Alpbach Summer School July 15-25,2019

POLITECNICO
MILANO 1863

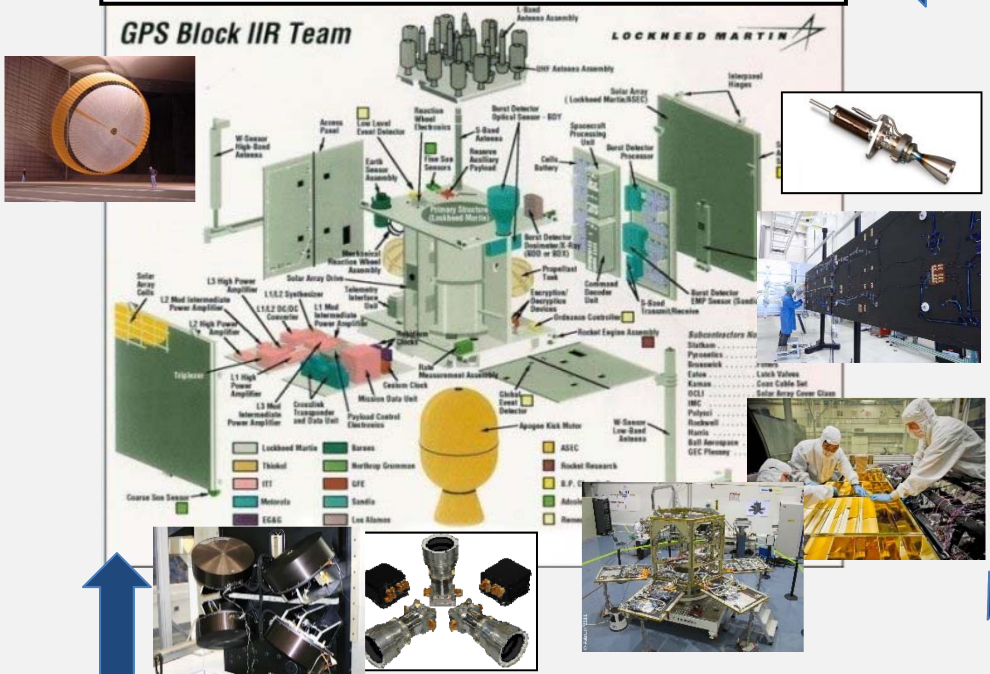
System Engineering and Technology

Michèle Lavagna

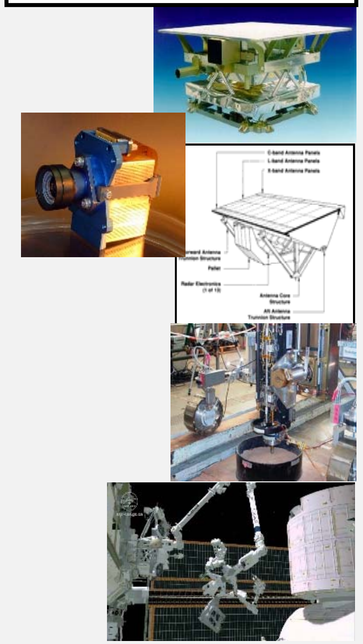
michelle.lavagna@polimi.it

Space Missions and System Engineering

Space segment Design

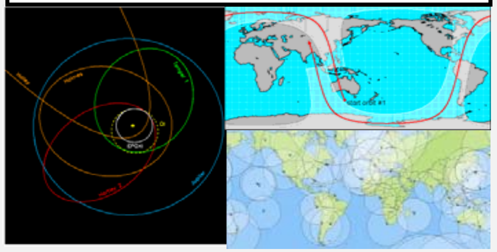


Payload

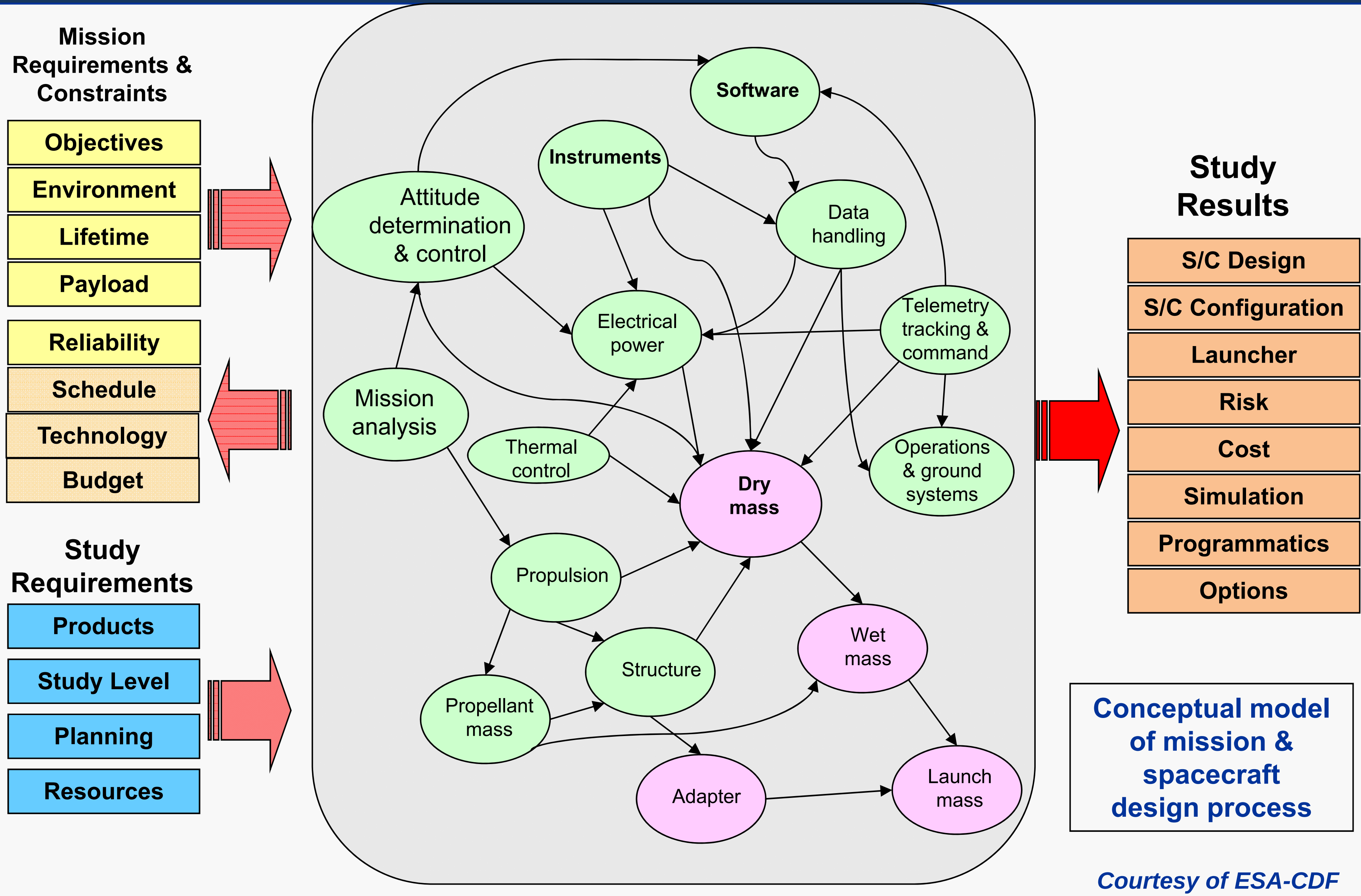


Ground Segment Design

MA &GNC



Design Process: strongly multidisciplinary & interconnected



Courtesy of ESA-CDF

Mission Design→ how

Space Mission Design: any design process is made of the following bricks:

***Goal analysis*→requirements identification and definition (functional, operation, technical, etc)**

↓

***Alternatives identification*, drivers\criteria definition and comparison** → **Analysis & Design**

↓

Baseline motivated selection → **Analysis & Design**

↓

Requirements **refinement\revision** and critical aspects identification to **assess the mission feasibility and address next project phases→ to build & to launch**

Mission Design → building blocks: the technologies

- Power Generation, distribution & Control systems
- Propulsion systems
- Dynamics control
  - ✓ *Trajectory\attitude Guidance Navigation & Control*
  - ✓ *Rendez-vous & docking*
  - ✓ *Landing*
- On board software
- Avionics
- Robotics
- Materials\thermal-structural components
- Communications
- .Environmental protection
  - ✓ *Rad-hard (manned\unmanned)*
  - ✓ *Sample curation: biological protection\sterilization*
  - ✓ *Impacts*
  - ✓ *Ionization*
- Sensors\detectors
- Environmental control & Life Cycle systems
- Inhabited modules\surface infrastructures

→ **Technology development**

Mission design: building blocks - technologies

Technology areas *(examples)*

Power

- Solar Cells
- Nuclear Systems
- Fuel Cells
- High efficiency batteries
- Nuclear plants

Propulsion

- Electric
- Solar thermal
- Sails
- Air-breathing

Entry

- Aerobraking/aerocapture/ aerogravity assist
- Inflatable
- Thermal shields

Comms\data handling

- Wireless data exchange
- Laser\optical comms

Descent/Ascent

- Ascending/Launchers
- Descending Mechanism
- Soft Landing (airbags,crashable legs skycrane)
- Parachutes

Materials

- High temperature
- Environmental aggressions
- Low density High performance

infrastructures

- In situ chemical plants
- Habitats
- greenhouse

Human Factor

- Body\Radiation protection
- EVA support
- Ergonomy

Sw Engineering

- Distributed systems
- Learning units
- AI techniques

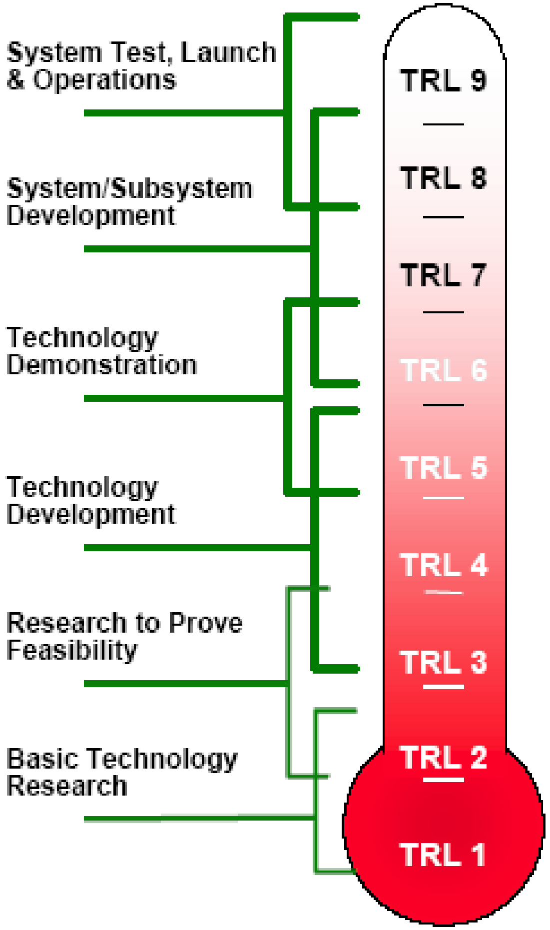
GNC

- Deep Space Nav
- EDL Navigation
- Precision Landing
- Autonomous navigation/scheduling/FDIR
- Proximity maneuvering

Robotics

- Grasping mechanisms
- manipulators
- Instrumentation: sample collection (mole, drill etc), in –situ soil analysis, storage systems
- Mobility\locomotion

Technologies Assessment: Tech Readiness Levels - TRL



- Actual system “flight proven” through successful mission operations
- Actual system completed and “flight qualified” through test and demonstration (Ground or Flight)
- System prototype demonstration in a space environment
- System/subsystem model or prototype demonstration in a relevant environment (Ground or Space)
- Component and/or breadboard validation in relevant environment
- Component and/or breadboard validation in laboratory environment
- Analytical and experimental critical function and/or characteristic proof-of-concept
- Technology concept and/or application formulated
- Basic principles observed and reported

System Engineering notes: Tech Readiness Levels - TRL

- Technology Readiness Levels Handbook for Space Applications

- Guidelines for the use of TRLs in ESA programmes (2013)

- Tailored ECSS Engineering Standards for In-Orbit Demonstration CubeSat Projects

Design process flow: I/O

Area	Input\IF	Task\goal
Mission Analysis MA	• *Mission objectives* • *Requirements* • *P/L* • *TMTC-ADCS-TCS-EPS* • *Operations-GS*	• **Orbit selection** • **Δv or propellant budget** • **Launcher selection** • **Eclipse, visibility and coverage** • **Station keeping needs**
Propulsion	• *MA* • *ADCS* • *Environment* • *Configuration*	• **Propulsion system selection** • **Propellant budget** • **Propellant feeding definition & sizing** • **Tank sizing**

Design process flow: I/O

Area	Input\IF	Task\goal
Telecom TT&C	•*P/L* •*operations* •*MA* •*environment* •*ADCS-EPS-OBDH*	• **Antenna selection** • **Ground segment definition** • **Tx and Rx component definition** • **Link budget**
Thermal Control TCS	•*All s\s* •*configuration* •*Environment* •*P/L-MA*	• **Thermal analysis** • **Thermal control definition** • **Material selection** • **Thermal budget**
Attitude Determination &Control ADCS	•*P/L- MA* •*TCS* •*TMTC* •*Configuration* •*EPS*	• **Sensors set definition** • **Actuators set definition** • **Control definition** • **pointing budget**

Design process flow: I/O

Area	Input\IF	Task\goal
On-Board Data Handling OBDH	• *P/L* • *all s/s*	• Architecture definition • Bus design • Computer budget • Data management • Component list • Interface definition
Operations	• *P/L* • *MA* • *TMTC*	• On board activities planning design • Mission phases & modes definition • Mission autonomy level definition • FDIR logic definition

Design process flow: I/O

Area	Input\IF	Task\goal
Power EPS	• *All s/s* • *MA* • *environment*	• Power budget • Array surface (or primary power unit sizing) • Bus definition • Battery selection and sizing (or storage unit sizing) • Power control definition
Configuration	• *MA* • *Launcher* • *all s/s*	• Mass budget • Geometry • Components allocation
Structures\mechs	• *MA* • *Launcher* • *all s/s* • *configuration*	• Structural analysis • Structural mass budget • Mechanisms sizing\selection

Design process flow: I/O

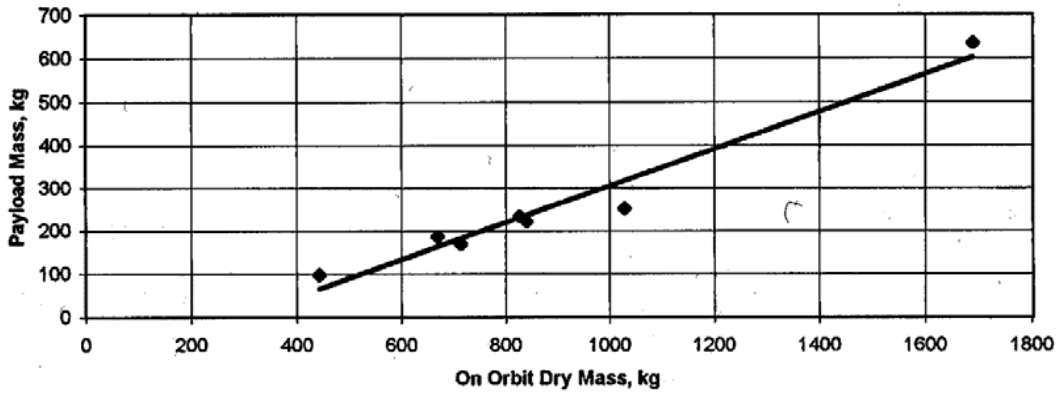
Area	Input\IF	Task\goal
Costs	• *Launcher* • *Ground\space segments* • *Operations\AIV-AIT*	• **Cost analysis** • **Cost budget**
Risks	• *Launcher* • *All s/s* • *Operations* • *AIV* • *programmatics*	• **Risk analysis** • **Mitigation actions definition**
AIV/AIT	• *Mission objectives* • *Modes* • *all s/s* • *Cost & programmatics*	• **Programmatic analysis** • **Test plan definition**

Let's practice

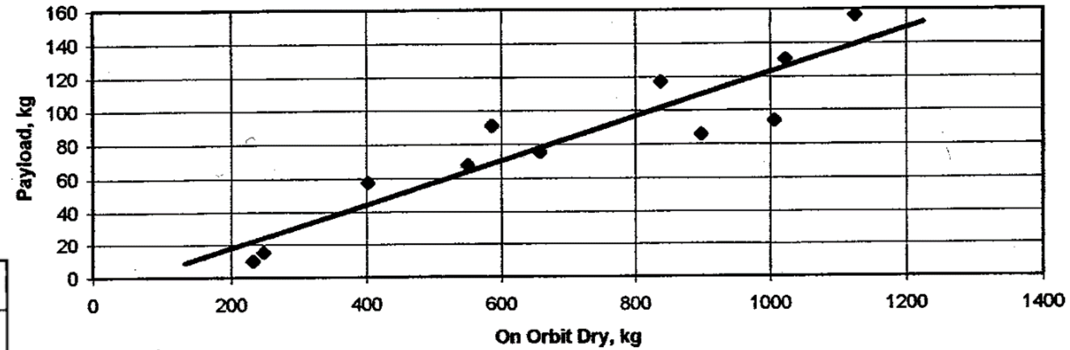
System Engineering notes - Mass allocation

1. Determine the **rough mass** for the imposed\selected payload
2. Determine **the mission class** and, accordingly, the on-orbit dry mass from statistical data
3. Determine the **total allowable on-orbit mass** from current launchers
4. Deduct launch vehicle adapter mass from the launch mass
5. Determine the propellants and pressurants required for the mission
6. Verify the on-orbit needed mass and **launchable mass** consistency
7. **Distribute the consistent gross mass** (m_0) among the on-board subsystem to impose preliminary constraints to start sizing
8. Start looping the design refinement

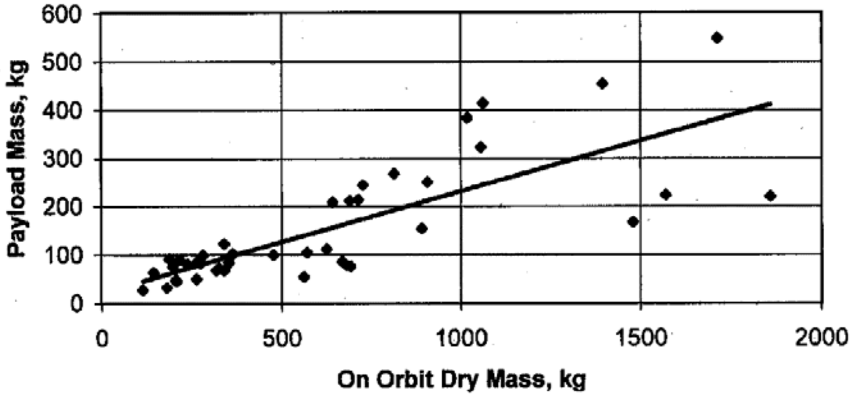
Step 1-2 Mass versus p/l



Communication spacecraft mass.



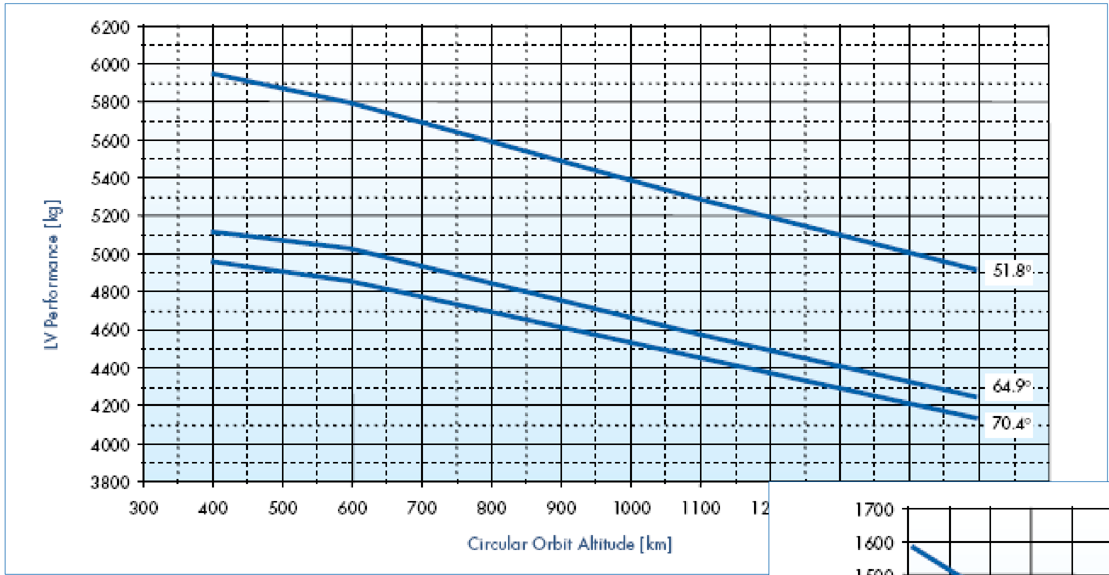
Planetary spacecraft mass.

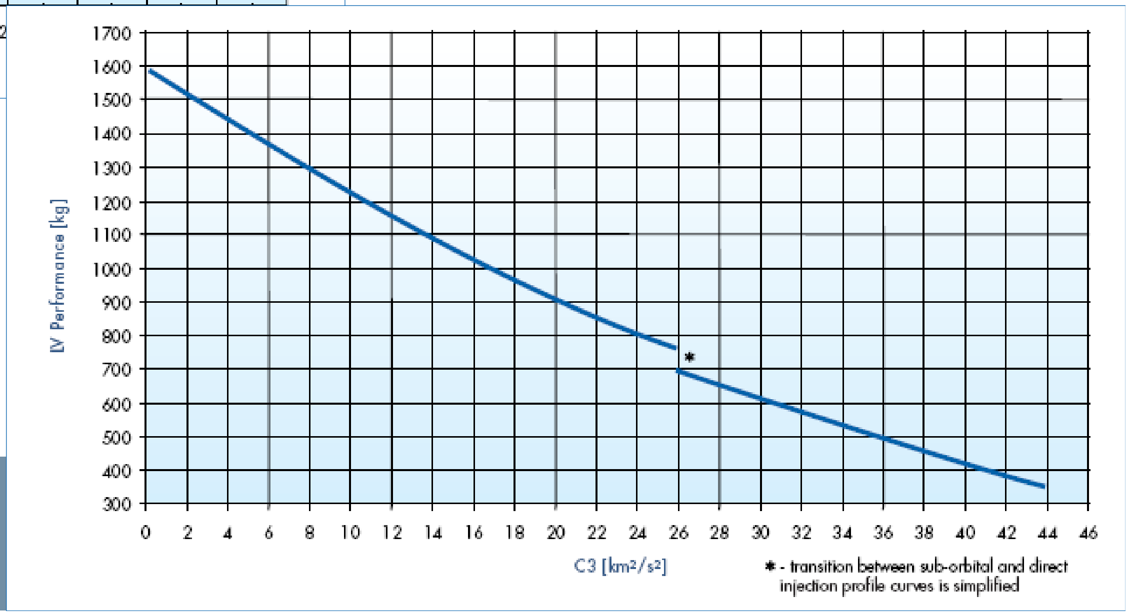


Earth-orbiting spacecraft mass.

Step 3-4 Launcher's performance

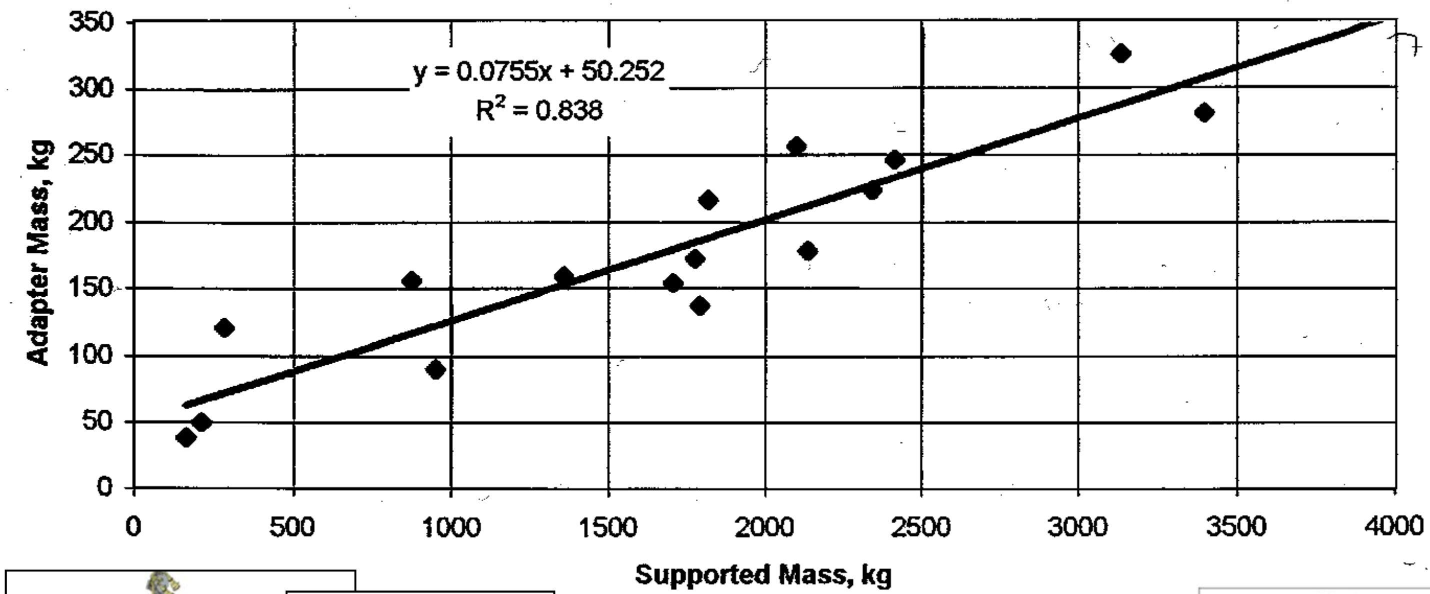
3. **Determine the maximum launch mass for the mission**: directly derived from the launcher capabilities



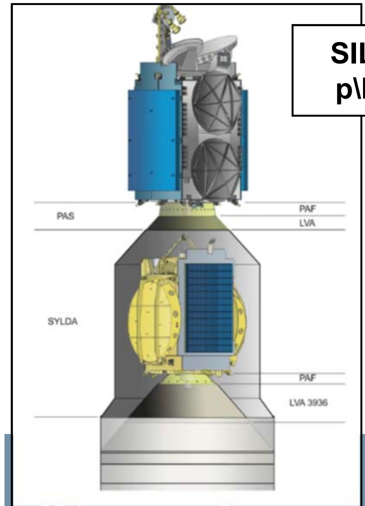


Step 3-4 launcher's interface

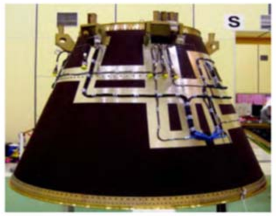
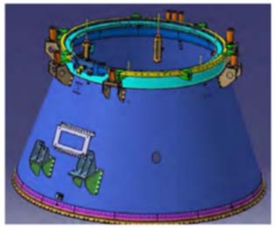
4. Deduct launch vehicle adapter mass from the launch mass:



VESPA double p\l VEGA



SILDA double p\l ARIANE V

Adapter	Description	Separation system
PLA 937 VG 	Total height: 1461 mm Total mass: 77 kg	Clamp-band ∅937 mm with low shock separation system
PLA 1194 VG 	Total height: 1071.5 mm Total mass: 78 kg (TBC)	Clamp-band ∅1194 mm with low shock separation system

Step 5-6 size the wet mass

5. **Determine the propellant required for the mission**: preliminary propellant mass computation (chemical propulsion) with the rocket equation:

$$\Delta V = I_{sp} g_0 \ln(m_0/m_{dry})$$

or assuming m_{prop}=60-70%m_o

6. **Verify the total allowable on-orbit dry mass:**

Dry mass+Margin+p/l+propellant < = LM-LVA

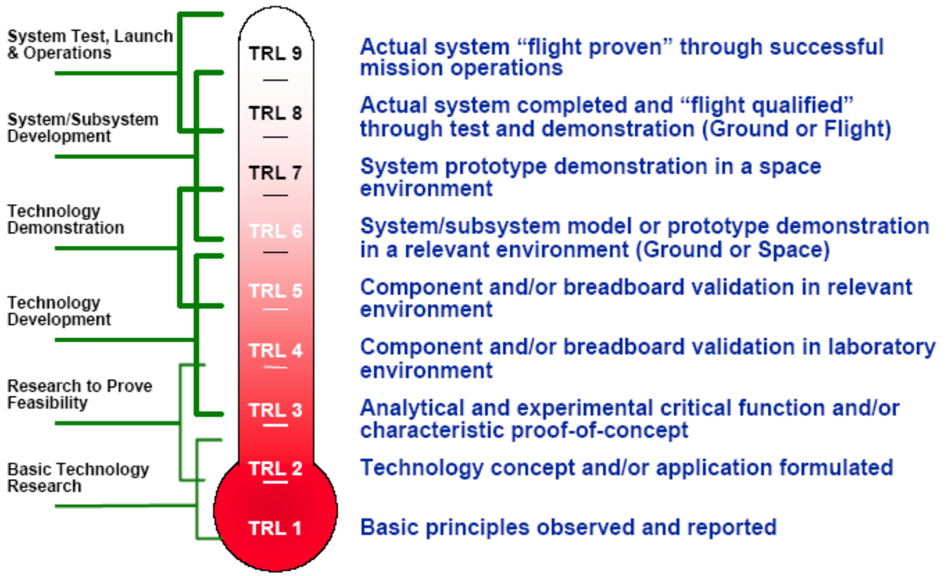
LVA=launch vehicle adapter

If the left member is **greater than the right** member either a different launcher shall be selected or a decrease of any of the left terms shall be imposed but the margin)

Technology Readiness Level and Margin philosophy

HRST TECHNOLOGY ASSESSMENTS

TECHNOLOGY READINESS LEVELS



Readiness Level	Definition	Explanation
TRL 1	Basic principles observed and reported	Lowest level of technology readiness. Scientific research begins to be translated into applied research and development. (See Paragraph 4.2)
TRL 2	Technology concept and/or application formulated	Once basic principles are observed, practical applications can be invented and R&D started. Applications are speculative and may be unproven. (See Paragraph 4.3).
TRL 3	Analytical and experimental critical function and/or characteristic proof-of-concept	Active research and development is initiated, including analytical / laboratory studies to validate predictions regarding the technology. (See Paragraph 4.4)
TRL 4	Component and/or breadboard validation in laboratory environment	Basic technological components are integrated to establish that they will work together. (See Paragraph 4.5)
TRL 5	Component and/or breadboard validation in relevant environment	The basic technological components are integrated with reasonably realistic supporting elements so it can be tested in a simulated environment. (See Paragraph 4.6)
TRL 6	System/subsystem model or prototype demonstration in a relevant environment (ground or space)	A representative model or prototype system is tested in a relevant environment. (See Paragraph 4.7)
TRL 7	System prototype demonstration in a space environment	A prototype system that is near, or at, the planned operational system. (See Paragraph 4.8)
TRL 8	Actual system completed and "flight qualified" through test and demonstration (ground or space)	In an actual system, the technology has been proven to work in its final form and under expected conditions. (See Paragraph 4.9)
TRL 9	Actual system "flight proven" through successful mission operations	The system incorporating the new technology in its final form has been used under actual mission conditions. (See Paragraph 4.2.10)

Mass margins philosophy

6. **Evaluate the mass margin to be set aside**

1=new s/c

2=next generation s/c

3=existing design s/c

Abbrev.	Review name
CoDR	Conceptual design review
PDR	Preliminary design review
CDR	Critical design review
PRR	Preshipment readiness review
FRR	Flight readiness review

Description/ categories	Minimum standard weight contingencies, %														
	Proposal stage			Design development stage											
	Bid Class			*CoDR* Class			*PDR* Class			*CDR* Class			*PRR* Class		
	1	2	3	1	2	3	1	2	3	1	2	3	1	2	3
Category AW, 0–50 kg 0–110 lb	50	30	4	35	25	3	25	20	2	15	12	1	0	0	0
Category BW, 50–500 kg 110–1102 lb	35	25	4	30	20	3	20	15	2	10	10	1	0	0	0
Category CW, 500–2500 kg 1102–5511 lb	30	20	2	25	15	1	20	10	0.8	10	5	0.5	0	0	0
Category DW, 2500 kg and up	28	18	1	22	12	0.8	15	10	0.6	10	5	0.5	0	0	0

[a]Data copyright AIAA, reproduced with permission; Ref. 11.

