PH547 Space Mission Analysis Primer

Students taking PH547 Space Mission Analysis have different academic backgrounds and possess varying degrees of knowledge in orbital mechanics, the space environment, and space systems. This document provides fundamental information that will be covered in more detail during the course.

This primer contains selected lecture notes from the PHE 255: Introduction to Space Science course offered at the Royal Military College of Canada (Fall Term 2017).

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PHE 255
Introduction to Space Science

Module 4: Space Environment
Part 1: The Sun

- Introduction
- Sun Interior
- Sun Atmosphere
- Solar Activity

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Introduction — The Big Picture

- There is an estimated 10^{11} to 10^{12} galaxies in the Universe, each containing billions of stars
- Our galaxy, the Milky Way, has a disk of about 160,000 light years in diameter and 2,000 light years thick
  - Milky Way speed is ~300 km/s
- There are about 200 billion stars in the Milky Way
  - Our sun is an average star
- The sun orbits around the center of the Galaxy at ~220 km/s
  - It takes about 220 million years to complete one orbit
- Earth orbits the sun at ~ 30 km/s

Depiction of the Milky Way

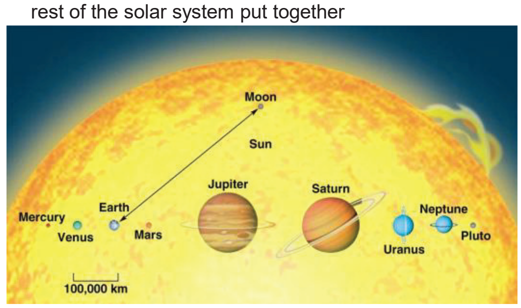

Edge on view of a spiral galaxy

Introduction — Sun Parameters

- The sun has the biggest effect on the space environment
  - Distance from Earth
    - ✓ Mean = 149,598,000 km = 1 Astronomical Unit (1 AU)
    - ✓ Maximum = 152,000,000 km (Aphelion)
    - ✓ Minimum = 147,000,000 km (Perihelion)
  - Physical Parameters
    - ✓ Radius: 696,000 km (109 Earth radii)
    - ✓ Mass: 2.0×10^{30} kg (3.3×10^5 Earth masses)
    - ✓ Mean Density: 1410 kg/m^3 (Earth = 5540 kg/m^3)
    - ✓ Mean Temperature: 5800 K (surface), 1.6×10^7 K (core)
      - ➢ Note: 0 Kelvin (K) is equal to -273.15 ° C
    - ✓ Composition by Mass: 74% Hydrogen, 25 % Helium

Introduction — Size of Sun

- The sun is by far the largest object in our solar system
  - Contains about a thousand times more mass than the rest of the solar system put together



Relative sizes of the planets with respect to the sun

Sun Interior

- The radiant power of the sun, or *luminosity,* is 3.9×10^{26} Watts
  - The solar constant, or *solar flux*, at the Earth is approximately 1368 W/m^2 on average
- Sun has been burning for 4.56 billion years
  - This length of sustained energy output cannot be explained by:
    - ✓ Kelvin-Helmholtz contraction
    - ✓ Chemical reactions
- Einstein discovered the key to this mystery in 1905
  - Special Theory of Relativity

Sun Interior

- Hydrogen fusion $\boxed{\text{4 Hydrogen} \rightarrow \text{1 He + energy}}$

Mass of 4 Hydrogen atoms = 6.693×10^{-27} kg
Mass of 1 Helium atom = 6.645×10^{-27} kg
Mass loss = 0.048×10^{-27} kg

$$E = mc^2$$

$$E = (0.048 \times 10^{-27}\,\text{kg})(3 \times 10^8\,\text{m/s})^2$$

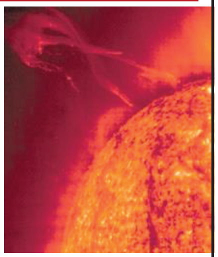
$$E = 4.3 \times 10^{-12}\ \text{Joules}$$

Amount of energy released by the formation of a single helium atom

- When 1 kilogram of Hydrogen is converted to helium, 6.3×10^{14} Joules are produced
  - This is equivalent to burning 20,000 metric tons of coal

Sun Interior

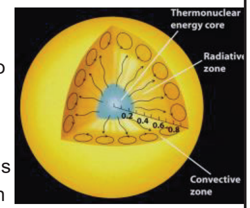
- The sun converts 600 million metric tons of hydrogen into helium every second
  - The sun's core has enough hydrogen to continue at this rate for another 6 billion years
- Thermonuclear fusion occurs only at very high temperatures
  - Hydrogen fusion occurs at temperatures of 1.6×10^7 K



- In the sun, fusion occurs only in the dense, hot core
  - Byproducts of the fusion process include *neutrinos, positrons* and *gamma-ray photons*
    - ✓ 1×10^{14} neutrinos should pass through every square meter of the Earth each second

Sun Interior — Zones

- The sun's interior consists of three zones
  - Thermonuclear Core: Hydrogen fusion takes place in a core extending from the sun's center to about 0.25 solar radius
  - Radiative Zone: The core is surrounded by a radiative zone extending to about 0.7 solar radius
    - ✓ Energy travels outward through radiative diffusion

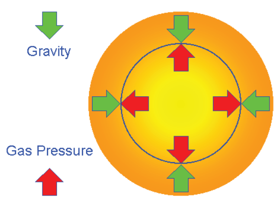


  - Convection Zone: The radiative zone is surrounded by an opaque convective zone of gas at relatively low temperature and pressure
    - ✓ Energy travels outward primarily through convection

Sun Interior

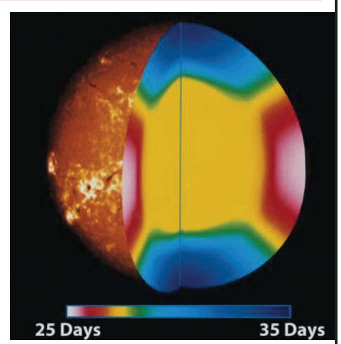
- The sun is in thermal equilibrium
  - The core is neither heating up or cooling down
  - Heat generated in the core is radiated outward and eventually into space
- The sun is also in hydrostatic equilibrium
  - It is neither expanding nor contracting

The internal pressure balances the weight of the material above it (gravity)



Sun Interior — Differential Rotation

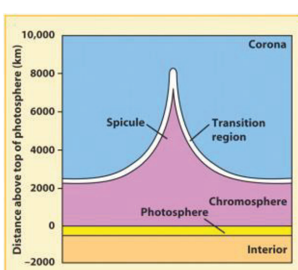
- The sun does not rotate like a solid disc
  - Rotation changes with depth and latitude
  - Faster rotation at the equator than the poles
  - Faster rotation towards the surface
  - The radiative zone and core seem to rotate like a rigid sphere



Rotation of the Solar Interior

Sun Atmosphere

- The sun's atmosphere has three main layers
  - Photosphere (bottom layer)
    - ✓ Considered the surface of the sun
    - ✓ ~ 400 km thick
    - ✓ Opaque
  - Chromosphere (mid layer)
    - ✓ ~ 2000 km thick
    - ✓ Contains spicules, which are jets of rising gas
  - Corona (top layer)
    - ✓ Extends millions of kilometers



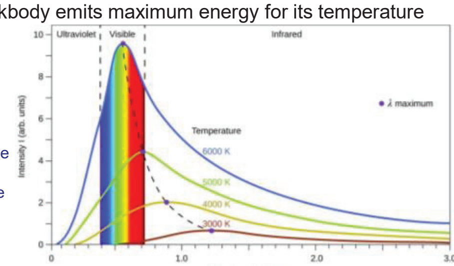
Solar Atmosphere

Sun Atmosphere — Photosphere

- The photosphere, which is the lowest part of the solar atmosphere, is the visible part of the sun
  - The spectrum of the photosphere is similar to that of a blackbody at a temperature of ~5800 K
    - ✓ A blackbody emits maximum energy for its temperature

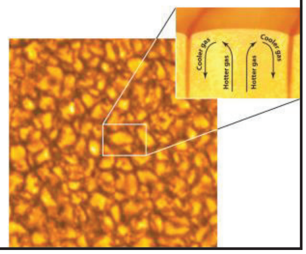
Black Body Curves (3000 to 6000 K)

Peak energy is the visible range for the surface of the sun (yellow/orange)



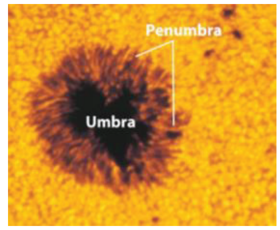
Sun Atmosphere — Granules

- High resolution photographs of the photosphere reveals a blotchy pattern known as granulation
- Granules are convection cells about 1000 km wide in the photosphere
  - Convection transports heat from the sun's radiative zone to the solar atmosphere
- Supergranules are larger convection cells super-imposed on the granules
  - 35,000 km in diameter
  - Spicules occur at the boundaries of supergranules

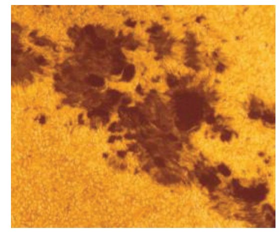


Sun Atmosphere — Sunspots

- Sunspots are low-temperature regions in the photosphere
  - Often appear in groups, usually lasting about two months
  - Typically a sunspot is ~ 20,000 km across



Penumbra: 5500 K Umbra: 4300 K

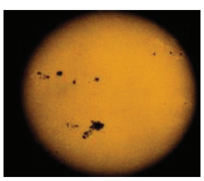


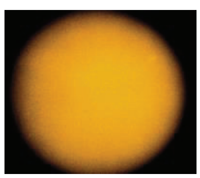
Sunspot Groups

Sun Atmosphere — Sunspot Cycle

- The number of sunspots varies with a period of about 11 years
  - This is known as the sunspot cycle
- The average number of sunspots increases and decreases in a regular cycle of approximately 11 years, with reversed magnetic polarities from one 11-year cycle to the next
- Two such cycles make up the 22-year solar cycle

Sunspot maximum

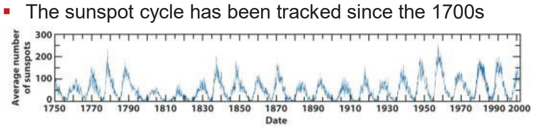




Sunspot minimum

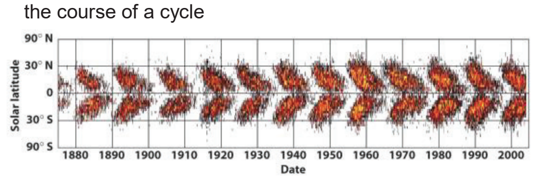
Sun Atmosphere — Sunspot Cycle

- The sunspot cycle has been tracked since the 1700s



Sunspots (1750 to 2000)

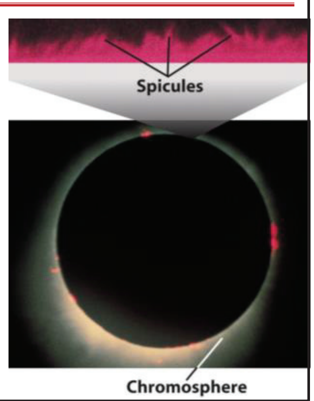
- Sunspots form closer and closer to the sun's equator during the course of a cycle



Graph showing sunspot migration toward equator during sunspot cycles

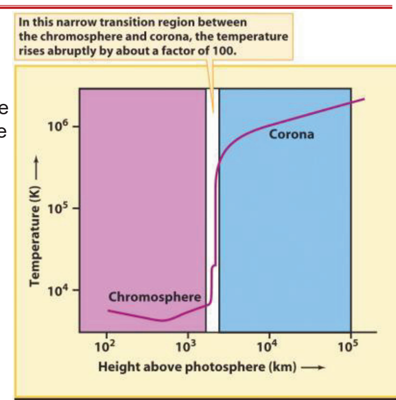
Sun Atmosphere — Chromosphere

- The chromosphere is a layer of less dense but higher temperature gases above the photosphere
- Jets of rising gas called spicules extend upward from the photosphere into the chromosphere
  - Spicules last about 15 minutes
  - 300,000 spicules exist at any one time
  - 1% of the sun's surface



Sun Atmosphere — Corona

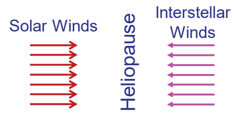
- The outermost layer of the solar atmosphere, the corona, is made of very high-temperature gases at extremely low density



Sun Atmosphere — Corona

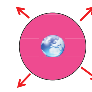
- The corona ejects mass into space to form the solar wind
  - The solar wind is a stream of charged particles, or plasma, ejected from the upper atmosphere of the sun
  - It consists mostly of electrons and protons with energies of about 1 keV
  - Speed is ~400 km/s
  - Solar wind is felt out to 110 to 170 AUs (AU = 150 million km)
    - ✓ Out to the heliopause


Sun's corona during a solar eclipse



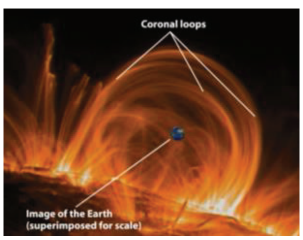
Solar Activity — Relationship to Sunspots

- There is an increase in solar activity with increasing number of sunspots
  - Higher amounts of solar radiation interacting with the Earth's outer atmosphere
- This increased solar radiation has two main effects on spacecraft:
  - Expansion of the thermosphere, leading to increased drag on satellites in low Earth orbit
  - Increased wear on satellite component such as electronics and outer surfaces
- Solar events such as solar flares and coronal mass ejections (see following slides) are commonly associated with sunspot activity

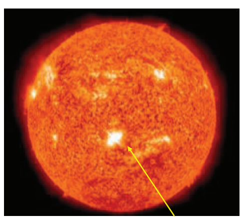

Thermosphere expansion

Solar Activity — Solar Flare

- A solar flare is a brief eruption of hot, ionized gases from a sunspot group
  - The gases may become trapped in magnetic field loops
    - ✓ Magnetic storm related to sunspot fields
  - Solar flares can lead to coronal mass ejections (next slide)

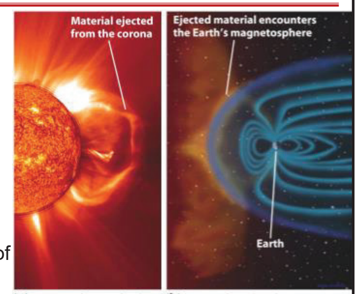


Magnetic field loops


13 Feb 2011, sunspot 1158 unleashed a strong solar flare

Solar Activity — Coronal Mass Ejection

- Coronal mass ejections cause billions of tons of coronal gas to be blasted into space
- If pointed at Earth, they may interfere with satellites and electronic equipment on Earth
- Earth's magnetosphere protects us from much of the material ejected by coronal mass ejections

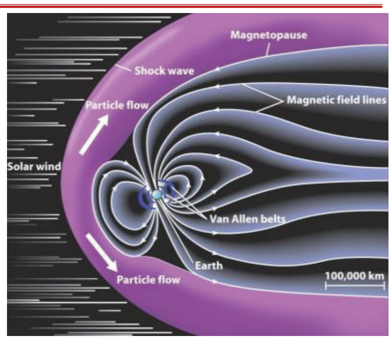


(a) A coronal mass ejection (b) Two to four days later

The magnetosphere is the region of space surrounding Earth where the dominant magnetic field is the magnetic field of Earth, rather than the magnetic field of interplanetary space. The magnetosphere is formed by the interaction of the solar wind with Earth's magnetic field.

Solar Activity — Van Allen Belts

- Most of the particles of the solar wind are deflected around the Earth by the magnetosphere
- Some of the charged particles leak through the magnetopause and become trapped in the Earth's magnetic field to form the Van Allen Belts

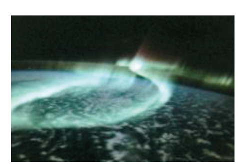


Dangerous for satellites and space travel

Solar Activity — Aurora

- An increased flow of charged particles from the Sun can overload the Van Allen belts and cascade toward the Earth, producing aurora known as the northern lights and the southern lights
- The high speed particles cascade into the upper atmosphere and excite the atoms to higher energy states
  - The atoms emit visible light when they return to their original states


Alaska


Antarctica

PHE 255
Introduction to Space Science

Module 4: Space Environment
Part 2: Effects on Spacecraft

- Gravity
- Atmosphere
- Vacuum
- Space Debris
- Radiation
- Charged Particles

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Gravity

- Satellites orbiting the Earth are **not** in zero gravity
- Force of gravity decreases as function of the distance form the center of the Earth where:

$$F_g = G\frac{m_1 m_2}{r^2} \qquad G = 6.67 \times 10^{-11}\ \mathrm{N \cdot m^2 \cdot kg^{-2}}$$

m_1 = Mass of Earth = 6×10^{24} kg m_2 = Mass of satellite
r = Distance to the center of the Earth where Earth radius is 6378 km

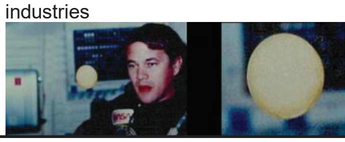
- Example: How much gravity is present at an altitude of 400 km (Low Earth Orbit – e.g. International Space Station)

$$\frac{F_{g\_300km} = 6.67\times10^{-11}\dfrac{6\times10^{24} m_2}{(6778)^2}}{F_{g\_Surface} = 6.67\times10^{-11}\dfrac{6\times10^{24} m_2}{(6378)^2}} = \frac{(6378)^2}{(6778)^2} = 0.885$$

88.5% of Earth's gravity is still present

Gravity — Microgravity

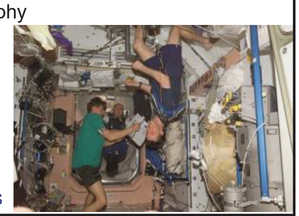
- The spacecraft and everything in it is in free fall
  - Also known as microgravity
- Microgravity environment of space offers a unique environment to perform research for new products and technologies
  - No sedimentation/stratification
  - No hydrostatic pressure
  - Reduced contact with vessel walls
- Microgravity has been a major area of research in space
  - May revolutionize traditional Earth-bound processing methods, especially in the biomedical and drug development industries



Astronaut Joe Allen with his morning orange juice on an early Shuttle mission

Gravity — Microgravity

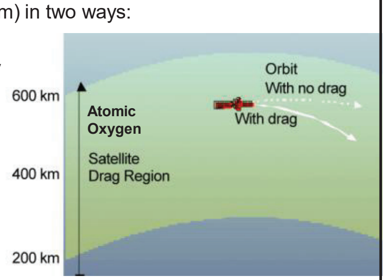
- Microgravity has no direct effect on the spacecraft, but the speed of free-fall causes problems with atmospheric drag and space debris, both of which will be discussed later
- Human exposure to microgravity for extended periods can have hazardous effects on human health
  - In response to an extended period of weightlessness, various physiological systems begin to change while unused muscles may atrophy
    - ✓ This subject will be discussed in the next lecture, Living and Working in Space



Russian, Dutch and American astronauts in the Unity Node of the ISS

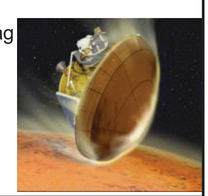
Atmosphere

- The Earth's atmosphere does not end abruptly
  - Extends hundreds of kilometres above the surface
- The atmosphere affects a spacecraft in low Earth orbit (below approximately 600 km) in two ways:
  - Drag: shortens orbital lifetimes by slowing the satellite orbital speed
  - Atomic Oxygen: degrades spacecraft surfaces



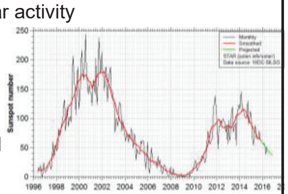
Atmosphere — Drag

- Atmospheric density indicates how many molecules are packed into a specific volume
  - Molecular density decreases dramatically with altitude:
    - ✓ 100 km: 1×10^{20} molecules/m^3
    - ✓ 300 km: 1.4×10^{9} molecules/m^3
- Despite the decrease in atmospheric density, satellites are effected by drag below altitudes of approximately 600 km (limit of the thermosphere)
  - Lower altitude = More atmospheric drag
- The amount of drag depends on:
  - Atmospheric density
  - Spacecraft parameters
    - ✓ Size, shape, speed and orientation to air flow



Atmosphere — Drag and Sunspot Cycle

- Drag and Sunspot Cycle:
  - As solar activity increases ultraviolet radiation heats the thermosphere causing it to expand
  - Leads to increased drag of satellites in LEO due to higher atmospheric density
    - ✓ Few sunspots = low solar activity
    - ✓ Many sunspots = high solar activity
- Example for a satellite at 400 km altitude:
  - During solar minimum orbital lifetime is about 552 days
  - During solar maximum orbital lifetime is about 77 days



Recent and projected sunspot numbers

Atmosphere — Atomic Oxygen

- Atomic Oxygen
  - At sea-level air is composed of 78% nitrogen, 21% oxygen and 1% of other gases
    - ✓ Normally oxygen is in the form of O_2 at sea level
  - Radiation and charged particles cause O_2 to split apart in the upper atmosphere
    - ✓ Left with atomic oxygen, O
  - Atomic oxygen causes a breakdown of satellite surfaces
    - ✓ Similar process as rusting
  - Atomic oxygen chemical reactions can produce significant amounts of background radiation

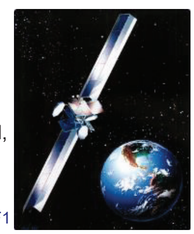

Shuttle Glow (red arrows) on the space shuttle due to atomic oxygen reactions

Vacuum

- Atmospheric density decreases dramatically with altitude
  - At 1000 km there is about a trillion times less air than at the surface
    - ✓ However, there is still about a million particles per cubic centimeter
    - ✓ A pure vacuum is nearly unobtainable
      - ➢ Space is called a *near vacuum* or *hard vacuum*
- The near vacuum of space causes three potential problems:
  - Outgassing
  - Cold welding
  - Heat Transfer

Vacuum — Outgassing

- Outgassing
  - Some materials such as plastics or composites may release trapped gasses when exposed to a vacuum
    - ✓ Mass is loss in the process
    - ✓ Process will increase with higher temperatures
  - The material's molecules may coat delicate sensors or cause electronic components to arc
  - Example: The solar panels of Anik F1 degraded more rapidly than expected, likely due to outgassing


Anik F1

Vacuum — Cold Welding

- Cold Welding
  - Occurs between mechanical parts that have very little separation between them
  - In the vacuum of space these parts may be come stuck, essentially welding together
  - Controllers try various techniques to 'unstick' the two parts
    - ✓ May expose one part to the sun and the other to shade so that differential heating may cause the parts to separate
  - Designers try to avoid moving parts as much as possible to prevent cold welding
  - Dissimilar materials will not cold weld

Vacuum — Heat Transfer

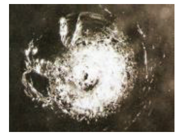
- Heat Transfer
  - The space vacuum environment creates a problem with heat transfer since there is no atmosphere to distribute heat
  - Spacecraft is very cold in eclipse (no sun) and very hot when in sunlight
  - Parts of the spacecraft not in direct sunlight may be cold while exposed areas are hot
  - Heat may be transferred through conduction, convection and radiation
    - ✓ Conduction and convection require a medium and can be used within the spacecraft for heat transfer
    - ✓ Radiation is the primary method for moving heat into and out of a spacecraft

Space Debris

- The risk of a satellite colliding with a natural object is remote
- A larger problem is the accumulation of manmade space debris, or *space junk*
  - With nearly every mission broken spacecraft parts, booster segments and other assorted pieces are left behind, posing serious hazards
- Objects that are sufficiently large are tracked by the US Space Surveillance Network (SSN)
  - Mix of radars and optical sensors
- SSN can track objects > 5 to10 cm in LEO and > 1 m in GEO
  - Objects < 5 cm cannot be tracked
- SSN tracks more than 17,000 objects, most it space junk
- There are also millions of pieces of space debris that are too small to be tracked

Space Debris

- Debris between 1 mm and 1 cm can damage a satellite due to the high speeds of orbiting objects
  - Shielding increases the cost of a satellite so most have minimal protection against collision with debris



In 1983 a paint flake 0.2 mm in diameter hit the Challenger wind-shield, making a gouge 4 mm wide

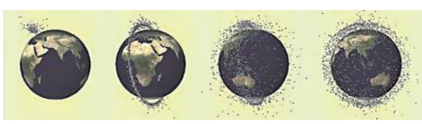
- Debris is not uniformly distributed in space, but is concentrated in regions that are heavily used by satellites
  - Debris is concentrated at GEO (35,800 km altitude) and in LEO (< ~2000 km altitude)

Space Debris

- The greatest risk for collision occurs in LEO
- Collision with an object >1 cm expected every 3 to 4 years
  - Risk will become more severe with increasing space junk
- The Chinese destruction of their own satellite (Feng Yun 1C) in 2007 increased the risk of collision considerably at ~ 800 km
- In 2009 Cosmos-2251 collided with Iridium-33
  - Both satellites were destroyed, creating large debris clouds
- After a satellite break-up the debris disperses

Dispersion of space junk after satellite break-up

Spreading of the pieces is due to different orbital speeds after satellite break-up



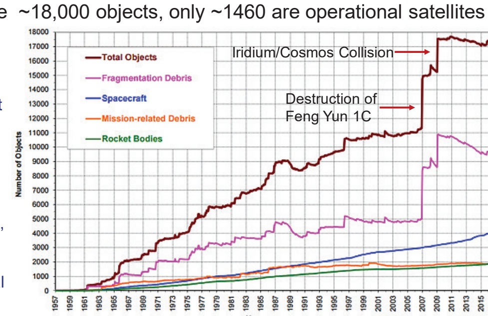
15 min 10 days 6 months 3 years

Space Debris

- The chart below shows the number of space objects tracked by the SSN by year
- Of the ~18,000 objects, only ~1460 are operational satellites

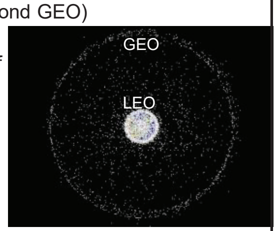
Objects tracked in Earth orbit by type and year

Of the 4,000 spacecraft, about 1,460 are operational



Space Debris

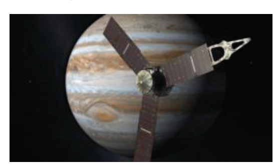
- There are currently no plans to clean up space junk
  - Some international agreements aim at decreasing the rate at which it is accumulating
    - ✓ Require operators in GEO to boost worn out pieces into graveyard orbits (beyond GEO)
    - ✓ Satellites in LEO must de-orbit within 25 years of mission completion
- Developing international measures to prohibit the use of kinetic energy antisatellite (ASAT) weapons should become a priority
  - Currently there are no legal restrictions



Distribution of space debris looking downward at the North Pole

Radiation

- Visible light hitting the spacecraft can be converted to electric energy using photovoltaic cells, or *solar cells*
  - Cheap, reliable and abundant source of power



Solar Panels on Juno

- The radiation can also cause problems for the spacecraft
  - Heating on exposed surfaces
  - Degradation or damage to surfaces and electronic components
  - Solar radiation pressure (SRP)

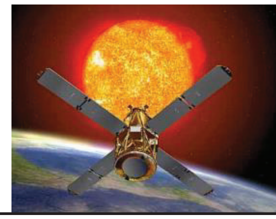
Radiation — Heating

- Heating on Exposed Surfaces
  - Infrared or thermal radiation can be harmful if electronics in the satellite are overheated
    - ✓ Most electronic devices and batteries have a relatively narrow temperature operating range
    - ✓ Thermal control system has to moderate the temperature
      - ➢ Must design the satellite's thermal control system to accommodate the most temperature sensitive piece of equipment

Satellite thermal and vacuum test chamber

Radiation — Degradation

- Degradation or damage
  - Long exposure to ultraviolet (UV) radiation can degrade spacecraft coatings
    - ✓ UV can also damage solar cells
  - During solar flares, bursts of radiation in the radio regime can interfere with satellite communications

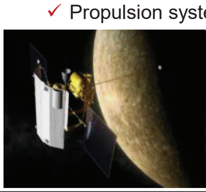


Satellite sensors can be scrambled by excessive 'noise' from solar flares

GPS and satellite television can be hampered by increased solar activity

Radiation — Solar Radiation Pressure

- Solar Radiation Pressure (SRP)
  - Photons can disturb the orientation of the spacecraft, causing it to point in the wrong direction
    - ✓ Attitude control system required to orient the spacecraft correctly
  - Photons can also change the orbital parameters
    - ✓ Satellites in GEO are particularly vulnerable
    - ✓ Propulsion system required to correct the orbit



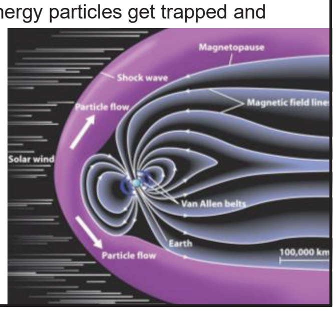
For the Mercury Messenger mission, controllers creatively used solar radiation pressure instead of traditional thrusters to help control trajectory

Charged Particles

- Perhaps the most dangerous aspect of the space environment is the influence of charged particles
- Sources for these particles include:
  - Galactic Cosmic Rays (GCRs)
    - ✓ Charged particles that originate outside the solar system
    - ✓ Extremely energetic
  - Solar wind and solar flares
    - ✓ The solar wind is a stream of charged particles that are ejected from the upper atmosphere of the sun
      - ➢ Solar flares and sunspots increase the number of charged particles emanating from the sun
    - ✓ Van Allen Belts form due to the trapping of charged particles from the solar wind

Charged Particles — Van Allen Belts

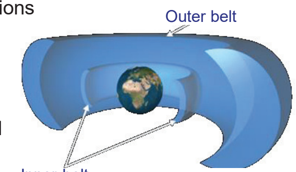
- The Earth's magnetic field produces a magnetosphere that blocks the solar wind from hitting the atmosphere
- As the solar wind interacts with Earth's magnetic field and atmosphere, some high-energy particles get trapped and become concentrated between field lines
  - These areas of are the Van Allen belts, which contain charged particles that can damage spacecraft



Interaction between solar wind and the Earth's magnetic field

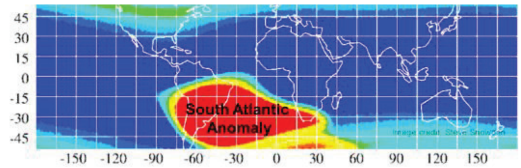
Charged Particles — Outer and Inner Belt

- The outer Van Allen radiation belt extends from 13,000 to 60,0000 km above the surface of the Earth, but the greatest intensity is at 25,000 to 31,000 km
  - Consists mainly of high energy electrons with energies of 0.1 to 10 MeV
- The inner Van Allen Belt radiation belt extends from 1,000 to 7,000 km above the surface of the Earth
  - Contains high concentrations energetic protons with energies exceeding 100 MeV
- Outside the belts there is still high levels of charged particles compared to the surface of the Earth



Charged Particles — South Atlantic Anomaly

- The inner Van Allen Belt radiation belt dips down as low as 200 km off the lower coast of Brazil, creating a phenomenon known as the South Atlantic Anomaly
  - Weakest part of the magnetic field
  - A satellite in a typical low Earth orbit remains safely below the inner belt, except at the South Atlantic Anomaly
- Mission planners must be aware of this anomaly and avoid if possible

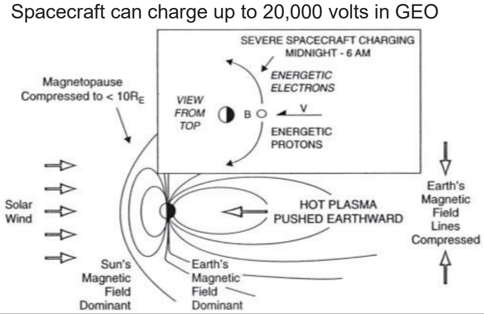


Charged Particles

- While the Earth's magnetic field protects us from charged particles on the surface, spacecraft orbiting the Earth encounter problems such as: Charging, Sputtering and Total Dose Effects
- Charging
  - Charging occurs when electrical charges build up on different parts of the spacecraft as it moves through concentrated areas of charged particles
  - Once this charge builds up, discharge can:
    - ✓ Damage surface coatings, degrade solar panels, cause loss of power
    - ✓ Switch off electronics → may need to restart system
    - ✓ Permanently damage electronics
  - Charged particles can also give spurious readings to sensors and cameras

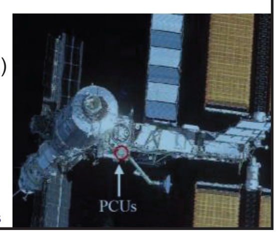
Charged Particles — Charging

- Severe spacecraft charging occurs in the geostationary belt when the Earth's magnetic field compresses and injects hot plasma toward the 'night side' of Earth
  - Spacecraft can charge up to 20,000 volts in GEO



Charged Particles — Charging

- To reduce the effects of spacecraft charging:
  - Make the surface conductivity as uniform as possible
  - Use electronics resistant to electrostatic discharge
  - Use active current balance methods such as a plasma contactor
    - ✓ Discharges accumulated electrons
      - ➢ ISS uses Plasma Contactor Units (PCUs)



Location of ISS PCUs

Charged Particles — Sputtering and Total Dose

- Sputtering
  - Refers to the damage done by high speed particles to the spacecraft surface
  - Analogous to sandblasting
  - Over the long term it can damage a spacecraft's thermal coating and sensors
- Total Dose Effects
  - Long term damage by high energy protons and electrons to the crystal structure of semi-conductors
  - Efficiency of the material is degraded over time, causing computer problems
  - Can be mitigated by shielding computer components and using redundant computer code

Charged Particles — Protection

- Low Earth orbits near the equatorial plane are somewhat protected by the Earth's magnetic field
- Polar orbits have increased exposure to charged particles
- Highly elliptic orbits are exposed to the Van Allen Belt belts on every orbit if the perigee low enough
- Geosynchronous / geostationary and higher orbits are less protected by the Earth's magnetic field
- Protection from high speed particles include:
  - Passive bulk shielding
    - ✓ Aluminum, H_20 and polyethylene are good shielding material
  - Magnetic shielding can deflect particles with kinetic energies below a cut-off value
    - ✓ High energy particles such as GCRs are not deflected

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Introduction to Space Science

Module 4: Space Environment
Part 3: Living and Working in Space

- ➢ Introduction
- Freefall
- Radiation
- Psychological Effects

Dr. Ron Vincent,
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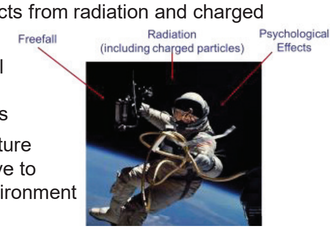
Introduction

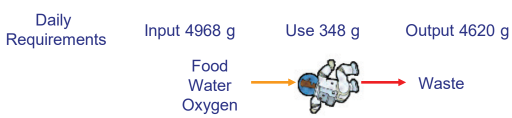
- Humans have evolved to deal with Earth's environment
  - We have bones, muscles and connective tissue to support ourselves against the pull of gravity
  - The ozone protects us from UV rays while the magnetosphere protects from radiation and charged particles
    - ✓ We have no natural biological defence against these things
    - ✓ When humans venture into space they have to adapt to a new environment
- Also have to consider psychological effects



The three main issues with living and working in space

Introduction

- Life support systems are necessary for spacecraft carrying humans, including:
  - Comfortable temperature range
  - Oxygen at the right pressure for comfortable breathing
    - ✓ Mix with nitrogen to decrease fire hazard
  - Water for drinking, hygiene and humidity
  - Food
  - Waste Management
  - Removal of CO_2 from the air



Freefall

- A person in orbit is in constant freefall
- A freefall environment poses significant hazards to humans:
  - Decrease in hydrostatic gradient or fluid shift
  - Altered vestibular functions (inner ear) leading to motion sickness
  - Bone degeneration
    - ✓ Bone marrow, which produces red blood cells, is affected (i.e. less red blood cells)
    - ✓ Unacceptable levels of bone fragility could develop in a person living in microgravity for 1 to 2 years
  - Muscle atrophy
  - Loss of cardio vascular conditioning
  - Loss of lean body mass

Due to reduced load on weight-bearing tissues

Freefall

- Hydrostatic Gradient
  - Refers to the distribution of fluids in our body
    - ✓ Gravity pulls fluid toward our legs so that blood pressure is normally higher in our feet than our heads
- In freefall the fluid is equally distributed through our body
  - As a result, pressure in the lower part of the body decreases while pressure in the upper body increases
    - ✓ This shift is called a decrease in hydrostatic gradient or fluid shift
  - Each leg can lose as much as 1 litre of fluid and about 10% of its volume
    - ✓ Leads to several changes
      - ➢ See next slide

