Alabama Space Grant Consortium
Alabama CubeSat Initiative

ABEX
Alabama Burst Energetics eXplorer

Intro to Spacecraft Engineering

Small Satellite Engineering Primer for New Spacecraft Engineers

University of Alabama in Huntsville – Management Team
Michael Halvorson, ABEX Chief Engineer

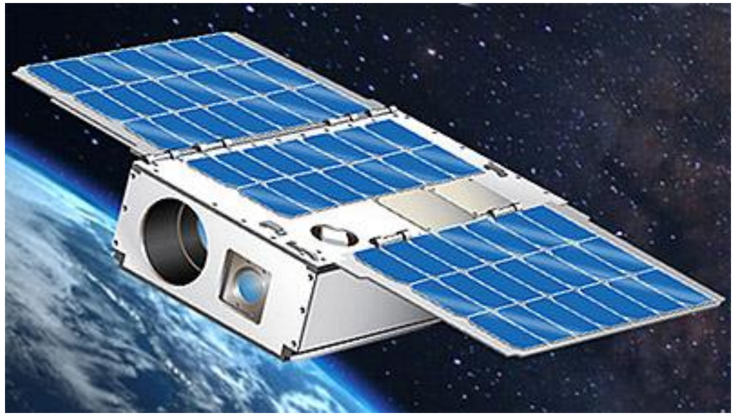
Agenda

1. Spacecraft Overview
2. Orbital Mechanics
3. Structural Integrity
4. Thermal Control
5. Command & Data Handling
6. Telemetry, Tracking, & Command
7. Guidance, Navigation, & Control
8. Electrical Power System

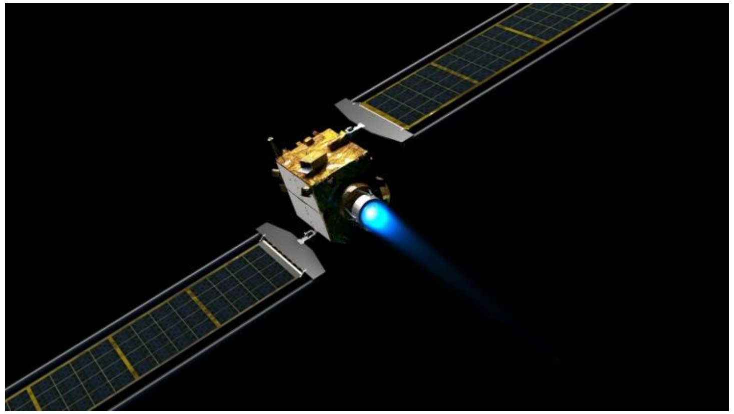
Spacecraft Overview

The Realm of the Possible

- Satellite: Any object in space that orbits another, bigger object
  - Natural vs. artificial
- Types of spacecraft
  - Vehicles (human-rated)
  - Satellites (artificial)
    - Big Bois
    - SmallSats
    - Fabricators
    - Mining Robots
  - Landers
  - Rovers
  - Aircraft* (Ginny)
- Today's Focus: SmallSats



ASTERIA [1]



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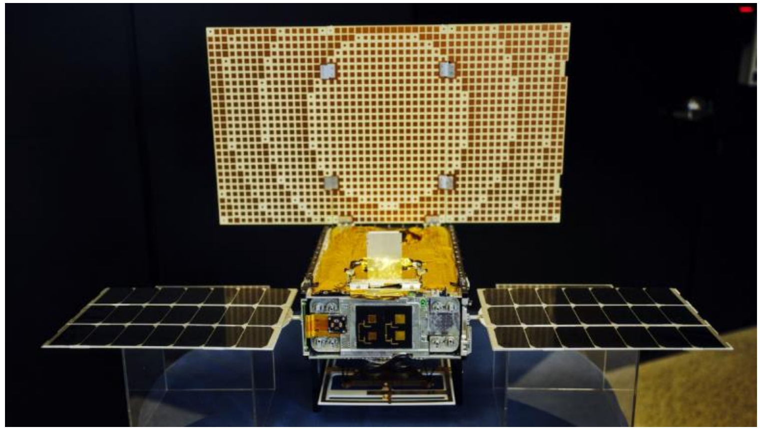


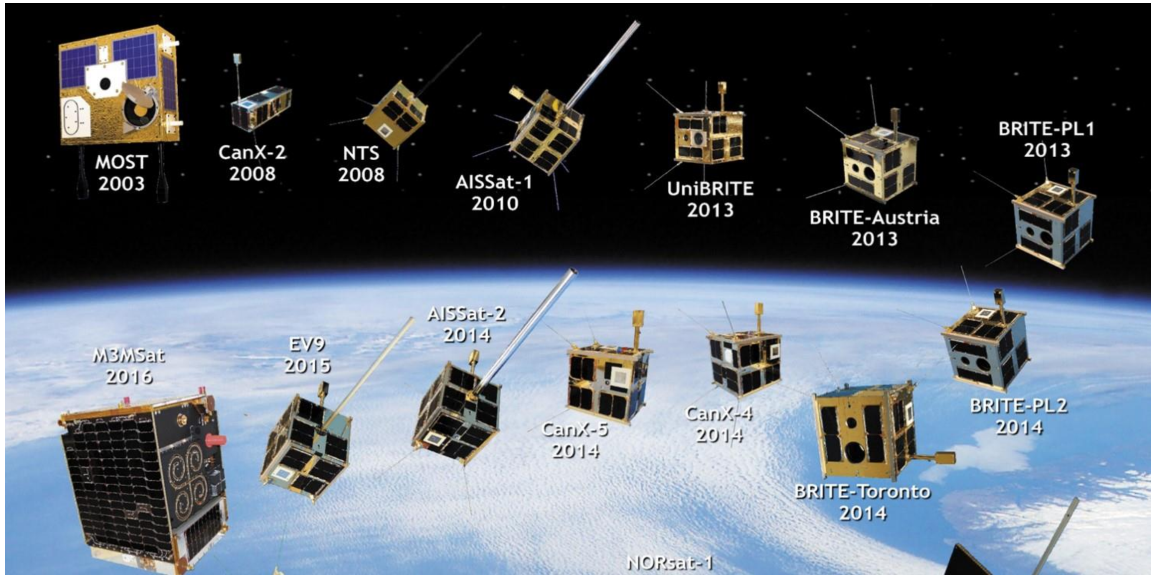
MarCO [3]

[1] O'Neill, 2021 [2] GSP, 2022 [3] Stirone, 2019

What Kinds of Small Satellites Exist?

Classes

- Civil
- Commercial
- Military
- University



Space Flight Laboratory Heritage Satellites [4]

Starlink Constellation Deployment [5]

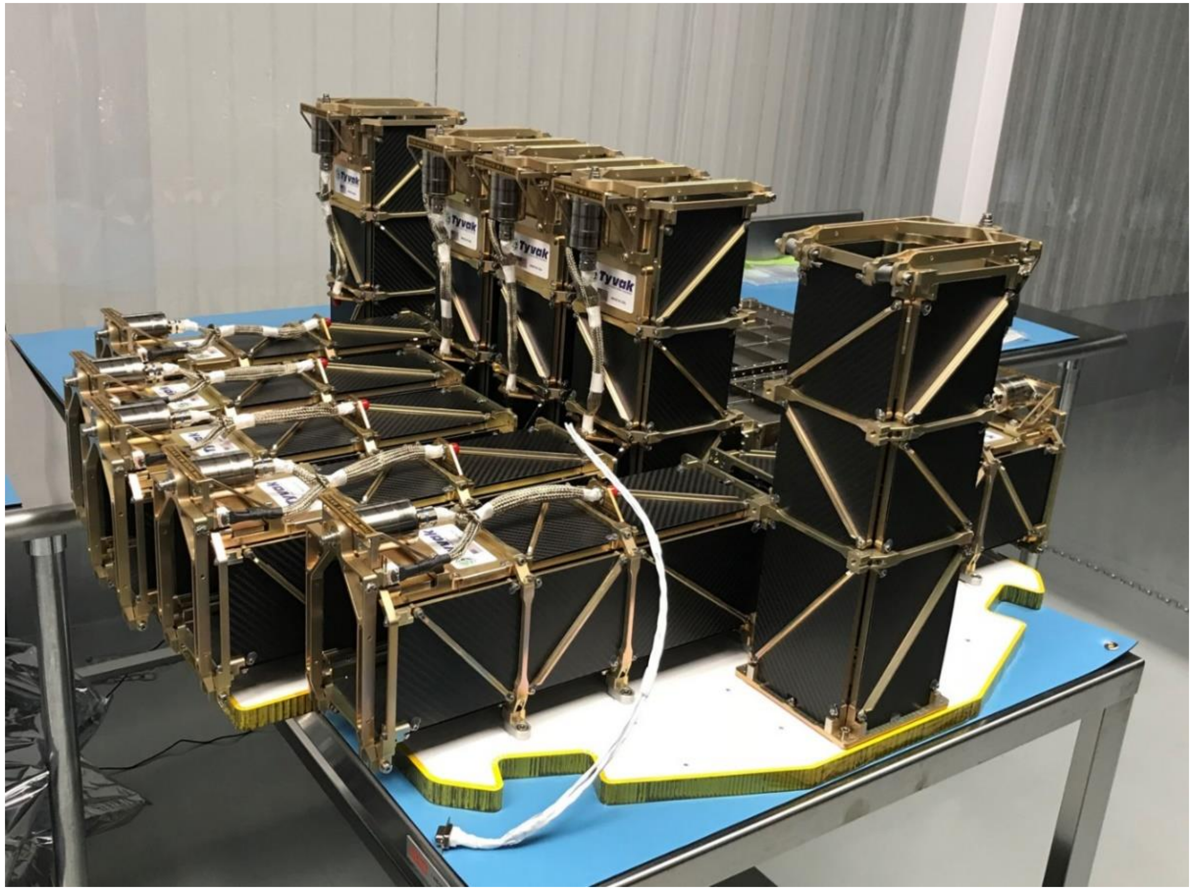
Mission Types

- Communications
- Earth Imaging
- Military
- Science
- Technology Demonstration

[4] GW, 2018 [5] Wikimedia, 2019 [6] Swartwout, 2018

Why Small Satellites?

- Hardware Cost Savings
  - Mars Reconnaissance Orbiter, $416,600,000 [7]
  - Lunar Reconnaissance Orbiter, $583,000,000 [8]
  - First 10 GPS satellites, ~$577,000,000 [9]
  - ABEX, ~$3,000,0000
- Size
  - Can fit several onto one LV
- Component Availability
  - Entire markets exist for parts



Tyvak Fit Check on Rocket Lab's Photon Platform [10]

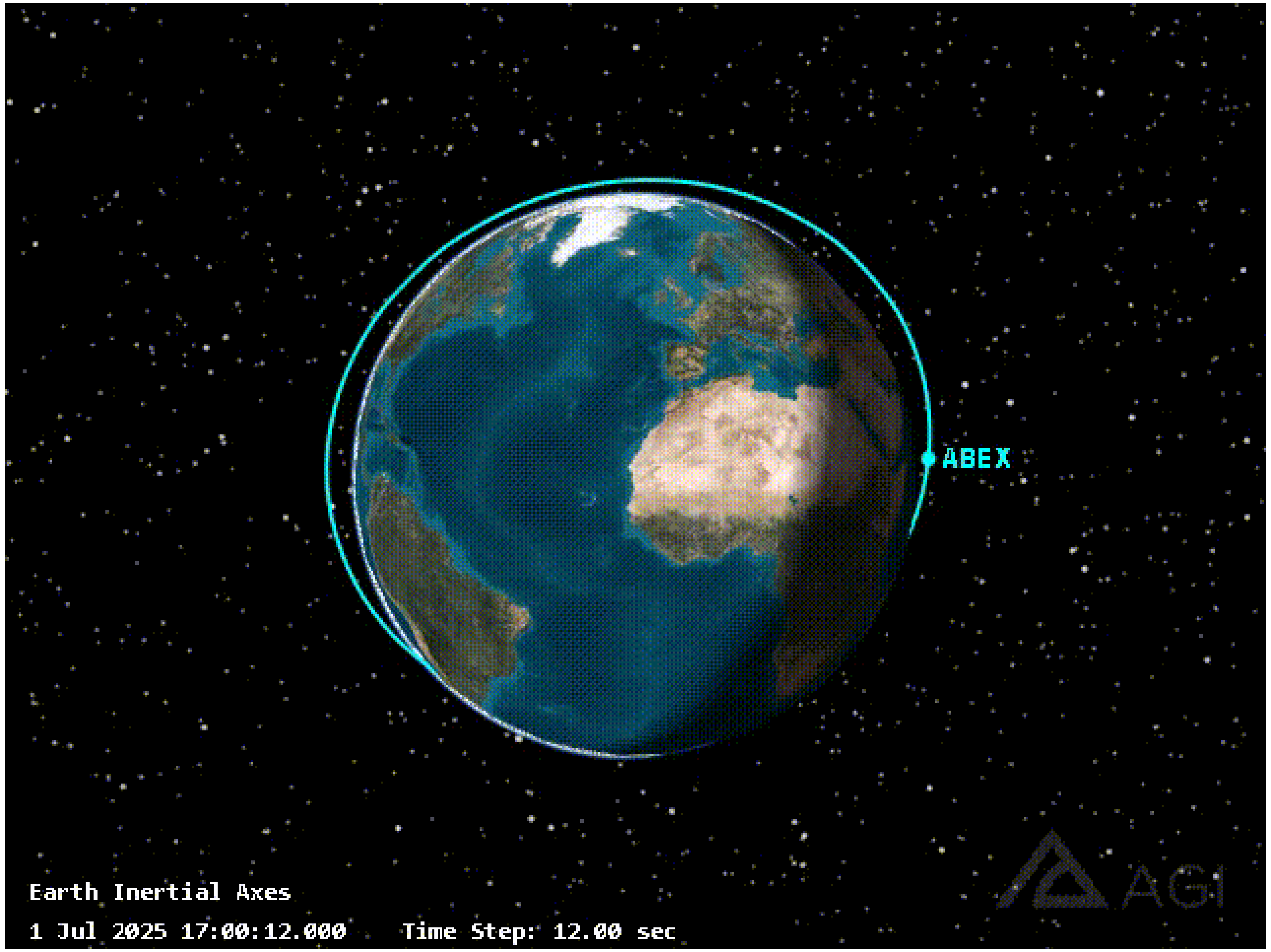
[7] Planetary Society, 2020 [8] CNET, 2009 [9] Elliot, 2018 [10] Rocket Lab, 2018

What Does a Small Satellite Program Need?

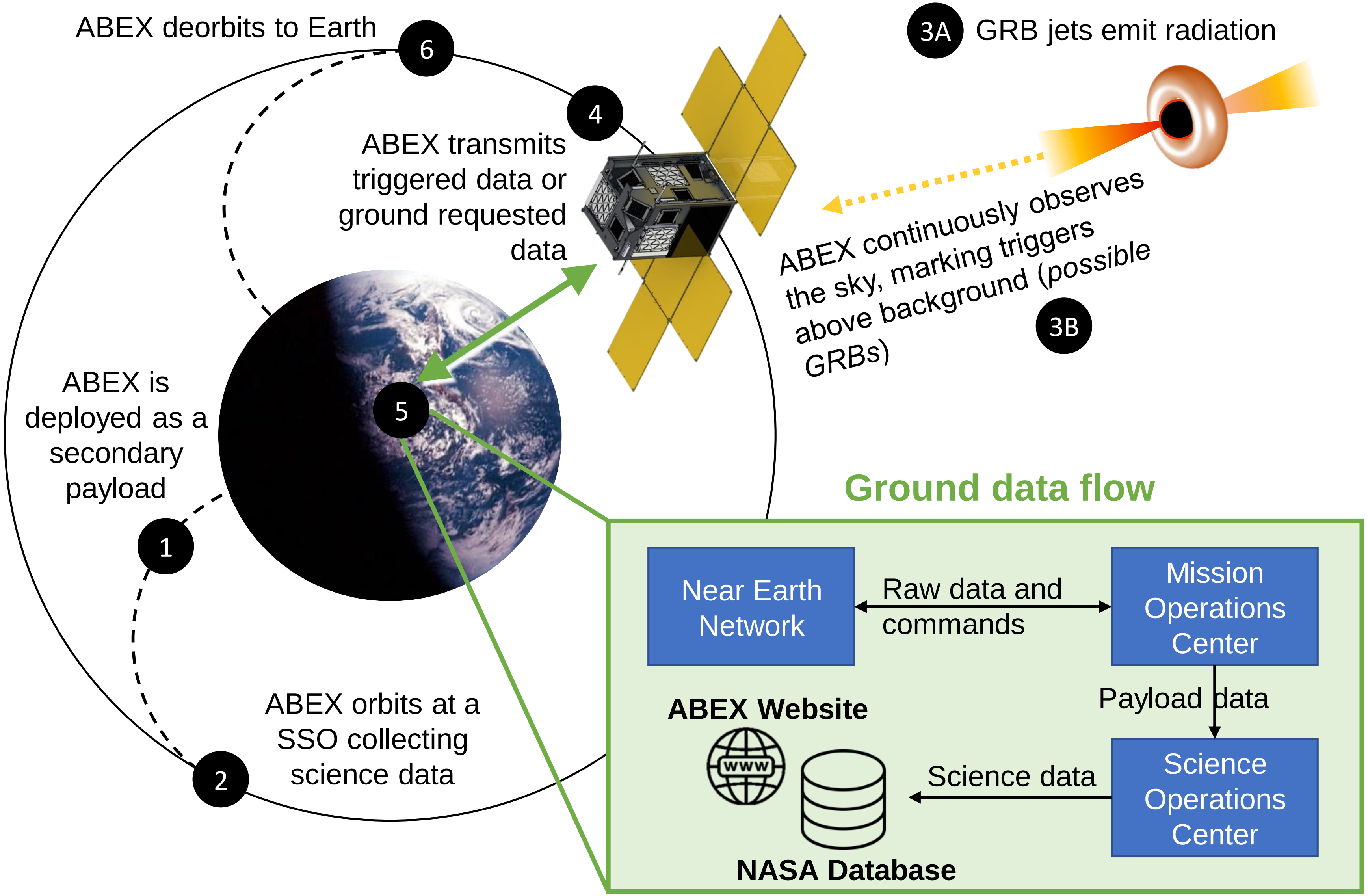
- Mission
  - Operational Concept (OpsCon)
    - What and Why
  - Concepts of Operations (ConOps)
    - How and To What End
- Management
  - Principle Investigator, Project Manager, Chief Engineer, Deputies
- Systems Engineering
  - Life Cycles, Requirements, V&V, AI&T, Safety & Mission Assurance
- Hardware/Software Development Teams
- Launch Vehicle Manifest Opportunities
- Funding

What is ABEX?

- Satellite: CubeSat
- Class: University
- Mission Type: Science
- Size: Microsatellite (10-100 kg)
- Form Factor: 12U
- Orbit: Sun-Synchronous
- Excellent: Yes



ABEX ConOps


ABEX deorbits to Earth
6
4
ABEX transmits triggered data or ground requested data
3A
GRB jets emit radiation
ABEX continuously observes the sky, marking triggers above background (possible GRBs)
3B
ABEX is deployed as a secondary payload
5
1
2
ABEX orbits at a SSO collecting science data
Ground data flow
Near Earth Network
Raw data and commands
Mission Operations Center
Payload data
ABEX Website
www
Science data
Science Operations Center
NASA Database

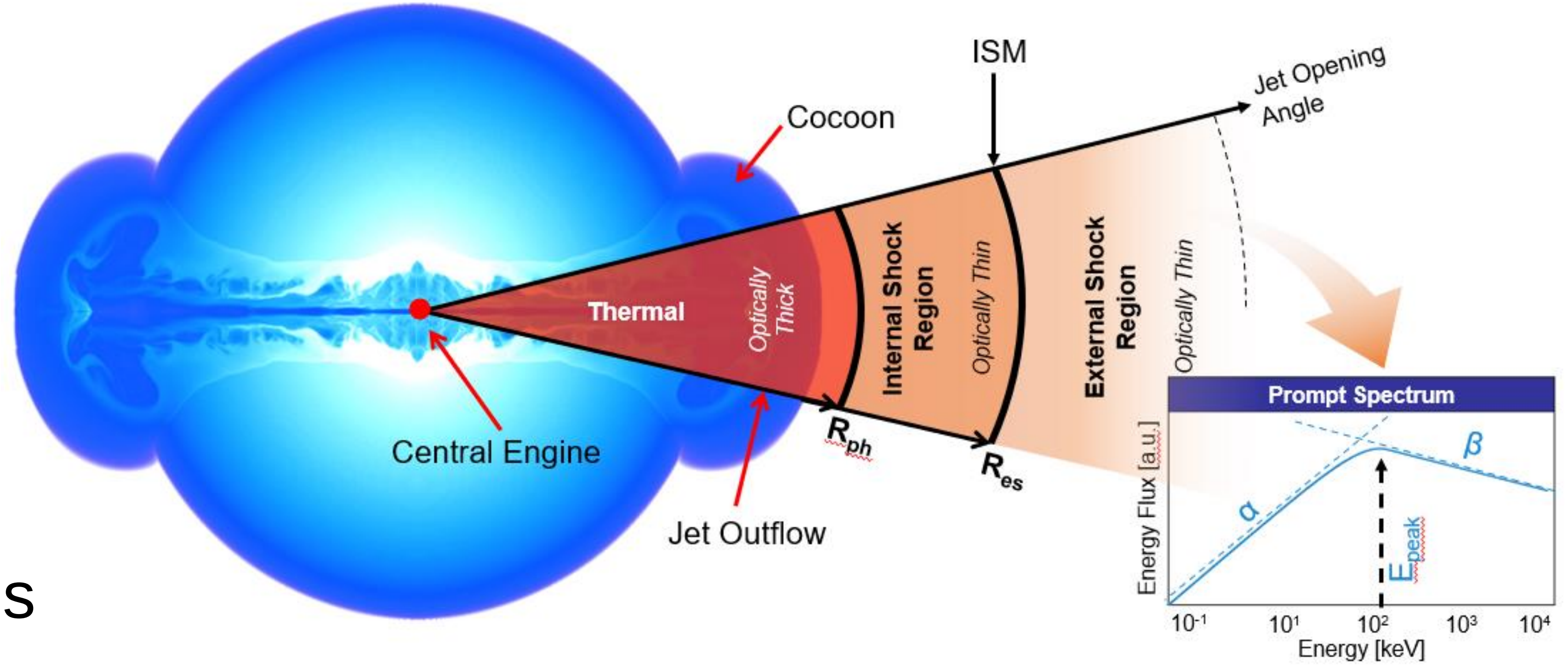
The ABEX Science Mission: Gamma-Ray Bursts

Gamma-ray Bursts (GRBs)

- Bright events of gamma-ray emissions that are **isotropic** in the sky and **random in time**

Features:

- Prompt emission from seconds to minutes (*γ/X-ray*)
- Afterglow from hours to months (*X-ray/Radio*)
- Short and long types



The ABEX Science Mission: Gamma-Ray Bursts

Unresolved Key Questions (KQ)

KQ1: How is energy dissipated within the jet?

