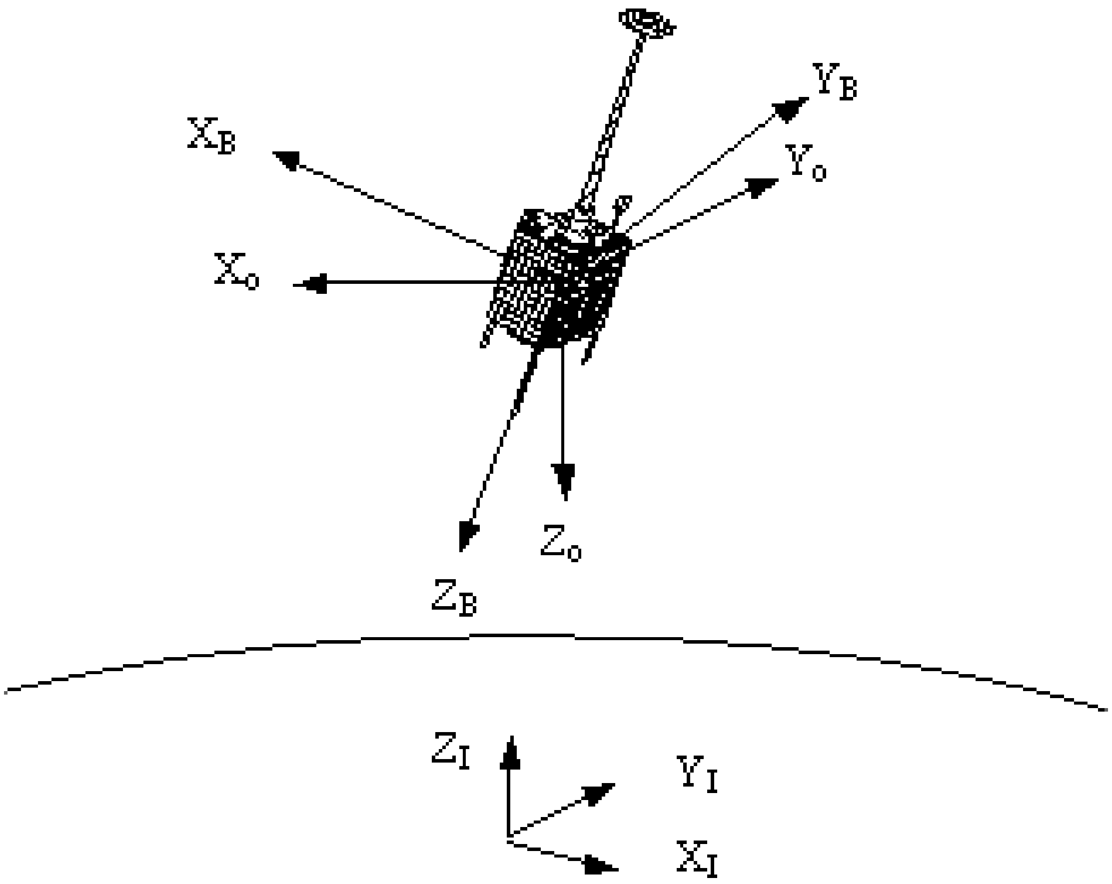


Reference Frames

i. The Inertial Reference Frame: The origin of the inertial reference frame X_I $_,Y_I$ $_,$ Z_I is the centre of the Earth. Z_I is in the same direction as the Earth's geometric north pole and X_I is defined in the vernal equinox direction and Y_I is perpendicular to these. This particular reference frame is used to calculate the latitude and longitude of the satellite's centre of mass as it moves along its orbit.

ii The Body Frame: The second set of coordinates X_B $_,Y_B$ $_,$ Z_B, are referred to as the body frame. The origin of this frame is placed at the spacecraft centre of mass. This frame is considered to be fixed in the satellite's body and is used to determine the satellite's orientation with respect to other reference frames (e.g. orbit reference frame)

Reference Frames



Attitude Control Types

a. Gravity Gradient: Uses the inertial properties of a vehicle to keep it pointed to the Earth. It relies on the fact that an elongated object in a gravity field tends to align its longitudinal axis through the Earth's centre. Example: UoSAT-12 Earth Observation satellite

b. Spin Stabilsation: A passive control technique in which the entire spacecraft rotates as to enable its angular momentum vector to be approximately fixed in inertial space. Uses gyroscopic properties to compensate/cancel external disturbance. Example: Boeing 376 Communications satellite

c. 3-axis Control: Spacecraft is stabilised about three its three body referenced axes of rotation using active means of control such as thrusters, reaction wheels, magnetorquers. Ikonos-1 Earth Observation satellite

External Disturbances

- In order to determine and size of the ACS, one must first quantify the torques acting on a spacecraft. These can be distinguished into controlled actuator torques (e.g. magnetic torquers, reaction wheels,) and external torques (e.g. gravity gradient, aerodynamic, solar pressure, etc.).

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Gravity Gradient

The gravity gradient torque is a torque that originates from the "dumb bell" effect on a long thin rotating object. This torque is created due to the finite distance between the opposite ends of the spacecraft, causing a slight difference in the forces acting on those ends, resulting in a torque about the spacecraft's centre of mass. Gravity gradient torque for satellites with small products of inertia is defined as

$$\mathrm{N}_{\mathrm{GG}} = \frac{3\mu}{2R_e^{\ 3}}\left[I_{zz} - \frac{I_{xx} + I_{yy}}{2}\right](\mathbf{z}_0 \cdot \mathbf{z})(\mathbf{z}_0 \times \mathbf{z})$$

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Gravity Gradient

μ is Earth's gravitational constant ($3.986 \times 10^{14}\ m^3s^{-2}$)

I_{xx} is the spacecraft's moment of Inertia about the x-axis

I_{yy} is the spacecraft's moment of Inertia about the y-axis

I_{zz} is the spacecraft's moment of Inertia about the z-axis

R_e is the spacecraft orbit radius (700km)

$\mathbf{z}_0 = [A_{13}\ A_{23}\ A_{33}]^T$ is the nadir unit vector in body coordinates

$\mathbf{z}$ is the principal body Z-axis unit vector

Solar Radiation Pressure

This torque is caused mainly due to the difference in location of the satellite's centre of pressure and its centre of gravity. Solar radiation will reflect off the satellite in parts of the spacecraft's orbit and this will create a torque about the spacecraft's centre of gravity. This torque is defined:

$$\mathbf{N}_{\mathbf{sp}} = F(\mathbf{C}_{\mathbf{ps}} - \mathbf{C}_{\mathbf{g}})$$

where,

$$F = \frac{F_s}{c} A_s (1 + q)\cos(i)$$

Solar Radiation Pressure

F_s is the average solar constant (1358Wm^{-2})

c is the speed of light (3.0 x 10^8 ms^{-1})

$\mathbf{C_{ps}}$ is the centre of solar pressure vector

$\mathbf{C_g}$ is the centre of gravity

A_s is the spacecraft's surface area projected towards Sun

i is the sun incidence angle

q is the reflectivity/transparency factor

For the $\mathbf{C_{ps}}$-$\mathbf{C_g}$ term, an estimated value of 0.1 m is used and for reflectivity q a value of 0.6 is typical of small spacecraft. The cross sectional area of the SSTL Microsatellite bus is 0.129 m^2.

Aerodynamic Disturbance

In Low Earth Orbits (LEO, i.e. < 2000 km), one can not dismiss the effect of Earth's atmosphere (drag). The atmospheric torque disturbance N_A is directly proportional to the cross sectional area A_p and to atmospheric density ρ.

$$\mathbf{N}_{\mathbf{A}} = \frac{1}{2}(\rho C_D A_p V^2)(\mathbf{C}_{\mathbf{pa}} - \mathbf{C}_{\mathbf{g}})$$

where,

ρ is the atmospheric density (kgm^{-3})

C_D is the drag coefficient

A_p is the spacecraft projected area (m^2)

V is the spacecraft velocity (ms^{-1})

$\mathbf{C}_{\mathbf{pa}}$ is the centre of aerodynamic pressure of the spacecraft

$\mathbf{C}_{\mathbf{g}}$ is the centre of gravity

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$\mathbf{C_g}$ is the centre of gravity