Freefall

- Effects of Fluid Shift
  - Kidneys work overtime to eliminate what appears to be extra fluid in the upper part of the body
    - ✓ Increased urination
    - ✓ Body fluid may decrease by as much as 20%
  - Edema to the face
    - ✓ Redness, puffiness
  - Heart beats faster and with greater irregularity
    - ✓ Heart loses mass because it doesn't have to work as hard
  - Orthostatic intolerance on return to Earth
    - ✓ Similar to a head rush if you stand too quickly
    - ✓ May cause black outs

Freefall

- Counteracting the effects of freefall:
  - Vigorous exercise offers some promise in preventing long term atrophy of muscles
  - Flight doctors recommend fifteen minutes of exercise daily on 7 to 14 day missions and thirty minutes of exercise daily on 30 day missions
  - Exercising also helps people re-adjust more quickly to Earth's gravity when they return home
  - Preventing changes within bones is a more complicated problem
    - ✓ The development of artificial gravity may be necessary for very long missions
      - ➢ e.g. Long voyage to reach Mars and then only 38% of Earth's gravity on Mars

Freefall

- Valeri Polyakov holds the record for the longest single spaceflight in human history, staying aboard the Mir space station for 437 days during one trip
  - Upon landing, Polyakov managed to walk the few feet between the Soyuz capsule and a nearby chair
  - Polyakov did not suffer from any prolonged performance impairments after returning to Earth
  - Researchers concluded that overall function could be maintained during extended duration spaceflights

Valeri Polyakov (left) arriving back from Mir in 1989

Mir 1986-2001

Radiation

- Radiation and Charged Particles
  - From the standpoint of biological damage, we can treat exposure to EM radiation and charged particles in much the same way
  - The overall severity of damage depends on total dosage
    - ✓ Measure of accumulated radiation or charged particle exposure
  - Quantifying the dosage depends on two factors:
    - ✓ Energy contained in the radiation or particles
      - ➢ Measured in Radiation Absorbed Doses (RADs)
    - ✓ Relative biological effectiveness (RBE) rating of the exposure

Radiation

- Radiation Absorbed Doses (RADs)
  - Is a measure of the energy contained in radiation or particles
  - 1 RAD = 0.01 J/kg
    - ✓ About the energy it would take to lift a paper clip
  - The SI Unit is the Gray
    - ✓ 1 Gray = 1 J/kg = 100 RADs
- Relative Biological Effectiveness (RBE)
  - Represents the destructive power of dosage on living tissue
    - ✓ EM radiation (photons) has an RBE of 1
    - ✓ Charged particles can have an RBE of 10 or more
      - ➢ At least 10 times more destructive than photons

Radiation

- Röentgen Equivalent in Man (REM)
  - The total dosage, or REM, is quantified as the product of the RAD and RBE
    - ✓ REM dosage is cumulative over a person's lifetime
  - In SI units Grays are multiplied by RBE to get total dose, H, in Sieverts (Sv)
    - ✓ 1 Sievert = 100 REMs

Total Dose - SI Units

$$\text{Dose equivalent in Sieverts (Sv)} \longrightarrow H = \underbrace{Q}_{\text{RBE (also known as the quality factor)}}D \longleftarrow \text{Dose in Grays (Gy)}$$

Radiation

- The effects on humans exposed to radiation depends to some extent on the time over which a dose occurs
  - 0.5 Sv (50 REMs) in one day will be more harmful than the same dosage spread over one year
    - ✓ Short term doses are called acute doses
      - ➢ Effects fast producing cells within the body, specifically in the gastrointestinal tract and bone marrow
        - » Interferes with cell division
      - ➢ Physical manifestations include blood count changes, vomiting, diarrhea and death
    - ✓ High dosages spread over a long time can lead to cataracts and various types of cancer such as leukemia

Radiation

- Galactic Cosmic Radiation (GCR) cause the most problems since these particles are most massive
- Protons are more damaging than electrons since they are 1×10^4 more massive

RBE and Occurrence of Different Kinds of Radiation

GCRs interacting with shield material produce bremsstrahlung radiation

Radiation	RBE	Occurrence
X-rays	1	Radiation belts, Solar radiation, Bremsstrahlung
5 MeV γrays	0.5	Radiation belts, Solar radiation, Bremsstrahlung
1 MeV γrays	0.7	Radiation belts, Solar radiation, Bremsstrahlung
200 keV γrays	1.0	Radiation belts, Solar radiation, Bremsstrahlung
Electrons	1.0	Radiation belts
Protons	2.0 - 10.0	Cosmic radiation, Inner radiation belt
Neutrons	2-10	Close to the Earth, the Sun and any matter
α-particles	10-20	2 neutrons and 2 protons Cosmic radiation
Heavy particles		GCRs Cosmic radiation

Radiation

- The higher the dosage and the faster it comes, the worse the effects on humans

Dose [Sv]	Probable effects
0 - 0.5	No obvious effects; possibly minor blood changes
0.5 - 1	Radiation sickness in 5 - 10 % of exposed personnel; no serious disability
1 - 1.5	Radiation sickness in about 25 % of exposed personnel
1.5 - 2	Radiation sickness in about 50 % of exposed personnel; no deaths anticipated
2 - 3.5	Radiation sickness in nearly all the personnel; about 20 % deaths
3.5 - 5	Radiation sickness; about 50 % deaths
10	Probably no survivors

Probable radiation dose effects
Recall: Cosmos 954 had pieces with readings of 1.1 Sv

Radiation

- Radiation dosage is a concern for very long missions
  - Extended space station mission
  - Manned Mars expedition

Example of Radiation of Dosages

Event	Dosage (Sv)
Fly across North America	0.00004
Chest X-ray	0.0001
Living 1 year a sea level	0.001
Living 1 year at 1600 m	0.002
Space Shuttle Mission	0.0065
Apollo Mission	0.05
Space Station 84 days	0.1785

Radiation

- Onboard a spacecraft the crew is most sensitive to radiation doses, followed by the electronics
- Structural metals have a high tolerance and may be used for shielding

Material	Damage Threshold (gray)
Humans and animals	10^{-1}–10^{0}
Electronics	10^{0}–10^{4}
Lubricants, hydraulic fluid	10^{3}–10^{5}
Ceramics, glasses	10^{4}–10^{6}
Polymeric material	10^{5}–10^{7}
Structural metals	10^{7}–10^{9}

Typical Sensitivities to Radiation Doses

Radiation

- Relatively easy to build shielding made of aluminum or other light metals to protect astronauts from solar EM radiation and protons from the solar wind
  - In the case of solar flares, long missions may require *storm shelters,* which are areas deep within the spacecraft that offer increased protection for a few days
- More difficult to protect from massive GCRs
  - When GCRs interact with shield material they produce secondary radiation called *bremsstrahlung*, which is also harmful
- Mission planners avoid extended periods in the Van Allen belts
  - Also avoid the South Atlantic Anomaly

Psychological Effects

- Psychological Effects
  - Excessive workload
    - ✓ Getting into space is expensive, so the demands on the crew are high once they are there
      - ➢ Leads to degraded performance
      - ➢ Important to schedule regular breaks
  - Living in confined quarters with other people for extended periods is stressful
    - ✓ Must select crews carefully
  - On long missions the crew's isolation also adds to the stress level and may lead to depression
    - ✓ Need frequent contact with friends and family
    - ✓ Psychological diversions such as music, movies and video games help relieve boredom on long missions

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Introduction to Space Science

Module 5: Orbits
Part 1: Orbital Mechanics

- History of Orbital Mechanics
- Orbital Elements
- Orbital Equations
- Two Line Element (TLE) Set

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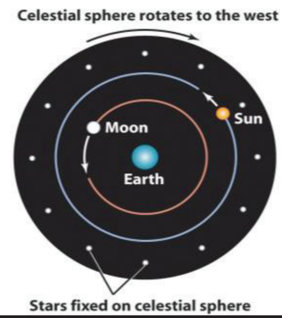
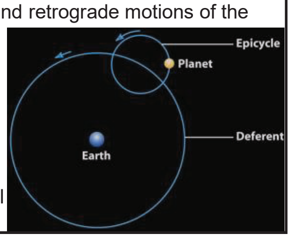
History of Orbital Mechanics

- Ancient astronomers believed the Earth to be at the center of the universe and invented geocentric models to explain planetary motions
  - According to ancient astronomers the sun and moon, as well as the planets, moved on the celestial sphere with respect to the background of stars
  - The Earth was at the center and did not move



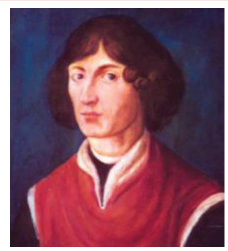
History of Orbital Mechanics

- From the Earth's perspective a planet generally moves eastward in the same direction as the sun and the moon, but from time to time it moves westward in retrograde motion
  - This is due to Earth passing slower moving outer planets
    - ✓ e.g. Mars has an apparent retrograde motion at times
- Astronomers invented a complex system of epicycles and deferents to explain the direct and retrograde motions of the planets on the celestial sphere
- Ptolemy (2nd century AD) produced 13 volumes that predicted the paths of the sun, moon and planets based on deferents and epicycles
- This collective work, the Almagest, was the astronomical bible for over 1000 years



History of Orbital Mechanics — Copernicus

- Copernicus's heliocentric (sun-centered) theory simplified the general explanation of planetary motions
- The use of epicycles was still necessary to explain all planetary motion since circular orbits were assumed
- Subsequently there was little support from astronomers to abandon the idea of a geocentric model



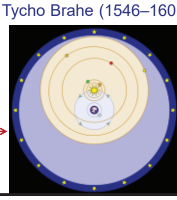
Nicolaus Copernicus (1473-1543)

Note: The notion that the Earth revolves around the sun had been proposed as early as the 3rd century BC by Aristarchus but received no support from other ancient astronomers

History of Orbital Mechanics — Brahe

- Tycho Brahe measured the position of stars and planets with unprecedented accuracy
- Brahe set out to disprove the Copernicus heliocentric model
  - He argued that if the Earth was in motion, the stars should appear to shift their position
    - ✓ Known as parallax
  - Without a telescope Brahe did not detect any parallax and concluded that the Earth was at rest
- Developed the Tychonic System
  - Earth centered, but other planets orbited the sun

Tycho Brahe (1546–1601)



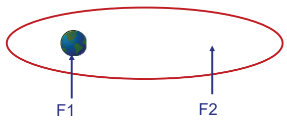
History of Orbital Mechanics — Kepler

- At the beginning of the 17th century, Johannes Kepler took on the task of finding a planetary model that agreed with the voluminous and accurate observations made by Brahe
- Using Brahe's data, Kepler deduced three laws of planetary motion
- Although these laws were based on planetary observations they also hold true for satellite orbits

Johannes Kepler (1571-1630)

History of Orbital Mechanics — Kepler's 1st Law

- Kepler's First Law: The orbit of a planet about the sun is an ellipse with the sun at one focus
  - The second focus of the ellipse is empty but has geometric significance since it helps to define the shape of the ellipse

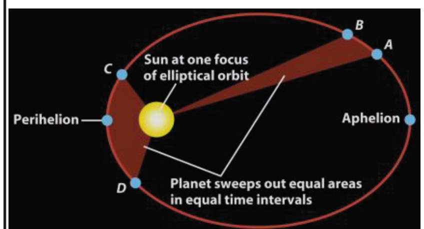


For a satellite, the Earth is at one of the focci for the orbital ellipse

- Astronomer's had long assumed that heavenly objects moved in circles, which were considered the most perfect and harmonious of geometric shapes
- Kepler's suggestion of elliptical motion of the planets was revolutionary

History of Orbital Mechanics — Kepler's 2nd Law

- Kepler's Second Law: A line joining a planet and the sun sweeps out equal areas in equal intervals of time
  - According to Kepler's 2nd law a planet moves fastest when closest to the sun and slowest when farthest from the sun



General Terms:
→Periapsis
→ Apoapsis

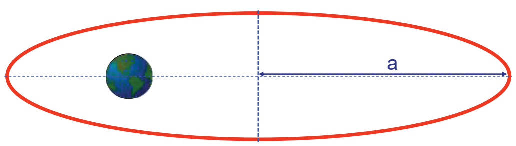
Peri: closest point of approach
Apo: most distant point of orbit

Helio: Sun centered
Geo: Earth centered

- By using the first and second laws, Kepler found a perfect fit to the apparent motions of the planets

History of Orbital Mechanics — Kepler's 3rd Law

- Kepler's Third Law: The square of the orbital period of a planet is directly proportional to the cube of the semi-major axis of the orbit
  - $P^2 \alpha\ a^3$

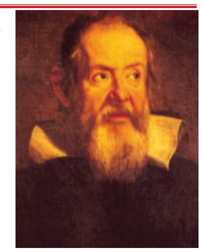


P = time for one complete orbit

- Although Kepler' s three laws supported a heliocentric solar system, many remained unconvinced since there was no hard evidence

History of Orbital Mechanics — Galileo

- Dutch opticians invented the telescope in the first decade of the 17th century
  - Galileo was the first to point one of these new inventions at the sky
- In 1610 Galileo discovered four moons, now called the Galilean satellites, orbiting Jupiter
  - His observation of these orbits proved Kepler's 3rd Law



Galileo Galilei (1564-1642)

- Galileo also saw the phases of Venus
  - This proved that Venus must orbit the sun, thus supporting the heliocentric solar system theory
- As a result of publicizing his findings Galileo spent his latter years under house arrest for vehement suspicion of heresy

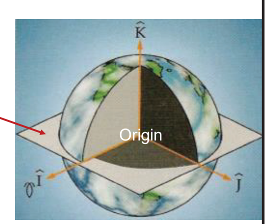
History of Orbital Mechanics — Newton

- While Galileo's work showed that a heliocentric model was correct, he was unable to explain why the Earth should orbit the sun
- The first person to provide such an explanation was Sir Isaac Newton
- Newton deduced the basic laws that govern motion on Earth and in space
- The orbital equations seen later in this lecture are based on Newton's laws

Sir Isaac Newton (1642–1727)

Orbital Elements — IJK Coordinate System

- It is necessary to use a coordinate system to describe the position of a satellite in Earth orbit
- A commonly used system is the Geocentric-equatorial coordinate system (IJK)
  - Origin = Earth's center
  - Fundamental Plane = Equatorial plane
  - I: Principle Direction = Vernal equinox, ♈
    - ✓ See next slide
  - K: Perpendicular to equatorial plane in the direction of the north pole
  - J: Right Hand Rule
    - ✓ I × J = K (cross-product)



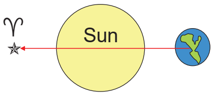
$\hat{I}$, $\hat{J}$, $\hat{K}$ are unit vectors for the system

Orbital Elements — Vernal Equinox

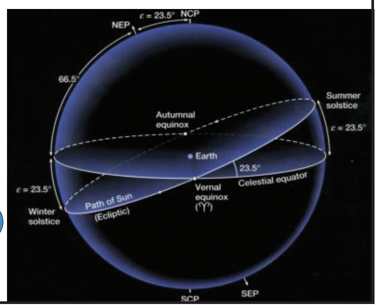
- The vernal equinox is found by drawing a line from the Earth through the sun on the first day of spring

The ecliptic is the apparent path of the sun from the perspective of the Earth, which is tilted at 23.5°

At the equinoxes (spring and fall) the Earth is in line with the sun, resulting in 12 hours of daylight/night all over the world.



First point in Aries, ♈



Orbital Elements — IJK Coordinate System

- Six elements can completely define the position of a satellite in the IJK coordinate system
  - For example:

$$\overline{R} = 10{,}000\hat{I} + 8000\hat{J} - 7000\hat{K}\ \text{km} \longrightarrow \text{Position Vector}$$

$$\overline{v} = 4.4\hat{I} + 3.1\hat{J} - 2.7\hat{K}\ \text{km/s} \longrightarrow \text{Velocity Vector}$$

- Position and velocity vectors mathematically describe the 3-dimensional state of the spacecraft at a specific time, however they **do not** help us visualize:
  - The shape of the orbit
  - The size of the orbit
  - The spacecraft's position within the orbit

Orbital Elements — Classical Orbital Elements (COEs)

- Kepler developed a method for describing orbits that allows us to visualize their shape, size and orientations, as well as the satellite's position within them
  - Classic Orbital Elements (COEs)
    - ✓ Semi-major axis, *a*
    - ✓ Eccentricity, *e*
    - ✓ Inclination, *i*
    - ✓ Right ascension of the ascending node (RAAN), Ω
    - ✓ Argument of the perigee, ω
    - ✓ True anomaly, ν *(nu)*

Orbital Elements — Eccentricity and Semi-Major Axis

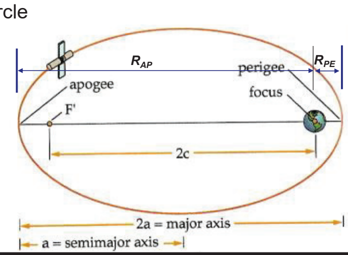
- The size of the orbit is defined by the semi-major axis, *a*, of the orbit ellipse
- Orbital shape is defined by eccentricity, *e*, of the orbital ellipse
  - Ratio of the major axis and the distance between the foci
  - As *e* approaches 1 the orbit becomes more elliptical
  - *e* = 0 is a perfect circle

$$e = \frac{2c}{2a} = \frac{R_{AP} - R_{PE}}{R_{AP} + R_{PE}}$$

R_{AP} = Range from center of Earth to Apogee

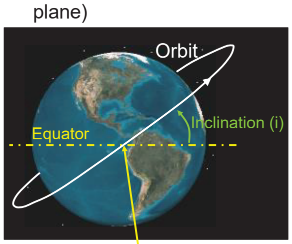
R_{PE} = Range from center of Earth to Perigee

Note: The radius of the Earth, R_e, is 6378 km

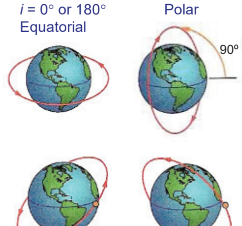


Orbital Elements — Inclination

- Inclination, *i*, describes the tilt of the orbit with respect to the equator (fundamental plane)

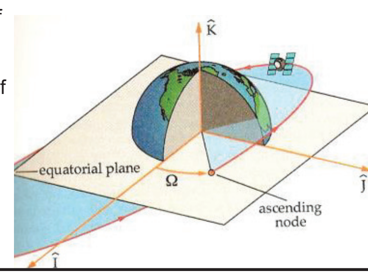


Ascending Node: Point in orbit where satellite crosses the equator in a northward direction



Orbital Elements — Right Ascension of the Ascending Node

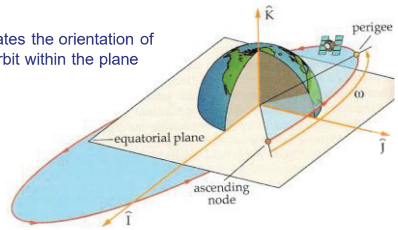
- Right ascension of the ascending node (RAAN), Ω, is the angle between the vernal equinox (I of the IJK coordinate system) and the ascending node
  - The angle is measured eastwards or, as seen from the north, counterclockwise
    - ✓ The range of values are 0 to 359°
- The intersection of the fundamental plane and orbital plane forms the *line of nodes*



Orbital Elements — Argument of the Perigee

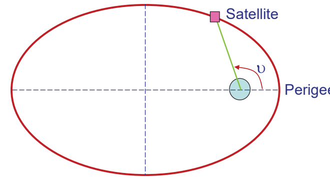
- The argument of the perigee, ω, is the angle between the ascending node and perigee
  - Tells us where the perigee of the orbit occurs
  - Measured in the direction of spacecraft motion
    - ✓ The range of values are 0 to 359°

Indicates the orientation of the orbit within the plane



Orbital Elements — True Anomaly

- The true anomaly, ν *(nu)*, is the angle along the orbital path from perigee to the spacecraft's position
  - Measured in the direction of spacecraft motion
    - ✓ Range of values are 0 to 359°
  - Indicates the position of the spacecraft within the orbit
  - Sometimes referred to as θ



- The true anomaly is not an easy calculation in an eccentric orbit (see next two slides)

Orbital Elements — Mean Anomaly

- In perfectly circular orbits it is easy to determine a future position of a spacecraft since the motion is constant
- In an elliptical orbit the speed changes with position
- Kepler defined an angular speed, *n*, that will give us an average speed for the elliptical orbit

Radians/second $\quad n = \dfrac{2\pi}{P}$ — 2π: 2 π radians in a circle; P: Period in seconds

- The Mean Anomaly, *M*, can then be expressed as:

$$M = nt$$

t = Time elapsed since the last perigee passage

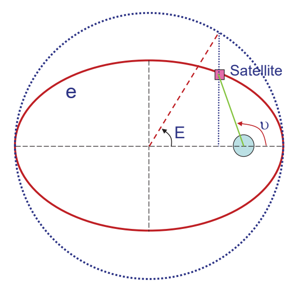
- The Mean Anomaly is an angle that has no physical meaning and can't be drawn in a picture

Orbital Elements — Eccentric Anomaly

- To physically relate circular elliptical motion to circular motion we use the Eccentric Anomaly, *E*
- *E* is defined by circumscribing an elliptical orbit with a circle and relating *M* to *E* using the true anomaly, ν
  - *M* and *E* related by Kepler's Equation

$$M = E - e\sin E$$

where *E* and *M* are in radians

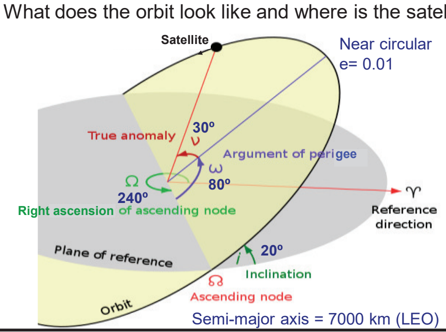


- Once *E* is determined using iterative techniques solve for ν

$$\upsilon = \cos^{-1}\left[\frac{\cos E - e}{1 - e\cos E}\right]$$

Orbital Elements — Example

- Suppose a satellite has the following COEs:
  a = 7000 km, e = 0.01, i = 20°, Ω = 240°, ω = 80° ν = 30°
  - What does the orbit look like and where is the satellite?



Orbital Equations — Law of Universal Gravitation

- Newton's Law of Universal Gravitation: The force of gravity between two bodies is directly proportional to the product of their two masses and inversely proportional to the square of the distance between them
- The universal gravitational constant, *G*, allows us to calculate the force of gravity:

$$F_g = G\frac{m_1 m_2}{R^2}$$

- F_g: Gravitational Force (N)
- $m_1 m_2$: Masses of Body-1 and Body-2 (kg)
- R^2: Distance between bodies (m)
- G: Universal gravitational constant G = 6.67 × 10^{-11} N·m^2·kg^{-2}

Orbital Equations — Gravitational Parameter

- The Gravitational Parameter (μ) is defined as the mass of Earth multiplied by the gravitational constant (G) :

Mass of Earth M_e = 5.98 × 10^{24} kg

$$GM_e = \mu = 3.986\times10^{14}\ \mathrm{m^3s^{-2}}$$

- The gravitational parameter is often expressed in terms of km^3/s^2

$$\mu = 398{,}600\ \mathrm{km^3/s^2}$$

- Acceleration due to gravity is a function of the gravitational parameter and distance from the Earth's centre

Acceleration due to gravity at distance R from the Earth's center ← $a_g = \dfrac{\mu}{R^2}$ → Distance to Earth's center

Orbital Equations — Orbital Period

- Orbital Period (P): The satellite orbital period is the time it takes to make one complete orbit
  - In accordance with Kepler's Third Law, P^2 is proportional to a^3, where P is the period and a is the semi-major axis of the orbit
  - Using this relationship we can derive an expression for P

$$P = 2\pi\sqrt{\frac{a^3}{\mu}}$$

Orbital Period; Semi-major axis of orbit; Gravitational Parameter (398,600 km^3/s^2)

- If we know the period of an orbit we can re-arrange this equation to find the semi-major axis, a

$$a = \left(\frac{P^2\mu}{4\pi^2}\right)^{1/3}$$

Semi-major Axis Calculation

Orbital Equations — Distance to Satellite

- Distance to satellite at any point in the orbit can be calculated:

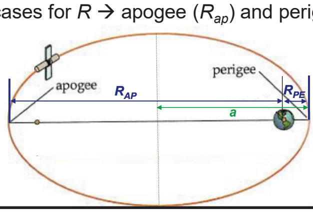
$$R = \frac{a(1-e^2)}{1+e\cos\nu}$$

R is range to satellite from centre of Earth
Note that distance changes as position changes, ν

- **NOTE:** For altitude of spacecraft ('H' for height):

H = R - R_e where R_e = *radius of Earth* = 6378 km

- Special cases for R → apogee (R_{ap}) and perigee (R_{pe}):



At apogee ν = 180°

$$R_{AP} = a(1+e)$$

Distance to apogee from Earth's centre

At perigee ν = 0°

$$R_{PE} = a(1-e)$$

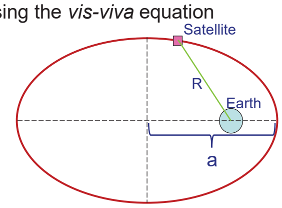
Distance to perigee from Earth's centre

Orbital Equations — Satellite Velocity

- Velocity (v): The velocity of a satellite at a particular point in the orbit can be determined using the *vis-viva* equation

$$\mathrm{v} = \sqrt{\mu\left(\frac{2}{R}-\frac{1}{a}\right)}$$

Velocity; Gravitational Parameter (398,600 km^3/s^2); Range to satellite from centre of Earth (Satellite altitude + 6378 km); Semi-major axis of orbit (**NOT** satellite altitude)



- Speed is constantly changing in an orbit unless it is a perfect circle
  - i.e. Range to satellite from Earth's centre (R) changes throughout the orbit

Two Line Element (TLE) Set

- Keplerian elements parameters can be encoded as text in a *two-line element* (TLE) set, which is used by NASA/NORAD

Satellite # · International Designator · Year and Day Fraction · 1st and 2nd derivative of Mean Motion · B* · Check Sum · Element #

```
1  25544  98067A   08264.51782528  -.00002182  00000-0  11606-4  0  292  7
2  25544  51.6416° 247.4627° 0006703  130.5360° 325.0288° 15.72125391 56353 7
```

i degrees · Ω degrees · e · ω degrees · M degrees · Mean Motion · Revolution #

- Decimal point assumed in front of eccentricity
- Mean Motion refers to orbits per day of the satellite
  - ✓ Therefore Period = 24 hours/Mean Motion
- Position is given in terms of the Mean Anomaly, M
- B* indicates the amount of drag on the satellite

Two Line Element (TLE) Set — Example

- The following is a TLE for the International Space Station
  - What is the orbital altitude?

```
25544U 98067A   090 25.90753036  0.00009376  00000-0  79134-4  0 1400 8
25544  51.6429 315.3294 .0007688 309.6325  94.9788 15.70580475  58357 8
```

i · Ω · e · ω · M · Mean Motion Orbits per day

Determine the semi-major axis, a

$$a = \left(\frac{P^2\mu}{4\pi^2}\right)^{1/3} \quad \text{where } P = \frac{24\ \text{hrs}}{\text{Mean Motion}} = \frac{24\times60\times60\ \text{s}}{15.70580475} = 5501.2\ \text{s}$$

$$\therefore a = \left(\frac{(5501.2\ \text{s})^2\,398{,}600\ \mathrm{km^3/s^2}}{4\pi^2}\right)^{1/3} = 6735.4\ \text{km}$$

Distance to the centre of the Earth

Near circular orbit (e ~ 0) therefore, orbit altitude is 6735 – 6378 = 357 km

PHE 255

Introduction to Space Science

Module 5: Orbits

Part 2: Ground Tracks, Maneuvers and Perturbations

- Satellite Ground Track
- Satellite Maneuvers
- Orbital Perturbations

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Royal Military College of Canada
Department of Physics and Space Science

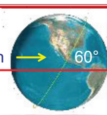
Satellite Ground Track

- A satellite **ground track** is the path on the surface of the Earth directly below a satellite
  - Note: Directly below the satellite = nadir
  - For many space applications it is important to know what part of the Earth a satellite is passing over at any given time
    - ✓ e.g. Remote sensing, communication with ground station
- Since the Earth rotates eastward at 15°/hr (360°/24 hrs), the orbit track appears to move westward on consecutive orbits from the perspective of a person on the ground
- For a prograde orbit (*i* < *90°*) the satellite will appear to move eastward, while for a retrograde orbit (*i* > *90°*) the satellite will appear to move westward

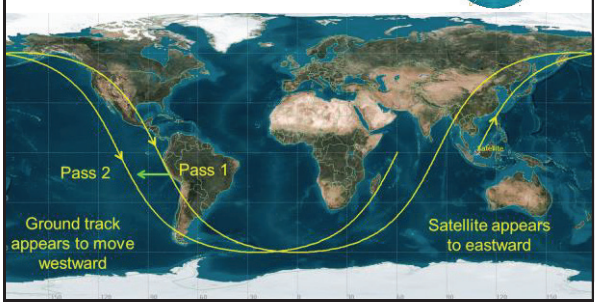
Satellite Ground Track — LEO Prograde

- Example: LEO, *i* = 60° prograde orbit

Earth Rotation



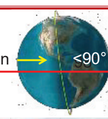
Ground track for two orbits



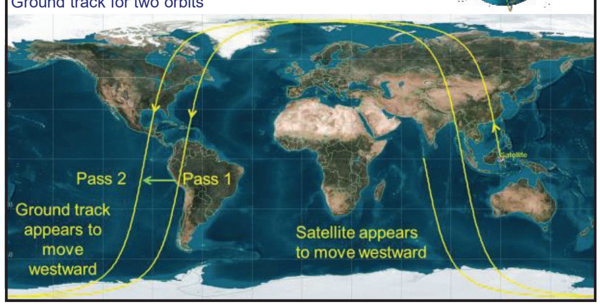
Satellite Ground Track — LEO Retrograde

- Example: LEO, *i* = 98° retrograde orbit

Earth Rotation



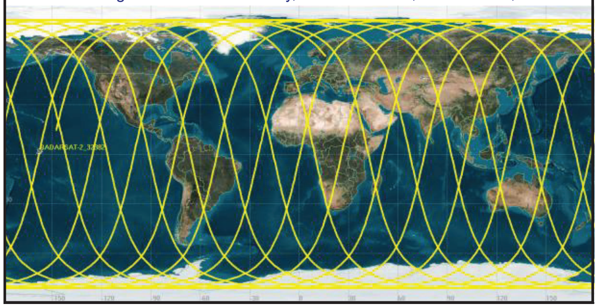
Ground track for two orbits



Satellite Ground Track — LEO Orbits Per Day

- A LEO orbit will orbit the Earth ~14 t0 15 times per day since the period is in the order of 100 minutes (1 day = 1440 min)

RADARSAT-2 ground track for one day, 792 km altitude, near circular, i = 98.6°

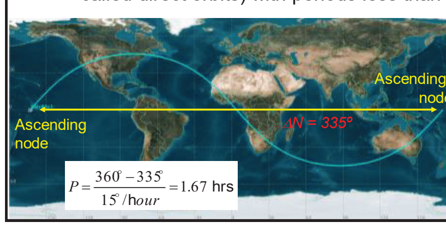


Satellite Ground Track — Nodal Displacement

- *Nodal displacement, ΔN,* can tell us the period of the orbit
  - *ΔN* is the displacement in degrees between successive ascending nodes

$$P = \frac{360^{\circ} - \Delta N}{15^{\circ}\,/\,hour} \longrightarrow \text{Period in hours}$$

- ✓ Note: This equation only works for prograde orbits (also called direct orbits) with periods less than 24 hours



$$P = \frac{360^{\circ} - 335^{\circ}}{15^{\circ}\,/\,hour} = 1.67 \text{ hrs}$$

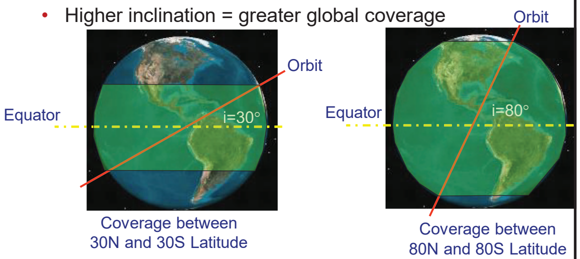
To find a:

$$a = \left(\frac{P^2 \mu}{4\pi^2}\right)^{1/3}$$

The inclination of this orbit is 45°
→ See next slide

Satellite Ground Track — Inclination

- Since the inclination relates the angle between the orbital plane and the equatorial plane, the highest latitude reached by a spacecraft equals its inclination for direct orbits
  - Higher inclination = greater global coverage



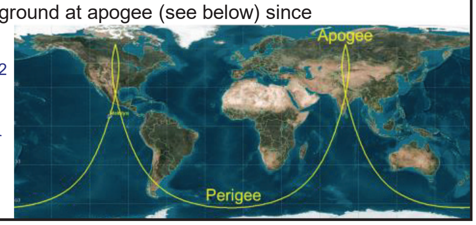
- For indirect orbits (retrograde), subtract the maximum latitude from 180° to get the inclination

Satellite Ground Track — Perigee and Apogee

- Eccentricity and the location of the perigee also affect the shape of the ground track
  - (Near) Circular Orbit: Ground track is symmetrical about the equator
  - Highly Elliptical Orbit: Ground track is asymmetrical about the equator
    - ✓ The satellite appears to cover more ground at perigee and less ground at apogee (see below) since

Example: Molniya Orbit (2 orbits shown)

Satellite is moving slower than Earth rotation at apogee



Satellite Maneuvers — Total Mechanical Energy

- Total energy in an orbiting satellite is the combination of kinetic energy (*KE*) and potential energy (*PE*)

$$E = KE + PE \quad \text{where} \quad KE = \frac{1}{2}mv^2 \quad PE = -\frac{m\mu}{R}$$

- For space applications we choose the zero of gravitational potential energy to be at infinity
  - The negative sign indicates the direction is toward the Earth for a geocentric orbit
- The total mechanical energy of a satellite in orbit is the total of its kinetic and potential energy

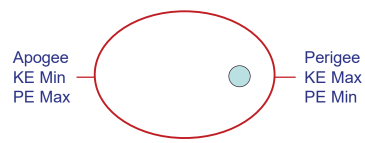
Total Mechanical Energy $\longrightarrow$ $E = \frac{1}{2}mv^2 - \frac{m\mu}{R}$ Units kg·m^2·s^{-2} = Joule

Satellite Maneuvers — Conservation of Energy

- Conservation of Energy
  - The initial *KE* and *PE* of the system must equal the final *KE* and *PE* of the system

$$E = KE_{initial} + PE_{initial} = KE_{final} + PE_{final}$$

- Since *E* is conserved, it is constant throughout the orbit
  - Trade-off between *KE* and *PE* throughout the orbit



Satellite Maneuvers — Escape Velocity

- The escape velocity, v_{esc}, is the speed needed to break free from a gravitational field
  - For an object under the influence of Earth's gravity, the minimum escape velocity would result in the total energy of the spacecraft equalling zero

$$E = \frac{1}{2}mv^2 - \frac{m\mu}{R} = 0$$

Rearrange to find v_{esc}

Recall from last lecture: Gravitation Parameter μ = 398,600 km^3/s^2

$$\therefore v_{esc} = \sqrt{\frac{2\mu}{R}}$$

~ 11.2 km/s from the Earth's surface

Satellite Maneuvers — Specific Mechanical Energy

- Specific mechanical energy (ε) indicates the energy in an orbit regardless of mass, where:

$$\varepsilon \equiv \frac{E}{m}$$

Recall: $E = \frac{1}{2}mv^2 - \frac{m\mu}{R} \longrightarrow \varepsilon = \frac{v^2}{2} - \frac{\mu}{R}$ km^2·s^{-2}

- Semi-major axis, *a*, can be expressed as a function of specific mechanical energy,

$$a = -\frac{\mu}{2\varepsilon}$$

Satellite Maneuvers — Hohmann Transfer

- One of the first problems faced by space-mission designers was figuring out how to go from one orbit to another
  - Gemini missions of the 1960s refined the process in preparation for the Moon missions
- The simplest manoeuvre is co-planar
  - Initial and final orbits are in the same plane
- Fuel is critical for all missions, so it is necessary to plan orbit changes in the most efficient manner
  - The *Hohmann Transfer* is the most fuel-efficient way to carry out co-planar orbital manoeuvres
    - ✓ Theorized by Walter Hohmann in 1925