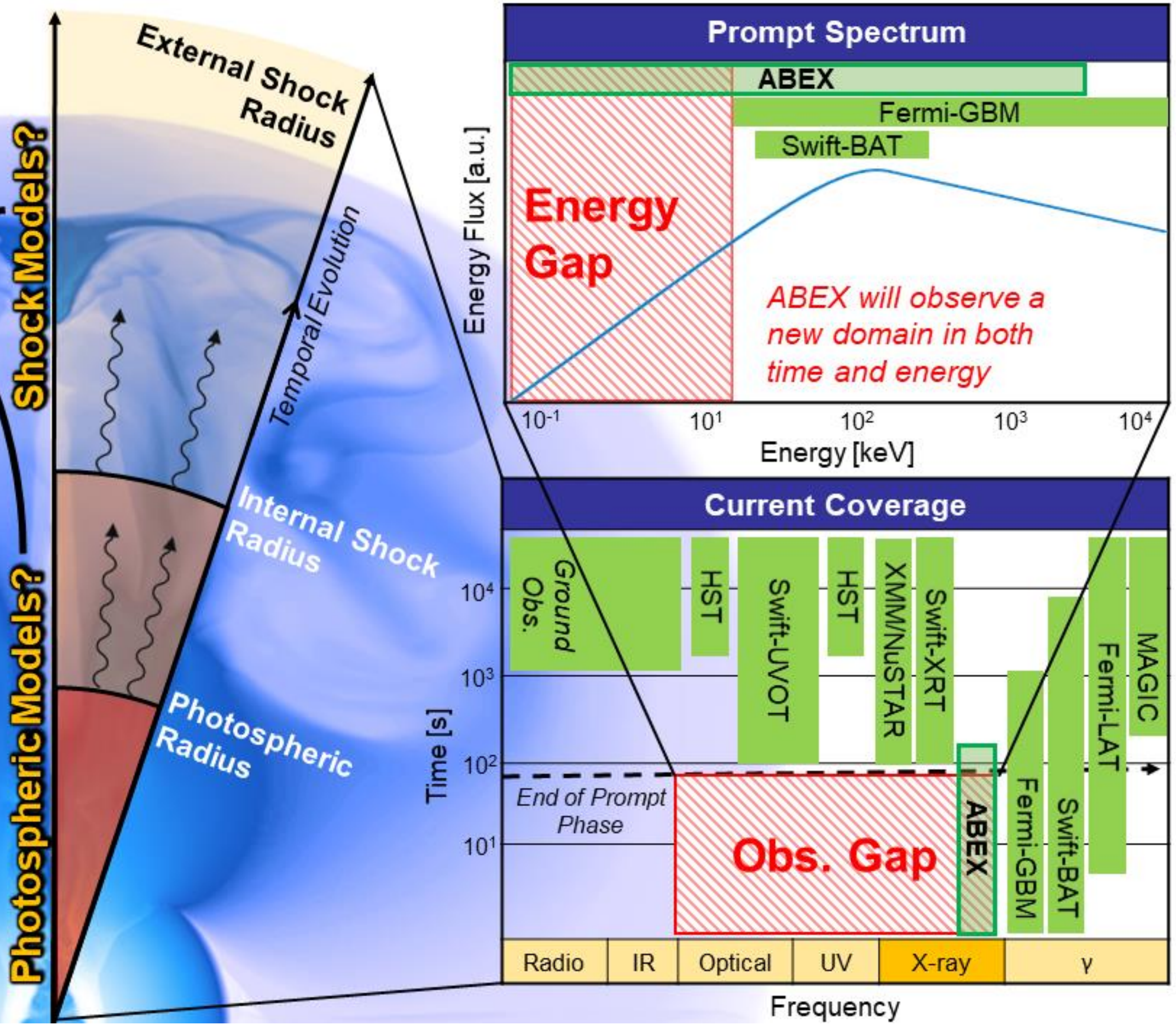
KQ2: What is the composition of the jet?

KQ3: What mechanisms are responsible for the prompt radiation?

Address KQs in the GRB community

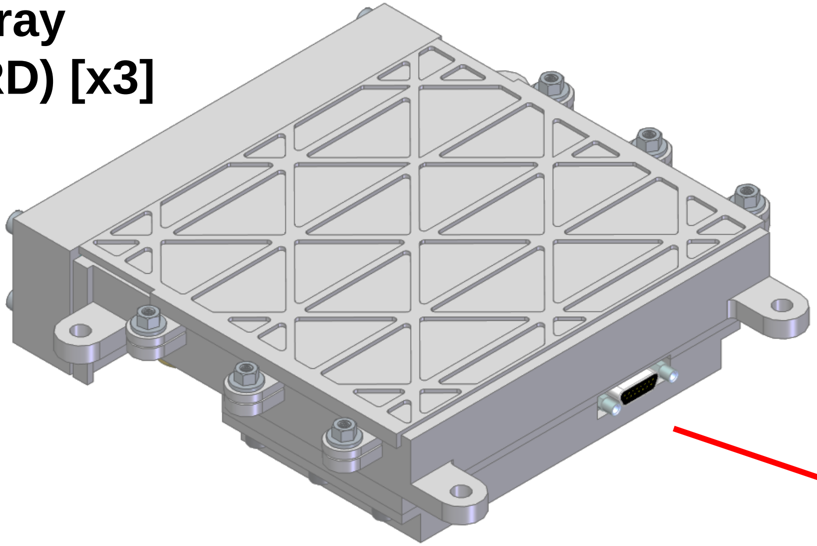
Science Objective 1:
Distinguish between photospheric and internal shock models.

Science Objective 2:
Distinguish between local and continuous dissipation around the photosphere.

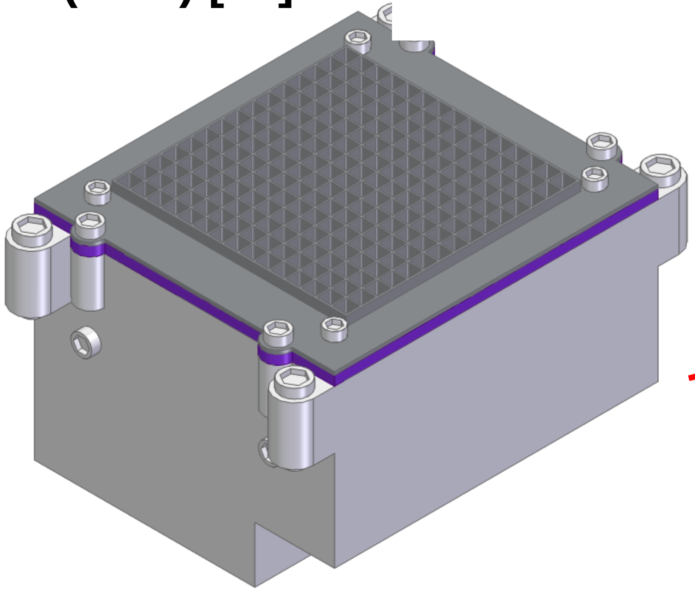


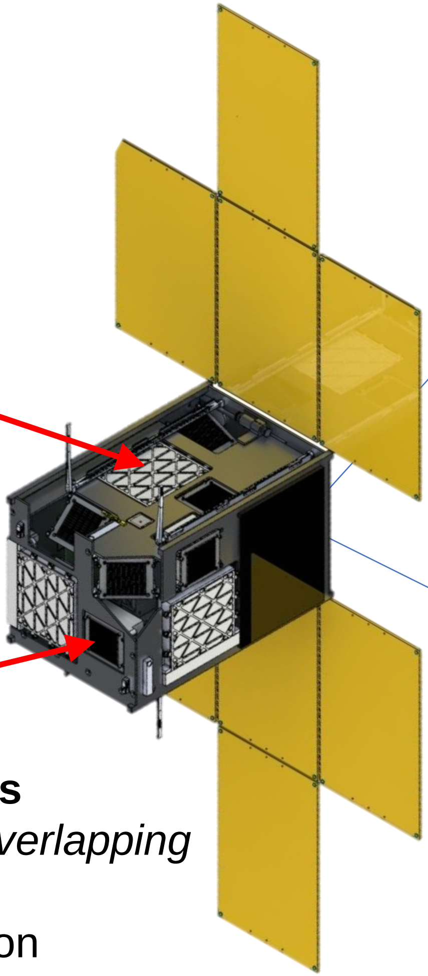
Elements of the Science Payload

Gamma-ray Detector (GRD) [x3]

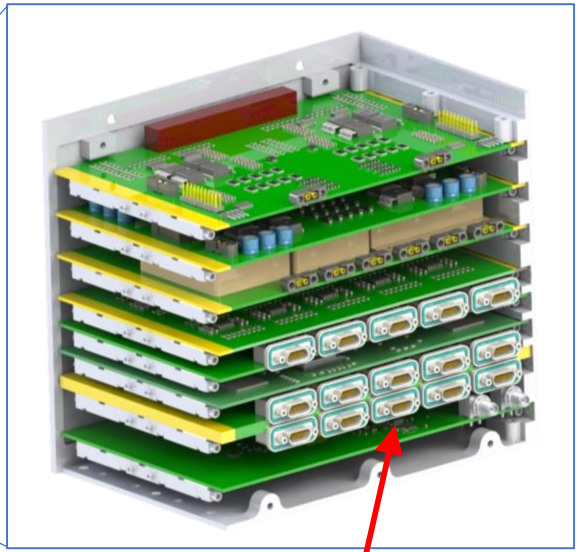


X-ray Detector (XRD) [x6]





Avionics Stack



Payload Interface Unit (PIU)

Key Objectives

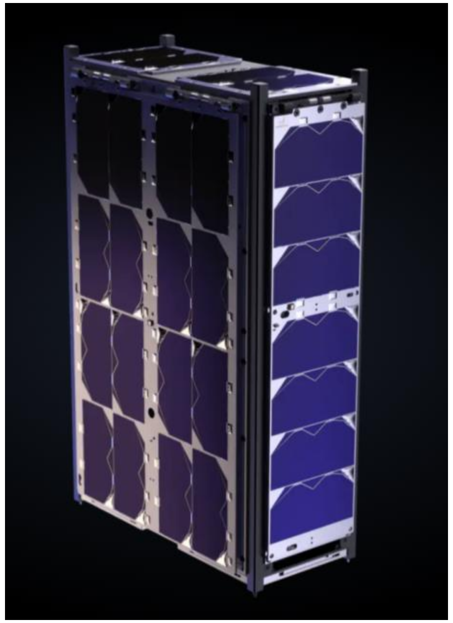
- High-field of view (*with overlapping XRD and GRD*)
- High uptime of observation
- Large energy range (1 keV – 1 MeV)

What's In A Spacecraft?

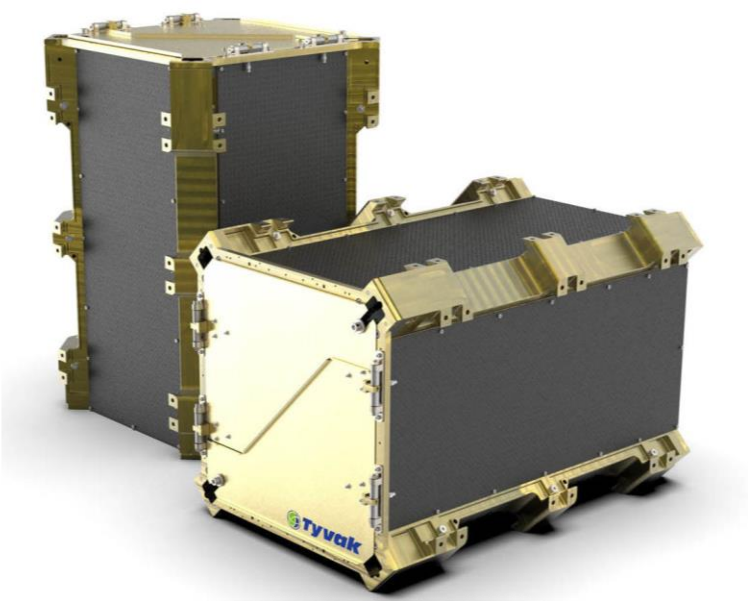
- Hardware Subsystems
  - Structural Integrity
  - Thermal Control
  - Command & Data Handling (C&DH)
  - Telemetry, Tracking, & Command (TT&C)
  - Guidance, Navigation, & Control (GN&C)
  - Electrical Power System (EPS)
  - Payload
  - Propulsion (maybe)
- Flight Software – Not discussed in this presentation
- Honorable Mention: Orbital Mechanics & Astrodynamics

What Kind of Commercial Support Exists?

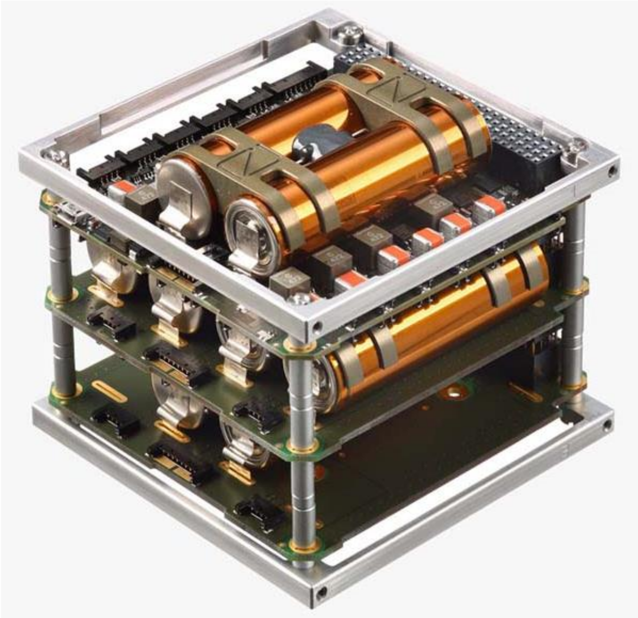
- SmallSats are generally divided into a Payload & a "Bus"
  - CubeSat Buses can be purchased. CubeSats are SmallSats
- CubeSats (still SmallSats) are deployed from dispensers
- Components are designed to CubeSat form factors



EnduroSat 6U CubeSat Bus [11]



Tyvak 12U CubeSat Dispenser [12]



NanoAvionics CubeSat EPS [13]

[11] EnduroSat, 2021 [12] Tyvak, 2022 [13] NanoAvionics, 2022

Subsystem Technical Performance Measures

- TPMs will be provided for all hardware/software subsystems
  - Big ones will be mentioned, but your team can add more as needed

- Mass, Volume, and Power aren't on the lists
  - They're considered givens
  - Need to be reported to the Systems Engineering team each Design Analysis Cycle

Orbital Mechanics

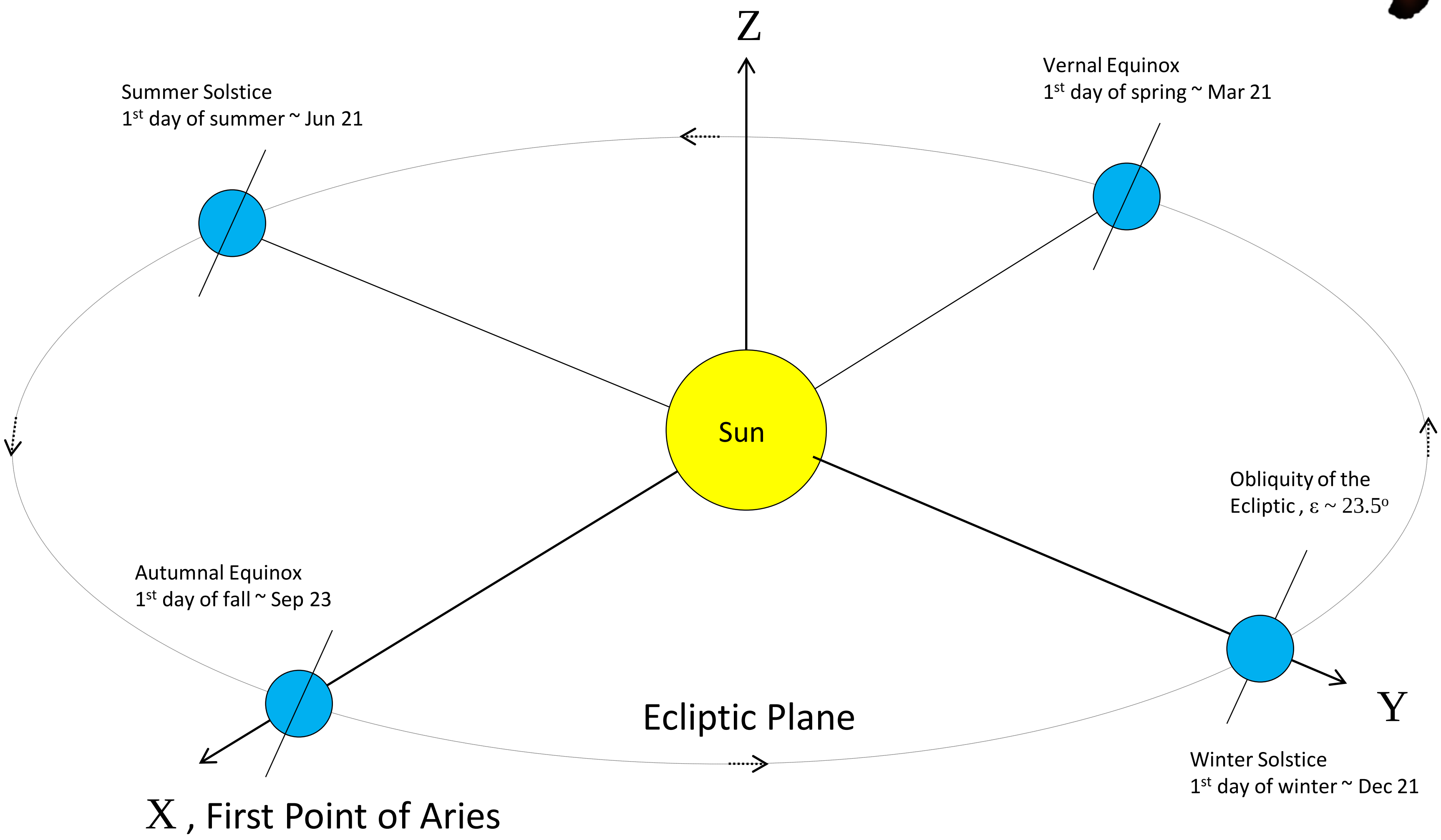
Orbital Mechanics Concepts

- Coordinate Frames
- Determining Time
- Two-Body Equations of Motion
- Kepler’s Laws & Conic Section Motion
- Position and Velocity as a Function of Time
- Three-Dimensional Orbit in Space
- Launching & Ground Tracks
- Orbital Maneuvers
- Interplanetary Trajectories

[14] Cicci, 2010

Coordinate Frames – Heliocentric Ecliptic

- Origin at center of Sun
- X axis points toward First Point of Aries, which is now at Pisces
- XY plane is Earth orbit
- Z axis parallel to Sun North Pole

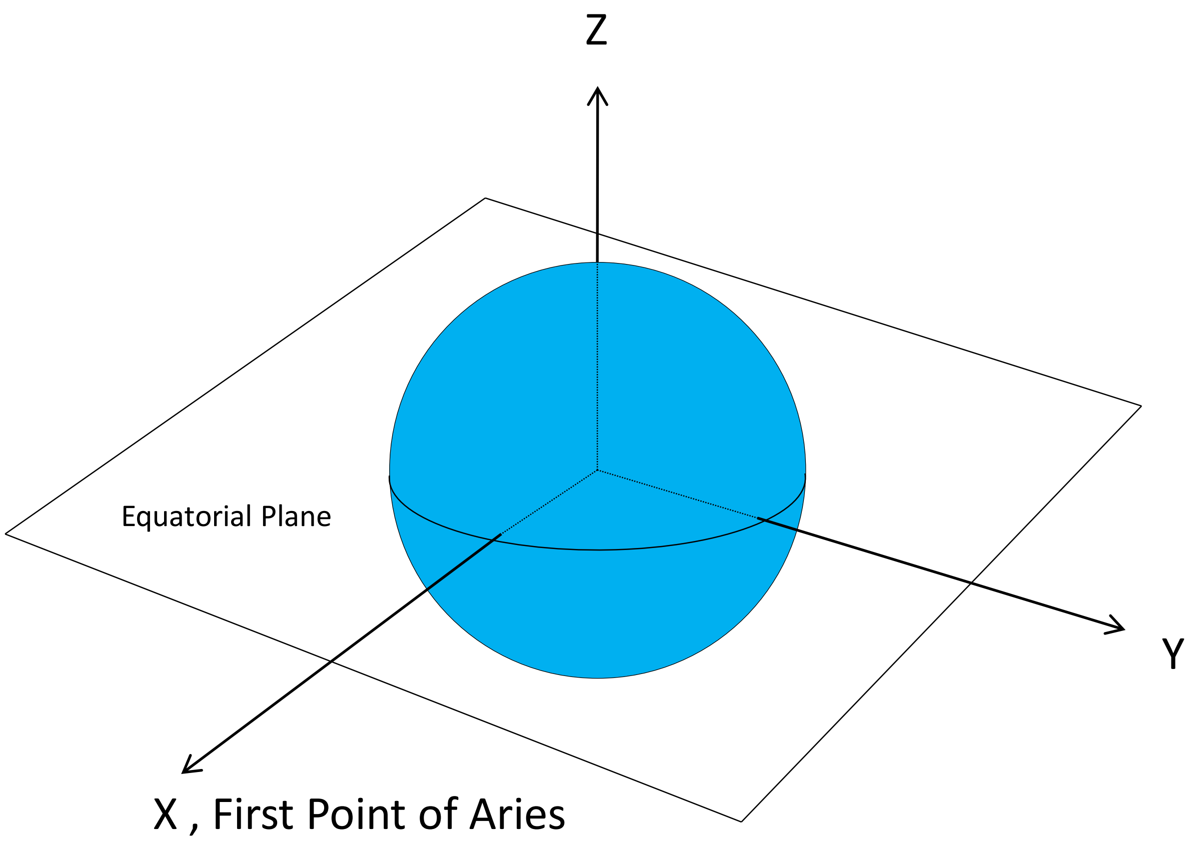


Heliocentric-Ecliptic Coordinate System [14]