Mass preliminary breakdown

7. **Allocate mass percentage for each s/s:** preliminary mass percentage distribution (as % of dry mass)

GFE=government furnished equipment)

Subsystem	Comsats[a] with P/L[c]	Comsats[a] GFE P/L	Metsats[b] with P/L	Metsats[b] GFE P/L	Planetary with P/L	Planetary GFE P/L	Other with P/L	Other GFE P/L
Structure, %	21	29	20	29	26	29	21	30
Thermal, %	4	6	3	4	3	3	3	4
ACS, %	7	10	9	13	9	10	8	11
Power, %	26	35	16	23	19	21	21	29
Cabling, %	3	4	8	12	7	8	5	7
Propulsion, %	7	10	5	7	13	15	5	7
Telecom, %	—	—	4	6	6	7	4	6
CDS, %	4	6	4	6	6	7	4	6
Payload, %	28	—	31	—	11	—	29	—

[a]Comsat = communication satellite. [b]Metsat = meteorology or weather satellite. [c]P/L = payload.

Start sizing and iterate

Power preliminary allocation

Total power versus p/l power and mission category first estimate

Spacecraft mission	Power estimating relationship
Communications	$P_t = 1.1568 P_{pl} + 55.497$
Meteorology	$P_t = 602.18 \ln(P_{pl}) - 2761.4$
Planetary	$P_t = 332.93 \ln(P_{pl}) - 1046.6$
Other missions	$P_t = 210 + 1.3 P_{pl}$

Power margins philospophy

Evaluate the power margin to be set aside

1=new s/c

2=next generation s/c

3=existing design s/c

Abbrev.	Review name
CoDR	Conceptual design review
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Description/ categories	Minimum standard power contingencies, %														
	Proposal stage			Design development stage											
	Bid Class			*CoDR* Class			*PDR* Class			*CDR* Class			*PRR* Class		
	1	2	3	1	2	3	1	2	3	1	2	3	1	2	3
Category AP, 0–500 W	90	40	13	75	25	12	45	20	9	20	15	7	5	5	5
Category BP, 500–1500 W	80	35	13	65	22	12	40	15	9	15	10	7	5	5	5
Category CP, 1500–5000 W	70	30	13	60	20	12	30	15	9	15	10	7	5	5	5
Category DP, 5000 W and up	40	25	13	35	20	11	20	15	9	10	7	7	5	5	5

Power preliminary breakdown

Allocate power percentage for each s/s: preliminary mass percentage distribution

	Percentage of subsystem total			
Subsystem	Comsats	Metsats	Planetary	Other
Thermal control	30	48	28	33
Attitude control	28	19	20	11
Power	16	5	10	2
CDS	19	13	17	15
Communications	0	15	23	30
Propulsion	7	0	1	4
Mechanisms	0	0	1	5

Other margins

Computer resources

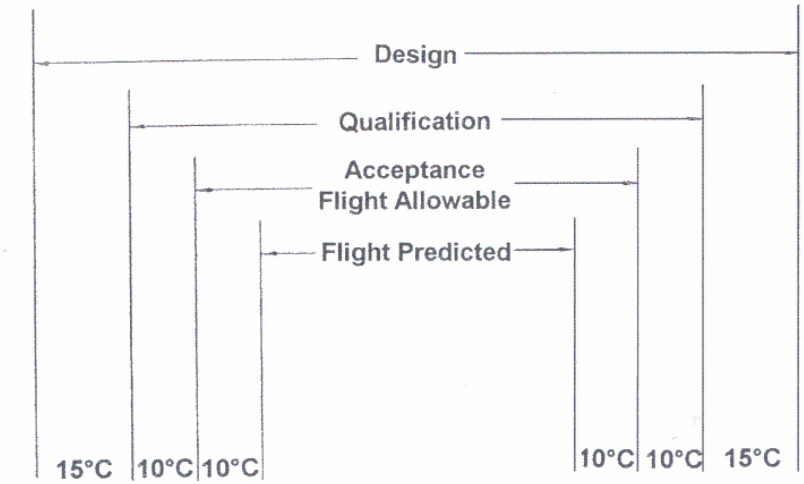
Jet Propulsion Lab suggests:

- At computer selection 400%
- At start of phase C/D 60%
- At launch 20%

Processing time and data bus usage

The computer processing requirement should **not exceed the 50%** of computer capacity at the computer selection

Thermal sizing



e.g. environment [0; 50]°C

→ accepted for [-10; 60]°C

→ Qualified for [-20; 70]°C

→ designed for [-35; 85]°C

System margins

- **10% mass margin at subsystem level** must be considered if the related technology is well known and already space proven
- **20% mass margin at subsystem level** must be considered if the related technology is not well known and already space proven
- **20% mass margin at system level** is strongly recommended in general and is compulsory if a new technology is necessary. 10% must be considered the minimum for a reused system.
- **20% at system level** shall be considered for launcher capabilities: the overall mass shall be at least 20% less than the launcher capability.

Mass Budget example

Example: mass budget

Target Spacecraft Mass at Launch	1250
Total % at Launch	73,6

	Subsystems	Without Margin [kg]	Maturity Level	Margin [%]	Margin [kg]	With Margin [kg]	% of Total
1	Structure	100	*certified*	5	5	**105**	11,4
2	Thermal Control	10	*to be modified*	10	1	**11**	1,2
3	TT&C	10	*to be developed*	20	2	**12**	1,3
4	ADCS	10	*new technology*	25	2,5	**12,5**	1,4
5	EPS	10	...	25	2,5	**12,5**	1,4
6	Propulsion	50	...	25	12,5	**62,5**	6,8
7	Payload	500	...	10	50	**550**	59,8
8	...	0	...	25	0	**0**	0,0
9	...	1	...	25	0,25	**1,25**	0,1

Total		766,75	83,3
System Margin	20	153,35	
Total With Margin		920,1	100,0

Mission Design

- **Satellite Constellation**

  Use of two or more satellites (sometimes a lot of S/C) to satisfy spatial and temporal coverage/observation needs which cannot be met with a single satellite. In an Earth Observation context such missions have applications in disaster monitoring, forest fire detection, ocean sampling, virtual payload synthesis, etc.

- **Satellite Formation Flying (FF)**

  Use of more than one satellite either to enable a mission whose objectives cannot be satisfied by a single satellite (e.g. to synthesise a larger aperture than could sensibly be carried on one platform), or to achieve the mission objectives more cost effectively (e.g. by co-operation between agencies or by taking advantage of low-cost satellite approaches and/or cheaper launchers)

enabled by suitable **orbit and attitude control**

Mission sizing: MA architecture for constellations

- Mission and payload requirements
- Constellation performance criteria
- Constellation orbital design

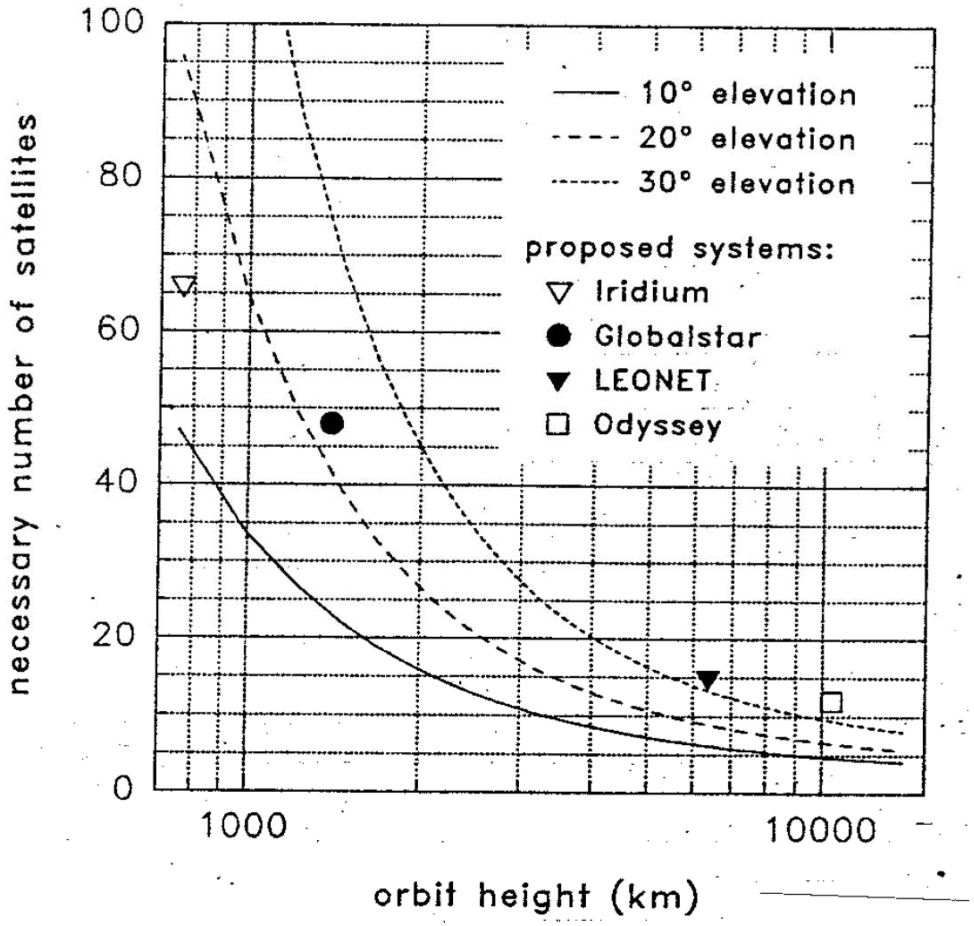
- Constellation launch procedure (launcher, single/multiple launch, direct or parking orbit injection)
- Constellation build-up strategy (global/regional build-up, performance acquisition by plateau, random build-up, ...)
- Constellation "back-up" strategy (spare satellite placed within the constellation, on parking orbit, spare satellite kept on ground, ...)

- Constellation maintenance strategy
- Constellation end-of-life procedure (no end-of-life procedure foreseen, de-orbiting strategy, graveyard orbit, ...)

Constellations design Parameters

Factor	Effect	Selection Criteria
Number of Satellites	Principal determinant of cost and coverage	Minimize number of s/c while fulfilling other criteria
Constellation Pattern	Determines coverage vs. latitude, plateaus	Select for best coverage and spatial sampling performance
Minimum Elevation Angle	Principal determinant of single satellite coverage for a given altitude	Minimum value consistent with payload performance and constellation pattern
Altitude	Coverage, environment, launch and transfer cost; has direct impact on the total s/c number	System level trade of cost vs. performance
Number of Orbit Planes	Determine coverage plateaus, growth and degradation	Minimize for launch and s/c replacement considerations, consistent with coverage needs.
Collision Avoidance Parameters	Key to preventing constellation self-destruction	Maximize the inter-satellite distances at plane crossing
Inclination	Determines latitude distribution of coverage. Combined with altitude, it drives selection of candidate launchers	Compare latitude coverage with launch cost, and fine-tune for collision avoidance
Between Plane Phasing	Determines coverage uniformity	Select best coverage among discrete phasing option, and fine-tune for collision avoidance, if needed
Eccentricity	Mission complexity and coverage vs. cost	Normally zero; non-zero may reduce the number of satellites needed
Size of Station-keeping Box	Coverage overlap needed; cross-track pointing	Determined by mission objectives, perturbations selected to be overcome, and method of control. Minimize consistent with low cost maintenance approach
Lifetime	Depends on space environment. Limited by the on-board resources, and by acceptable orbit degradation vs. available fuel margin	Select for mission fulfilment and required fuel allocation for orbit maintenance
End-of-life Strategy	Elimination of orbital debris, planetary protection	Any mechanism that allows to solve the most important of the two aspects, mission-wise

Sizing a constellation: # Sats for Earth global converage



Main design parameters:

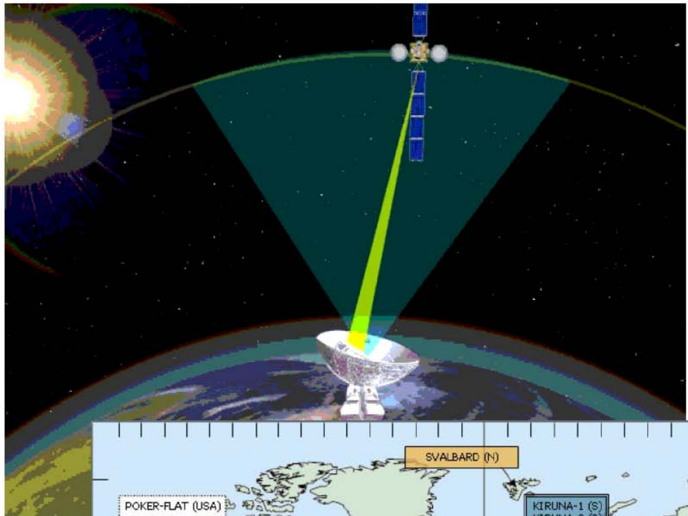
- **Orbit altitude**
  - LEO
  - MEO
  - GEO
- **Minimum elevation angle** for appropriate link set-up
  - ~10º for telecommunications
  - ~5º for ground station coverage
  - ~5º ÷ 40º for navigation or Earth observation purposes

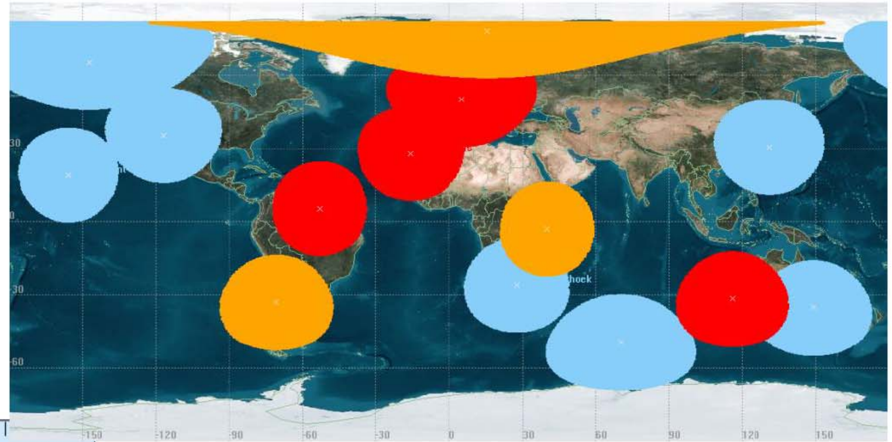
CRITERIA to trade a EO constellation design

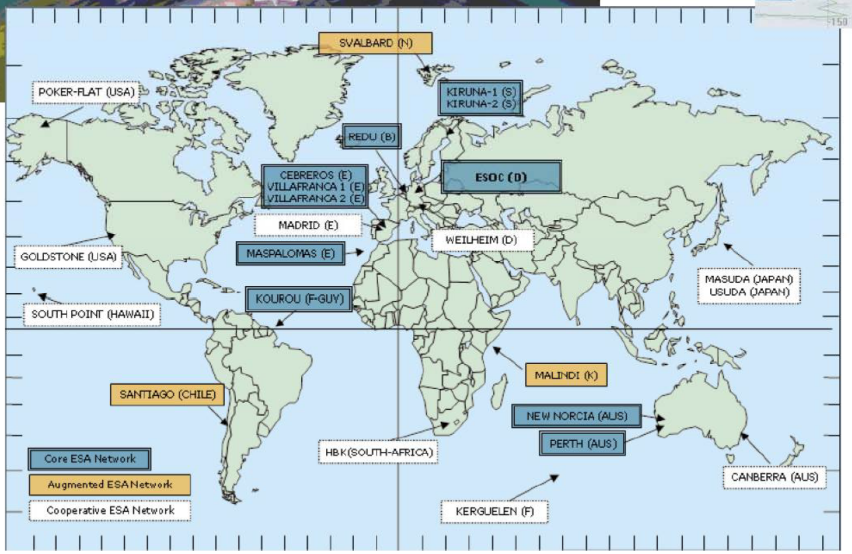
- Most common **"Figures of Merit" for Earth Observation Constellations**:
  - Max/Min/Average coverage percentage at any grid point and globally over a zone
  - Max/Min/Average revisit time at any grid point and globally over a zone
  - Instrument duty cycle (data acquisition time per orbit)
  - Zone, DRS and ground stations visibility for different instruments or antennas
  - Data timeliness/latency (time interval from data acquisition by the instrument to the delivery as data product at the user segment interface)
  - Max/Min/Average response time at any grid point and globally over a zone
  - Illumination conditions (eclipse analysis) and Sun geometry (limitations can be imposed on the basis of the Sun position and the min/max "solar β angle", i.e. the angle of the vector to the Sun relative to the satellite orbital plane)

- The instrument and payload features and modes of operation are the driving factors when analysing the performance of an Earth observation satellite system

CRITERIA to trade a EO constellation design





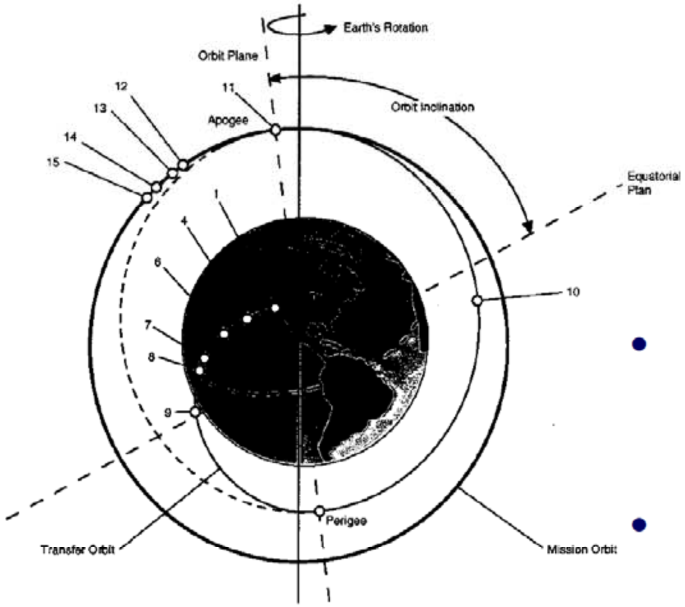


ESA Ground Station Networks

- ESA ESTRACK (red) + ESA Augmented Network (orange) + ESA Cooperative Network (light blue)
- Identify adequate network in terms of GS geographical locations to fulfil GS visibility and data timeliness performance requirements, and to guarantee efficient data flows

CRITERIA to trade a EO constellation design

Orbit Injection and Transfer Strategies



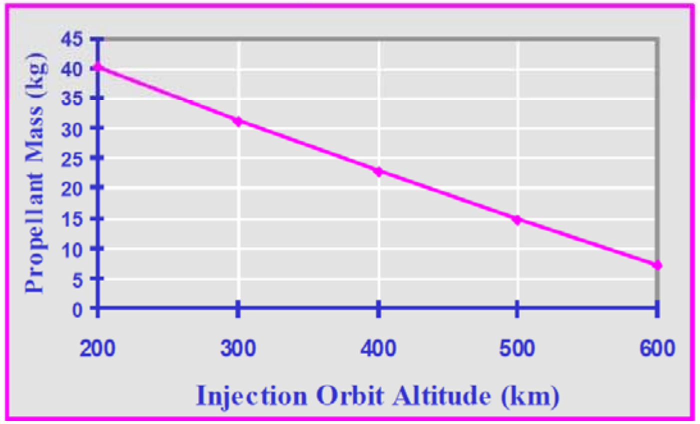
Satellite Orbit Injection Techniques

- **Direct injection into the final orbit**
  - The launcher may require an upper stage, adding to the launch cost
  - Quick deployment, so as to start early system operations
  - Feasibility depends on final orbit altitude and inclination
- **The s/c performs propulsive manoeuvres to reach the final orbit** from an initial injection orbit
  - additional propellant and structure
- **Indirect injection** to populate several operational orbital planes with a launch: uses the differential effect of the Earth oblateness on the node
  - to launch several small s/c using a launcher with considerable injection capability
  - No quick deployment, due to the required drift time

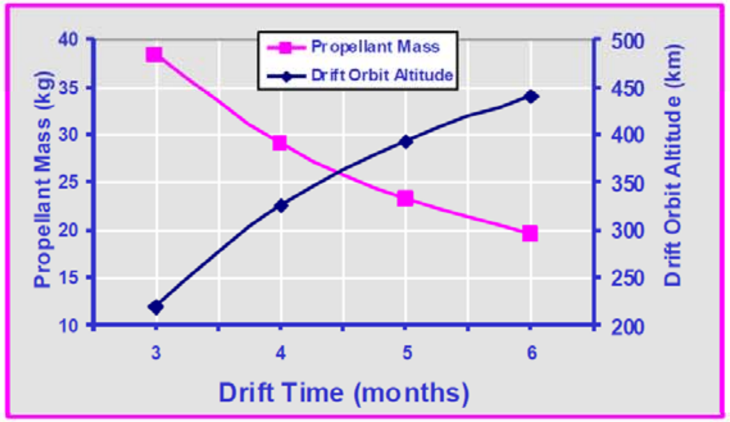
CRITERIA to trade a EO constellation design

Orbit Injection and Transfer Strategies

Propellant Mass for the Impulsive Transfer



Indirect Injection



Rocket equation:

$$M_p = M_f \left[e^{(\Delta V / Isp \cdot g)} - 1\right] = M_0 \left[1 - e^{-(\Delta V / Isp \cdot g)}\right]$$

Orbital plane drift due to J2:

$$\dot{\Omega}_{J_2} = -\frac{3}{2} \cdot J_2 \cdot \sqrt{\frac{\mu}{a^3}} \cdot \left[\frac{Re}{a \cdot (1 - e^2)}\right]^2 \cdot \cos i$$

Constellation maneuvers sizing

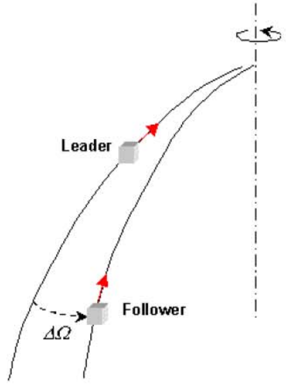
- **Constellation Maintenance**
  - Absolute orbit control: applied to each satellite independently w.r.t. its ref. trajectory
  - Relative orbit control: to guarantee the global geometry of the constellation
- **Collision Avoidance**
  - Avoidance manoeuvres between operational S/C and catalogued debris objects
  - Avoidance manoeuvres between two satellites of the constellation in case of failures or unforeseen events that might trigger a non-negligible collision risk
- **Constellation End-Of-Life Disposal**
  - If necessary, disposal strategy of a LEO S/C shall foresee a manoeuvre to lower the orbit perigee to an altitude that guarantees safe uncontrolled decay within 25 years
  - Of particular concern with regards to LEO constellations is the possibility of a collision having a domino effect and wiping out all of the satellites within a particular orbit band

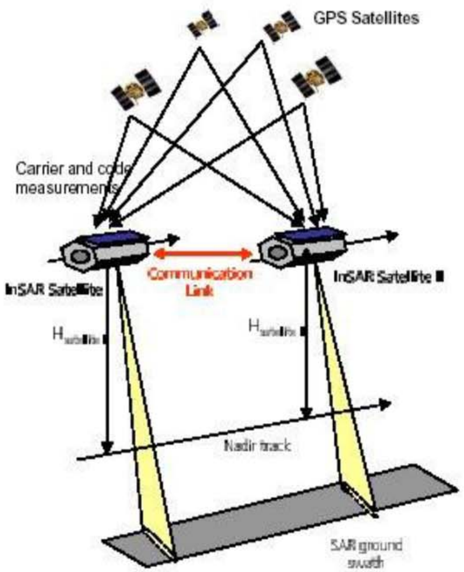
Formation Flying - Design

- **FF Definition and Main Properties**
  - The mission consists of 2 or more spacecraft
  - The spacecraft states are directly coupled such that changing the state of one satellite affects the state of all others
  - The relative position and velocity between the satellites are controlled, and possibly also the relative attitudes
  - The satellites are moving on quasi coplanar orbits or perhaps Lagrange points
  - The spacecraft are in close proximity, which means typically below a few-km separation where the relative motion is in a linear domain (though some FF mission concepts foresee rather large distances)
  - A plane is defined for the inter-spacecraft positions with an arbitrary orientation in space and with respect to a possible local orbital frame. Spacecraft do not all have to be in that plane in their nominal position
  - Guidance Navigation and Control (GNC) requirements are typically high to very high

Formation Flying - Design

- **Ground-track-oriented FF missions**: objective is to achieve a relative position between ground tracks of FF satellites (typically, same ground track), so that they observe a same area with possibly some time delay between the observations
  - Controlling ground track of the S/C composing the FF relatively to each other
  - An overlapping ground track or slightly shifted ground tracks are obtained by flying the spacecraft on the same orbit with a shift in argument of latitude

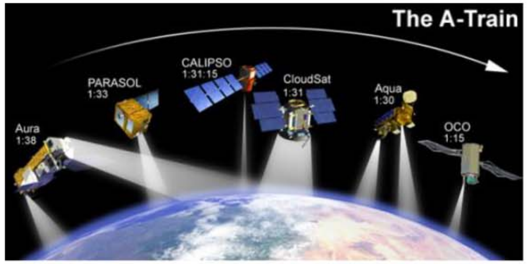




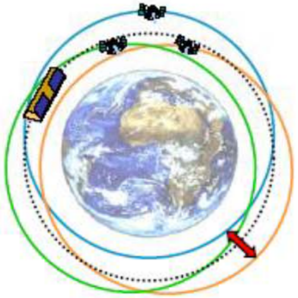
  - Depending on the value of this shift in argument of latitude – along-track separations can range from a few kilometres to thousands of kilometres – a shift in right ascension of ascending node is also introduced in order to compensate for the Earth rotation

Formation Flying - Design

- **Geometry-oriented FF missions**: objective is to compose an instrument distributed on several S/C ⇒ S/C have to maintain a given geometry (1D, 2D, 3D)
  - Geometry can be a fixed segment obtained by having the S/C on a same orbit with an along-track shift



  - Triangle varying at the orbital period can be obtained by introducing a shift in argument of latitude and in right ascension of ascending node
  - S/C evolving on a circle or an ellipse around a reference orbit can compose a geometric figure. This ellipse is obtained by introducing shifts in cross-track and radial direction and by properly phasing obtained relative oscillations at orbital period

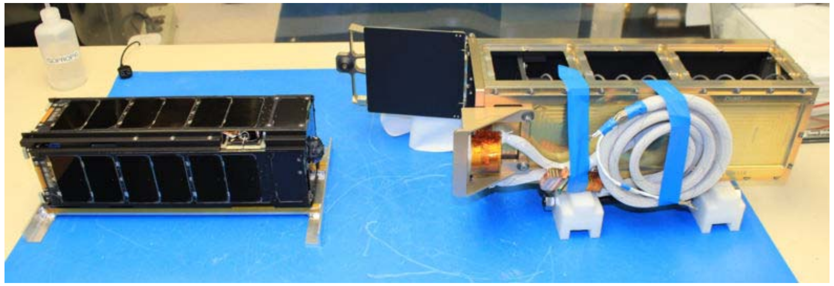


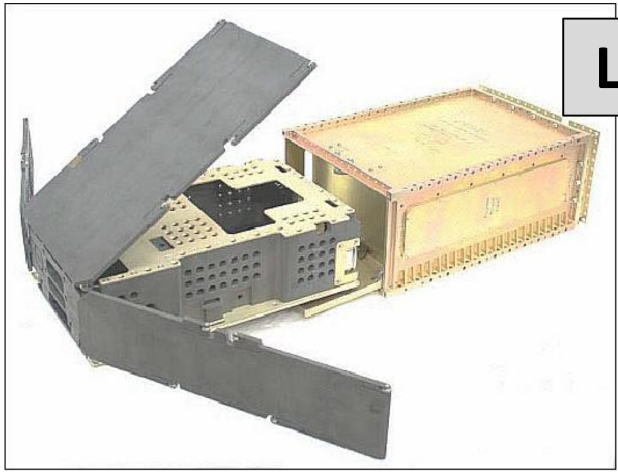
- Design driver of these FF geometries being always to stay as close as possible to a natural evolution of the formation, so as to minimise control manoeuvres to be implemented during the mission lifetime