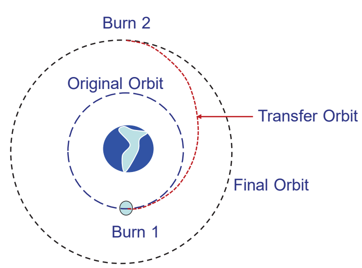
Satellite Maneuvers — Hohmann Transfer

- Hohmann transfers are limited to:
  - Orbits in the same plane
  - Orbits with the major axes aligned
  - Instantaneous velocity changes are tangent to the initial and final orbits
- Velocity change is referred to as Δv, or *impulsive burn*
  - i.e. Burning fuel to achieve thrust
  - $\Delta v = | v_{selected} - v_{present} |$
  - Absolute value used since fuel is burned whether the spacecraft accelerates or decelerates
- Whenever we change the velocity we change the orbit's specific mechanical energy

i.e. $\varepsilon = \frac{v^2}{2} - \frac{\mu}{R}$

Satellite Maneuvers — Hohmann Transfer Example

- Step 1: The first burn takes the spacecraft out of it's initial circular orbit and puts it into an elliptical transfer orbit
- Step 2: The second burn takes the spacecraft from the transfer orbit and puts it in the final circular orbit



Satellite Maneuvers — Hohmann Transfer

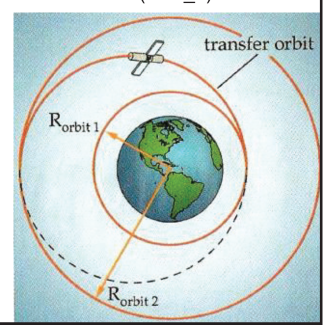
- To solve for the required Δv for the two burns we need to know the specific mechanical energy (ε) of the initial orbit (orbit_1), transfer orbit and the final orbit (orbit_2)

$$\varepsilon_{orbit\_1} = -\frac{\mu}{2a_{orbit\_1}}$$

$$\varepsilon_{transfer} = -\frac{\mu}{2a_{transfer}}$$

$$2a_{transfer} = R_{orbit\ 1} + R_{orbit\ 2}$$

$$\varepsilon_{orbit\_2} = -\frac{\mu}{2a_{orbit\_2}}$$



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Satellite Maneuvers — Hohmann Transfer

- The Δv values for the two burns can be calculated:

$$\Delta v_1 = \left| v_{transfer\_orbit\_1} - v_{orbit\_1} \right|$$ where

$$v_{transfer\_orbit\_1} = \sqrt{2\left(\frac{\mu}{R_{orbit\_1}} + \varepsilon_{transfer}\right)}$$

$$v_{orbit\_1} = \sqrt{2\left(\frac{\mu}{R_{orbit\_1}} + \varepsilon_{orbit\_1}\right)}$$

$$\Delta v_2 = \left| v_{orbit\_2} - v_{transfer\_orbit\_2} \right|$$ where

$$v_{orbit\_2} = \sqrt{2\left(\frac{\mu}{R_{orbit\_2}} + \varepsilon_{orbit\_2}\right)}$$

$$v_{transfer\_orbit\_2} = \sqrt{2\left(\frac{\mu}{R_{orbit\_2}} + \varepsilon_{transfer}\right)}$$

The total Δv for the Hohmann transfer is:

$$\Delta v_{total} = \left| \Delta v_1 + \Delta v_2 \right|$$

Time of Flight (TOF) of the transfer orbit is

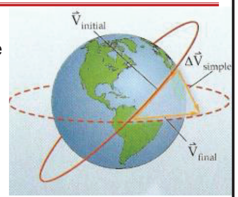
$$TOF = \pi\sqrt{\frac{a^3_{transfer}}{\mu}}$$

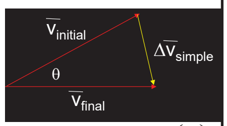
Satellite Maneuvers — Plane Change

- There are two types of orbital plane changes
  - Simple Change
    - ✓ Only the direction changes
  - Combined Plane Change
    - ✓ Change both the direction and the magnitude
- Example:
  - A satellite intended for a geostationary orbit is launched due east from the Kennedy Space Station (latitude 28.6ºN) into a LEO *parking* orbit
    - ✓ It will have an inclination of 28.6°
    - ✓ It will be in an orbit less than a 1,000 km in altitude
    - ✓ It needs to have an inclination of 0° and an altitude of 35,800 km
      - ➢ Combined plane change required

Satellite Maneuvers — Simple Plane Change

- A simple plane change only affects the direction and not the magnitude of the original velocity vector
  - If we only want to change the inclination, we must change the velocity at either the ascending or descending node
- Generally, since $v_{initial} = v_{final}$ we can use geometry on the resulting isosceles triangle to determine Δv_{simple}
- In order to reduce fuel used, make Δv as small as possible do the manoeuvre at apogee
  - Lowest value for the satellite initial velocity

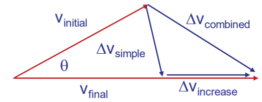




$$\Delta v_{simple} = 2 v_{initial} \sin\left(\frac{\theta}{2}\right)$$

Satellite Maneuvers — Combined Plane Change

- For a combined plane change we want to change the plane and the magnitude of the velocity



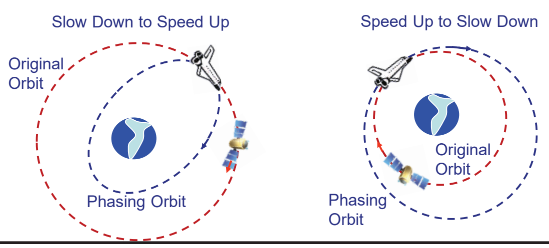
Apply the law of cosines to solve for the required Δv

$$\Delta v_{combined} = \sqrt{v_{initial}^2 + v_{final}^2 - 2\left|v_{initial}\right|\left|v_{final}\right|\cos\theta}$$

- To reduce fuel consumption conduct manoeuvre at slow velocities and small angular changes

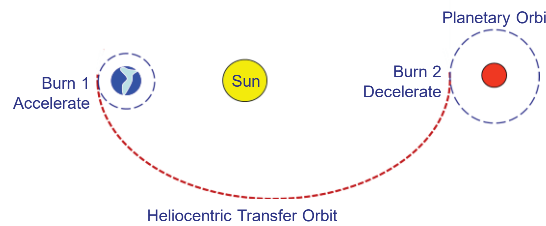
Satellite Maneuvers — Orbital Rendezvous

- To catch another spacecraft ahead of it in the same orbit, an interceptor decelerates, entering a lower, faster phasing orbit with a shorter period
- If the target is behind the interceptor in the same orbit, the interceptor must accelerate to enter a higher, slower orbit thereby allowing the interceptor to catch up



Satellite Maneuvers — Planetary Rendezvous

- A planetary rendezvous requires two burns
  - First Burn or ΔV_{Boost} transfers from a circular parking orbit around Earth to a hyperbolic departure trajectory that patches into an elliptical orbit around the sun
  - Second Burn or ΔV_{Retro} slows the spacecraft down to enter orbit around the destination planet

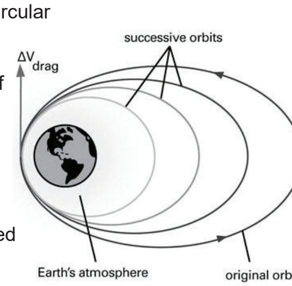


Orbital Perturbations

- External forces that change any of the classical orbital elements of the orbit are called *perturbations*
  - Perturbations do not include the main force of gravity or Δv (thrust) of the satellite
- Orbital perturbations include:
  - Atmospheric Drag
  - Earth Oblateness
  - Solar Radiation Pressure (SRP)
  - Third Body Effects
- Orbit prediction is important for operations, since we need to know where the spacecraft will be at some point in the future
  - To predict the location of a satellite in the future, we need to take all perturbations into account

Orbital Perturbations — Drag

- Drag is a non-conservative force that takes energy out of the orbit in the form of friction
  - The semi-major axis becomes smaller
  - The orbit becomes more circular
- As the spacecraft passes through the atmosphere at perigee energy is taken out of the orbit
  - This will lower the apogee altitude, circularizing the orbit until it decays and re-enters
  - Once the orbit is circularized at ~ 200 km it will de-orbit will within a few days



Orbital Perturbations — Drag

- Drag is difficult to model because there are many factors that influence the Earth's upper atmosphere
  - E.g. more drag during solar maximum
- Drag is also a function of the satellite's ballistic coefficient (*BC*)
  - Higher *BC* = less effected by drag

$$BC = {}^{m}\!/_{C_d A_d}$$

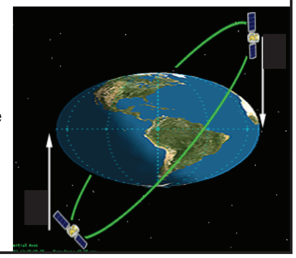
m - satellite mass
C_d - coefficient of drag
A_d - Area exposed to drag

	ESTIMATED ORBIT LIFETIME			
Alt (km)	Solar Min 50 kg/ m² (days)	Solar Max 50 kg/ m² (days)	Solar Min 200 kg/ m² (days)	Solar Max 200 kg/ m² (days)
0	0.00	0.00	0.00	0.00
100	0.06	0.06	0.06	0.06
150	0.24	0.18	0.54	0.48
200	1.65	1.03	5.99	3.6
250	10.06	3.82	40.21	14.98
300	49.9	11.0	196.7	49.2
350	195.6	30.9	615.9	140.3
400	552.2	77.4	1024.5	346.9
450	872	181	1,497	724
500	1,205	393	2,377	3,310
550	1,638	801	5,470	4,775
600	2,580	3,430	14,100	13,400

Orbit lifetime for solar maximum and solar minimum for *BC* = 50 and 200 kg/m²

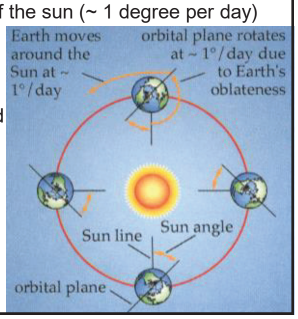
Orbital Perturbations — Earth Oblateness

- Earth Oblateness
  - The Earth is an oblate spheroid that is 22 km larger around equator than from pole to pole
    - ✓ This gravitational asymmetry cause changes in RAAN and argument of the perigee
  - The effect is a twisting force on the spacecraft orbit
    - ✓ Results in movement of the ascending node
      - ➢ Westward for prograde orbits, eastward for retrograde orbits
      - ➢ Polar orbits not effected



Orbital Perturbations — Earth Oblateness

- Sun synchronous orbits take advantage of the change in RAAN
  - By selecting the proper semi-major axis (a), inclination (i), and eccentricity (e) it is possible to match the change in RAAN with the movement of the sun (~ 1 degree per day)
  - The same angle between the orbital plane and the sun will be maintained
  - Satellite passes over ground at the same local time
  - Possible to follow the day/night line (terminator) to minimize eclipses
    - ✓ Example: RADARSAT *a* =7178 km, *i* = 98.6°, *e* = 0.01

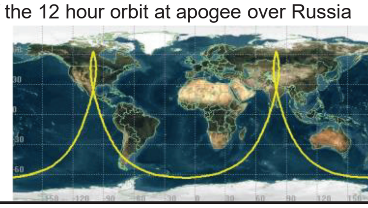


Orbital Perturbations — Earth Oblateness

- The Earth's oblateness also causes the perigee to rotate
  - Argument of the perigee (ω) changes
  - By selecting the proper semi-major axis (a), inclination (i), and eccentricity (e) it is possible to make the change in $\omega = 0$
- The Russian *Molniya* orbit uses *a = 26,554 km, e = 0.72* and *i = 63.4°* so that the change in *ω = 0°*
  - Useful for northern communications since the satellite spends most of the 12 hour orbit at apogee over Russia

24-hour ground track of a Molniya satellite

Perigee and, therefore, apogee, do not move



Orbital — Solar Radiation Pressure

- Solar Radiation Pressure (SRP) is the pressure exerted on a satellite by light photons
  - Can cause long-term orbital perturbations and unwanted spacecraft rotation
  - Strongest for light satellites with large cross-sectional areas exposed to the sun
  - Above ~ 500 to 800 km the effects of SRP are greater than atmospheric drag
  - The dominant effect of solar radiation pressure is to increase the eccentricity of GEO satellites

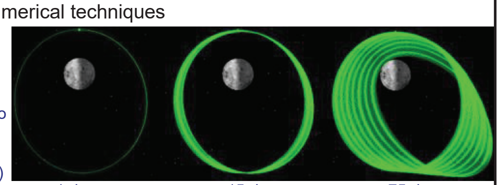
In 1960, the large balloon-like satellite Echo-1 felt the effects of solar pressure radiation.

"Photon pressure played orbital soccer with the Echo-1 thin-film balloon in orbit...."

Orbital Perturbations — 3rd Body Gravitational Effects

- Third Body Gravitational Effects
  - The sun, moon and other planets can perturb orbits at high altitudes and on interplanetary trajectories
  - For geocentric orbits the sun and moon have about the same effect on a satellite
  - The solution for the 3-body acceleration term in the force model can not be solved analytically and requires numerical techniques

Change in polar orbit around the Moon due to Earth's gravity (3rd body effect)



1 day 15 days 75 days

PHE 255
Introduction to Space Science

Module 6: Space Mission Design

- Introduction
- Design Process
- Space Mission Components

Dr. Ron Vincent,
Royal Military College of Canada
Department of Physics and Space Science

Introduction

- How does a space mission begin?
  - There is a perceived requirement for satellite derived data or services
    - ✓ This may come from the government, private sector, military, educational institution etc.
- It is important to clearly define the mission requirements
  - What end result do we want to achieve or accomplish?
  - What is the objective?
- The more clear and specific the requirements the easier it is to chart a course to achieve them

Introduction

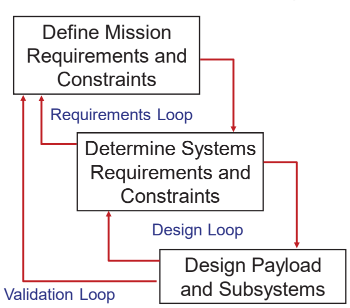
- The motivation for a space mission falls into 4 broad categories
  - Science
    - ✓ Research, increase knowledge of the universe
      - ➢ Includes exploration
  - Global Requirement
    - ✓ Monitor climate change, pollution, natural disasters, weather for global well-being
  - Service
    - ✓ Telecom, broadcasting, navigation
    - ✓ Private benefit from existing market
  - Military
    - ✓ Surveillance, secure communications
    - ✓ Potential for anti satellite (ASAT) vehicles, missile defence and weapons

Introduction

- Space mission design is not necessarily limited to a single satellite
  - Satellite constellation
    - ✓ Example: GPS (24 satellites), Iridium (66 satellites)
  - Composite design
    - ✓ Integration with other assets

Introduction

- Satellite design is an *iterative* approach
  - Between each step in the process, there are loops that take us back to review decisions in the previous step



Design Process
Mission Statement

- The design process starts with a mission statement
  - Defines the purpose of the mission and what services or information it will deliver to the user
    - ✓ Users are the customers, who give the reason for the mission and usually supply the funding
  - Can include information on how the satellite contributes to a need or a void in space technology
  - Brief and to the point (1 or 2 paragraphs)
  - The mission statement can change as a project is developed

Design Process — Mission Statement Example

- Space ADS-B Experiment (SABRE) Mission Statement

> Automatic dependent surveillance-broadcast (ADS-B) is a system in which aircraft continually transmit their identity and GPS derived navigational information. ADS-B networks for air traffic monitoring have already been implemented in areas around the world, but ground stations cannot be installed in mid ocean and are difficult to maintain in the Arctic, leaving a coverage gap for oceanic and high latitude airspace. A potential solution for worldwide tracking of aircraft is through the monitoring of aircraft transmitted ADS-B signals using satellite-borne receivers. SABRE will demonstrate the technical feasibility of a space based ADS-B receiver system, paving the way to a worldwide system of air traffic control.

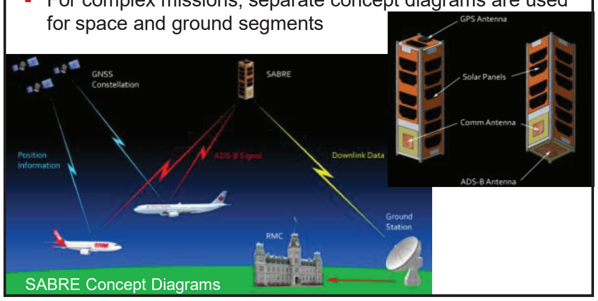
Design Process — Operations Concept

- The operations concept describes how people, systems and all the elements of the mission architecture will interact to satisfy the mission requirements
- Example: Space ADS-B Experiment (SABRE) Operations Concept

> SABRE is a low cost satellite that will collect ADS-B data over the North Atlantic from a low Earth orbit. Data will be stored onboard the satellite and then downlinked to RMCC for analysis. Truth data will be supplied from NAV Canada to determine the effectiveness of the ADS-B receiver.

Design Process — Concept Diagram

- Concept diagrams (and videos, mock-ups etc.) illustrate how the mission will be carried out
  - Promotion, quick description of mission and vehicle
- For complex missions, separate concept diagrams are used for space and ground segments



SABRE Concept Diagrams

Design Process — Mission Constraints

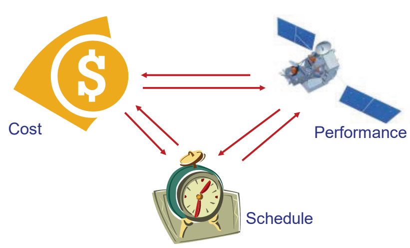
- Mission design constraints include:
  - Cost
    - ✓ Due to budget constraints mission planners are often use a design-to-cost approach
      - Will dictate size of satellite(s) and complexity of mission
    - ✓ Amount of funding will depend on user & sponsor interest
  - Time
    - ✓ Missions often must conform to a specific schedule to meet a particular launch window or are required to supply a service to paying customers on a specified date
  - Political and legal restraints
    - ✓ Orbital restrictions, radio frequency and power restrictions, dangerous material etc.

Design Process — Performance Requirements

- Performance requirements are defined early in the mission design process
  - High level performance requirements include:
    - ✓ Payload requirement
      - What is required of the payload to perform the mission
    - ✓ Length of the satellite mission
    - ✓ Areas covered and re-visit times of satellite
    - ✓ Data requirements and distribution of data
    - ✓ etc.
  - Use these performance requirements to decide if technology exists or needs to be developed
  - More specific requirements will be developed later

Design Process — Cost - Performance - Schedule

- Cost, schedule and performance represent a three-dimensional trade-space in which all space missions are constrained
- Systems engineers must constantly trade these competing objectives to achieve a well-balanced solution



Design Process — Cost Analysis

- Life Cycle Cost Estimate
  - The life cycle cost estimate is typically 1:3:6 for Development: Construction: Operations
    - ✓ 1: Development cost (paper)
      - As much as possible is done on paper before anything is built → paper is cheap
    - ✓ 3: Construction cost
      - Satellite
      - Ground station
    - ✓ 6: Operational cost
      - On orbit costs
        - » Need people to run the mission

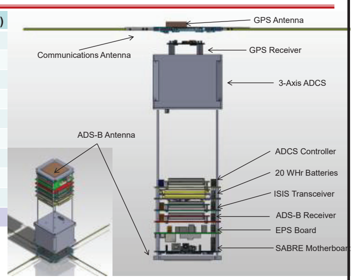
Design Process — Costing Methods

- Three types of costing methods are listed below
- Cost by Analogy
  - Based on past experience
    - ✓ Example: Radarsat-1 cost \$X , Radarsat-2 cost \$(A)*X
- Parametric Costing
  - Use formulae to determine the cost of a system
    - ✓ Example: IR sensor = \$ 35,000 $(x)^{0.562}$ ± 25% where x = optics diameter
- Engineering Costing
  - Estimate the cost and labour of each element and compile the results using a constant financial reference frame
    - ✓ e.g. Year 2015 Canadian dollars
  - Very time consuming for large projects but, appropriate for small single satellite missions
    - ✓ Example: RMCC's SABRE (see next slide)

Design Process — SABRE Engineering Costing

Satellite Components	Cost (K)
Pumpkin Structure	8*
Pumpkin Solar Panels	30*
Pumpkin C&DH	1*
ISIS Radio and Antenna	10*
OEMV-1 GPS	3*
ADS-B Payload	5*
Clydespace 20 W·Hr Battery	3*
ISIS MAI-100 ADCS	35*
Miscellaneous Hardware	5*
TOTAL	**100**

* Purchased 2015 CAD



Other Costs:

Integration, Assembly and Testing of components - \$15K
Ground Control Station - \$15K (system currently operational)
Launch Cost: \$250K (estimated)
Operations - 50 man hours per week for 2 years = \$100K
Insurance - \$20K (estimated)

\$400 K

\$500 K Total

Design Process — Insurance

- Insurance
  - Another cost associated with a satellite mission is insurance
  - Most spacecraft are insured during design, launch, and operations phases
  - Launch companies will require a signed Memorandum of Understanding regarding liability and insurance
    - ✓ If your satellite causes damage, then you are responsible
  - Typical values for insurance with respect to total cost are:
    - ✓ Pre-launch 0.35 to 1.75%
    - ✓ Launch 16 to 30%
    - ✓ Operations 1.5 to 3.5% per year
    - ✓ Liability 0.15%

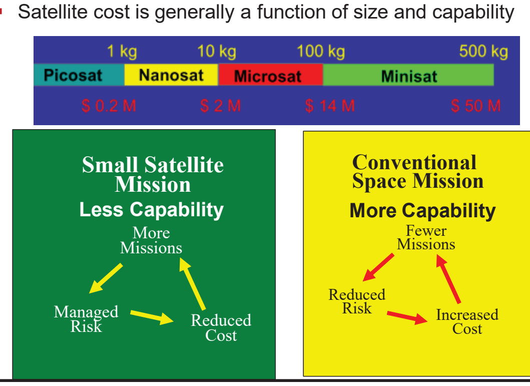
Launch is the riskiest part of any space mission

Design Process — COTS Components

- To mitigate cost, risk and development time, it is desirable to use space-proven, commercial-off-the-shelf (COTS) components
- COTS considerations include:
  - Vacuum effects
    - ✓ Outgassing from plastic material
    - ✓ Electrostatic discharge
  - Thermal control
    - ✓ Operating temperature range
  - Ionizing radiation
    - ✓ Radiation resistance
  - Mechanical effects
    - ✓ Shock loadings during launch
    - ✓ Low frequency mechanical resonances

Design Process — Approximate Satellite Costs

- Satellite cost is generally a function of size and capability

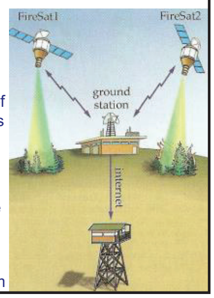


Design Process — Trading Requirements

- Trading requirements
  - Few projects take a linear course to their final design
  - Detailed analysis of mission requirements may make mission planners re-think them
- As an example, let's consider a hypothetical mission called FireSat

Mission Statement: Detect and locate forest fires worldwide and provide timely notification. Users of this service are national and international agencies responsible for fighting forest fires.

Operations Concept: There will be a number of satellites in LEO that are capable of detecting and locating forest fires. The system will communicate this information to users through the internet. The mission will be controlled via a single dedicated ground system.



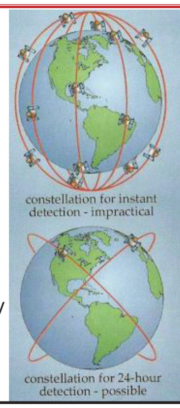
Concept Diagram

Design Process — Trading Requirements Example

- The FireSat mission statement says:
  - "Detect and locate forest fires worldwide and provide timely notification to users."
    - ✓ A preliminary analysis of the number of spacecraft needed to provide global coverage showed that to provide instant notification, the manufacturing, launch and operations costs are prohibitive
      - ➢ Over 30 satellites required in low Earth orbit
    - ✓ The question becomes, how fast is *timely* ?
      - ➢ Is instant notification necessary?
    - ✓ Another question is what constitutes *worldwide* ?
      - ➢ No forest fires in the polar regions

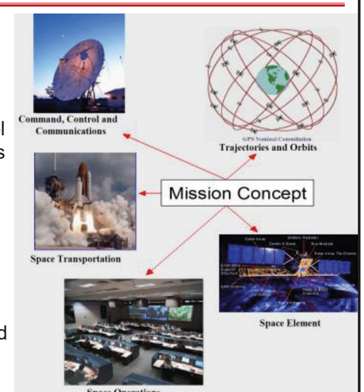
Design Process — Trading Requirements Example

- FireSat Requirements
  - Mission designers and sponsors would need to work together to re-define 'timely' and 'worldwide' coverage
  - By changing timely to 'within 24-hours' and changing worldwide to 'forested areas', a two satellite constellation with inclinations of 60 and 120° can meet the mission objective
    - ✓ Thus, a mission that was originally too expensive or impossible to carry out may become affordable and reasonable to accomplish



Space Mission Components

- Space Mission Components include:
  - Trajectories and orbits
  - Command, Control & Communications
    - ✓ C3
  - Space Operations
  - Space transportation
    - ✓ Launch
  - Space element
  - Spacecraft payload and subsystems



Space Mission Components — Trajectories and Orbits

- Trajectories
  - Launch
  - Re-entry
  - Escape
    - ✓ Interplanetary

- Orbits
  - Geostationary (GEO)
  - Geosynchronous (GSO)
  - Semi synchronous (12 hours)
    - ✓ MEO
  - Sun synchronous
  - Molniya
  - Tundra
  - Low Earth orbit (LEO)
  - Highly elliptical orbit (HEO)
  - Transfer orbits
    - ✓ e.g. Geosynchronous Transfer Orbit (GTO)

Space Mission Components — Trajectories and Orbits

- Decommissioning Phase
  - GEO positions are valuable
    - ✓ Satellites at the end of their mission life are raised to a super-synchronous orbit
      - ➢ A graveyard burn raises the orbit altitude by 200 to 300 km
  - LEO satellites that are projected to remain in orbit more than 25 years after mission completion must be de-orbited
    - ✓ Need enough fuel at end-of-life to safely de-orbit the satellite
    - ✓ What about satellites without propulsion system?
      - ➢ e.g. Most Cubesats do not have propulsion
        - » CanX-7 demonstrating a drag sail

Space Mission Components — C3

- Command, Control and Communications (C3)
  - Necessary to track the satellite in order to communicate with it for data uplink (commands) and downlink
  - Track with optical instruments, radar, telemetry
    - ✓ e.g. United States Space Surveillance Network (SSN)

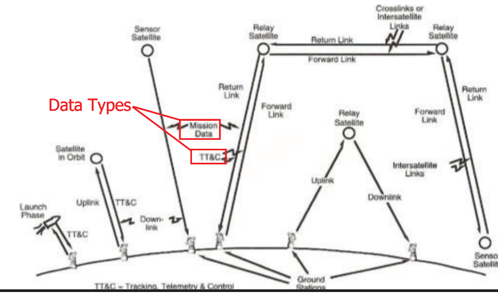
Radar Ground Tracking (Gladstone 70 m radar antenna)

Optical Ground Tracking RMCC Castor Telescope

Tracking from Space (Tracking and Data Relay Satellite - TDRS)

Space Mission Components — C3

- Communication is accomplished with uplinks, downlinks and crosslinks
  - Data types include satellite Telemetry, Tracking and Control (TT&C) data and payload data (mission data)



Space Mission Components — Space Operations

- Space operations deals with controlling the spacecraft payload and disseminating data to the users
- Ground control elements include:
  - Mission Control Centre (MCC) plans and operates the entire space mission, including the configuration and scheduling of resources for both the space and ground segment (Managers)
  - Spacecraft Operations Control Centre (SOCC) monitors and commands the spacecraft (Engineers)
  - Payload Operations Control Centre (POCC) analyzes telemetry and mission data from spacecraft payload instruments and issues commands to those instruments through the SOCC (Scientists)
- Depending on the size of the mission, these elements may be co-located or widely distributed

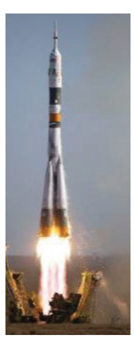
Space Mission Components — Operations Example

- MCC and SOCCs are shown for the GPS constellation
  - Continuity of contact and security of the constellation is key

Space Mission Components — Space Transportation

- Space transportation is the means to get into orbit
- Must consider a number of issues:
  - Launch Environment
    - ✓ e.g. Vibrations
  - Final Orbit
  - Satellite mass and volume
  - Launch Opportunities
    - ✓ Small satellites often rely on 'hitching a ride' with larger payloads to reduce costs

Atlas V



Soyuz

Space Mission Components — Satellite Components

- The three main components of the spacecraft are:
  - Payload
    - ✓ Mission of the satellite
      - ➢ Only part of spacecraft that realizes the objectives of the mission
  - Bus
    - ✓ Subsystems to support payload
  - Adapters
    - ✓ Interface with launcher

Space Mission Components — Payload

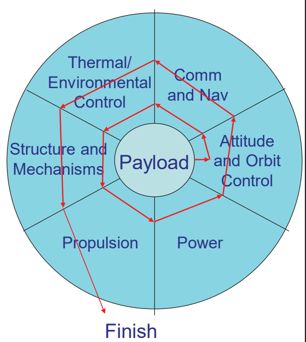
- Payload
  - The payload is the reason for the mission
    - ✓ Example: For the FireSat mission, the payload would be required to detect fire through thermal contrast or visible means
  - As we design the payload, there are a number of spin-off requirements for the spacecraft bus, which exists to support the payload
    - ✓ Where and how precisely must the spacecraft point?
    - ✓ What is the spatial resolution of the sensor(s)
    - ✓ How much data must be processed and transmitted?
    - ✓ How much electrical power is required?
    - ✓ Range of operating temperatures for components?

Space Mission Components — Subsystems (Bus)

- The subsystems that make up the spacecraft bus must satisfy all the payload requirements
  - Subsystems Include:
    - ✓ Power
    - ✓ Thermal Control
    - ✓ Propulsion
    - ✓ Navigation
    - ✓ Communication and Data Handling
    - ✓ Attitude Determination and Control
    - ✓ Structure
  - Mission designers define these requirements in terms of performance budgets (not monetary)
    - ✓ e.g. Power Budget, Link Budget (communications), Mass Budget etc.

Space Mission Components — Spacecraft Integration

- In the spacecraft design process all the payload and subsystems are integrated
- If the design of one subsystem is adjusted, it is likely that some, or all, of the other subsystems will also have to be adjusted
- One of the biggest challenges in the design phase is keeping the entire mission in perspective



Space Mission Components — Integration Example

- A planned Mars rover mission was designed to have the capability to investigate 1 km radius from the landing site
  - Scientists wanted to change the capability to 10 km
- The rover was redesigned
  - Originally it was 66 kg and after redesign it was 72 kg
- The results were:
  - The c of g of the lander was higher due to size increase
  - The lander needed another leg for stability
  - Top deck needed redesigning
    - ✓ Fuel tanks and thrusters moved
  - More mass = more thrust needed
    - ✓ Bigger propellant tanks
  - Larger launch vehicle required
  - Heating shields expanded etc. etc.

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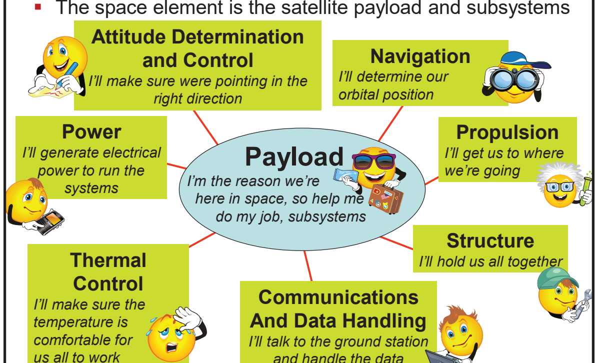
Space Mission Components — Space Element

- The space element is the satellite payload and subsystems



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PHE 255
Introduction to Space Science

Module 7: Propulsion
Part 1: Rocket Science

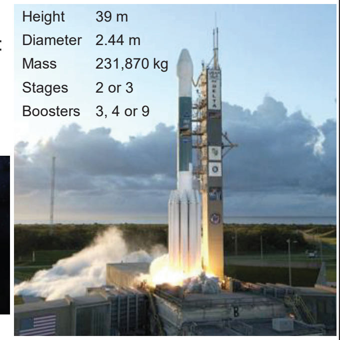
- Propulsion Basics
- Rocket Performance
- Launcher Staging

Dr. Ron Vincent,
Royal Military College of Canada
Department of Physics and Space Science

Propulsion Basics

- Spacecraft propulsion consists of two elements:
  - Getting into space (Launcher)
  - On Orbit Manoeuvres (Satellite Propulsion)

Dawn visited Vesta and Ceres in the asteroid belt with an ion propulsion system



Height	39 m
Diameter	2.44 m
Mass	231,870 kg
Stages	2 or 3
Boosters	3, 4 or 9

Delta II launch of the Dawn spacecraft in 2007

Propulsion Basics — Functions of a Propulsion System

- There are four primary functions of the propulsion system
  - Launch: To lift the launch vehicle and its payload from the launch pad, and to place the payload in Low-Earth Orbit (LEO) or higher
  - Orbit insertion: To transfer the payload from LEO into higher orbits necessary for mission operations
  - Orbit maintenance and maneuvering: Maintaining the desired mission orbit or moving to other desired orbits
  - Attitude control: To provide torque to maintain desired spacecraft orientation
- The three most important qualities that define rocket performance are thrust (F_{thrust}), specific impulse (I_{sp}) and delta v (Δv)
  - We will study these aspects in this lecture

Propulsion Basics — Newton's First Law

- Newton's First Law of Motion: A body continues in its state of uniform motion, or state of rest, unless acted upon by an external force.
- Linear momentum is the amount of resistance an object in motion has to changes in speed or direction

$$\bar{p} = m\bar{v}$$

Velocity ($m \cdot s^{-1}$)

Linear momentum ($kg \cdot m \cdot s^{-2}$)

Mass (kg)

- A satellite in orbit will continue in that orbit unless acted upon by an external force such as thrust from a propulsion system or orbital perturbations (drag, solar radiation pressure, oblate Earth, 3rd body effects)

Propulsion Basics — Newton's Second Law

- Newton's Second Law of Motion: The time rate of change of an object's momentum equals the applied force

$$\bar{F} = \frac{d\bar{p}}{dt} \rightarrow \bar{F} = \frac{d(m\bar{v})}{dt} \rightarrow \bar{F} = \boxed{\bar{v}\frac{dm}{dt}} + m\frac{d\bar{v}}{dt}$$

If the mass is constant this term goes to zero

$$\therefore \bar{F} = m\frac{d\bar{v}}{dt} \rightarrow F = m\bar{a}$$

On a spacecraft the mass changes when fuel is consumed → we will look at this in a few slides

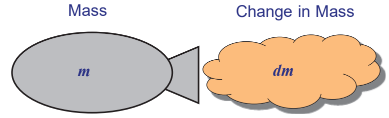
---

Propulsion Basics — Newton's Third Law

- Newton's Third Law of Motion: For every action there is an equal and opposite reaction.
  - This is the basic principle for the thrust provided by rocket engines
    - ✓ The rocket ejects mass in one direction, propelling the vehicle in the opposite direction

Mass

Change in Mass



← dv

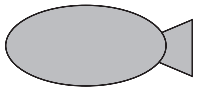
v_e →

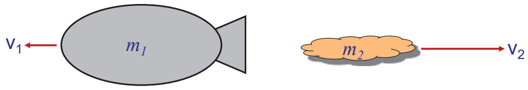
Change in Velocity

Exhaust Velocity

Propulsion Basics — Conservation of Momentum

- Conservation of momentum: Total amount of momentum in a system remains constant





$\bar{p}_1 = m_1\bar{v}_1$ $\bar{p}_2 = m_2\bar{v}_2$

$\bar{p}_1 + \bar{p}_2 = 0$

Final Momentum = 0

Propulsion Basics — Momentum Thrust

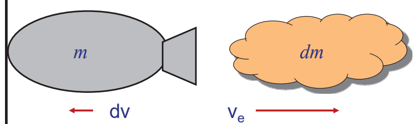
- Recall Newton's 2nd Law: $\bar{F} = \bar{v}\dfrac{dm}{dt} + m\dfrac{d\bar{v}}{dt}$

$$m\frac{dv}{dt} = \frac{dm}{dt}v_e \quad \text{where} \quad \frac{dm}{dt} \approx \frac{\Delta m}{\Delta t} = \dot{m}$$

Mass of fuel burned

mass flow rate

Burn time



$\therefore F = \dot{m}v_e$ Momentum thrust

Rocket Performance — Thrust

- For a chemical rocket, F_{thrust}, is the total force applied to a space vehicle due to momentum thrust and pressure thrust
  - Pressure thrust is a function of the combustion chamber pressure and nozzle design
  - Will effect v_e

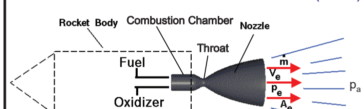
Propellant mass flow rate

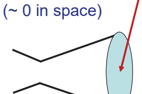
Propellant exhaust velocity

Units = Newtons $F_{thrust} = \dot{m}v_e + (p_e - p_a)A_e$ Nozzle exit area (m^2)

Gas Pressure at nozzle exit area (N/m^2)

Ambient Pressure (~ 0 in space)