[14] Cicci, 2010

Coordinate Frames – Geocentric-Equatorial

- Origin at center of Earth
- Also called Earth Centered Inertial (ECI)
- X axis to vernal equinox
- XY plane is equatorial
- Z axis parallel to Earth North Pole

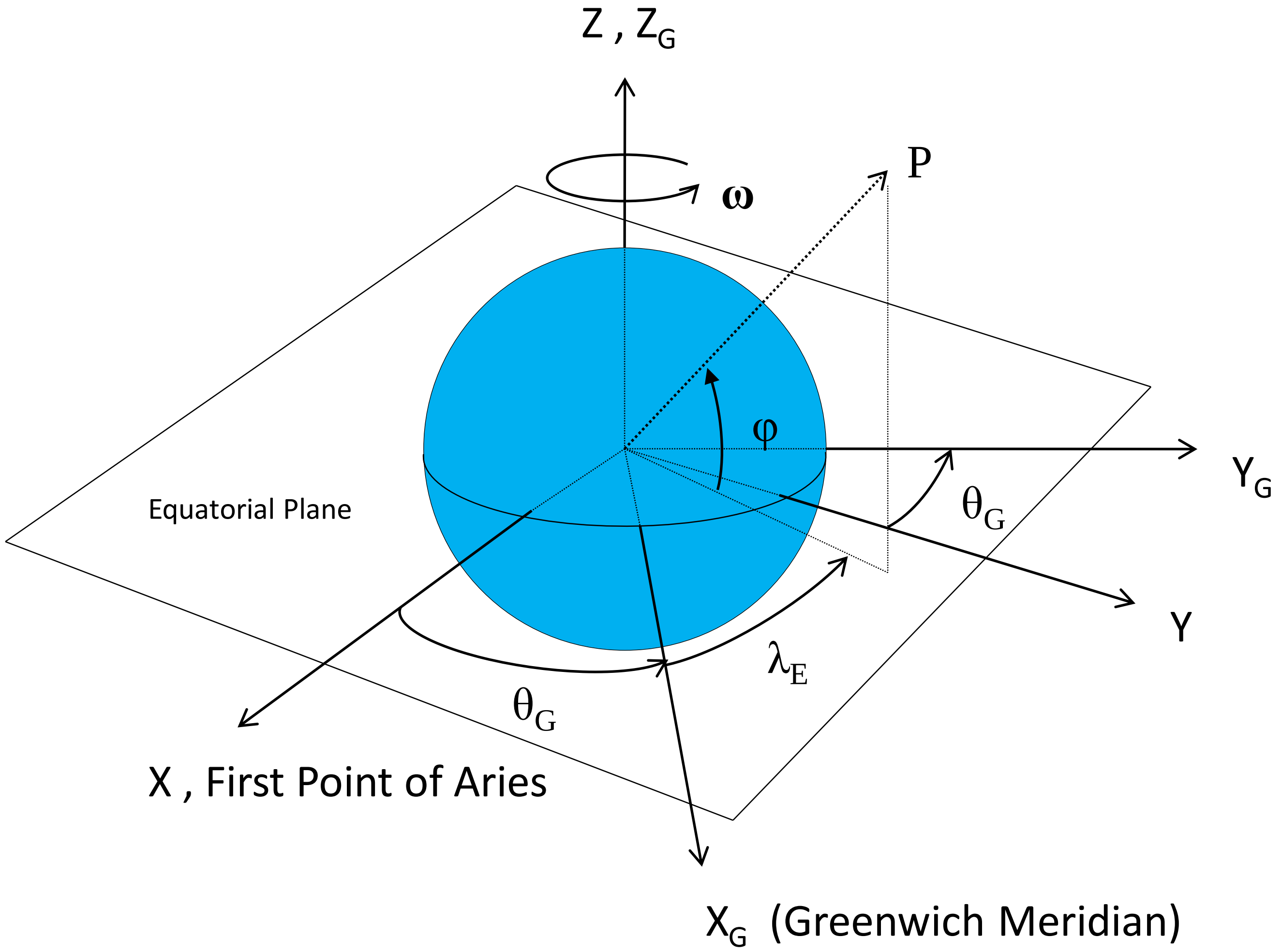


Geocentric-Equatorial Coordinate System [14]

[14] Cicci, 2010

Coordinate Frames – Earth-Centered, Earth-Fixed

- Origin at center of Earth
- X_g passes through 0° Longitude
- X_g, Y_g axes rotate with Earth, plane coincides with equator
- Z axis same as ECI
- Time matters here



Earth-Centered, Earth-Fixed Coordinate System [14]

[14] Cicci, 2010

Orbital Laws: Newton and Kepler

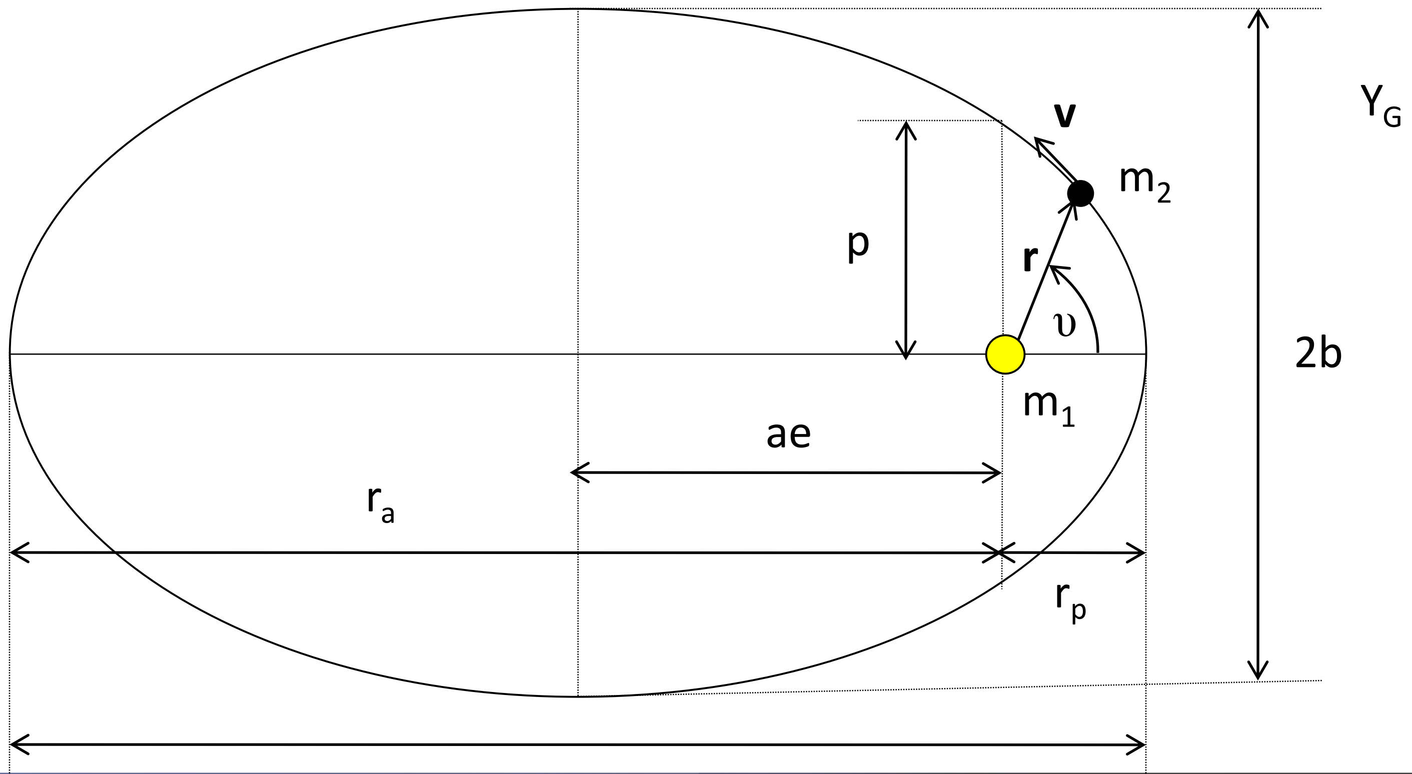
- Johan Kepler published two laws in 1609 and a third in 1619
  - Explained how planets moved, but not why they moved that way

- Isaac Newton described gravitational forces in *Principia*, 1687

Kepler's First Law: The orbit of each planet is an ellipse, with the Sun at a focus. (1609)

Newton's Law of Universal Gravitation,
$F_{12} = G\, m_1\, m_2 / r^2$

(G = Universal Gravitational Constant = 6.6742 x 10^{20} $km^3/kg\text{-}s^2$)



[14] Cicci, 2010

Orbital Laws: Newton and Kepler

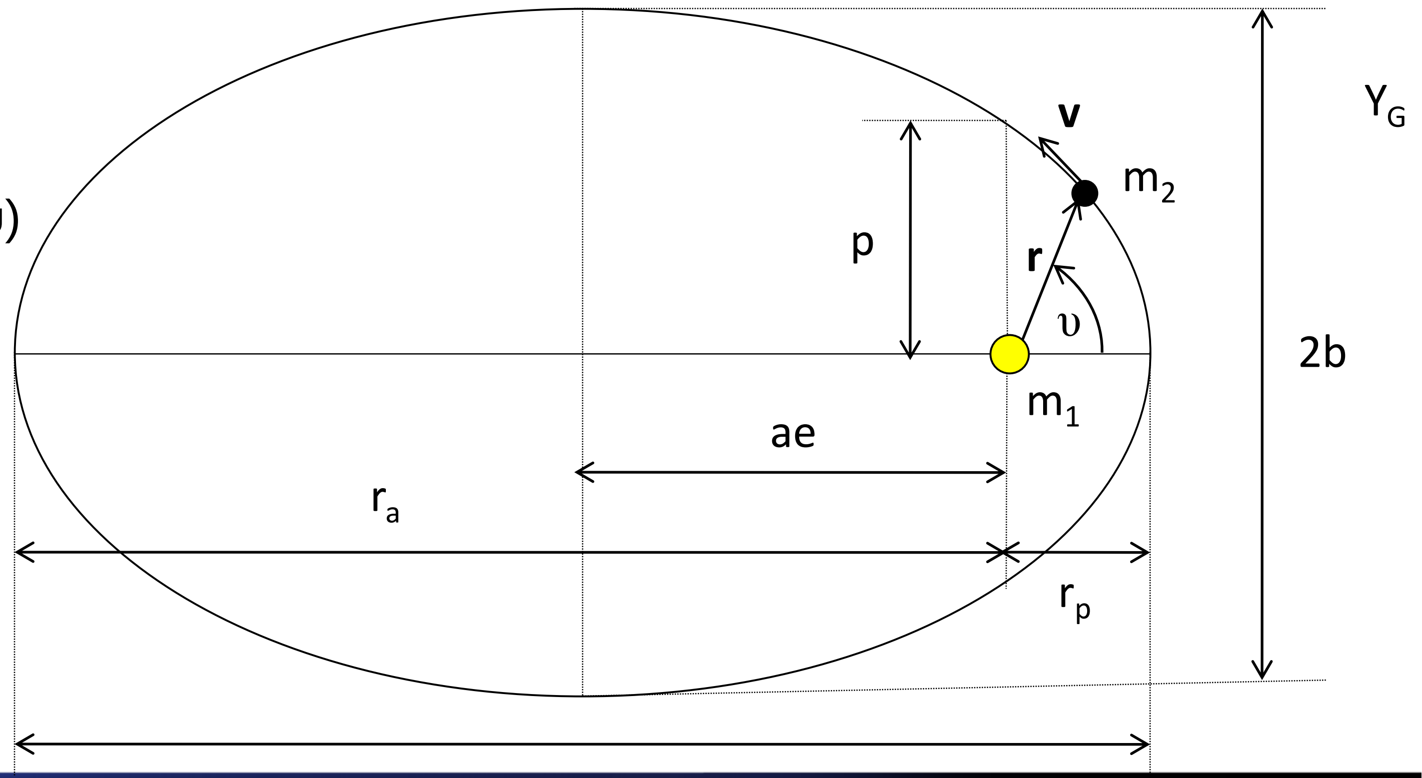
- r = distance between m_1 and m_2
- a = semi-major axis → size of ellipse
- e = eccentricity → shape of ellipse
- u = true anomaly → angular position
- p = parameter
- b = semi-minor axis

- The geometry of an ellipse yields the expression:
  - $r = p / (1 + e \cos \upsilon) = a(1 - e^2) / (1 + e \cos \upsilon)$
- All based on geometry, no dynamics
- Dynamics relationships come from Newton
- Closest Point, r_p, is perigee; farthest, r_a, is apogee

$e = (r_a - r_p) / (r_p + r_a)$ $\qquad r_p = a(1 - e)$

$p = (1 - e^2)$ $\qquad r_a = a(1 + e)$

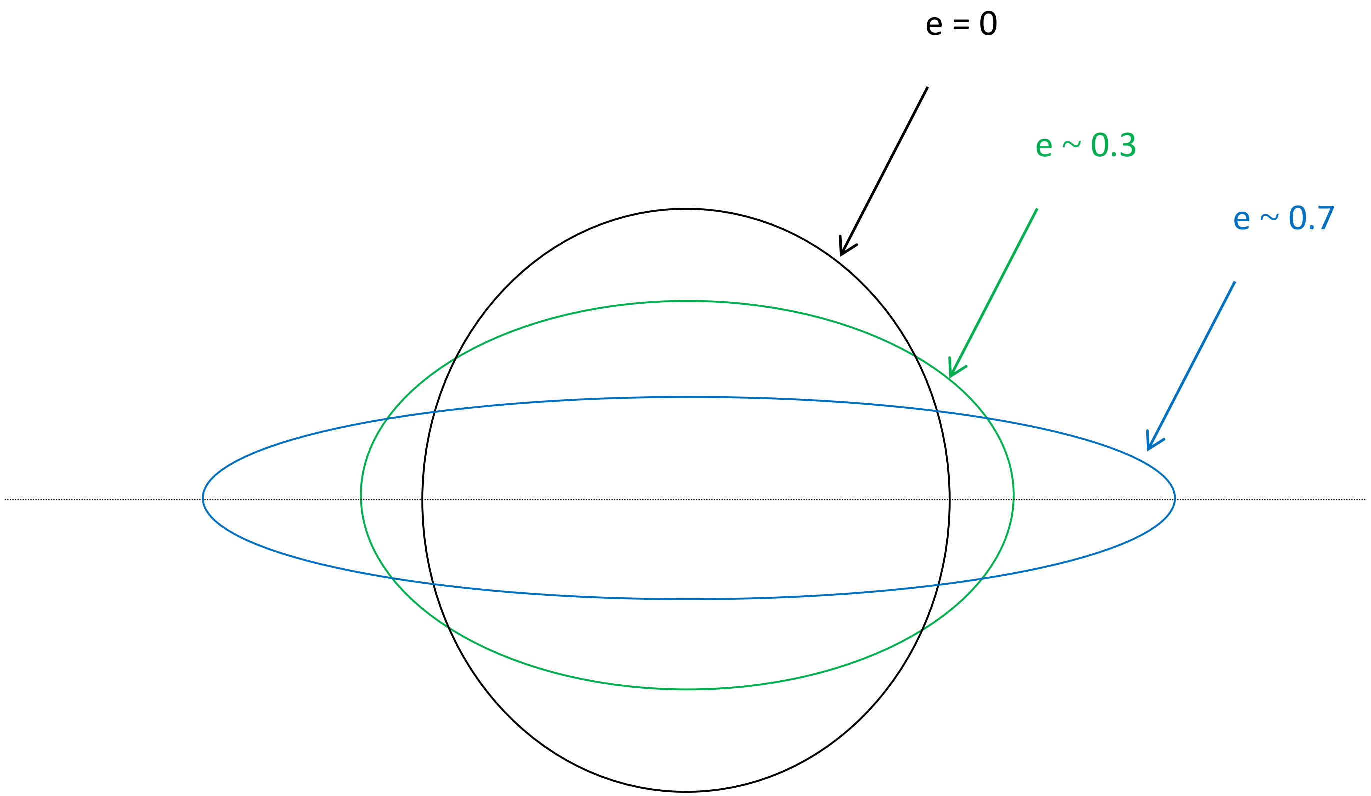
$b = (1 - e^2)^{1/2}$ $\qquad a = (r_p + r_a) / 2$



[14] Cicci, 2010

Kepler was Eccentric

- As eccentricity, e, gets larger, the ellipse becomes elongated
- Orbital period and inclination don't need to change



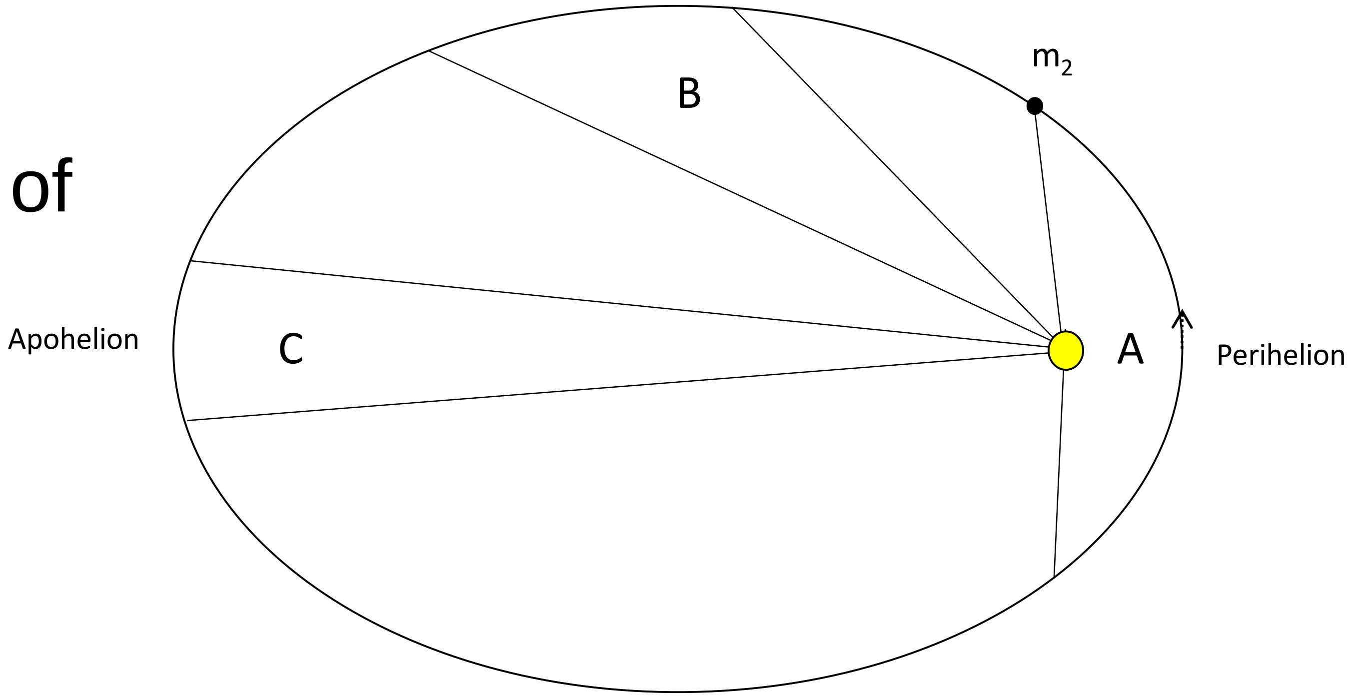
[14] Cicci, 2010

Kepler's Second & Third Laws

- Kepler's Second Law: The line joining the planet to the Sun sweeps out equal areas in equal times
  - Effectively means the orbiting body gets slower when farther away and faster when closer

- Kepler's Third Law: The square of the period of a planet is proportional to the cube of its mean distance from the Sun