Aerodynamic Disturbance

Taking as an example a SSTL microsatellite, orbiting at an altitude of 700km, the spacecraft velocity will be:

$$V = \left(\frac{\mu}{r}\right)^{1/2} = 7452\text{ms}^{-1}$$

where,

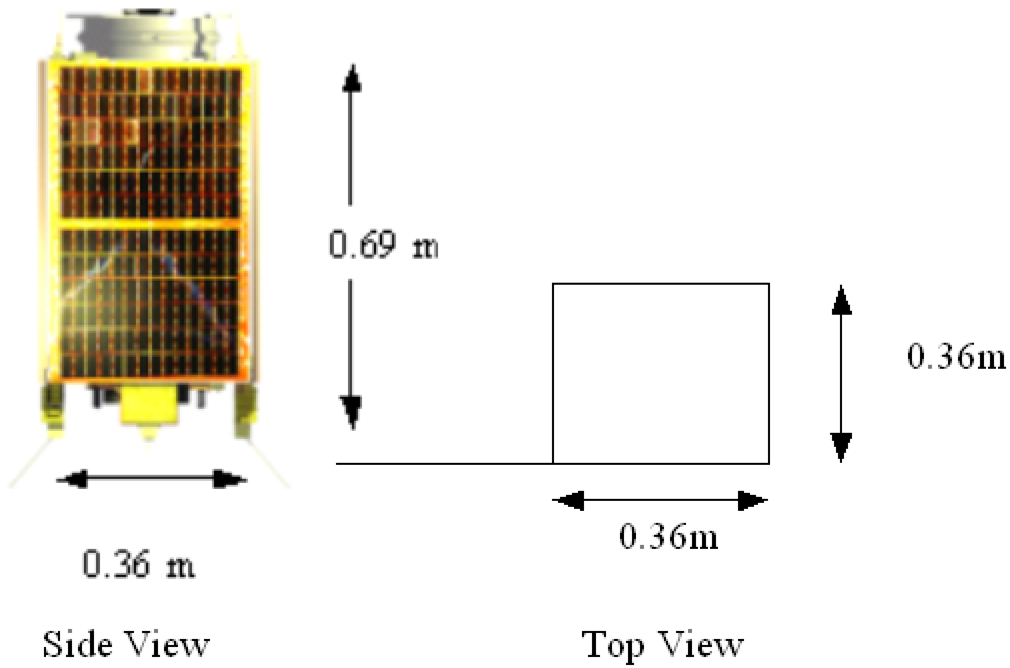
μ is the gravitational parameter of the Earth

r is the spacecraft's orbital radius

From the 700 km orbit we can find the average atmospheric density, ρ_{ave}=10^{-14} kgm^{-3} The drag coefficient is also estimated (C_D= 2.0) and the cross sectional area is already known as 0.129 m^2. The torque generated by aerodynamic disturbances is in the order of 10^{-7} N m.

Example Problem

- Assuming that only the disturbance on a spacecraft in a circular LEO is aerodynamic, use the data on table 1, to calculate the torque needed for an actuator to compensate for aerodynamic disturbance



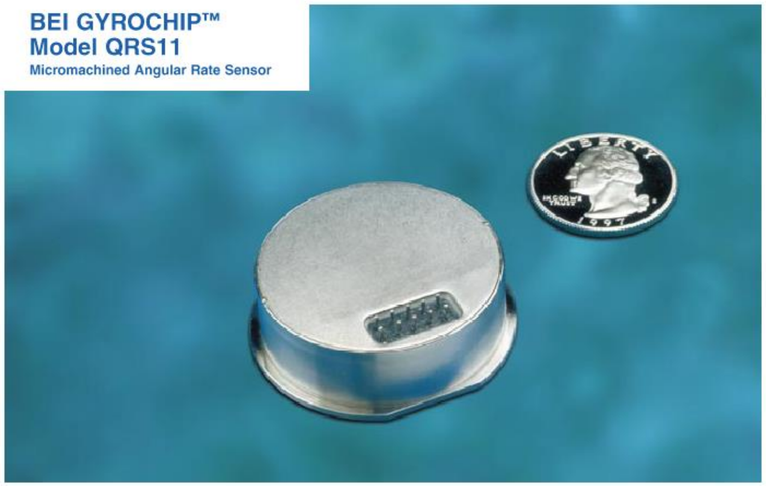
μ	3.9×10^{14} m^{3}s^{-2}
r	700 km
C_D	2
ρ_{ave}	10^{-14} gm^{-3}
C_{pa}-C_g	0.1 m

Table 1

Attitude Determination

- Need to know where and with what rate of rotation spacecraft is pointing to
- Then we can control or 'actuate' the spacecraft
- We can choose sensors depending on the pointing (deg) and stability (deg/s) requirements of a mission
- Pointing and stability requirements are variable and depend on the instruments of each spacecraft (e.g. hi resolution camera)
- Selecting attitude sensors is also a function of cost and mass availability

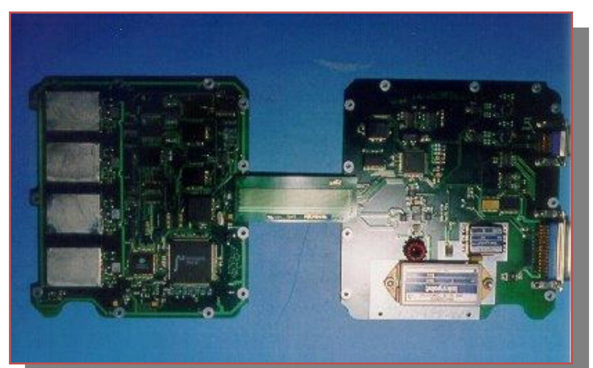
Sensors for Attitude Determination

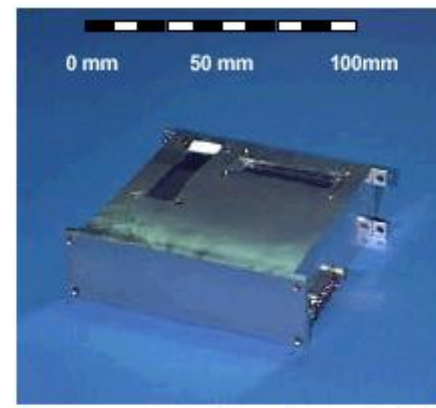


Gyros

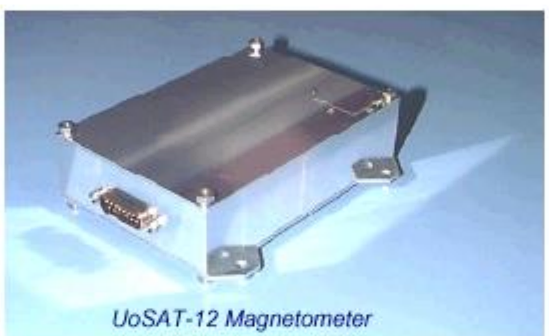
Star Camera

GPS





Sun Sensor



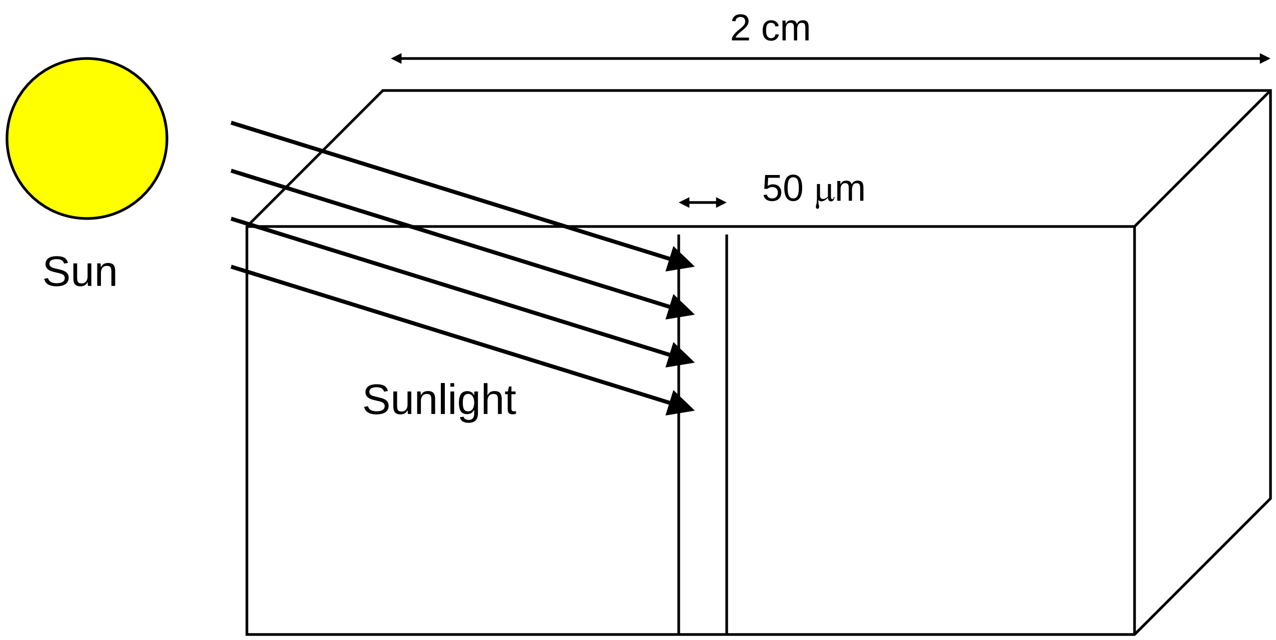
Sensors

Attitude Sensor	Performance	Mass (kg)	Power (W)
Inertial Measurement Unit	Gyro drift rate: 0.003° /hr to 1°/hr	3-25	10-200
Sun Sensor	Accuracy: 0.001° to 3°	0.5-2	0-3
Star Sensor	Accuracy: 0.0003° to 0.1°	0.5-7	4-20
Horizon Sensor	Accuracy: 0.05° to 1°	2-5	0.3-10
Magnetometer	Accuracy: 0.5° to 3°	0.15-1.2	<1