Rocket Performance — Thrust

- Total thrust of a rocket can be thought of as Momentum Thrust (left hand side of equation) + Pressure Thrust (right hand side of the equation)

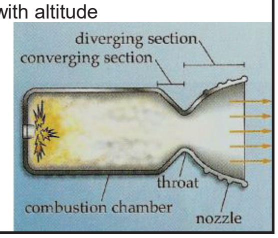
$$F_{thrust} = \dot{m}v_e + (p_e - p_a)A_e$$

- Higher mass flow = greater thrust
- p_a decreases exponentially with altitude
  - ✓ Equal to zero in orbit
- p_e is governed by the Nozzle Efficiency (NE)

$$NE = \frac{A_e}{A_t}$$

A_e → Nozzle exit area

A_t → Area of the nozzle throat

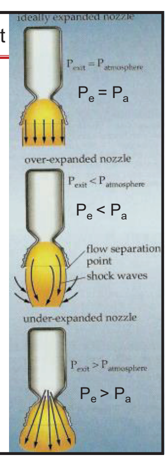


Rocket Performance — Thrust

- The thrust equation implies that thrust is higher in a vacuum since p_a goes to zero, however, the highest v_e and occurs when $p_e = p_a$, resulting in maximum F_{thrust}

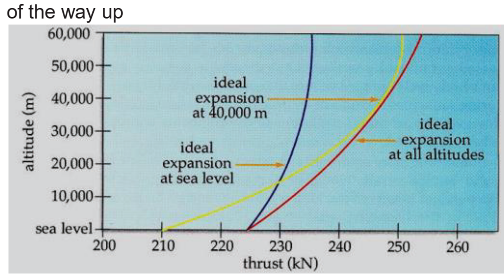
$F_{thrust} = \dot{m}v_e + (p_e - p_a)A_e$ Maximum when $p_e = p_a$

- Over-expanded nozzle:
  - ✓ $p_e < p_a$, reduced v_e and F_{thrust}
- Under-expanded nozzle:
  - ✓ $p_e > p_a$, reduced v_e and F_{thrust}
    - ➢ Spacecraft rocket engines on an orbiting satellite work in a vacuum where $p_e > p_a$, so designers use the greatest practical expansion ratio



Rocket Performance — Thrust

- For launch vehicles the atmospheric pressure decrease with altitude so $p_e = p_a$ cannot be maintained
- Typically, rocket nozzes are designd for ideal expansion 2/3 of the way up



The example above shows the rocket thrust between the surface and 60,000 m, showing ideal expansion at the surface, 40,000 m and all altitudes

Rocket Performance — Specific Impulse

- Specific impulse, I_{sp} , is a measure of the energy content of the propellants and how efficiently it is converted into thrust
  - Represents rocket efficiency
  - Higher I_{sp} = more efficient rocket,
  - I_{sp}, is the ratio of the thrust to the mass flow rate of the propellant

Units = seconds
$$I_{sp} = \frac{F_{thrust}}{\dot{m} g_o}$$
g_o = 9.8 m/s²

or
$$F_{thrust} = I_{sp} \dot{m} g_o$$

Rocket Performance — Δv

- An important measure of propulsion system performance capability is the velocity change, *Δv*, that it can produce
  - Quantified by the *Rocket Equation*

$$\Delta v \equiv g_o I_{sp} \ln\left(\frac{m_i}{m_f}\right)$$
m_i: Initial vehicle mass
m_f: Final vehicle mass

- The rocket equation tells us the amount of velocity change the propulsion system can impart on the spacecraft for the duration of the mission
  - *Δv* is used for every maneuver → need to know how much is required to perform the entire space mission

Rocket Performance — Δv

- Re-arranging the equation on the previous slide allows us to calculate the mass of the propellant, m_p, required for a given increment of *Δv*
  - Can be based on the initial mass of the spacecraft, m_i :

$$m_p == m_i \left[1 - e^{-\left(\Delta v / I_{sp} g_o\right)}\right]$$

  - Or the final mass of the spacecraft, m_p :

Note: $m_i - m_f$ represents the mass of the propellant used

$$m_p = m_f \left[e^{\left(\Delta v / I_{sp} g_o\right)} - 1\right]$$

Launcher Staging

- Launcher Staging
  - Over 80% of a typical launch vehicle is propellant
    - ✓ 15% are structures, tanks, etc.
    - ✓ 5% is the payload
  - One way of reducing the vehicle's mass on the way to orbit is to discard mass that is no longer needed
  - Stages consist of propellant tanks, rocket engines and other supporting subsystems
    - ✓ As the propellant is used up in a stage, it is dropped off
    - ✓ Mass decreases with each stage dropped, so a smaller engine can keep the vehicle on track to get into orbit

Launcher Staging

- The Δv of a multi-staged rocket is the sum of the Δv associated with each stage

$$\Delta v_{total} = \Delta v_{stage\_1} + \Delta v_{stage\_2} + \Delta v_{stage\_3} + \; . \; . \; .$$

$$\Delta v_{total} = g_o I_{sp\_stage\_1} \ln\left(\frac{m_{i\_stage\_1}}{m_{f\_stage\_1}}\right) + g_o I_{sp\_stage\_2} \ln\left(\frac{m_{i\_stage\_2}}{m_{f\_stage\_2}}\right) + g_o I_{sp\_stage\_n} \ln\left(\frac{m_{i\_stage\_n}}{m_{f\_stage\_n}}\right) + \; . \; . \; .$$

$m_{i\_stage\_1}$: initial mass of vehicle

$m_{f\_stage\_1}$: $m_i - m_{propellant\_stage\_1}$

$m_{i\_stage\_2}$: $m_{f\_stage\_1} - m_{structure\_stage\_1}$

$m_{f\_stage\_2}$: $m_{i\_stage\_2} - m_{propellant\_stage\_2}$

Launcher Staging

- Staging Advantages
  - Reduces the vehicle's total mass for a given payload and Δv requirement
  - Increases the total payload mass delivered to space for the same sized vehicle
  - Increase the total velocity achieved for the same size vehicle
  - Decreases the engine efficiency required to deliver a same-sized payload into orbit
- Staging Disadvantages
  - Increased complexity
  - Decreased reliability
  - Increased total cost

  $\}$ Due to added engines
  - Diminishing returns beyond two stages
    - ✓ Performance gains overshadowed by complexity and cost

PHE 255

Introduction to Space Science

Module 7: Propulsion

Part 2: Propulsion Systems

- Chemical Rockets
- Electric Rockets
- Other Systems
- Launch Vehicles

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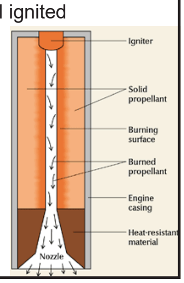
Chemical Rockets

- Chemical rockets are a type of thermal rocket that transfer thermodynamic energy to a propellant and then converts the energized propellant into high speed exhaust using nozzles
- The majority of rockets in use today are chemical
  - These are the only rockets with enough thrust to get a vehicle into orbit
  - Require a fuel and oxidizer (or catalyst) for combustion
- Types of chemical rockets include:
  - Solid propellant rockets
    - ✓ Used for launch
  - Liquid fuel rockets (Monopropellant and Bipropellant)
    - ✓ Referred to as engines or thrusters
  - Hybrid Systems
    - ✓ Combination of solid and liquid propellants

Chemical Rockets — Solid Propellant Rockets

- Solid Propellant Rockets
  - Fuel and oxidizer are mixed together (called the grain) and cast into a solid material that surrounds a void in the center
    - ✓ Types of grain are charcoal/potassium or zinc/sulfur
    - ✓ Fuel and oxidizer will not combust until ignited
  - Once ignited, a flame travels through the empty central region and consumes the propellant radially outward
  - Byproducts expand through a nozzle
  - ☺ Simple, reliable, cheap
  - ☹ Performance not adjustable, cannot turn off once it starts → safety issues

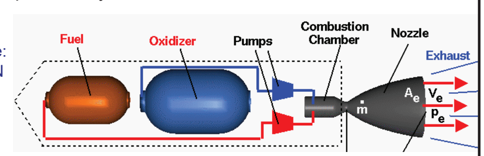
Thrust Range: 50 to 5 × 10^6 N
Vacuum I_{sp} : 280 to 300 S



Chemical Rockets — Bipropellant Rockets

- Liquid Rockets - Bipropellant
  - A liquid oxidizer and a liquid fuel react in a combustion chamber, with the byproducts expanding through the nozzle
    - ✓ Fuel types include liquid hydrogen, kerosene or hydrazine
    - ✓ A common oxidizer is liquid oxygen
    - ☺ High performance, can be throttled and restarted
    - ☹ Complicated system

Thrust Range: 5 to 5 × 10^6 N
Vacuum I_{sp} : 350 to 450 S



Chemical Rockets — SRB/SRMs

- Many launchers use bipropellant rockets for the main engine and then add Solid Rocket Boosters(SRBs) to enhance lift capacity
- SRBs are used during liftoff

Space shuttle solid rocket boosters being jettisoned

Ariane 5 Launcher with solid rocket boosters

Chemical Rockets — Monopropellant Rockets

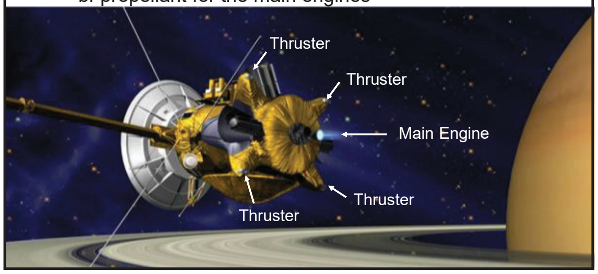
- Liquid Rockets - Monopropellant
  - A single propellant decomposes using a catalyst, releasing heat and creating byproducts that expand through a nozzle
    - ✓ Common fuels are Hydrazine (N_2H_4) and Hydrogen Peroxide (H_2O_2)
    - ✓ The decomposition chamber contains a silver or platinum sponge catalyst
    - ☺ Simple, reliable, cheap
    - ☹ Low performance

Thrust Range: 0.05 to 0.5 N
Vacuum I_{sp} : 150 to 225 S

Monopropellant hydrazine thruster for reaction control of the Mars Viking lander

Chemical Rockets — Example

- Different types of propulsion systems may be used for different applications on the same spacecraft
  - The Cassini probe uses liquid monopropellant thrusters for precise maneuvers (attitude control) and liquid bi-propellant for the main engines

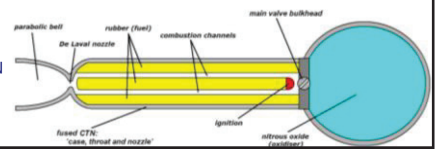


Chemical Rockets — Hybrid Rockets

- Hybrid Rockets
  - Propellants are stored in different forms
  - Fuel is normally a solid and the oxidizer a liquid or gas
    - ✓ Oxidizer is allowed to reach the fuel, and an ignition source is supplied to start the combustion
    - ☺ Safety: Non explosive mixture of fuel and oxidizer
    - ☺ Throttling: Modulate the oxidizer flow to regulate thrust
    - ☺ Restart: Can shut it off and restart it
    - ☹ Low I_{sp}

Thrust Range: 225 to 3.5 × 10^5 N

Vacuum I_{sp}: 225 S



Chemical Rockets — Hybrid Rocket Example

- SpaceShipOne was a suborbital spaceplane that completed the first manned private spaceflight in 2004
  - Won the $10 million Ansari X Prize
  - Used a hybrid rocket motor
- Dream Chaser (in development) will be propelled by twin hybrid rockets that will be fueled by rubber and nitrous oxide

SpaceShipOne

Dream Chaser

Chemical Rockets — Uses

- Uses Chemical rockets of rockets are listed below

Rocket Type	Launch	Orbit Insertion	Orbit Maintenance	Attitude Control
Solid	✓			
Liquid Monopropellant		✓	✓	✓
Liquid Bipropellant	✓	✓	✓	✓
Hybrid		✓		

Electric Rockets

- Electric Rockets use electric or magnetic fields to accelerate a charged propellant
  - More efficient than a thermodynamic rocket
    - ✓ Very high I_{sp} compared to chemical rockets
  - Thrust is low compared to chemical rockets
    - ✓ High Δv total can be reached but it takes extended periods of thrusting to accomplish it
  - Applications include:
    - ✓ Orbit maintenance and manoeuvring, de-orbiting
    - ✓ Attitude Control
    - ✓ Propulsion for small satellites
      - ➢ High velocities achieved through long periods of thrusting

Electric Rockets — Types

- Electric Rocket types include:
  - Electrostatic Thrusters
    - ✓ Gridded Ion Thruster
    - ✓ Hall Effect Thruster (HET)
  - Plasma Thrusters
    - ✓ Pulsed Plasma Thruster (PPT)
  - Electrothermal Thrusters
    - ✓ Resistojet
    - ✓ Arcjet

Dawn: Gridded Ion Thruster

Smart-1: Hall Effect Thruster

Electric Rockets — Electrostatic Thrusters

- For an electrostatic thruster positive ions are accelerated out the back of the spacecraft by an electric field
  - The expelled ions propel the spacecraft in the opposite direction according to Newton's 3rd law

Acceleration of ion (m/s^2)

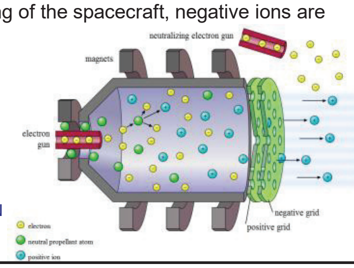
Electrostatic Force (N) ← $\bar{F} = m\bar{a} = q\bar{E}$ → Electric field (V/m)

Mass of ion (kg) Charge on the particle (C)

  - High charge density and strong electric field lead to higher mass flow and greater exhaust velocities

Electric Rockets — Gridded Ion Thruster

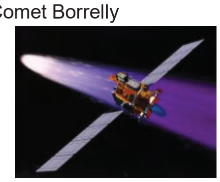
- Gridded Ion Thruster
  - Propellant (typically xenon) is ionized by stripping off the outer electrons through collisions with injected electrons
  - The positive ions are then accelerated (> 30 km/s) through a grid with the application of a strong electric field
  - To prevent charging of the spacecraft, negative ions are injected into the exhaust to neutralize it
  - ☺High I_{sp}
  - ☹Low thrust, complicated



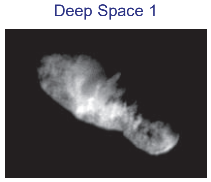
Thrust : 0.02 to 2 N
Vacuum I_{sp} : 5,000 to 10,000 S

Electric Rockets — Gridded Ion Thruster Example

- Deep Space 1 (1998) - Intercepted Comet Borrelly
  - Used an ion thruster
  - I_{sp} ~ 3,000 seconds
  - Maximum thrust ~ 0.09 N
  - The engine fired for 678 total days to achieve high Δv values
  - The ion engine required 2100 W of power


Deep Space 1

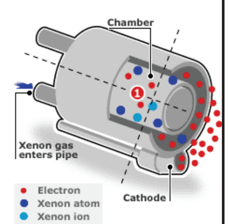

Deep Space-1 Ion Thruster


Comet Borrelly

Electric Rockets — Hall Effect Thruster

- Hall Effect Thruster (HET)
  - The propellant (typically xenon gas) is introduced near the anode, where it becomes ionized through collisions with electrons
  - Ions are accelerated by an electric field towards the cathode at 10 to 80 km/s and exit to create thrust
  - Ions attract electrons suspended by a magnetic field as they are accelerated out the back, which neutralizes the beam
  - ☺ High I_{sp}
  - ☹ Low thrust, complicated

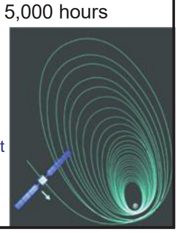
Thrust Range: 0.005 to 3 N
Vacuum I_{sp} : 1,200 to 1,800 S



Electric Rockets — Hall Effect Thruster Example

- SMART-1 (2003) - Went to the Moon
  - Propelled by a solar-powered HET using 82 kg of xenon propellant,
  - I_{sp} was 1,640 s
  - The thruster had a weight of 29 kg with a peak power consumption of 1,200 watts
  - Δv ~ 4 km/s over the thrusting lifetime of 5,000 hours

Orbital path taken by SMART-1 to get to the moon

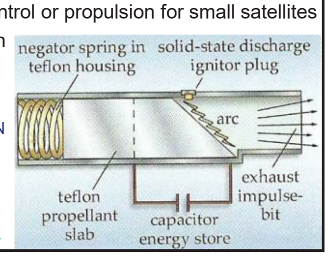


Electric Rockets — Pulsed Plasma Thruster

- Pulsed Plasma Thruster
  - Use an arc of electric current adjacent to a solid propellant, to produce a quick and repeatable burst of impulse
  - High voltage arc vapourizes propellant and creates plasma
  - Precisely controlled, low thrust
    - ✓ Useful for attitude control or propulsion for small satellites
    - ☺ Simple, robust design
    - ☹ A lot of propellant for little thrust

Thrust Range: 0.00001 to 0.001 N
Vacuum I_{sp} : 500- 700 s

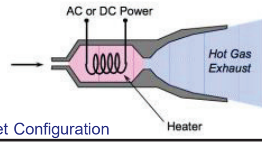
Clydespace PPT for Cubesats



Electric Rockets — Electrothermal - Resistojet

- For electrothermal rockets, heat comes from an electric resistance or a spark discharge inside a chamber
  - Propellant travels through the chamber, absorbs heat and then expands through a nozzle
- Resistojet
  - The resistojet is the simplest form of electric propulsion
    - ✓ Propellant fluid, such as water or nitrous oxide, is superheated over an electrically-heated element, allowing the resulting hot gas to escape through a converging-diverging nozzle
    - ☺ High performance
    - ☹ Low thrust

Thrust Range: 0.005 to 0.5 N
Vacuum I_{sp} : 150 to 700 S

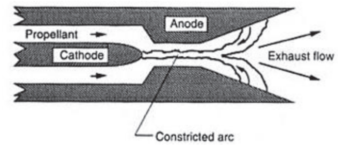


Resistojet Configuration

Electric Rockets — Electrothermal - Arcjet

- Arcjet
  - Electrical discharge (arc) is created in a flow of propellant, which imparts additional energy to the propellant
    - ✓ Propellant is typically hydrazine or ammonia
    - ☺ Gives more energy per kilogram of propellant than the resistojet
    - ☹ More complicated than resistojet, increased power consumption

Thrust Range: 0.05 to 5 N
Vacuum I_{sp} : 450 to 1500 S



Arcjet Configuration

Electric Rockets — Electrothermal Example

- UoSAT-12 (1999) - Comms/Remote Sensing
  - A 60-minute resistojet firing period would raise the 650 km orbit of the spacecraft by 3 km
  - UoSAT-12 carried 2.5 kg of nitrous oxide, sufficient for 14 hours of thrust


UoSAT-12

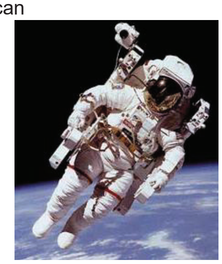
- Advanced Research and Global Observation Satellite (ARGOS)
  - Demonstrated a high-powered electric propulsion provided by a 26 kilowatt ammonia fueled arcjet

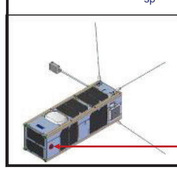

ARGOS

Other Systems — Cold Gas

- Cold Gas
  - Uses mechanical energy of gas stored under pressure
    - ✓ Similar to spraying an aerosol can
  - Simplest form of rocket
  - ☺ Reliable, low cost
  - ☹ Low performance, heavy given performance level

Thrust Range: 0.05 to 200 N
Vacuum I_{sp} : 50 to 75 S


Astronaut Cold Gas Propulsion


CanX-2 used an experimental cold gas propulsion system on a 3U nanosatellite

Other Systems — Solar Sail

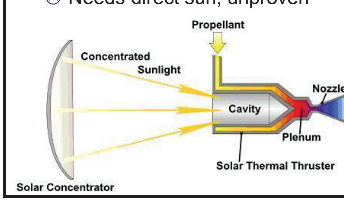
- Solar Sail
  - Uses a large membrane mirrors to take advantage of radiation pressure from the sun
  - ☺ Potentially limitless fuel for propulsion in near sun applications
  - ☹ To produce 5 N of force, the solar sail would have to be 1 km²

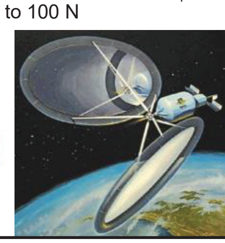

Light photons reflect off the sail, transferring momentum


LightSail-1 is due for launch in 2017 (32 m² sail)

Other Systems — Solar Thermal

- Solar Thermal
  - Lenses or mirrors concentrate solar energy onto a heat transfer chamber
  - Propellant, such as liquid hydrogen, flows through the chamber, absorbs heat and expands through a nozzle
  - ☺ Limitless energy supply if near sun, potentially high I_{sp} (~800s), thrust in the order of 10 to 100 N
  - ☹ Needs direct sun, unproven

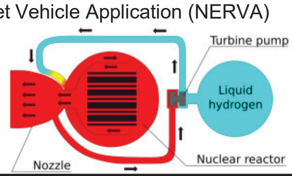




Other Systems — Thermonuclear

- Thermonuclear
  - A working fluid, usually hydrogen, is heated in a high temperature nuclear fission reactor, and then expands through a rocket nozzle
  - ☺ High I_{sp} and thrust, long-term energy supply
  - ☹ Nuclear contamination, shielding required, political problems
  - No thermonuclear rocket has flown in space, though NASA programs have existed in the past
    - ✓ Nuclear Engine for Rocket Vehicle Application (NERVA) and Prometheus projects

Thrust Range: 3000 to 6 × 10^5 N
Vacuum I_{sp} : 700 to 1100 S



Thermonuclear Propulsion

Other Systems — Unproven Systems

- Magnetoplasmadynamic (MPD)
  - Ionized particles are accelerated by the interaction of electric and magnetic fields → Lorentz Force
    - ✓ High I_{sp} and 200 N thrust possible in theory but a lot of power is required to run the system
- Variable Specific Impulse Magnetoplasma Rocket (VASIMR)
  - Uses radio waves to ionize a propellant and magnetic fields to accelerate the resulting plasma to generate thrust
    - ✓ Can vary the I_{sp} depending on mission requirements
- Nuclear Fusion/Fission
  - Achieve thrust through explosive nuclear fusion or fission
- Electrospray
  - RMCC research for small satellite propulsion
    - ✓ Ionized spray ejected through a small nozzle

Launch Vehicles

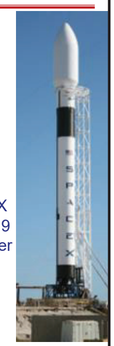
- Launch vehicles are responsible for getting satellites into orbit
- Only a limited number of countries possess the ability to launch a satellite into orbit
  - Russia, United States, France, Japan, China, India, Israel, Iran and North Korea
  - Launch capability range from ~450 kg (Pegasus) to >20,000 kg LEO
  - Heavy launchers include the Delta IV (USA) and Proton-M (Russia)
    - ✓ ~23,000 kg to LEO
  - Note: Saturn 5 could lift 119,000 kg to LEO
    - ✓ Retired in 1973


Delta IV Heavy


Proton-M

Launch Vehicles — Characteristics

- General characteristics of a launch vehicle
  - Thrust-to-weight ratio must be greater than one
    - ✓ Eg. Delta IV Heavy has a mass of 733,000 kg
  - Throttling and thrust-vector control required
  - Optimized nozzle design
  - Navigation, guidance and control subsystem
  - Communication subsystem
  - Electrical power
  - Structures and mechanisms
  - Mass is ~80% propellant, 5% payload
  - Liquid bipropellant rockets
    - ✓ May have solid rocket boosters
- More lift capacity = more fuel = bigger launcher = higher cost
  - $140 to $170 million to launch a Delta IV Heavy


SpaceX Falcon 9 Launcher

Launch Vehicles — Single Stage To Orbit

- Most launchers have stages in order to increase efficiency, however, the ultimate goal is a single-stage-to-orbit (SSTO), re-useable launcher
  - ☺ Operates like an airliner with routine maintenance → reducing launch costs is the main advantage
  - ☹ Technical challenges are difficult to overcome
- The aerospike engine is a possible SSTO
  - Maintains efficiency across a wide range of altitudes


Standard Rocket


Aerospike Engine


X-33 Aerospike Rocket Plane Concept

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PHE 255
Introduction to Space Science

Module 8: Attitude Determination and Control
Part 1: Attitude Dynamics

- Introduction
- Rotational Motion
- Disturbance Torques

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Introduction — Objectives of Attitude Control System

- Attitude Control System Objectives
  1. Point communication antennas in specific directions
     - ✓ Antennas for payload data and telemetry, tracking and control (TT&C) need to point at the ground station
  2. Point payload sensors in designated directions
     - ✓ e.g. Remote sensing payloads need to point at a specific point on the surface
     - ✓ Need to meet sensor pointing accuracy requirements
       - Dependent on mission objectives
  3. Point propulsion system in the correct direction when applying thrust for orbital changes
  4. Stabilize the spacecraft against external torques
     - ✓ Small environmental effects (perturbations) will cause the satellite to drift from its desired attitude

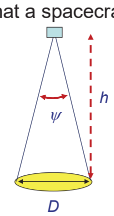
Introduction — Pointing Accuracy

- Typically, attitude control requirements are stated in terms of pointing accuracy, ψ, and slew rate
  - Slew rate is the angular speed in rad/s that a spacecraft can change its attitude

Target distance (m)

Target diameter (m) → $D = h\psi$

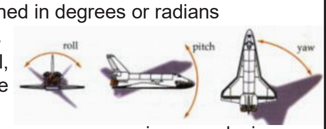
Pointing accuracy (rads)



- The need and accuracy for spacecraft pointing depends on the mission pointing requirements
  - At 500 km altitude, a spacecraft would need a pointing accuracy of 0.115° to hit a 1 km target on Earth

Introduction — Pitch, Roll, Yaw

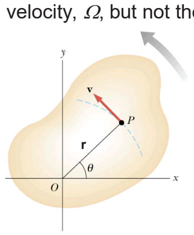
- When describing the motion of an object we use an x, y, z coordinate system coordinate
  - In the case of attitude control the motion is rotational, so coordinates are determined in degrees or radians
  - Space vehicle attitude is described in terms of roll, pitch and yaw around the axes of the body frame



- Satellites generally do not have a nose or wings, so designers pick convenient, preferred directions to define the body frame
- Since the motion is rotational for attitude control we need to know something about the mechanics of a rotating object:
  - Position = θ (degrees), Velocity = Ω (radians/s), Acceleration = α (radians/s^2), Moment of Inertia = I (kg·m^2), Center of Mass = CM (this is a location), Angular Momentum = H (kg·m^2·s^{-2}), Torque = T (N·m)

Rotational Motion — Angular Velocity, Ω

- Every point on a rigid rotating object has the same angular velocity, Ω, but not the same tangential velocity, v



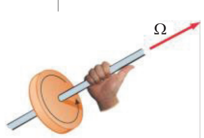
$$\Omega = \frac{d\theta}{dt}$$

Change in θ with respect to time gives the angular velocity

θ, r defines position, P

$$v = r\Omega$$

Tangential velocity v, increases further away from the centre of rotation



Direction of Ω is obtained using the right hand rule

Rotational Motion — Angular Acceleration, α

- Angular acceleration, α, can be determined by taking the derivative of the angular speed with respect to time

$$\alpha = \frac{d\Omega}{dt}$$ Angular acceleration

- Rotational kinematic equations:

$$\theta_f = \theta_i + \Omega_i t + \frac{1}{2}\alpha t^2 \qquad \Omega_f = \Omega_i + \alpha t$$

f = final
i = initial

- Kinetic energy of a rotating mass:

$$KE_{Rotational} = \frac{1}{2} I \Omega^2$$

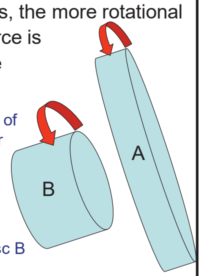
Moment of Inertia (see next slide)

Rotational Motion — Moment of Inertia, I

- Moment of Inertia (I) is the tendency of a body to resist angular acceleration (changes in the angular velocity)
  - Analogous to mass in a linear system
  - Function of the total mass **and** how this mass is distributed
    - ✓ The farther out the object's mass is, the more rotational inertia the object has and more force is required to change its rotation rate
    - ✓ Example:

Discs, A and B, are made of the same material and of equal mass. Disc A is larger in diameter but thinner than Disc B. If both discs have the same angular velocity, which requires more force to change its rotation rate?

More force is needed to change the rotation rate of Disc A $\Rightarrow$ it has a larger moment of inertia than Disc B since more mass is distributed toward the edges



Rotational Motion — Centre of Mass, CM

- There is a special point in a system or object, called the *center of mass* (CM), that moves as if all of the mass of the system is concentrated at that point
- Determining a satellite's *CM* is critical, since rotations will occur around this point
  - Also critical for propulsion
  - Need to place components in the satellite such that the *CM* is located near the central axes (*x,y and z*)

x, y, z coordinates of CM

$$x_{CM} = \frac{1}{M}\int x\,dm \qquad y_{CM} = \frac{1}{M}\int y\,dm \qquad z_{CM} = \frac{1}{M}\int z\,dm$$

Integrate with respect to mass along each axis and multiply by one over the total mass, $1/M$

Rotational Motion — Angular Momentum, H

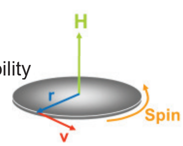
- Angular momentum, H, is the amount of resistance a spinning object has to changes in spin rate or direction

$$\overline{H} = I\overline{\Omega}$$

- We can also describe H in terms of a vector cross-product

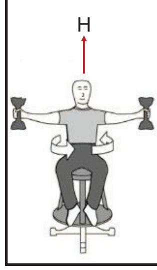
$$\overline{H} = \overline{r} \times m\overline{v} \qquad \overline{a} \times \overline{b} = ab\sin\theta$$

  - Can find the direction of **H** by using the right-hand rule
    - ✓ Fingers of right hand in direction of rotation and thumb will point in direction of H
- By imparting spin to a spacecraft we use angular momentum to help maintain stability
  - ✓ Higher spin means greater stability



Rotational Motion — Conservation of Angular Momentum

- The angular momentum of an isolated system (like a spacecraft) remains constant in both magnitude and direction
  - Conservation of Momentum
- The example below illustrates conservation of momentum



A man sits on a rotating stool holding out two dumbbells.

What happens if he brings the dumbbells towards his body?

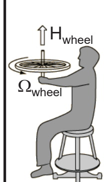
$$\overline{H} = I\overline{\Omega}$$

Bringing in the dumbbells will decrease the moment of inertia, since mass is shifted inward

In order for H to be conserved, the angular velocity must increase

Rotational Motion — Conservation of Angular Momentum

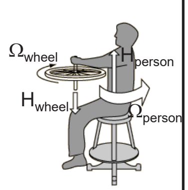
- The sum of the angular momenta of the parts of an isolated system (like a spacecraft) is constant
  - If one part of the system is given an angular momentum in a given direction, then some other part or parts of the system must simultaneously be given exactly the same angular momentum in the opposite direction
    - ✓ See example below



A man on a rotatable stool holds a spinning wheel. What happens if the wheel is turned over?

If the wheel is turned over, its angular momentum is now downward

The man and the stool will rotate with angular momentum twice that of the wheel



Rotational Motion — Torque, T

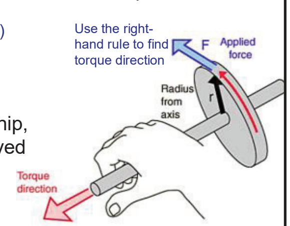
- When force is applied to a rigid body it will rotate about the CM
- The tendency of this force to cause the object to rotate is measured by torque, T
  - Torque is not a force but a consequence of force
  - We can express torque as a vector cross-product

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

$\overline{F}$ — Applied force (N)

$\overline{r}$ — Distance from the center of mass to where the force is applied (m)

  - According to this relationship, more torque can be achieved with the same force by applying the force further from the center of rotation



Rotational Motion — Torque

- Torque can also be written as the time rate of change of angular momentum, which is related to angular acceleration

$$\bar{T} = \frac{d\bar{H}}{dt} = I\bar{\alpha}$$

- If torque is added to the spacecraft, an angular acceleration is created that leads to a change in attitude

Torque T → Spacecraft I → Angular Acceleration α → Leads to a change in angular velocity Ω → Leads to a change in position θ

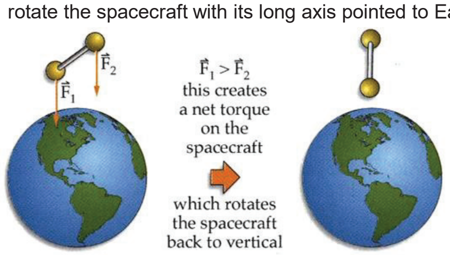
- Torque may come from the spacecraft's attitude control system to give a desired attitude → discussed next lecture
- Unwanted torque may come from the space environment and give an undesired attitude that needs to be corrected → discussed in the next section

Disturbance Torques

- Environmental effects called *disturbance torques* drive a spacecraft from its original attitude
  - Most of these torques are extremely small, but over time they can rotate even the largest spacecraft
- Four main sources of disturbance torques include:
  - Gravity gradient torque
  - Solar radiation pressure torque
  - Earth's magnetic field torque
  - Atmospheric drag torque

Disturbance Torques — Gravity Gradient Torque

- Gravity Gradient Torque
  - Results from the difference in gravitational force exerted on different parts of a spacecraft
  - The slight difference in gravitational force between the upper and lower part of the spacecraft will tend to rotate the spacecraft with its long axis pointed to Earth



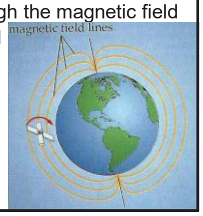
Disturbance Torques — Solar Radiation Pressure Torque

- Solar Radiation Pressure (SRP) Torque
  - Light photons strike exposed surfaces creating differential pressure across the surface of the spacecraft
    - ✓ Causes the spacecraft to rotate
  - Differential pressure is a result of different reflective properties of the spacecraft surface
    - ✓ More reflective surface = higher SRP
    - ✓ Less reflective surface = lower SRP

A typical satellite would reflect light photons differently from its solar panels, bus and protruding antennas leading to rotation

Disturbance Torques — Magnetic Torque

- Magnetic Torque
  - Because of the impact of charged particles in space, the surface of a spacecraft can develop a charge of its own, giving it a distinct dipole
    - ✓ North and a south like a compass
  - Just as a compass needle rotates to align with the Earth's magnetic field, the dipole-charged spacecraft will attempt to do the same when it passes through the magnetic field
  - This is a significant concern for small satellites in LEO, polar orbits where the magnetic field is strongest
  - The effect is far less for large satellites in geostationary orbit



Satellite lining up with the Earth's magnetic field

Disturbance Torques — Atmospheric Drag Torque

- Atmospheric Drag Torque
  - In low Earth orbit, the atmosphere applies a drag force to the vehicle

$$\underbrace{F_{drag}}_{\text{Drag force}} = \frac{1}{2}\underbrace{\rho}_{\text{Atmospheric density}} \overbrace{v^2}^{\text{Velocity}} \underbrace{C_D}_{\text{Coefficient of drag}} \underbrace{A_D}_{\text{Impacted area of drag}}$$

  - Since parts of a spacecraft may have different drag coefficients (e.g. large solar panels), drag forces on different parts of the spacecraft may also differ
    - ✓ This creates a drag torque which can rotate the spacecraft
  - Spacecraft designers can do little to prevent drag torque, so the attitude control system has to deal with it

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Introduction to Space Science

Module 8: Attitude Determination and Control
Part 2: Sensor and Actuators

- Introduction
- Attitude Determination
- Attitude Control
- ADCS Controller

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Introduction

- A space mission that requires pointing will need:
  - **Attitude Determination**
    - ✓ Determine the current orientation of the spacecraft in a rotational coordinate system (pitch, roll and yaw)
    - ✓ Attitude determination is accomplished with **sensors**
  - **Attitude Control**
    - ✓ Change the orientation of the spacecraft to counter external torques or to meet operational requirements (point antennas, payload or thrusters)
    - ✓ Attitude control is accomplished with **actuators**
- These two processes form the Attitude Determination and Control System (ADACS) of the spacecraft
  - It is necessary to first determine the spacecraft orientation with sensors before changing it with actuators

Attitude Determination — Sensors

- For a spacecraft there are 3 types of out the window sensors
  - Earth Sensors
  - Sun Sensors
  - Star Sensors

  Need a visible reference
  - ☹ These sensors can accurately measure attitude in only two of pitch, roll and yaw
    - ✓ Need more than one of these sensors or combine with another type of sensor for equal accuracy in all 3 axes
- Spacecraft attitude can also be determined by sensors that do not require visible references, including:
  - Magnetometer
  - Gyroscopes
    - ✓ Mechanical
    - ✓ Ring Laser/Fibre Optic