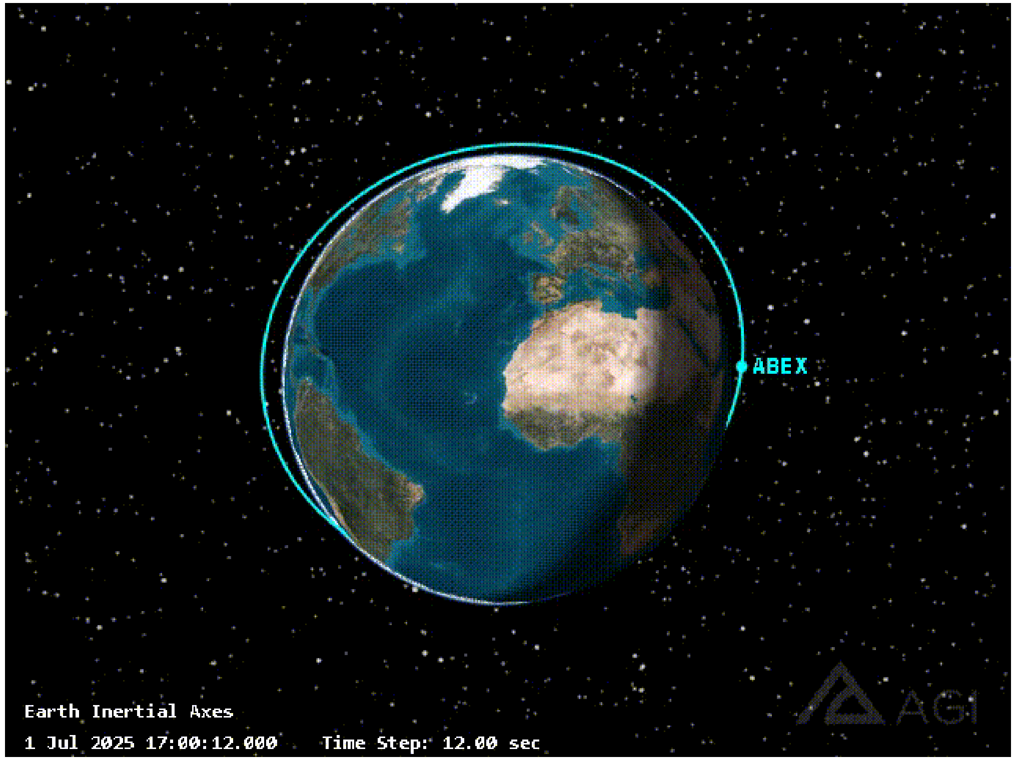
$$T^2 = (4\pi^2 / \mu)\, a^3$$

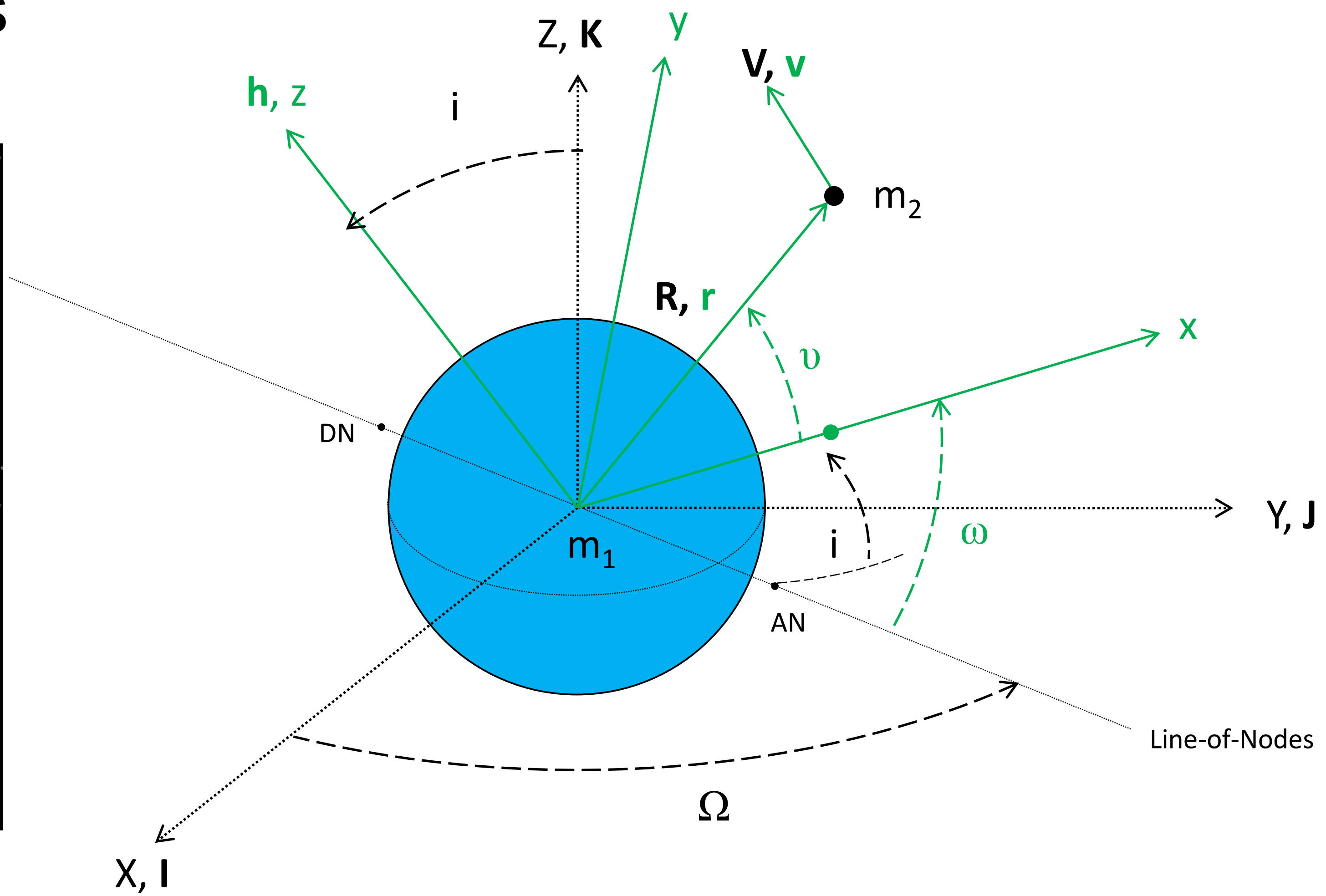


[14] Cicci, 2010

Orbits in Three Dimensions: Orbital Elements

- If you put Newton and Kepler's work together with fancy math, you get orbital elements





[14] Cicci, 2010

Types of Orbits

- $0^o \leq i < 90^o \rightarrow$ ‘direct’ orbit
- $90^o < i \leq 180^o \rightarrow$ ‘retrograde’ orbit (east → west)
- $i = 90^o \rightarrow$ ‘polar’ orbit (over North and South poles)
- $i = 0^o \rightarrow$ ‘equatorial’ (direct) orbit
- $i = 180^o \rightarrow$ ‘equatorial’ (retrograde) orbit

- Geosynchronous orbit = an orbit with a period equal to the rotation rate of the Earth = 23 h, 56 m, 4.09 s

- Geostationary orbit = a geosynchronous orbit having: $i = 0^o$

[14] Cicci, 2010

Types of Orbits

- Low Earth Orbit (LEO): ~ 300-1,500 km alt
- Medium Earth Orbit (MEO): ~1,500-35,800 km alt

- Geosynchronous / Geostationary Orbits (GEO):
  - 35,800 km alt, circular (orbital period = sidereal day)

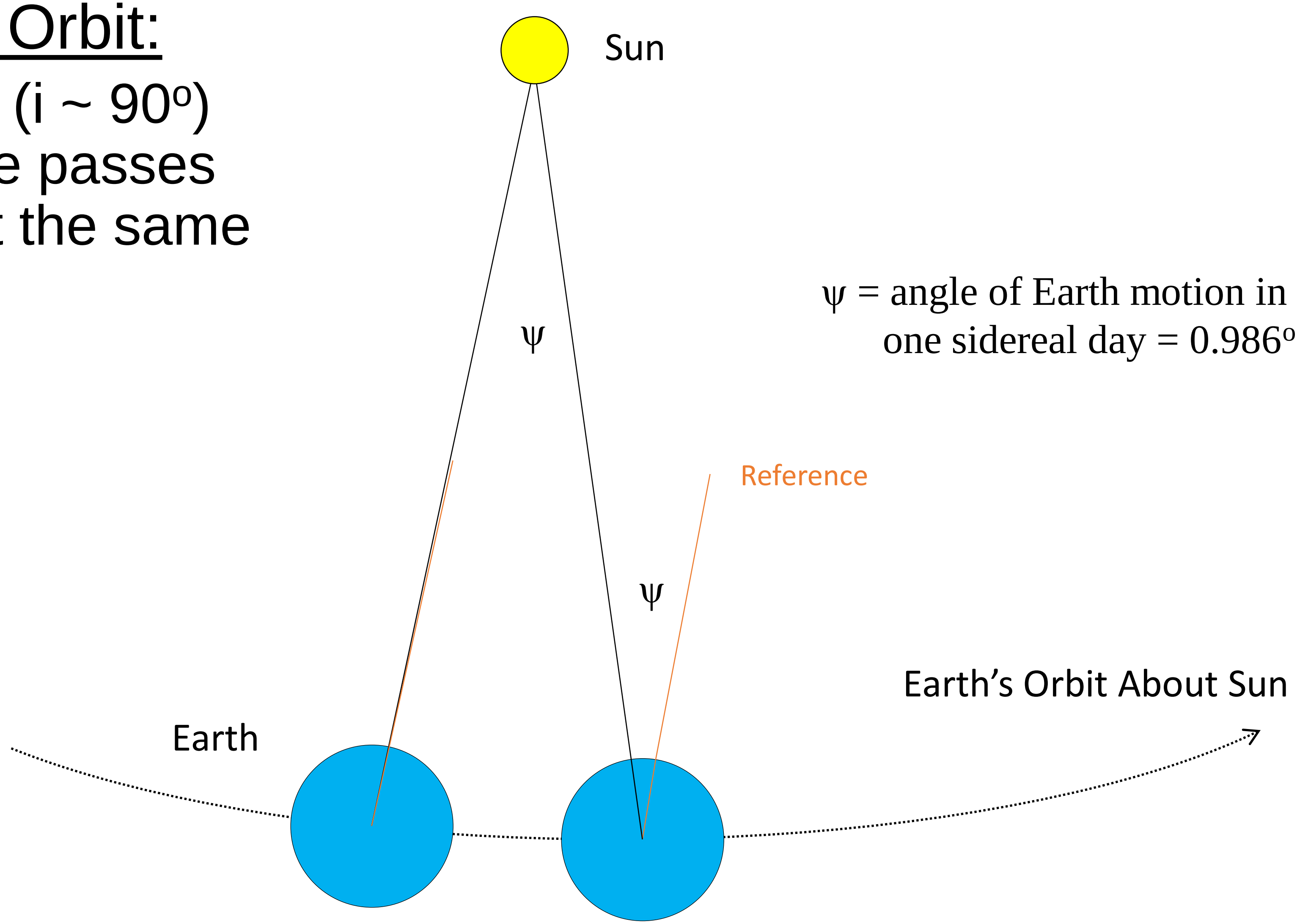
- Highly Elliptical Orbit (HEO):
  - perigee <1,000 km alt; apogee ~15,000-40,000 km alt

[14] Cicci, 2010

Types of Orbits

- <u>Sun-Synchronous Orbit:</u>
  - Nearly polar orbit (i ~ 90^o) where the satellite passes over the planet at the same mean solar time.

- Mean Solar Day:

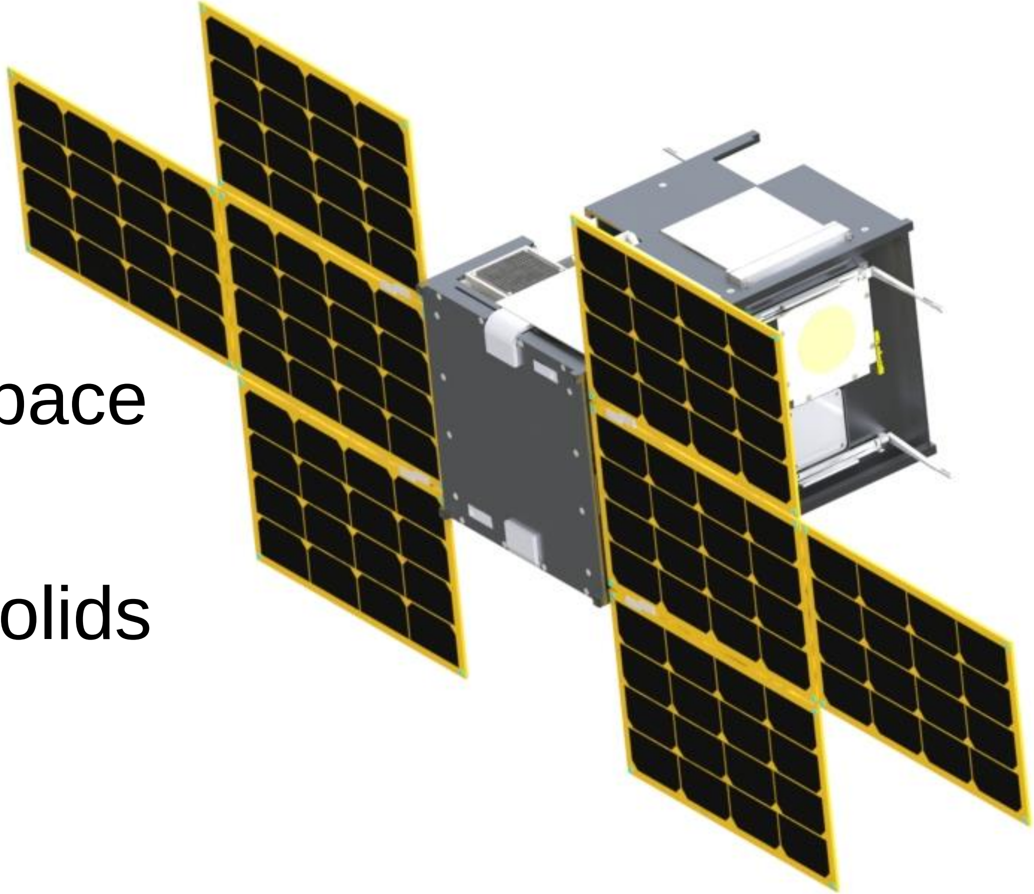


[14] Cicci, 2010

Structural Integrity

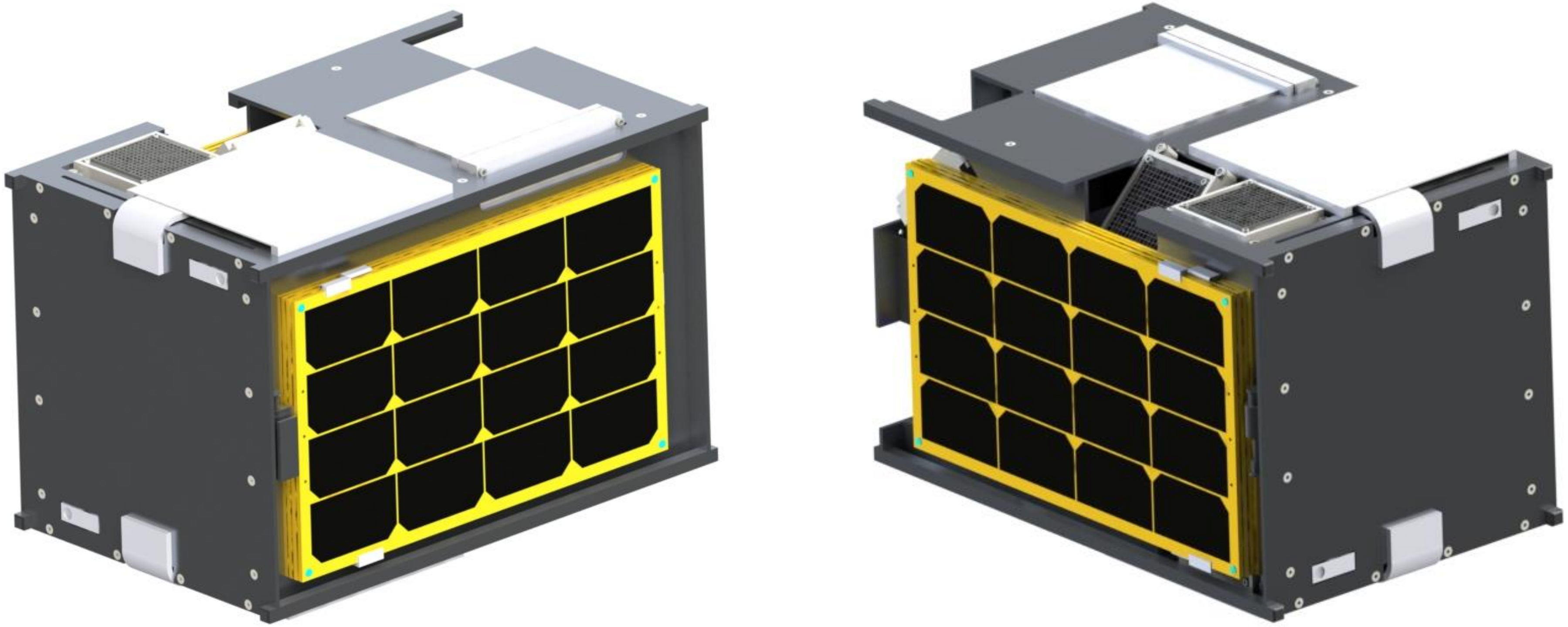
That’s Pretty Metal

- Spacecraft undergo three primary types of stresses
  - Static stresses
  - Dynamic stresses
  - Thermal stresses
- Metals are fairly well suited for space
- In a vacuum, gas comes out of solids
  - Called outgassing



How to Pick Your Pretty Metal

- Select structural materials for:
  - ability to deal with stress (various meanings)
  - thermal expansion coefficient (both metals and fasteners)
  - Outgassing considerations
  - Cost
  - The ‘ilities
    - Availability
    - Machinabilty
    - Reliability



Stresses: Where do they Come From?

- Different structural life cycle segments have different stresses
  - Static stresses are applied once (resultant vector, three components)
    - Directions matter
  - Dynamic stresses are the Hz multiplied by duration
    - Multiple frequencies may be present, can define an equivalent value
  - Thermal Stresses are a function of sunlight vs. eclipse
    - On the ground it's mostly day/night unless climate-controlled
    - ABEX has ~16 orbits per day, assuming eclipses present half the time here

Phase	Thermal Stress Cycles Per Year	Static Stress Cycles Per Year	Dynamic Stress Cycles Per Year
Terrestrial Transport	0	1	1350000
VAB Storage	0	1	0
Pre-Launch	1	1	45000
Dynamic Ascent	1	1	360000
Static Ascent	1	1	1620000
Launch Vehicle Coast	1*16 = 16	0	0
CubeSat Deployment	0.25*16 = 4	1	0
Mission Phase 1: LEOP	7*16 = 112	0	0
Mission Phase 2: Commissioning	1*30*16 = 480	0	0
Mission Phase 3: Operations	11*30*16 = 5280	0	0
Mission Phase 4: Disposal	7*16 = 112	0	0

Stresses: Where do they Come From?

- Static Stresses = LV Thrust-Induced Stress, Deployment Stress
- Dynamic Stresses = LV Vibration-Induced Stress, Acoustic Stress, Solar Array Deployment Stress
  - Interactions with s/c Fundamental Vibrational Mode cause resonance
- Thermal Stresses = Heating Up & Cooling Down
  - Occurs both in fasteners and in structural elements

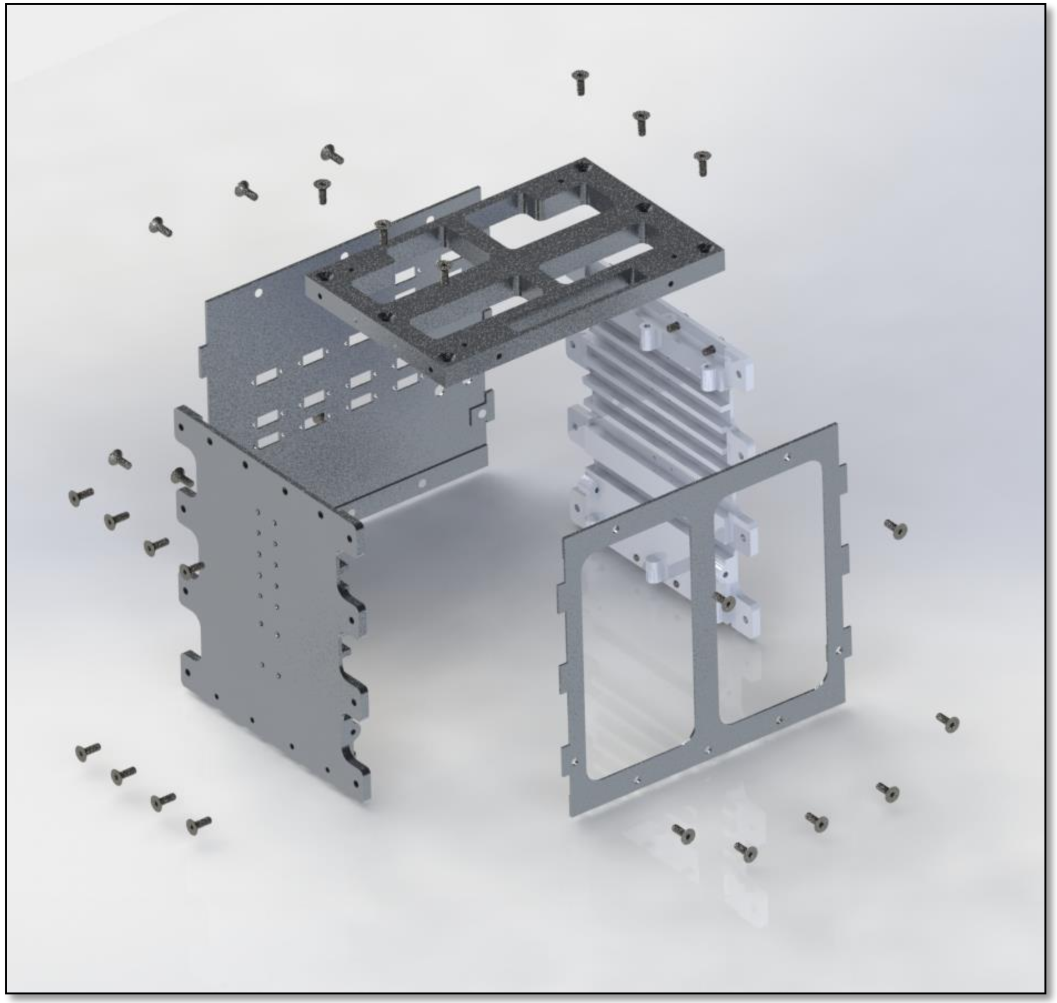
Phase	Thermal Stress Cycles Per Year	Static Stress Cycles Per Year	Dynamic Stress Cycles Per Year
Terrestrial Transport	0	1	1350000
VAB Storage	0	1	0
Pre-Launch	1	1	45000
Dynamic Ascent	1	1	360000
Static Ascent	1	1	1620000
Launch Vehicle Coast	1*16 = 16	0	0
CubeSat Deployment	0.25*16 = 4	1	0
Mission Phase 1: LEOP	7*16 = 112	0	1000
Mission Phase 2: Commissioning	1*30*16 = 480	0	0
Mission Phase 3: Operations	11*30*16 = 5280	0	0
Mission Phase 4: Disposal	7*16 = 112	0	0

Structural Technical Performance Measures

TPM	Rationale	Units
Available External Surface Area	How much surface area is taken up by external components directly affects mounting decisions for other teams.	cm^2
Rotational Inertia Matrix	Rotational Inertia is an object’s resistance to rotation. It is a function of mass and distance to the spacecraft center of gravity and is provided to the GN&C team.	$kg{\cdot}m^2$
Metallic Structure Minimum Factor of Safety	Structures undergo static, thermal, and dynamic stresses in space. All metallic structures must be analyzed for their structural factor of safety. This TPM is how the structure is deemed not to fail due to non-fatigue related failure modes.	-
Metallic Structure Minimum Fatigue Life	The metallic structure’s anticipated fatigue life must be greater than that of the mission duration to ensure the spacecraft will not fail due to fatigue. NASA requires the fatigue life be 4x greater.	Years
Solar Array Stowed Natural Frequency	Solar arrays experience severe vibration when stowed (not deployed) if their natural frequency is within an octave of the launch vehicle frequencies. Solar array linear stiffness, dimensions, and constraints determine the natural frequency.	Hz
Solar Array Hinge Deployment Factor of Safety	When solar arrays are deployed via spring-loaded hinge, their hinge backstop experiences a significant structural load. The Factor of Safety must be analyzed to ensure the hinge won’t break upon deployment	-

Structural Materials

- For the body, many materials are options:
  - Al 6061-T6, Al 7075-T73 are common
- Metal fasteners are generally used
  - SS 316, SS A286 are common
- Polymer Matrix Composites outgas
  - Why carbon fiber isn't used much
- Metal Matrix Composites are expensive
  - Ceramic MC's too



[15] Halvorson, 2020

Stress Example: Static

- Artemis I Launch Vehicle:
  - Thrust = 37,365,061 N
  - Total Mass w/Fuel = 2,608,156 kg
  - Total Mass w/o Fuel = 1,247,379 kg

- Use Newton's second law on LV, then apply acceleration to CS
  - LV acceleration?
  - Force on CubeSat?

$$F = m \cdot a$$

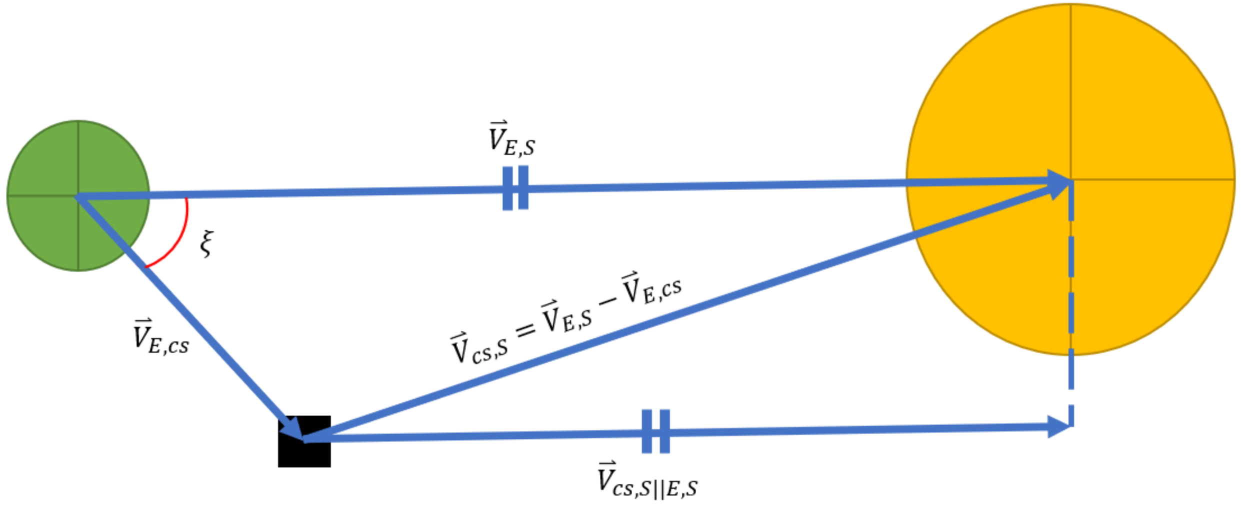
- Accelerations generally put into Finite Element Analysis, not forces
  - Dynamic and Thermal best left to FEA
  - Can be done by hand, but it's longer than an example

Thermal Control

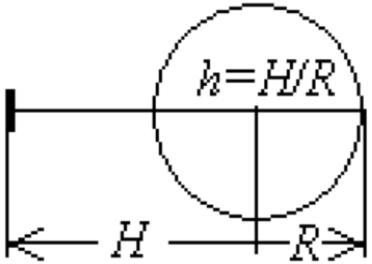
Topics in Spacecraft Thermal Engineering

The purpose of Thermal Spacecraft Engineering is to determine the necessary radiator area and heater wattage required to maintain a nominal energy balance; **the purpose is not to predict temperatures**.