From Wetz & Larson : Space Mission Analysis and Design

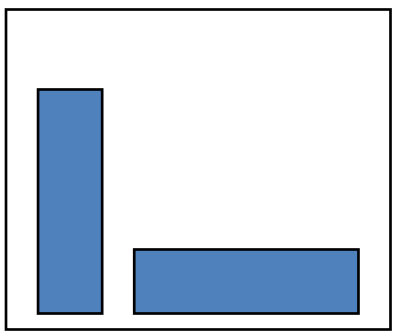
Sun Sensor (1)

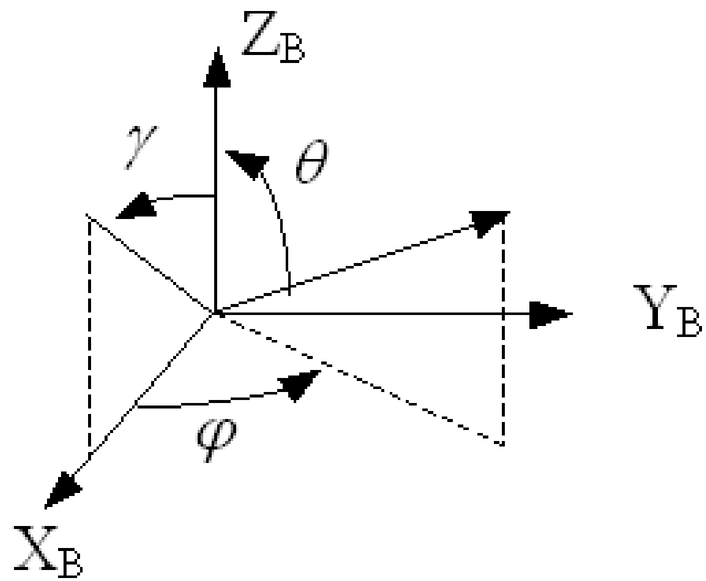
- When a satellite is not in eclipse we can determine the direction from which the Sun's light falls upon it. This is called a sun sensor



Sun Sensor (5)

- We use 2 orthogonal slits to measure the elevation & azimuth angles of the Sun, it is considered a point light source at infinity.

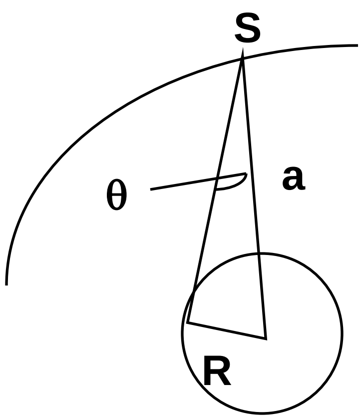




- One detector detects ϕ, but the other detector detects angle γ which is the projection of the sun direction on to the Y_B plane
- X_B plane: $F\sin\theta\cos\phi = F\sin\gamma$, $\sin\gamma = \sin\theta\cos\phi$, hence γ is modulated by ϕ

Horizon Sensor (1)

- A satellite in LEO can also detect the limb of the Earth
- This is because the Earth is much brighter than the background sky
- The ability to detect the Earth’s horizon enables us to detect roll and pitch on the satellite but as the Earth is almost spherical we cannot detect yaw



For a satellite S at orbital attitude a, then the angle θ to the horizon is

$$\sin\theta = R/a$$

Horizon Sensor (2)

- We use a simple single pixel array canted at this angle, so then the satellite is perfectly nadir pointing, the horizon of the Earth should be in the middle of the array.
- Knowing FOV of sensor and how many pixels, we can determine angular resolution per pixel.
- This converts number of pixels shifted into a pitch angle
- A similar sensor at $\pi/2$ gives roll angle

Star Sensor

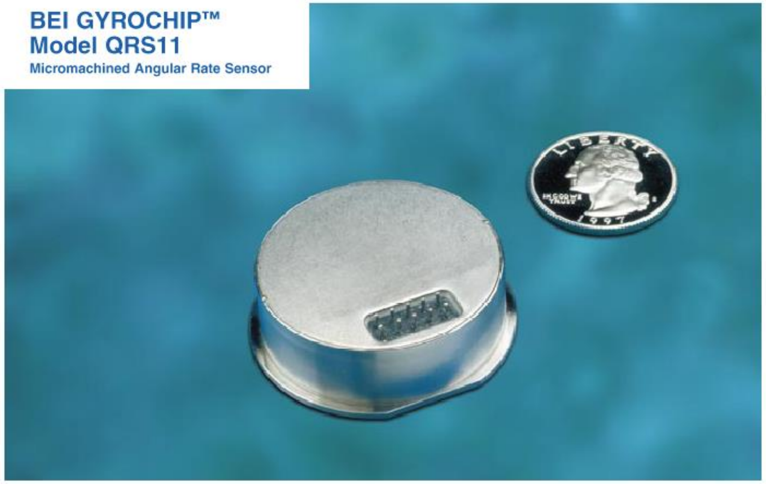
- A star sensor is a telescope that images star patterns for bright stars in the sky
- Once a pattern has been formed it is then compared with a catalogue of bright stars held in memory
- Sophisticated pattern matching algorithms are used to make the match
- This then allows us to determine directly the attitude of the spacecraft as we have identified several inertial directions
- Does it matter where we are in orbit? No. Stars are at infinity so small shifts in satellite position are negligible
- Star trackers latch on to particular stars or planets and track their progress in time across star camera images.
- The provide not only attitude information (kinematics), but also estimates of angular relation (dynamics)

Magnetometers

- Satellites in LEO move through the Earth's B field and we have sophisticated models of the magnetosphere
- Knowing the satellites position in orbit, we know the direction of the local B field vector.
- So by measuring the B field with a magnetometer we can estimate the attitude changes as we move around in orbit
- The B field model is the IGRF (International Geomagnetic Reference Frame) and expresses the B field in spherical harmonics

Rate Gyroscope

- Single gimbal wheel with spinning & damper on the wheel axis
- External torque from the satellite cause the gimbal to rotate
- Equilibrium is established with the spring resulting in a fixed angular offset which is proportional to the rotation rate of the satellite



Actuators

- Actuators can be divided into inertial and non-inertial actuators. Inertial actuators are devices that generate torques, by modifying their angular momentum. They can be grouped into three categories:
  - Momentum Wheels (MW): They provide constant angular momentum for gyroscopic stabilisation. Orientation of the spin axis is fixed with respect to inertial space. Attitude Control is achieved by varying the spin speed of the wheel about some nominal value.
  - Reaction Wheels (RW): They provide torque to a vehicle by increasing or decreasing the speed of the wheel, with the wheel nominally at rest.
  - Control Moment Gyroscopes (CMG): A momentum wheel gimballed in one or two axes. Control torques are generated by changing the direction of the momentum vector, by changing the direction of the spinning wheel's axis.

Actuators

- **Non-inertial actuators:**
- 1. Magnetic torquers (MT): Magnetic coils or electromagnets that generate magnetic dipole moments, **M**. A magnetic torquer produces torque proportional (and perpendicular) to Earth's magnetic field, **B**. It is often used as a second actuator on spacecraft to desaturate momentum exchange systems.
- Thrusters: Produce a thrust (force) or torque around the centre of mass by expelling mass.

Rotational Motion

- ADCS requires concepts of *rotational motion*, which are less intuitive than those of translation. However, the concepts involved, such as *torque* and *angular momentum*, are analogous to those of translational motion.
- To fully understand these concepts, and to appreciate phenomena such as *precession*, you will need to be conversant with vector algebra and matrix operations.