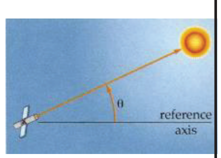
    Not appropriate for small satellites

Attitude Determination — Earth Sensors

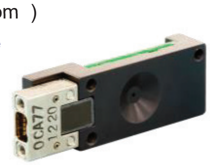
- Earth Sensors
  - In LEO the Earth fills a significant portion of the sky, so a sensor would have to focus on a small portion of the Earth for greater accuracy
    - ✓ Sensors that scan for the Earth horizon can be more accurate
- Earth sensor example (CubeSatShop.com)
  - Infrared Earth horizon sensor
  - Dimensions: 4.33 x 3.18 x 3.18 cm
  - Mass: 33g
  - Operating voltage: 3.3V
  - Coarse field of view: 60 deg
  - Resolution (coarse FOV): 1 deg
  - Fine field of view: >7 deg
  - Resolution (fine FOV): <0.25 deg

Attitude Determination — Sun Sensors

- Sun Sensors
  - Finds the sun and determines its direction with respect to the spacecraft
  - Most widely used spacecraft attitude sensor

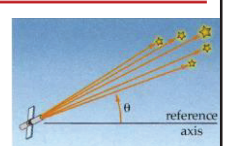


- Sun sensor example (CubeSatShop.com)
  - Six sensors can achieve full sky coverage
  - Field of view: 114°
  - Update Rate: >10 Hz (limited by ADC)
  - Accuracy: <0.5°
  - Mass: < 5 g
  - Power: < 10 mA
  - Size: 33mm x 11mm x 6mm
  - Operating temperature: -25°C to +50°C
  - 10krad total dose (component level)



Attitude Determination — Star Sensors

- Star Sensors
  - Compares the pattern of stars seen by the sensor to a star catalog and uses astrometry to determine where the sensor is pointed
  - More accurate than a sun sensor



- Star sensor Example (CubeSatShop.com)
  - 5 VDC required
  - Peak current: 0.2 A
  - Weight with baffle: 245 g
  - Dimensions with baffle: 50 mm x 50 mm x 113 mm
  - Operating temperature: -30 deg C to +60 deg C
  - Accuracy: <0.02 deg
  - Acquisition time: <1 sec

Attitude Determination — Magnetometer

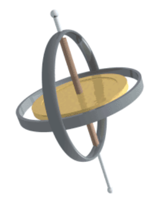
- Magnetometer
  - A magnetometer functions as a highly accurate compass that measures the direction and strength of the local magnetic field
    - ✓ Earth's magnetic field ranges from 0.25 to 0.6 Gauss at the surface
  - By comparing this measurement to a model of the Earth's field it can determine the spacecraft's attitude
- Magnetometer example (CubeSatShop.com)
  - Accuracy: +/- 1°
  - Measurement range: 60,000 nT
  - Sensitivity: 6.5 nT
  - Update rate: up to 10Hz
  - Power consumption: <700 mW
  - Size: 96 x 43 x 17mm, Mass: <200g
  - Operating temperature: -35°C to +75°C
  - 10 krad total dose

Attitude Determination — Mechanical Gyroscope

- Mechanical Gyroscope
  - The simplest type of gyroscope is a spinning mass
  - With no torque applied it will always point in the same direction
  - Higher moment of inertia (I) and higher angular velocity (Ω) will result in greater stability
    - ➢ Higher angular momentu

$$\overline{H} = I\overline{\Omega}$$

  - With torque applied, a mechanical gyroscopes will precess in a predictable direction with a predictable magnitude



Gyroscope Precession

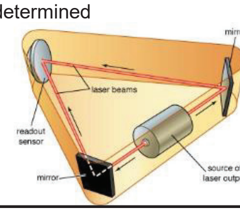
Attitude Determination — Mechanical Gyroscope

- Mechanical Gyroscope (continued)
  - There are two methods to measure spacecraft rotation due to external torques using a mechanical gyroscope
    - ✓ Mount the gyroscope directly to the spacecraft frame
      - ➢ When the spacecraft rotates the gyroscope will precess in a predictable fashion
      - ➢ By measuring the precession angle and rate, the system can compute the spacecraft's rotation
    - ✓ Isolate the gyroscope from external torques by mounting it on a gimbal
      - ➢ Measure spacecraft rotation with respect to the stationary gyroscope

Gimballed Gyroscope

Attitude Determination — Ring Laser/Fibre Optic Gyros

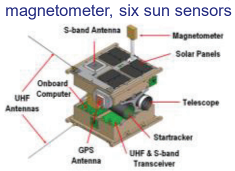
- Ring Laser Gyroscope
  - Consists of a circular cavity containing a closed path, through which two laser beams shine in opposite directions
    - ✓ When beams meet, an interference pattern is formed
  - As the spacecraft rotates the path lengths traveled by the beams change, causing a shift in the interference pattern
  - By measuring shift of the interference pattern, changes in the vehicle's orientation can be determined
  - ☺ No moving parts, good accuracy
- Fibre Optic Gyroscope
  - Same idea as the Ring Laser Gyrospcope but with fibre optics to guide laser instead of mirrors



Attitude Determination — Examples

- BRIght-star Target Explorer (BRITE) 20 × 20 × 20 cm cubesat:

  Attitude determination accuracy of 10 arc seconds is made possible with a magnetometer, six sun sensors (1 per side) and a star tracker



1 degree = 1/60 of a circle ~ 17.5 rad
1 arc minute = 1/60 of a degree ~ 290.9 μrad
1 arc second = 1/60 of an arc minute ~ 4.5 μrad

- Hubble Telescope (13.2 m × 4.5 m diameter)

  Able to lock onto a target without deviating more than 0.007 arc seconds, which is about the width of a human hair seen at a distance of 1.6 km

  Hubble has 6 gyroscopes, each of which consists of a wheel spinning at 320 revolutions per second inside a sealed cylinder

Attitude Control — Actuators

- Once the spacecraft orientation is determined, we may need to change it to a different orientation
- Actuators provide torque on demand to rotate a spacecraft as needed to collect data, downlink data, compensate for disturbance torques, apply thrust or meet other mission requirements
- Passive Actuators
  - Gravity-gradient stabilization
  - Spin or dual-spin stabilization

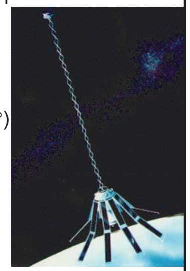
  } Require little or no input
- Active Actuators
  - Thrusters
  - Momentum Control Devices
  - Magnetic Torquers

  } Require continuous feedback and adjustment

Commonly use more than one type of actuator for attitude control

Attitude Control — Gravity-Gradient

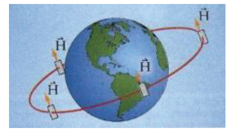
- Gravity-Gradient Stabilization
  - Takes advantage of the gravity-gradient disturbance torque discussed last lecture
  - Can exploit this free torque to keep a spacecraft oriented in a vertical orientation
    - ☺ Cheap and simple
    - ☹ Pitch and roll only (no yaw)
    - ☹ Limited accuracy depending on spacecraft's mass distribution ($\pm$ 5°)
      - ➢ These satellites are usually dumbbell shape
    - ☹ Used in LEO only



Example: GeoSat Radar Altimeter
Launched in 1985 to measure sea surface height
Used gravity gradient to keep radar altimeter pointing at the Earth

Attitude Control — Spin Stabilization

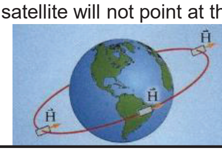
- Spin Stabilization (Passive)
  - Spin the entire satellite to create angular momentum to maintain a constant inertial orientation of one of its axes



$$\overline{H} = I\overline{\Omega}$$

Angular momentum, *H*, will stabilize the spacecraft

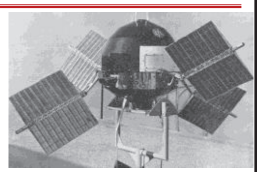
☹ Spin stabilization isn't useful for Earth pointing missions since the satellite will not point at the Earth for part of the orbit



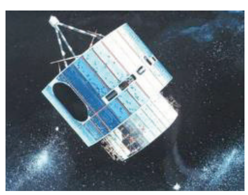
For a spinning spacecraft angular momentum will point in the same direction

Attitude Control — Spin Stabilization

- Spin Stabilization Examples
  - Explorer 6 (1959)
    - ✓ Designed to detect radiation
    - ✓ Spin stabilized at 168 rpm
  - Geostationary Operational Environmental Satellite (GOES)
    - ✓ GOES 1 to 7 (1975 to 1987) were spin stabilized at 100 RPM
    - ✓ Visible and Infrared Spin Scan Radiometer used the spin of the satellite to scan the Earth



Explorer-6



GOES-2

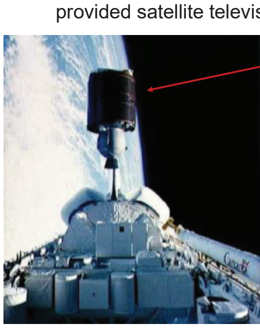
Attitude Control — Dual-Spin Stabilization

- Dual-Spin Stabilization (Passive)
  - One way to avoid Earth-pointing limitations of spin stabilization is to use a dual-spin system
    - ✓ Outer section spins to achieve high angular momentum
    - ✓ Inner section is 'despun' so it can point at the Earth
    - ☺ Appropriate for a geostationary comms satellite

Dual spin geostationary satellite with 'despun' inner section pointing at Earth

Attitude Control — Dual-Spin Stabilization

- Dual-Spin Stabilization Example
  - Anik-C Series (1983 - 2002) geostationary orbit, provided satellite television to Canada

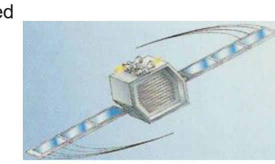


Anik C-2 deployment from the Space Shuttle

Anik C-2 coverage

Attitude Control — Thrusters

- Thrusters (Active)
  - Thrusters are rockets that rotate the spacecraft
    - ✓ Apply a balanced force with a pair of rockets on opposite sides of a spacecraft to produce torque
    - ✓ By varying thruster power, the satellite can be rotated in any direction
    - ✓ Placing the thrusters as far from the satellite's center of mass as possible allows them to exert a greater torque
    - ☺ Can produce well-defined torque on demand allowing a spacecraft to slew quickly from one attitude to another
    - ☹ Amount of propellant required limits their use

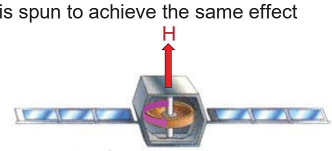


Attitude Control — Momentum-Control Devices

- Momentum-Control Devices (Active)
  - Small rotating masses in the spacecraft may serve two purposes
    - ✓ Give the spacecraft gyroscopic stiffness
      - Stable pointing
    - ✓ Change the angular momentum of the spacecraft by changing the rotation rate of the small rotating mass
      - Reaction Wheels
- Three types of momentum-control devices are:
  - Biased momentum system
  - Zero-biased system
  - Control-moment gyroscope

Attitude Control — Biased Momentum System

- Biased Momentum System (Active)
  - Simplest type of momentum-control device
  - Uses one or two spinning momentum wheels
    - ✓ Because the wheels are spinning rapidly, they give the spacecraft a large angular momentum vector
  - Similar to spin-stabilization, except instead of spinning the whole spacecraft only a small wheel inside the spacecraft is spun to achieve the same effect

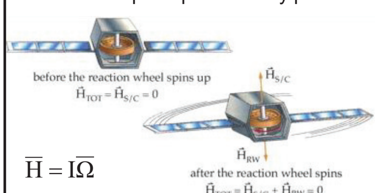


Wheel inside satellite spins, imparting angular momentum on the satellite

$$\overline{H} = I\overline{\Omega}$$

Attitude Control — Reaction Wheel

- Reaction wheel
  - A reaction wheel is a spinning mass used to impart changes in spacecraft attitude
    - ✓ Utilizes conservation of angular momentum
    - ✓ Since the spinning mass is a small fraction of the spacecraft's total mass, easily-measurable changes in its speed provide very precise changes in attitude



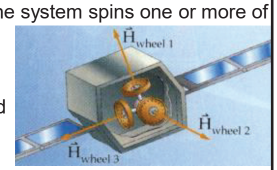
$$\overline{H} = I\overline{\Omega}$$

In this simple single-axis example we start with a non-rotating spacecraft that has zero angular momentum

To rotate the spacecraft in one direction, the reaction wheel is spun up in the opposite direction

Attitude Control — Zero-Biased System

- Zero-Biased System (Active)
  - A zero-biased system has 3 independent reaction wheels at right angles to each other with little or no initial momentum
  - When the spacecraft needs to rotate to a new attitude, or to absorb disturbance torques, the system spins one or more of these wheels
    - ☺ Provides precise attitude control
    - ☹ Complex, expensive, limited operational lifetime



- Example - SABRE ADCS

The MAI-100 is a 3-axis (3 reaction wheels) ADCS in a 10 x 10 x 7.5 cm enclosure, 865 g, 8°/sec slew rate, 5 Hz update, pointing accuracy = 1°, max T = 0.635 mNm

Includes external 3-axis magnetometer and sun sensors

Attitude Control — Control-Moment Gyroscope

- Control-Moment Gyroscope (Active)
  - Consists of 3 or more spinning reaction wheels, each mounted on gimbals that allow them to rotate in all directions
  - Momentum is changed by changing the magnitude and direction of the spinning wheels
  - Provide pointing accuracy equivalent to the zero-biased system, but offer higher skew rates
    - ✓ Used on large satellites

Gimballed reaction wheel

Attitude Control — Momentum Dumping

- A limitation of all momentum-control devices is that there is a practical limit to how fast a given wheel can spin
  - During operation, these systems must gradually spin faster and faster to rotate the spacecraft and absorb disturbance torques
    - ✓ Wheels become saturated
- Momentum dumping is a technique for decreasing the angular momentum of a reaction wheel by applying a controlled torque to the spacecraft
  - ✓ Spacecraft needs an independent way of applying an external torque to slow down the reaction wheel
  - ✓ Magnetic torquers and/or thrusters are normally used for momentum dumping

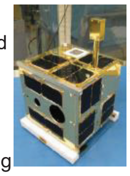
Attitude Control — Magnetic Torquers

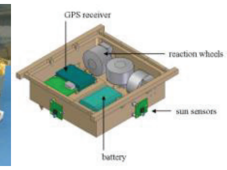
- Magnetic Torquers (Active)
  - Takes advantage of Earth's magnetic field
  - System switches electromagnets on and off as needed
    - ✓ Electromagnet aligns with the Earth's magnetic field, dragging the spacecraft with it
  - Important means of momentum dumping for reaction wheels for satellites in highly inclined LEO
  - Used by small satellites in LEO as primary attitude control actuator
    - ☺ Cheap and simple
- Magnetic Torquer Example (CubeSatShop.com)

Length: 7 cm Diameter < 9 mm
Mass: 0.6 kg
Power: 200 mW
Magnetic Moment: 0.2 Am^2

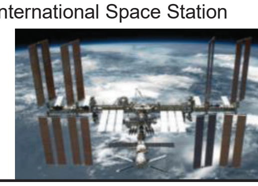
Attitude Control — Examples

- Attitude Control Example: BRITE (5 kg)
  - Three orthogonal reaction wheels and three orthogonal magnetic torquer coils for three-axis attitude control and momentum dumping





- Attitude Control Example: International Space Station (450,000 kg)
  - Four control moment gyroscopes, attitude control system thrusters



Attitude Control — Summary of Capabilities

Type	Pointing Options	Attitude Maneuverability	Typical Accuracy	Lifetime Limits
Gravity-Gradient	Earth local vertical only	Very limited	±5 deg (2 axes)	None
Pure Spin Stabilization	Inertially fixed any direction	Repoint with precession maneuvers; very slow with torquers, faster with thrusters	±0.1 deg to ±1 deg in 2 axes (proportional to spin rate)	Thruster propellant (if applies)*
Dual-Spin Stabilization	Limited only by articulation on despun platform	Same as above	Same as above for spun section. Despun dictated by payload reference and pointing	Thruster propellant (if applies)* Despun section bearings
Bias Momentum (1 wheel)	Local vertical pointing or inertial targets	Fast maneuvers possible around momentum vector Repoint of momentum vector as with spin stabilized	±0.1 deg to ±1 deg	Propellant (if applies)* Life of sensor and wheel bearings
Magnetic Torquer	Any, but may drift over short periods	Slow (several orbits to slew); faster at lower altitudes	±1 deg to ±5 deg (depends on sensors)	Life of sensors
Zero Momentum (thruster only)	No constraints	No constraints High rates possible	±0.1 deg to 5 deg	Propellant
Zero momentum (3 wheels)	No constraints	No constraints	±0.0001 deg to ±1 deg (determined by sensors and processor)	Propellant (if applies)* Life of sensors and wheel bearing
Zero Momentum (CMG)	No constraints Short CMG life may require high redundancy	No constraints High rates possible	±0.001 deg to ±1 deg	Propellant (if applies)* Life sensors and CMG bearings

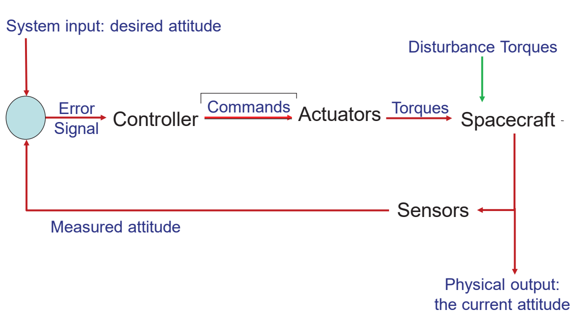
NOTE: Momentum wheels require momentum dumping
- Thrusters may be used for momentum dumping at all altitudes
- Magnetic torquers generally used only in LEO for momentum dumping

ADCS Controller — Types of Control

- The ADCS controller generates commands for the actuators to make the spacecraft point in the right direction
- Integral Control → Predict disturbance torques
  - Controller determines change in attitude over time
  - Controller then calculates how much torque to add in a steady-state mode to compensate for disturbance torques
    - ✓ Used when highly accurate pointing is desired
- Derivative Control → Respond to disturbance torques
  - Sensors determines current attitude and compares it to desired attitude
  - The difference between the measured and desired attitude is the error signal
  - The controller then sends a message to the actuators to correct to the desired attitude (see next slide)

ADCS Controller — Derivative Control

- Derivative Control is shown below



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PHE 255
Introduction to Space Science

Module 9: Power

Part 1: Solar Arrays

- Introduction to Power Systems
- Solar Cells
- Solar Array Sizing

Dr. Ron Vincent,
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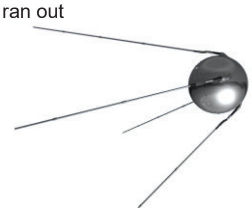
Introduction to Power Systems

- Power is the most fundamental requirement for a payload
  - Almost all others failures can be worked around but loss of power results in the loss of a space mission
- Several early satellite systems failed due to the power system
- Even today the power system may still cause problems
  - Anik F1 satellite is losing power at a very fast rate
    - ☹ The solar panels of Anik F1 degraded more rapidly than expected

Anik F1

Introduction to Power Systems

- Early satellites were power limited
  - Powered by batteries
  - When batteries died, satellite died
  - Example:
    - ✓ Sputnik-1 had 3 silver-zinc batteries
    - ✓ 1 W of power transmitted signals at 20 and 40 MHz for 22 days until batteries ran out



Sputnik-1, launched in 1957 by the USSR, was the first satellite in space

It was an 84 kg sphere 58 cm in diameter with 4 radiating antennas

Introduction to Power Systems

- Introduction of solar cells allowed space missions to be extended
  - Example:
    - ✓ Launched in 1958, Vanguard 1 had a solar powered transmitter that worked until 1964
    - ✓ Also contained a Mercury battery that worked for 20 days
      - ➢ Not rechargeable

Vanguard 1 is a 1.47 kg aluminium sphere, 152 mm in diameter, that is still in orbit today

A 5 mW, 108.03 MHz transmitter was powered by six solar cells mounted on the body of the satellite

- The advent of rechargeable batteries allowed operations during eclipse periods when no solar energy was available
  - Solar panels re-charge batteries when in daylight

Introduction to Power Systems

- The vast majority of satellites use solar panels for power, however, for deep space probes that venture far from the sun, non solar power options are essential

New Horizons Cassini Voyager-1

New Horizons (Pluto), Cassini (Saturn) and Voyager-1 (edge of solar system) are three examples of satellites powered by Radioisotope Thermoelectric Generators (RTGs)

Introduction to Power Systems

- Typical power requirements for modern satellites can range from a watt to thousands of kilowatts
- Satellites typically have peak power demands during payload operations and downlink periods

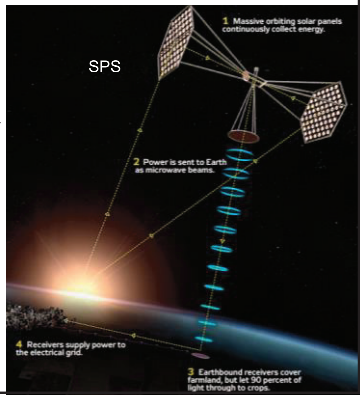
Average and Peak Power for Selected Satellite Missions

Spacecraft type	Average Power, kW	Peak Power, kW
Small Satellites	0.1...0.3	0.2...0.4
GEO ComSats	1.5...5.5	2.0...6.5
LEO ComSats	0.5...0.8	0.7...1.2
LEO Remote Sensing Satellites	2.0...6.5	2.8...8.7
Interplanetary Probes	0.3...0.5	0.8...1.0
Space Shuttle	10...15	13...17
Space Station	25...70	50...150
Micro - satellites	10^{-3}...10^{-1}	10^{-1}... 1

Note: SABRE can produce 10 W with the solar panels and 20 W-hrs with the lithium ion batteries

Introduction to Power Systems

- RADARSAT-2 has power capability of over 2.4 kW
- The Sirius satellites have a total power of 18 kW
- The ISS can demand as much as 150 kW
- If space-based solar power systems (SPS) come to fruition, 5 GW of power may be generated



Introduction to Power Systems

- Types of power sources for satellite missions
  - Solar Cells (Photovoltaic Cells)
    - ✓ Primary
  - Fuel Cells
    - ✓ Primary or secondary
  - Radioisotope Thermoelectric Generators (RTGs)
    - ✓ Primary
  - Nuclear Reactors
    - ✓ Primary
  - Batteries
    - ✓ Secondary

In this lecture we will cover solar cells

These systems will be covered in the next lecture

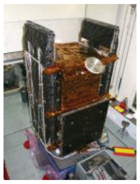
Solar Cells

- Solar Cells
  - Photovoltaic cells convert solar photons to create a flow of electrons (i.e. a current)
    - ✓ Uses the visible spectrum of light
  - Constructed of a thin of a semi-conductor crystal such as silicon (Si) or of gallium arsenide (GaAs)
  - As light photons strike the cell they transmit their energy to the atoms within the cell
    - ✓ Electrons are freed and begin to flow
  - Due to manufacturing limitations, the maximum size of a solar cell is relatively small
    - ✓ Thousands of solar cells may be wired together to form a complete solar array

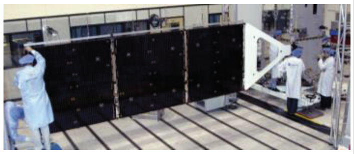
Solar Cells

Example

- Example: RADARSAT-2 solar panels are 13.4 m^2 in total area
  - Each 'wing' is composed of three solar array panels, with each panel carrying a total of 798 solar cell assemblies divided into 14 strings



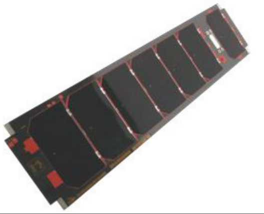
Solar panels folded for launch



Solar panel 'wing' being tested at David Florida labs, Ottawa

Solar Cells

- Silicon solar cells were used on the majority of spacecraft until the mid 1990s
- Today most spacecraft solar cells are GaAs
  - Provide a more efficient energy conversion and have higher operating temperature limits than Si
  - They are also more radiation tolerant than Si cells



7 GaAs solar cells on a 10 × 30 cm side panel for a 3U nanosatellite

4 side panels purchased for SABRE mission

Solar Cells

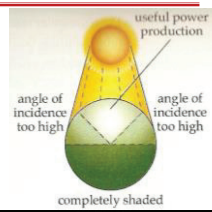
- The intensity of solar energy depends on the distance from the sun
  - Solar flux density decreases with the square of the distance from the sun
    - ✓ Varies between 1317 W/m^2 and 1419 W/m^2 near Earth due to the elliptical orbit around the sun
      - ➢ The average, 1368 W/m^2, is known as the solar constant → we will use 1368 W/m^2 for calculations when considering satellites in geocentric orbits
    - ✓ Solar flux density for other planets:
      - ➢ 2600 W/m^2 at Venus
      - ➢ 585 W/m^2 at Mars
      - ➢ 50 W/m^2 at Jupiter

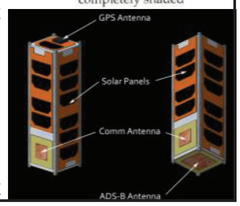
Juno in orbit around Jupiter since 2016 is the most distant mission ever to use solar panels

- Other energy sources must be used for mission that venture to the outer planets of the solar system or beyond

Solar Cells — Body-Mounted Arrays

- Body-Mounted Arrays
  - Body-mounted arrays are used on spacecraft with spin or dual-spin stability
    - ✓ Simple but the Sun shines on less than half the cells at any one time
  - For a cylindrical spacecraft, only $1/\pi$ of the total surface is illuminated at any one time
  - Also common on small satellites
    - ✓ For example, a CubeSat may have solar cells on all faces that may be exposed to the Sun during an orbit





SABRE

Solar Cells — Sun Tracking Solar Arrays

- Sun-Tracking Solar Arrays
  - Higher power output is achieved by keeping the sun's rays perpendicular to the solar array
  - Sun-tracking solar arrays have a control system that move the arrays to keep them pointed directly at the sun
    - ✓ Common method but makes it more complicated

In 2007 a spacewalk to inspect a faulty solar panel on part of the ISS revealed metallic debris that prevented the array from tracking the sun

The ISS solar arrays are 58 m long and have an area of 375 m^2

Solar Cells — Efficiency

- Solar Cells
  - The *solar cell efficiency*, η, indicates the percentage of solar energy that is converted to electrical energy
    - ✓ Si ~ 15% (η ~ 0.15)
    - ✓ GaAs cells ~ 20% (η ~ 0.20)
    - ✓ Multi-junction ~ 25 to 28% (η ~ 0.25 t0 0.28)
  - The potential power output (P_o) per unit array from the array is a function the power density from the Sun

$$P_o = P_{in}\eta$$

    - ✓ Example: Multi-junction with 25% efficiency in LEO
      - ➢ Sun power density near Earth = Solar Constant = 1368 W/m^2

$$\therefore P_o = 1368 \times 0.25 = 342 \ \ \mathrm{W/m^2}$$

Solar Cells — Solar Cell Degradation

- Solar cell performance is degraded by:
  - Temperature
    - ✓ Typical solar arrays lose from 0.025% to 0.075% efficiency per degree Celsius as the temperature increases above 28° C
  - Radiation and Charged Particles
    - ✓ Solar cells can lose 30% of their efficiency over 10 years as a result of ultraviolet radiation and charged particles
      - ➢ Beginning of Life (BOL) power must be high enough so that End of Life (EOL) power is sufficient to run all the spacecraft systems
      - ➢ Any excess power must be shed at BOL

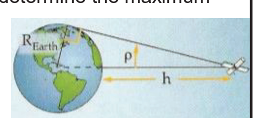
Solar Cells — Eclipses

- Many satellites pass into the Earth's shadow once each orbit
  - Need secondary power during eclipse periods
    - ✓ Batteries are used to supply power during eclipse periods for geocentric orbits
  - Need to charge batteries with solar arrays when the satellite is in sunlight

Eclipse Condition

Solar Cells — Eclipses

- When determining the amount of power needed for a satellite mission it is important to determine the maximum eclipse time for an orbit
- The duration of an orbital eclipse depends mostly on the satellite's altitude, which determines the radius, ρ



$$\rho = \sin^{-1}\left(\frac{R_{Earth}}{h + R_{Earth}}\right)$$

R_{Earth} = 6378 km

h = altitude of satellite above Earth`s surface

- A maximum time in eclipse (TE) for a **circular** orbit is:

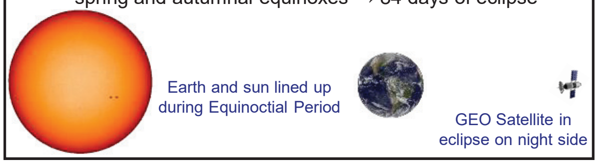
Eclipse times may be shorter due to orbital inclination and sun orientation

$$\boxed{T_E = \frac{2\rho}{360^\circ}\cdot \mathrm{P}}$$

P = Period $\xrightarrow{\text{Recall}}$ $P = 2\pi\sqrt{\frac{a^3}{\mu}}$

Solar Cells — Eclipses - LEO

- Eclipse Examples
  - LEO: ~ 14 eclipses per day, ~ 30 to 40 minutes each
  - Sun synchronous on the terminator: Eclipses only occur during summer and winter solstice periods, max ~ 17 min
  - MEO (e.g. GPS): Eclipses occur in January and July only, ~54 minutes maximum
  - GEO: Eclipses occur during the equinox periods, 70 minutes maximum, declining 21 days before and after the spring and autumnal equinoxes → 84 days of eclipse



Solar Cells — Advantages and Disadvantages

- Solar Cell Advantages:
  - ☺ Can provide abundant power
  - ☺ Safe and clean to launch
  - ☺ Known technology
- Solar Cell Disadvantages
  - ☹ Large and rather fragile - difficult to test on ground
  - ☹ Vulnerable to external failures (solar flares, meteorites)
  - ☹ Not useful in an eclipse
  - ☹ Not useful large distances from sun
  - ☹ Not particularly efficient for their size and mass
  - ☹ Atmospheric drag an issue for solar 'wing' arrays in LEO
  - ☹ Mechanical failures
    - ✓ Large arrays need to deploy
    - ✓ Steering for sun tracking arrays

Solar Array Sizing — Power Requirement

- To size a solar array for a space mission, first determine how much power the solar array must produce for one orbit
  - Required solar array power, P_{sa} , is calculated as:

$$P_{sa} = \frac{\left(\frac{P_e T_e}{X_e} + \frac{P_d T_d}{X_d}\right)}{T_d} \quad \text{Units = W}$$

- ✓ P_e – power requirement during eclipse (W)
- ✓ P_d – power requirement during daylight (W)
- ✓ T_e – time duration of eclipse
- ✓ T_d – time duration of daylight
- ✓ X_e – transfer efficiency (Arrays to batteries ~ 0.65)
  - ✓ Batteries are used for back-up power during eclipse
- ✓ X_d – transfer efficiency (Arrays to loads ~ 0.85)

Solar Array Sizing

- The solar array area , A_{sa} ,is the required solar array power, P_{sa} , divided by the end-of-life power, P_{EOL} ,
  - It is necessary to meet the power requirements at the end of the projected mission lifetime

$$A_{sa} = \frac{P_{sa}}{P_{EOL}}$$

Solar array area (m²) ← A_{sa}; P_{sa} → Solar array power requirement (W); P_{EOL} → End-of-life power (W/m²)

- P_{EOL} is a function of beginning of life power (P_{BOL}) multiplied by the lifetime degradation of the solar array, (L_d)

$$P_{EOL} = P_{BOL} L_d \quad \text{(W/m}^2\text{)}$$

- See next slide for calculating P_{BOL} and L_d

Solar Array Sizing

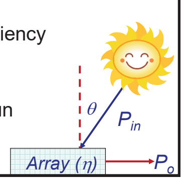
- End-of-life power calculation $P_{EOL} = P_{BOL} L_d$ (W/m²)

where $L_d = (1 - \text{degradatio n/year})^{\text{satellite lifetime}}$

- Degradation/year of a solar cell is typically ~0.5 to 3.75%

and $P_{BOL} = P_o I_d \cos\theta$ (W/m²)

- P_o: Power output per unit area
  - ✓ Recall $P_o = P_{in}\eta$
- I_d: Inherent degradation (no units)
  - ✓ Several cells combined have less efficiency than a single cell
  - ✓ Typical value ~ 0.49 to 0.88
- cos θ : Angle between solar panel and sun
  - ✓ Maximum when θ = 0°
    - ✓ 0° for sun tracking solar arrays



Solar Cell Sizing — Problem

- Calculate the required area of the solar array for the following LEO mission:
  - There is a power requirement of 600 W during eclipse (P_e) and 1200 W during daylight (P_d)
  - 35 minutes of the orbit is in eclipse (T_e) and 65 minutes is in daylight (T_d)
  - The transfer efficiency of arrays to batteries is 0.65 (X_e)
  - The transfer efficiency of arrays to loads is 0.85 (X_d)
  - The efficiency (η) of the GaAs array is 22.09%
  - Inherent degradation (I_d) of the arrays is 0.8
  - The arrays degrade by 2.5% every year
  - The satellite lifetime is 6 years
  - The solar arrays are sun-tracking to ensure maximum illumination(θ = 0°)
  - Solar constant = 1368 W/m²

$$\boxed{P_{sa} = \frac{\left(\frac{P_e T_e}{X_e} + \frac{P_d T_d}{X_d}\right)}{T_d}}$$

$$\boxed{P_o = P_{in}\eta}$$

$$\boxed{P_{BOL} = P_o I_d \cos\theta}$$

$$\boxed{L_d = (1 - \text{degradatio n/year})^{\text{satellite lifetime}}}$$

$$\boxed{P_{EOL} = P_{BOL} L_d}$$

$$\boxed{A_{sa} = \frac{P_{sa}}{P_{EOL}}}$$

$$P_{sa} = \frac{\left(\frac{600 \cdot 35}{0.65} + \frac{1200 \cdot 65}{0.85}\right)}{65} = 19088 \text{ W}$$

$$P_o = 1368 \cdot 0.2209 = 302.2 \text{ W/m}^2$$

$$P_{BOL} = 302.2 \cdot 0.8 \cdot \cos 0 = 241.8 \text{ W/m}^2$$

$$L_d = (1 - 0.025)^6 = 0.86$$

$$P_{EOL} = 241.8 \cdot 0.86 = 207.9 \text{ W/m}^2$$

$$\boxed{A_{sa} = \frac{1908.8 \text{ W}}{207.9 \text{ W/m}^2} = 9.2 \text{ m}^2}$$

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Introduction to Space Science

Module 9: Power

Part 2: Power Systems

- Fuels Cells
- RTGs
- Nuclear Reactors
- Batteries
- Power System Design

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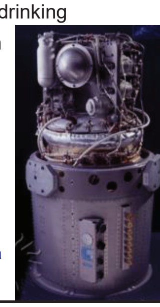
Fuel Cells

- Fuel Cells
  - Rely on energy released in reaction of oxygen and hydrogen
  - By-product is water which can used for drinking
  - Fuel cells have been a power source on every manned spacecraft launched by NASA



An Apollo spacecraft carried three hydrogen-oxygen fuel cells in the service module

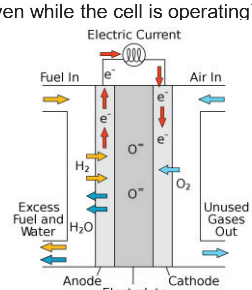
1.1 m in length × 0.56 m in diameter

Each unit contained 31 individual fuel cells connected in series

Normal power output is 563 to 1420 watts, with a maximum of 2300 watts

Fuel Cells

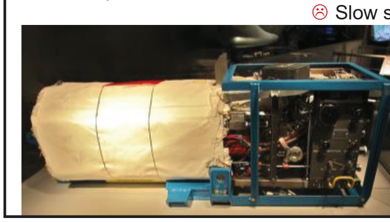
- A fuel cell is like a battery, converting energy released in a chemical reaction to produce electrical power
- A fuel cell supplies current as long as chemical reactants are available or replenished (even while the cell is operating)
- Fuel cells produce several times as much energy per equivalent unit of mass over conventional batteries
  - More efficient



Fuel Cell

Fuel Cells

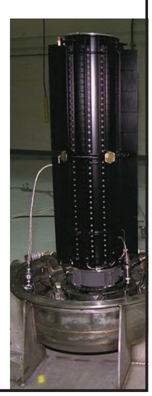
- Fuel Cell Advantages:
  - ☺ Long life span
  - ☺ Can be refueled
  - ☺ Useful by-product
    - ✓ Water
  - ☺ Very efficient
- Fuel Cell Disadvantages
  - ☹ Long flights need lots of fuel
  - ☹ Run very hot (200 to 550° C)
    - ✓ Need to radiate heat into space
  - ☹ Need hydrogen
  - ☹ Slow start up time