Formation Flying maneuver sizing

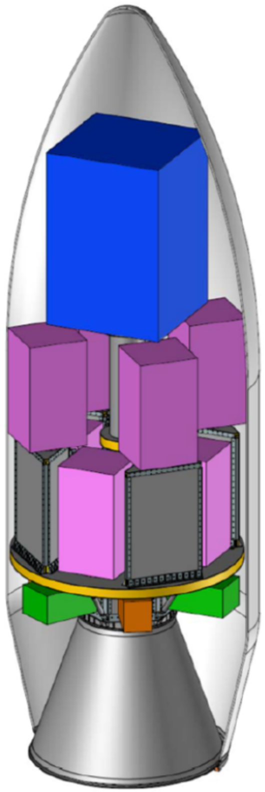
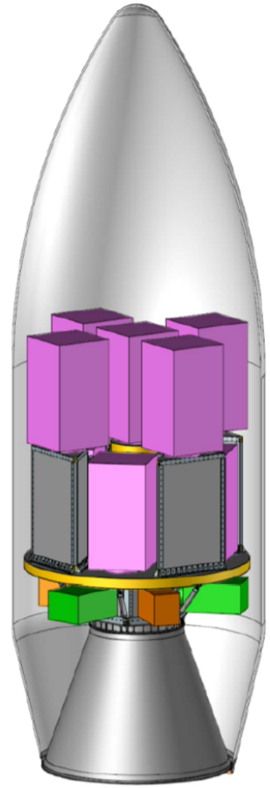
- **Formation Acquisition**
  - Distribute the S/C in space at the beginning of the mission, so that they can achieve the nominal FF configuration to perform the observations foreseen
  - Correct FF initialisation errors (initial position and velocity dispersions)
- **Formation Reconfigurations and Recovery (i.e., FF re-initialisation)**
  - Modify FF spatial configuration (inter-satellite distances and/or angles), so as to tune characteristic observation scale (e.g. interferometric baseline) or measurement type
  - Recover unforeseen perturbation events and re-initialise the FF configuration following an anomaly affecting one or more FF satellites
- **Formation Maintenance**
  - Maintain the FF configurations within given control deadbands in terms of relative satellite positions, distances, angles, etc. (tight vs. loose FF control)
- **Formation Attitude Control**
  - Needed to comply with stringent pointing requirements, for instrument FOV co-registration, simultaneity of measurements, interferometry, etc.
- **Formation End-Of-Life Disposal**
  - If necessary, disposal strategy of a LEO S/C shall foresee a manoeuvre to lower the orbit perigee to an altitude that guarantees safe uncontrolled decay within 25 years

Constellation\FF launch configurations





Launcher's IF

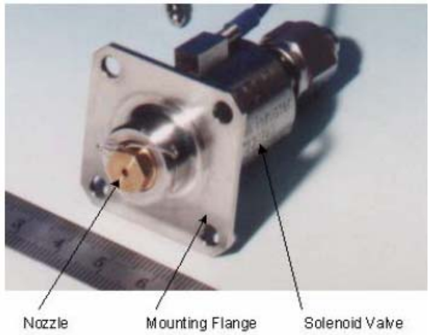
SSMS Stack conf#1	SSMS Stack conf#2
FLEXI 4 configuration with Mini and Micro S/C	FLEXI 4 configuration with Micro S/C
One (1) Mini satellite on a stretched central column Four (4) Micro satellites on Deck#1 on four tower modules Four (4) Micro satellites on Deck#2 Three (3) Nano satellites on Base module Three (3) Cubesats deployers on Base module	One (1) Micro satellite on the central column Four (4) Micro satellites on Deck#1 on four tower modules Four (4) Micro satellites on Deck#2 Three (3) Nano satellites on Base module Three (3) Cubesats deployers on Base module
	

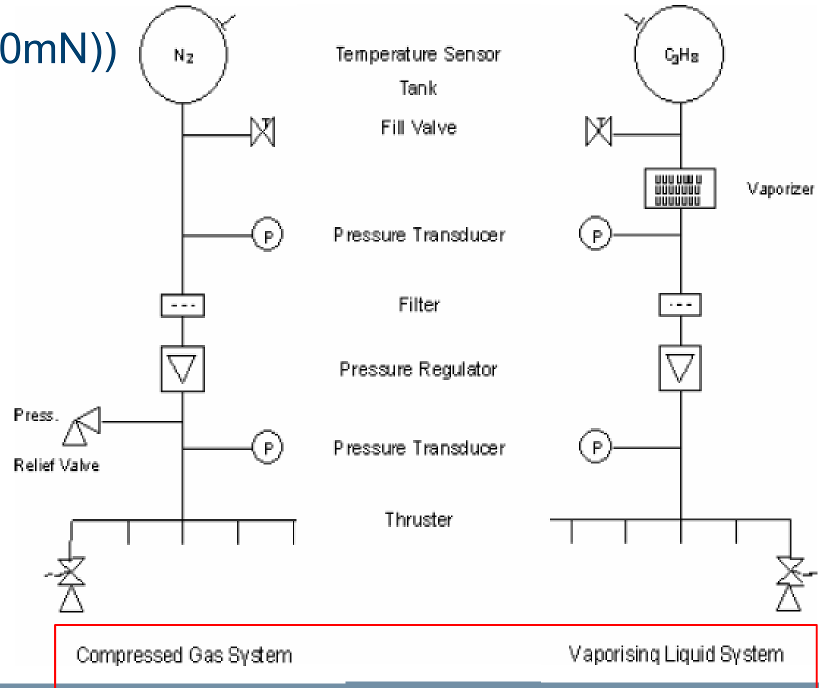
Propulsion s\s

Propulsion system types: Chemical

Cold gas:

- the simplest the cheapest (O(2*10^3) €)
- Propellants are compressed inertial gases (N_2) - at high storage pressure - or high vapour pressure hydrocarbons (Propane C_3H_8)
- No heating, kinetic energy depends only on the storage pressure
- very low impulse (~40-60s) and very low thrust (O(10mN))
- Multiple starts
- Pulsing
- No throttling
- **Applications**:
- Attitude control
- Fine control
- Nano-platforms

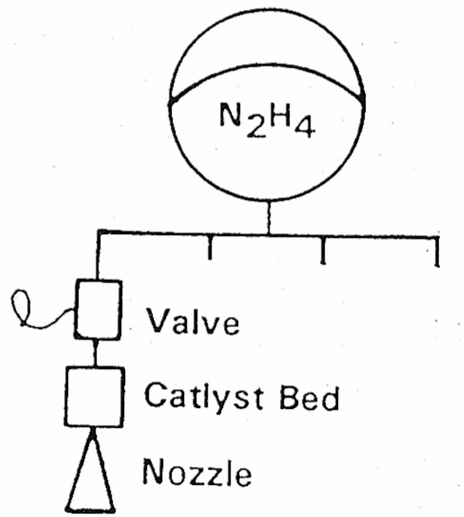




Propulsion system types: Chemical

Engine	Manufacturer	Vacuum Thrust [N]	Vacuum Specific impulse [s]	Cycle life [Cycles]	Engine mass [kg]	Inlet pressure [bar]	Input power [Watt]	Voltage range [volt]	Envelope[4] [mm] (LxD)
	Bradford	55e-3[3]	77						
VP-03-001	AMPAC-ISP	0.001	>70		< 0.300	1		-20 to +150	87 x 16 x 91
58-125	Moog	0.0045	65		0.00734		2.4		
	Marotta	0.05			0.07	6.9	<1		
CGT1	DASA	0.02	67		0.120/	7.0			64 (L)
	Sterer	1[2]	68	250,000	0.174	3.5	5-6	24-32	66 x 31
58-102	Moog	1.11		10,000	0.015	8.8-6.3	30	24-32	24.7 x 14.5
58-112	Moog	1.11		10,000	0.015	7.4-4.9	30	24-32	24.7 x 14.5
58-115	Moog	2.89			0.013		30		
58-113	Moog	3.33		10,000	0.015	8.8-6.3	30	24-32	24.7 x 14.5
58-103	Moog	5.55		10,000	0.015	8.8-6.3	30	24-32	24.7 x 14.5
50-673	Moog	44.5		5,000	0.231	10.5-4.9	6-12	24-32	87 x 80 x 64[1]
50-820	Moog	52					6-12		
58-126	Moog	266		10,000	0.181	10.5-4.9	30	24-32	70 x 63(e)

Propulsion system types: Chemical monopropellant



Monopropellant gas:

- moderately cheap (O(8*10^3) €)
- specific impulse (~200-300s)
- Large thrust range: (O(10^2) N)
- Multiple starts
- Pulsing
- Throttling
- Moderated lifetime (>12y, limited by the catalyst bed lifetime)
- Fuel: hydrazine (N_2H_4), stored as liquid (melting point 2°C;boiling point 114°C) exhaust gases are corrosive
- Heating may increase the thruster effectiveness

Propulsion system types: Chemical monopropellant

Performance Characteristics

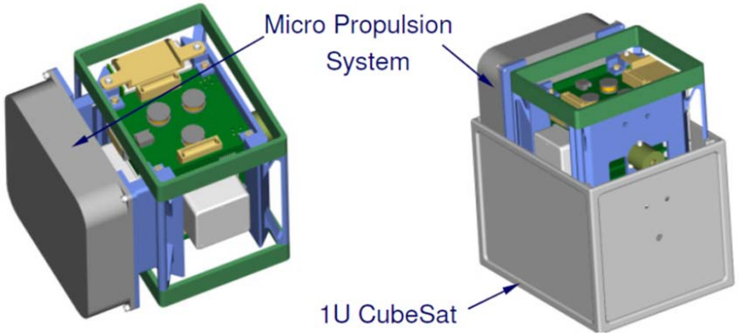
Engine	MONARC-1	MONARC-5	MONARC-22-6	MONARC-22-12	MONARC-90LT	MONARC-90HT	MONARC-445
Steady State Thrust	0.22 lbf (1N) @275 psia	1.0 lbf (4.5 N) @325 psia	5 lbf (22N) @275 psia	5 lbf (22N) @190 psia	20 lbf (90 N) @ 235 psia	26 lbf (116 N) @ 235 psia	100 lbf (445N) @ 275 psia
Feed Pressure	70 – 400 psia (4.8 – 27.6 bar)	80 – 420 psia (5.5 – 29.0 bar)	70 – 400 psia (4.8 – 27.6 bar)	70 – 400 psia (4.8 – 27.6 bar)	80 – 400 psia (5.5 – 27.6 bar)	80 – 370 psia (5.5 -25.5 bar)	70 – 400 psia (4.8-27.6 bar)
Nozzle Expansion	57:1	135:1	60:1	40:1	40:1	50:1	50:1
Valve Power	18 watts	18 watts	30 watts	30 watts	72 watts	72 watts	58 watts
Mass	0.83 lbm (0.38 kg)	1.08 lbm (0.49 kg)	1.58 lbm (0.72 kg)	1.51 lbm (0.69 kg)	2.47 lbm (1.12 kg)	2.47 lbm (1.12 kg)	3.5 lbm (1.6 kg)
Engine Length/Exit Diam	5.2 in (13.3 cm) / .2 in (0.5 cm)	9.4 in (41.8 cm) /1 in (2.5 cm)	8 in (20.3 cm) / 1.5 in (3.8 cm)	9 in (22.9 cm) / 1.2 in (5.3 cm)	12 in (30 cm) / 3.3 in (8.4 cm)	12 in (30 cm) / 3.3 in (8.4 cm)	16 in (41 cm) / 5.8 in (14.8 cm)
Specific Impulse	227.5 sec	226.1 secs	229.5 secs	228.1 secs	232.1 secs	234.0 secs	234.0 secs
Minimum Impulse Bit	0.0006 lbf-sec (2.6 mN-sec)	0.0007 lbf-sec (3.1 mN-sec)	0.07 lbf-sec (312m N-sec)	0.12 lbf-sec (526m N-sec)	0.04 lbf-sec (1.8 N-sec)	0.26 lbf-sec (1.16 N-sec)	2.59 lbf-sec (11.52 N-sec)
Total Impulse	25,000 lbf-sec (111,250 N-sec)	138,000 lbf-sec (613,852 N-sec)	120,000 lbf-sec (533,784 N-sec)	263,720 lbf-sec (1,173,085 N-sec)	786,000 (3,500,000 N-sec)	459,100 lbf-sec (2,042,178 N-sec)	1,250,000 lbf-sec (5,600,000 N-sec)
Pulses	375,000	205,000	230,000	160,000	50,000	70,000	12,000

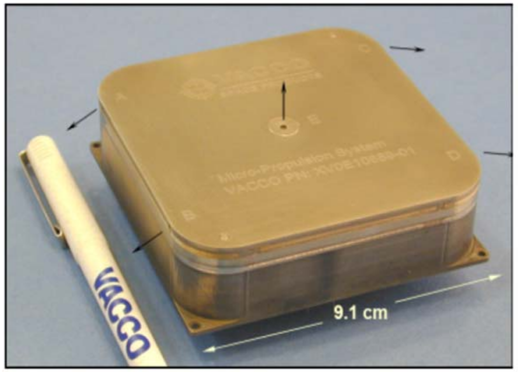
http://www.moog.com/

Propulsion system types: Chemical monopropellant

CubeSat Propulsion System	Size (U)	End/Cener Mount	Propellants	Thruster type	Thrusters	Nomina Thrust (mN)
PUC	0.14U-1U	End	R236FA/ SO2	Warm Gas	1	5.4
CPOD	1U	Center	R134a/ R236FA	Cold Gas	8	25
MarCO	2U	End	R236FA	Cold Gas	8	50
Green Mono Prop System	0.5-1U+	End	ADN/AF-M315E	Mono-Prop	4	400
End mounted standard	0.25-1U	End	R134a/ R236FA	Cold Gas	5	10
Hybrid Green Monoprop	0.5-1U+	End	ADN/AF-M315E	Mono-Prop	1 Hot, 4 Cold	100
Standard	0.3-1U	End	R134a/ R236FA	Cold Gas	5	10
MEPSI	0.25U	End	Isobutane	Cold Gas	5	53
Palomar	1U	Center	Isobutane	Cold Gas	8	35

Nano-sats VACCO propulsion





http://www.cubesat-propulsion.com/vacco-systems/

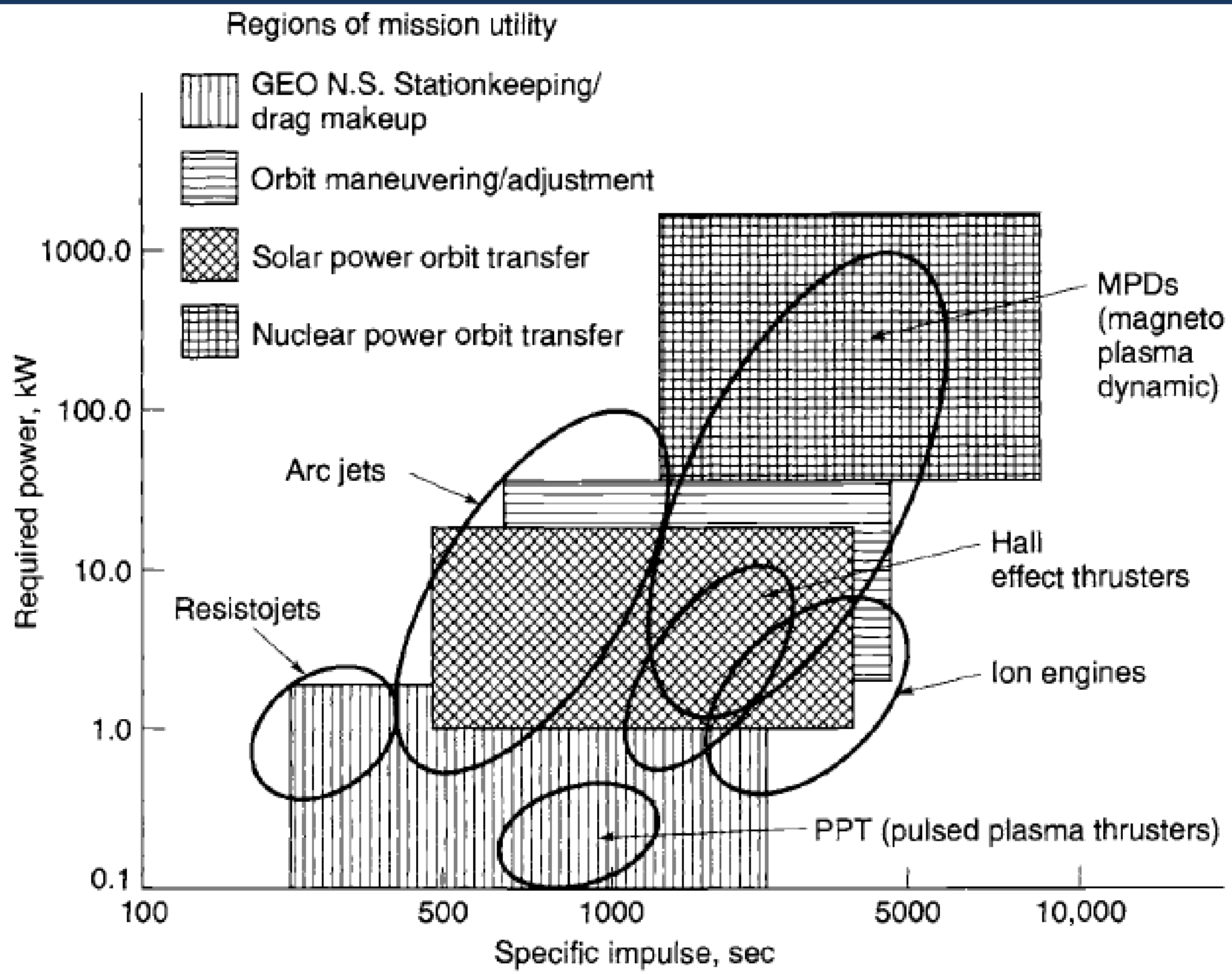
TAXONOMY

- **Electrothermal (T/W < $1E^{-3}$)**
- Type: resistojet, arcjet
- Principle: thermal\mechanical energy exchange

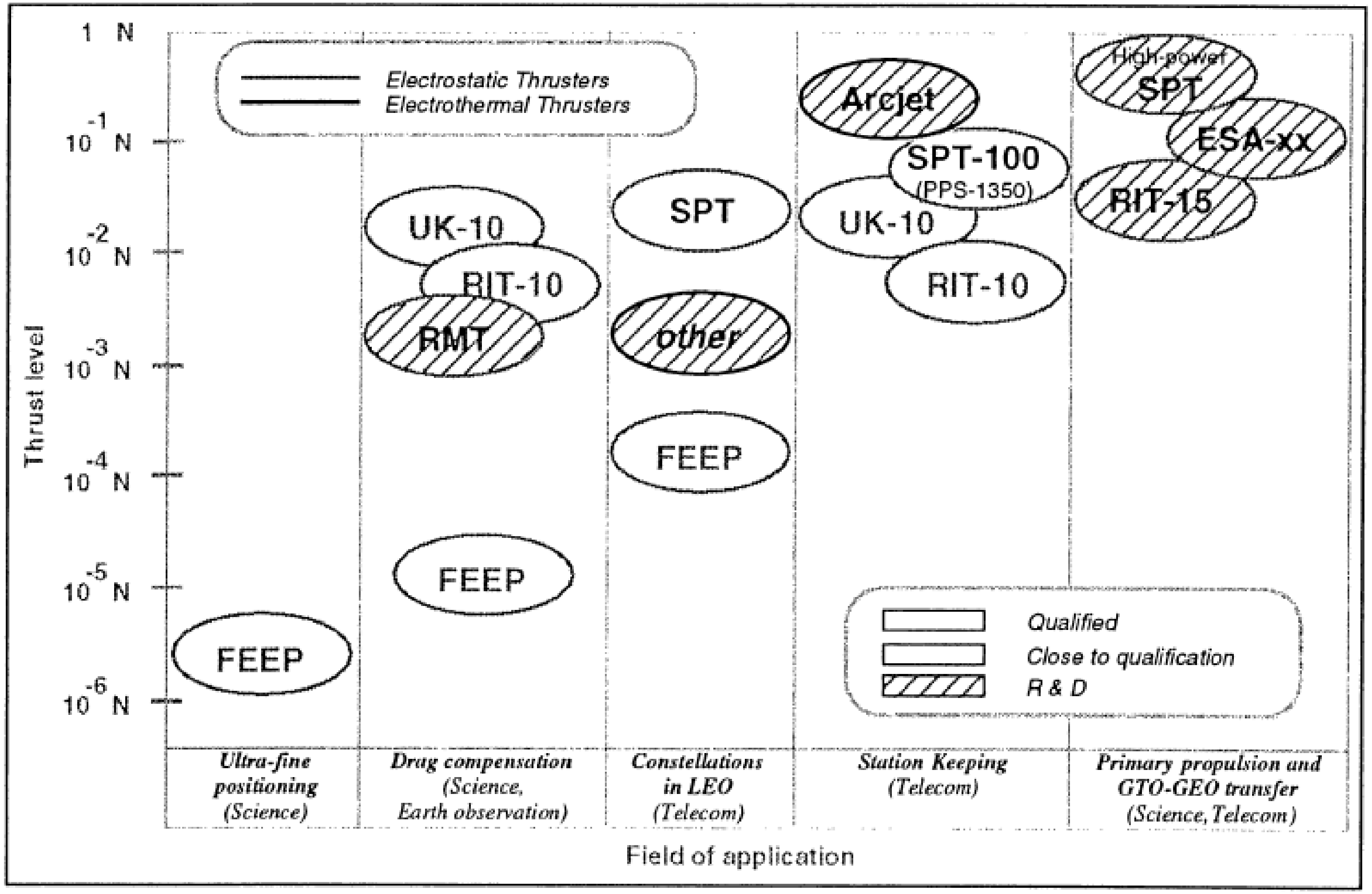
- **Electrostatic (T/W < $1E^{-4}$)**
- Type: Gridded EP, Field Emission EP
- Principle: electric\mechanical energy exchange

- **Electromagnetic (T/W < $1E^{-4}$-$1E^{-6}$)**
- Type: Hall (magneto static), pulsed plasma (PPT)
- Principle: magnetic\mechanical energy exchange

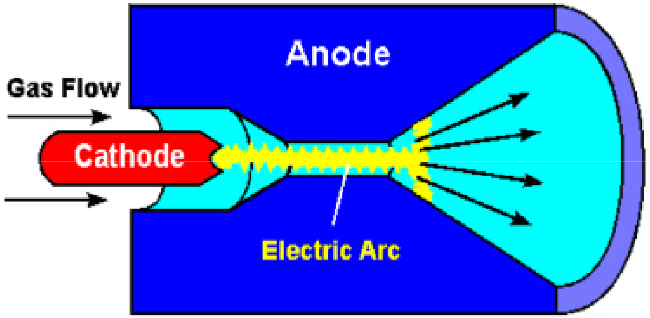
Electric Propulsion

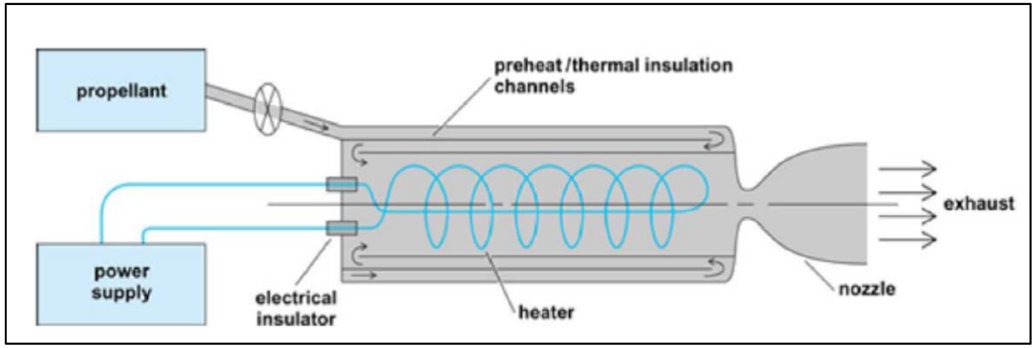


European electric Propulsion: fields of application



Electric Propulsion: electrothermal





Arcjet

heat propellant using an <u>electric arc</u> generated between the anode and the cathode (η<50%)

- low thrust level 0.1-0.3 N
- **High** specific impulse 500-1500s
- good for **SK**

Resistojet

heat propellant by passing it on an <u>electrically heated</u> surface(η=65-85%)

- low thrust level 0.2-0.3 N
- **low** specific impulse 100-400s
- good for **attitude control** system

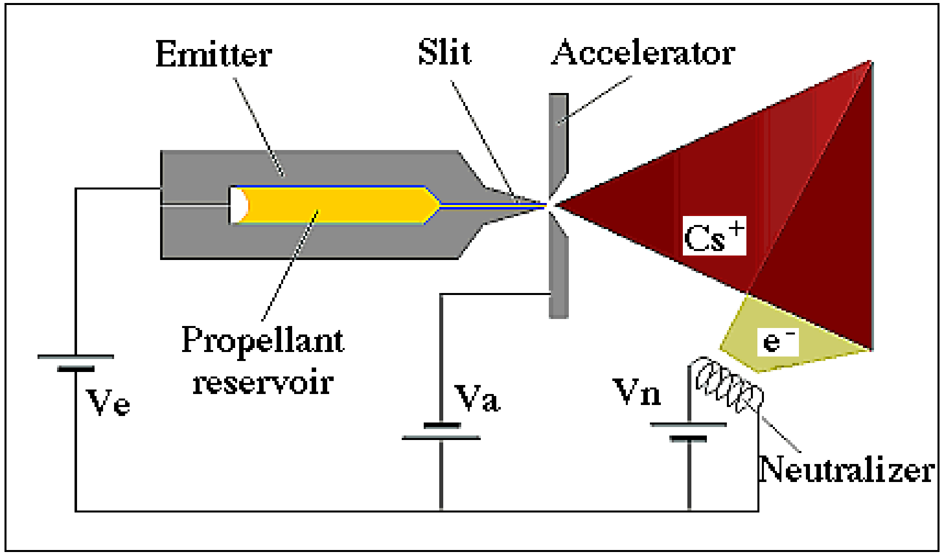
Propellant: Hydrazine, Nitrogen, Ammonia (low mass; High specific heat preferred)

Electric Propulsion: Gridded Ion Thrusters

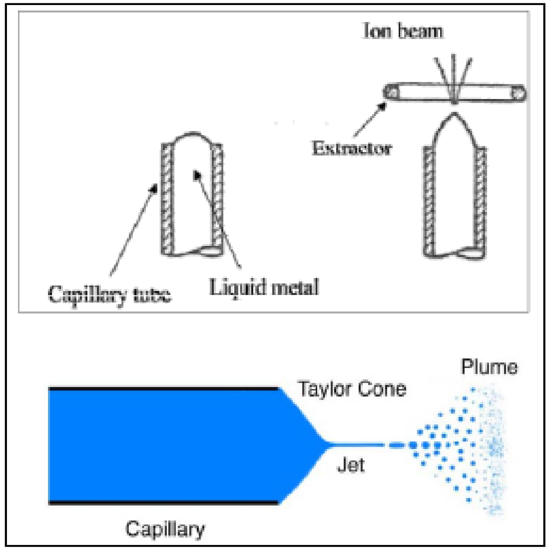
Radiofrequency Ionised Thruster RIT (ASL)

	RIT µX	RIT 10 EVO	RIT 2X
Thrust & Power			
Nominal Thrust	50 - 500 µN	5 mN | 15 mN | 25 mN	80 mN | 115 mN | 168 mN | 200 mN
nom. Power	< 50 W	145 W | 435 W | 760 W	2185 W | 2985 W | 4650 W | 5785 W
Functional Performance			
extended / on request	10-100 µN, 300 - 3000 µN		
Isp	300 - 3000s	> 1900s | > 3000s | > 3200s	> 3400s | > 3434s | > 4000s | > 4300s
max. demonstrated	> 3500s	> 3400s	> 6000s (RIT 22)
Divergance angle*	< 17°	< 15°	< 25°
Lifetime			
Total Impulse	> 10kNs up to 200kNs	> 1.1 MNs	> 10 MNs
Max Operational cycles	> 10000	> 10000	> 10000
Total Lifetime	> 20000 h	> 20000 h **	> 20000 h
Technology			
Ionisation	RF-Principle	RF-Principle	RF-Principle
Acceleration	Electrostatic	Electrostatic	Electrostatic
Gridsystem	2 Grids	2 Grids	2 Grids
Propellant	Xenon	Xenon	Xenon
Design			
mass	440 g	1.8 kg	8.8 kg
Dimensions			
Diameter	78 mm	186 mm	308 mm
Length	76 mm	134 mm	215 mm

Electric Propulsion: Field Emission Electric Propulsion - FEEP thrusters

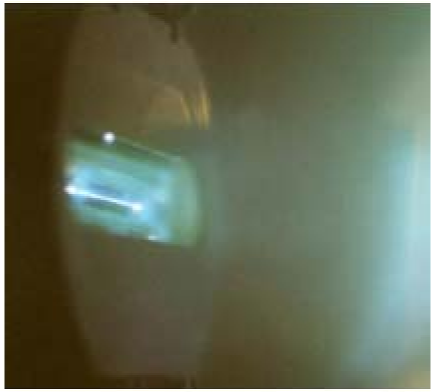


- Ve;Va ~5keV
- T∝L= aperture length
- Propellant: liquid Cs



- Thrust is produced by exhausting a beam of mainly singly-ionized cesium atoms, produced by field evaporation.
- thrust level very low 1-100μN, good for fine attitude control
- thrust to power ratio 16mN/W
- High impulses: 6000-10000s; efficiency 98%

Electric Propulsion: Field Emission Electric Propulsion - FEEP thrusters



Manufacturer	Centrospazio (Italy)	Centrospazio (Italy)	Centrospazio (Italy)	Centrospazio (Italy)
Propellant	Cs	Cs	Cs	Cs
Slit Width (mm)	2	70	5	70
Configuration	Single module	Single module	Cluster of 2 thrusters	Cluster of 4 thrusters
Nominal Thrust (μN) *	40	1,400	2 x 100	4 x 1,400
Isp (sec)	9000	9000	9000	9000
Power (W) **	2.7	93	13	370
Specific Power (W/mN)	66	66	66	66
Max Emitter Voltage (kV)	+5.5	+5.5	+5.5	+5.5
Accelerator Voltage (kV)	-5	-5	-5	-5
Thruster Mass (kg)	0.6	1.2	1	3.2
Thruster or Cluster Size (cm)	8 x 6 x 8	13 x 7 x 9	10 dia. x 10	18 dia. x 15
PPU Mass (kg)	1	1.2	2	5.5
PPU Size (cm)	8 x 12 x 16	8 x 16 x 16	16 x 12 x 16	20 x 25 x 16
Comments	Qualification model	Qualification model	Under development	Under development

* Maximum attainable thrust may be larger by a factor 2.
** Assuming all thrusters operating at nominal thrust.