- Important Concepts
  - Heat Source Determination
  - Heat Flux Estimation
  - Absorbed Heat Determination
  - Thermal Control Strategies



- Note: ABEX includes particle radiation in this team



Thermal Technical Performance Measures

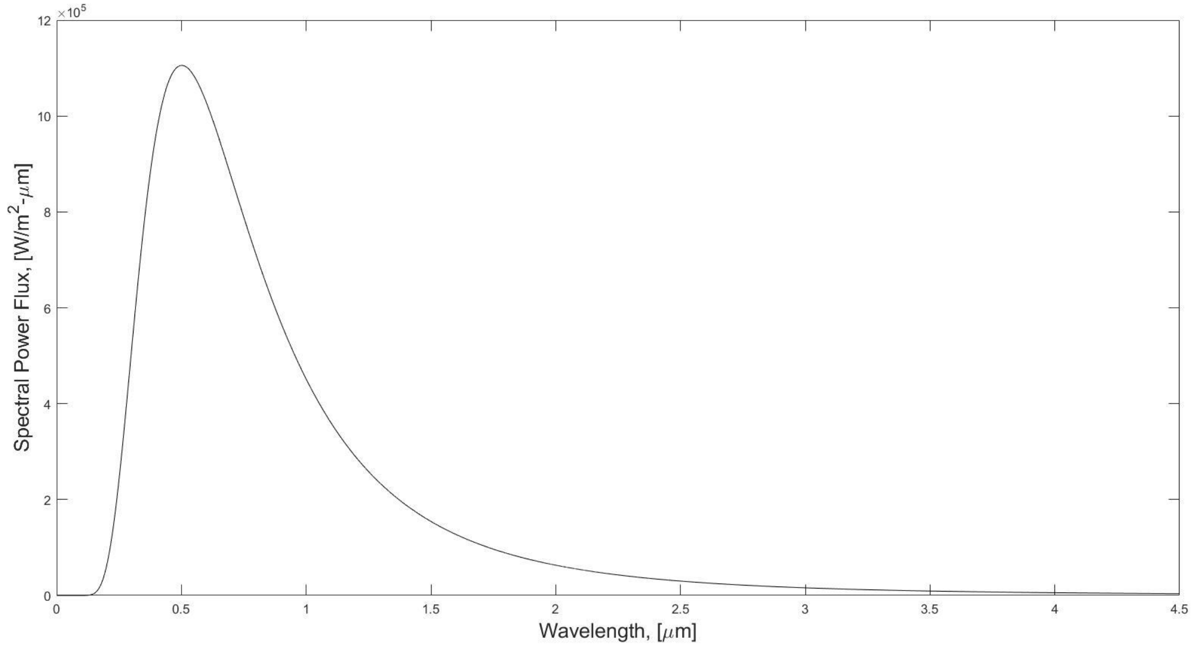
TPM	Rationale	Units
Minimum Thermal Condition	Minimum total heat absorbed at any point during the spacecraft lifespan.	W
Maximum Thermal Condition	Maximum total heat absorbed at any point during the spacecraft lifespan.	W
Radiation Shield Areal Density	For either the material in the structural chassis or material in a dedicated radiation shield, it is the shield mass per unit area of a defined shield structure.	g/cm^2
Solar Array Coverglass Mass	For the coverglass protecting the Coverglass-Interconnected Cells on the solar arrays, this is the mass of the coverglass required to prevent 10% solar array efficiency degradation within a year.	m
Heater Wattage	Wattage provided to heaters to maintain components within operational temperature conditions at End of Life.	W
Radiator Area	Area required of the radiator to maintain components within operational temperature conditions at End of Life.	m^2

Electromagnetic Radiation is Difficult

Parameter	Variable	Unit
Spectral Emissive Power	$E_{\lambda b}$	W/m^2-µm
Planck's Constant	h_p	J-s
Speed of Light in a Vacuum	c_0	m/s
Refractive Index	n	-
Boltzmann's Constant	k_B	J/K
Wavelength	λ	µm
Temperature	T	K
Emissivity of a Blackbody	ε_b	-
Stefan-Boltzmann Constant	σ	W/m^2-K^4
Heat Flux from the Surface of the Sun	$Q''_{Solar,Surf}$	W/m^2

Spectral Emissive Power of a Hemispherical Blackbody

$$E_{\lambda b} = \frac{2 \cdot \pi \cdot h_p \cdot c_0^2}{n^2 \cdot \lambda^5 \cdot \left[\exp\left(\frac{h_p \cdot c_0}{n \cdot k_B \cdot \lambda \cdot T}\right) - 1\right]} = \left[\frac{W}{m^2 - \mu m}\right]$$



[15] Halvorson, 2020

Electromagnetic Radiation is Difficult

$$T_{sun} = 5780\,[K] \quad \sigma = 5.67 \cdot 10^{-8} \left[\frac{W}{m^2 - K^4}\right] \quad \varepsilon_b = 1\ [\text{-}]$$

$$f_s = 0\ [\text{-}]$$

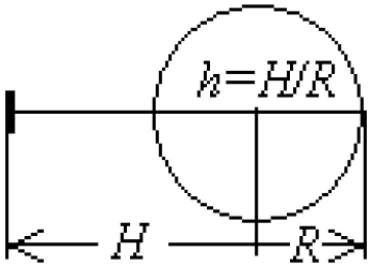
$$Q''_{Sun,Surf} = \varepsilon_b \cdot \sigma \cdot {T_{sun}}^4 = \sim 63{,}300{,}000 \left[\frac{W}{m^2}\right]$$

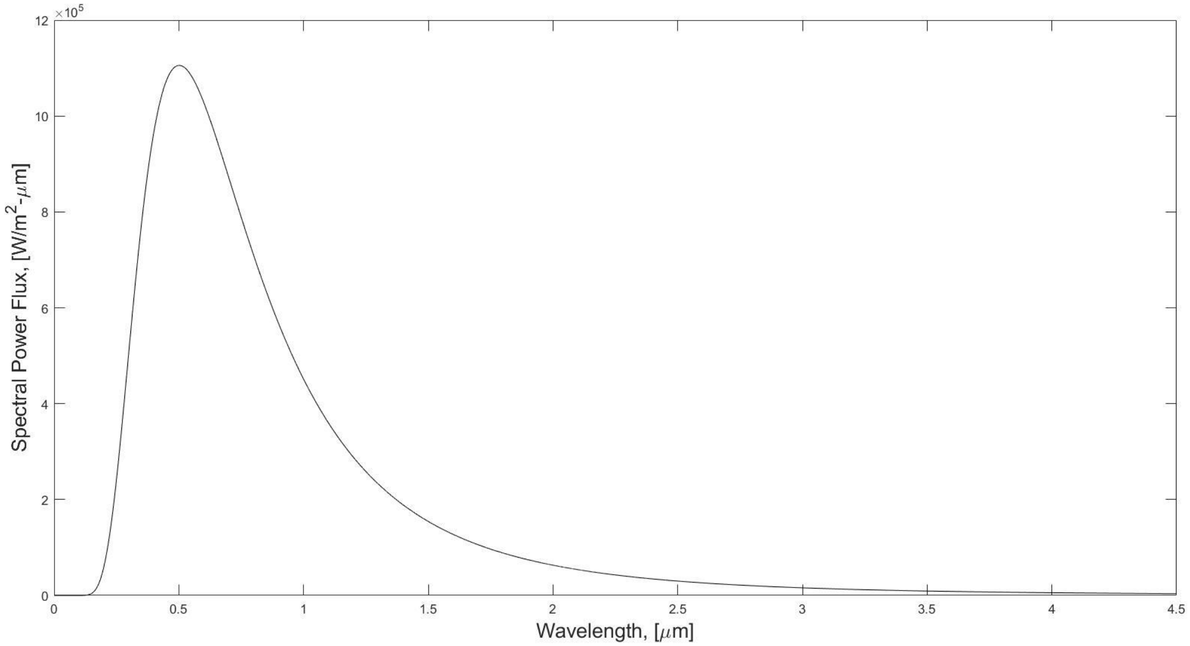
Spectral Emissive Power of a Hemispherical Blackbody

$$E_{\lambda b} = \frac{2 \cdot \pi \cdot h_p \cdot c_0^2}{n^2 \cdot \lambda^5 \cdot \left[\exp\left(\frac{h_p \cdot c_0}{n \cdot k_B \cdot \lambda \cdot T}\right) - 1\right]} = \left[\frac{W}{m^2 - \mu m}\right]$$

Radiative View Factor

$$F_{rad} = \frac{1}{\left(\frac{H}{R}\right)^2}$$





$$Q''_{Sun,Ems} = Q''_{Sun,Surf} \cdot F_{rad} \cdot (1 - f_s) = \frac{Q''_{Sun,Surf}}{\left(\frac{D_{Sun} + R_{Sun}}{R_{Sun}}\right)^2} \cdot (1 - f_s)$$

$$Q''_{Sun,Ems} = \frac{63{,}300{,}000 \frac{W}{m^2}}{\left(\frac{1.496 \cdot 10^{11}\ m}{6.95 \cdot 10^{8}\ m}\right)^2} \cdot (1 - 0) = \sim 1{,}366 \frac{W}{m^2}$$

Heat Sources In Space

• Solar Emission	(Electromagnetic Radiation)
• Earth Emission	(Electromagnetic Radiation)
• Earth Albedo	(Electromagnetic Radiation)
• Free Molecular Heating	(Physical Interactions with Atmosphere)
• Charged Particle Heating	(Corpuscular/Particle Radiation)
• Operational Heating	(Power Conversion Efficiency Losses)
• Intentional Heating	(Heater Power)

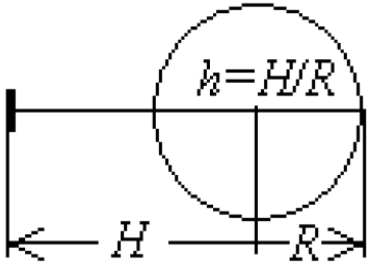
- Doing an example, assuming $Q''_{Sun,Ems} = \frac{63{,}300{,}000 \frac{W}{m^2}}{\left(\frac{1.496 \cdot 10^{11}\ m}{6.95 \cdot 10^{8}\ m}\right)^2} = {\sim}1{,}366 \frac{W}{m^2}$

[16] Halvorson et al, 2021

Example: Thermal Energy Balance

- Solar Emission
  - $Q''_{Sun,Ems}$ =1366 W/m^2
- Earth Emission
  - $Q''_{Earth,Surf} = \varepsilon_b \cdot \sigma \cdot T_{Earth}{}^4$
  - $Q''_{Earth,Ems} = Q''_{Earth,Surf} \cdot F_{rad} \cdot (1 - f_s) = \dfrac{Q''_{Sun,Earth}}{\left(\frac{D_{Orbit}+R_{Earth}}{R_{Earth}}\right)^2} \cdot (1 - f_s)$

$$F_{rad} = \frac{1}{\left(\frac{H}{R}\right)^2}$$

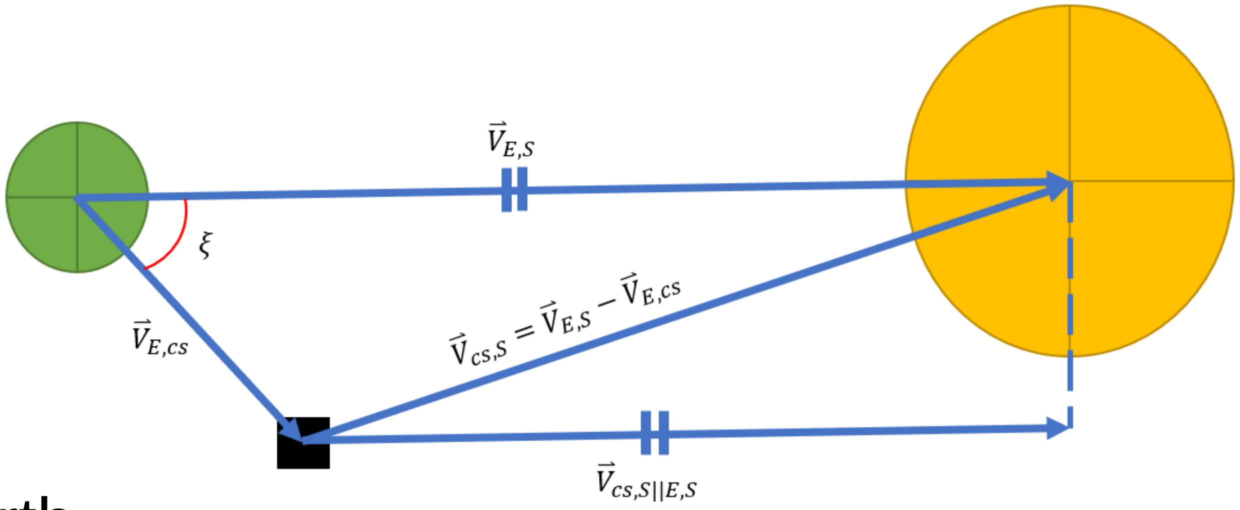


  - Assume T_{Earth} = 300 K, R_{Earth} = 6,378 km, D_{Orbit} = 600 km
  - Assume Shadow Fraction, f_s = 0

[16] Halvorson et al, 2021 [17] Martinez, 2021

Example: Thermal Energy Balance

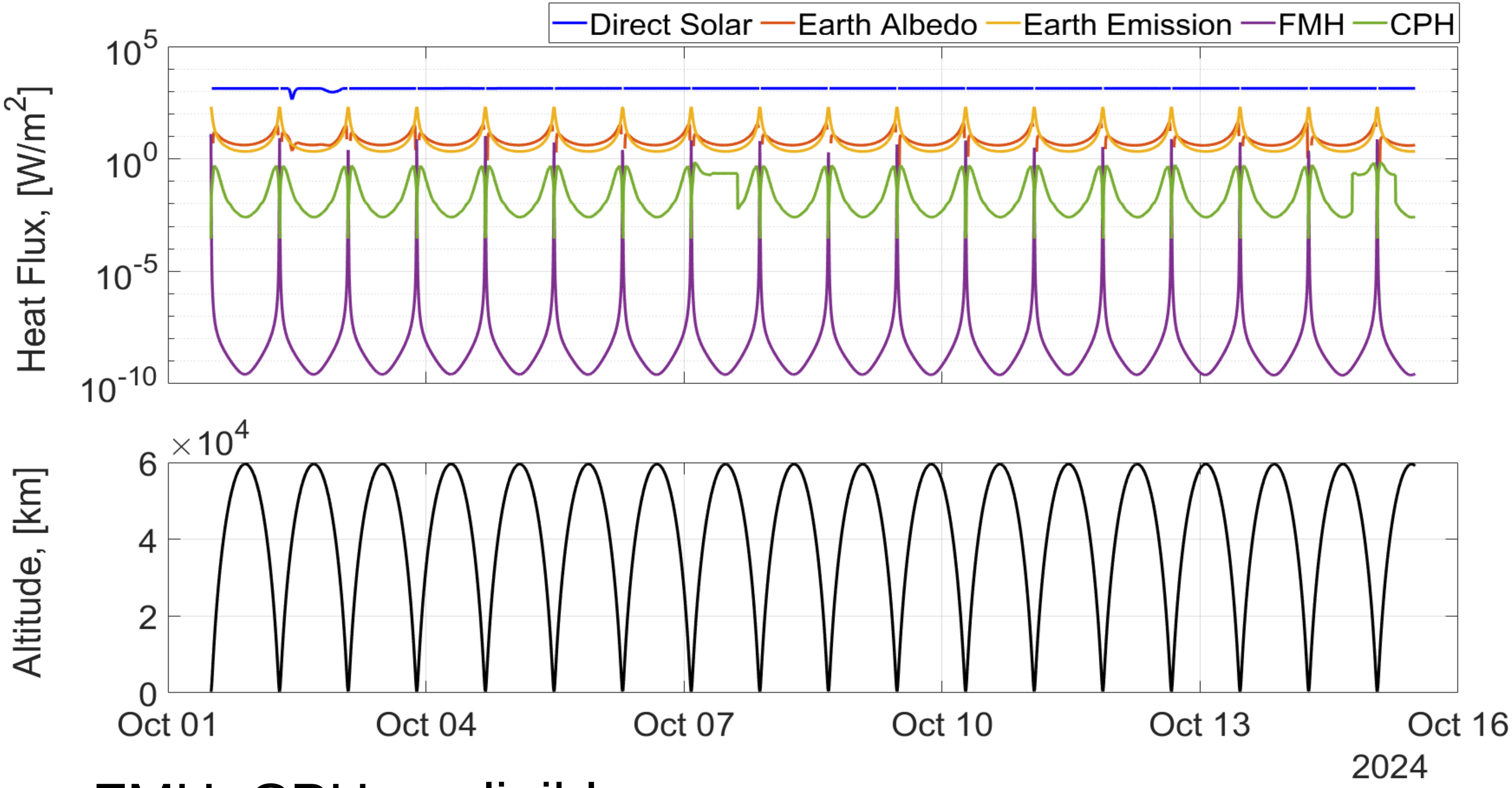
- Solar Emission
  - 1366 W/m^2
- Earth Emission
  - 384 W/m^2
- Earth Albedo
  - aF = 0.33 on average for Earth
  - Assume T_{Earth} = 300 K, R_{Earth} = 6,378 km, D_{Orbit} = 600 km
  - Assume Shadow Fraction, f_s = 0
  - Assume ξ = 30 deg



$$Q"_{Earth,Alb} = \frac{Q"_{Sun}}{\left(\frac{D_{Orbit} + R_{Earth}}{R_{Earth}}\right)^2} \cdot aF \cdot \cos(\xi) \cdot (1 - f_s)$$

[16] Halvorson et al, 2021

Relative Contributions of Heat Sources



- Assume FMH, CPH negligible

[16] Halvorson et al, 2021

Example: Thermal Energy Balance

$$Q_{in} - Q_{out} + Q_{gen} = Q_{Stored}$$

$$Q = Q'' \cdot A$$

$$Q_{in} = Q''_{Sun,Ems} \cdot A_{Sun} + Q''_{Earth,Ems} \cdot A_{Earth} + Q''_{Earth,Alb} \cdot A_{Earth}$$

$$Q_{gen} = Q_{ops} \cdot (1 - \eta_{PCE}) + Q_{Heat}$$

$$Q_{out} = A_{SC} \cdot \varepsilon_{SC} \cdot \sigma \cdot T_{SC}{}^4$$

- Solar Emission
  - 1366 W/m^2
- Earth Emission
  - 384 W/m^2
- Earth Albedo
  - 326 W/m^2
- Assume FMH, CPH negligible
- Assume $A_{SC} = (2 * 0.226\,m * 0.226\,m) + (4 * 0.226\,m * 0.366\,m)$
- Assume $A_{Sun} = 0.226\,m * 0.226\,m$, $A_{Earth} = 0.226\,m * 0.366\,m$
- Assume $\varepsilon_{SC} = 0.6, Q_{ops} = 42\,W, \eta_{PCE} = 0.25$
- $What\ is\ the\ expression?$

Example: Thermal Energy Balance

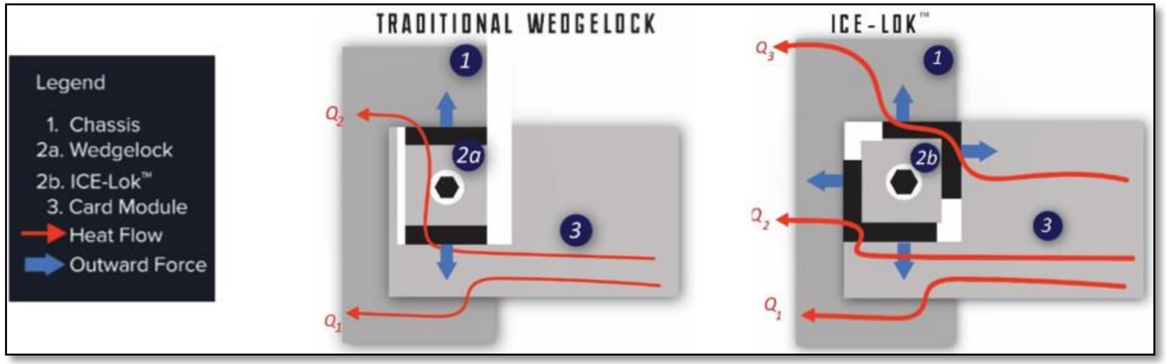
$$Q_{in} - Q_{out} + Q_{gen} = Q_{Stored}$$

$$(Q''_{Sun,Ems} \cdot A_{Sun} + Q''_{Earth,Ems} \cdot A_{Earth} + Q''_{Earth,Alb} \cdot A_{Earth}) - [(A_{SC} + A_{rad}) \cdot \varepsilon_{SC} \cdot \sigma \cdot {T_{SC}}^{4}] + [Q_{ops} \cdot (1 - \eta_{PCE}) + Q_{Heat}] = 0$$