Space Shuttle Fuel Cell

RTGs

- A Radioisotope Thermoelectric Generator (RTG) uses the heat generated by the decay of radioactive isotopes to produce electricity
  - Use thermocouples attached to the outside of the heated chamber
    - ✓ Thermocouples are metallic strips formed from two different metals
    - ✓ As one side of the thermocouple heats, the difference in resistance causes electrons to flow
      - ➢ Seebeck effect

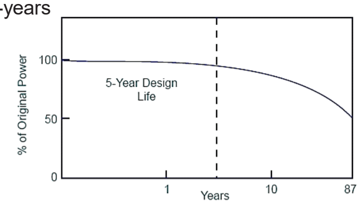


Cassini probe has three RTGs each with a mass of 60 kg and able to produce 296 Watts of power

RTGs

- RTGs were first flown in 1961 on US satellites
- RTG power levels are moderate but life time is long and no energy from the sun is needed, making them uniquely suited for interplanetary missions
  - Examples: Voyager, Pioneer, Cassini, New Horizons
- RTGs typically use Plutonium-238 (Pu-238) isotope, which has a half-life of 87-years

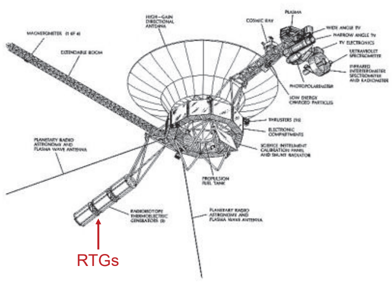
The 87-year half-life of Pu-238 results in 96% of the original power after five years



% of Original Power versus Time

RTGs

- RTG Example: Voyager 2
  - Three RTG's provide electric power to Voyager 2
  - RTGs produce ~1800 W of heat, which is converted to ~ 400 W of power by thermocouples



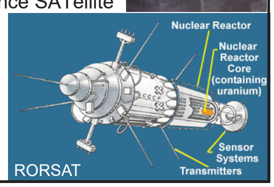
  - The RTGs are mounted on a boom to protect the scientific instruments from excess heat and radioactivity

RTGs

- RTG Advantages
  - ☺ Reliable
    - ✓ No moving parts
  - ☺ Flight proven
  - ☺ Maintenance free
  - ☺ Consistent
  - ☺ No sun required
  - ☺ Long life time
  - ☺ Can be use as thermal heaters
- RTG Disadvantages
  - ☹ Expensive
  - ☹ Public opposition
  - ☹ Contrary to international laws?
  - ☹ Hard to turn off
  - ☹ Hot
    - ✓ Needs cooling
  - ☹ Other spacecraft components need shielding
  - ☹ Not very efficient

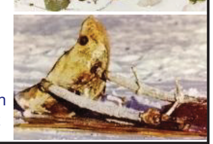
Nuclear Reactors

- Nuclear Reactors
  - Relies on fission reaction and control of radioisotopes
  - US flew a reactor aboard SNAP 10A in 1965
    - ✓ Only US satellite to use a nuclear reactor
  - Former Soviet Union flew them on a regular basis
    - ✓ Radar Ocean Reconnaissance SATellite (RORSAT)
      - ➢ Cosmos series
    - ☺ High power and long life
    - ☹ Disadvantage is human and political consequences of nuclear safety



Nuclear Reactors

- January 24, 1978: A nuclear powered Soviet satellite, Cosmos 954, re-entered the atmosphere and landed in northern Canada
  - Cosmos 954 was a RORSAT with an onboard nuclear reactor
- Operation Morning Light, recovered materials from Cosmos 954
  - They found 12 large pieces
  - These pieces displayed radioactivity of up to 1.1 Sieverts per hour, yet they only comprised an estimated 1% of the fuel



Pieces of Cosmos 954 in Operation Morning Light

Batteries

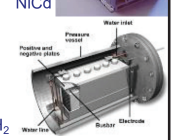
- Short duration missions, typically less than one month, may use batteries as a primary power source
  - Non-rechargeable batteries used
- The vast majority of space missions last many months or years, so batteries are a secondary source of power
  - Rechargeable batteries are ideal for as a secondary power source for solar cells
  - Can charge while the satellite is in sunlight
  - Discharge during an eclipse, providing back-up power
    - ✓ Not as efficient as non-rechargeable batteries
- **NOTE**: Batteries can also support operations during daylight
  - ✓ Supplement the solar array during times of peak power usage

Batteries

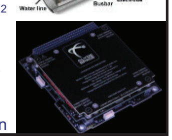
Types

- First batteries on spacecraft were Mercury or Zinc-Silver
  - No longer used
- Nickel Cadmium (NiCd) batteries were the batteries of choice for many years
  - Known lifetime of 10-20 years in space
  - Still used on LEO mission
- Nickel Hydrogen (NiH_2) batteries are desirable for applications where higher energy and longer life is important
  - Has a pressure cell ~ 1000 psi
- Lithium Ion batteries are commonly used for cubesat missions
  - Much lighter and can operate over wider temperature ranges than NiCd or NiH_2

NiCd



NiH_2



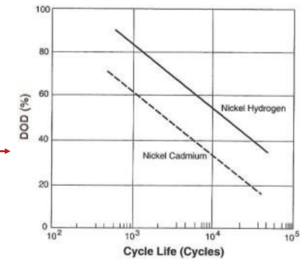
Li Ion

Batteries — Depth of Discharge

- A battery's lifetime depends on how deeply it discharges and the total number of charge/discharge cycles it experiences
- Depth of Discharge (DOD) is the percentage of the total energy stored that is removed from a battery during a discharge period
  - The smaller the DOD the more times a battery can be recycled before it dies
    - ✓ For example, a NiCd battery can be cycled more than 20,000 times at 25% DOD, but only 800 times at 75% DOD before it is unable to hold a charge
- For satellite applications charge/discharge cycles depend on the number of eclipses
  - LEO ~ 5500 cycles per year
  - GEO ~ 84 cycles per year

Batteries — Depth of Discharge

- Follow DOD recommendations for the battery
  - NiH has a larger DOD than NiCd
    - ✓ See graph →
  - SABRE Li ion batteries have a recommended DOD of 19%



DOD versus Number of Cycles for NiCd and NiH_2 Batteries

- The life of the battery can be enhanced by having a flat discharge curve
  - Discharge at a constant rate
- Overcharging can degrade a batteries performance and should be avoided

Batteries — Specific Energy Density

- Battery capacity is measured in Watt·hours (W·hr)
  - For example, a 10 W·hr battery with a DOD of 50%, has 5 W·hr available for use
    - ✓ Since this is stored energy, could use 10 W for 30 min
- A performance characteristic of re-chargeable batteries is the **specific energy density**, which indicates how much energy a battery can hold in relation to its mass
  - Measured in W·hr/kg
- Some representative values are:
  - Nickel Cadmium (NiCd): 25 to 30 W·hr/kg
  - Nickel Hydrogen (NiH) 35 to 57 W·hr/kg
  - Lithium Ion 70 to 110 W·hr/kg
- Use specific energy density to calculate how many kilograms of batteries are required for the satellite mission
  - See next slide

Batteries — Battery Sizing

- As *secondary* power sources batteries are sized using the following equations:

$$C_r = \frac{P_e T_e}{DOD}$$

C_r : Required battery capacity in Watt·hours
P_e : Power required during eclipse (Watts)
DOD : Depth of discharge
T_e : Maximum time duration of eclipse (Hours)

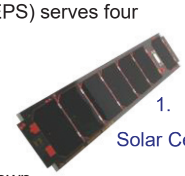
Recall from last lecture: $T_E = \frac{2\rho}{360^\circ}\cdot \mathrm{P}$ where $\rho = \sin^{-1}\left(\frac{R_{Earth}}{h+R_{Earth}}\right)$ and $P = 2\pi\sqrt{\frac{a^3}{\mu}}$

- Can then calculate how many kilograms of batteries are needed to meet power requirements during an eclipse
  - Determine the battery's specific energy density in W·hr/kg and use the following relationship:

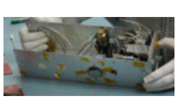
$$\text{Mass of batteries}(\text{kg}) = \frac{C_r(\text{W}\cdot\text{hr})}{\text{Specific Energy Density}(\text{W}\cdot\text{hr/kg})}$$

Power System Design

- A satellite electrical power system (EPS) serves four functions:
  1. Provide power
  2. Store power
  3. Distribute power
  4. Regulate and control power
- SABRE cubesat components are shown


1. Solar Cells


2. Batteries


3. Wiring

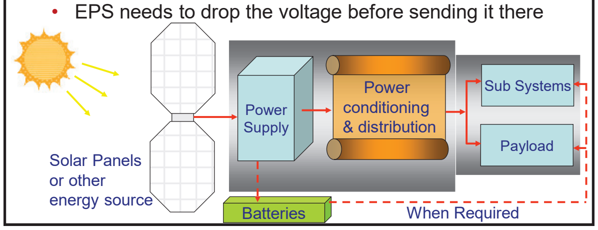

4. EPS Control Board

Power System Design — Elements

- To meet the EPS functionality requirements, the power system is broken into the following elements:
  - Primary energy source
    - ✓ Convert a fuel into electrical power
  - Secondary energy source
    - ✓ Store energy and deliver it to the satellite system when the primary source is not available (e.g eclipse) or if the primary source needs to be supplemented
  - Power control and distribution
    - ✓ Deliver appropriate voltage loads to all spacecraft systems
  - Battery Charge Control (BCR)
    - ✓ Maintain charge of batteries to ensure maximum life

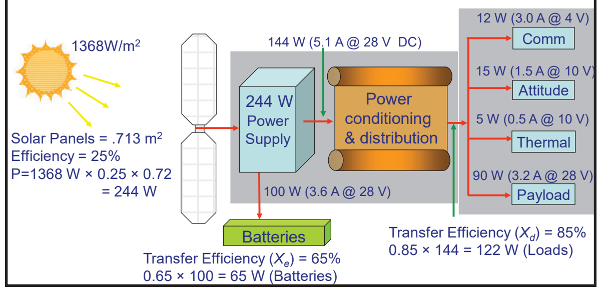
Power System Design — EPS

- An Electrical Power Subsystem (EPS) converts solar energy (or other energy source) into DC electrical power
- Power is conditioned and then distributed to the payload and various subsystems in accordance with mission needs
  - e.g. Bus voltage may be 28 V DC while the communications subsystem only needs 4 V DC
  - EPS needs to drop the voltage before sending it there



Power System Design — Example

- The example has the solar panels generating 244 W of power (P) →100 W goes to the batteries for recharging while 144 W is sent to the various subsystems with the prescribed amount of voltage (V) and amperage (I), where $P = VI$

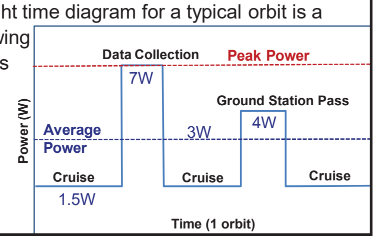


Power System Design — Considerations

- The power system is designed to meet End-Of-Life (EOL) power requirements
  - For example, degradation of solar cells require designers to take this loss of power into account
- Need to consider the dynamic demands of the mission
  - Duty Cycle defines the use timeline for each component
    - ✓ Payload, communications, ADACS, navigation, etc.
- Orbit **average power** (average power for one orbit) is a useful measure for designing the EPS
  - Solar arrays are normally sized to meet average power demands
- **Peak power** is when maximum power is used during an orbit
  - May require a contribution from secondary sources such as batteries to supplement the main power

Power System Design — Considerations

- The power system must satisfy peak power demand periods
  - Secondary power systems may be required
  - Certain subsystems may not operate at the same time to reduce power demands
    - ✓ e.g. No communication during payload operation
- A power versus flight time diagram for a typical orbit is a useful tool for showing power requirements for a typical orbit



Power vs Flight time diagram showing phases of a CubeSat mission for one orbit

Peak and average power are clearly indicated

Power System Design — Power Budget

- Power Budget
  - A power budget is a table showing the power requirements of various systems and the payload during different phases of the mission
  - For each phase the power output of the primary power source is listed along with all systems drawing power
- The goal is to show the ability to power all systems for all phases of the mission
- A power margin is always included in the design process
  - Ensures that there will be enough power for the satellite
  - For new technology satellites the power margin can be as high as 90%
  - For mature designs power margins are 5% to 25%

Power System Design — Requirements and Constraints

- Power System Requirements
  - Performance
    - ✓ The capability of a power source is quantified by specific power
      - ➢ How much power per kilogram is produced?
  - Life time (length of mission)
  - Reliability
  - Safety
- Power System Constraints
  - Mass
  - Size
  - Cost
  - Schedule

Cubesat Deployable Solar Panels

PHE 255
Introduction to Space Science

Module 10: Thermal Control
Part 1: Thermal Environment

- Introduction
- Heat In
- Heat Out
- Heat Transfer
- Orbital Considerations

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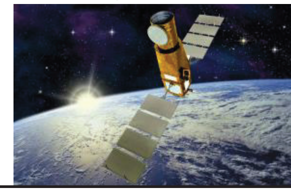
Royal Military College of Canada
Department of Physics and Space Science

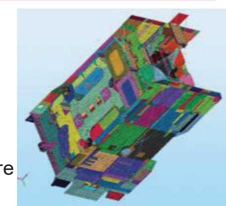
Introduction

- The thermal environment in space is complex since there is no atmosphere to distribute the heat
  - Very hot when in direct sunlight
  - Very cold when in eclipse or shadowed by satellite external structures
- Electronic components and batteries are sensitive to temperature extremes, so it is necessary to thermally control the environment of the spacecraft



Introduction

- Since the geometry and internal heating of the spacecraft is complicated, thermal variations are difficult to determine
  - Typical thermal design is completed through numerical modeling with specialized software
    - ✓ e.g. Finite Element Modeling and Post-processing (FEMAP)
- However, we can make a reasonable first order approximation of thermal conditions using simple principles and making some basic assumptions
  - For example, we will approximate the spacecraft shape as a sphere or some other basic shape to simplify calculations



FEMAP thermal model

Introduction

- A spacecraft is a good example of an isolated system
  - In principle it is easy to analyze everything that goes in and out, such as heat
- For simplicity, most mission scenarios are initially assumed to be in a steady-state energy balance
  - Referred to as thermal equilibrium where the change in the spacecraft temperature (T) with respect to time (t) is 0

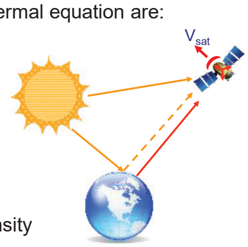
$$\frac{dT}{dt} = 0$$

- Basic components of the heat balance are:

External heat from space environment + Internal heat generated by spacecraft components → **Heat In = Heat Out** → Heat radiated away by the spacecraft

Heat In — Sources

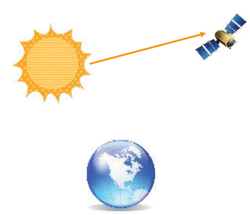
- The **Heat In** components of the thermal equation are:
  - External Heat Sources
    - ✓ Direct solar energy
      - Solar flux density
    - ✓ Reflected solar energy
      - Albedo flux density
    - ✓ Earth emissions
      - Earth Infrared (IR) flux density
    - ✓ Atmospheric drag heating
      - Free molecular heating (FMH) flux density
  - Satellite Internal Heat
    - ✓ Heat generated by spacecraft electrical components (IR)



Note:
$Q \rightarrow$ Flux density = W/m^2
$q \rightarrow$ Flux = W

Heat In — Direct Solar Energy

- Solar energy directly striking the spacecraft is a function of the *solar flux density* (Q_{sun})
- Average solar flux density is the solar constant which is 1368 W/m^2
  - Use this value for thermal calculations
- The solar flux density peaks in the visible range at ~ 0.5 μm



The solar flux density (Q_{sun}) is the solar constant, which is the same parameter used when calculating the power output of a solar array (Module 9 Part 1)

Heat In — Direct Solar Energy

- The solar flux density, or solar constant, is different for other parts of the solar system
  - See table below

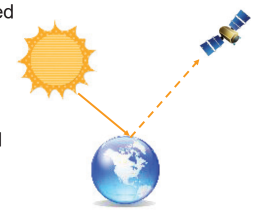
Location	Solar Flux Density (W/m^2)
Mercury	9,058
Venus	2,600
Earth	1,368
Mars	585
Jupiter	50
Saturn	15
Uranus	3.7
Neptune	1.5
Pluto	0.9

Heat shields used by Mercury Messenger satellite

For the outer planets there is essentially no warmth provided by the sun, so heaters are required

Heat In — Reflected Solar Energy

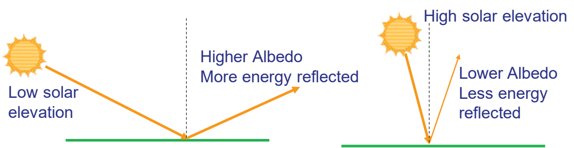
- The spacecraft is also heated by solar energy that is reflected off the Earth's surface
  - Albedo represents the reflectivity of the surface
    - ✓ Albedo is a unitless number between 0 and 1
- Albedo flux density, Q_{albedo}, is the energy input due to reflection of solar radiation from the Earth
- $Q_{albedo} = 0$ when satellite is in eclipse



Albedo flux density is the percentage of the solar constant reflected by the surface

Heat In — Reflected Solar Energy

- Albedo varies with the reflecting surface:
  - Greater over continents than the oceans
  - Increases with cloud, snow and ice cover
  - Increases with decreasing local solar elevation angle



The albedo will decrease as the solar elevation increases

Heat In — Reflected Solar Energy

- Albedo can be averaged out for a single orbit
  - Changes in albedo from 0.05 (ocean) to 1 (ice and snow), are rapid and beyond the thermal inertia of spacecraft
    - ✓ Thermal inertia is a measure of the resistance of a material to temperature change
      - ➢ Higher thermal inertia = more resistant to temperature change
- Average albedo for Earth is 0.30
  - 30% of solar energy is reflected back into space
- Therefore, for geocentric orbits the albedo flux density can be approximated as 30% of solar flux density
  - Average albedo flux density at the surface of the Earth:

$$Q_{albedo} = 1368\ \mathrm{W/m^2} \times 30\% = 410.4\ \mathrm{W/m^2}$$

    - ✓ This value will diminish with altitude from the surface

Heat In — Reflected Solar Energy

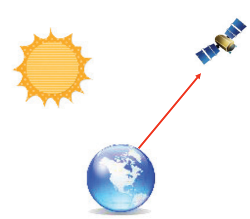
- For interplanetary missions, the solar flux and albedo for that planet must be taken into consideration

Location	Solar Flux Density (W/m^2)	Average Albedo
Mercury	9058	0.06
Venus	2594	0.61
Earth	1358	0.30
Moon	1358	0.07
Mars	585	0.15
Jupiter	50	0.41
Saturn	15	0.42
Uranus	3.7	0.42
Neptune	1.5	0.52
Pluto	0.9	0.16

Example: Venus has a high solar flux density and a high albedo, so a significant amount of solar energy is directly hitting the spacecraft and also being reflected from the planet in a Low Venus Orbit

Heat In — Earth Emissions

- A satellite in orbit around the Earth is heated by the Earth's thermal energy
- Earth temperature is ~300 K
  - Emits long wavelength infrared (IR)
    - ✓ ~10 μm wavelength
- Known as Earth IR flux density, Q_{Earth_IR}
- Present in daylight and eclipse



The effect of Earth IR emissions becomes more predominant the closer the satellite is to the Earth

Heat In — Earth Emissions

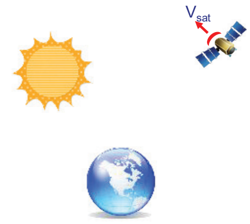
- Earth IR flux density will vary depending on a number of factors:
  - Surface and air temperature
    - ✓ Higher temperature = more IR energy
  - Atmospheric moisture and clouds
    - ✓ IR energy absorbed by clouds, water vapor and CO_2, reducing amount of energy going into space
  - Highest values occur in clear tropical areas
- Earth emissions at the surface varies between 94 W/m² at the Poles to 276 W/m² at the Equator
  - Average IR flux density at the surface of the Earth is:

$$Q_{Earth\_IR} = 236 \text{ W/m}^2$$

    - ✓ This value will diminish with altitude

Heat In — Atmospheric Drag Heating

- If a satellite is low enough it will experience heating as a result of collisions with atmospheric molecules
  - Lower altitude = more heating since atmospheric density is greater
- This type of heating is called free molecular heating (FMH) flux density,
- Only a significant heat source for satellites below an altitude of 500 km
- Present in daylight and eclipse



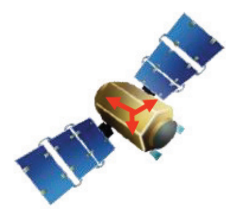
Q_{FMH} is a function of satellite size, speed and altitude

Heat In — Atmospheric Drag Heating

- FMH flux density is in the order of thousands of W/m² at an altitude of 100 km and decreases rapidly with altitude
  - Molecular heating is negligible above ~ 500 km
  - FMH is only a factor for altitudes below 500 km
- A satellite is potentially vulnerable to FMH in the following circumstances:
  - Leaving the launch fairing
    - ✓ May be as low as 100 km
      - ➢ The satellite leaves the launch vehicle, then boosts itself into a higher operational orbit
  - LEO circular orbits less than ~ 500 km
  - HEO with low perigee (< 500 km)

Heat In — Satellite Internal Heat

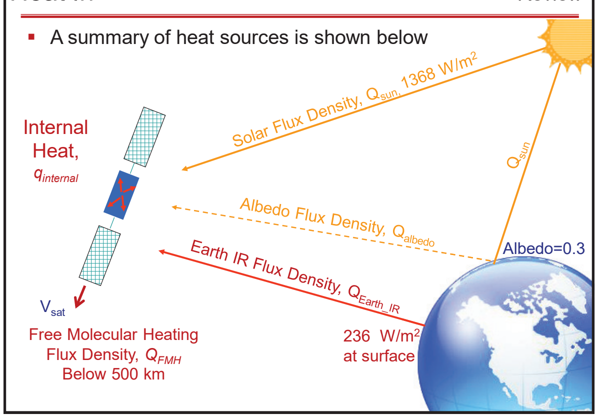
- Electrical components in the satellite will dissipate energy and heat up the satellite internally
- Expressed in in Watts, $q_{internal}$
- May vary throughout the orbit, depending on what systems are running
  - Plan for worst case scenarios when conducting thermal analysis
    - ✓ Low in eclipse
    - ✓ High in sunlight



During eclipse periods satellite internal heating is the main heat source

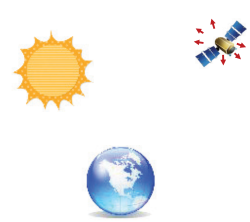
Heat In — Review

- A summary of heat sources is shown below



Heat Out — Satellite IR Emissions

- Any object above absolute zero will emit heat
- The heat inputs discussed in the previous section will cause the satellite to heat up
- Subsequently, the satellite will radiate energy in the infrared (IR) regime
- In this section we will discuss the factors affecting the heat dissipation of a satellite

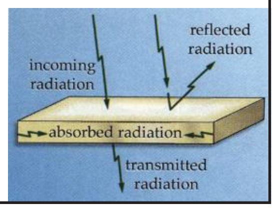


Remember, for thermal equilibrium

Heat In = Heat Out

Heat Out — Satellite IR Emissions

- When radiation strikes a surface, the material reflects, absorbs or transmits it
  - Absorptivity (α) is defined as the percentage of incident radiant energy that is absorbed by a surface
  - Reflectivity (ρ) is the percentage of energy reflected by a surface
  - Transmissivity (τ) is the percentage of radiation transmitted through the body
    - ✓ Most solid materials do not transmit energy
      - ➢ In these cases $\tau = 0$, which is the case for satellite surfaces



Heat Out — Satellite IR Emissions

- With respect α, ρ and τ all of the incident radiation must be accounted for:

$$\alpha + \rho + \tau = 1$$

  - Since $\tau = 0$ for satellites

$$\alpha + \rho = 1$$

  - Therefore all energy incident on the satellite is either reflected or absorbed
- The absorbed energy will heat up the satellite
  - This absorbed energy in conjunction with internal heat will cause the satellite to emit energy in the IR regime

Heat Out — Satellite IR Emissions

- The amount of radiated energy depends on the emissivity, ε, of the object
  - Emissivity is a measure of how well an object emits radiation
    - ✓ Varies between 0 and 1 (no units)
  - A blackbody is a perfect emitter where $\varepsilon = 1$
  - For a grey body $\varepsilon < 1$
- The Stefan-Boltzmann Law quantifies the amount of heat energy radiated.

Surface area of the material (m²)

Heat energy emitted (W) → $q_{radiation} = \sigma \varepsilon A T^4$ ← Temperature (K)

Emissivity

Stefan-Boltzmann Constant (5.67×10^{-8} Wm^{-2}K^{-4})

Heat Out — Satellite IR Emissions

- An important property of materials is heat capacity
  - Ability of a material to store or dissipate heat
  - This means that the satellite will not cool down immediately when going into eclipse
    - ✓ It takes time for the heat to dissipate from the satellite

Mass of the system (kg)

Temperature increase (or decrease) in K

Input (or output) energy in Watts → $q_{heat\_capacity} = m c_p \left(\frac{\Delta T}{\Delta t} \right)$

Specific heat capacity (J kg^{-1} K^{-1})

Duration of input (or output) energy (s)

Measure of the heat energy required to raise the temperature 1 K (See next slide for values)

Satellite IR Emissions

- Specific heat capacity, c_p, of selected materials is shown

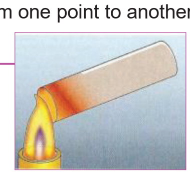
Material	Specific Heat Capacity—c_p (J·kg^{-1}K^{-1})	Material	Specific Heat Capacity—c_p (J·kg^{-1}K^{-1})
Copper	390	Nylon	1,680
Aluminum Alloys	920	Carbon Fiber	840
Stainless Steel	503	Air (1 atm.)	1,006
Beryllium	1,800	Water	4,182
Magnesium	1,050	Hydrazine (liquid)	3,100
Titanium	520	Gallium Arsenide	335
Nickel	460	Ceramics	800
Tungsten	142	Silicon	712
Teflon	1,006	Kapton	1,006

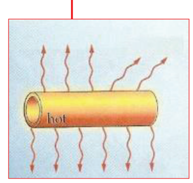
$$q_{heat\_capacity} = m c_p \left(\frac{\Delta T}{\Delta t} \right)$$

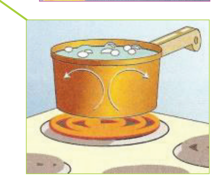
A higher specific heat capacity results in a higher heat capacity, which means that the material is less sensitive to a change in temperature

Heat Transfer — Conduction

- Heat may be transferred from one point to another in a spacecraft in three ways
  - Conduction
  - Convection
  - Radiation

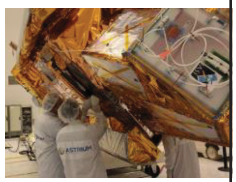






Heat Transfer — Conduction

- Conduction occurs when heat flows through a physical medium to a cooler point
  - Transfer of kinetic energy
    - ✓ Atoms vibrate but do not move
- The thermal conductivity, *k*, of a material indicates how well it will transfer heat
  - Units of *k* are W/m·K
- A material that has a low *k*:
  - Insulate a spacecraft system from heat or cold
- A material that has a high *k*:
  - Transfer heat from one part of a spacecraft to another



METOP-A being wrapped with multi-layer insulation

Heat Transfer — Conduction

- Thermal conductivity of selected materials is shown below

Material	Conductivity ($W \cdot m^{-1} \cdot K^{-1}$)	Material	Conductivity ($W \cdot m^{-1} \cdot K^{-1}$)
Copper	389	Ethylene Glycol	2.42
Aluminum Alloy 2017	164	Water	0.536
Aluminum Alloy 3003	156	Silicone	0.151
Aluminum Alloy 308	96.9	Perlite	0.0692
Aluminum Alloy 2219-0	172.9	Calcium Silicate	0.0548
Aluminum Alloy 6061-T6	167.7	Mineral Fiber Board	0.9548
Aluminum Alloy 7075-T6-T7	121	Mineral Fiber Block	0.0360
Steel 17-4PH	18	Phenolic	0.0332
Inconel 718	11.2	Glass Fiber Block	0.0317
Magnesium AZ31B H24	76.1	Urea Formaldehyde	0.0317
Titanium Ti-6Al-6V	7.26	Polystyrene	0.0288
Beryllium	150.4	Air	0.0260
Plain Carbon Steel	51.9	Polyurethane	0.0231
Stainless Steel Type 304	17.3		

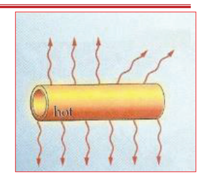
Heat Transfer — Convection

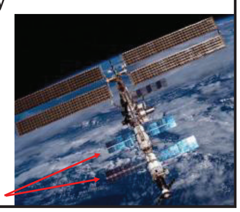
- Convection occurs when some driving force, such as gravity, moves the medium past a heat source
  - Medium is normally a liquid or gas
- In the free-fall of space there are no forces to cause cooler fluid/gas to replace the warmer fluid/gas
  - Everything free-falls together
- Some Russian spacecraft use forced convection to cool their spacecraft components
  - Components are in a large pressure vessel filled with nitrogen
  - Fans circulate the nitrogen, cooling the electronics

Meteor Spacecraft

Heat Transfer — Radiation

- Radiation is a method of transferring energy through space
  - No medium necessary for radiation
- Heat may only be removed from the spacecraft through radiation
  - If a spacecraft is too warm, it may be necessary to radiate excess heat into space
- Heat will be also be dispersed internally in the spacecraft through radiation





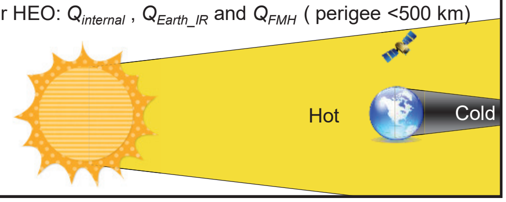
ISS Space Radiators

Orbital Considerations — General Description

- The values of solar flux density, albedo flux density, Earth IR flux density and free molecular heating (FMH) will depend on the orbit (LEO, MEO, GEO, HEO)
- FMH is primarily a concern for launch and re-entry, but is also a factor for orbits 500 km and below
  - Below 500 km: Calculate FMH
    - ✓ Applicable to LEO < 500 km and HEO in which perigee < 500 km
  - Above 500 km: FMH is negligible and does not need to be calculate
- Albedo flux and Earth IR flux are higher at lower altitudes, but should be calculated for all orbits
- During eclipse there is no illumination from the sun
  - Direct solar flux and Albedo flux are zero

Orbital Considerations — Eclipses

- Over the course of a mission it is necessary to measure the range of temperatures the satellite will experience
  - Determine daylight (hot) and eclipse (cold) temperatures
- During eclipse, calculate the following:
  - There is **no** Q_{sun} or Q_{albedo}
  - For LEO: $Q_{internal}$, Q_{Earth_IR} and Q_{FMH} (<500 km)
  - For MEO and GEO: $Q_{internal}$, Q_{Earth_IR}
  - For HEO: $Q_{internal}$, Q_{Earth_IR} and Q_{FMH} (perigee <500 km)



PHE 255
Introduction to Space Science

Module 10: Thermal Control
Part 2: Thermal Control Systems

- Review
- Thermal Calculations
- Thermal Design
- Thermal Control Systems

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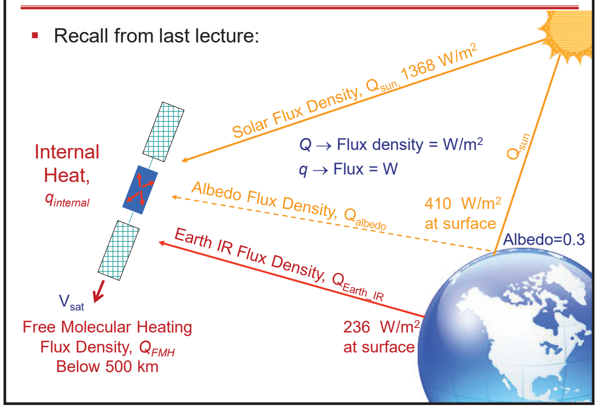
Royal Military College of Canada
Department of Physics and Space Science

Review — Thermal Environment

- Recall from last lecture:



Review — Thermal Environment

- Basic spacecraft temperatures are calculated with the following equation:

Heat In* = *Heat Out → Heat radiated by satellite

Earth Emissions, Free molecular heating, Solar Flux, Albedo Flux, Internal Heat → Heat In

- For thermal equilibrium $\dfrac{dT}{dt} = 0$

Thermal Calculations — Heat In - q_{sun}

- q_{sun} calculation:

$$q_{sun} = Q_{sun} A_s \alpha_s \mu_i \quad \text{Watts}$$

- Q_{sun} : Solar flux density = solar constant = 1368 W/m^2
- A_s : Satellite surface area
- α_s : Absorptance of the spacecraft surface in the solar regime (visible light) between 0 and 1
  - ✓ Percentage of solar flux density absorbed by spacecraft
- μ_i : Solar aspect coefficient = percentage of satellite area exposed to solar flux density
  - ✓ Depends on the shape of the satellite
    - ➢ For first order analysis assume a sphere $\rightarrow \mu_i = 0.25$
- q_{sun} is applicable for all geocentric orbits in daylight
- q_{sun} = 0 during eclipse

Thermal Calculations — Heat In - q_{albedo}

- q_{albedo} calculation is:

$$q_{albedo} = Q_{albedo} A_s \alpha_s F_s \quad \text{Watts}$$

- Q_{albedo} at surface: 1368 W/m^2 × 0.30 = 410 W/m^2
- A_s : Satellite surface area (same as previous slide)
- α_s : Absorptance (same as previous slide)
- F_s : View factor quantifies the percentage of reflected energy reaching a spherical satellite at altitude, H
  - ✓ View factor for a sphere is:

$$F_s = \frac{1}{2}\left(1 - \frac{\sqrt{H^2 + 2HR_e}}{H + R_e}\right)$$

R_e = Radius of Earth = 6378 km
H = Altitude of satellite

- q_{albedo} applicable in LEO only for daylight
- q_{albedo} = 0 during eclipse

Thermal Calculations — Heat In - q_{Earth_IR}

- q_{Earth_IR} calculation:

$$q_{Earth\_IR} = Q_{Earth\_IR} A_s \varepsilon_{IR} F_s \quad \text{Watts}$$

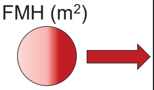
- Q_{earth_IR} : Earth IR flux density at the surface = 236 W/m^2
- A_s : Satellite surface area (same as previous slides)
- F_s : View factor (same as previous slide)
- ε_{IR} : Emissivity of the spacecraft in the IR regime
  - ✓ Why use emissivity since this equation tells us how much IR energy is absorbed?
    - ➢ Kirchoff's Law: At thermal equilibrium $\varepsilon(\lambda) = \alpha(\lambda)$
      - » Therefore, we can assume that emissivity(ε) = absorptivity(α) for IR wavelengths
- q_{Earth_IR} applicable in LEO for daylight and eclipse

Thermal Calculations — Heat In - q_{FMH}

- q_{FMH} calculation:

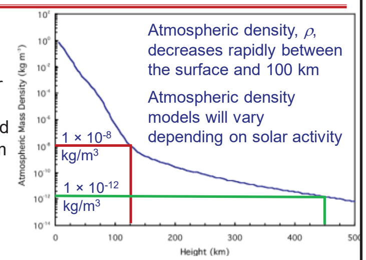
$$q_{FMH} = \frac{1}{2}\alpha_{FMH}\rho V^3 A_{FMH} \quad \text{Watts}$$

  - α_{FMH} : Accommodation coefficient (0.6-1.0)
    - ✓ Heat transfer percentage between molecule and surface
  - ρ : Atmospheric density (kg/m^3)
  - V : Velocity of spacecraft (m/s)
  - A_{FMH} : Area of the spacecraft exposed to FMH (m^2)
    - ✓ A_{FMH} = 0.25 for a sphere
  - q_{FMH} applicable when altitude < 500 km
  - q_{FMH} applicable in daylight and eclipse



Thermal Calculations — Heat In - q_{FMH}

- Sample calculation for q_{FMH}
  - Calculate FMH flux for circular orbits with altitudes of 110 km and 450 km for A_{FMH} = 1 m
  - Accommodation coefficient is 0.8
  - Use the table for atmospheric density



110 km

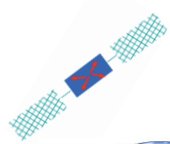
$$q_{FMH} = \frac{1}{2}\alpha_{FMH}\rho V^3 A_{FMH} = \frac{1}{2}(0.8)(1\times10^{-8}\text{kg/m}^3)(7800\ \text{m/s})^3(1\ \text{m}^2) = 1898\ \text{W}$$