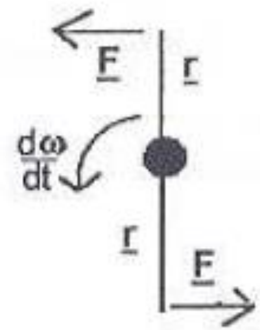
Torque

The following applies when the reference point is the C.o.M., *C*, or an inertially "fixed" point, I.

- Let us define "*external torque*" as being due either to an external *couple*, or to a force which has a *moment* about our reference point.
- An *external torque*, *T*, will cause a *rate of change* of angular momentum:

➜ $\underline{T} = d\underline{H}/dt$

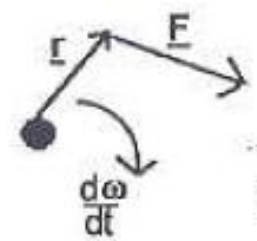
(w.r.t. *C* or I as appropriate)



A COUPLE

$\underline{T} = 2\,\underline{r} \times \underline{F}$

(Forces equal and opposite Acting at an equal distance, r)



BASIC TORQUE

$\underline{T} = \underline{r} \times \underline{F}$

$$\underline{T} = \frac{d(\underline{H})}{dt} = \frac{d(I\underline{\omega})}{dt} = I\,\frac{d\underline{\omega}}{dt}$$

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Torque

- *Internal torques*, acting between particles or bodies will *not* change the total momentum.
  - Thus mechanisms, fuel movement, etc. will not change the spacecraft's angular momentum.
- However, there will always be *external disturbance torques* occurring naturally for spacecraft.
  - This gives rise to a progressive build-up of angular momentum, leading to unacceptable rotation rates!
  - Thus, spacecraft need external torquers in order to control this build-up.
  - These might be magnetic coils, rocket thrusters which are not aligned with the C.o.M., etc.

Torque

We can now analyse 3 simple cases:

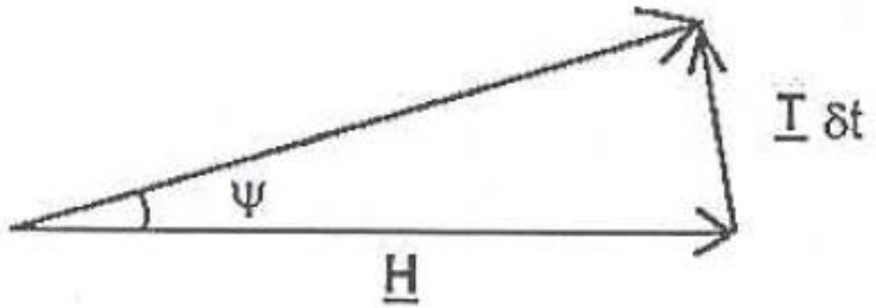
① *Zero* external torque, $\underline{T}$, => angular momentum, $\underline{H}$, is *constant* in magnitude *and* direction.

② An external torque applied in the *same* direction as $\underline{H}$ will *increase* the magnitude of $\underline{H}$ *without* changing its direction.

③ An external torque which is at all times *perpendicular* to $\underline{H}$, will change the *direction* of $\underline{H}$ but *not* its magnitude.

- This is the effect which is illustrated by the precession of a gyroscope.

Gyroscopic Rigidity



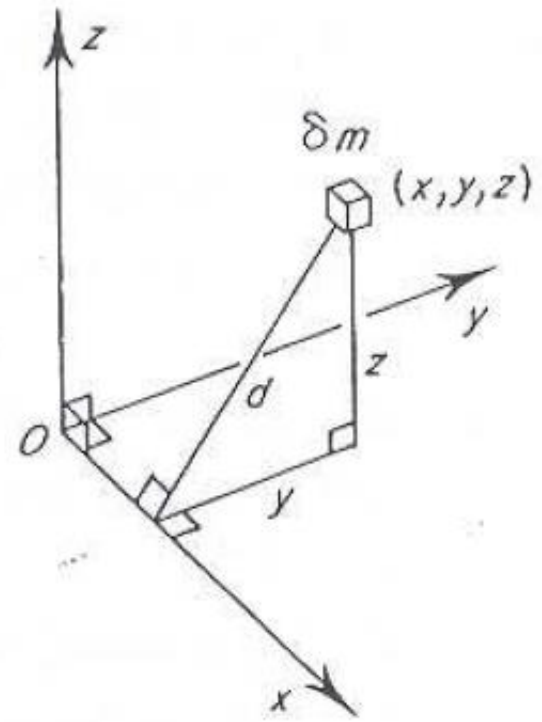
- If H is large, then the change in direction of *H*, $\delta\psi$, is small for a given torque impulse, $T\delta t$, i.e.

  ➡ $d\Psi/dt = T/H$ (precession rate)

- Thus, by giving a satellite a *momentum bias*, we can make the direction of this bias insensitive to disturbance torques: *gyroscopic rigidity*.

Inertia

- <u>Moments of Inertia:</u>
  I_{xx}, I_{yy}, I_{zz}
- The *moment of inertia*, *I*, of a body about any axis is the sum of the products of mass, *dm*, of each element of the body and the square of *d*, its distance, from the axis, e.g.:
  - $I_{xx} = \int \left(y^2 + z^2 \right) dm$
  - I_{yy} and I_{zz} are defined in an analogous way.



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Inertia

Products of Inertia: I_{xy}, I_{yz}, I_{zx}

- The product of inertia associated with the *x*-axis is:
  - $I_{yz} = \int yz\,dm$
  - I_{xy} and I_{zx} (associated with the *z*- and *y*-axes) are defined in an analogous way.
- Products of inertia are measures of the lack of symmetry in a mass distribution.
- If there are planes of symmetry, then the products of inertia for all axes in those planes are zero.

Principal Axes

- *Principal axes* are sets of orthogonal axes for which all three *products of inertia* are zero.
- There is always *at least* one such set at each point in the body (the point of most interest being the C.o.M.).
- In symmetrical or axisymetrical bodies, the principal axes are intuitively obvious.
- Principal axes are eigenvectors of the *inertia matrix* (see next).

Inertia Matrix

- We define the *inertia matrix* as follows:

$$[I_o] = \begin{bmatrix} I_{xx} & -I_{xy} & -I_{zx} \\ -I_{xy} & I_{yy} & -I_{yz} \\ -I_{zx} & -I_{yz} & I_{zz} \end{bmatrix}$$

  - where $[I_O]$ is referred to some point $O\ (x,y,z)$;
  - Notice that the elements on the diagonal are the *moments of inertia*, whilst those of the diagonal are the negative *products of inertia*.
  - $[I_C]$, the inertia matrix for the C.o.M. has a special significance (see next).

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Angular Momentum of Rigid Bodies

- Thus, in general:

$$\vec{H}_C = \begin{bmatrix} \left(I_{xx}\omega_x - I_{xy}\omega_y - I_{zx}\omega_z\right) \\ \left(I_{yy}\omega_y - I_{yz}\omega_z - I_{zx}\omega_x\right) \\ \left(I_{zz}\omega_z - I_{zx}\omega_x - I_{yz}\omega_y\right) \end{bmatrix}$$

- If *principal axes* are used, this reduces to:

$\Rightarrow \underline{H}_C = (I_{xx}\omega_x, I_{yy}\omega_y, I_{zz}\omega_z)^T$

It is usual to develop the rotational equations for a body by choosing axes in which the inertias are constant, such as axes fixed in the body.

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Angular Momentum of Rigid Bodies

- The angular momentum of a rigid body may be expressed in terms of its angular velocity, $\underline{\omega}$ (measured relative to an inertial frame).
- The resulting equations can then be used to describe the attitude and rotational motion of such a body or systems comprising such bodies.
- The angular momentum, $\underline{H}_C$ of a single *rigid* body about its C.o.M., *C*, is:

$$\Rightarrow \underline{H}_C = [I_C]\,\underline{\omega}$$

where $\underline{\omega} = (\omega_x, \omega_y, \omega_z)^T$

and $[I_C]$ is the inertia matrix at *C*.