Pulsed Plasma Thrusters: PPT

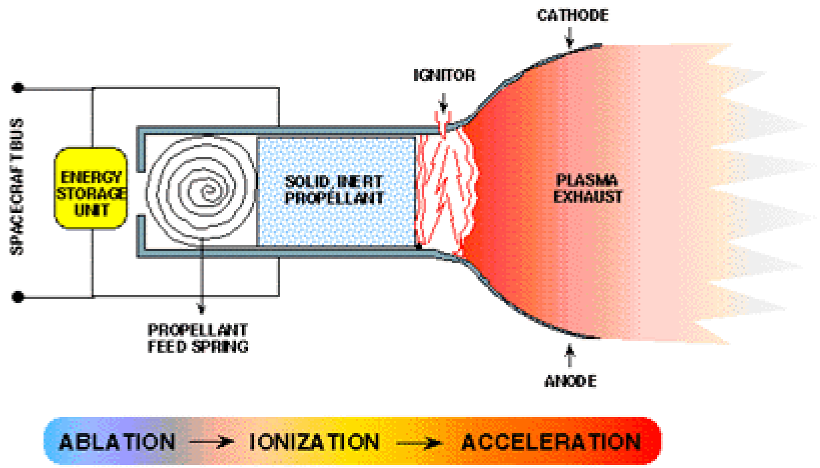
Features: high specific impulse and low power and fuel requirements; pulsing

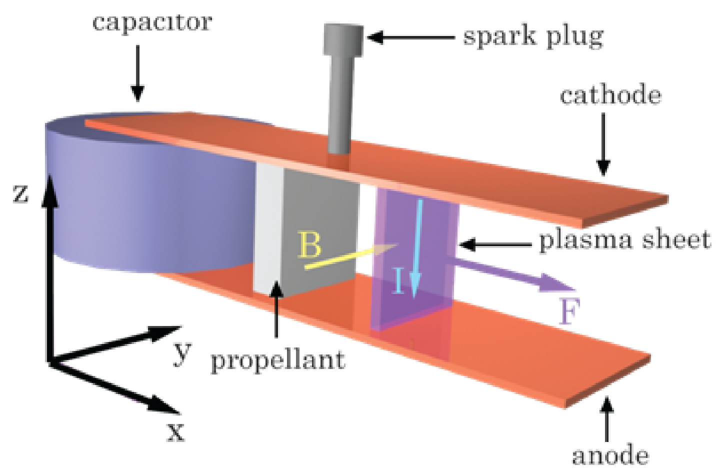
Application: SK maneuvers .

Principle:

Energy is stored in a capacitor; an ignitor shoot electrons between anode and cathode to discharge the capacitor and create and arc; the arc evaporates and ionizes the solid fuel which accelerates out the thruster by Lorentz forces provoked by the induced electromagnetic field.

The capacitor is then charged up again from a power supply and the pulse cycle repeated.

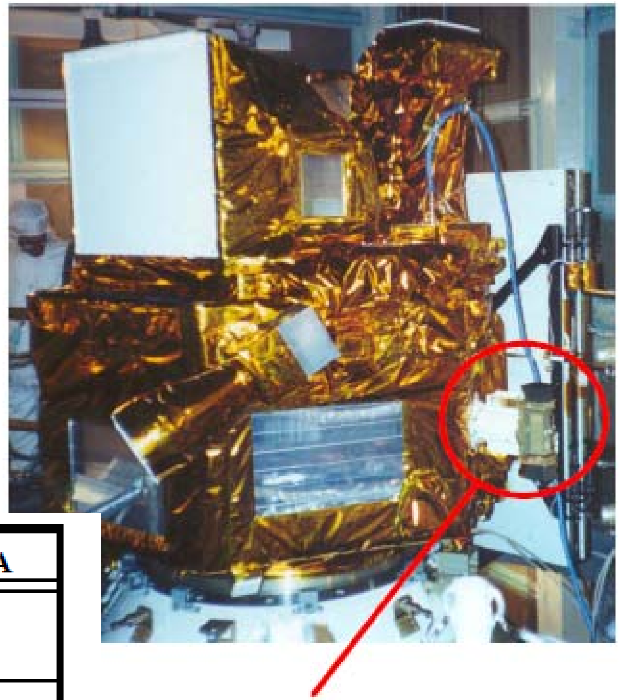




Pulsed Plasma Thrusters: PPT

Features

- Non toxic propellant
- Low power demand (50-70W)
- High specific Impulse 650-1350s
- Very small bits 90-860 μN-s
- Single capacitor→multiple thrusters
- Mass 5-6 kg



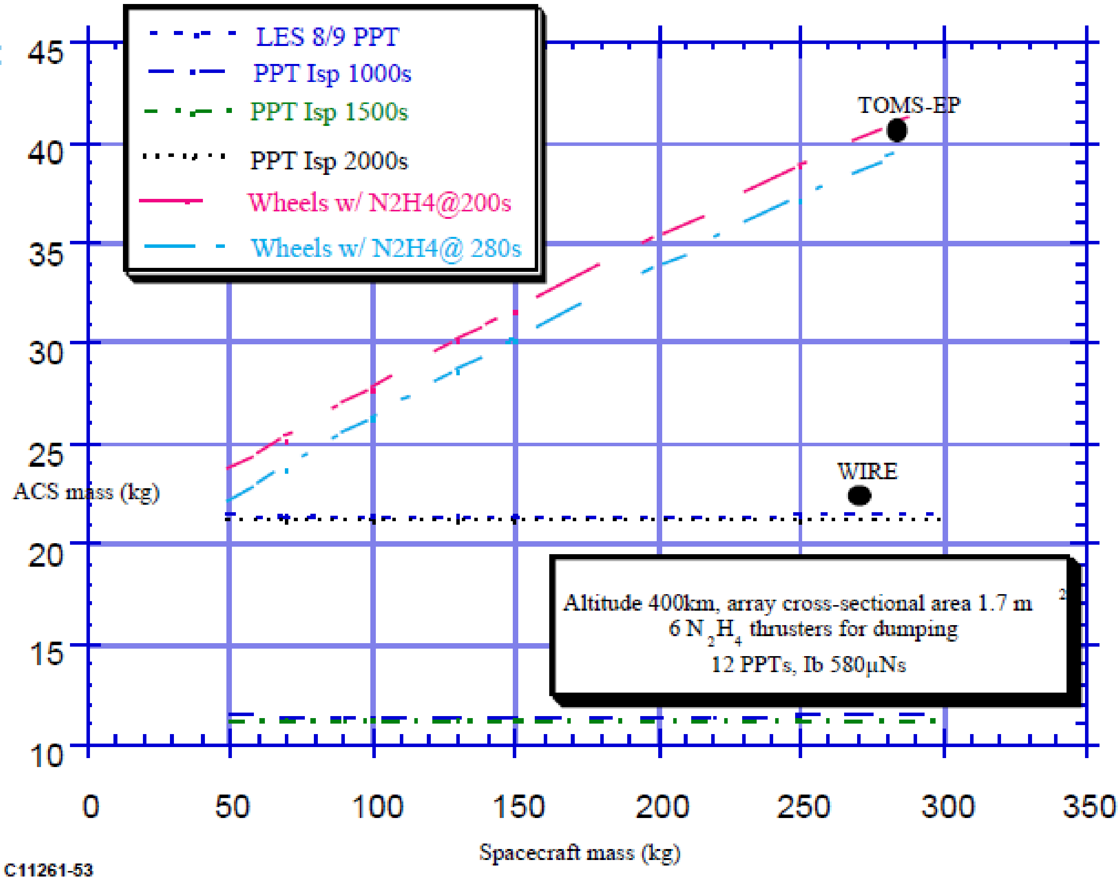
Pulsed Plasma Thruster

Parameter	Unit	LES 6	SMS	LES 8/9	TIP/NOVA
Ibit, (Thrust @ 1 Hz)	μ Newton - second	26.7	111	300	400
Specific Impulse	Seconds	312	505	1000	543
Thrust to Power	μN/Watt	10.6	12.2	12	13.3
Capacitor Energy	Joules	1.85	8.4	20	20
Total Impulse	N-Sec	320	1779	5560	2450
Life	Pulses	12,000,000	13,000,000	18,500,000	10,000,000
Mission		East-West Stationkeeping	Attitude Control	Attitude Control	Orbit Insertion & drag make-up

Pulsed Plasma Thrusters: PPT

- 50 - 300 kg spacecraft
- 400 km circular orbit, 0° inclination
- Disturbance torques per orbit (all N-m):
  - Solar Pressure = 1.9×10^{-6}
  - Aerodynamic = 8.7×10^{-5}
  - Gravity Gradient = 3.9×10^{-7}
  - Magnetic Field = 2.6×10^{-5}

    Total = 1.1×10^{-4}
- 5 year mission life

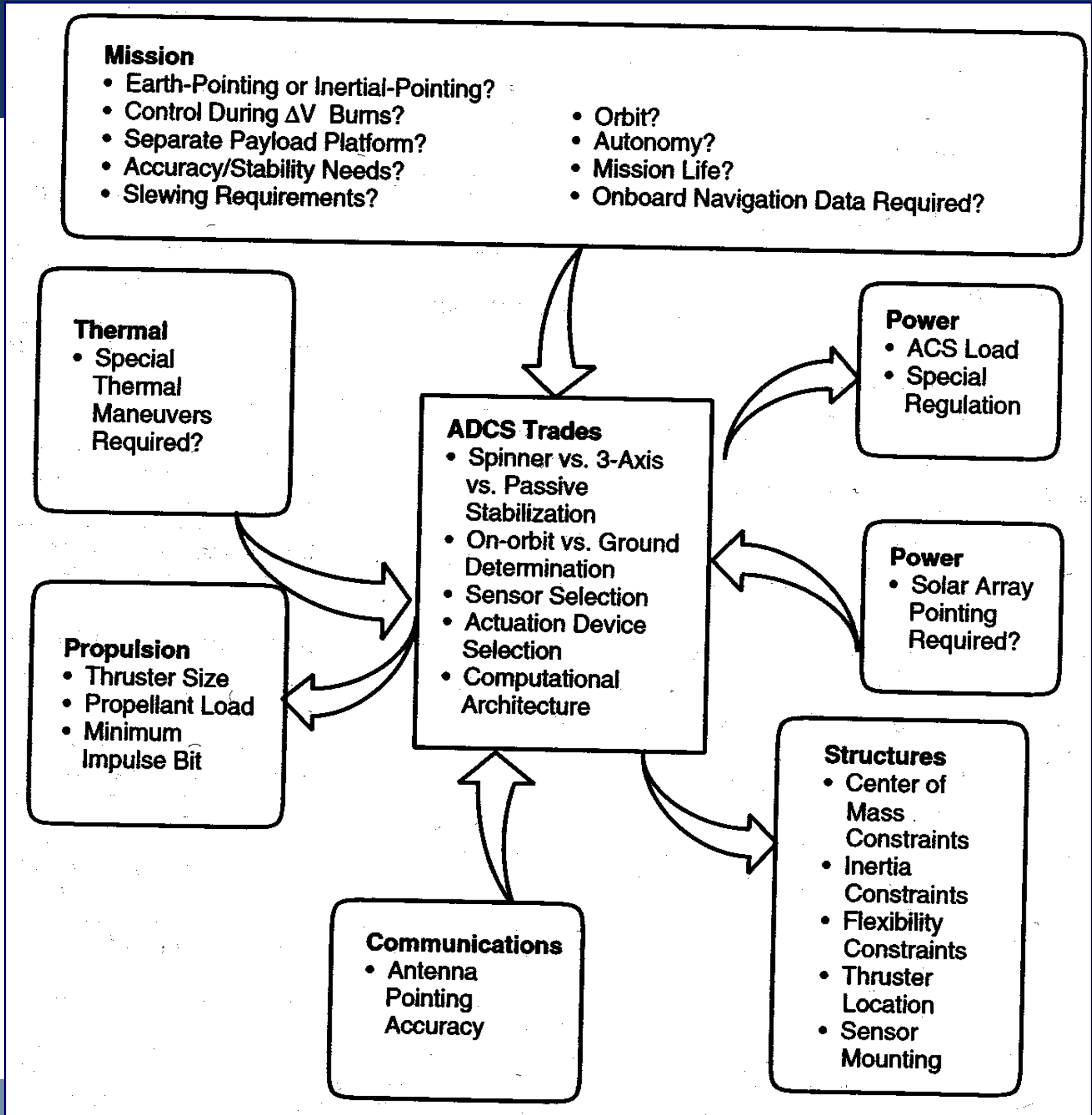


Propulsion solution comparison

Type of Propulsion System	Thrust level [N]	Exhaust Velocity [m/s]	Advantages	Disadvantages
Cold Gas (N_2)	0.0045 - 10	700	Extremely simple, reliable, very low cost	Very low performance, highest mass of all systems
Monopropellant (Hydrazine)	0.5	2 200 – 2 300	Simple, reliable, relatively low cost	Low performance, higher mass than bipropellant
Bi-Propellant (MMH/MON)	4 – 500	2 850 – 3 110	High performance	More complicated system than monopropellant
Solid Propellant	50 – 50 000	2 400 – 3 000	Simple, reliable, low cost	Limited performance, higher thrust
PACT, Hydrazine (Power Augmented Catalytic Thruster)	0.1 – 0.5	3 000	High performance, low power, simple feed system	More complicated interfaces, more power than chemical thrusters, low thrust
ARC-JET (Hydrazine)	0.2	5 000	High performance, simple feed system	High power, complicated interfaces (specially thermal)
Stationary Plasma SPT 100 (Ion Engine)	0.08	16 000	High performance	High power, low thrust, complicated
Kaufman, UK-10 (Ion-Engine)	0.011	30 000	Very high performance	Very high power, low thrust, complicated
Radio-frequency RIT 10 (Ion-Engine)	0.01	31 400	Very high performance	Very high power, low thrust, complicated
Field-Emission	$10^{-5} – 2 \cdot 10^{-3}$	60 000 -100 000	Extreme high performance	Very high power, very low thrust

Attitude Determination and Control

How other S\S influence the ADCS reqs



Attitude control Architectures

Method	Notes
3 axis stabilized	The s\c axes are kept aligned with a reference either **inertial** or **nadir** reference, thanks to gyros periodically updated by star scanning.
Gravity Gradient	Passive method. Effective below 1000 km orbits (for Earth) **Roll and pitch** axes can be controlled **yaw** axis is stabilized by means of a momentum wheel Ipitch>Iroll>Iyaw always stable; Iroll>Iyaw>Ipitch sometime stable
Momentum bias	A momentum wheel spins at nearly constant high speed • It provides inertial **stiffness in two axes** • control of **wheel speed** provides control in the **third axis.**

Attitude control Architectures

Method	Notes
Spin stabilized	The s\c is stabilized along an axis by keeping an angular velocity around it. With no disturbance the angular momentum keeps constant. Perpendicular disturbances make the rotational axis to precess . Parallel disturbances change the angular momentum modulus. **Translational manoeuvres** may occur **only along the spinning axis.**
Dual spin stabilized	A compromise between the three-axis and the spin stabilized solution. The major mass is spun while a platform with p/l or antenna is de-spun.

Attitude control architectures versus reqs

Requirement	Gravity gradient	Spin	Dual spin	Three axis	Momentum bias
Nadir pointing	Yes	No	Poor	OK	OK
Geosynchronous	No	OK	OK	OK	OK
Planetary	No	OK	OK	OK	NO
Thrust vector control	No	Good	Good	OK	NO
Maneuvering	No	Limited	Limited	Good	Poor
Pointing accuracy, deg	5	1	0.1	0.001	0.1 to 3
Relative cost	——	1.00	1.19	2.10	1.45

Disturbance Torques

Source	Type	Influenced primarily by
Gravity gradient	Constant torque for Earth-oriented vehicles, cyclic for inertially oriented vehicles	Spacecraft inertias Orbit altitude (significant below 500 km)
Solar radiation	Cyclic torque for Earth-oriented vehicles, constant for solar-oriented vehicle or platform	Spacecraft geometry and location of center of gravity Spacecraft surface reflectivity
Magnetic Field	Cyclic	Orbit altitude (significant out to GEO) Residual spacecraft magnetic dipole Orbit inclination
Aerodynamic	Constant for Earth-oriented vehicles, variable for inertially oriented vehicles	Orbit altitude (significant out to GEO) Spacecraft geometry and location of center of gravity

ADCS Sizing: actuators

Actuators must have sufficient torque authority to counteract disturbances.
Control Autority: Control Torque-disturbance torque

ex. CT=2DT→ 100% C.A. margin

Actuator	Typical Performance Range	Weight (kg)	Power (W)
Thrusters			
Hot Gas (Hydrazine)	0.5 to 9,000 N*	Variable†	N/A†
Cold Gas	< 5 N*	Variable†	N/A†
Reaction and Momentum Wheels	0.4 to 400 N·m·s for momentum wheels at 1,200 to 5,000 rpm; max torques from 0.01 to 1 N·m	2 to 20	10 to 110
Control Moment Gyros (CMG)	25 to 500 N·m of torque	> 10	90 to 150
Magnetic Torquers	1 to 4,000 $A \cdot m^2$‡	0.4 to 50	0.6 to 16

ADCS Sizing: Reaction Wheels

Characteristic	Mini-wheel	HR 0610	HR 12	HR 14	HR 16	HR 4820	HR 2010	HR 2020	HR 2030	HR 4520
Angular momentum, N-m-s	0.2 to 1.0	4 to 12	12 to 50	20 to 75	75 to 150	65	33.2 to 68.4	27	19.5 to 45.6	60.75
Output torque, N-m	>0.028	0.07 to 5	0.1 to 0.2	0.1 to 0.2	0.1 to 0.2	0.14	0.1	0.13	0.21	0.135
Wheel rpm, ±	9000	6000	6000	6000	6000	6000	6000	6500	6000	5400
Power, W[b]	>6	<15	22	22	22	20	17	35	20	35
Bus voltage, dc	12 to 34	14 to 35	23 to 57	23 to 57	23 to 57	22 to 36	27 to 44	70	27.7 to 31.3	51
Mass, kg	1.3	3.6 to 5.0	7.0	8.5	12	10.2	9.2 to 10.9	7.9	8.9 to 11.2	11.1
Integral electronics	Y	Y	Y	Y	Y	Y	N	Y	Y	Y
Diameter, mm[c]	108	267	316	368	418	405	406	300	305	406
Height, mm	54	12.0	159	159	178	214	235	172	191	215
Op temperature,[d]										
Low	−25	−15	−30	−30	−30	−15	−15	−13	−15	−24
High	+60	+60	70	+70	+70	+71	+70	+75	+80	+61

[a]Reproduced with permission of Honeywell International, Inc.
[b]Power values are steady-state power at maximum wheel speed, W.
[c]Dimensions are overall envelope, mm.
[d]Temperature ranges are qualification limits, operating, °C.

ADCS Sizing: sensors

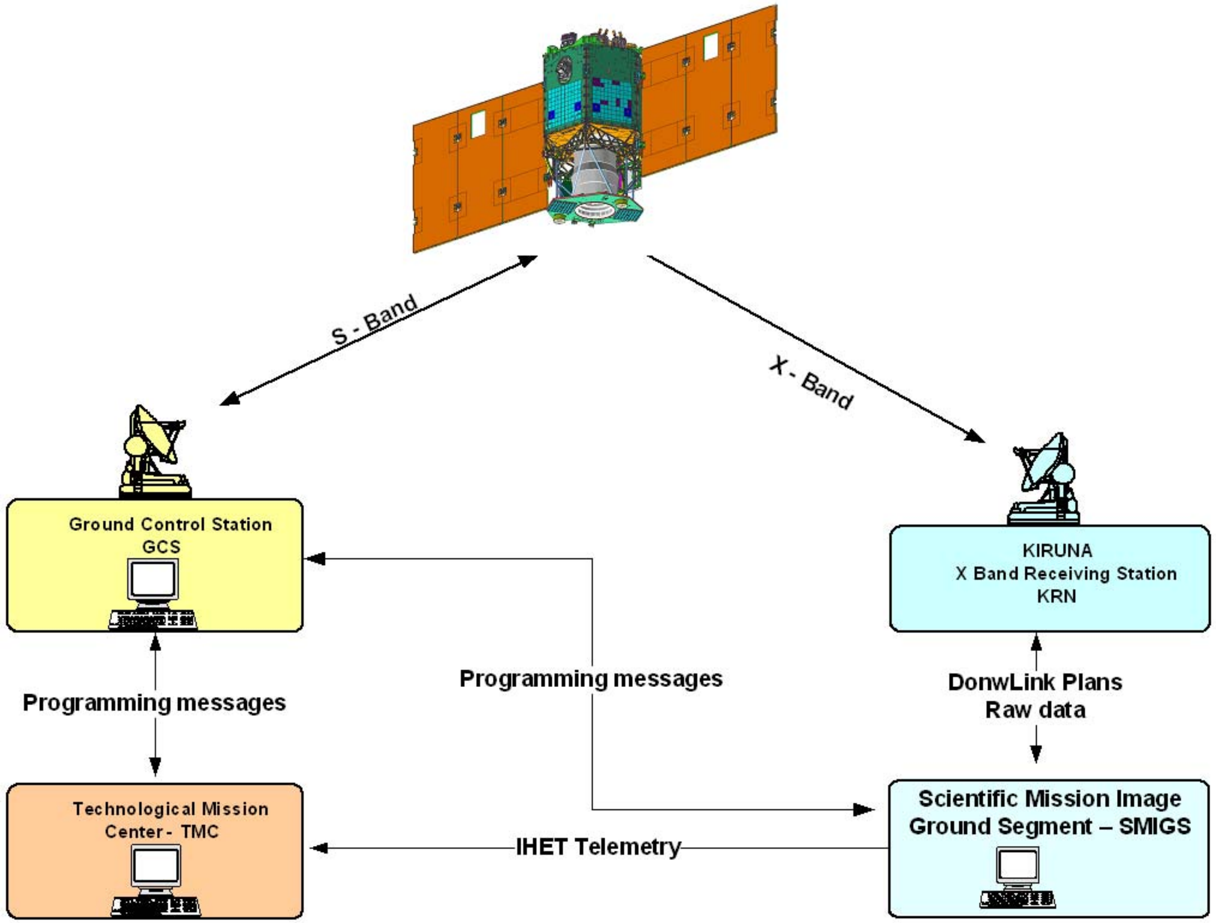
Sensor	Typical Performance Range	Weight (kg)	Power (W)	Comments
Inertial Measurement Unit (gyros and accelerometers)	Gyro Drift Rate = 0.003°/hr to 1°/hr Accel linearity = 1 to 5 x 10^{-6} g/g^2 (over range of 20 to 60 g)	1 to 15	10 to 200	No external inputs required Very high short-term accuracy, but poor long-term accuracy Normally requires periodic updates from other sensors to reset reference
Sun sensors	Accuracy: 0.005° to 3°	0.1 to 2	0.1 to 3	Bright, unambiguous target Target not available at all times due to eclipses
Star sensors (scanners & mappers)	Accuracy: 1 arc sec to 1 arc min (0.0003° to 0.02°)	2 to 5	5 to 20	High accuracy Orbit independent Tends to be heavier and require more power than other sensors
Horizon sensors Scanner/Pipper Fixed Head (static)	 Accuracy: 0.1° to 1.0°(LEO) Accuracy: < 0.1° to 0.25°	 1 to 4 0.5 to 3.5	 5 to 10 0.3 to 5	Bright target that is always available Direct measurements of pitch and roll Limited accuracy due to difficulty finding the Earth's horizon
Magnetometer	Accuracy: 0.5° to 3°	0.3 to 1.2	<1	Cheap, reliable, and light weight Magnetic field uncertainties and variability dominate accuracy. Usable only below ~6,000 km.
GPS	Accuracy: ~0.1°	~5	~15	Requires one receiver and multiple antennas separated appropriately No moving parts Convenient mainly in low Earth orbit

ADCS architectures & HW

Hardware	Gravity gradient	Spin	Dual spin	Three axis	Momentum bias
Sensors	Sun or horizon	Star, horizon, or sun	Gyros, star, or horizon scanner	Precision gyros, sun sensor, star tracker, or horizon sensor	Sun sensors, horizon sensor
Control	Control electronics	Control electronics, damper	Control electronics, damper, programmable computers, I/O and software	Control electronics, programmable computers, I/O and software	Control electronics, programmable computer
Torquers	Boom, momentum wheel	Thrusters	Thrusters	Thrusters, reaction wheels, magnetic torquers	Momentum wheel, thrusters
Mechanisms	None	Dampers	Despin drive, dampers, slip rings	Antenna pointing, solar array pointing	Antenna pointing, solar array pointing, slip ring

Telemetry\tracking & Telecommands

Ground Segment: architecture



Ground Station networks: Estrack

The Link budget

Link budget equation →

$$\frac{E_b}{N_0} = \frac{P_{tx} L_l G_{tx} L_s L_a G_{rx}}{k T_s R}$$

GOAL = energy per bit E_b versus noise density N_0 (Rx)ratio → containment
Tx Power P_t. → Minimization

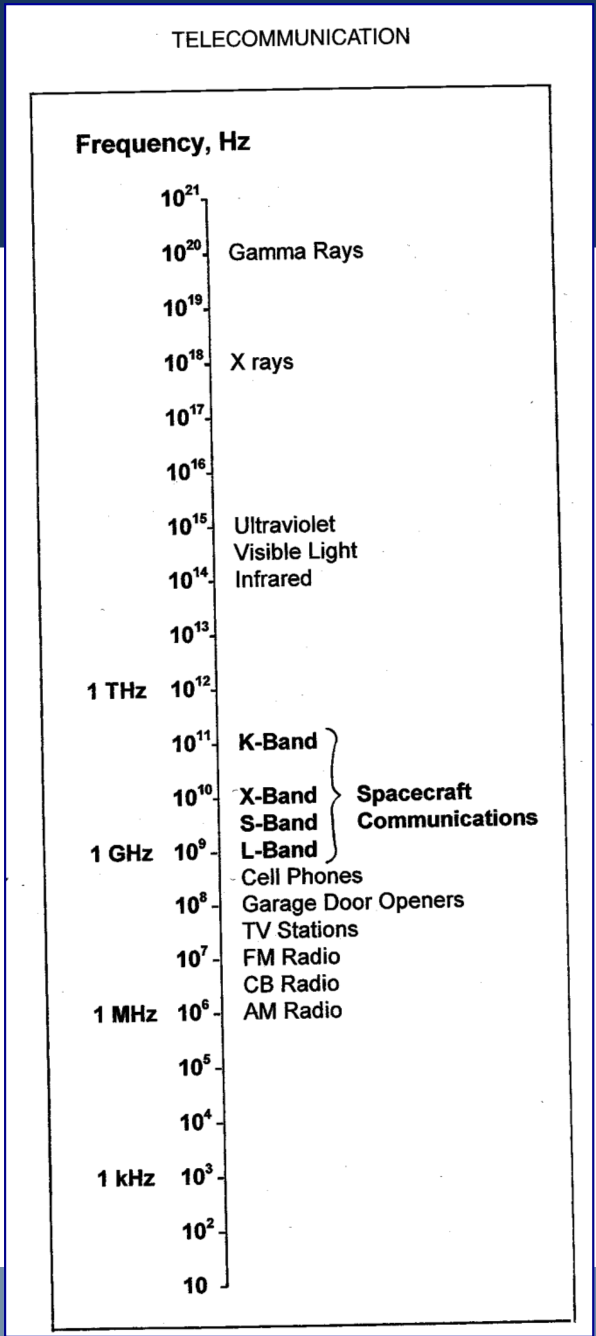
P_e = emitted power/area
L = Losses
R = data rate
G = gain of the transmitting/receiving antenna
k= Boltzmann's constant
T_s= Noise temperature

Digital transmission

$$E_b = \frac{P_{rx}}{R} [Ws]$$

Design variables		**Parameters**	
G	→antenna diam and frequency selection	R	→payload/orbit
L	→architecture optimization	T_s	→environment/sys design
E_b/N_0	→error containment		

EM Spectrum



Band	Freq. range, GHz
P	0.225–0.39
J	0.35–0.53
L	0.39–1.55
S	1.55–3.9
C	3.9–6.2
X	6.2–10.9
K	10.9–36.0
Ku	10.9–18
Ka	18–31
Q	36.0–46.0
V	46.0–56.0
W	56.0–100.0

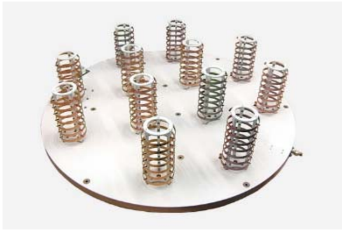
Intersatellite links (Q, V, W)

Space operations (S, C, X, K, Ku, Ka)

Antennas characteristics

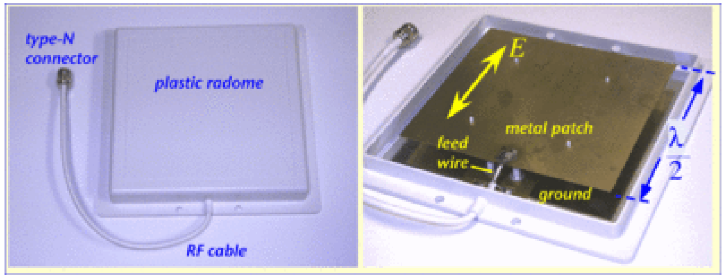
- **Horn antennas:** small aperture for Earth coverage with 4GHz (C band)

- **Helical antennas:** Earth coverage for frequencies below 4GHz (S band - MGA; Navstar)