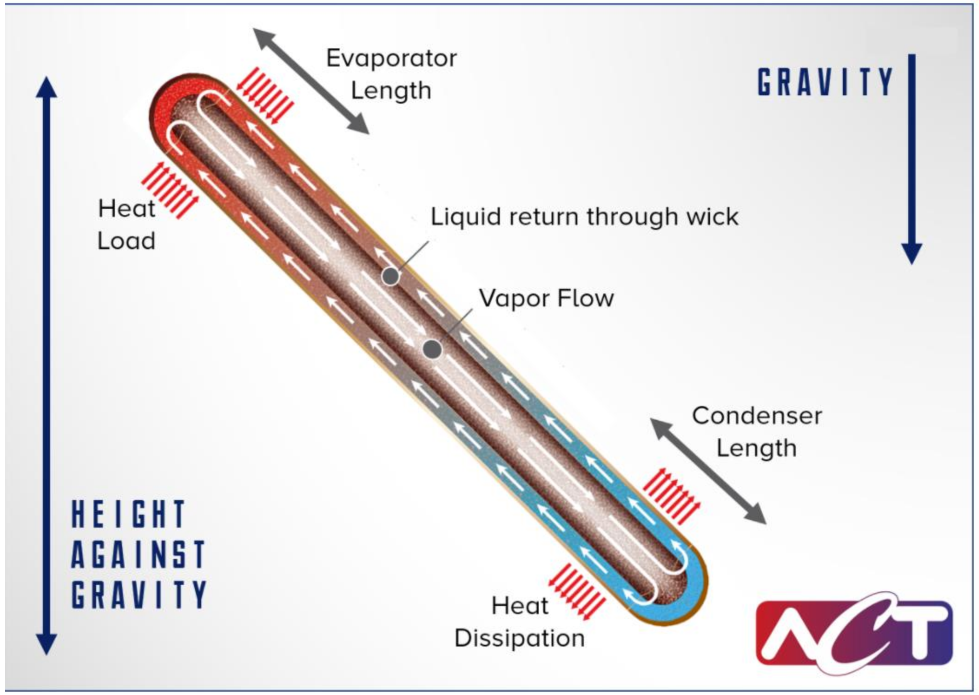
- Hot State: Assume T_{SC} = 313 K, Q_{Heat}= 0 W, solve for A_{rad}
- Cold State: Assume T_{SC} = 263 K, A_{rad}= 0 m^2, solve for Q_{Heat}

Thermal Control Components

- Heat Pipes
- Radiators
- Radiators with Heat Pipes
- Thermal Paints
- Thermal Straps
- Heat Switches
- Electronics Thermal Management
  - Wedge-Locks
  - Hi-K cards



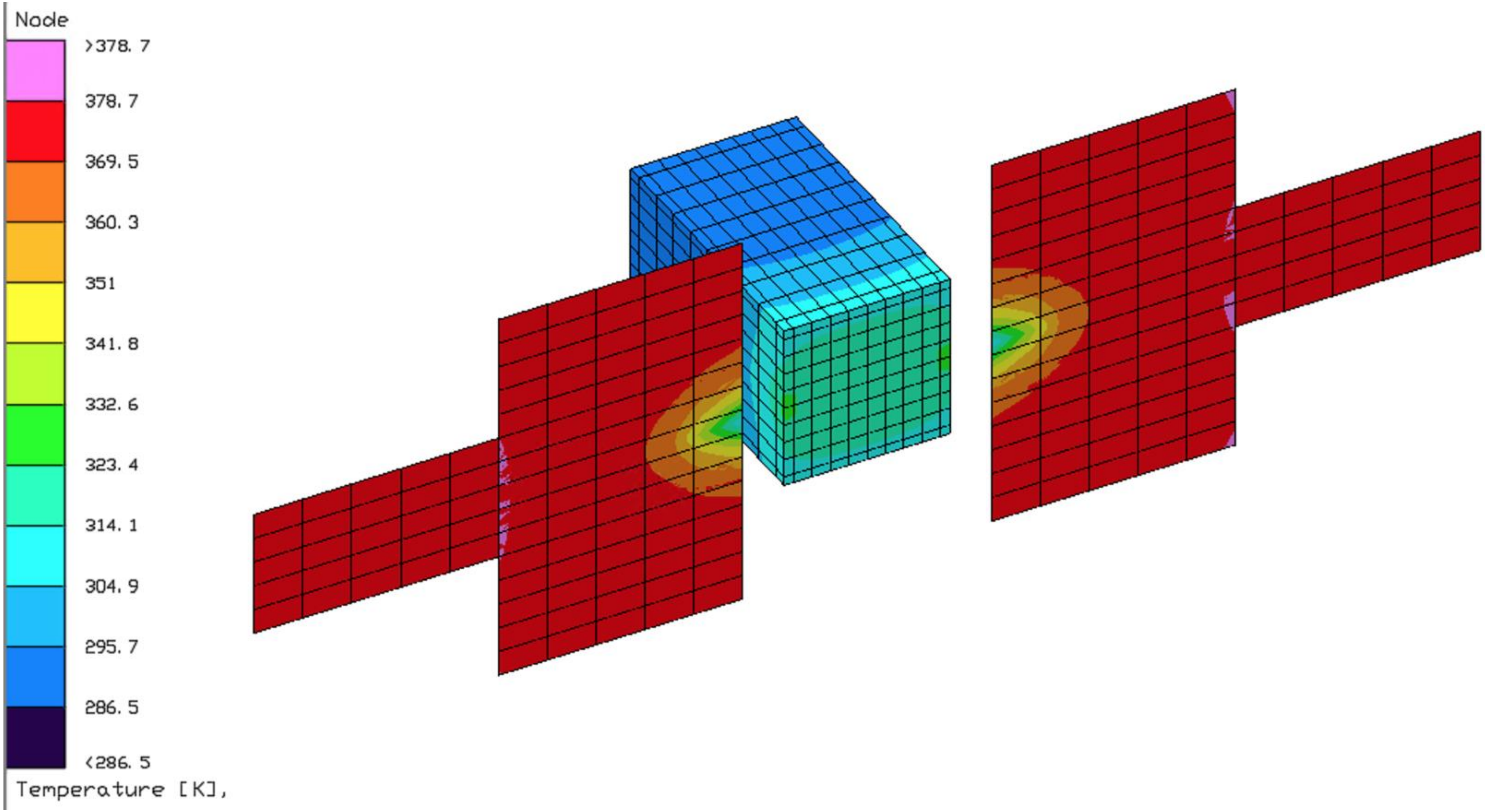
ACT ICE-LOK [18]



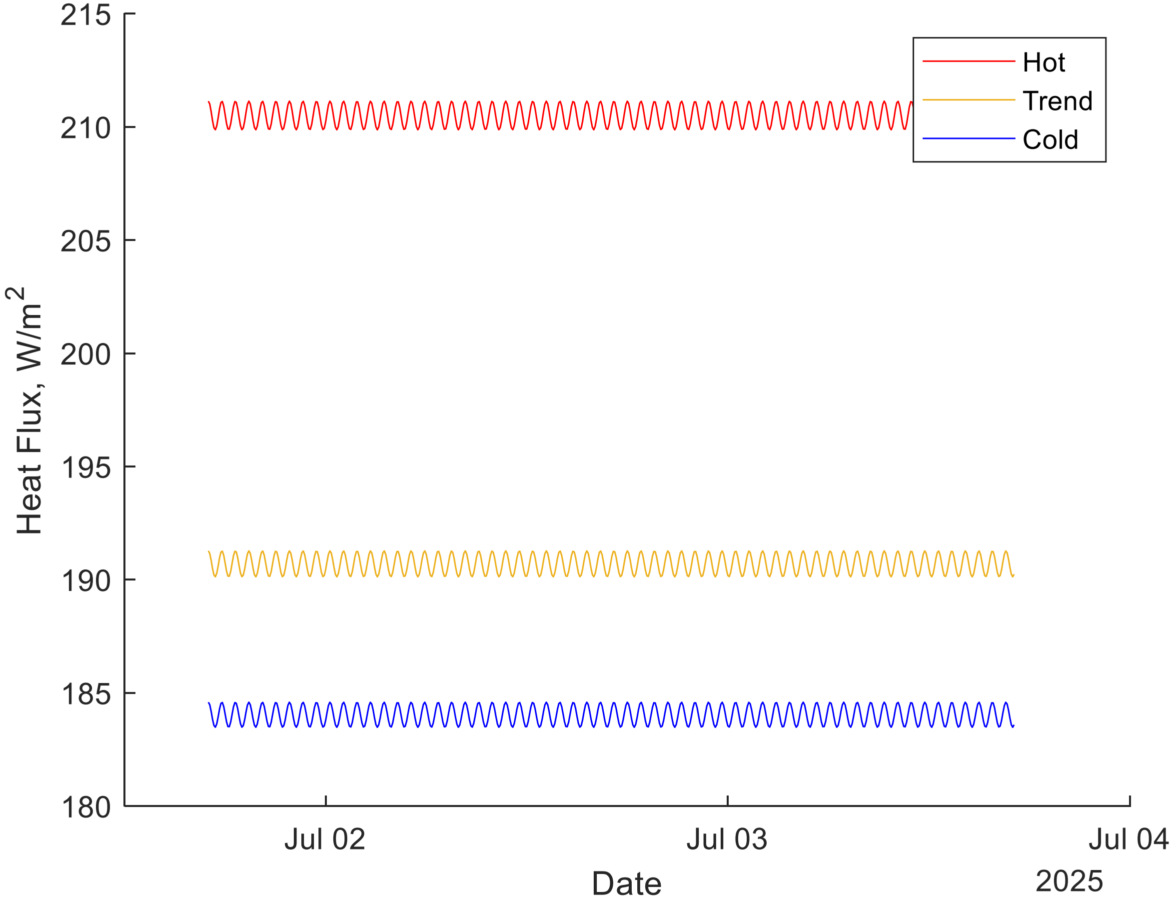
Heat Pipe Overview [18]

[18] ACT, 2022

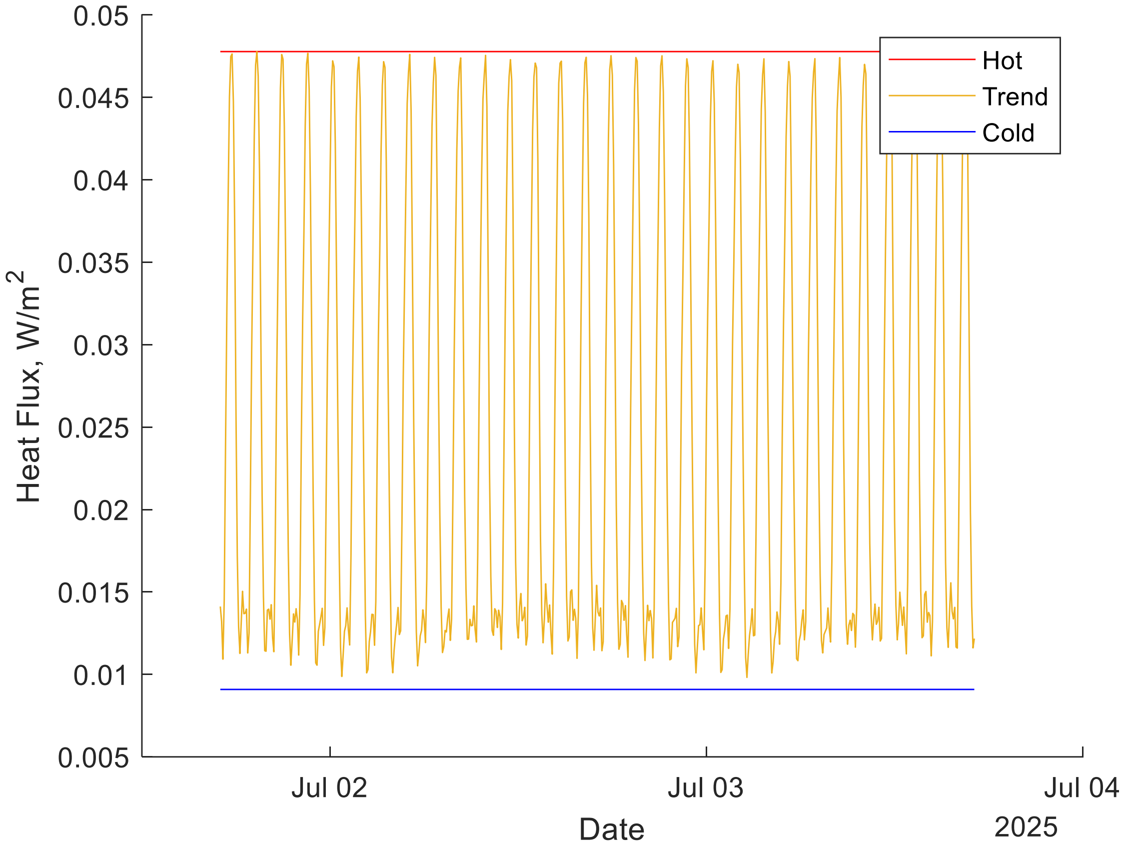
Thermal Analysis Techniques



Thermal Analysis Techniques



Incident Earth Emission Heat Flux, All TES's, Two Days, July 1, 2025 – July 3, 2025



Incident Free Molecular Heat Flux, All TES's, Two Days, July 1, 2025 – July 3, 2025

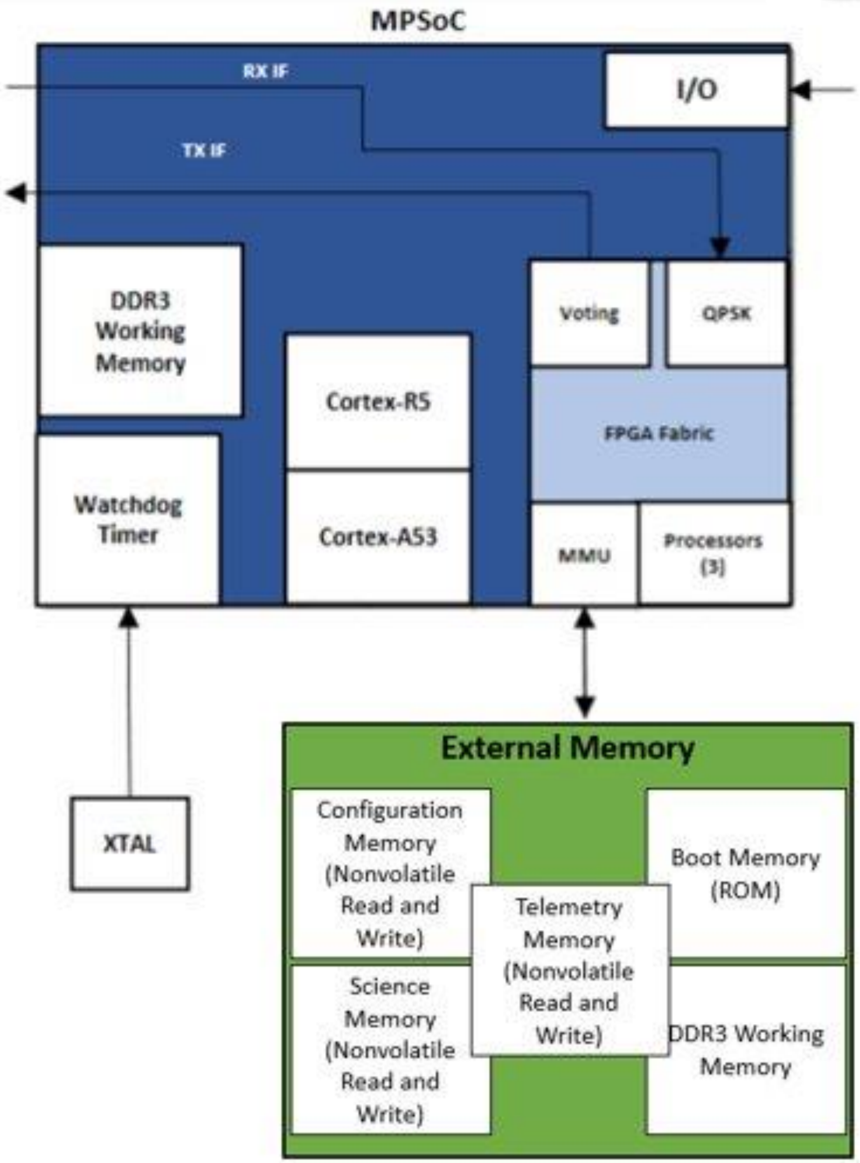
Command & Data Handling

Commanding the Spacecraft

- The C&DH Subsystem is the brains of the spacecraft
  - Connection to all components
  - Tells components what to do
  - Allows power to flow to components
  - Generally exists within an "Avionics Stack"
- Main component is called the On-Board Computer (OBC)
- Interface boards might be necessary to provide adequate space for peripheral connections
  - ABEX has Auxiliary Interface Board (AIB)

Commanding the Spacecraft

- Parts of a Spacecraft Computer
  - CPU
  - Memory
    - Working (RAM, Volatile)
    - Boot (Special, Non-Volatile)
    - Configuration (Non-Volatile)
    - Telemetry (Non-Volatile)
- Watchdog Timer
- Memory Management Unit
  - Error Detection and Correction
  - Error Correcting Codes
- I/O Interfaces

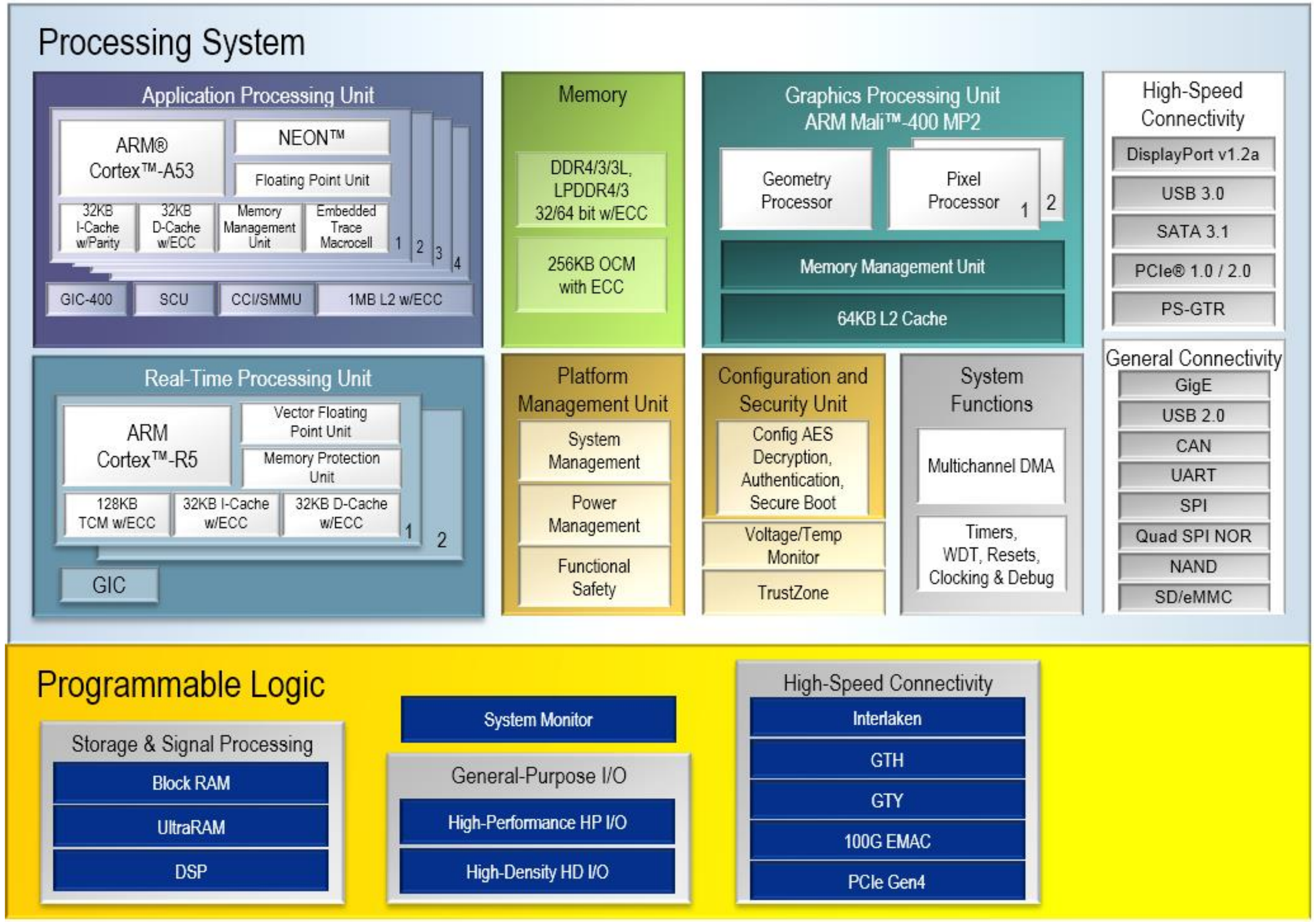


C&DH Technical Performance Measures

TPM	Rationale	Units
Processor Stress Ratio	The ratio of required processor throughput to available processor throughput with processor throughput measured in DMIPS/MHz	-
Boot Memory Capacity Ratio	The ratio of required boot memory to the available boot memory with memory measured in MB.	-
Configuration Memory Capacity Ratio	The ratio of required configuration memory to the available configuration memory with memory measured in MB.	-
Telemetry Memory Capacity Ratio	The ratio of required telemetry memory to the available telemetry memory with memory measured in MB.	-
Working Memory Capacity Ratio	The ratio of required working memory to the available working memory with memory measured in MB.	-
Number of Header Pins	The number of avionics stack header pins required for the execution of all C&DH operations	-

All-In-One OBCs: The System-on-Chip Architecture

- System-on-Chip (SoC)
  - Based on Field Programmable Gate Array (FPGA)
  - Hosting software
- MPSoC
  - 2 processors
  - FPGA fabric
    - Can add additional soft processors
  - FinFET
- Latch Circumvention Circuitry



[19] Xilinx, 2021

What Should We Remember?