Euler′s Equations

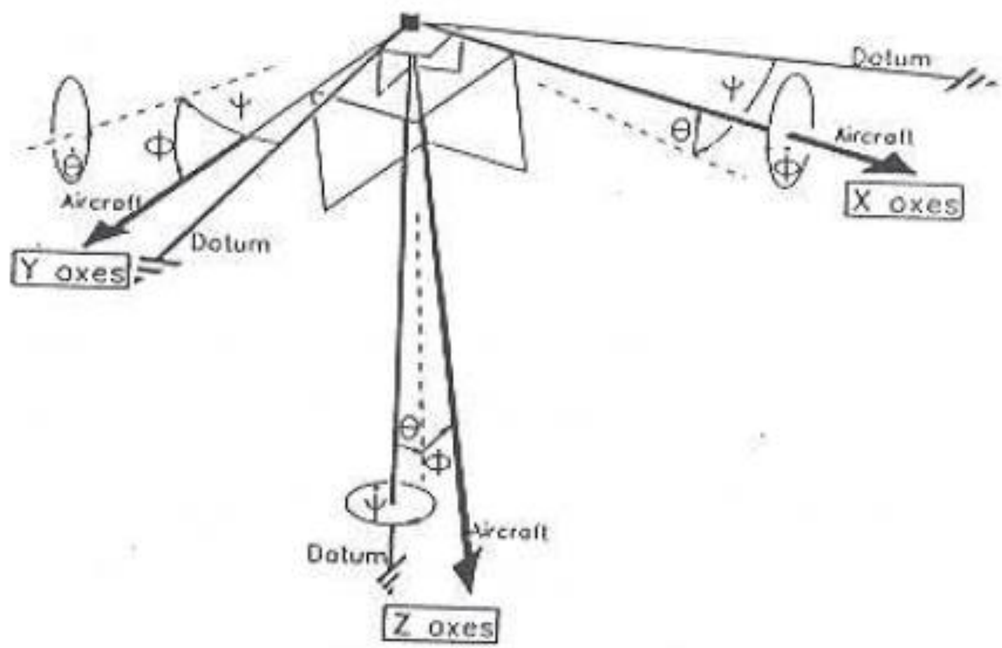
- Thus, *Euler's equations* of motion for the principal body axes of a rigid body are:

$$T_x = I_x\dot{\omega}_x - \left(I_y - I_z\right)\omega_y\omega_z$$
$$T_y = I_y\dot{\omega}_y - (I_z - I_x)\omega_z\omega_x$$
$$T_z = I_z\dot{\omega}_z - (I_x - I_y)\omega_x\omega_y$$

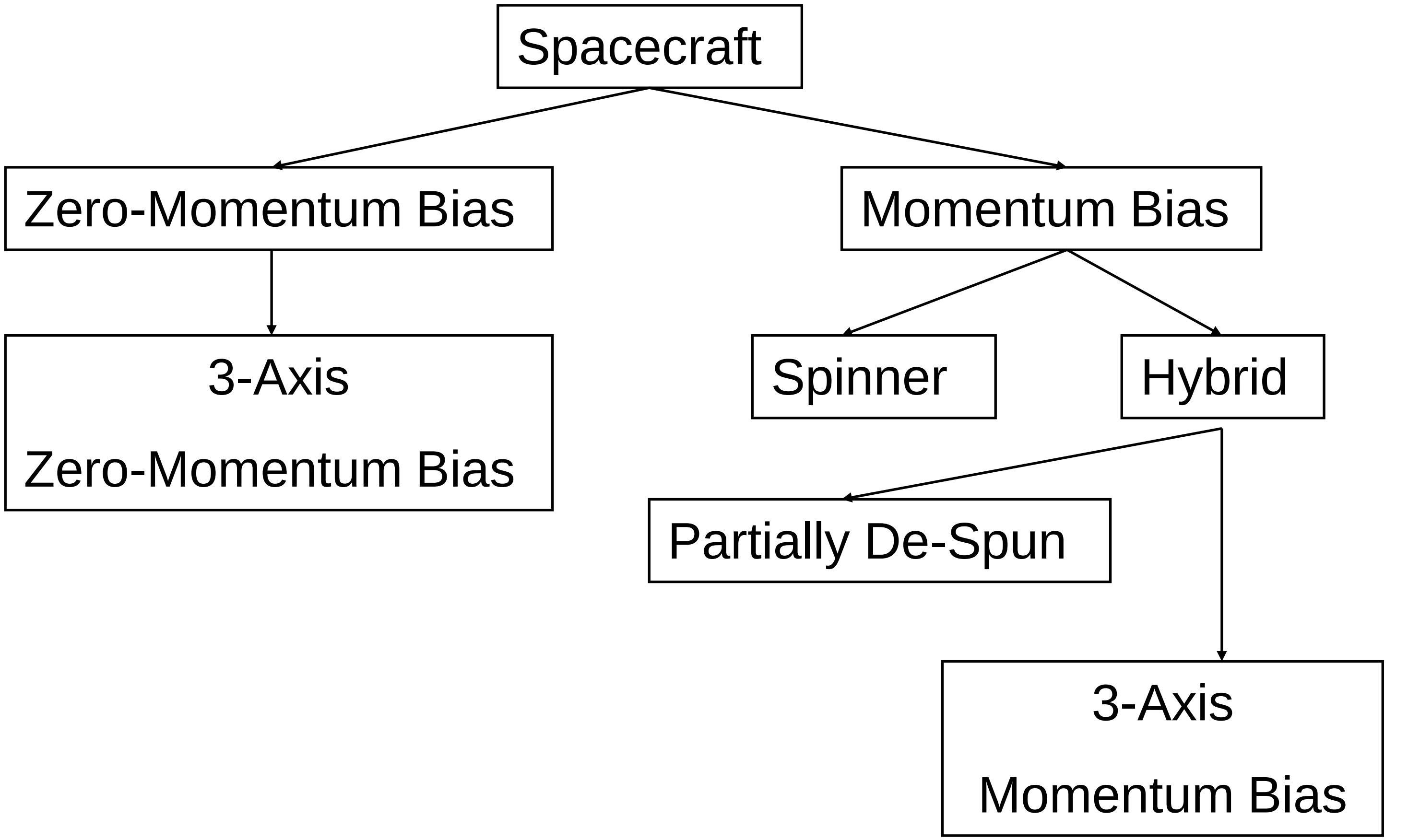
- We now have the means of predicting the *rotational motion* of a spacecraft under the action of external torques.

Roll, Pitch and Yaw

- So far we have use a body-fixed frame of reference and an inertial frame of reference. But there is another system in common use: *roll*, *pitch* and *yaw* where:
  - ***roll axis*** is in the orbit plane along the direction of motion;
  - ***pitch axis*** is perpendicular to the orbit plane (RH set);
  - ***yaw axis*** is the local vertical.
- The *Euler angles* (ϕ, θ, ψ) are measured about the *roll*, *pitch* and *yaw* axes, which are labeled (X, Y, Z) or (1, 2, 3).

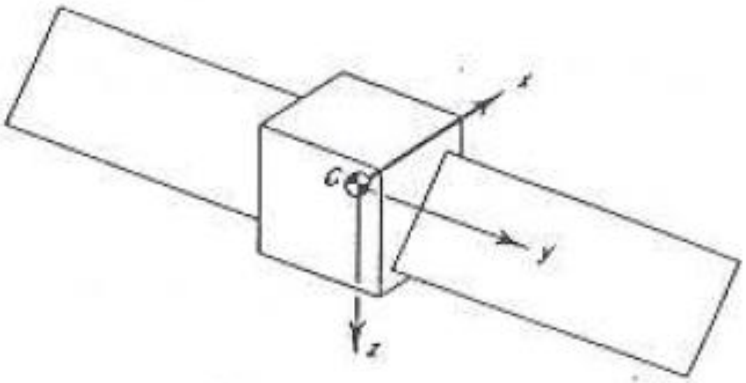


Spacecraft Attitude Control Types



3-Axis Stabilised (Zero Momentum Bias)

- Such spacecraft are usually large, with extensive solar arrays: e.g. typical "Box-shaped" GEO communication satellite.
- Their angular velocity (w.r.t. inertial space) is small, perhaps one revolution per orbit in order to maintain one facet pointing at Earth: e.g. antennas.
- The solar arrays have even less angular velocity as they are maintained Sun-pointing: (from the satellite, the Sun "rotates" once per year w.r.t. inertial space).



3-Axis Stabilised (Zero Momentum Bias)

- The response would be an angular acceleration (from rest), e.g.:

  ➡ $d\omega_x/dt = T_x / I_{xx}$

  Since torquers produce couples, all that is required for such a pure response is that the axes of the torquers are parallel to the principal axes through the C.o.M.