450 km

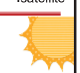
$$q_{FMH} = \frac{1}{2}\alpha_{FMH}\rho V^3 A_{FMH} = \frac{1}{2}(0.8)(1\times10^{-12}\text{kg/m}^3)(7600\ \text{m/s})^3(1\ \text{m}^2) = 0.2\ \text{W}$$

Thermal Calculations — Heat In - $q_{internal}$

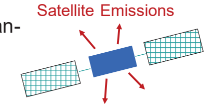
- Electrical components within the satellite will dissipate heat
  - Heat source within the spacecraft in Watts
  - Add up the amount of Watts to run each component
  - Expressed as $q_{internal}$
- For thermal calculations you need to determine amount of power used during daylight and eclipse
  - e.g. Electrical components dissipate 500 W of heat energy in eclipse and 700 W in daylight



Thermal Calculations — Heat Out - $q_{satellite}$



- The satellite will radiate heat as a result of the all the external and internal heat inputs
  - Expressed as $q_{satellite}$
- Recall from last lecture: The emitted radiation flux is governed by the Stefan-Boltzmann law



$$q_{satellite} = \varepsilon_{IR}\sigma A_s T^4 \quad \text{Watts}$$

  - $q_{satellite}$ - Radiation flux of the satellite (W)
  - ε_{IR} - Emissivity in IR band
  - A_s - Surface area of satellite (m^2)
  - σ - Stefan-Boltzmann constant (5.67×10^{-8} Wm^{-2}K^{-4})
  - T - Temperature (K)

Thermal Calculations — Satellite Temperature

- The heat balance equation for an isothermal spacecraft can be expressed by the following equation
  - Generally, we will model the spacecraft as a sphere to simply calculations

Change in temperature with respect to time
Internal heat from satellite components
Heat radiated away by satellite
(-ve)

$$mC_p\frac{dT}{dt} = q_{sun} + q_{albedo} + q_{Earth\_IR} + q_{FMH} + q_{\text{internal}} - q_{satellite}$$

Mass of Satellite
Specific heat capacity
Direct Input from Sun
Reflected Input from Sun
Earth IR flux
Free Molecular Heating

Prevents satellite from changing temperatures instantaneously

Thermal Calculations — Satellite Temperature

- At thermal equilibrium, the change in temperature with respect to time is 0 ($dT/dt = 0$)

$$0 = q_{sun} + q_{albedo} + q_{Earth\_IR} + q_{FMH} + q_{\text{internal}} - q_{satellite}$$

Expand

$$0 = q_{Sun} + q_{albedo} + q_{Earth\_IR} + q_{FMH} + q_{\text{internal}} - \varepsilon_{IR}A_s\sigma T^4$$

- Rearrange to solve for temperature (T):

$$T = \left(\frac{q_{sun} + q_{albedo} + q_{Earth\_IR} + q_{FMH} + q_{\text{internal}}}{\varepsilon_{IR}A_s\sigma}\right)^{1/4}$$

- This equation will solve for the steady state temperature of the spacecraft in Kelvin
  - Put temperature in degrees Celsius by subtracting 273.15

Thermal Calculations — Satellite Temperature

- Equation summary:

$$T = \left(\frac{q_{Sun} + q_{albedo} + q_{Earth\_IR} + q_{FMH} + q_{\text{internal}}}{\varepsilon_{IR} A_S \sigma} \right)^{1/4}$$

where:

$$q_{sun} = Q_{Sun} A_S \alpha_S \mu_i \qquad q_{albedo} = Q_{albedo} A_S \alpha_S F_S \qquad F_s = \frac{1}{2}\left(1 - \frac{\sqrt{H^2 + 2HR_e}}{H + R_e}\right)$$

$$q_{Earth\_IR} = Q_{Earth\_IR} A_S \varepsilon_{IR} F_S \qquad q_{FMH} = \tfrac{1}{2} A_{FMH} \alpha_{FMH} \rho V^3 \qquad q_{\text{internal}} = \text{internal heat}$$

- Use equations to solve for temperature extremes of the orbit
  - Eclipse at apogee and daylight at perigee
- Satellite temperature is largely dependent of the spacecraft coating
  - The values of ε_{IR} and α_S are a function of the coating
  - The importance of thermal coatings is demonstrated in the current assignment

Thermal Calculations — Example

Calculate the steady state temperature of satellite in a 500 km altitude circular orbit for daylight and eclipse using the following parameters. Satellite area = 10 m^2 , α_s = 0.2, ε_{IR} = 0.4, the satellite generates 500 W at all times, σ = 5.67×10^{-8} W/m^{2}K, solar constant = 1368 W/m^2 , R_e = 6378 km. Simplify the satellite shape to a sphere.

a. What is the value of μ_i and F_s?
b. What are the values of Q_{albedo}, Q_{Earth_IR}, and q_{FMH}?
c. What is the daylight temperature in Celsius?
d. What is the eclipse temperature in Celsius?

$$\mu_i = 0.25 \text{ for a sphere}$$

$$F_s = \frac{1}{2}\left(1 - \frac{\sqrt{500^2 + 2 \cdot 500 \cdot 6378}}{500 + 6378}\right) = 0.31$$

$$Q_{albedo} = 1368 \cdot 0.3 = 410 \text{ W/m}^2$$

$$Q_{Earth\_IR} = 236 \text{ W/m}^2$$

$$q_{FMH} \approx 0 \text{ for orbits } \geq 500 \text{ km}$$

$$T = \left(\frac{q_{Sun} + q_{albedo} + q_{Earth\_IR} + q_{FMH} + q_{\text{internal}}}{\varepsilon_{IR} A_S \sigma} \right)^{1/4}$$

$$q_{sun} = Q_{Sun} A_s \alpha_s \mu_i = 1368 \cdot 10 \cdot 0.2 \cdot 0.25 = 684 \text{ W}$$

$$q_{albedo} = Q_{albedo} A_s \alpha_s F_s = 410 \cdot 10 \cdot 0.2 \cdot 0.31 = 254 \text{ W}$$

$$q_{Earth\_IR} = Q_{Earth\_IR} A_s \varepsilon_{IR} F_s = 236 \cdot 10 \cdot 0.4 \cdot 0.31 = 293 \text{ W}$$

Day: q_{Sun} q_{Albedo} q_{Earth_IR} $q_{Internal}$
q_{FMH} *negligible at 500 km*

$$T_{day} = \left(\frac{684 + 254 + 293 + 0 + 500 + 0}{0.4 \cdot 10 \cdot 5.67 \times 10^{-8}} \right)^{1/4} = 295.6 \text{ K} = 22.4^\circ\text{C}$$

Eclipse: q_{Earth_IR} $q_{Internal}$ *only*
q_{FMH} *negligible at 500 km*

$$T_{eclipse} = \left(\frac{293 + 0 + 500 + 0}{0.4 \cdot 10 \cdot 5.67 \times 10^{-8}} \right)^{1/4} = 243.16 \text{ K} = -30.0^\circ\text{C}$$

Thermal Design

- The objective of the thermal control system is to provide the proper heat transfer between all spacecraft elements so that the temperature-sensitive components will remain within their specified temperature limits during all mission environmental conditions
- Thermal design is considered in all phases of the mission:
  - Manufacturing and testing
  - Launch preparation
  - Launch
  - Parking orbit (if applicable)
  - On orbit (operations)
  - Re-entry and recovery

Thermal Design

- Temperature tolerances can be divided into three categories:
  - Operating temperature range
    - ✓ Temperature range within which the equipment fulfills all specified operating performance and life requirements
  - Switch-on temperature limit
    - ✓ Lowest temperature for safe activation of equipment
  - Non-operating temperature range
    - ✓ The temperature limits within which equipment that is "off" can be safely left

Thermal Design

- At the initial design stages it is necessary to determine the following information:
  - The space environment definitions, which involves characterizing the amount of heating the spacecraft is exposed to with respect to time
    - ✓ Thermal calculations
  - The temperature tolerance for each of the spacecraft components
- Thermal margins are also placed on components
  - Equipment must be kept inside allowable limits by 5 degrees
    - ✓ Example: If a component temperature tolerance is 0 to 40° C, then the thermal design should ensure a range of 5 to 35° C for this component

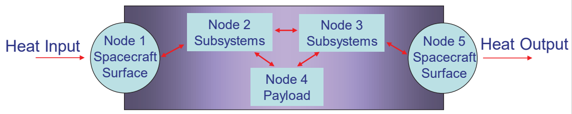
Thermal Design

- Typical operating temperatures in degrees Celsius are shown for spacecraft components
- Must identify the component(s) with the least tolerance and design to that range
- Humans have the narrowest temperature range if onboard

Telecommunications			
Payload units	−10 to +50		
Remote Sensing Payloads			
Optical sensors	+15 to +25	Infrared module[3]	−40 to +30
Radiometric units	−10 to +50	Radar units	−10 to +40
Onboard Computer	−10 to +50		
Telemetry & Command units	−10 to +50		
Electrical Power			
Batteries (NiH_2)	−5 to +20	Batteries (NiCd)	0 to +25
Solar Arrays	−105 to +110	Power control unit	−20 to +55
Attitude Control			
Sun & Earth sensors	−30 to +50	Magnetometer	−80 to +80
Electronic units	−10 to +55	Momentum and reaction wheels	−5 to +45
Gyro package	0 to +50		
Propulsion			
Propellant tank, filters, valves, lines	+7 to +55	Thrusters	+7 to +65
Harness			
Spacecraft internal	−15 to +55	Spacecraft external	−100 to +100
Thermal Control			
Multilayer Insulation (MLI)	−160 to +250	Heaters, thermostats, heat pipes	−35 to +60
Radiators	−95 to +60		
Structures			
Nonalignment critical	−45 to +65		
Mechanisms			
Pyrotechnics	−100 to +120	Electric motors	−45 to +80
Deployment hinge	−45 to +65	Solar array drive assembly	−35 to +60
Antennas			
Parabolic reflector	−160 to +95	TT&C	−65 to +95
GPS antenna	−95 to +70		

Thermal Design

- When determining thermal requirements in the design phase one method is to divide the spacecraft into nodes
  - A thermal node is any part of the spacecraft that has unique thermal properties to consider
    - ✓ For thermal equilibrium, Heat Input = Heat Output



- Computer analysis techniques are then used to determine the overall system temperatures
  - Finite element modeling

Thermal Control Systems

- Thermal control methods are categorized as passive and active
- Common methods of thermal control are shown below:

Passive	**Active**
Thermal coatings	Heat pipes
Multilayer insulation	Electric heaters
Space radiators	Refrigerators
	Louvers

85% to 100% of a spacecraft's thermal control demands can be met by choosing the right coatings and insulation

Thermal Control Systems — Passive - Coatings

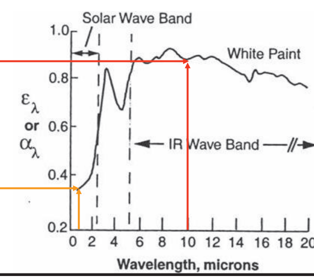
- Satellite temperature can be regulated by choosing an appropriate external coating
  - Coating dictates overall absorptance (α) and emissivity (ε) of the satellite
- α and ε values depend on wavelength
  - Peak α is in visible range (sun)
  - Peak ε is in IR range (satellite emission)

Surface	Absorptance α	Emittance ε
Gold plate on Al 7075	0.3	0.03
Polished aluminium	0.35	0.04
Polished beryllium	0.4	0.05
Gold on aluminium	0.26	0.03
Polished stainless steel	0.5	0.13
Polished copper	0.28	0.13
Grafoil	0.66	0.34
Vapour-blasted stainless steel	0.6	0.33
Gold/Kapton/aluminium	0.53	0.42
Epoxy black paint	0.95	0.85
Acrylic black paint	0.97	0.91
Silicone white paint	0.19	0.88
—after 3 years' u.v. irradiation	0.39	0.88
Silicate white paint	0.14	0.94
—after 3 years' u.v. irradiation	0.27	0.94
Silicon solar cell, bare	0.82	0.64
Silicon solar cell, silica cover	0.82	0.81
Silicon solar cell, silica cover, blue filter	0.78	0.81
Silicon solar cell, silica cover, red filter	0.7	0.81
Kapton (5 mil)/aluminium	0.48	0.81
In_2O_3/Kapton/aluminium	0.4	0.71
Quartz fabric/tape	0.19	0.6
OSR (quartz mirror) silvered Teflon	0.08	0.81
FEP (5 mil)/silver	0.11	0.8
FEP (2 mil)/silver	0.05	0.62

Absorptance and Emittance for Selected Coatings and Surfaces

Thermal Control Systems — Passive - Coatings

- Example - White Paint
  - Low absorptivity in the solar band coupled with high emissivity in the IR band, means white-painted surface remains relatively cool → more heat emitted than absorbed



Absorptivity and Emissivity for White Paint, 0 to 20 μm

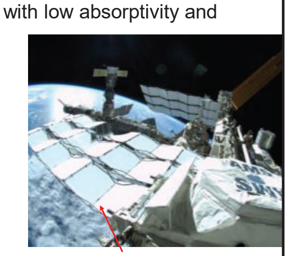
Thermal Control Systems — Passive - MLI

- Multi-layer Insulation (MLI)
  - Comprised of radiative reflecting shields (Kapton, aluminium, Mylar, titanium) separated by low conductivity spacers
  - Goal is to reduce the rate of heat flow between two boundaries
    - ✓ This minimizes temperature fluctuations due to eclipses and varying directions of incoming radiative heat
    - ✓ Isolate components that radiate heat or are sensitive to temperature fluctuations

The golden areas are MLI blankets on the Mars Reconnaissance Orbiter

Thermal Control Systems — Passive - Radiators

- Radiators
  - The most effective long-term method for ejecting heat is by radiation
  - To radiate heat, designers incorporate special surfaces on the spacecraft with low absorptivity and high emissivity
    - ✓ Allow hot components on the inside of the spacecraft to radiate heat into space
    - ✓ May be simply a section of glass coating over a particularly hot section that allows heat to escape into space



ISS Radiators

Thermal Control Systems
Passive - Radiators

- Heat dissipated through radiators is considered waste heat, q_w
- The flux of this *waste heat* is expressed as:

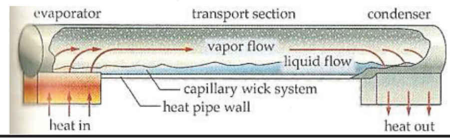
$$q_W = Q_W A_R \quad \text{Watts}$$

  - q_w : Waste heat dissipated by radiator (W)
  - Q_W : Waste heat energy flux density (W/m²)
  - A_R : Area of radiator (m²)
- Thermal equation with q_w :

$$mC_p \frac{dT}{dt} = q_{sun} + q_{albedo} + q_{Earth\_IR} + q_{FMH} + q_{internal} - q_{satellite} \boxed{- q_W}$$

Thermal Control Systems
Active - Heat Pipes

- Heat Pipes
  - As the amount of heat and the urgency to remove it increases, it is necessary to have more complex thermal control methods (active methods)
    - ✓ One method is the use of heat pipes
  - Heat pipes employ a liquid, such as ammonia, with a low boiling point inside a hollow tube
  - As the liquid absorbs heat at the hot end it vaporizes and carries heat to the cool end where it re-condenses
    - ✓ *Latent heat of vaporization*



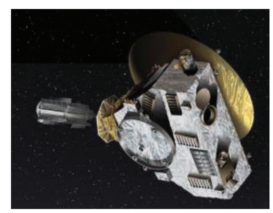
Thermal Control Systems
Active – Electric Heaters and Refrigerators

- Electric Heaters
  - Used when fine temperature control is required
    - ✓ Typically include temperature sensing items and electronic temperature controllers
  - Batteries commonly have integrated patch heaters
- Refrigerators
  - Infrared detectors require low temperatures
    - ✓ Locally generated heat is noise
  - IR sensors are placed on the opposite side of the spacecraft away from the sun
    - ✓ Performance increases with decreasing temperature

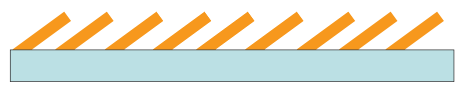
Cryogenic coolers use liquid helium to keep IR sensors at extremely low temperatures (< 20 K)

Thermal Control Systems
Active - Louvers

- Louvers
  - A controllable 'window' situated between space and a radiator surface
    - ✓ Louvers open and close similar to a Venetian blind
  - Louvers actively control the absorptivity and emissivity of the surface



New Horizons with louvers



Thermal Control Systems
Testing

- Once the design is finished and temperature control techniques have been implemented, it is necessary to validate it against operational requirements
- Testing process involves placing the spacecraft in an environment that simulates the vacuum and temperature of space
- To test extremes, thermal cycling tests are performed
  - Hot-cold-hot-cold etc.

Lockheed Martin Thermal Vacuum Testing Facility

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PHE 255
Introduction to Space Science

Module 11: Navigation and Communication
Part 1: Navigation

- Introduction
- Ground Based Systems
- Space Based Systems
- Autonomous Systems
- Orbit Maintenance and Control
- Navigation Guidance and Control System

Dr. Ron Vincent,
Royal Military College of Canada
Department of Physics and Space Science

Introduction

- The main function of a spacecraft's navigation system is to determine the current orbital parameters (i.e. position) of the satellite
- This is accomplished by determining:
  - Position and velocity vectors: x, y, z and v_x, v_y, v_z

    or
  - Classical orbital elements: a, e, i, Ω, ω, ν
- This position can then be propagated forward to project future position for operations and communications
- The type of navigation system or combination of systems chosen for a space mission depends on:
  - The required positional accuracy for the mission
  - The level of autonomy
    - ✓ Ground based or within the satellite?

Introduction — Orbit Determination

- Real Time Orbit Determination
  - Provides the best estimate where the satellite is at the present time
  - Important for spacecraft and payload operations, such as accurate pointing at some target
  - Can propagate the orbit forward for future operations
    - ✓ Determine future satellite position based on orbital data
- Definitive Orbit Determination
  - Is the best estimate of the satellite position and orbital elements at some earlier time
  - Carried out after gathering and processing all relevant observations
  - Can then propagate the orbit forward to the present and into the future

Introduction — Orbit Propagation

- Orbit propagation uses numerical techniques to determine where a satellite will be at some time in the future based on orbital elements
- Orbit propagators may take the main orbital perturbations into account:
  - Drag, Non Spherical Earth, Solar Radiation Pressure, 3rd Body Effects
    - ✓ Example**:** Simplified General Perturbations (SGP4) propagator is used with two-line element (TLE) sets
      - Considers variations due to Earth oblateness, solar and lunar gravitational effects, and orbital decay using a drag model
    - ✓ The latest satellite TLEs are readily available on-line
      - e.g. Celestrak

Introduction — Orbit Determination Systems

- Options for determining orbital elements are listed below:
  - Ground Based Tracking
    - ✓ Optical
    - ✓ Radio and Radar
    - ✓ Satellite Laser Ranging (SLR)
    - ✓ Doppler Orbitography and Radio-positioning Integrated by Satellite (DORIS)
  - Space Based Systems
    - ✓ Tracking Data and Relay Satellite System (TDRSS)
    - ✓ Sapphire → CF satellite
  - Spacecraft Autonomous Navigation
    - ✓ Satellite Positioning Systems
    - ✓ Systems for interplanetary missions

The system selected is a function of the positional accuracy required by the mission and cost

More than one system may be used

Ground Based Systems — Ground Tracking Station

- Traditionally, ground stations provide tracking data to a mission operation center
  - Accomplished by locating the satellite with radar, optical telescopes or laser ranging
    - ✓ May also track by analyzing spacecraft telemetry signals
  - Data used are range and change in range (range rate)
    - ✓ Requires data from a number of passes over a single station, or data may be passed from several stations to a central location for processing
  - When all data is available, definitive orbit determination gives the best estimate of the orbit
    - ✓ Orbit is then propagated forward
  - Ground based space surveillance networks exist in the US, Canada, Europe, Russia, China and India

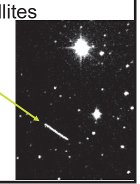
Ground Based Systems — SSN

- The U.S. military Space Surveillance Network (SSN) is the world's most comprehensive space surveillance system
  - Over 25 sites in the Northern Hemisphere
    - ✓ RADAR and optical ground stations
  - Information from other systems are fused into the SSN
  - Two-Line Element sets are produced and made available to the public
  - Track objects as small as 10 cm in LEO, 25 cm in GEO
  - 500,000 observations per day
  - Catalog contains >15,000 objects
    - ✓ 1,000 operational satellites
  - Uses different types of RADAR
    - ✓ Phased array, Dish, Space Fence (will discuss later)

Ground Based Systems — Optical

- Satellites can be tracked using optical telescopes
- Example: The Space Surveillance Research and Analysis Laboratory (SSRAL) at RMCC has been a part of the Canadian Surveillance of Space Concept Demonstrator
  - A fully automated network of small, satellite tracking telescopes across Canada
  - The primary targets of the network are GEO and Molniya
  - The optical tracking methods used to observe these satellites can similarly be applied other satellites


RMCC Castor Telescope


Time lapse Molniya

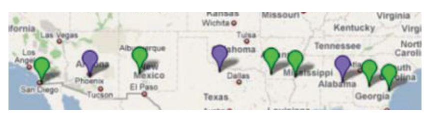
Ground Based Systems — Radio and Radar

- Information about a satellite's position can be determined from the reception and analysis of emitted radio signals
  - Doppler shift (apparent change in transmitted frequency due to spacecraft motion) and/or telemetry signals
- Example: The NASA Deep Space Network (DSN) is an international network of antennas (US, Spain, Australia) that supports interplanetary spacecraft missions
  - Also used for radio and radar astronomy observations for the exploration of the solar system and the universe
- The 70 m diameter antenna is capable of tracking a spacecraft more than 16 billion kilometers from Earth
  - A 26 m antenna is used for tracking satellites in LEO


70 m Antenna

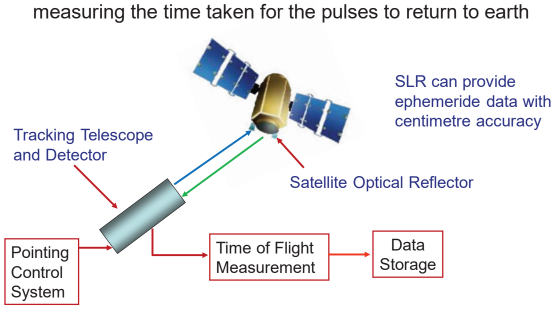
Ground Based Systems — Radio and Radar

- For Radar, a pulse of EM energy is sent out and the return is analyzed to give satellite position and velocity
- Example: The Space Fence is part of the Space Surveillance Network (SSN) to detect satellites and space junk
  - Continuous wave multistatic radar (transmitter and receiver are separated) that tracks objects in LEO and MEO
  - Approximate radiated power is 768 kW at 217 MHz
  - Capable of detecting basketball-sized objects as far away 30,000 km and collects an average of 5 million observations per month


Receiver Transmitter

Ground Based Systems — Satellite Laser Ranging

- Satellite Laser Ranging (SLR) involves the firing of laser pulses through a telescope at passing satellites and measuring the time taken for the pulses to return to earth

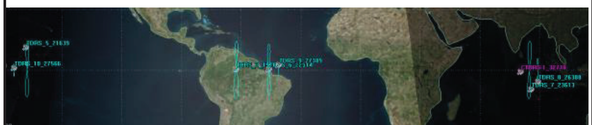


Ground Based Systems — DORIS

- Doppler Orbitography and Radiopositioning Integrated by Satellite (DORIS) is a tracking system providing range-rate measurements of signals from a network of 51 ground-based beacons
  - Receiver is on board the satellite while the transmitters are on the ground
  - A frequency shift of the signal occurs due to the movement of the satellite (Doppler effect)
  - Precision processing of Doppler information allows high accuracy of orbit position
  - Positional accuracy in the order of centimeters is possible
- Processing of position may be done on the ground or onboard the satellite
  - Autonomous operation possible

Space Based Systems — TDRS

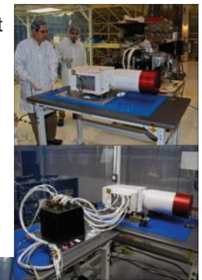
- Tracking and Data Relay Satellite System (TDRSS) replaced NASA's worldwide ground tracking network
- Constellation of geosynchronous satellites
  - TDRS-E (1991) to TDRS-L (2014) are active
- Supplies positional information for suitably equipped satellites
  - Determined with Doppler or time between interrogation and reply between TDRS and satellite
- If drag effects on a satellite is small, TDRS can achieve an accuracy of approximately 50 m



TDRS Locations

Space Based Systems — Sapphire

- Launched in 2013, Sapphire is the first Canadian Forces satellite
  - Contains an optical telescope that tracks satellites and space junk in the GEO belt from a sun synchronous orbit
  - Information is then shared with the US SSN



Bus Acceptance, Integration, and Testing at David Florida Labs in Ottawa

Autonomous Systems

- Determining the orbit onboard is technically easy with the advent of advanced spacecraft computers
  - Need to provide orbit determination that is reliable, robust and economical in both cost and weight
- The advent of positioning satellite constellations, such as the Global Positioning System (GPS) have enhanced the ability for a satellite to determine its own position
  - Useful on small platforms since receivers are small, light and do not require a lot of power
- Autonomous systems are inherently real-time
  - Highly accurate orbit propagation is less critical, although forward propagation is still necessary for prediction and planning

Autonomous Systems — Satellite Positioning Systems

- Satellite positioning systems include:
  - Navstar Global Positioning System (GPS)
    - ✓ United States system operational since 1994
    - ✓ MEO constellation, 24 satellites
  - GLONASS
    - ✓ Russian equivalent to GPS
    - ✓ MEO constellation, 24 satellites
  - Galilieo (Projected to be completed by 2019)
    - ✓ European equivalent to GPS
    - ✓ MEO constellation, 27 satellites
  - Beidou (Not globally operational)
    - ✓ Chinese (to completed by 2020)
    - ✓ 5 Geostationary, 3 Geosynchronous inclined, 27 MEO

Autonomous Systems — GPS

- Global Positioning System (GPS)
  - GPS is a constellation of navigation satellites intended to allow position determination by small receivers
    - ✓ 24 satellite constellation in circular MEO (20,200 km) Orbital period = 12 hours, 6 planes, $i = 55°$
  - GPS only available to LEO spacecraft
    - ✓ Have to be below the GPS constellation since signals are directed toward the Earth
    - ✓ Satellite GPS antenna must be pointed upward
  - Good accuracy available for space applications
    - ✓ 15 m or less

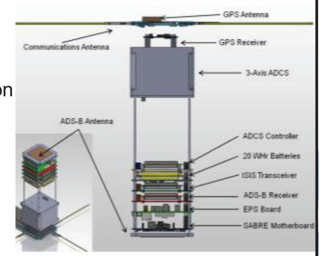
GPS Satellite

Autonomous Systems — GPS

- GPS works by measuring the time of signal propagation from the satellite to the receiver and converting this to a range
  - Requires:
    - ✓ Precise measurement of time
    - ✓ Precise knowledge of satellite position
      - ➢ Ephemeris data is transmitted from the satellite as part of the GPS signal
    - ✓ Precise synchronization between satellite and user
  - Four satellites are required for a fix
    - ✓ Three solve for x, y, z coordinates
    - ✓ The fourth satellite permits synchronization between the GPS atomic clock and receiver
      - ➢ This means that the user does not require an atomic

Autonomous Systems — SABRE Navigation

- The required temporal orbit propagation accuracy for SABRE for all operational modes is 10 seconds
- The chosen methods for determining SABRE's orbit position during nominal operations are:
  - TLEs from SSN
    - ✓ TLEs will be accurate enough for orbit determination requirements during nominal operations
  - Space qualified GPS
    - ✓ GPS positioning or time information for payload ops and communication



SABRE with GPS Receiver and Antenna

Autonomous Systems — Interplanetary Missions

- The following navigation methods could be used for interplanetary missions:
  - Landmark Tracking
    - ✓ Image based onboard navigation accomplished by tracking known landmarks such as craters
      - Near Earth Asteroid Rendezvous (NEAR) used such a system during rendezvous with asteroid Eros
  - Horizon Scanning
    - ✓ Scanning device scans the horizon of a celestial body to determine distance and range
      - e.g. Microcosm Autonomous Navigations System
  - Optical Stellar Navigation
    - ✓ Imaging device used to observe a local celestial body against a known background of stars
      - Deep Space 1 AutoNav system

Orbit Maintenance and Control

- Orbit Maintenance and Control is the process of changing the orbital parameters of the satellite
  - Propulsion system is required
- All satellites need navigation, while orbit maintenance and control is not required for all missions
- Many small spacecraft do not require orbit control and have no onboard propulsion
  - Once the satellite leaves the launch vehicle, no orbital control of the satellite is possible and it will be subject to orbital perturbations
  - Usually only acceptable for spacecraft with intended life of 1 to 3 years

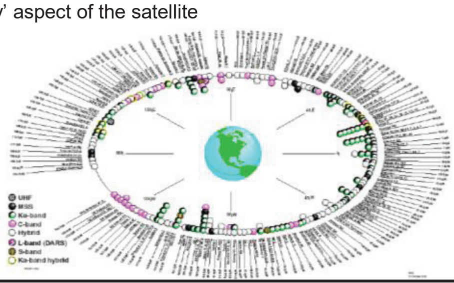
Orbit Maintenance and Control

- Types of orbit maintenance and control include the following:
  - Change the orbital parameters
    - ✓ May be necessary to meet mission objectives
      - e.g. Take pictures of an area for reconnaissance or rendezvous with another spacecraft
    - ✓ Avoid a collision with another orbiting object
  - Maintain the orbital elements
    - ✓ An example of orbit maintenance is the occasional firing of thrusters to overcome orbital perturbations
  - Maintain satellite within a predefined area (stationkeeping)
    - ✓ GEO stationkeeping maintains the satellite in a box over one place relative to the Earth
    - ✓ LEO, MEO and HEO stationkeeping may include constellation maintenance

Orbit Maintenance and Control — GEO

- Geostationary is a busy orbit, so keeping the satellite within a designated box is important
  - Stationkeeping
- Antennas pointed at GEO satellites from the Earth rely on the 'stationary' aspect of the satellite

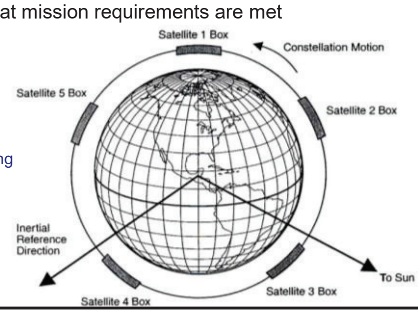
GEO slots, frequencies and transmitting power is controlled by the International Tele-communications Union (ITU)



Orbit Maintenance and Control — HEO, MEO, LEO

- Stationkeeping for constellations of satellites in HEO, MEO and LEO is important in order to maintain the proper orientation of the constellation
  - Ensure that mission requirements are met

LEO Stationkeeping



Orbit Maintenance and Control — HEO Example

- Three SIRIUS satellites in a Tundra Orbit (HEO) maintained in the correct configuration allow 24 hour coverage over North America

SIRIUS Constellation Parameters

Apogee = 47,008 km

Perigee = 24,559 km

e = 0.2684

ω= 270°

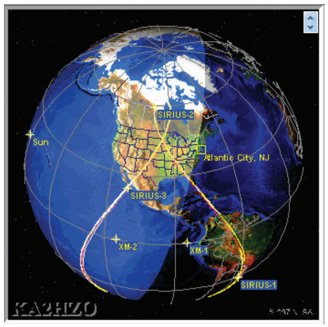
i = 63°

Ω = SIRIUS 1 285°

SIRIUS 2 165°

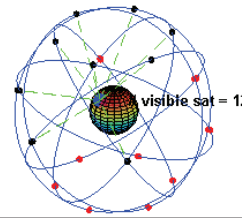
SIRIUS 3 45°

Apogee Longitude = 96°



Orbit Maintenance and Control — MEO Example

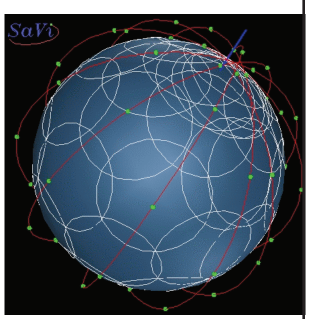
- The GPS constellation ensures that 6 to 11 satellites are 5° above the horizon anywhere in the world at any time
  - 6 orbital planes
  - 24 operational satellites
  - Orbital planes are 60° apart
- > 30° above the horizon ensures better fix geometry and less atmospheric distortion
- Fix geometry is degraded toward the poles due to the 55° inclination of the satellites



Animation showing number of visible GPS satellites in Kingston region during an orbital period (12 hours)

Orbit Maintenance and Control — LEO Example

- Iridium is an example of a complicated (and expensive) constellation
  - Altitude = 780 km
  - 86.4° near circular orbit
  - 6 orbital planes with 11 satellites in a plane
  - Overlapping of footprints in higher latitudes due to polar orbit
  - Satellites turned off in overlapping regions to prevent interference



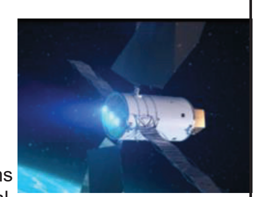
Iridium coverage

Orbit Maintenance and Control

- Ground-based
  - Orbit maintenance is traditionally done from the ground
    - ✓ By comparison, attitude control is done autonomously to prevent serious consequences such as tumbling or pointing sensitive equipment at the sun
      - ➢ Orbit control maneuvers occur less frequently
  - Autonomous systems can reduce operation costs
    - ✓ Lack of control from the ground may be a safety issue
    - ✓ Required for some missions, such as planetary flybys
  - Semi-autonomous operation possible
    - ✓ Orbit control commands are uplinked for execution at a later time

Navigation Guidance and Control System

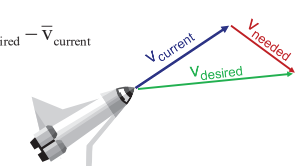
- Spacecraft position is a function of the velocity vector, which can be changed by applying thrust
  - The propulsion system provides thrust, which allows orbit maintenance or maneuvering
  - The Attitude Determination and Control System (ADCS) ensures the thrusters are oriented correctly
- The satellite navigation and guidance system needs to know how much thrust to expect, then calculate which direction to apply it and for how long
- This is an example of navigation, propulsion and ADCS subsystems working together to achieve a goal



Navigation Guidance and Control System

- For a spacecraft changing orbital parameters, start by checking the current velocity vector, $v_{current}$
  - Obtained through the navigation system
- Guidance system then compares this to a desired velocity, $v_{desired}$, set by pre-defined mission requirements or sent to the spacecraft by mission controllers
- The difference between these two vectors, Δv_{needed}, gives the required velocity to reach the new position

$$\Delta\bar{v}_{needed} = \bar{v}_{desired} - \bar{v}_{current}$$



Propulsion system works in conjunction with the navigation and ADCS for spacecraft guidance

PHE 255
Introduction to Space Science

Module 11: Navigation and Communication
Part 2: Communication

- Overview
- Communication Architectures
- Communications Criteria
- Link Budget
- Communication and Data Handling Subsystem (CDHS)

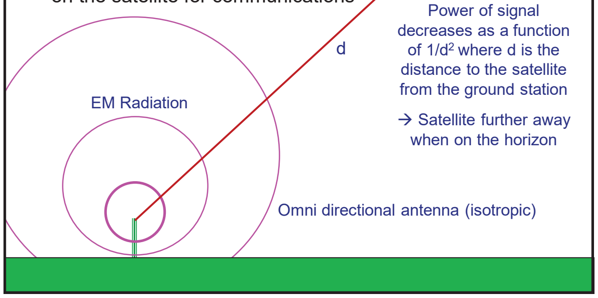
Dr. Ron Vincent,
Royal Military College of Canada
Department of Physics and Space Science

Overview

- Satellite communication involves the transmission of data to and from the satellite
- A satellite transmits two types of data:
  - Payload Data
    - ✓ Mission data from satellite
  - Telemetry, Tracking and Command (TT&C)
    - ✓ Satellite systems information
- Normally the quantity of payload data > TT&C data
- The ground station will send information to the satellite, including:
  - Commands
  - Updates to software
- The following slides illustrate the communications process

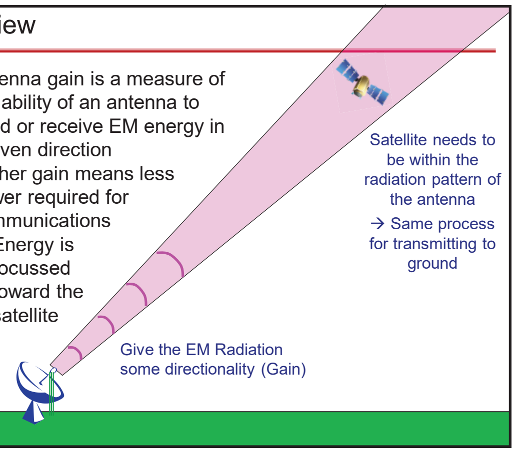
Overview

- An antenna is a device that sends and receives electromagnetic (EM) energy
- Need an antenna on the ground and on the satellite for communications