- Boot Memory (non-volatile)
  - Boot processes (buildroot, bootlin, Das U-boot)
  - Operating System settings
  - Extremely radiation-tolerant
  - Read-only

- Configuration Memory (non-volatile)
  - Critical spacecraft parameters
  - Software mode parameters
  - Navigation information (SPICE Library)
  - Look-Up Tables (LUT)
  - Very radiation-tolerant
  - Read and Write

What Should We Remember?

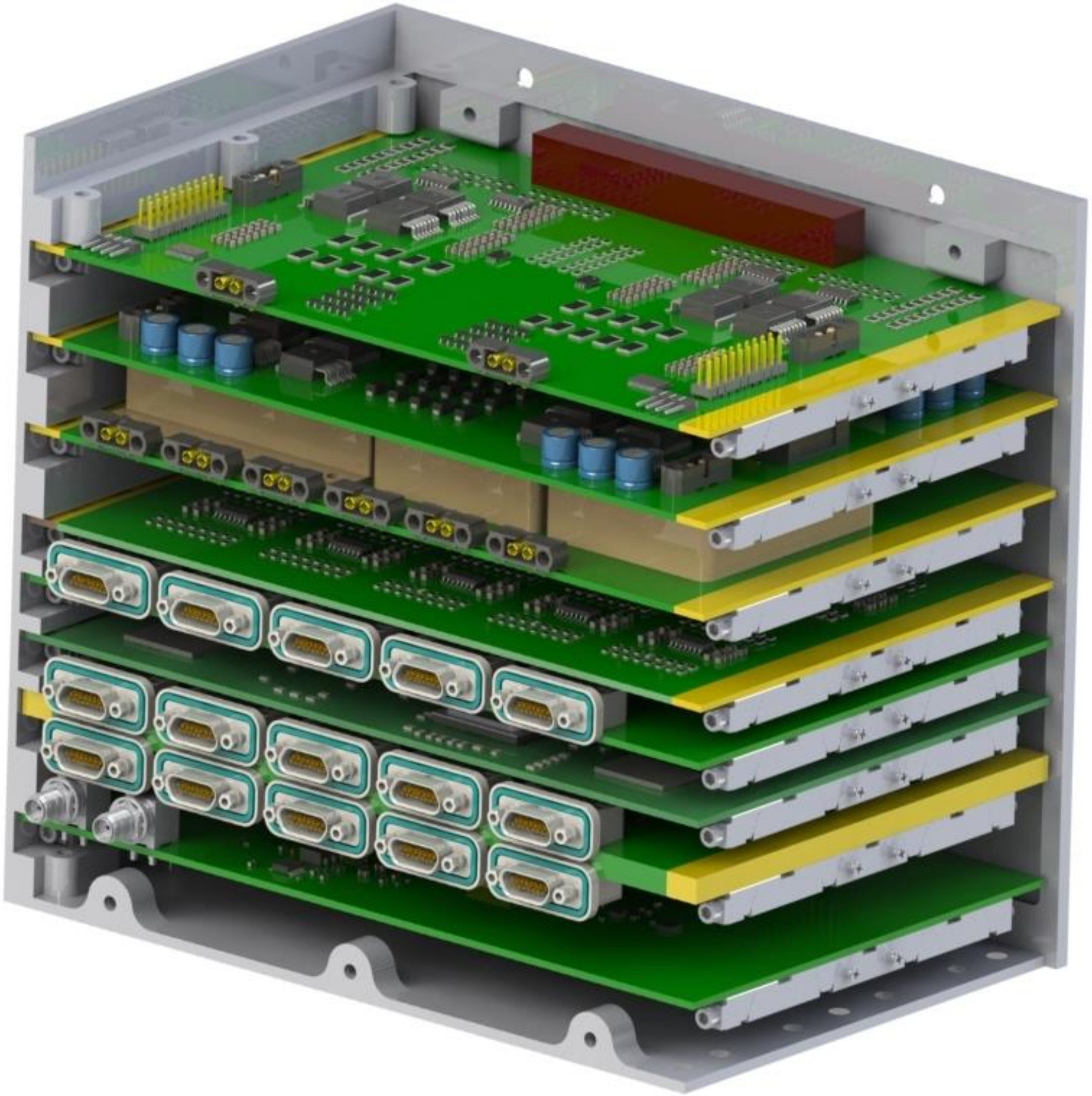
- Telemetry Memory (non-volatile)
  - Encoded data, not raw data
  - Organized Science data from Payload computer
  - Event logs from Flight Software
  - Diagnostics from Flight Software
  - Read and Write

- Working Memory (volatile)
  - Any spacecraft math on OBC lives here
  - While operational, OS and Flight Software values
  - Generally reads from configuration and boot and then works from here
  - Read and write

The C&DH Peripherals

- External to Avionics Box
  - GN&C Hardware
  - Payload Hardware
  - Mechanisms
  - Solar Arrays

- Internal To Avionics Box
  - Payload Interface Unit
  - Electrical Power System
  - Software-Defined Radio



Telemetry, Tracking, & Command

Can Anyone Hear Me?

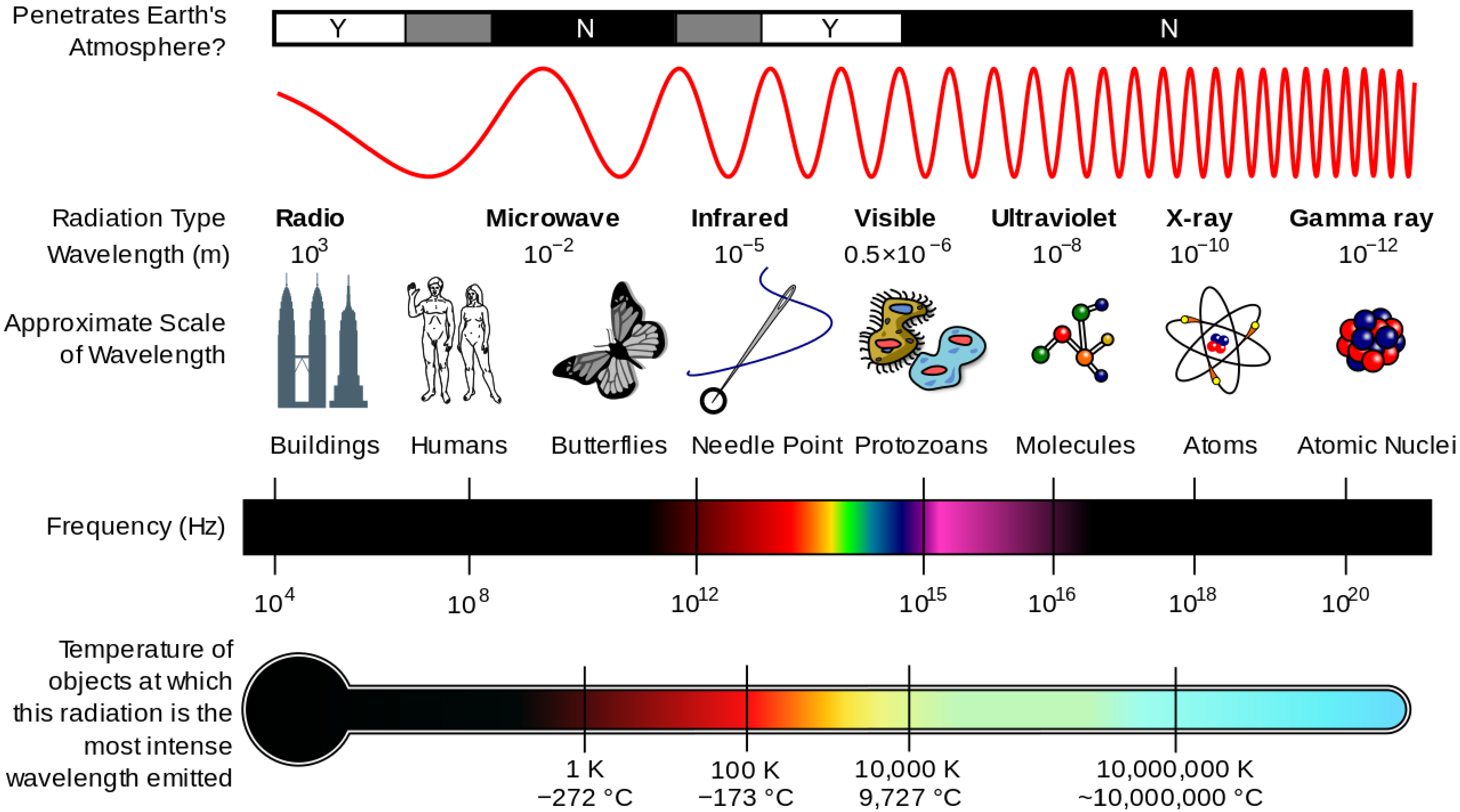
- TT&C talks to external entities, except for GPS satellites
  - Downlink: Spacecraft-to-ground communications
  - Uplink: Ground-to-spacecraft communications

- Three major components in a TT&C subsystem
  - Radio
  - Antenna
  - Encoding Scheme

TT&C Technical Performance Measures

TPM	Rationale	Units
Power Delivered to Antenna	The amount of power provided from the radio to the antenna during downlink	W
Transmitting Antenna Gain	The ratio of power produced by the antenna from a far-field source on the antenna's beam axis to the power produced by a hypothetical, lossless, isotropic antenna	dB
Receiving Antenna Gain	The ratio of power received by the antenna from a far-field source on the antenna's beam axis to the power received by a hypothetical, lossless, isotropic antenna	dB
Bit Error Rate	The number of bit errors incurred as a result of encoding, modulation, and electrical radio operations	Errors/bit
Signal to Noise Ratio	The ratio of signal power to the ratio of noise power	dB
Number of Header Pins	The number of avionics stack header pins required for the execution of all TT&C operations	-

Electromagnetic Spectrum



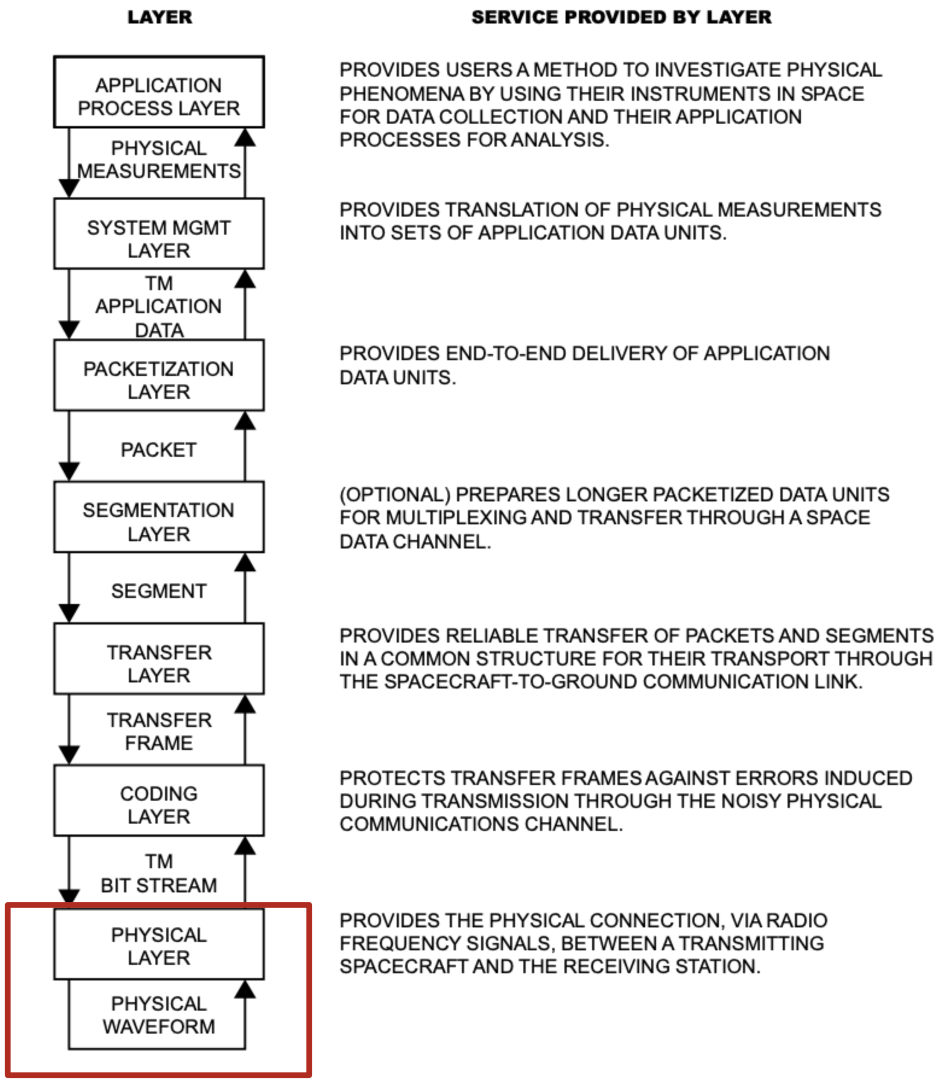
[19] Wikimedia, 2021

Radio Technology

- Radios can be built different ways, but they all do the same things
  - Transmitter
    - Digital-to-Analog Conversion (Data in from C&DH, analog signal out)
    - Signal conditioning (amplification, filtration)
    - Mixing the frequency from Intermediate Frequency (IF) to Carrier Frequency (CF)
    - Provide CF at appropriate power and Signal-to-Noise Ratio (SNR) to the Antenna
  - Receiver
    - Receive CF at some power and SNR at receiving antenna
    - Amplify signal power (Increases SNR)
    - Filter signal (Increases SNR)
    - Mix the CF down to the IF
    - Analog-to-Digital Conversion to get encoded data
      - FSW Decodes Data After

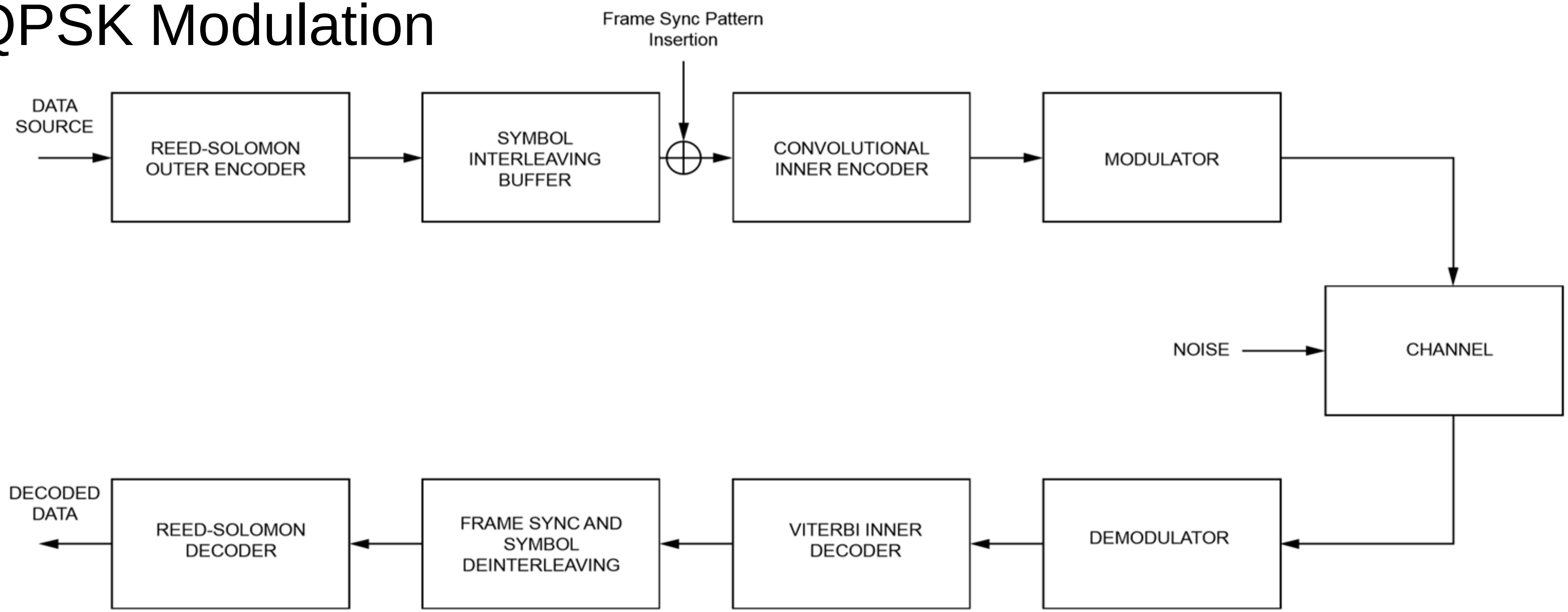
The Open Systems Interconnection (OSI) Model

- Set of formatting procedures
- Layers 2-7: Flight Software's Job
  - Organize Data
  - Format Data
  - Encode Data (TX) / Decode Data (RX)
- Layer 1: Physical Layer (TT&C's job)
  - Unstructured, raw data
  - Bits to RF and back
  - Analog-to-Digital Conversion
  - Digital-to-Analog Conversion
  - Waveform Modulation



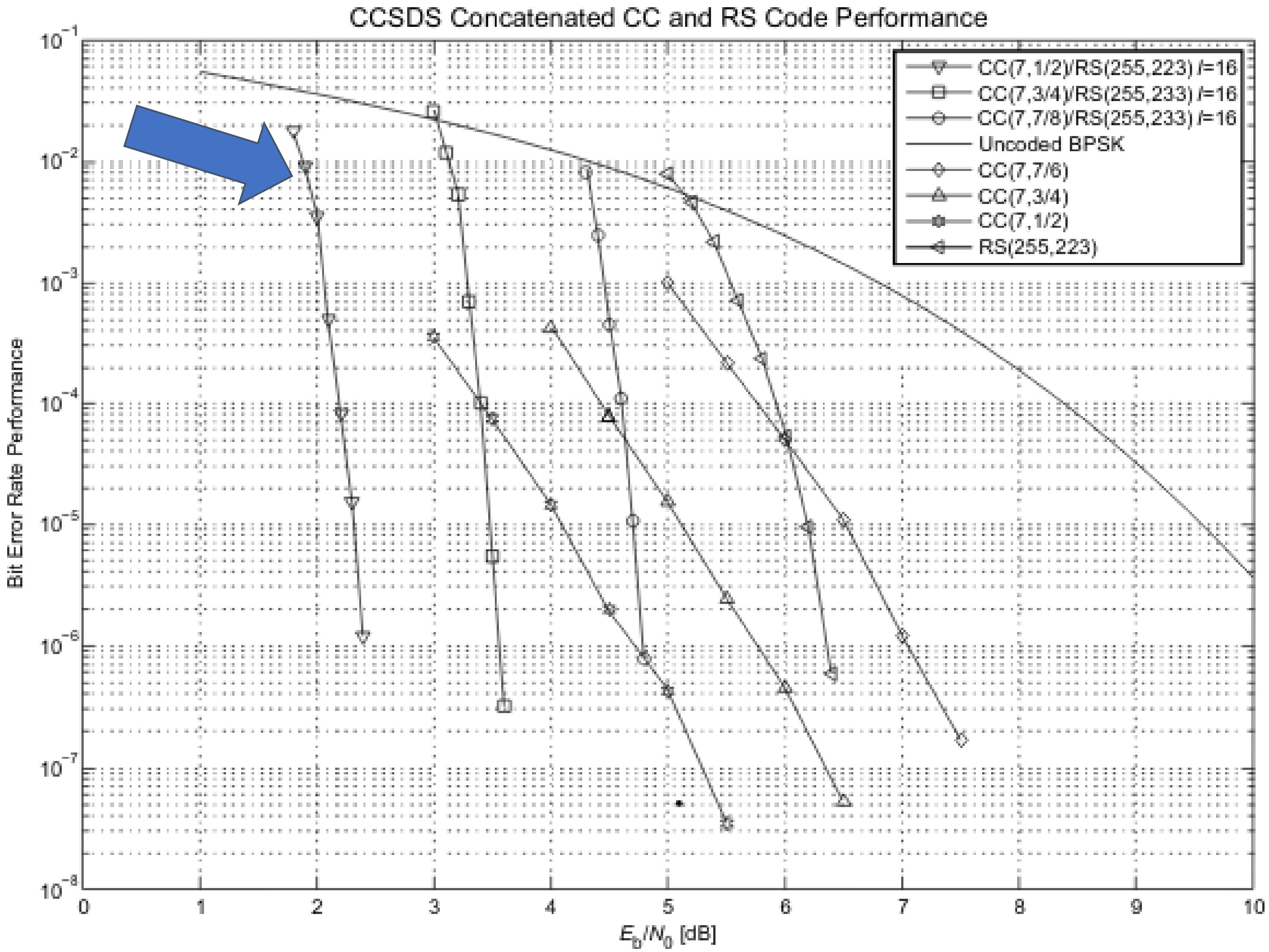
Consultative Committee on Space Data Systems

- CCSDS defines encoding scheme, ABEX uses this one
- Concatenated Encoding = Block Code + Convolutional Code
- Block Code = Reed-Solomon (255,223), Convolutional = (7,1/2)
- QPSK Modulation



Why do we Change the Data Waveform?