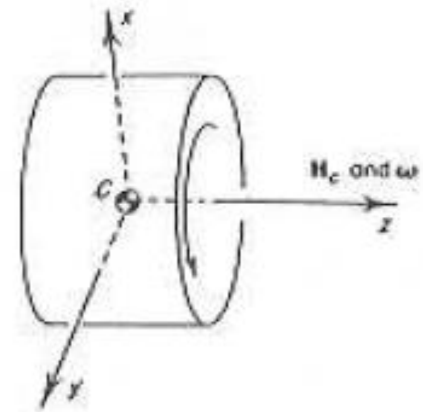
- More generally, when there are *products of inertia*:

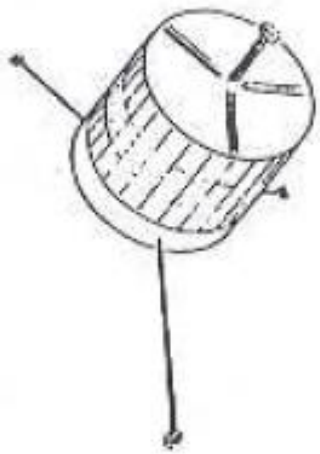
  ➡ $d\underline{\omega}/dt = [I_C]^{-1}\,\underline{T}$

  But in this latter case there will be *cross-coupling* between axes, and so to achieve a pure response ($d\omega_x/dt$ say), potentially all 3 torquers must be used - hence choosing principal axes and configuring the spacecraft torquer axes to be parallel to these, simplifies matters considerably.

Spinning Spacecraft

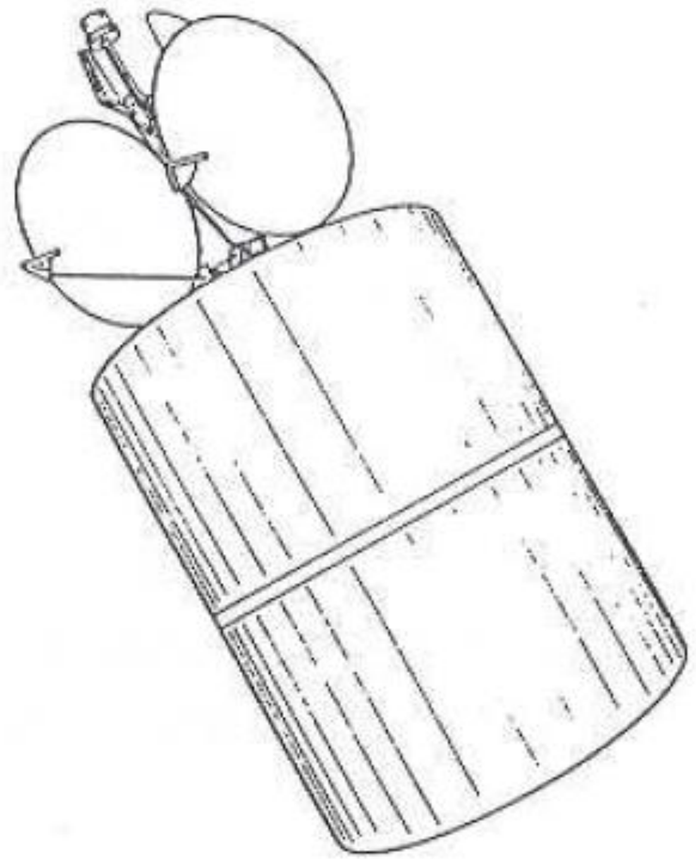
- The behaviour of spinning (or part-spinning) satellites (i.e. those with a significant momentum bias) is complex, but is still described by the *Euler equations*.
- Consider the case where the *whole* spacecraft spins:
- Small angular momentum => minor cross-couplings as already described.
- However, large angular momentum => significant cross-couplings
- Thus, we need to re-examine the motion.





Hybrid Spacecraft

- *Hybrid spacecraft* are a mixture of the two schemes already discussed i.e. they have a "fixed" component and a spinning component.
- An example is a *dual spinner* (such as a Hughes type Comsat).
- Another is a 3-axis controlled satellite with *momentum bias* - i.e. one containing a *momentum wheel*.



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External Torques

- Gas Jets:
  - (Advantages)
    - » Insensitive to altitude;
    - » Suit any orbit;
    - » Can torque about any axis;
  - (Disadvantages)
    - » Requires fuel;
    - » On-Off operation only;
    - » Has a minimum impulse;
    - » Exhaust plume contaminants;
- Magnetic:
  - » No fuel required;
  - » Torque magnitude may be controlled;
  - » No torque about the local field direction;
  - » Torque is altitude & lattitude dependent;
  - » Magnetic interference;
- Gravity Gradient:
  - » No fuel or energy needed;
  - » No torque about local vertical;
  - » Low torque, altitude sensitive;
  - » *Libration* mode requires damping.
- Solar Radiation:
  - » No fuel or energy required;
  - » Very low torque;

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Internal Torques

	(Advantages)	(Disadvantages)
• Reaction Wheel(s):	» Suits any orbit;	» Nonlinearity at zero speed;
• Momentum Wheel(s):	» Provides momentum bias;	» Moving parts;
• Control Moment Gyroscope(s):	» Suitable for 3-axis control;	» Complicated;
	» Provides momentum bias;	» Potential reliability problm;

Attitude Control

Typical ACS

Actuator	Typical Performance Range	Weight (kg)	Power (W)	Suppliers
Thrusters				
• Hot gas (hydrazine)	0.5 to 9,000 N (1)	Variable (2)	N/A (2)	Rocket Research, Hamilton Standard, TRW, Marquardt Walter Kidde, Hughes
• Cold gas	< 5 N (1)	Variable (2)	N/A (2)	
Reaction & momentum wheels	0.4 to 400 N·m·s for momentum wheels at 1200 to 5000 rpm; max torques from 0.01 to 1 N·m	2 to 20	10 to 110	Bendix, GE/RCA, Honeywell
Control moment gyros (CMG)	25 to 500 N·m of torque	> 40	90 to 150	Bendix, Honeywell
Magnetic torquers	10 to 4000 A·m^2 (1 A·m^2 = 1000 pole ·cm) (3)	0.4 to 50	0.6 to 16	Ithaco, Hughes, Lockheed, McDonnell Douglas

(1) Multiply by moment arm (typically 1 m to 2 m) to get torque.

(2) Chapter 17 discusses weight and power for thruster systems in more detail.

(3) For 800-km orbit & maximum Earth field of 0.4 gauss, the maximum torques would be 0.4 x 10^{-3} N·m to 0.16 N·m (see Table 11.6-6)

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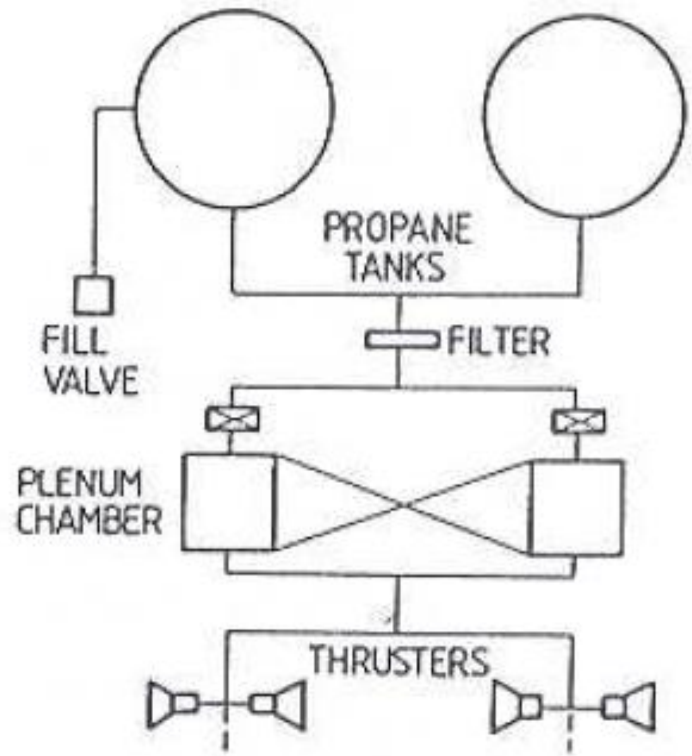
Attitude Control

- Thrusters (Gas Jets) [EXTERNAL TORQUE]
  - Mounted in clusters so as to provide torque about all principal axes.
  - Can share common fuel and control systems with orbit control (station keeping) thrusters - hence "AOCS".
  - Have on-off action => there is a minimum impulse that can be delivered (typically 10^{-4} Ns impulse at 10^{-2} N thrust).
  - If reaction wheels or momentum wheels are the primary method of attitude control, thrusters may be used to "unload" the wheels (i.e. to dump angular momentum).
  - May be cold gas (N_2); hot gas (N_2H_4); bi-propellant ($MMH+N_2O_4$); electric (Xe, NH_3, H_2O resistojet), etc.
  - When the fuel runs out - no more control!

Attitude Control

- Cold Gas Thrusters
  - Uses inert gas (N_2, Ar, Freon, Propane) stored at high pressure;
  - Low I_{sp} (~50 s);
  - Low thrust (~20 mN);
  - Simple;
  - Fine control (~ 0.1°) via min. impulse (~10^{-4} Ns);
  - No plume impingement;
  - Used on Science missions (e.g. EXOSAT).