Overview

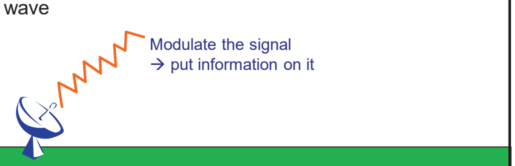
- Antenna gain is a measure of the ability of an antenna to send or receive EM energy in a given direction
- Higher gain means less power required for communications
  - Energy is focussed toward the satellite



Overview

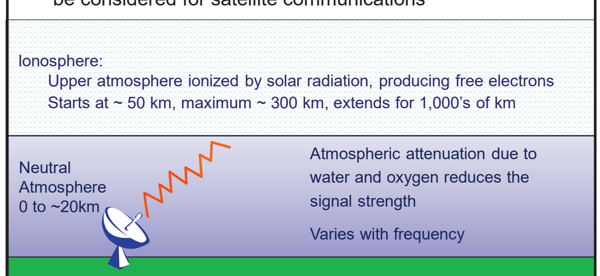
- Modulation is the process of putting information on an EM wave
- Without modulation there is no information contained in the EM wave

Satellite will demodulate the signal to extract the information



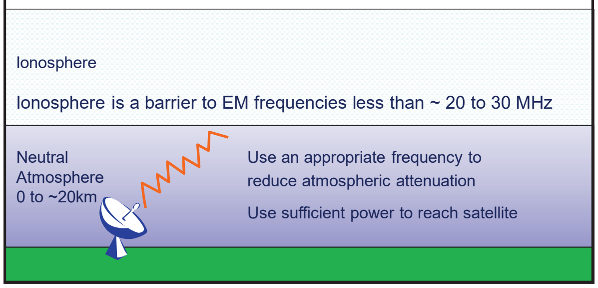
Overview

- The neutral atmosphere will attenuate the EM signal (i.e. reduce the strength)
- The ionosphere is also a factor that must be considered for satellite communications



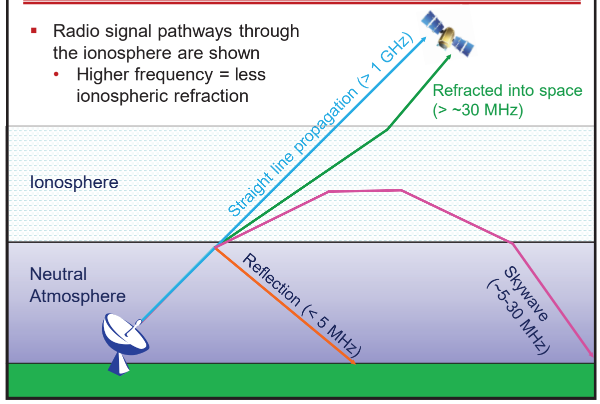
Overview

- Choose a frequency that is relatively unaffected by both the neutral atmosphere and the ionosphere



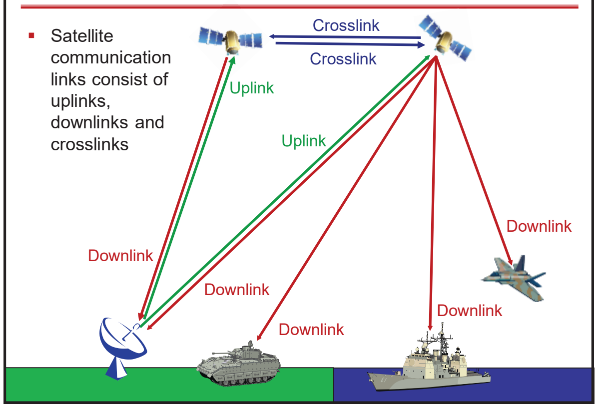
Overview

- Radio signal pathways through the ionosphere are shown
  - Higher frequency = less ionospheric refraction



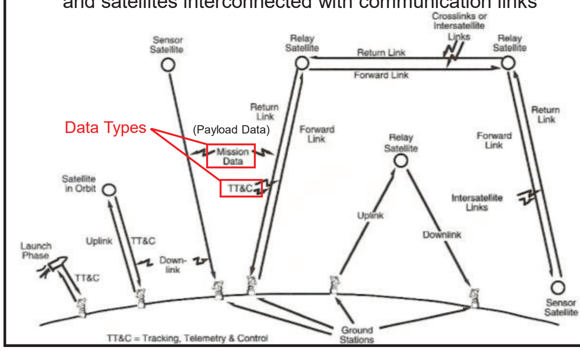
Overview

- Satellite communication links consist of uplinks, downlinks and crosslinks



Communications Architecture

- A communications architecture consists of ground stations and satellites interconnected with communication links



Communications Architecture — Ground Station

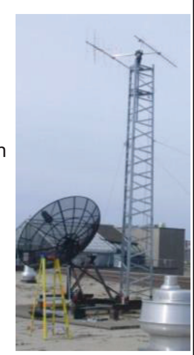
- Ground station refers to the antenna, transmitter receiver, and control equipment required to communicate with the satellite from Earth
  - Includes land, ship and airborne mobile terminals

Verhaert Space, ESA Ground Station in Belgium

- For communications between a satellite and ground station, the transmitter and receiver must be in view of each other (line of sight) and use frequencies that can easily pass through both the neutral atmosphere and the ionosphere
- Satellite to satellite communication (crosslink) is not affected by atmospheric attenuation, but satellites must be in view of each other

Communications Architecture — RMC Ground Station

- RMCC ground station:
  - Two Yagi antennas on the roof of the Sawyer Building operating at 130 to 144 MHz (VHF) and 430 to 440 MHz (UHF)
  - 1.5 m diameter high performance dish antenna operating at 2.4 GHz (S-Band) for future mission downlink
  - Radio room on 5th floor with antenna rotator controls and satellite tracking software
  - Ground station, callsign is VE3RMC
    - ✓ Have contacted the International Space Station



Roof of Sawyer Building

Communications Architecture — Types

- Five types of communications architecture are:
  - Store and Forward
  - Geostationary Orbit
  - Molniya Orbit
  - Geostationary Orbit with Crosslink
  - Low-Altitude Crosslink

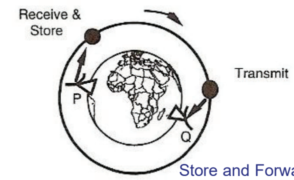
  (Geostationary Orbit with Crosslink and Low-Altitude Crosslink: Data Relay)
- Each will be discussed in the following slides

Communications Architecture — Store and Forward

- Store and Forward*:* Satellite collects payload data, stores it in memory and downlinks it at a later time
  - LEO only has about 10 minutes to communicate with ground station
  - When the satellite moves in view of a receiving ground-station the stored data is transmitted
    - ☺ No need to be in constant communication with ground station
    - ☹ Transmission delay

Viability depends on mission parameters and requirements

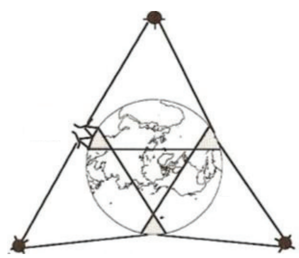
How important is it that the data be received real time?



Store and Forward

Communications Architecture — GEO Orbit

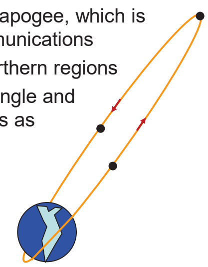
- Geostationary Orbit: Satellite with zero inclination at an altitude of 35,800 km
  - Used by communication, relay and meteorological satellites
    - ☺ Easy to set up and monitor, satellite always in view of ground-station
    - ☹ Lack of coverage above 70° Latitude
    - ☹ Require a lot of power (far away)



GEO coverage – one satellite covers ~ 1/3 of Earth not including polar regions

Communications Architecture — Molniya Orbit

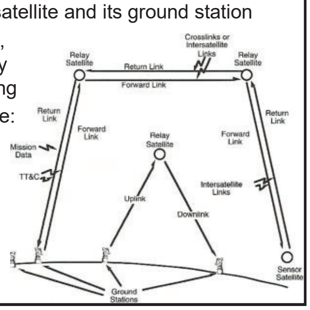
- Molniya Orbit: Highly elliptical orbit, ranging in altitude from ~500 to 40,000 km altitude in a 63.4° inclination
  - Russian communications satellites for northern regions
  - Need 3 satellites for continuous coverage
  - Each satellite spends ~ 8 hours at apogee, which is when the satellite is used for communications
    - ☺ Communications solution for northern regions
    - ☹ Constant changing of antenna angle and switching links between satellites as they move in and out of view
    - ☹ Need a lot of power (far away)



Molniya Constellation

Communications Architecture — Data Relay

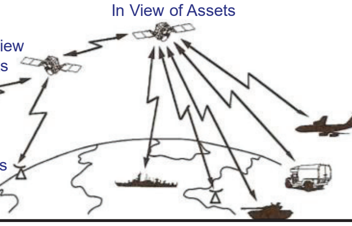
- Data Relay: A satellite, especially one in LEO, may be out of view of its ground station for extended periods
  - A second satellite or a series of satellites may be used to relay data between the satellite and its ground station
    - ✓ Known as a bent pipe, which is a simple relay with no data processing
- Common types of relays are:
  - Geostationary Orbit with Crosslink
  - Low Altitude Crosslinks



Data Relay

Communications Architecture — GEO with Crosslinks

- Geostationary Orbit with Crosslink*:* A GEO satellite is used as a relay between a ground station and a satellite that is not in view of the ground station
  - Can link LEO satellites or another GEO satellite
    - ☺ Increases coverage of a LEO satellite
    - ☹ Increased complexity

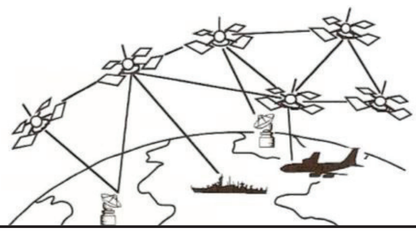


GEO with crosslinks

Communications Architecture — Low Altitude Crosslinks

- Low Altitude Crosslinks: Multiple LEO satellites are interconnected with crosslinks to extend LEO coverage
  - ☺ Highly survivable and less power required than GEO
  - ☹ Many satellites are required to maintain the LEO crosslink over a large area, leading to complex synchronization and control

LEO Crosslinks

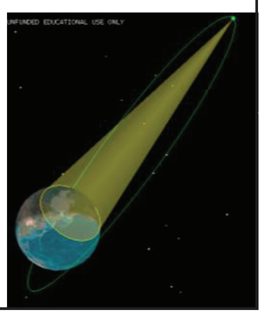


Communications Criteria

- The criteria for selecting a communications architecture are listed below
  - Orbit
  - Radio Frequency
  - Data Rate and Signal Modulation
  - Link Access
  - Threat
- These criteria will be discussed in the following slides

Communications Criteria — Orbit

- Orbit
  - Determines how long satellite is in view by the ground-station and the potential need for intersatellite links
  - Determines Earth coverage and the number of satellites required for a specified continuity of coverage
  - Is a factor in transmitter power and antenna size for communications



Molniya coverage at apogee

Communications Criteria — Radio Frequency

- Radio Frequency
  - The radio frequency used affects the satellite and ground station transmitter power and antenna size
    - ✓ Higher frequency = More power/smaller antenna
    - ✓ Lower frequency = Less power/larger antenna
  - Frequencies for satellite communications must be high enough to pass through the **ionosphere**
    - ✓ Must be higher than ~ 20 to 30 MHz to get through
    - ✓ Frequencies > 100 MHz normally used
  - Also need to avoid excessive attenuation (signal loss) in the neutral atmosphere due to water vapour and oxygen
    - ✓ Higher frequencies (>10 GHz) generally experience more attenuation due to **water vapour** and **oxygen**
    - ✓ Negligible attenuation < ~ 3 GHz

Communications Criteria — Radio Frequency

- Satellite communication frequencies, bandwidth and power are highly regulated
- Necessary to apply to and receive permission from the International Tele-communications Union (ITU)
- Frequencies are designated in bands with specific frequencies for uplink and downlink
  - Amateur satellite: VHF (~140 to 200 MHz

Frequency Band	Frequency Range (GHz)	
	Uplink	Downlink
UHF	0.2 - 0.45	0.2 - 0.45
L	1.635 - 1.66	1.535 - 1.56
S	2.65 - 2.69	2.5 - 2.54
C	5.9 - 6.4	3.7 - 4.2
X	7.9 - 8.4	7.25 - 7.75
K_u	14.0 - 14.5	12.5 - 12.75
K_a	27.5 - 31.0	17.7 - 19.7
SHF/EHF	43.5 - 45.5	19.7 - 20.7

Frequency Bands for Satellite Uplink and Downlink

Communications Criteria — Data Rate

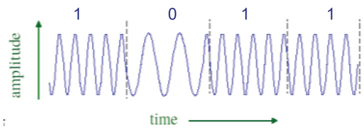
- Data Rate
  - Quantity of information per time transferred between the satellite and ground-station (or relay satellite)
    - ✓ Quantified in bits per second (bps)
  - The higher the data rate the larger the transmitter power and antenna required
    - ✓ Depends on mission requirements
    - ✓ Generally required TT&C data rate < payload data rate
  - Processing data onboard the satellite reduces the amount of data to be transmitted to the ground station but makes the satellite more complex
    - ✓ For scientific missions, scientists normally want all the data

Communications Criteria — Signal Modulation

- Data rate is a function of the signal modulation scheme
  - Signal modulation is the process by which a carrier wave is modified to carry information
- Example of modulation schemes for satellite applications:
  - Amplitude Shift Keying (ASK)
  - Phase Shift Keying (PSK)
  - Quadrature Phase Shift Keying (QSK) - phase and amplitude
  - Frequency Shift Keying (FSK)

Frequency Shift Keying (FSK)

Example: 1011 binary code

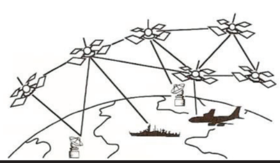


Communications Criteria — Link Access

- Link Access Time
  - Time users have to wait before they get their communication link
  - Depends on satellite function
    - ✓ Communication satellites demand link access in seconds
    - ✓ TT&C is normally in real-time
    - ✓ Meteorological data usually needed within an hour
    - ✓ Most types of remote sensing data can be stored and transmitted later
    - ✓ Real-time response for deep space missions is impossible due to propagation time of radio signals

Communications Criteria — Threat

- Threat
  - For military applications, choices of frequency, antenna, modulation need to be evaluated for susceptibility to jamming
  - If a physical threat to a satellite exists then multiple satellites may be necessary for redundancy
  - A physical threat to a ground station may demand a data relay satellite with crosslinks to allow the ground station to be relocated in safe territory



LEO Crosslinks

Link Budget — Link Equation

- The link budget describes the relationship between the mechanical properties of the communication system and the physical properties of signal propagation
  - The goal is to ensure that the satellite will be able to communicate with the ground station
- First, determine the ratio of received energy per bit to noise-density ratio, E_b/N_o (signal to noise ratio)
  - Values are in decibels (dBs) → See next slide

$$\text{Calculated } E_b/N_o = P + G_t + G_r - L_l - L_s - L_a - R - (k \cdot T_s)$$

P : Transmitter power G_t : Transmit antenna gain G_r : Receive antenna gain
L_l : Transmitter to antenna line loss
L_s : Space loss $(4\pi d / \lambda)^2$ where d = distance to satellite (use maximum d)
L_a : Transmission path loss due to water vapor and oxygen R: Data rate
kT_s : System noise temperature where T_s = system temperature and k = Boltzmann constant (1.380×10^{-23} m^2 kg s^{-2} K^{-1})

Link Budget — Decibels

- Gain is usually expressed in decibels (dB) since they are ratios
  - Gain (dB) = 10 log (Gain Ratio)
- To go from dB back to ratio:
  - Ratio = $10^{\text{dB Value}/10}$
- For powers of ten the decibel value is the number of zeros in the ratio followed by a zero
  - 0 db = 1
  - 10 db = 10
  - 20 db = 100
  - 30 db = 1000
- Negative decibel values are fractions
  - -10 db = 1/10
  - -20 db = 1/100
  - -30 dB = 1/1000

Can convert any number into dBs

Instead of multiplication values in dBs are added

Instead of division values in dBs are subtracted

Link Budget — Link Margin

- The link margin indicates how much excess signal power is available for communications
  - Link margin should be at least 3 dB (50% more than calculated) to ensure communication with the ground station

$$\boxed{\text{Link Margin} = \text{Calculated } E_b/N_o - \text{Required } E_b/N_o}$$

  - Required E_b/N_o depends on modulation scheme and bit error rate (BER)

Modulation	E_b/N_o for BER = 10^{-5} (dB)	Spectrum Utilization (bps/Hz)
BPSK	9.6	1.0
DPSK	10.3	1.0
QPSK	9.6	2.0
FSK	13.3	0.5

Bits per second per Hz ∴ higher frequency means more data per second transmitted or received

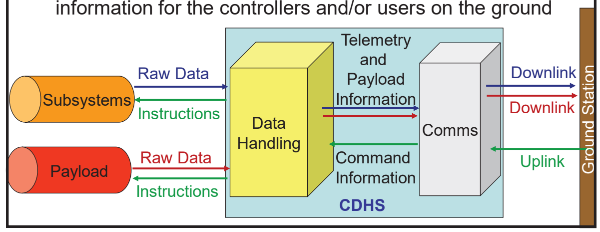
Required E_b/N_o

CDHS

- To define the Communication and Data Handling Subsystem (CDHS) it is necessary to determine:
  - The communication frequencies available
  - Physical design limitations
    - ✓ Mass, volume, power
  - The total amount of data to be handled
  - The amount of onboard processing
  - The amount of time available for down/uplink with ground stations and/or users
  - Signal modulation scheme
    - ✓ Determine if encryption is required
      - ➢ Pertinent for military applications

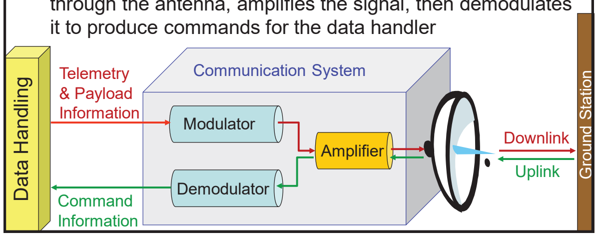
CDHS — Information Flow

- Information through the CDHS flows in two directions between the ground and the satellite
  - Commands from the ground
  - Telemetry and payload data from the subsystems and payload onboard the satellite
- The data handling unit converts raw data into useful information for the controllers and/or users on the ground



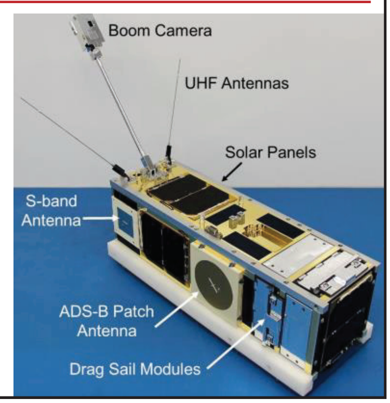
CDHS — Components

- Components of the CDHS consist of modulators, demodulators, amplifiers and antennas
- Payload and spacecraft information (i.e. telemetry) modulates onto the carrier signal, then is amplified and broadcast to the ground station through the antenna
- The spacecraft collects commands from ground stations through the antenna, amplifies the signal, then demodulates it to produce commands for the data handler



CDHS — Example - CanX-7

- CanX-7
  - S-Band uplink (~2 GHz)
  - UHF Uplink (~ 400 MHz)
  - ADS-B payload Antenna - receives in the L-Band (~ 1 GHz)



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PHE 255 Introduction to Space Science

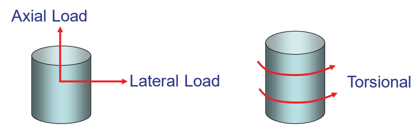
Module 12: Spacecraft Structures

- Structural Loads
- Material Properties
- Design and Testing
- Re-entry

Dr. Ron Vincent,
Royal Military College of Canada
Department of Physics and Space Science

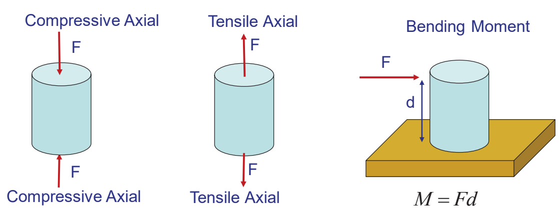
Structural Loads — Types

- There are different types of loads that are imparted on a spacecraft structure
  - Static loads are constant or relatively constant
  - Dynamic loads vary widely over time
- Static or dynamic loads can be classified based on how they are applied to a structure's axis of orientation
  - Axial loads are those applied longitudinally
  - Lateral loads are applied to the short axis of the structure
  - Torsional loads apply a torque to the structure (twist)



Structural Loads — Types

- Loads can also be compressive (push) or tensile (pull) loads
- A bending moment on an object results from a load applied at a distance away from the attachment point



$$M = Fd$$

Structural Loads — Stress and Strain

- The fundamental question with structural analysis is, "Will it break?"
  - Two parameters help to answer this question
    - ✓ Stress: Applied load over a unit area

$$\sigma = \frac{F}{A} \quad \text{N/m}^2$$

    - ✓ Strain: Changed length, or deformation, due to an applied load

$$\varepsilon = \frac{\Delta L}{L} \quad \text{No units}$$

Structural Loads — Thermal Expansion

- Stress and strain can result from a variety of loads, including environmental factors such as heating
- Most materials expand when they heat up and contract when they cool
  - A spacecraft heats up in direct sunlight and cools down in eclipse, causing materials to expand and contract

When the Hubble Telescope was first deployed, operators noticed vibrations each time it went from eclipse into sunlight

These vibrations were caused by thermal expansion and contraction

The vibrations made it difficult to focus on distant objects so the problem was eventually repaired by a shuttle mission in which thermal insulation was added

More insulation was added on the fifth and final repair mission

Structural Loads — Thermal Expansion

- The deformation caused by thermal expansion and contraction is generally not a problem if there is enough space to account for this deformation
- The amount of thermal deformation can be quantified by knowing the object's coefficient of thermal expansion

$$\Delta L = \alpha(\Delta T)L$$

Object's expansion or contraction (m) — Object's coefficient of thermal expansion (1/°C) — Change in temperature (°C) — Object's original length (m)

- Using the change in length we can find the corresponding strain

$$\varepsilon = \frac{\Delta L}{L}$$

Structural Loads — Vibrations

- Dynamic loads that vary widely and randomly are called vibrations
  - Present during launch and on orbit phases of the mission
    - ✓ Active or passive damping may be used to reduce the structure's response to vibrations
- Vibrations cause two important effects:
  - Fatigue: Vibrations have the same effect as if we bent a material back and forth many times
  - Resonance: Is the tendency of for an object to vibrate with increased amplitude due to a synchronized applied periodic force
    - ✓ More severe problem than fatigue

Structural Loads — Resonance

- The most important part of a resonance is its lowest natural frequency, known as the fundamental frequency
- If the input frequency of a vibration matches the fundamental frequency, each vibration will amplify the structure's oscillation
  - Can lead to very large-amplitude oscillations
- To calculate the fundamental frequency, f_{fund}, of a structure, we can assume that it acts like a large spring

$$f_{fund} = \sqrt{\frac{k}{m}}$$

k → Spring constant (Nm^{-1}) → High k = stiff object

m → Mass (kg) → High mass = lower fundamental frequency

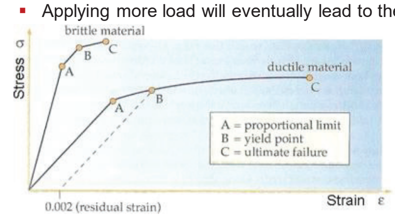
- In order to avoid resonance, ensure that the fundamental frequency of the structure is different from the frequency of the forcing vibration

Material Properties

- The ultimate effect of stress, strain, vibration, bending and thermal loading depends on the material used in the structure
- The main factors considered when building the satellite structure are:
  - Mechanical Properties
    - ✓ Indicate how materials respond when subjected to stress and strain
  - Physical Properties
    - ✓ Physical characteristics of the material
  - Ease of Fabrication
    - ✓ Is the material readily available and is it easy to use?
  - Cost
    - ✓ Must operate within the budget of the satellite mission

Material Properties — Mechanical Properties

- Relationship between stress and strain is linear in the elastic zone
  - The maximum amount of stress and strain in the elastic zone is referred to as the proportional limit (no deformation of material)
- Beyond the proportional limit, the yield point is where a residual strain of 0.2% remains after the load is removed
  - Material will be permanently deformed by 0.2%
- Applying more load will eventually lead to the ultimate failure point



Stress-Strain Curves for a Brittle and Ductile Material

Any material that can withstand a significant strain before failing is ductile

A material that breaks after only a small amount of strain is brittle

Material Properties — Mechanical Properties

- Based on stress-strain curves the strength of a material can be described in terms of Yield Strength and Ultimate Strength
  - e.g. Aluminum: yield strength = 95×10^6 N/m²
    ultimate strength = 110×10^6 N/m²
  - e.g. Steel: yield strength = 250×10^6 N/m²
    ultimate strength of 400×10^6 N/m²
- Young's Modulus describes the amount of deformation a material undergoes for a given amount of stress
  - e.g. Aluminum: Young's Modulus of 69×10^9 N/m²
  - e.g. Steel: Young's Modulus of 200×10^9 N/m²
    - ✓ Indicates how well the material will withstand vibrations

$$E = \frac{\sigma}{\varepsilon}$$

Young's Modulus (N/m^2) (Modulus of elasticity) → E

σ ← Stress (N/m^2)

ε ← Strain (ratio)

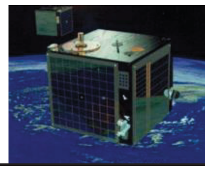
Material Properties — Physical Properties

- Physical material properties for a satellite structure include:
  - Outgassing occurs when minute quantities of gas trapped in a material leak out under vacuum conditions
    - ✓ Leaking gases can damage lenses or cause shorts in electrical equipment
  - Radiation tolerance indicates how well a material can withstand radiation doses in the space environment
  - Magnetic properties of materials must be considered since they could interfere with devices such as magnetometers
    - ✓ Don't want structure to become magnetized
  - Density of a material is also important since it is desirable to use the lightest materials possible
    - ✓ Aluminum: density ~ 2800 kg/m³
    - ✓ Steel: density ~ 7860 kg/m³

Material Properties — Physical Properties

- Normally we assume that a material's mechanical and physical properties are homogeneous
  - Not true for composite materials
- Composite materials are carefully tailored to carry a specific load in a specific way
  - Strong in a particular direction
- The advantage with composites is that they can have the strength of a metal at the fraction of the weight

Space Technology Research Vehicle (DRTV) built by the UK Defence Research Agency used carbon/PEEK thermosplastic and aluminum honeycomb panels to reduce mass



Minisat 45 cm × 45 cm

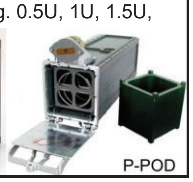
Material Properties — Ease of Fabrication

- Ease of fabrication is another important factor to consider
  - Certain materials may be able to handle the load, but may be too difficult or dangerous to work with
    - ✓ e.g. Titanium is very strong and has good high temperature qualities, but is more difficult to machine than Aluminum
    - ✓ e.g. Beryllium has many advantageous material properties, but it is toxic
  - Aluminum is the most popular construction material due to strength and light weight
- The structure of satellites is often based on a proven concept that has already been flown in space
  - Example: CubeSat kit structures

Material Properties — Cost

- Cost is one of the biggest drivers in deciding what type of material to use
  - Need to balance out the best material properties with cost
  - Space proven spacecraft structures reduce cost and risk
  - Example: Commercial-off-the-shelf (COTS) CubeSat kit
    - ✓ The 10 x 10 x 10 cm (1U) CubeSat standard has evolved to become the basis for one of the most widely accepted families of nanosatellite designs
      - Available in a variety of sizes: e.g. 0.5U, 1U, 1.5U, 2U or 3U
    - ✓ Works with the Poly Picosatellite Deployer (P-POD)

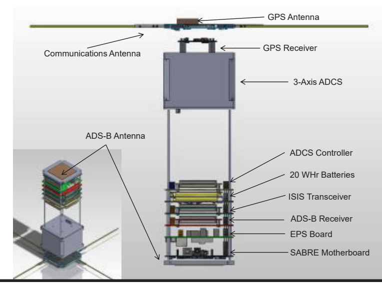
1U Cubesat



P-POD

Material Properties — Example - SABRE

- SABRE is a 3U satellite
  - Will use a Pumpkin 2U + 0.3U structure
    - ✓ Aluminum alloy (5052-H32)
  - ADCS comprises 0.7U

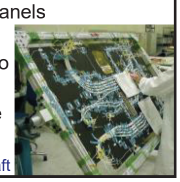


Design and Testing — System Overview

- Typically, the spacecraft structure accounts for about 10% to 20% of the spacecraft's total dry weight
  - Dry weight is the total weight minus propellant
- Primary Structure
  - Maintains the physical integrity of the spacecraft
  - Carries most of the loads that the spacecraft must withstand
- Secondary Structure
  - Accommodates the mechanical configuration of the payload and subsystems
  - Holds the together all the mechanisms, cameras, wires, pipes, doors and brackets inside and outside the spacecraft

Design and Testing — Mechanical Configuration

- The mechanical configuration must be designed to meet volume and mass constraints
  - Antennas, sensors, solar panels, radiators and payloads compete for valuable external space on the satellite
    - ✓ e.g. Do not want an antenna shading a solar panel
  - Propulsion system needs pipes to carry propellant between the storage tanks and thrusters
  - Mechanisms must be carefully integrated into the structure
    - ✓ e.g. Actuators that deploy the solar panels
  - All of the spacecraft components must be interconnected by a wiring harness to carry power and data
  - Must fit inside the payload fairing on the launch vehicle



Wiring on the Contour Spacecraft

Design and Testing — Systems Engineering

- Structural engineers translate all of the general requirements and constraints to determine:
  - Strength
    - ✓Ensure the structure won't break
  - Stability
    - ✓Ensure the structure remains stable in the launcher
  - Stiffness
    - ✓To resist vibrations in the launcher and on orbit
  - Fundamental frequency
    - ✓Avoid resonance with launch vehicle vibration modes
  - Physical properties
    - ✓Compute mass, volume, center of gravity, moments of inertia
  - Layout of mechanical interfaces
    - ✓What needs to attach where and how

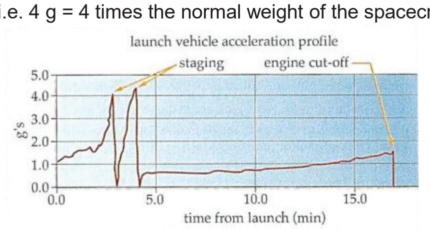
Design and Testing — Systems Engineering

- During the launch the most important structural aspects of a spacecraft are the *strength* and *stability*
  - This ensures the spacecraft can withstand the launch and is stable enough to not move in the launcher fairing
  - Also need to ensure that the fundamental frequency of the primary structure is well above or well below the major forcing frequency of the launch vehicle
    - ✓ Must avoid mechanical resonance
- Once the spacecraft is on orbit, *stiffness* is the most important structural property to counteract the effects of small vibrations

Satellite and Fairing

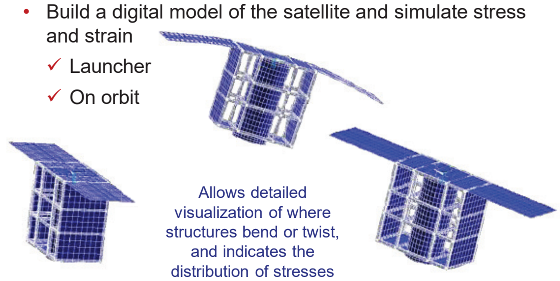
Design and Testing — Launch Environment

- Typically, the most demanding requirements come from the launch environment
- With respect to the strength of the structure, designers are most concerned with the rapid acceleration, or g-load
  - During the early phases of launch an excess of 4 g is not uncommon
    - ✓ i.e. 4 g = 4 times the normal weight of the spacecraft



Design and Testing — Finite element Modeling

- The structure can be evaluated using finite element modeling (FEM)
  - Build a digital model of the satellite and simulate stress and strain
    - ✓ Launcher
    - ✓ On orbit



Allows detailed visualization of where structures bend or twist, and indicates the distribution of stresses and displacements

Design and Testing — Pre-Launch Test

- The structure undergoes extensive testing prior to launch to estimate its strength, stiffness and natural frequency
- Once built, the structure undergoes qualification, which is a mandatory set of tests that simulate the launch environment
  - Vibrations, shocks, acoustic noise etc.
  - This complements thermal, radiation and vacuum testing
    - ✓ "Shake and bake"

ESA's Gaia payload module in its X-axis shake test configuration

Gaia will make the largest, most precise 3D map of our galaxy by surveying 1% of its 100 billion stars

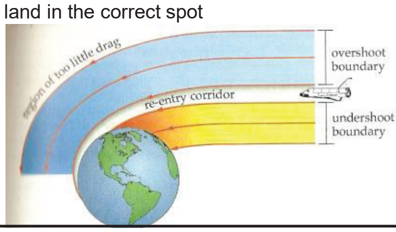
Re-Entry — Considerations

- If a spacecraft must re-enter the atmosphere safely, there are two major design concerns
  - Heating and deceleration
- Heating is due to friction between spacecraft and atmosphere
  - Temperatures are extremely high during re-entry
- The maximum deceleration a vehicle experiences during re-entry must be low enough to prevent damage or injury to the weakest part of the vehicle
  - Also, if there is too little deceleration, a spacecraft may bounce off the atmosphere
    - ✓ See next slide

Hayabusa canister with asteroid samples after re-entry

Re-Entry — Corridor

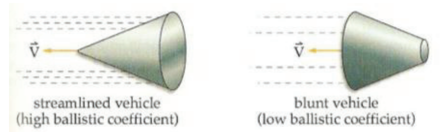
- The re-entry corridor is a narrow region in space that a re-entering vehicle must fly through
  - If the vehicle strays above the corridor, it may skip out
  - If the strays below the corridor, it may burn up
- If the space vehicle is unable to maneuver once in the lower atmosphere, it must enter this corridor in such a way that it will land in the correct spot



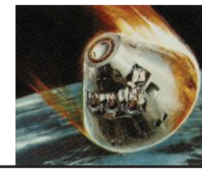
Re-entry Corridor

Re-Entry — Design

- Streamlined versus blunt re-entry vehicle
  - A light blunt vehicle will slow down more rapidly than a heavy, streamlined vehicle, but it will heat up more rapidly

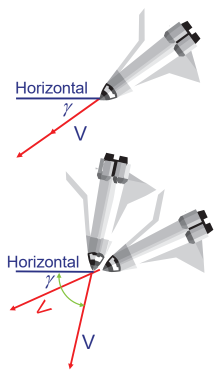


- The blunt side of the Apollo capsule allowed for aerodynamic braking
- Heat shields protected the astronauts inside



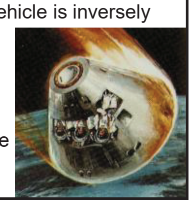
Re-Entry — Deceleration and Heating

- The deceleration and heating is a function of the flight path angle, γ, and the re-entry velocity, V
  - For a given γ, the greater the re-entry velocity the greater the maximum deceleration and heating
  - For a given re-entry velocity, the greater the γ the greater the maximum deceleration and heating



Re-Entry — Deceleration and Heating

- Deceleration is measured in g's
  - Number of g's for various re-entry vehicles
    - ✓ Space Shuttle: < 3 g
    - ✓ Soyuz Launcher: < 4 g
    - ✓ Apollo Missions: ~ 6 g
    - ✓ Mercury Missions: Upward to 11 g, which is the tolerance limit of humans
- The heat load experienced by an entry vehicle is inversely proportional to the drag coefficient
  - For a blunt re-entry vehicle, air cannot get out of the way quickly enough and acts as buffer zone for the spacecraft
  - This principle was incorporated into the shield designs of the Mercury, Gemini and Apollo space capsules



Re-Entry — Thermal Protection

- Thermal-protection systems available for re-entry, include:
  - Heat Sinks
    - ✓ Use material that absorbs the heat to keep the peak temperature lower
  - Ablation
    - ✓ Coat the vehicle with a material that has a very high latent heat of fusion
      - As material melts or vaporizes it absorbs heat
        - » Used by Apollo missions and Russian missions
  - Radiative Cooling
    - ✓ If an object being heated has a high emissivity it will emit almost as much as it absorbs
      - High emissivity radiators emit the heat before it is absorb by the structure
        - » Principle that was used by the Space Shuttle

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