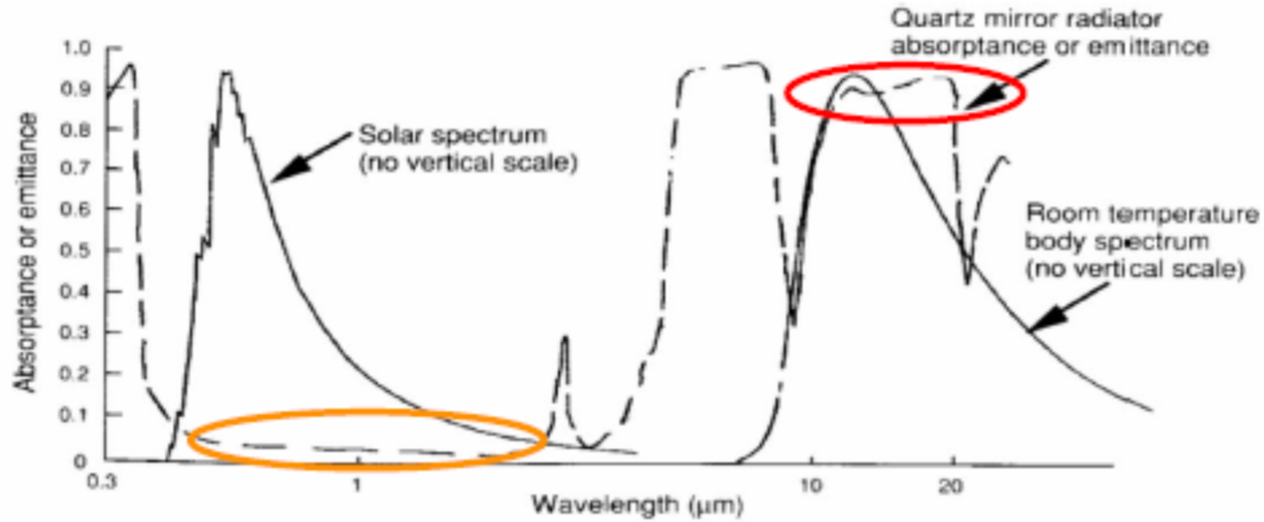
- **Reflectors:** narrow beam requirement (HGA)

- **Patch antenna**: flat metallic surface, low gain (3-7 dB, θ=65°)



Thermal Control

Radiation



- Visible [0.2μm – 2.8 μm] [800°C_14000°C] (= 95% of Sun energy,<0.5% of IR energy)
- Infrared [5μm – 50μm] [-200°C_300°C] (=92% of IR energy,< 1% of the Sun energy)

Depending on the wavelengths the following terminology is applied:

ABSORBIVITY – α – heat transfer in the 1) frequency band

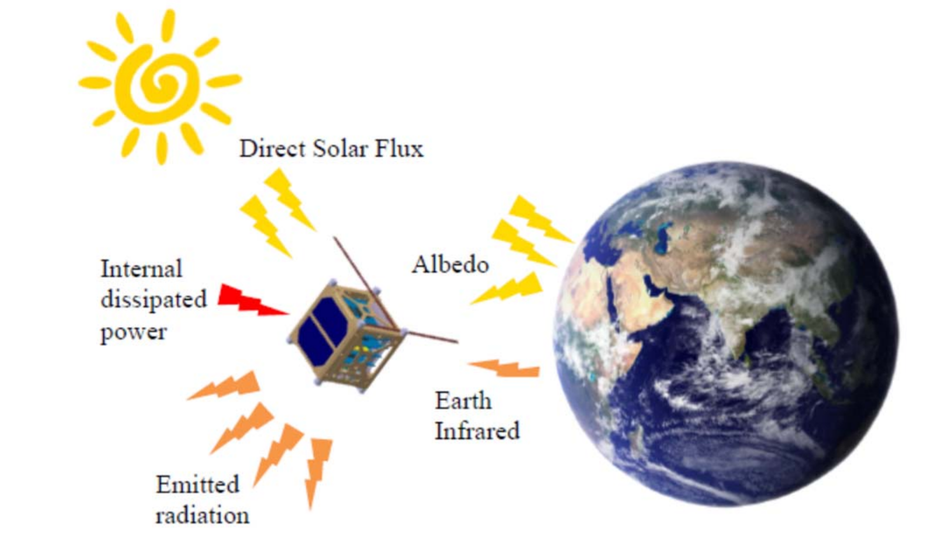
EMISSIVITY – ε - heat transfer in the 2) frequency band

Nb. Attention if wide temperature ranges exist in the scenario

Heat Sources & thermal design - preliminar

Environmental inputs

- **Sun radiation**
- Planetary sources:
  - **Albedo**
  - **IR emission**
- **Internal sources**



In general we can assume (unless internal heat source is particularly high in shadow):

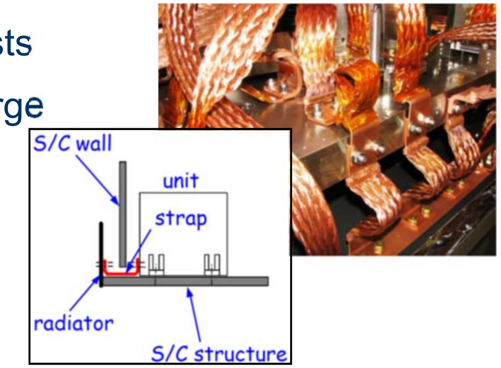
Hot Case $Q_i+Q_{IR}+Q_a+Q_s-Q_d=0$

Cold Case $Q_i+Q_{IR}-Q_d=0$

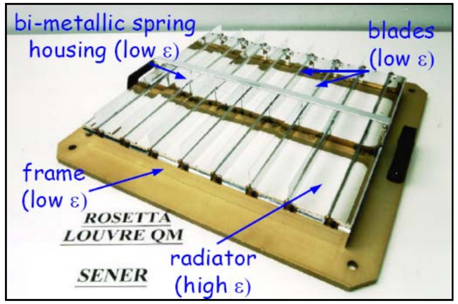
Thermal Control Components

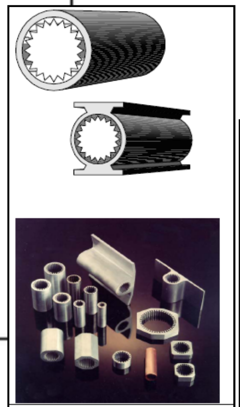
The main Thermal Control solutions are:

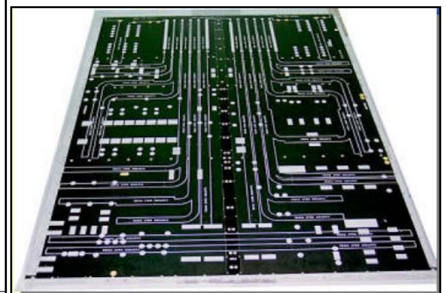
- **PASSIVE THERMAL CONTROL (PTC)** simple, reliable, low mass, power,costs
- **ACTIVE THERMAL CONTROL (ATC)** allowable ranges tight and precise, large power to be evacuated, variable environment, cryogenic application,…



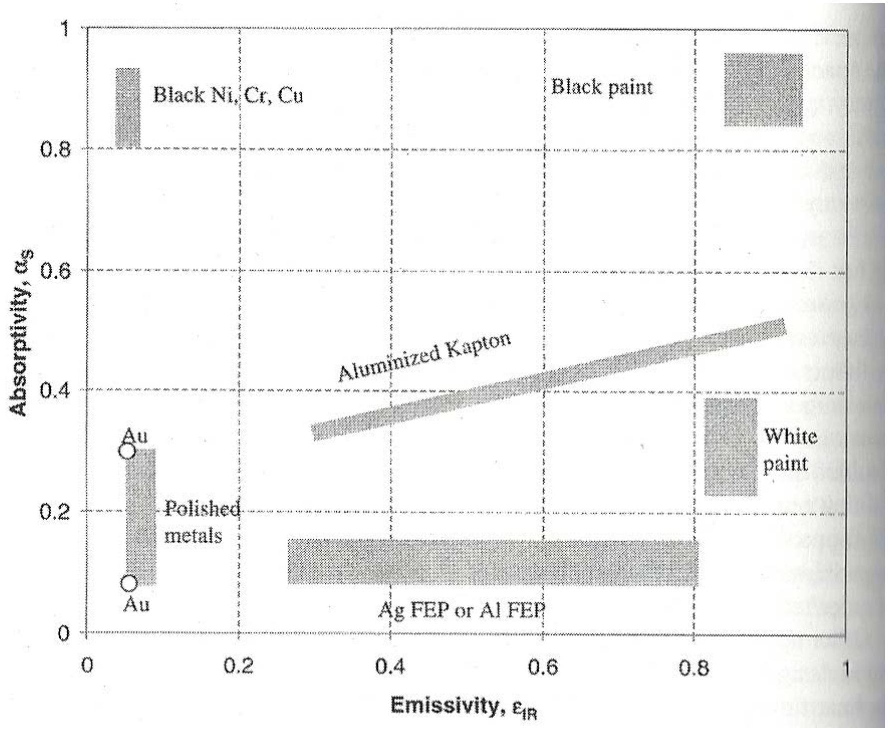
Passive	**Radiation** - Coating - MLI blanket - radiator **Latent heat & ablation** - Thermal protection system - Phase change material	**Conduction** - Structural material - Doubler, filler - Washer, strap, bolt - foam
Active	**Heater** - Thermostat control - Electronic control - Ground control **Peltier element**	**Heat pipes** - fixed/variable conductance **Fluid loops** - mono/diphasic fluid **Louvers** **Coolers**

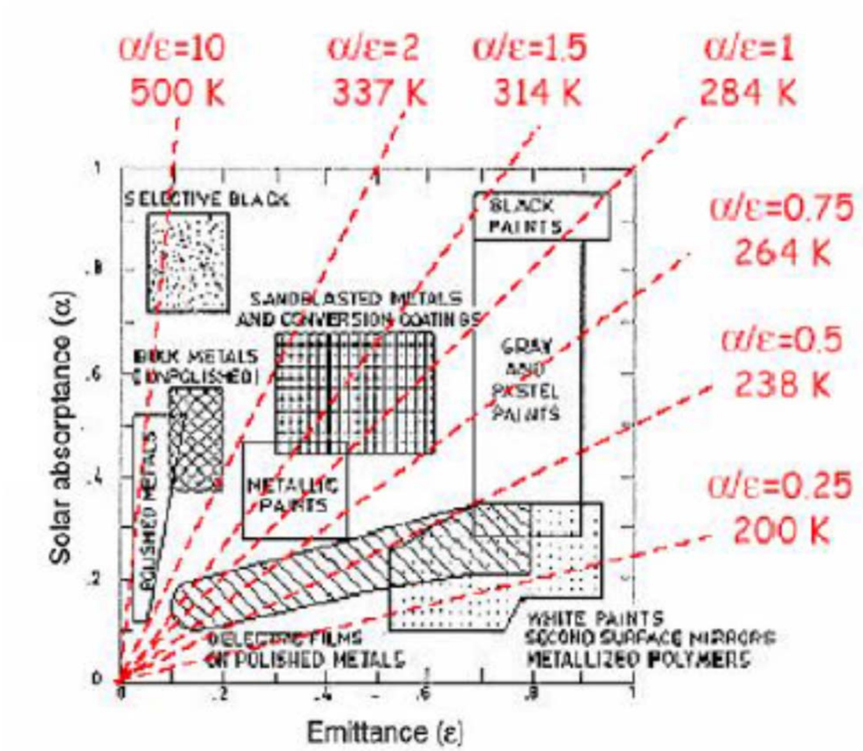






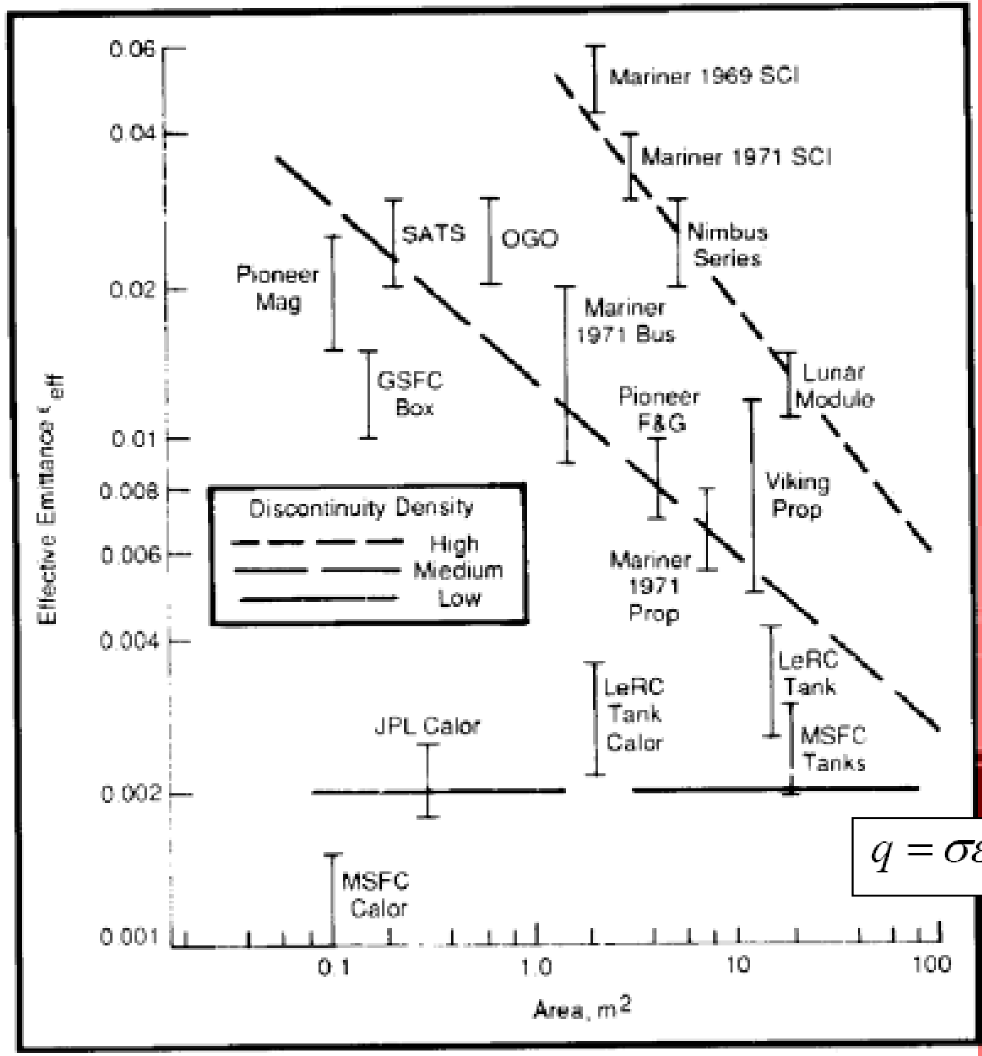
Thermal Control Components: Coatings



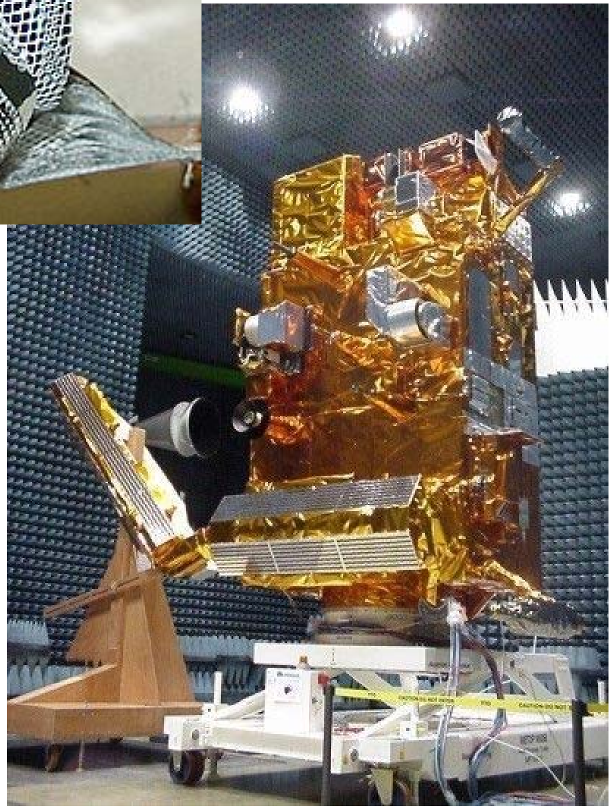


Coated Sphere Equilibrium Temperature in Sun

Thermal Control Components: Insulators - MLI



$$q = \sigma \varepsilon_{eff} (T_i^4 - T_o^4)$$



Electric power subsystem

Electric Power Subsystem

Electric Power

- **Power Source** — It converts any energy source into electric energy
- **Energy Storage** — It stores the primary source energy excess to provide electric power supply whenever the primary source is unavailable or for power peaks
- **Power Distribution** — Set of components (*harness)* devoted to the loads\sources interfaces
- **Power Regulation & Control** — Set of components devoted to the power control in *terms of V and I to provide to the loads*. Regulation is needed because of:
  - Loads requirements
  - Mission profile variability
  - Power source degradation
  - Batteries charge\discharge control

Electric Power Sources

Primary Power Sources

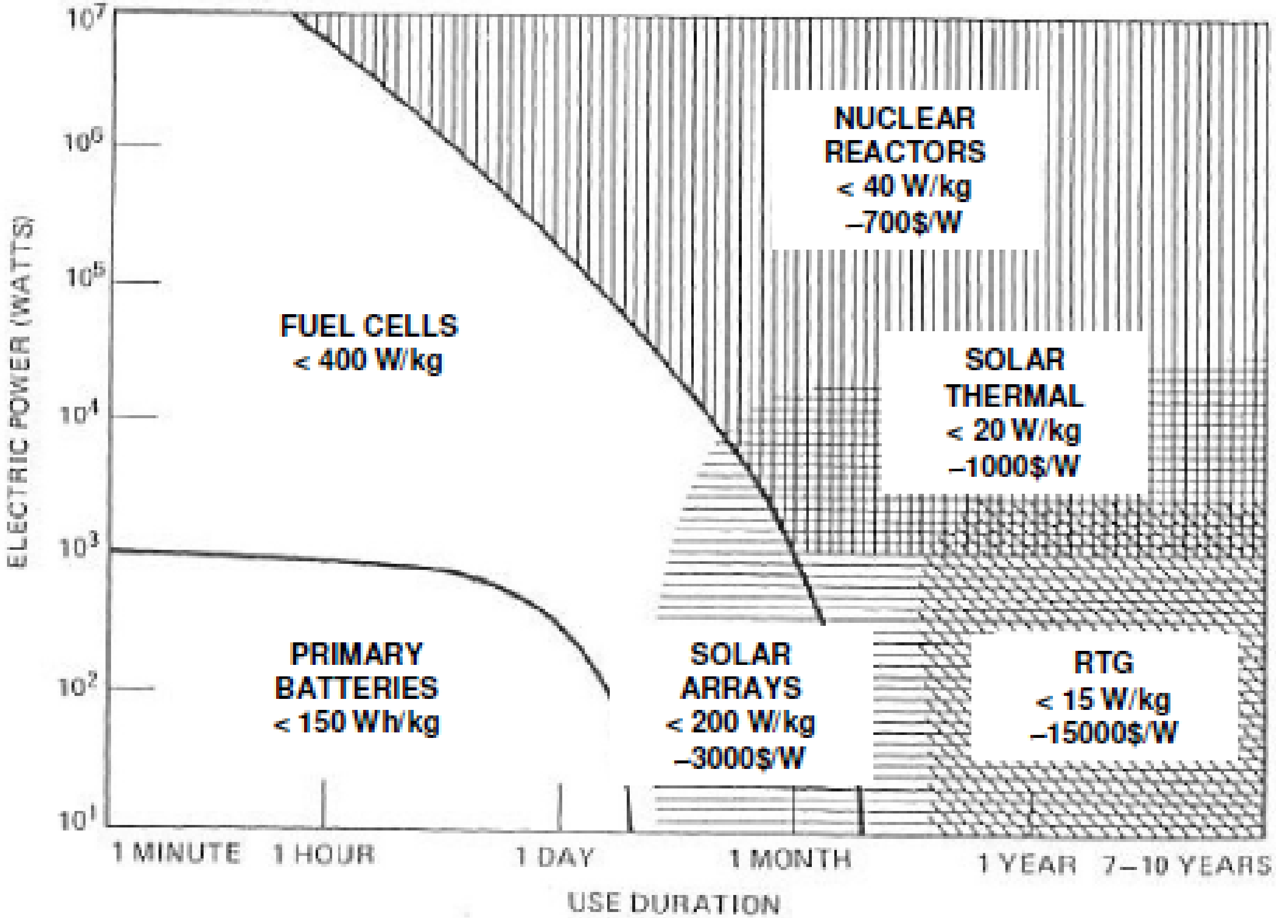
- Primary Batteries
- Solar array
- *RTGs (Radio Isotope Generators)*
- *Fuel Cells*
- *Solar Dynamics*
- *Nuclear Reactor*

Power Storage and Secondary Power Sources

- Secondary Batteries (accumulators)
- *Regenerative Fuel Cells*

Electric Power Sources: alternatives and taxonomy

Power density wrt mission lifetime

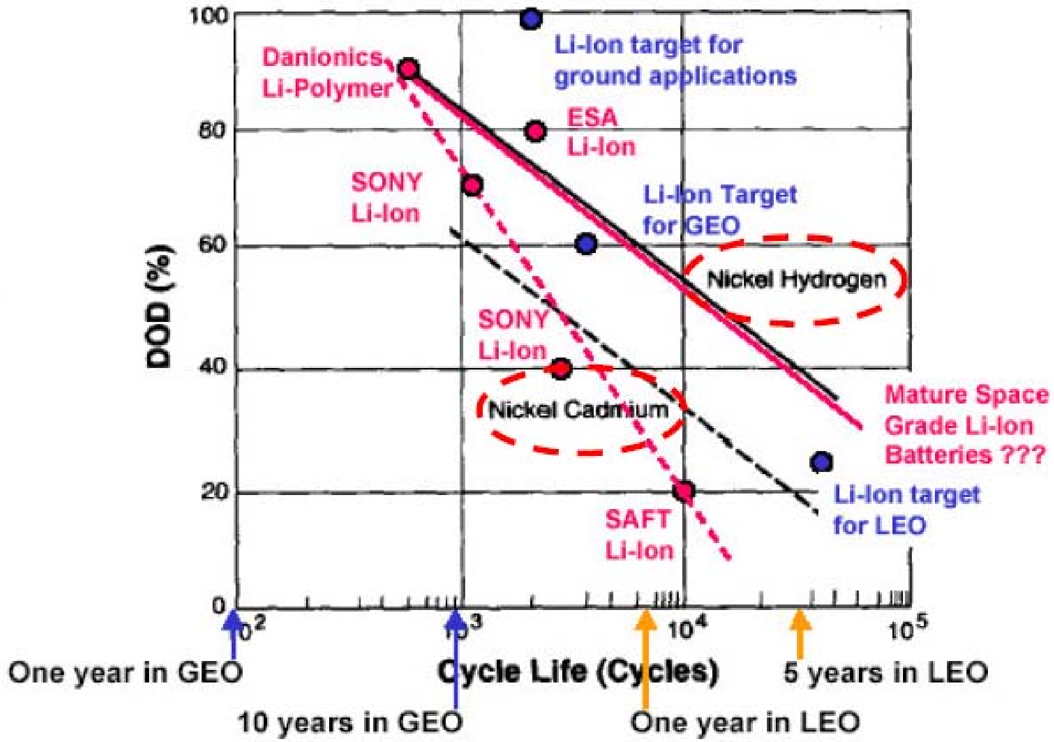


Secondary batteries sizing

Steps:

•Operational profile assessment:

- # and frequency of eclipse cycles



Energy Storage: Batteries sizing

The battery is sized computing its capacity as a function of the required power in eclipse and its characteristic DOD.

The capacity of a battery is computed as:

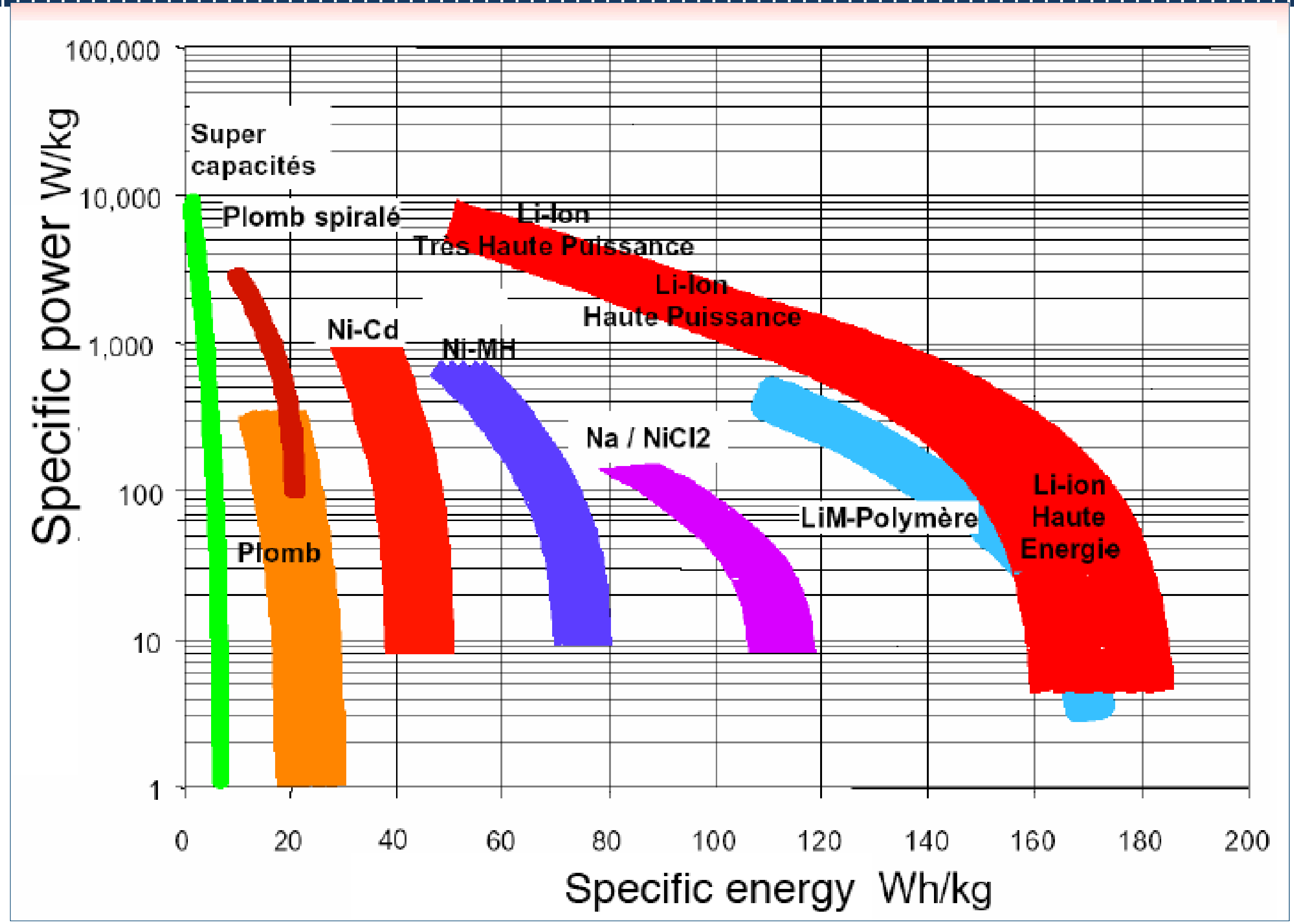
$$C_r = \frac{P_e T_e}{(DOD) N \eta} \text{ [Wh]} \qquad M = \frac{P_e t_e}{E_d}$$

where

- P_e is the average eclipse load in Watt
- T_e is the correspondent maximum eclipse time in hours
- *DOD* is the limit on battery's Depth-Of-Discharge
- *N* the number of batteries
- h transmission efficiency between batteries and load=f(T, C/rate)
- E_d energy density per unit mass

Batteries **need to be re-conditioned**→completely discharged to re-gain the global capacity

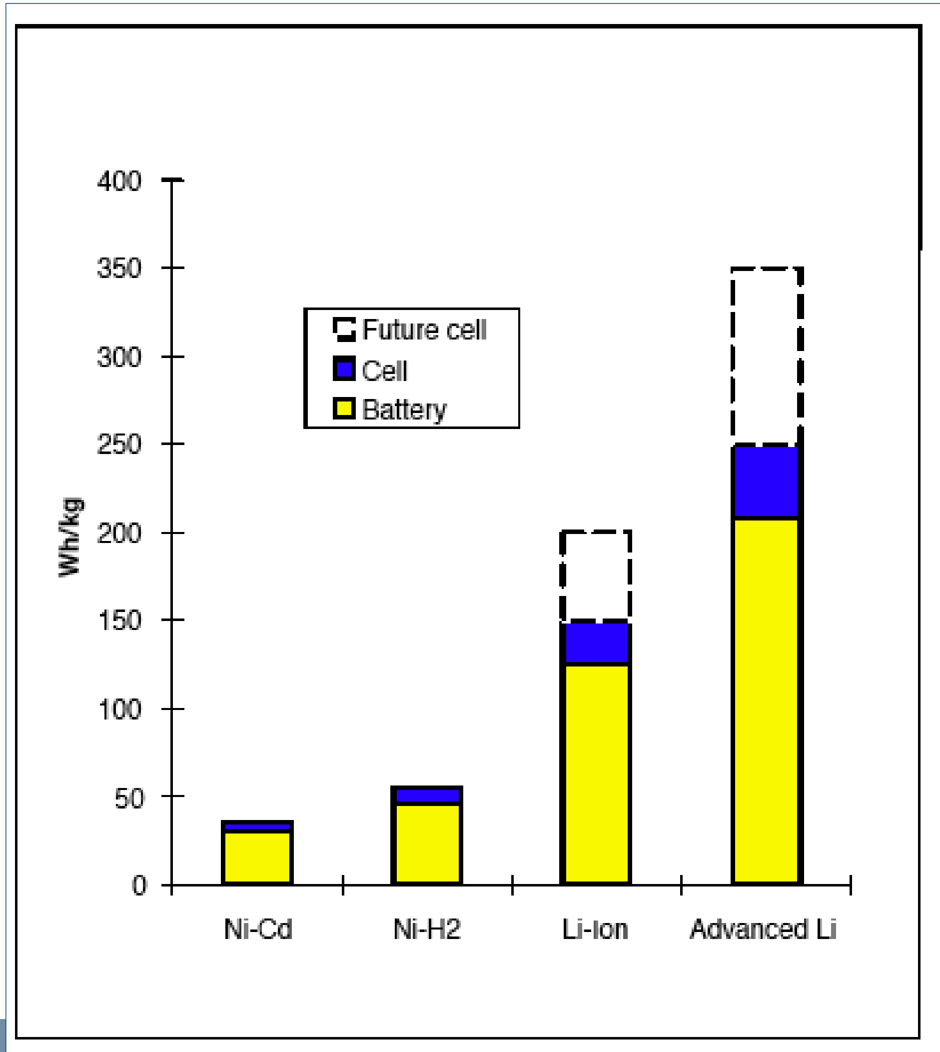
Secondary batteries: performance comparison