3-Axis Stabilised (Zero Momentum Bias)

Couple due to a pair of thrusters

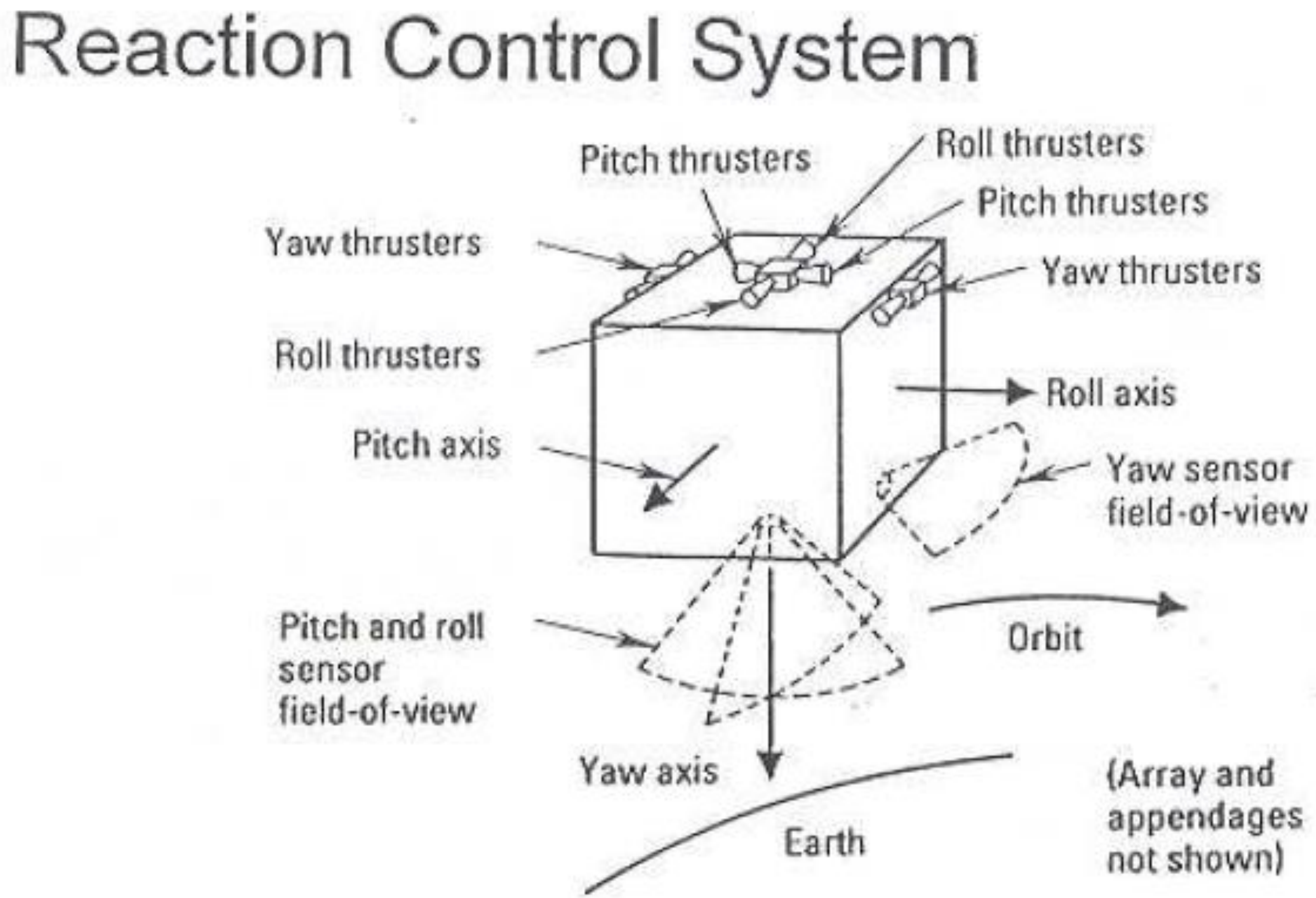
- Specific Impulse, I_{sp}, is the ratio of *effective exhaust velocity* ①, V_e^*, to the gravitational acceleration at the Earth's surface, g_0:

  ➜ $I_{sp} = V_e^* / g_0$ [seconds]

- Thrust (force), $F = V_e^*$ (dm/dt), where (dm/dt) is the rate of consumption of the propellents (= exhaust mass flow rate).
- So for two thrusters each at distance d from the CoM, pointing in opposite directions (so as to form a pure couple), the torque, T is:

  ➜ $T = 2\,F\,d = 2 \,.\, g_0\, I_{sp}\, (dm/dt) \,.\, d$ [Nm]