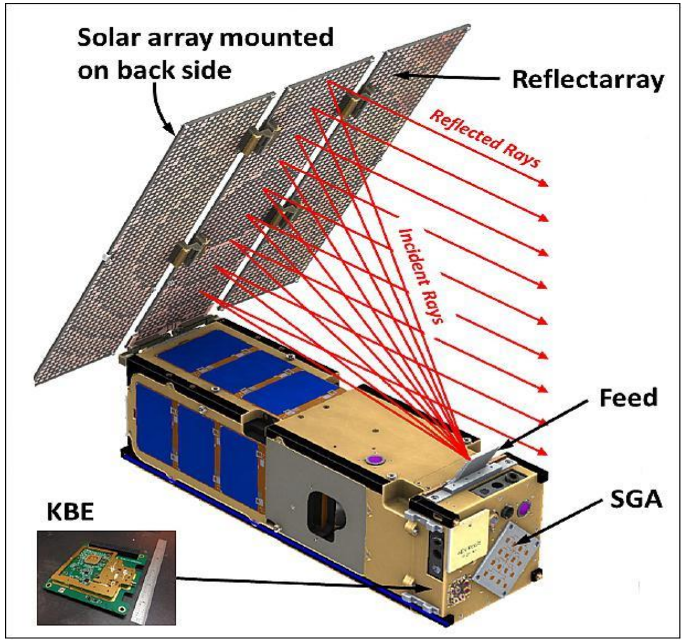
- Signals have Signal-to-Noise ratios (SNR)
- Per-bit SNR is E_b/N_0
- Increase E_b/N_0, Decrease Bit Error Rate (BER)
- BER of 10-6 means only one error in 1,000,000 bits



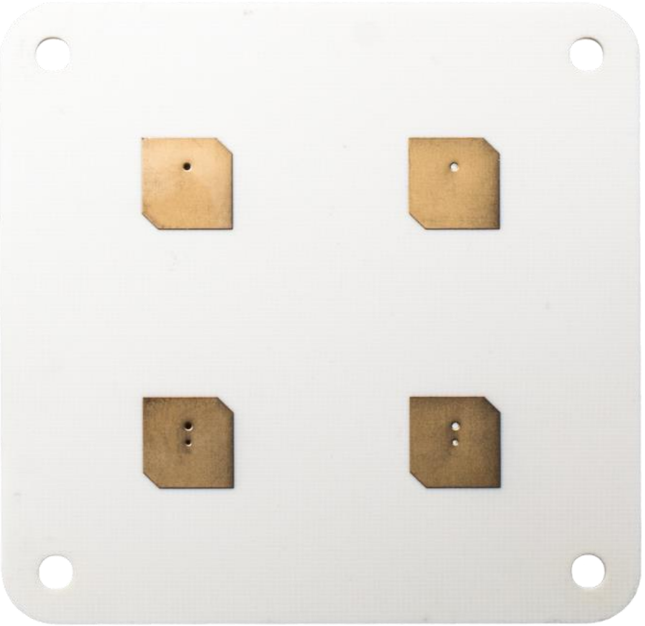
Antenna Technology

- Antennas on satellites are usually either parabolic or patch
  - Fancy ones use reflectarray

TDRS Reflector Antenna Design [21]



ISARA Reflectarray Design [22]



EnduroSat 4x4 X-band Patch [23]

[21] Dunbar, 2015 [22] Hodges, 2015 [23] EnduroSat, 2017

Guidance, Navigation, & Control

Trying to Improve Our Attitude

- A spacecraft's orientation in space is known as its attitude
- Attitude is disturbed by forces over a moment arm, or torques
  - Forces & Torques
    - Solar Radiation Pressure (SRP)
    - Aerodynamic Drag
    - Gravity Gradients
    - Residual Magnetic Dipoles
- When below 400 km, aerodynamic drag is a problem
- Between 400-1000 km altitude, gravitational and magnetic are the biggest concerns
- Above 500 km, SRP torques approach that of aerodynamic drag

GN&C Technical Performance Measures

TPM	Rationale	Units
Reaction Wheel Maximum Inertial Storage	How much angular momentum the attitude control system can dampen, which is a function of the maximum expected rotation rate	N-m-s
Attitude Control Authority	The precision at which the attitude control system can control the rate of spacecraft attitude change	deg/s/axis
Attitude Pointing Authority	The precision at which the attitude control system can control the spacecraft attitude	deg
Attitude Sensing Precision	The precision at which the spacecraft's attitude can be measured and reported to the attitude control system	deg
Navigator Position Drift Rate	The rate at which uncertainty in the spacecraft's orbital position state vector propagator is incurred	km/s/axis
Navigator Velocity Drift Rate	The rate at which uncertainty in the spacecraft's orbital velocity state vector propagator is incurred	km/s/s/axis

Attitude Determination

- We need to figure out which direction we're facing before we control our attitude
  - Accomplished using vector rotations about a known frame
  - Directional Cosine Matrices, Euler Angles, and Quaternions
    - Quaternions are the most useful because they take the least computational effort
    - Three vector components, one scalar component
    - p, q, r are rotation rates from Euler angles, not the same as q_x or $\dot{q}_y$

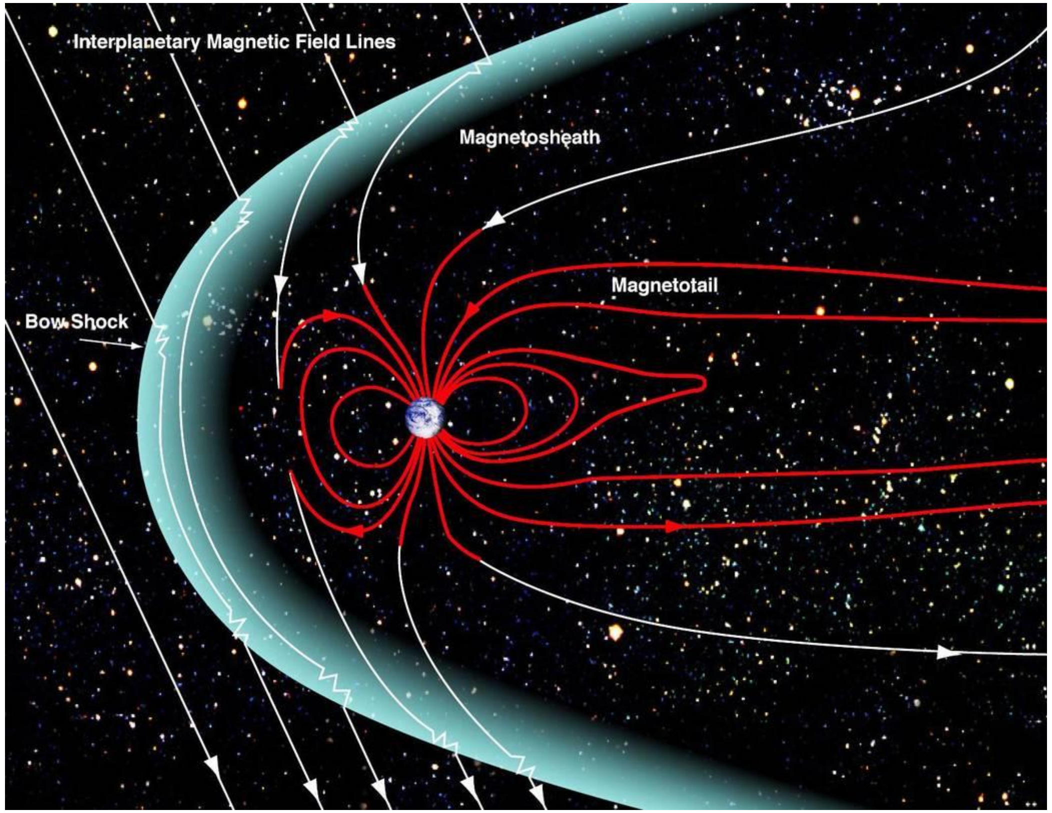
$$q = \begin{bmatrix} s \\ \vec{v} \end{bmatrix} = \begin{bmatrix} s \\ v_x \\ v_y \\ v_z \end{bmatrix} = \begin{bmatrix} q_s \\ q_x \\ q_y \\ q_z \end{bmatrix}$$

$$\begin{Bmatrix} \dot{q}_s \\ \dot{q}_x \\ \dot{q}_y \\ \dot{q}_z \end{Bmatrix} = \frac{1}{2} \begin{bmatrix} 0 & -p & -q & -r \\ p & 0 & r & -q \\ q & -r & 0 & p \\ r & q & -p & 0 \end{bmatrix} \begin{Bmatrix} q_s \\ q_x \\ q_y \\ q_z \end{Bmatrix}$$

Magnets: How do they Work?

- Earth has a Magnetosphere
  - Comes from spinning molten core
  - Generates a dipole around Earth
  - Traps radiation

- We can use the magnetic fields
  - Magnetometers measure magnetic North
  - Corrected for true North using World Magnetic Model
  - Magnetorquers (Torque Rods) can induce a torque on the satellite



Magnetosphere Shape & Field Lines [24]

[24] Zell, 2015

Attitude Determination Technology

- Sensors exist to help us figure out where we're looking
  - Star Trackers (multiple vectors, full quaternion)
  - Sun Sensors (one vector, half quaternion)
  - Inertial Measurement Units (rotation rates for Euler angles)
  - Magnetometers (one vector per unit, three for full quaternion)
  - Horizon Sensors (one vector, half quaternion)
- What combination of sensors is needed for a redundant system?

Attitude Control

- Use Cases:
  - Detumble the satellite
  - Point solar arrays at Sun
  - Point antennas at Earth

- Control Technologies:
  - Reaction Wheels (most common)
  - Torque Rods
  - Reaction Control System (Propulsive)
  - Propulsion (Thrust Vector Control)

Sizing Reaction Wheels: You Need to Know Your Inertia

Parameter	Description	Value	Unit
m_{cs}	CubeSat Mass	25	kg
w_{cs}	Width, X-direction	0.226	m
h_{cs}	Height, Y-direction	0.226	m
l_{cs}	Length, Z-direction	0.366	m
I_{cm}	Body Mass Moment of Inertia	Variable per Axis	kg-m^2
I	Total Mass Moment of Inertia	Variable per Axis	kg-m^2
m_{array}	Mass per Solar Array (all same)	1.5	kg
d_c	Distance from Centroid (all same)	0.1	m
ω	Maximum Expected Rotation Rate	5	°/s
L_{max}	Maximum Angular Momentum	0.0183	N-m-s
FOS	Factor of Safety	2.5	-

- Calculate your Body Mass Moment of Inertia,

$$I_{cm,l} = \frac{1}{12} \cdot (m_{cs} - m_{array}) \cdot (w_{cs}^2 + h_{cs}^2) = \frac{1}{12}(25\,kg - 1.5\,kg) \cdot [(0.226\,m)^2 + (0.366\,m)^2)] =$$

$$I_{cm,w} = \frac{1}{12} \cdot (m_{cs} - m_{array}) \cdot (l_{cs}^2 + h_{cs}^2) = \frac{1}{12}(25\,kg - 1.5\,kg) \cdot [(0.226\,m)^2 + (0.366\,m)^2)] =$$

$$I_{cm,h} = \frac{1}{12} \cdot (m_{cs} - m_{array}) \cdot (l_{cs}^2 + w_{cs}^2) = \frac{1}{12}(25\,kg - 1.5\,kg) \cdot [(0.226\,m)^2 + (0.226\,m)^2)] =$$

Sizing Reaction Wheels: You Need to Know Your Inertia

Parameter	Description	Value	Unit
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d_c	Distance from Centroid (all same)	0.1	m
ω	Maximum Expected Rotation Rate	5	°/s
L_{max}	Maximum Angular Momentum	0.0183	N-m-s
FOS	Factor of Safety	2.5	-

- Apply the Parallel Axis Theorem,

$$I_l = I_{cm,l} + m_{array,l} \cdot d_{c,l}^2 =$$

$$I_w = I_{cm,w} + m_{array,w} \cdot d_{c,w}^2 =$$

$$I_h = I_{cm,h} + m_{array,h} \cdot d_{c,h}^2 =$$

Sizing Reaction Wheels: You Need to Know Your Inertia

Parameter	Description	Value	Unit
m_{cs}	CubeSat Mass	25	kg
w_{cs}	Width, X-direction	0.226	m
h_{cs}	Height, Y-direction	0.226	m
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ω	Maximum Expected Rotation Rate	5	°/s
L_{max}	Maximum Angular Momentum	0.0183	N-m-s
FOS	Factor of Safety	2.5	-

- Assume a max rotation rate, then determine max angular momentum,

$$\omega = \left(5\frac{degree}{s}\right)\cdot\left(\frac{\pi}{180}\right) = 0.0873\frac{rad}{s}$$

$$L_{max} = \omega \cdot I_{max} =$$

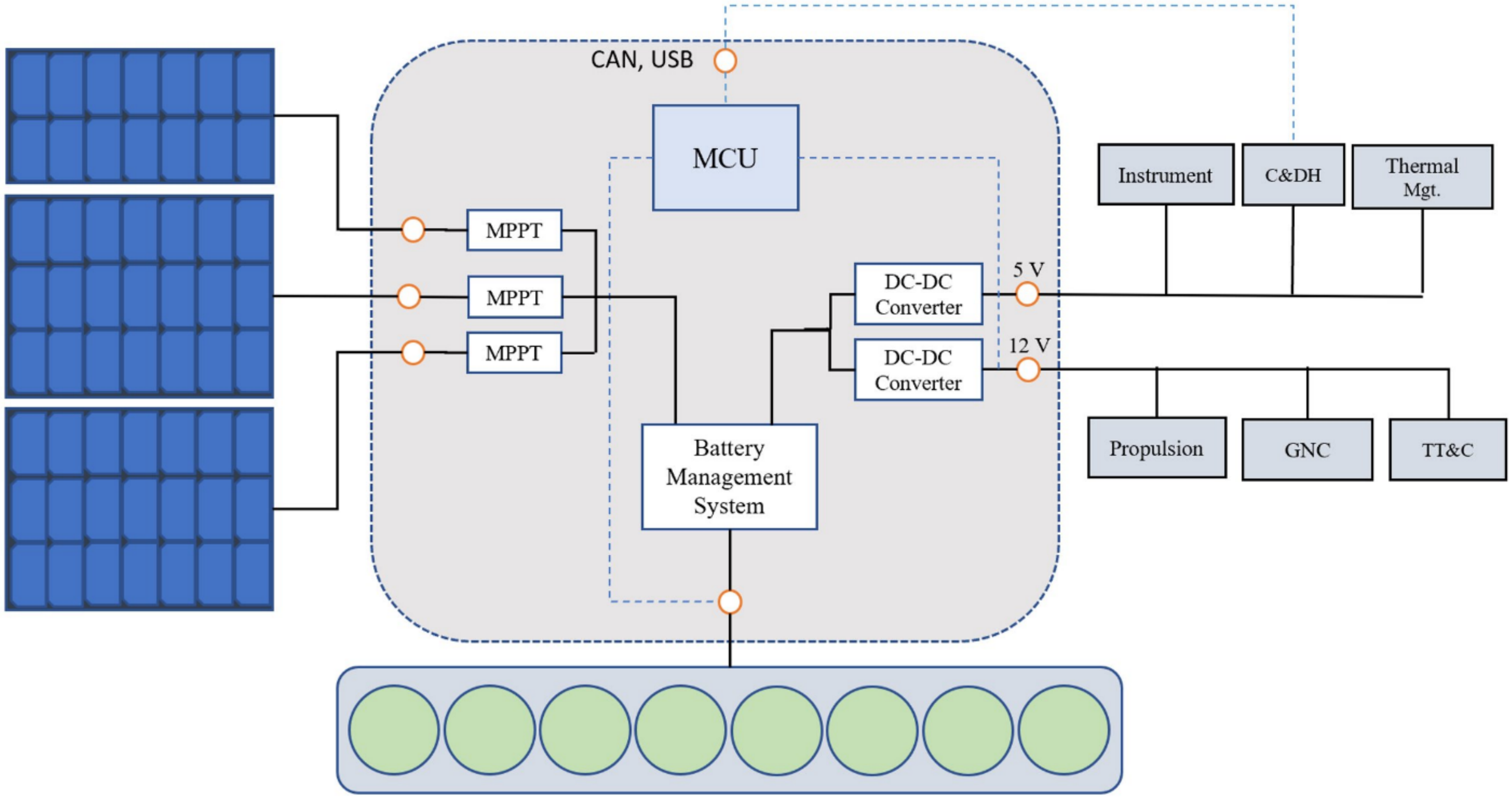
$$Inertial\ Storage = L_{max} \cdot FOS =$$

Electrical Power System

High-Voltage Spacecraft Education

- Three components are considered in power organization:
  - Power Generation
  - Power Distribution
  - Power Consumption
- A spacecraft EPS generally has three components:
  - Battery Management System
  - Maximum Power Point Trackers
  - DC/DC Conversion

Building an EPS is Difficult



EPS Technical Performance Measures

TPM	Rationale	Units
EPS Power Conversion Efficiency	The ratio of power available to the spacecraft for consumption over the power generated by the solar cells.	-
Battery Charging Efficiency	The ratio of power stored in the batteries over the power provided to the batteries.	s
Battery Charging Time	The time required to fully charge the batteries	s
Powered-On Time	The time between receiving the power-on signal and providing power to the rest of the spacecraft.	s
Solar Array Disconnect Time	The time required to disconnect the EPS from the solar arrays in the event the batteries are fully charged	s
Solar Array Reconnect Time	The time required to reconnect the EPS from the solar arrays in the event the batteries are no longer fully charged	s
Number of Header Pins	The number of avionics stack header pins required for the execution of all EPS operations	-

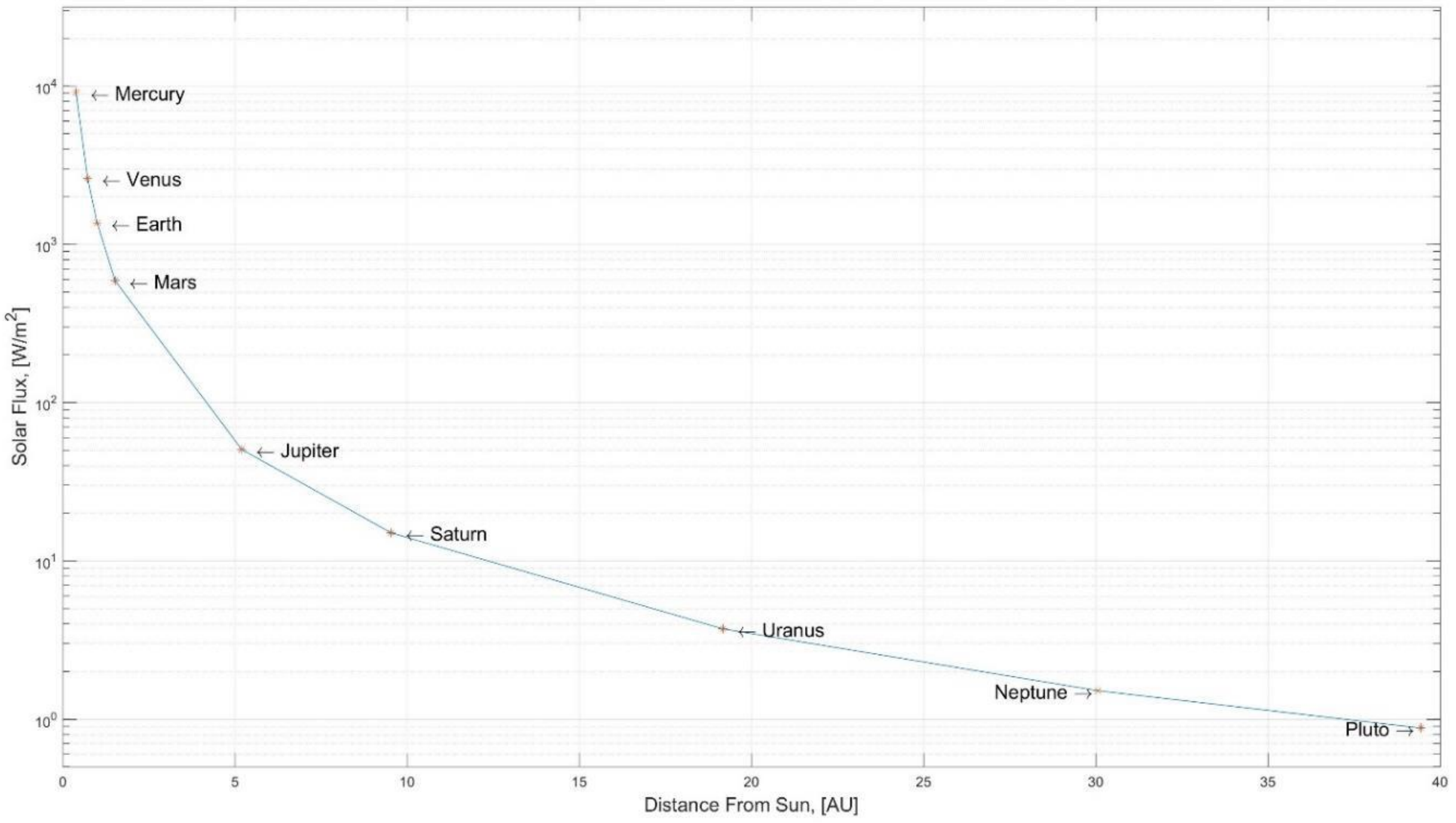
Building an EPS is Difficult

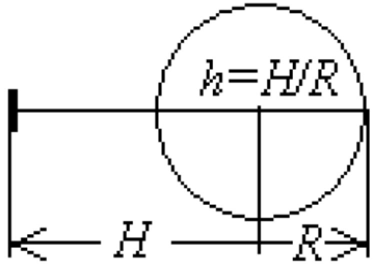
- The majority of spacecraft failures are due to the EPS or FSW
  - Lots of critical components
  - Electrostatic Discharge is a problem
  - Solar arrays degrade over time
  - Li-ion batteries don't like to get over 20 deg C or below -10 deg C

- Focusing on the math today rather than the composition

Power Generation

- How much flux is available is a function of distance





$$F_{rad} = \frac{1}{\left(\frac{H}{R}\right)^2}$$

$$Q''_{Sun,Ems} = \sim 1{,}366 \frac{W}{m^2}$$

[15] Halvorson, 2020

Power Generation

- Solar array power generation is a function of,
  - Surface material absorptivity
  - Solar array efficiency (band gap energy)
  - Temperature
  - Angle of Incidence
  - Array area
- Assume,
  - $\eta_{cell} = 0.32,\ \alpha_{cell} = 0.89, \varphi_{temp} = 0.88, A_{array} = 0.3136\ \text{m}^2, \theta = 10\ \text{deg}$

$$Q_{array} = Q''_{Sun,Ems} \cdot \eta_{cell} \cdot \alpha_{cell} \cdot \varphi_{temp} \cdot A_{array} \cdot \cos(\theta) =$$

[15] Halvorson, 2020

Power Distribution

- Distributing power incurs power losses
  - Power lines such as power channels in the avionics stack header
  - Diodes are particularly troublesome
- We look at both the margin and the efficiency
- Assume $\eta_{diode+line} = 0.9, M_{dist} = 0.9$

$$Q_{dist} = Q_{array} \cdot M_{dist} \cdot \eta_{diode+line} =$$

[15] Halvorson, 2020

Power Consumption

- The power during each software mode must be known
  - Sum the power from each component during the mode
  - Each component has its own power conversion efficiencies
  - Margins are applied here to overcome the component efficiencies
- Assume,
  - $\sum Q_{component} = 42.4\, W, M_{consumption} = 1.2, \eta_{EPS} = 0.75$

$$Q_{consumption} = \left(\frac{\sum Q_{component}}{\eta_{EPS}}\right) \cdot M_{consumption}$$

[15] Halvorson, 2020

Power Generation vs. Power Consumption

- How do our two values compare?

- If power generated isn't enough to supply for power consumed,
  - Increase solar array size
  - Decrease component power consumption
  - Increase power conversion efficiencies
  - Decrease Angle of Incidence via Control
  - Increase solar cell efficiencies

[15] Halvorson, 2020

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ABEX

Alabama **B**urst **E**nergetics e**X**plorer

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