Supplier: http://www.saftbatteries.com/

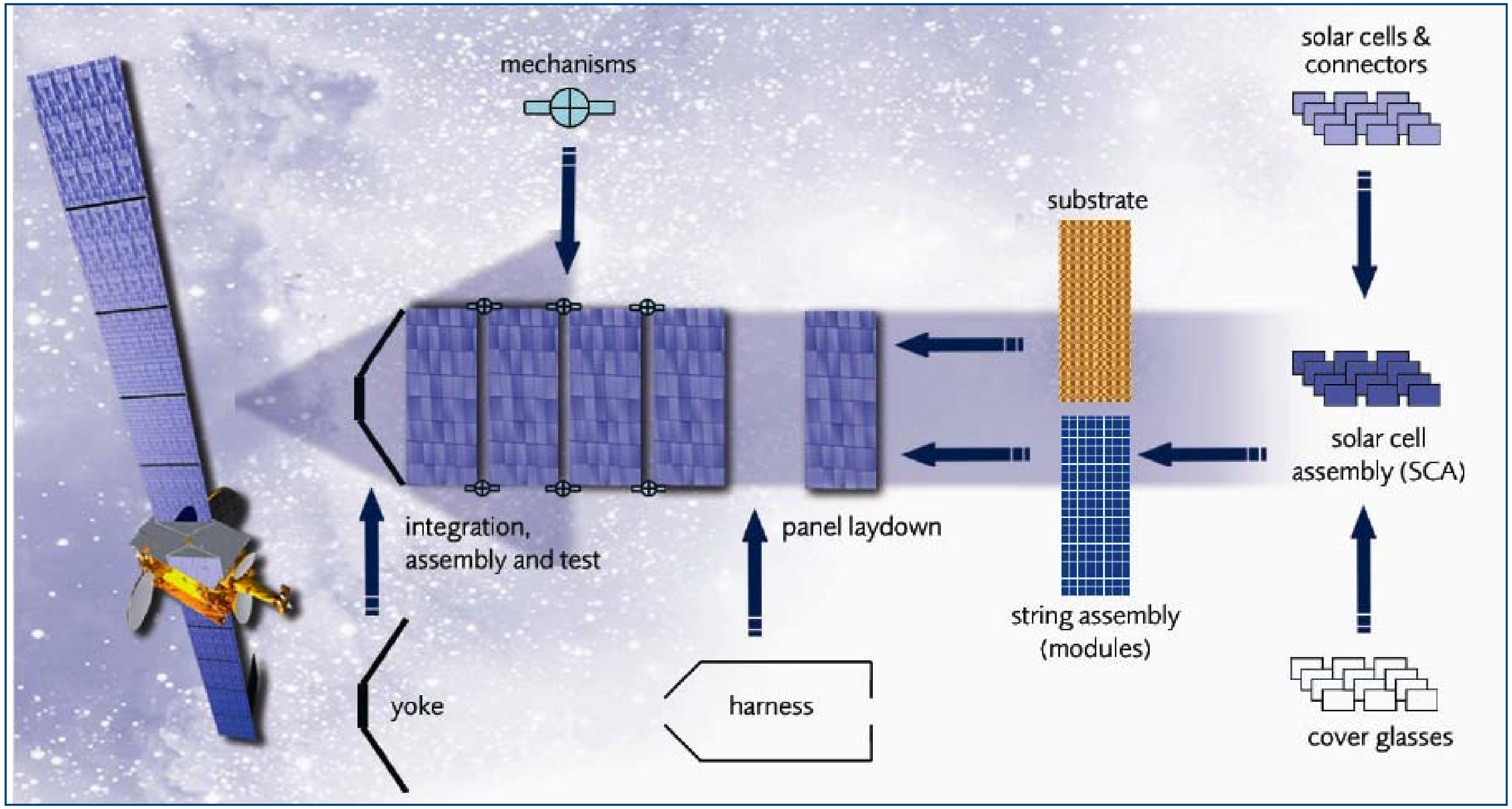
Energy storage: performance comparison

Cells packaging efficiency degradation
(~20%)



Supplier: http://www.saftbatteries.com/

Solar Arrays



EPS-Solar Arrays standards→ ECSS-E-20-08

Solar Array Sizing Process

First Step: identify worst case

- Maximum time in eclipse T_e and correspondent time in daylight T_d
- Maximum total power requirements in eclipse P_e and in daylight P_d
- Compute the total power required P_{sa} considering an efficiency factor in eclipse X_e and in daylight X_d : to this end a Power Budget table for different modes must be filled

$$P_{sa} = \frac{\frac{P_e T_e}{X_e} + \frac{P_d T_d}{X_d}}{T_d}$$

- DET (Direct Energy Transfer: Xe=0,65; Xd=0,85
- PPT (Peak Power Tracking): Xe=0,6; Xd=0,8

- Consider the required voltage from the loads to size the bus (string\cell number for SA and battery)

Solar Array Sizing Process

Second Step: identify power source characteristics

- For solar arrays compute the power generated at Beginning Of Life (BOL) P_{BOL}

$$P_{BOL} = P_0 I_d \cos\alpha$$

- I_d is the inherent degradation factor (0.49-0.88) and P_0 is in W/m^2 is the specific power at 1AU for the selected solar cells
- Estimate solar array degradation factor L_d

$$L_d = (1-d)^{lifetime}$$

- Compute the power produced at End Of Life (EOL) P_{EOL}

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

- Compute the total area required

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

- and the correspondent mass

Power Sources: different junction performances

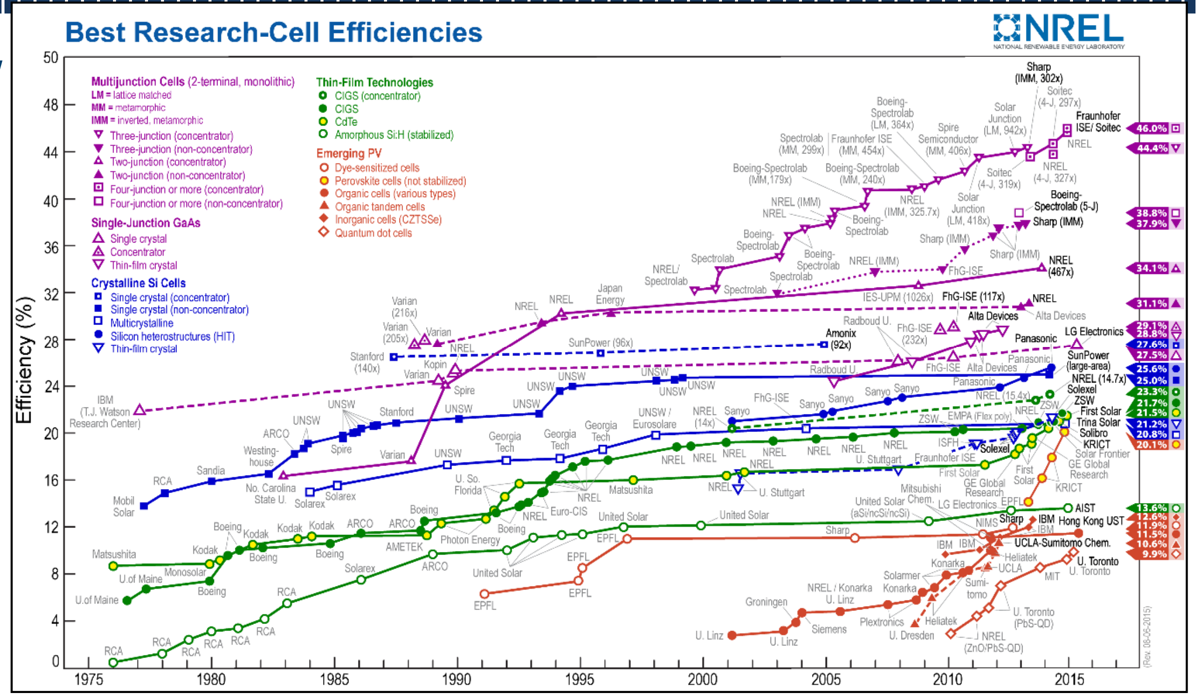
Deep Space1

→ 2J GaAs+Scarlet Concentrators 2.6kW

Juno

→ Si GaAs cells (Spectrolab) 420W (5,2AU) - 20x2,7 m

Array Technologies		
Cell Type	**BOL Efficiency**	**Specific Power**
	(%)	**(W/kg)**
Si	10	25
GaAs	19	40
GaInP/GaAs (2J)	23	60
GaInP/GaAs/Ge (3J)	26	80
InGaAlP/GaAs/InGaAs/Ge (4J)	**35**	100
Amorphous Si	10	100
CuInGaSe2(CIGS)	15	200



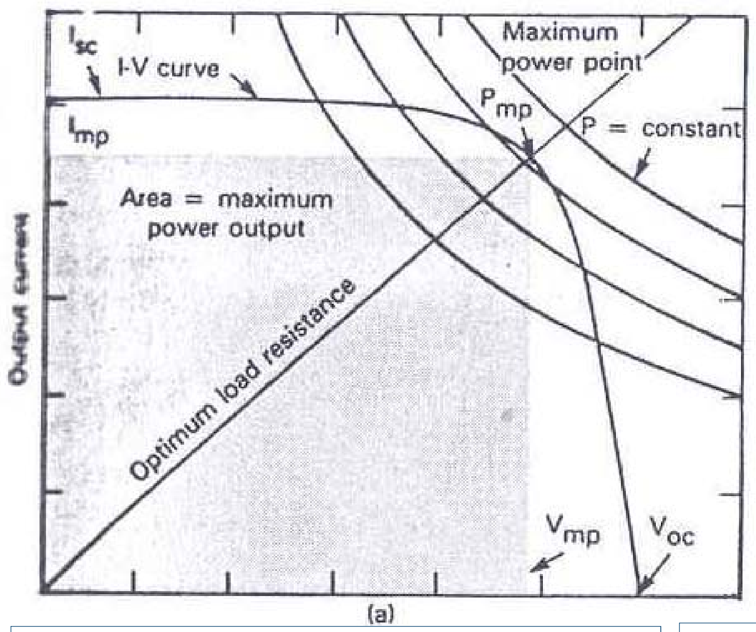
Suppliers

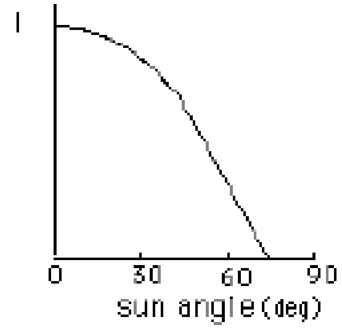
http://www.azurspace.com

http://www.spectrolab.com

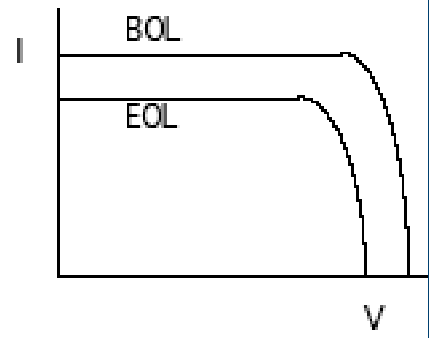
http://www.cesi.com

Photovoltaic source performance

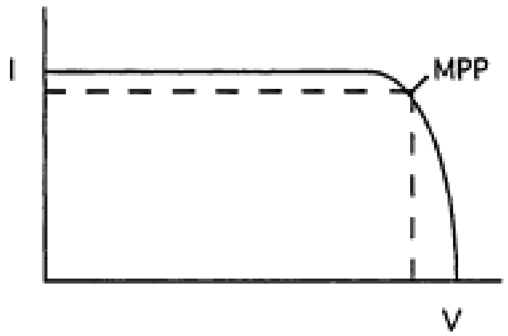




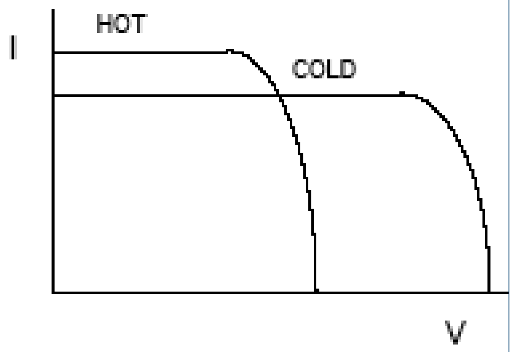
Effect of Sun Angle on Solar Array Power.



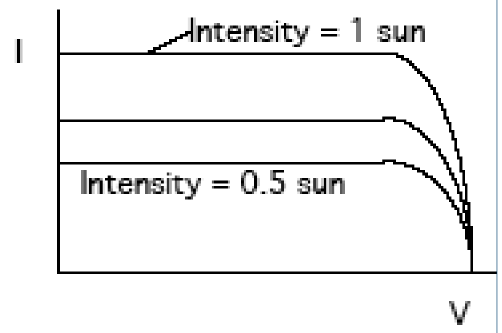
Effect of Radiation on Solar Cells.



Typical Solar Cell Power Characteristic.



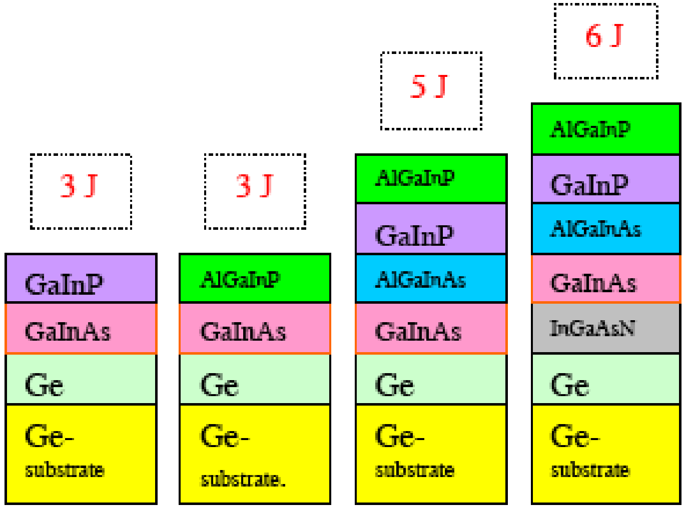
Effect of Temperature on Solar Cells.



Effect of Solar Distance.

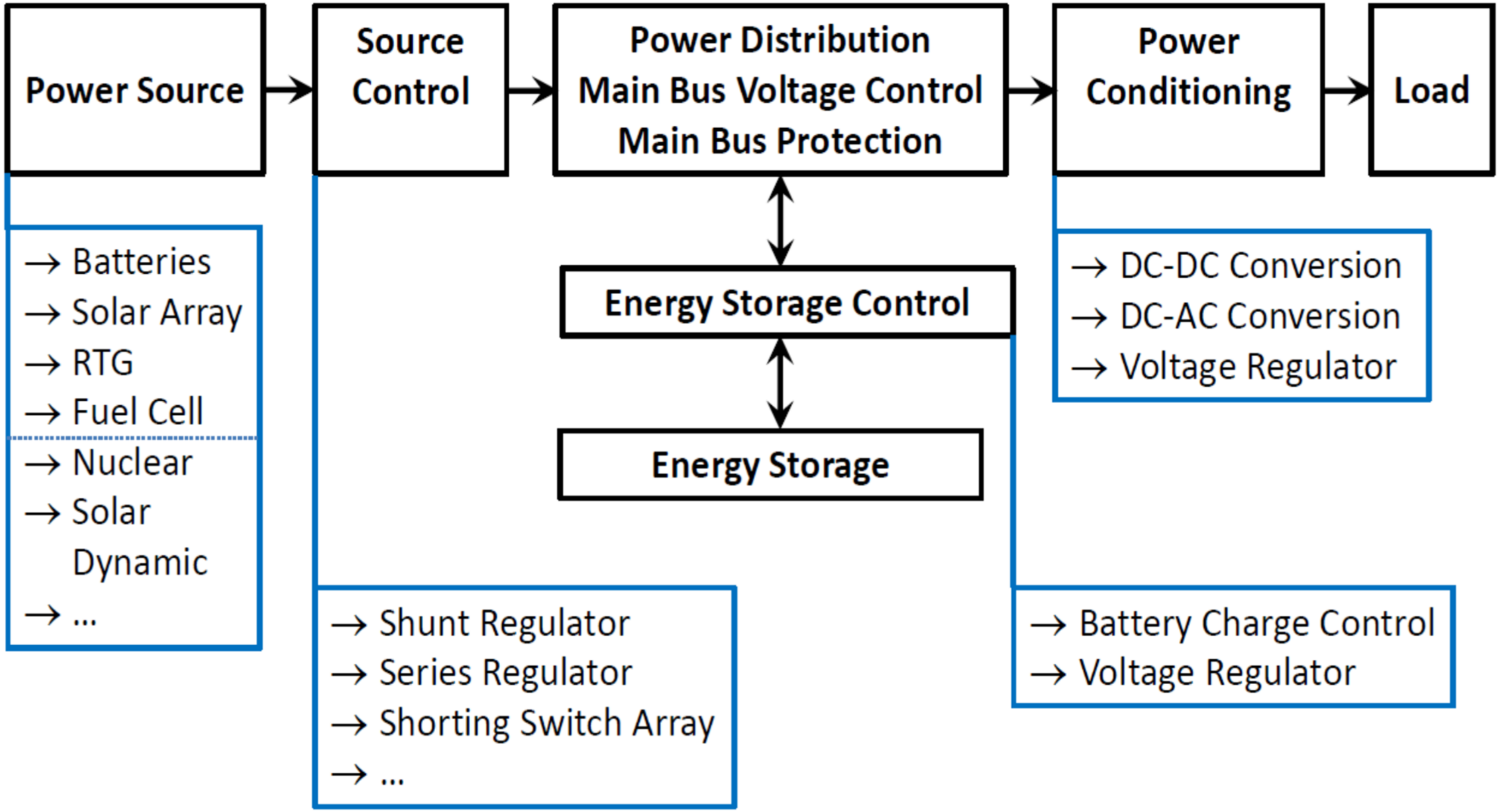
Photovoltaic source performance

Degradation



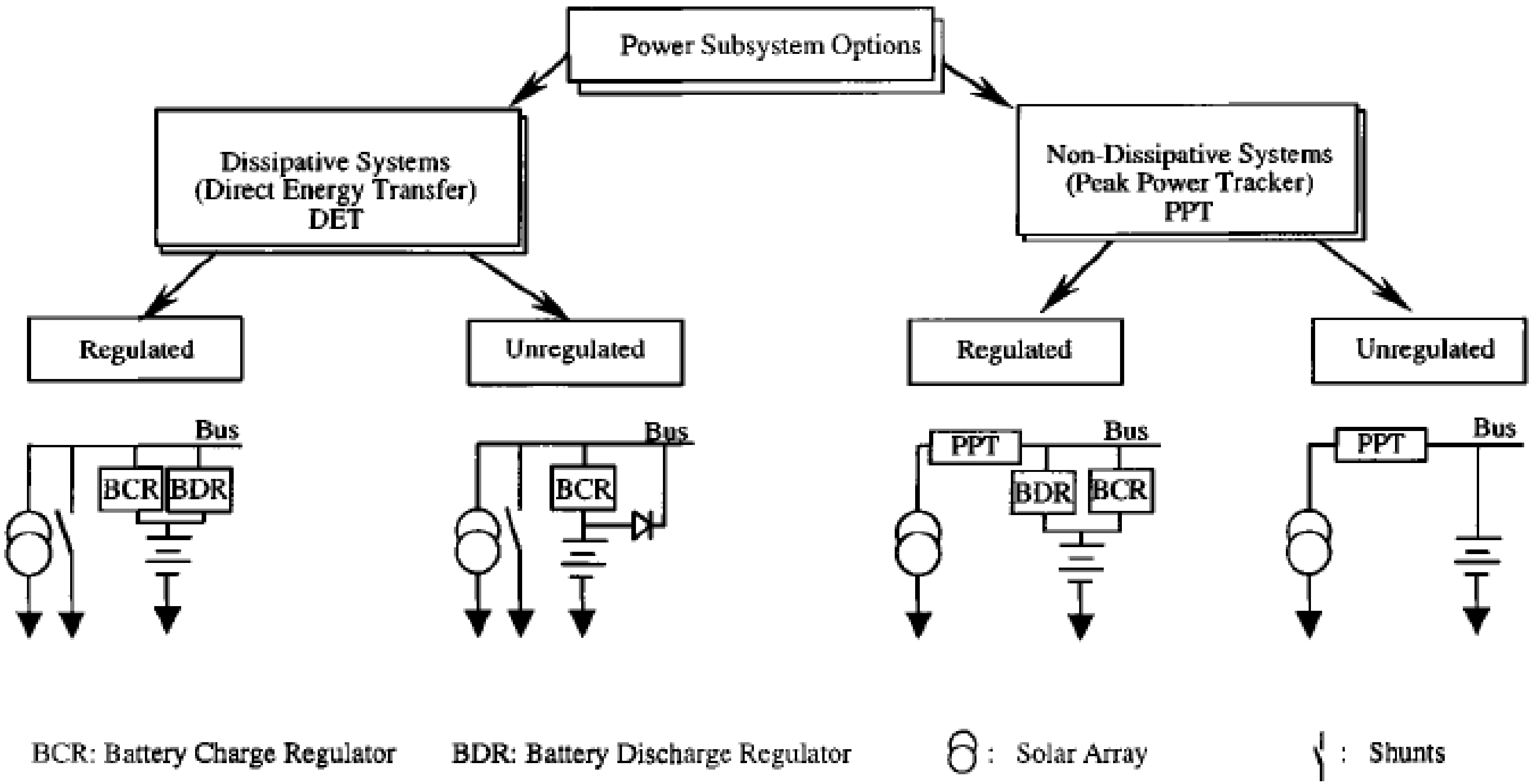
BOL □	target	28%	29%	29%	31%
(W/m^2)		(378)	(391)	(391)	(418)
EOL □	target	23.2%	25%	26%	28%
(W/m^2)		(313)	(337)	(351)	(378)

Power Control and Distribution & Conditioning



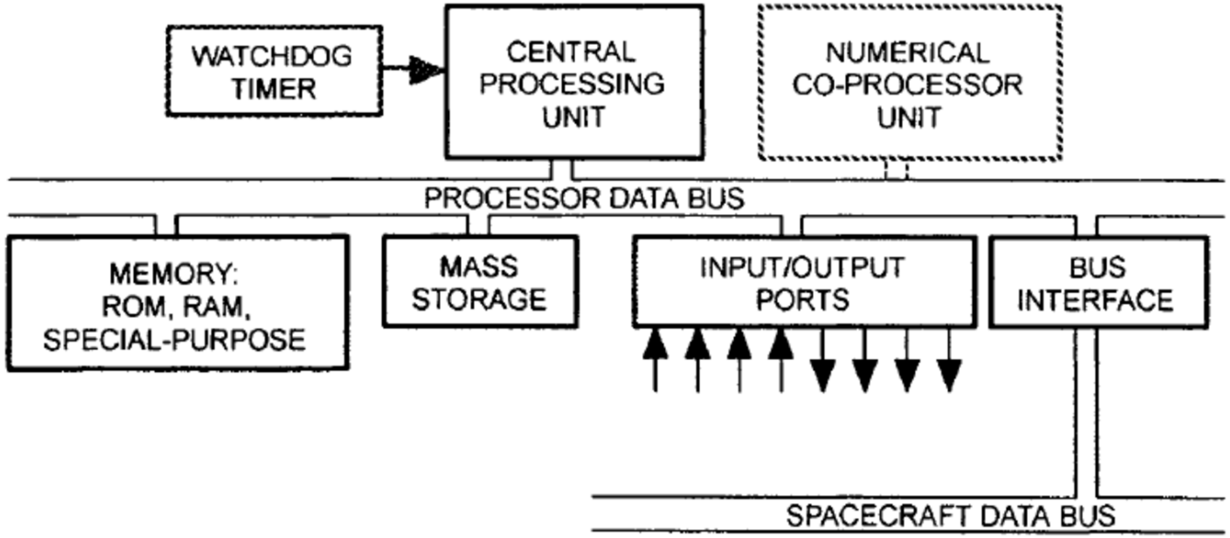
Power Regulation & Control

BUS REGULATION



On board Data Handling

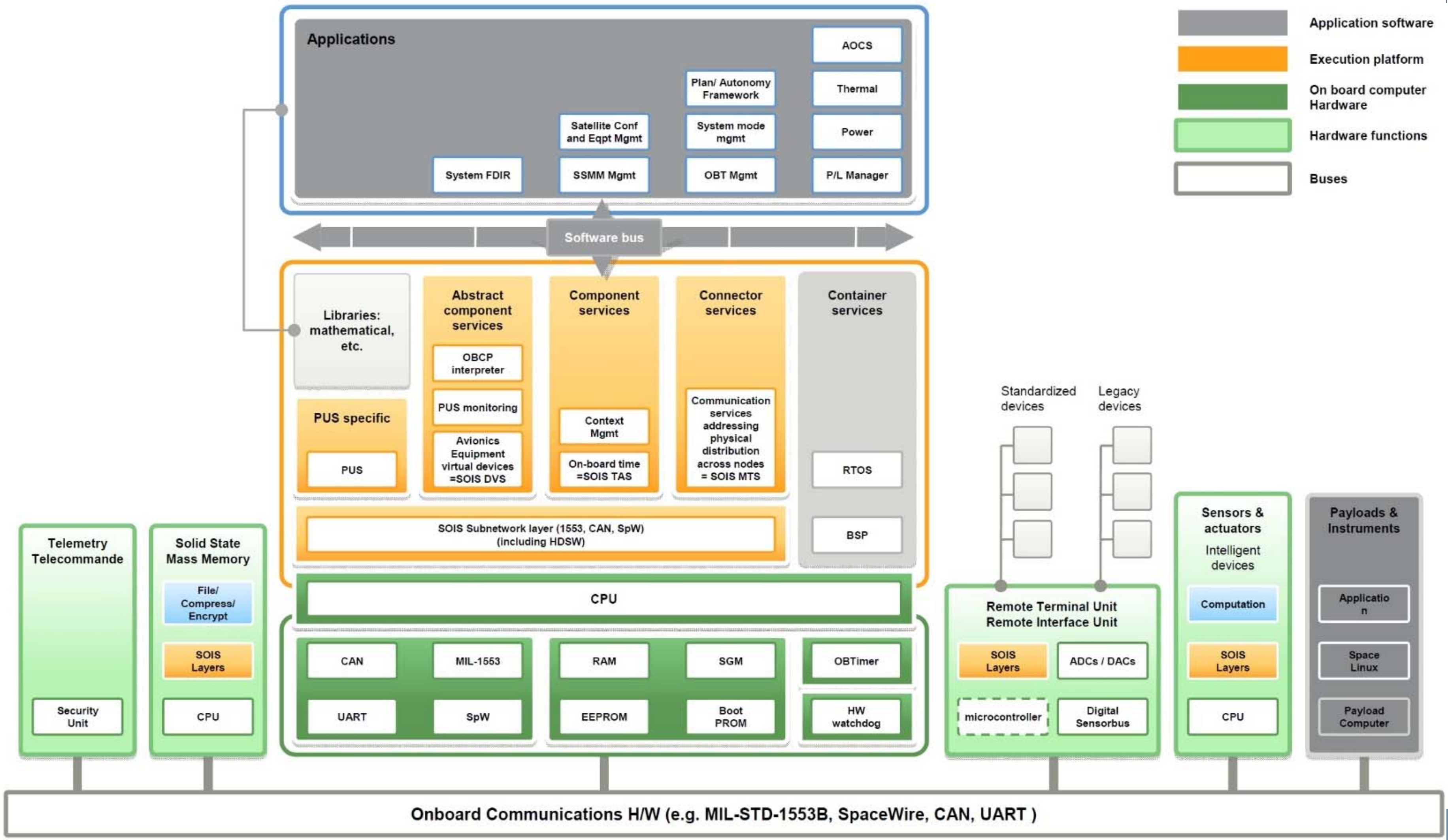
GENERAL ARCHITECTURE



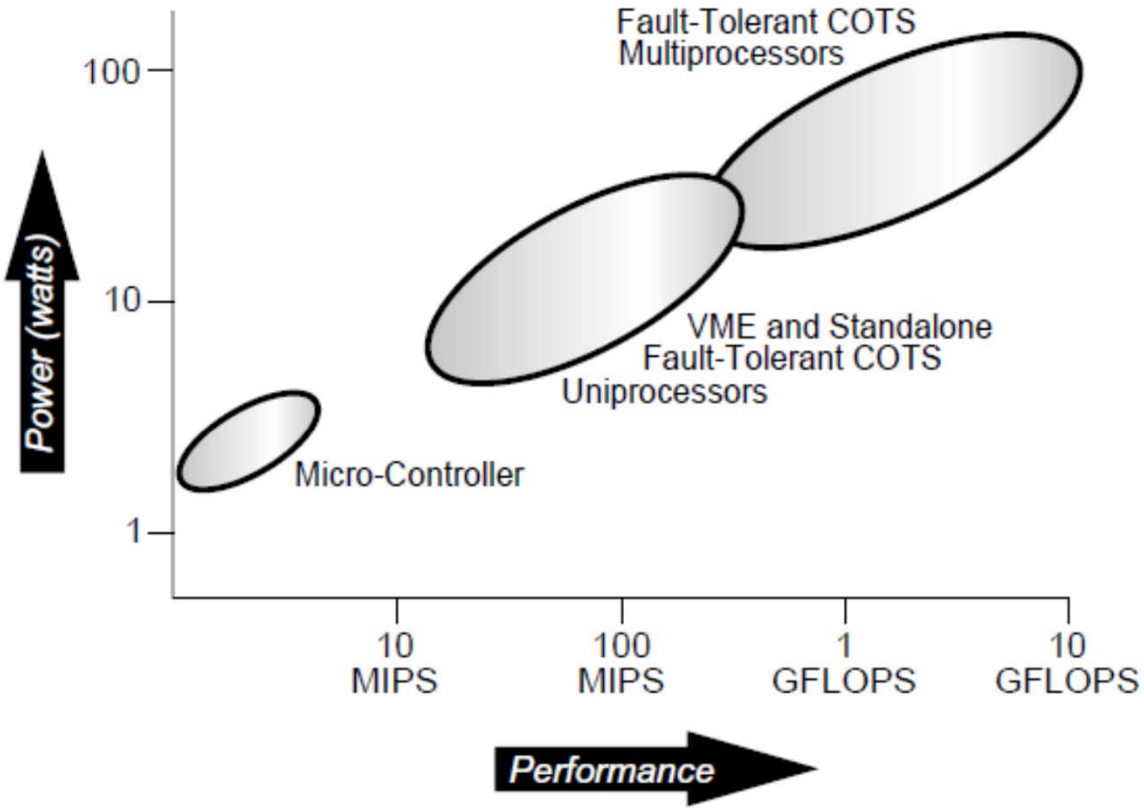
On-board Data Systems encompass a vast range of functional blocks that include:

- Telecommand and Telemetry Modules
- On-Board computers
- Data Storage and Mass memories
- Remote Terminal Units
- Communication protocols and Busses

GENERAL ARCHITECTURE



Building blocks - Microprocessor instruction\power



Highest Industry MIPS/Watt Performance

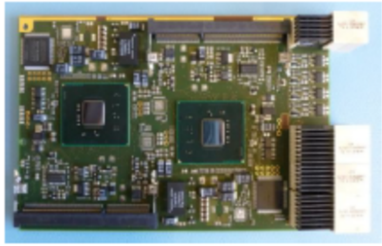
Building blocks - Microprocessor

Leon4 (HRPN-Leon4)

- **High Reliability Module**
- Standard: PICMG CPSI-S.0 compliant
- Processor: Rad Hard Leon 4
- Performance: 700 MIPS
- Power consumption: < 5 W
- Mass: < 0.5 kg
- TRL: 4

TMR Module (HAPN-Atom)

- **High Availability Module**
- Standard: PICMG CPSI-S.0 compliant
- Processor: intel Atom N270 in Tripple Module Redundant configuration with voting unit
- Performance: 2000 MIPS
- Power consumption: 5-10 W
- Mass: < 1 kg
- TRL: 4



P4080 (HPPN-P4080)

- **High Performance Module**
- Standard: PICMG CPSI-S.0 compliant
- Processor: Freescale P4080 in Dual Module Redundant configuration
- Performance: 60 GIPs, 12 GFLOPs
- Power consumption: 18-28 W
- Mass: < 1 kg
- TRL: 4

System Engineering Notes - Costs

ATTENTION!!
The software weights nothing but costs a lot and takes time to be developed and tested, often more than what required for hardware

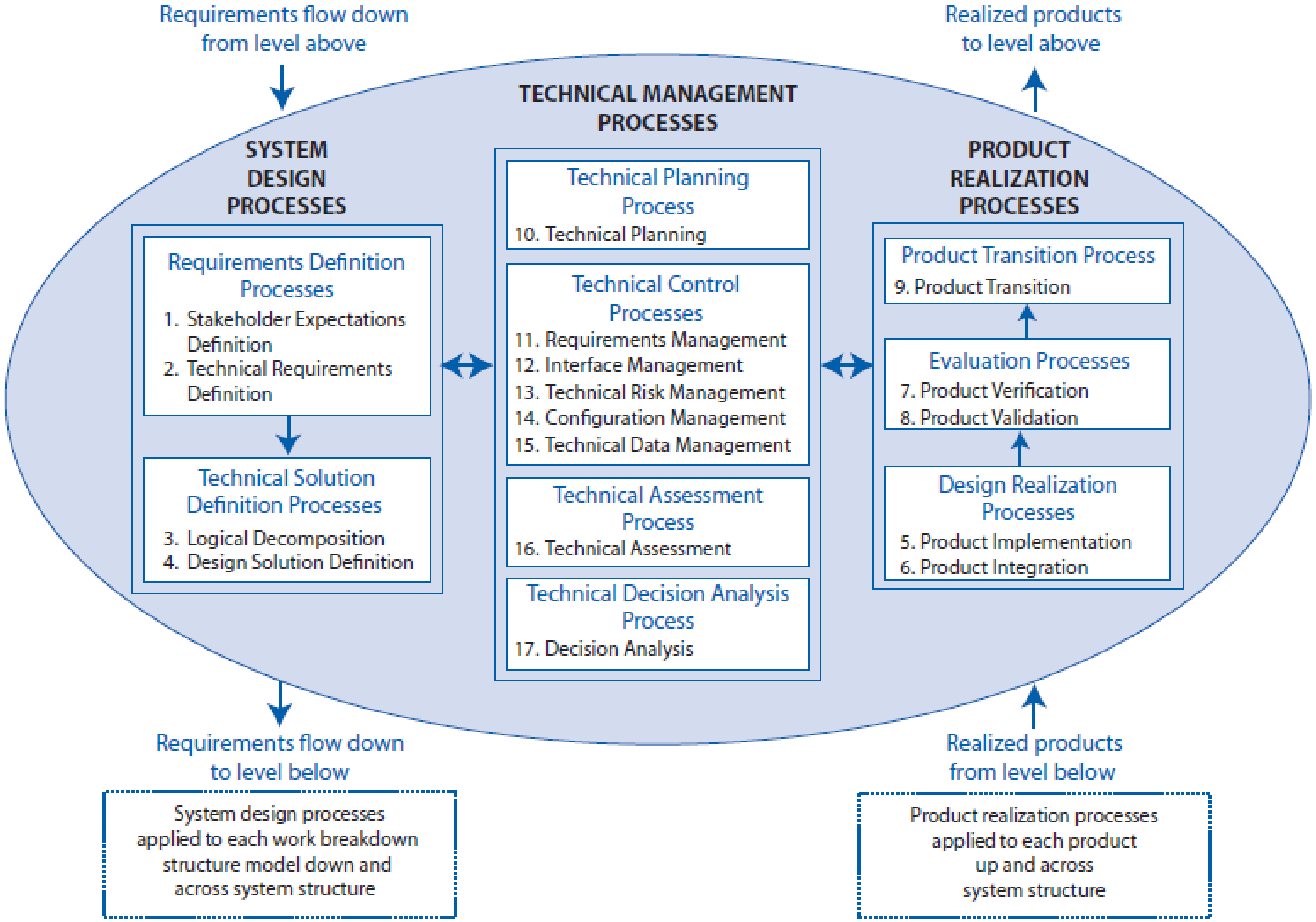
High level Cost Breakdown (highly mission dependent)

- **50% of mission cost is due to the ground segment**
- **50% of mission cost is due to the space segment**
  - 50% is due to the payload
  - 50% is due to launch +satellite

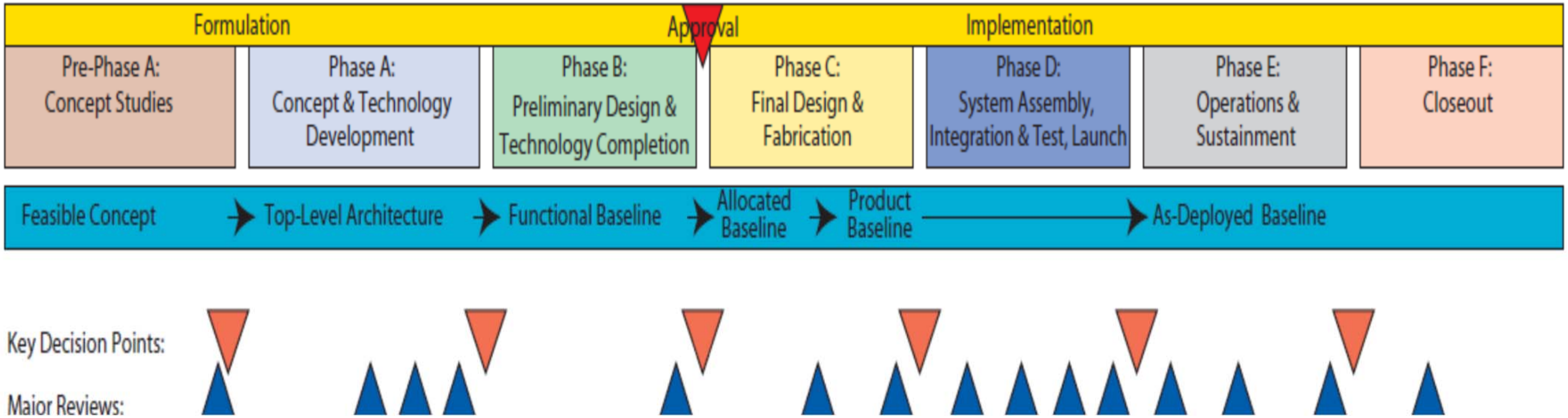
Note: for Europe, as far as scientific missions are considered, the payload s paid by national agencies, the platform by ESA

System Engineering

System Engineering notes: System Design process

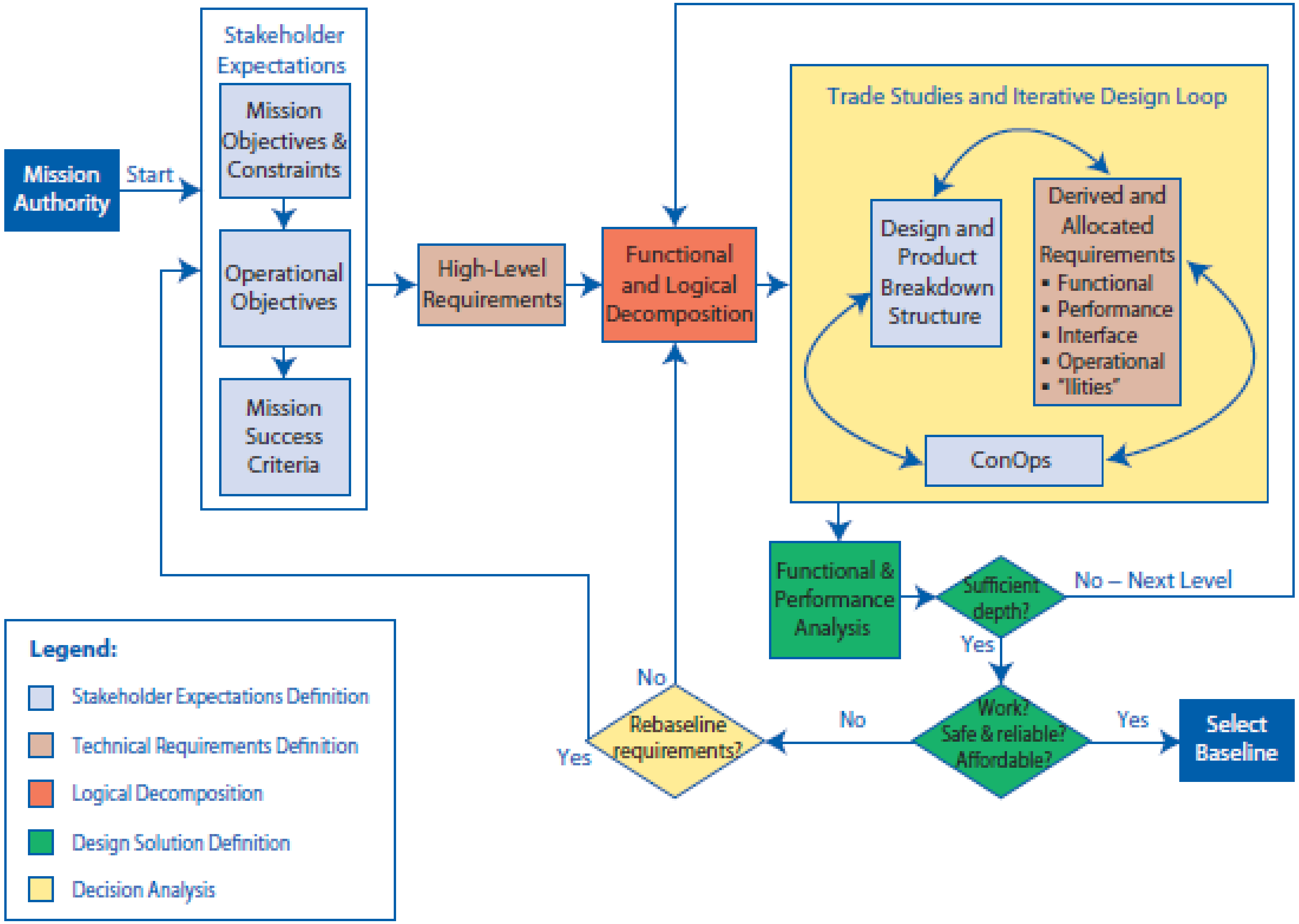


System Engineering notes: space projects lifecycle

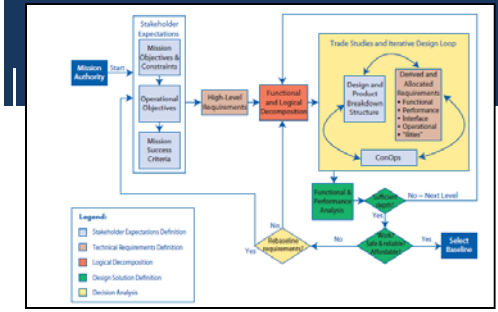


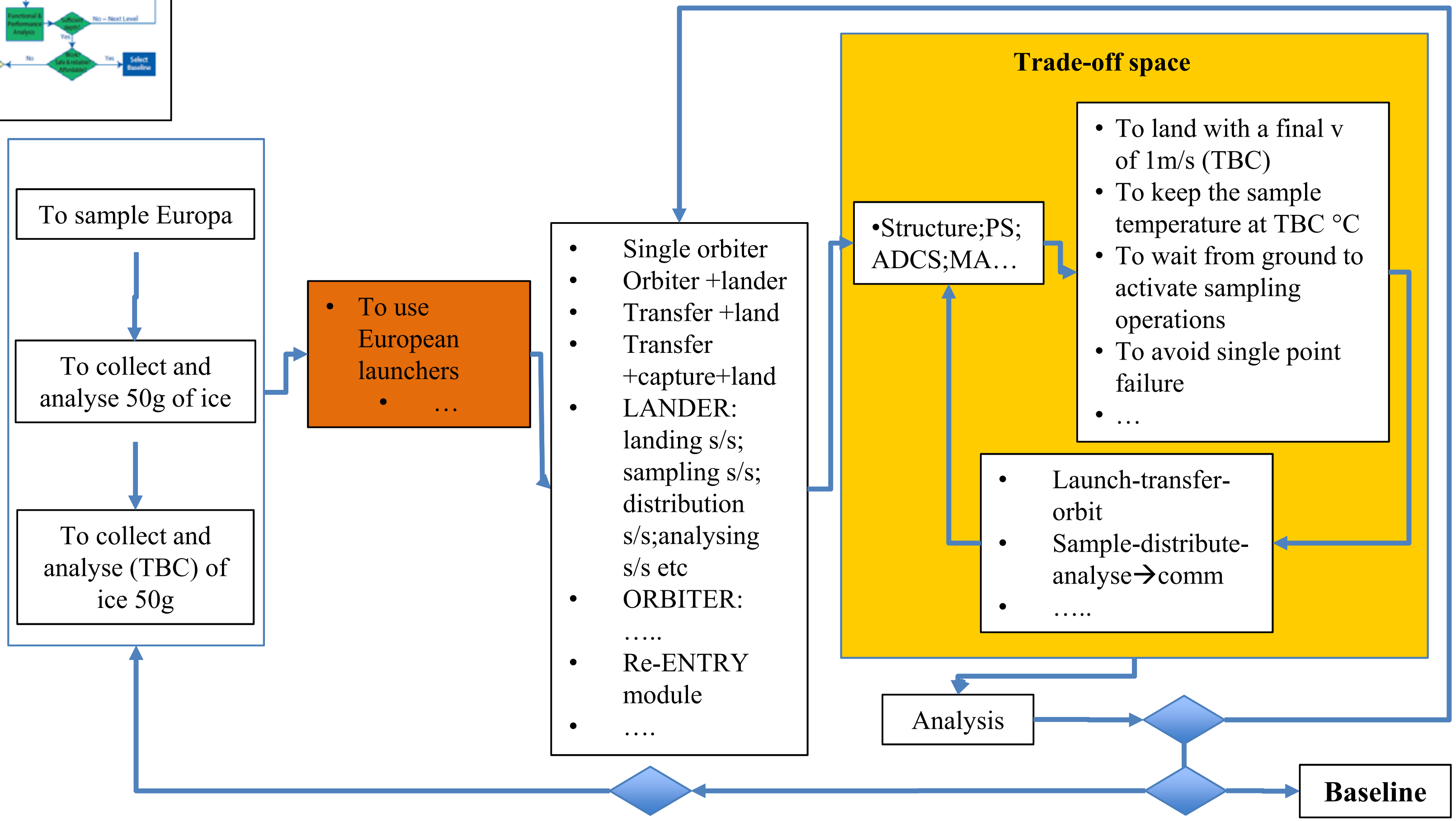
- **Pre-Phase A**—Advanced Studies ("find a suitable project")
- **Phase A**— Preliminary Analysis ("make sure the project is worthwhile")
- **Phase B**— Definition ("define the project and establish a preliminary design")
- **Phase C**— Design ("complete the system design")
- **Phase D** — Development ("build, integrate, and verify the system, and prepare for operations")
- **Phase E**— Operations ("operate the system and dispose of it properly")
- **Phase F** — Disposal

System Engineering notes: System Design process



System Engineering notes: System Design process





System Engineering notes: measure of Effectiveness

This is one of the Systems Engineer's **most important tasks**.

An elegant solution to the wrong problem is less than worthless.

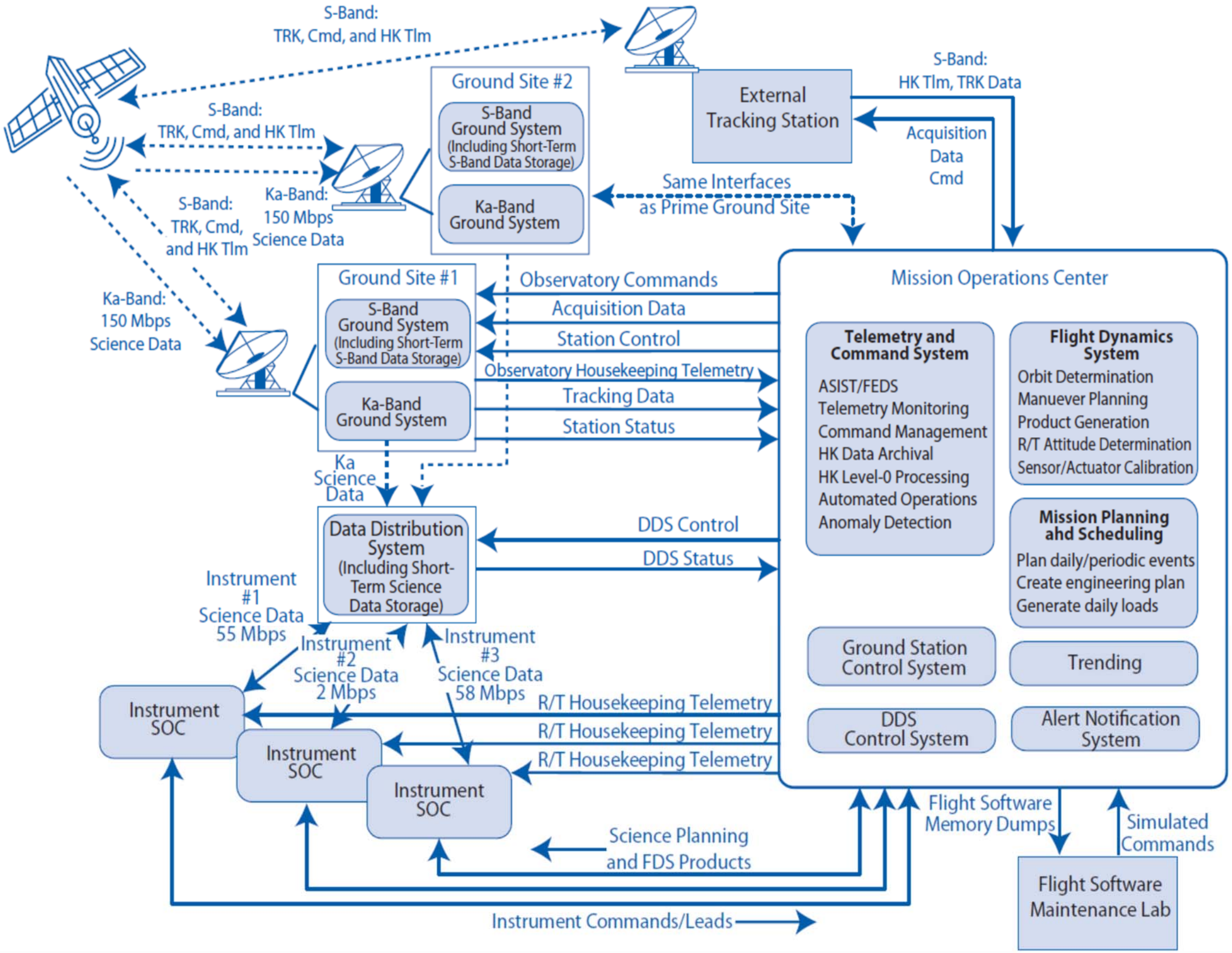
The word *optima*l should not appear in the statement of the problem, because **there is no single optimal solution to complex systems problems.**

Most system designs have several performance and cost criteria.

Typical **performance and cost criteria in space system design** may be the "budgets":

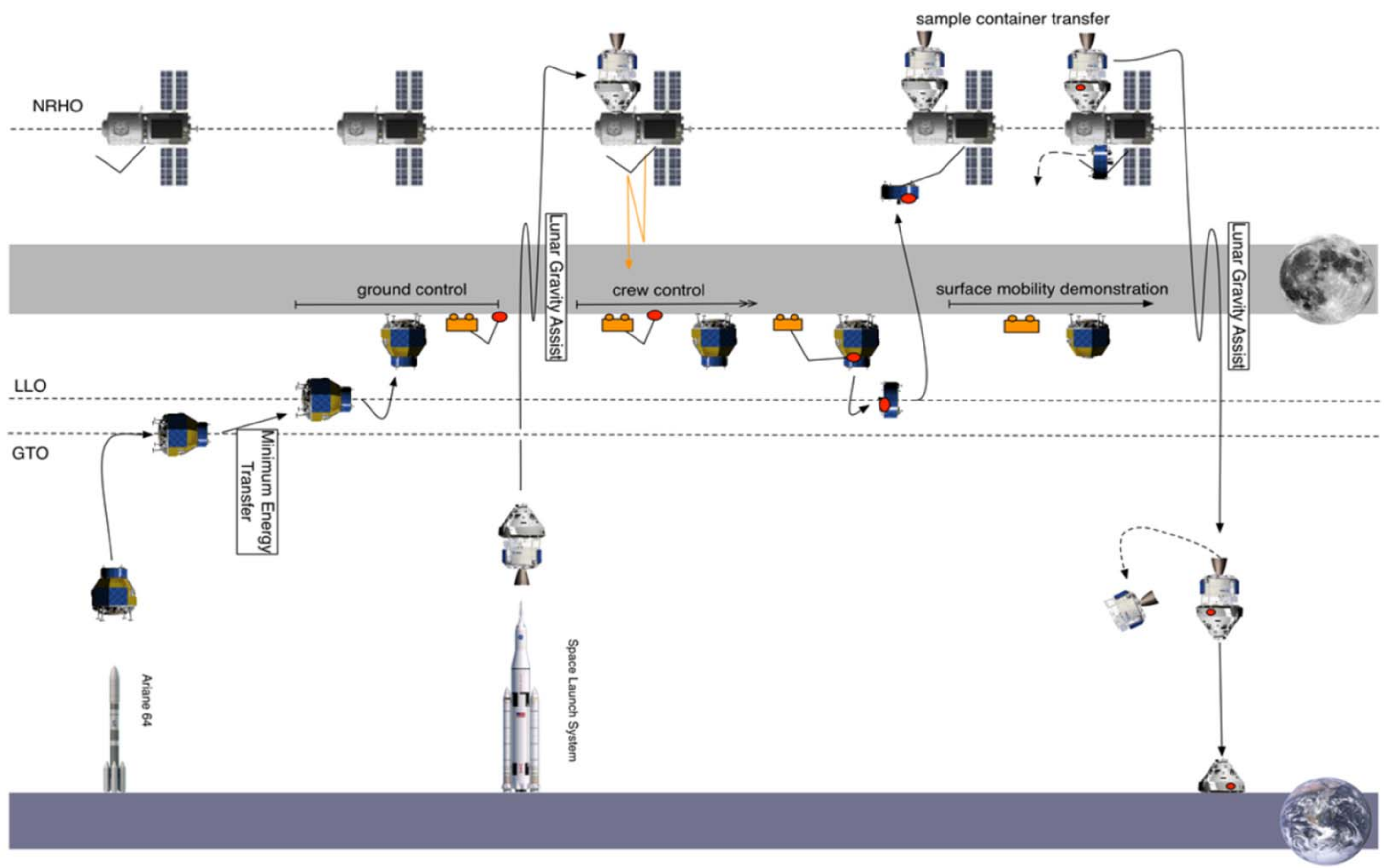
- **Mass budget**
- **Power budget**
- **Δv budget**
- **Pointing budget**
- **Link budget**
- **Cost budget**
- **Risk assessment**
- **Mission specific budgets (e.g.** *DOP, coverage, revisit time, etc*)
- **…**

System Engineering notes: Operations Architecture



System Engineering notes
Define mission timeline - phases

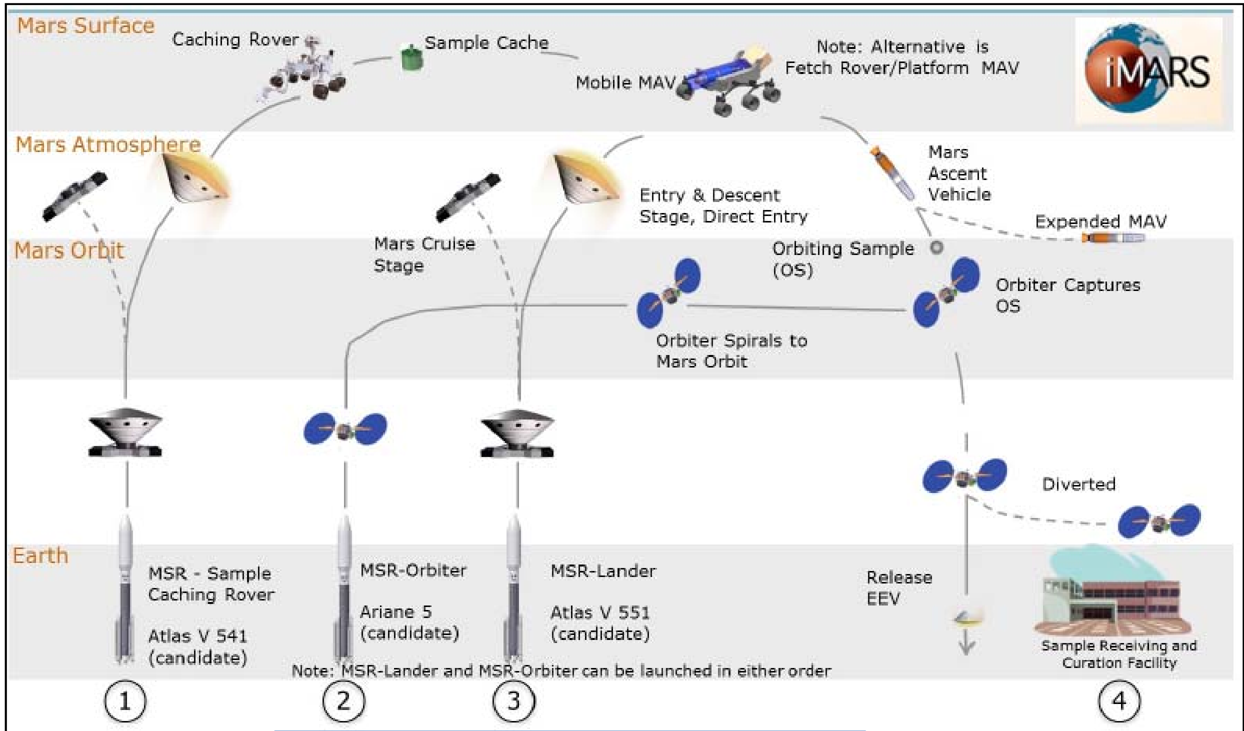
Define Conceptual Operations (ConOps) - Moon service example (LOP-DSG)



System Engineering notes
Define mission timeline - phases

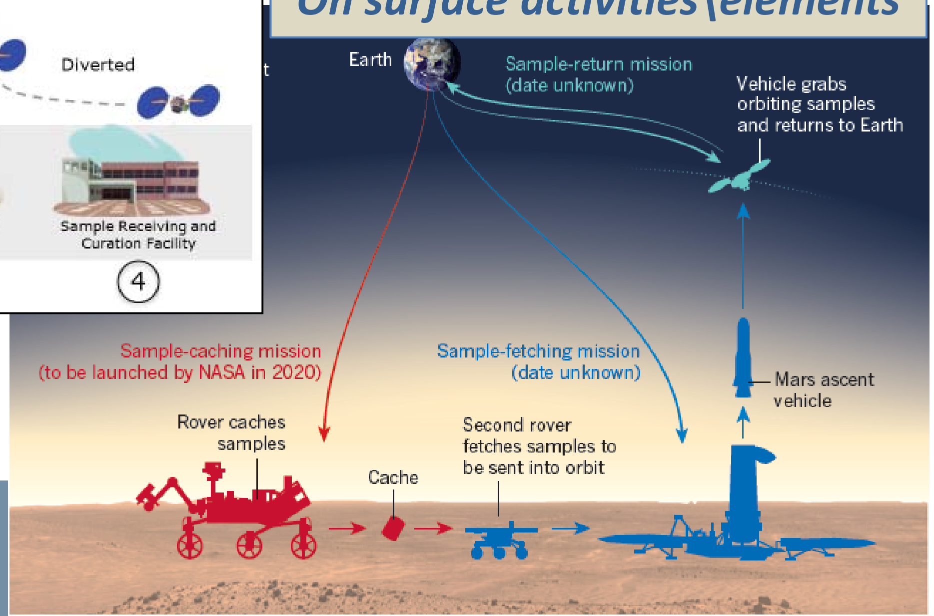
Define Conceptual Operations (ConOps)