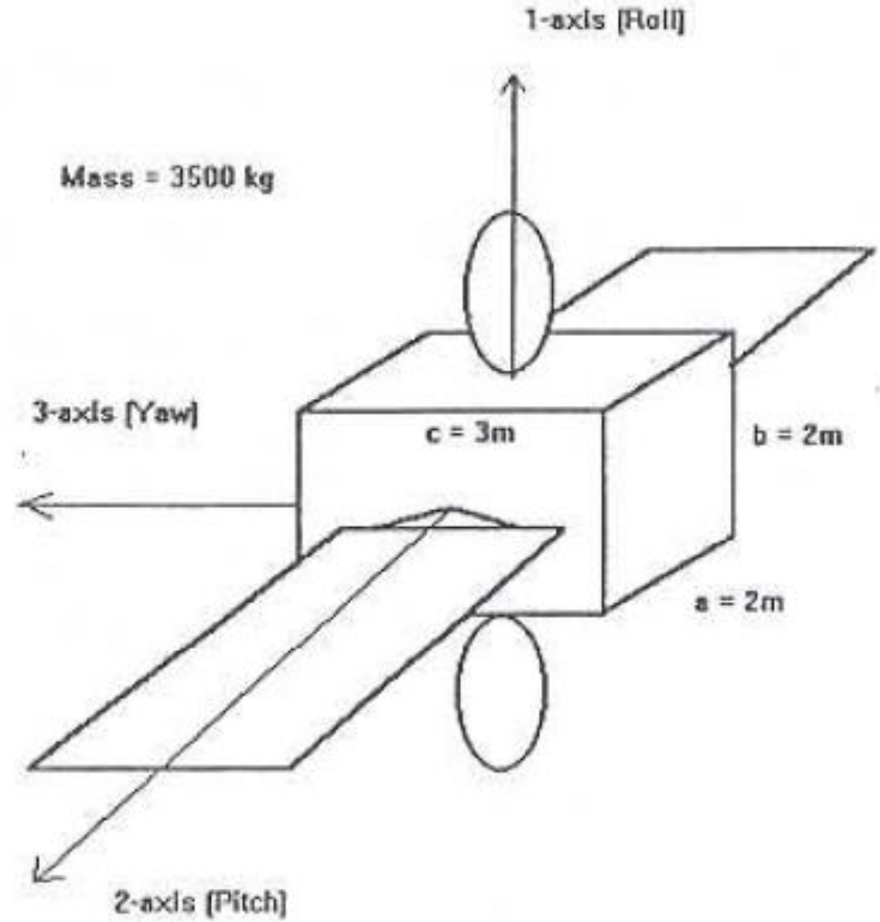
Attitude Control



Attitude Control

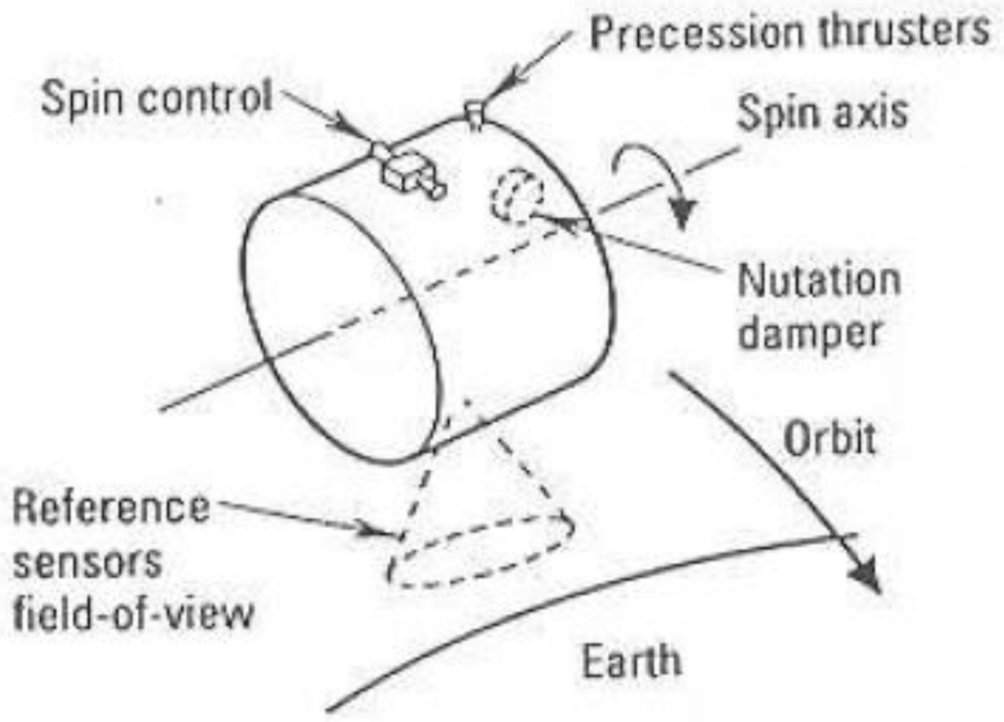
- EXAMPLE (de-stabilising a satellite!):
  Suppose we have a satellite of mass 3500 kg and size 2m x 2m x 3m, equiped with thrusters of 22 N thrust.
  How long will it take to spin the satellite at 1 rpm about the 2-axis?
- ANSWER:
  Estimate $I_{22} = m/12\ (b^2 + c^2) \sim 3800$ Nm
  (Std. Eqn. for M.o.I of a box)
  Assume lever of 1m; thrust = 22N
  Torque, $T = 2dF = 44$ Nm $= I_{22}\ d\omega_2/dt$
  $\Rightarrow d\omega_2/dt = 44 / 3800 = 0.012$ rad s^{-1}
  To get to $\omega_2 = 1$ rpm $= 2\pi / 60$
  Time taken $= \omega_2 / (d\omega_2/dt) = 9$ seconds!

  (small time errors can => big ω errors!)



Attitude Control

Spin



Attitude Control

- Magnetic Torque [EXTERNAL TORQUE]
  - The magnetic field generated by a spacecraft interacts wit the Earths's magnetic field to produce an external couple.
  - If the spacecraft's magnetic dipole moment is $\underline{\mathbf{m}}$, and the external field strength is $\underline{B}$, then the torque is:

$$\underline{T} = \underline{\mathbf{m}} \times \underline{B}$$

  - Electromagnets may be used to deliberately exert torques via this mechanism - e.g. for an "n" turn coil of cross-sectional area "A" carrying current "i", the torque is:

$$\underline{T} = niA\,(\underline{\hat{a}} \times \underline{B})$$

  - where $\underline{\hat{a}}$ is a unit vector normal to the plane of the coil (i.e along the coil's axis).

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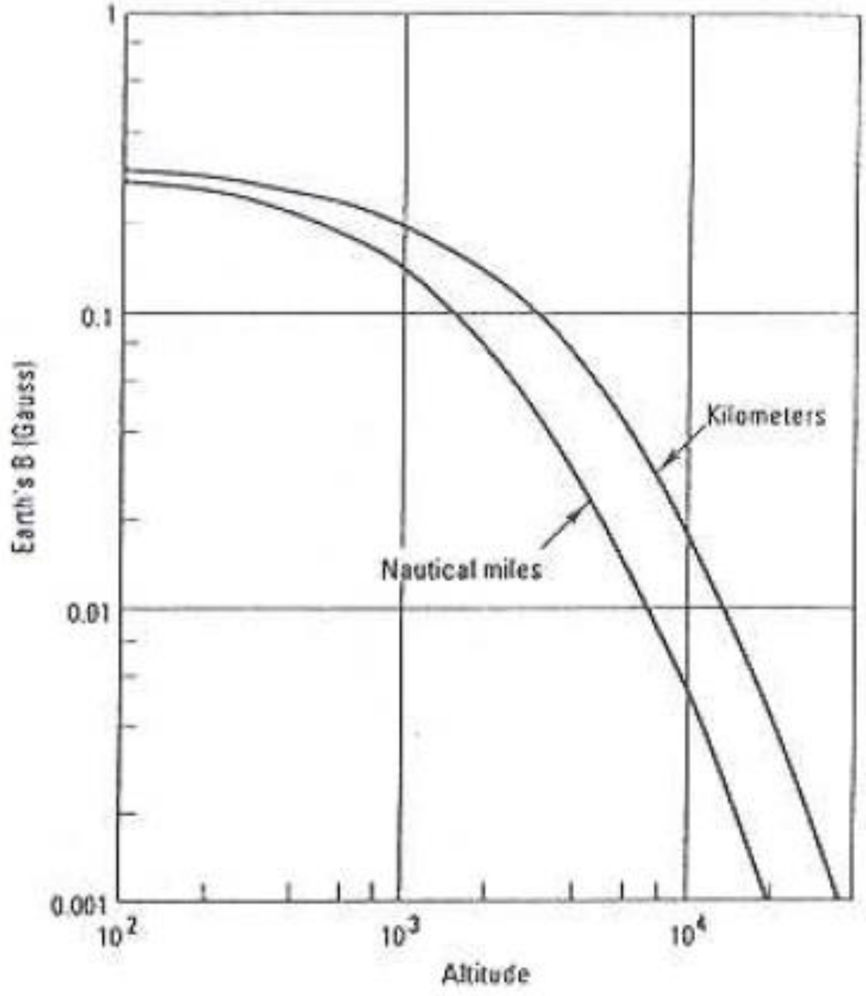
Attitude Control

- Magnetic Torque Cont'd
  - The Earth's magnetic field is (to a first approximation) dipolar, and therefore diminishes with the inverse cube of distance from the centre of the dipole (i.e $\propto 1 / r^3$). Therefore magnetorquers are most effective in LEO where B ~ 40 μT (e.g. UoSAT), but *may* be used all the way out to GEO.
  - In any case the torque available is very small ($< 10^{-1}$ Nm).
  - As the magnetic field varies in strength and direction around the orbit, spacecraft which use magnetorquers often carry *magnetometers* to measure the magnetic field.
  - There is *no* torque component about the local field direction.
  - Maximum torque is about an axis orthogonal to the local field direction.

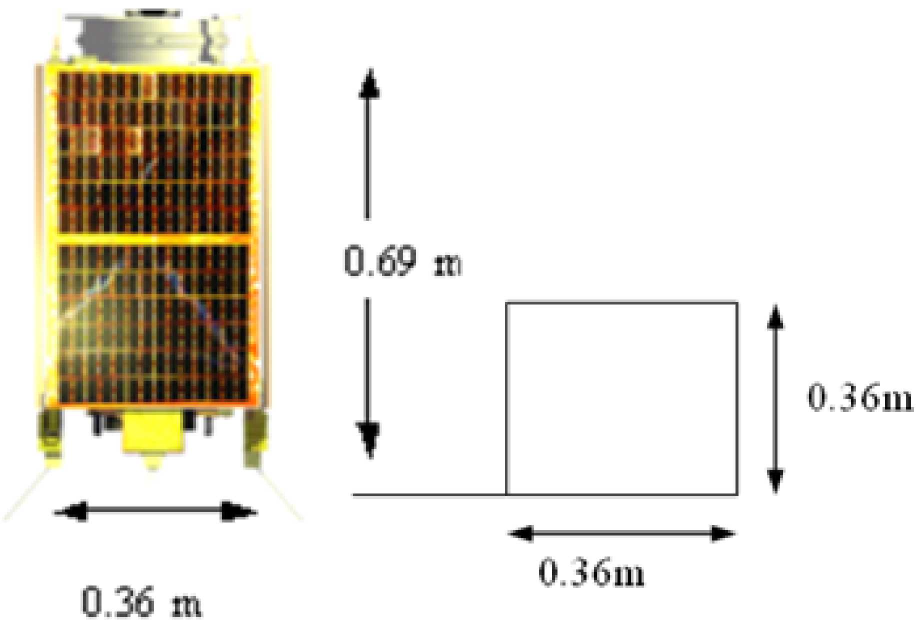
Attitude Control

Earth's Magnetic Field Strength



Example-Problem

Calculate the torque, angular momentum and acceleration for a rest-to-rest single axis manoeuvre of 90 degrees to be completed in 180 seconds. Use $I_{xx}=I_{yy}=I_{zz}=2.5$ kgm^2. Draw plots of the profiles of the attitude angle, angular rate and torque.



Side View

Top View

μ	3.9×10^{14} m^{3}s^{-2}
r	700 km
C_D	2
ρ_{ave}	10^{-14} gm^{-3}
C_{pa}-C_g	0.1 m

Table 1