Mars Sample Return Example

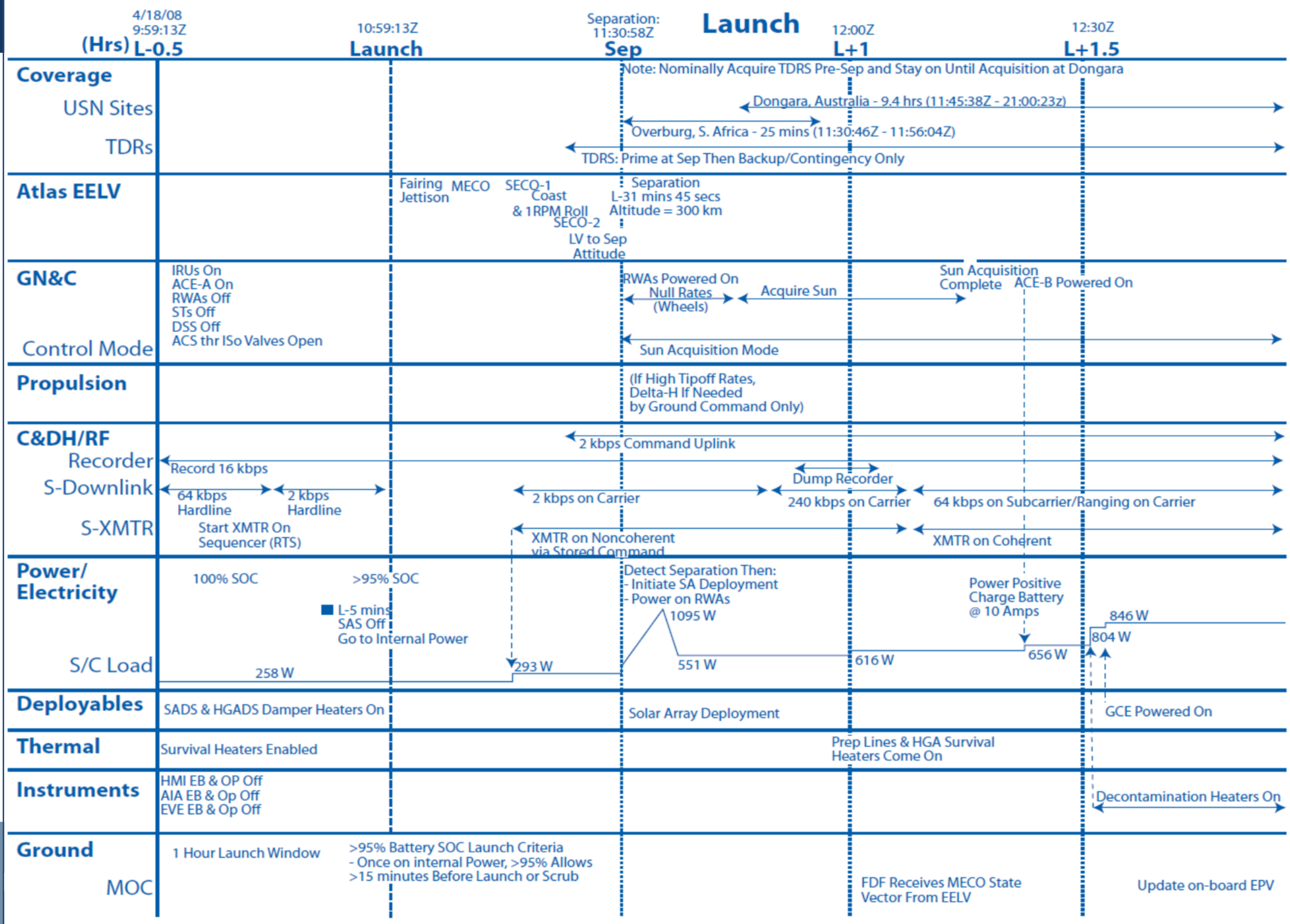


Overall mission

On surface activities\elements



System Engineering notes: Conceptual Operations - ConOps



System Engineering notes: the requirements

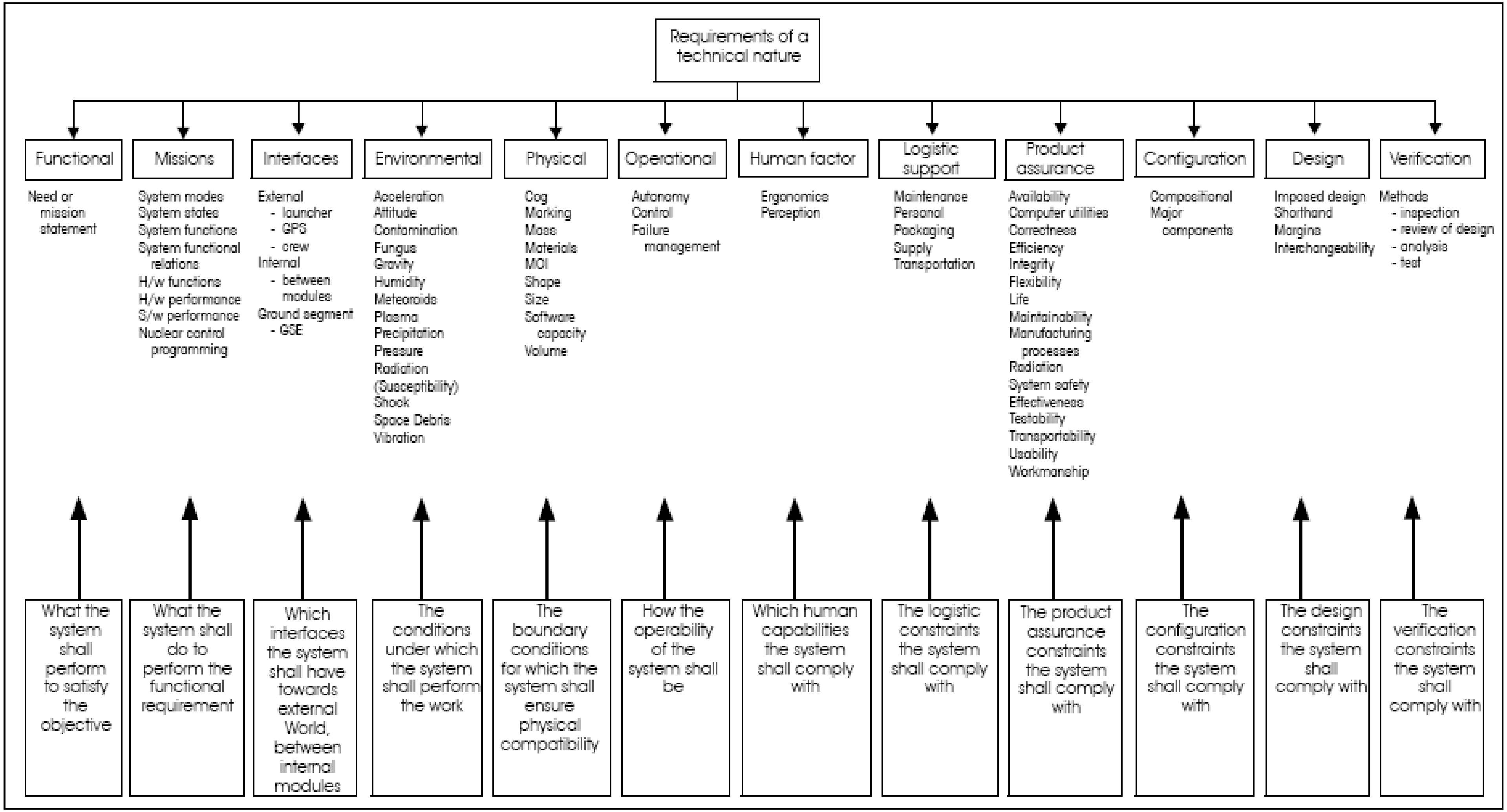
Requirements Engineering ECSS-E-ST-10-06C

A requirement transforms the **broad mission objectives →quantifiable design parameters** in terms of specific requirements and constraints

The key points to remember about **requirements** are the followings:

define "**what**" has to be done	→	**FUNCTIONAL**
define "**how**" it has to be done	→	**OPERATIONAL**
define "**how well**" the requirement should be satisfied	→	**PERFORMANCE**
define "**how**" the requirement should be verified	→	**VERIFICATION**

System Engineering notes: typical reqs



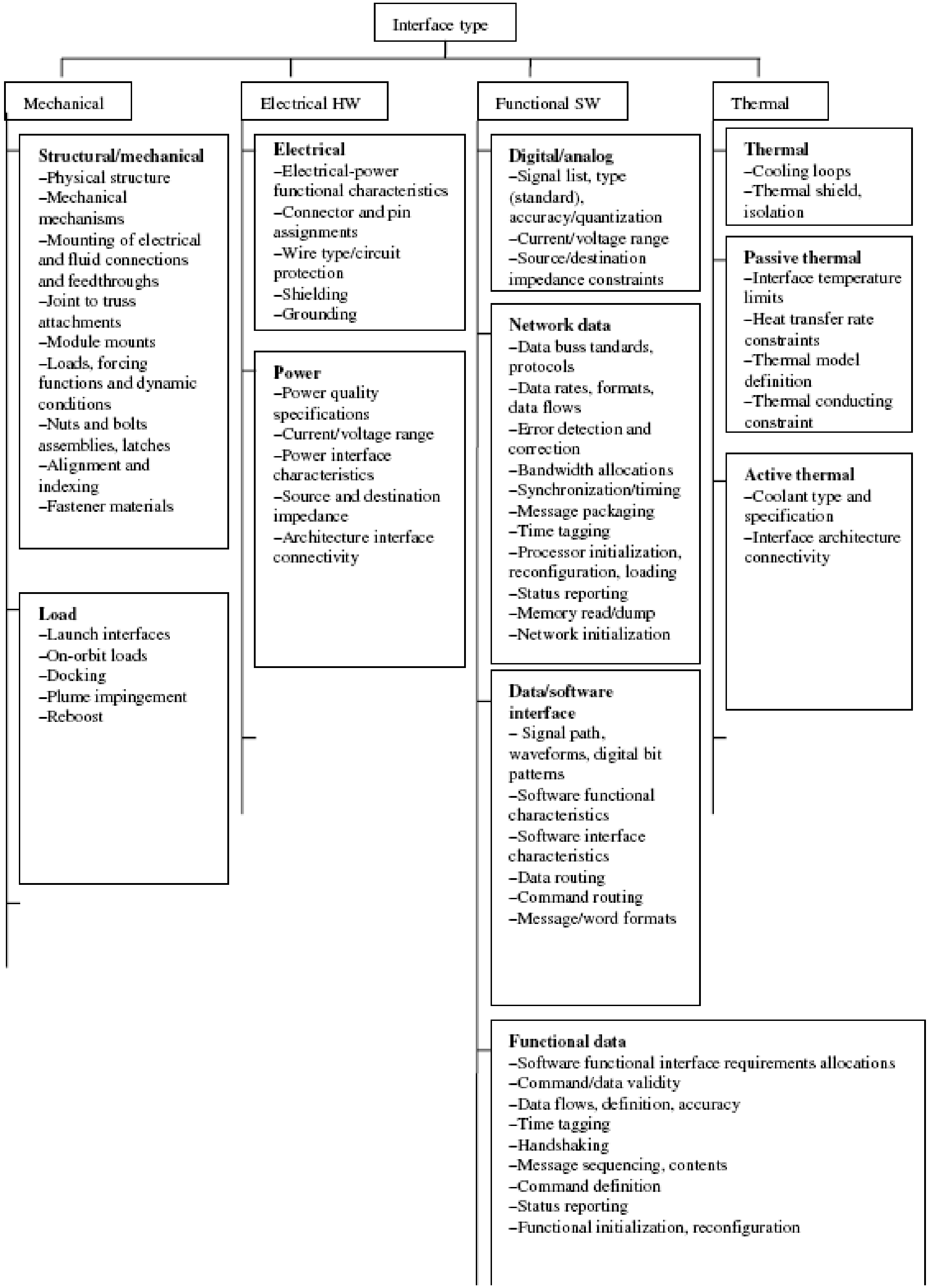
System Engineering notes - INTERFACES

Interfaces between subsystems and interfaces **between the main system and the external world** (launcher\payloads etc) must be designed.

- Subsystems should be defined to **minimize the amount of information to be exchanged between the subsystems**.
- Well-designed subsystems send **finished products** to other subsystems.
- Interfaces shall be clear, unambiguous, limited.

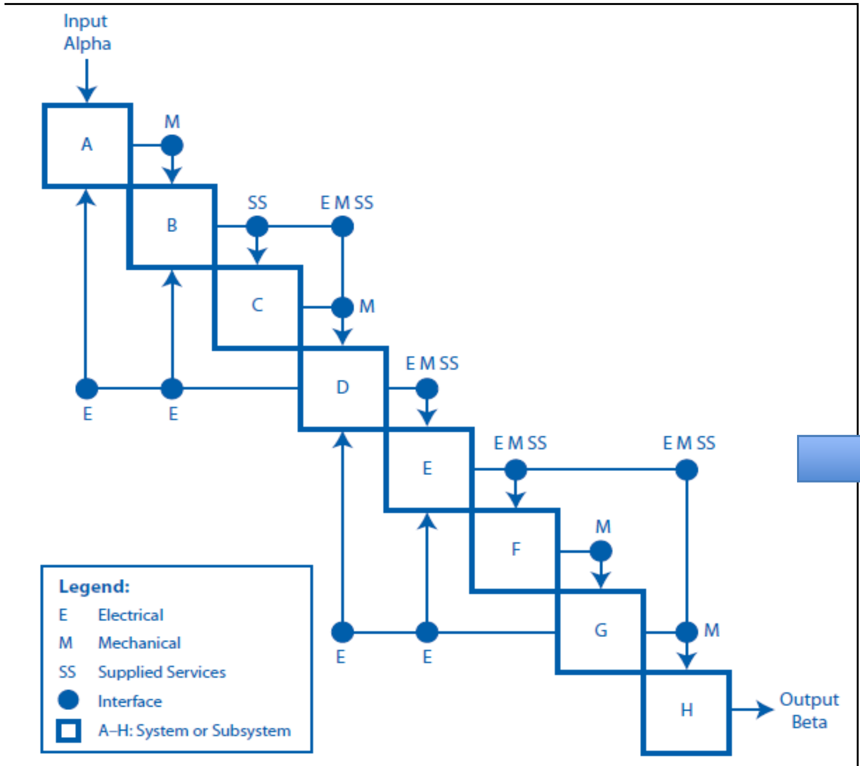
System Engineering INTERFACES notes

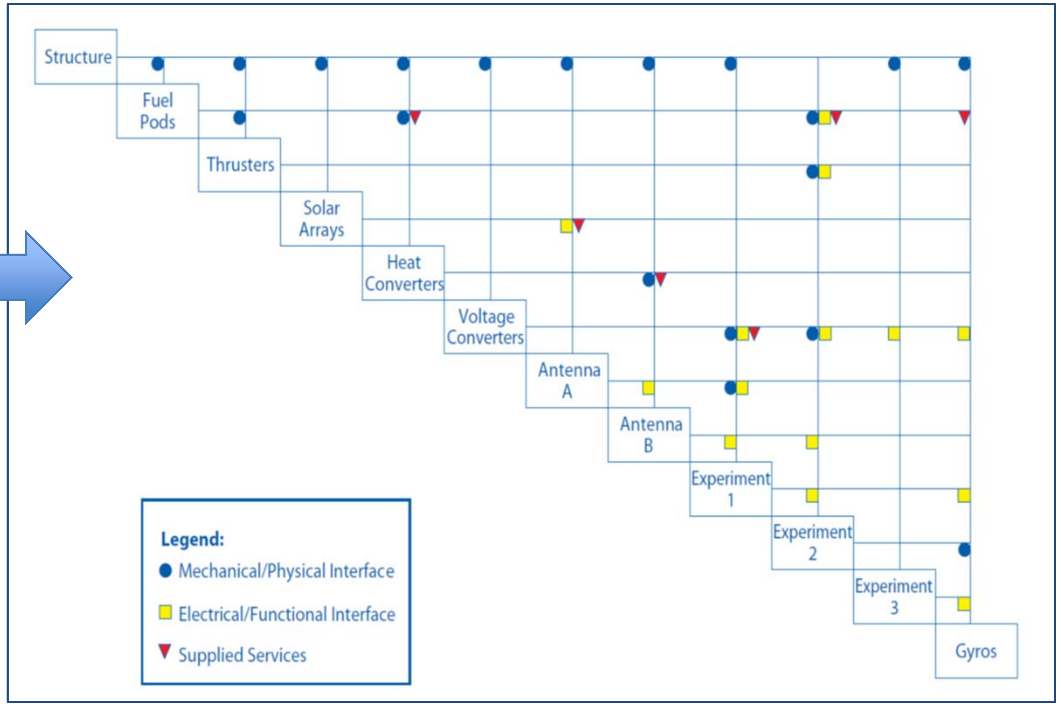
Existing interfaces



System Engineering notes: tools – INTERFACES - N2 diagram

The N2 diagram helps visualizing, highlighting and identifying **interfaces among subsystems**





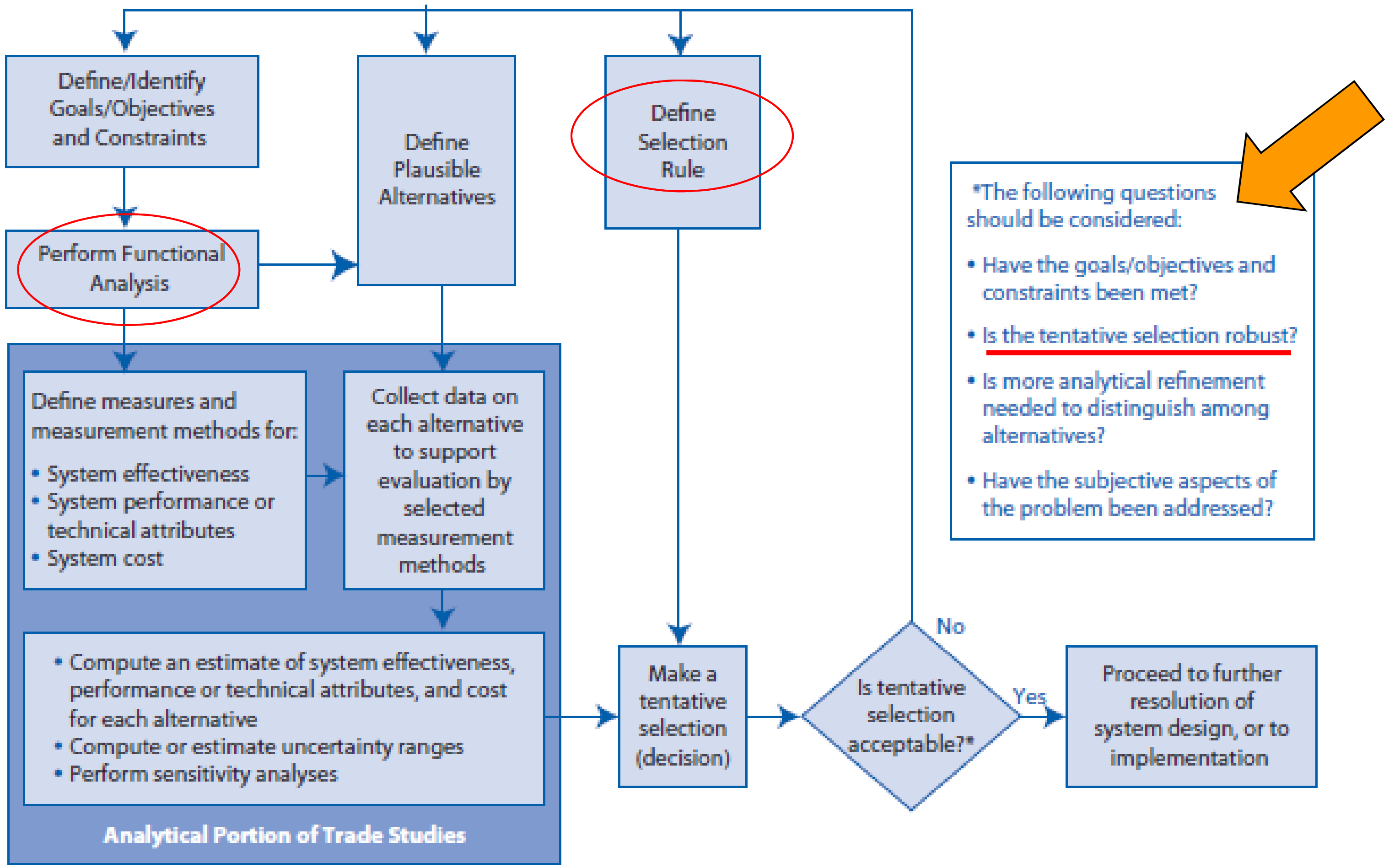
Exploring alternative desing concepts – trade-offs

Alternative designs shall be investigated and ranked according to multiple\ multidisciplinary CRITERIA

For the design of complex systems, alternative designs reduce project risk.

System Engineering notes

Explore Alternatives – TRADE OFF



System Engineering notes– trade process example

Manned Lunar south pole mission

Element Options:

A. Launch Crew and Cargo Separately

B. Launch Crew and Cargo Together

Launch Vehicle Choice: C. Ariane V, D. Shuttle, E. combination

F. Integrate at ISS

G. Launch pre-integrated

Transfer Propulsion Type: H. Chemical, I. Ion

J. Stage in lunar orbit

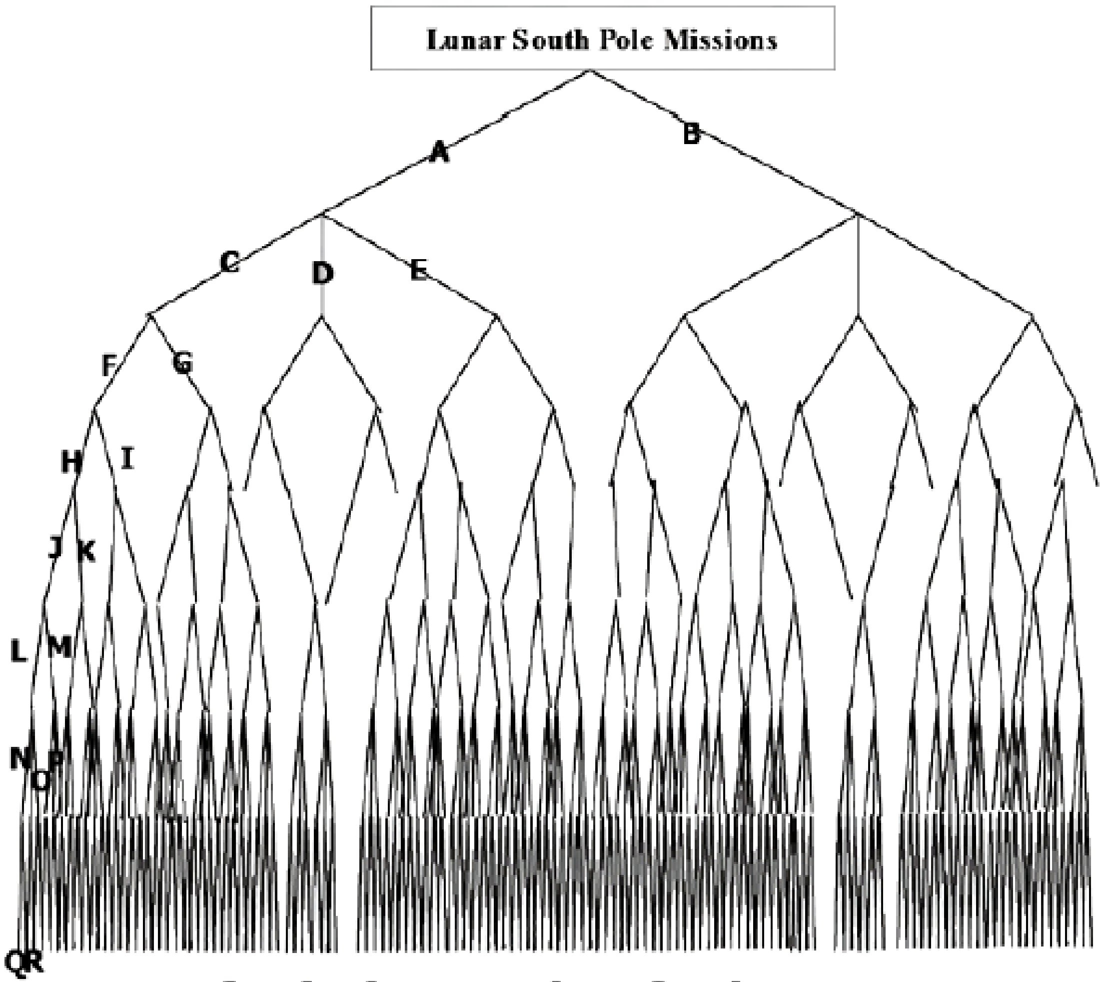
K. Direct to surface

Descent propulsion type: L. cryogenic or M. storable

Lunar Surface Duration: N. 3 days, O. 14 days, P. 28 days

Surface Infrastructure: Q. Self-contained in Lander, R. assembled base

...



576 Mission Design Options!

System Engineering notes– trade process options

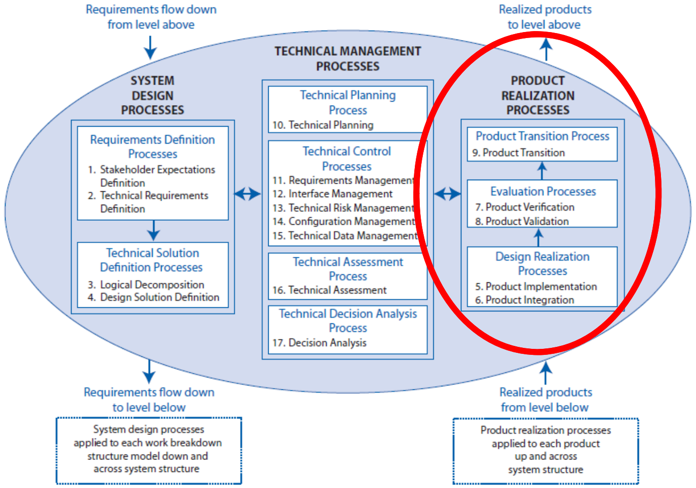
Development Related	Operations and Support Related
• Custom versus commercial-off-the-shelf	• Upgrade versus new start
• Light parts (expensive) versus heavy parts (less expensive)	• Manned versus unmanned
• On-board versus remote processing	• Autonomous versus remotely controlled
• Radio frequency versus optical links	• System of systems versus stand-alone system
• Levels of margin versus cost/risk	• One long-life unit versus many short-life units
• Class S versus non-class S parts	• Low Earth orbit versus medium Earth orbit versus geostationary orbit versus high Earth orbit
• Radiation-hardened versus standard components	• Single satellite versus constellation
• Levels of redundancy	• Launch vehicle type (e.g., Atlas versus Titan)
• Degrees of quality assurance	• Single stage versus multistage launch
• Built-in test versus remote diagnostics	• Repair in-situ versus bring down to ground
• Types of environmental exposure prior to operation	• Commercial versus Government assets
• Level of test (system versus subsystem)	• Limited versus public access
• Various life-cycle approaches (e.g., waterfall versus spiral versus incremental)	• Controlled versus uncontrolled reentry

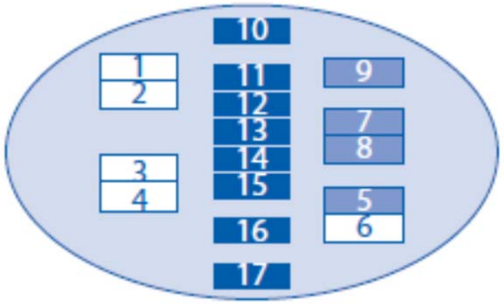
System Engineering notes– trade process options

Pre-Phase A	Phase A	Phase B	Phases C&D	Phases D&E	Phases E&F
• Problem selection • Upgrade versus new start	• On-board versus ground processing • Low Earth orbit versus geo-stationary orbit	• Levels of redundancy • Radio frequency links versus optical links	• Single source versus multiple suppliers • Level of testing	• Platform STS-28 versus STS-3a • Launch go-ahead (Go or No-Go)	• Adjust orbit daily versus weekly • Deorbit now versus later

Acquisition Phase	Trade Study Purpose
Mission needs analysis	Prioritize identified user needs
Concept exploration (concept and technology development)	1. Compare new technology with proven concepts 2. Select concepts best meeting mission needs 3. Select alternative system configurations 4. Focus on feasibility and affordability
Demonstration/validation	1. Select technology 2. Reduce alternative configurations to a testable number
Full-scale development (system development and demonstration)	1. Select component/part designs 2. Select test methods 3. Select operational test and evaluation quantities
Production	1. Examine effectiveness of all proposed design changes 2. Perform make/buy, process, rate, and location decisions

System Engineering notes: ingredients





DESIGN REALIZATION → **EVALUATION PROCESSES** → **PRODUCT TRANSITION PROCESS**

Product Implementation
- Acquire
- Make/Code
- Reuse

Product Integration
- Assembly
- Functional Evaluation

Product Verification
- Functional
- Environmental
- Operational Testing in Integration & Test Environment

Product Validation
- Operational Testing in Mission Environment

Product Transition
- Delivery to Next Higher Level in PBS
- Delivery to Operational System

System Engineering notes: MODELS

System implementation means the realization (buy or build) of the system elements.

- **Breadboard:** A **low fidelity unit** that **demonstrates function only**.It often uses commercial and/or ad hoc components
- **Engineering Unit:** A **high fidelity unit that demonstrates critical aspects** of the engineering processes involved in the development of the operational unit. Engineering test units are intended to closely resemble the final product (hardware/software) to the maximum extent possible and are built and tested so as to establish confidence that the design will function in the expected environments.
- **Prototype Unit:** The prototype unit demonstrates form, fit, and function **at a scale deemed to be representative** of the final product operating in its operational environment.
- **Qualification Unit:** A unit that is the **same as the flight unit** (form, fit, function, components, etc.) that will be **exposed to the extremes of the environmental criteria** (thermal, vibration, etc.). The unit will typically not be flown due to these off-nominal stresses.
- **Protoflight Unit:** In projects that will not develop a qualification unit, the flight unit may be designated as a protoflight unit and a limited version of qualification test ranges will be applied. This unit will be flown.
- **Flight Unit:** **The end product** that will be flown and will typically undergo **acceptance level testing**.

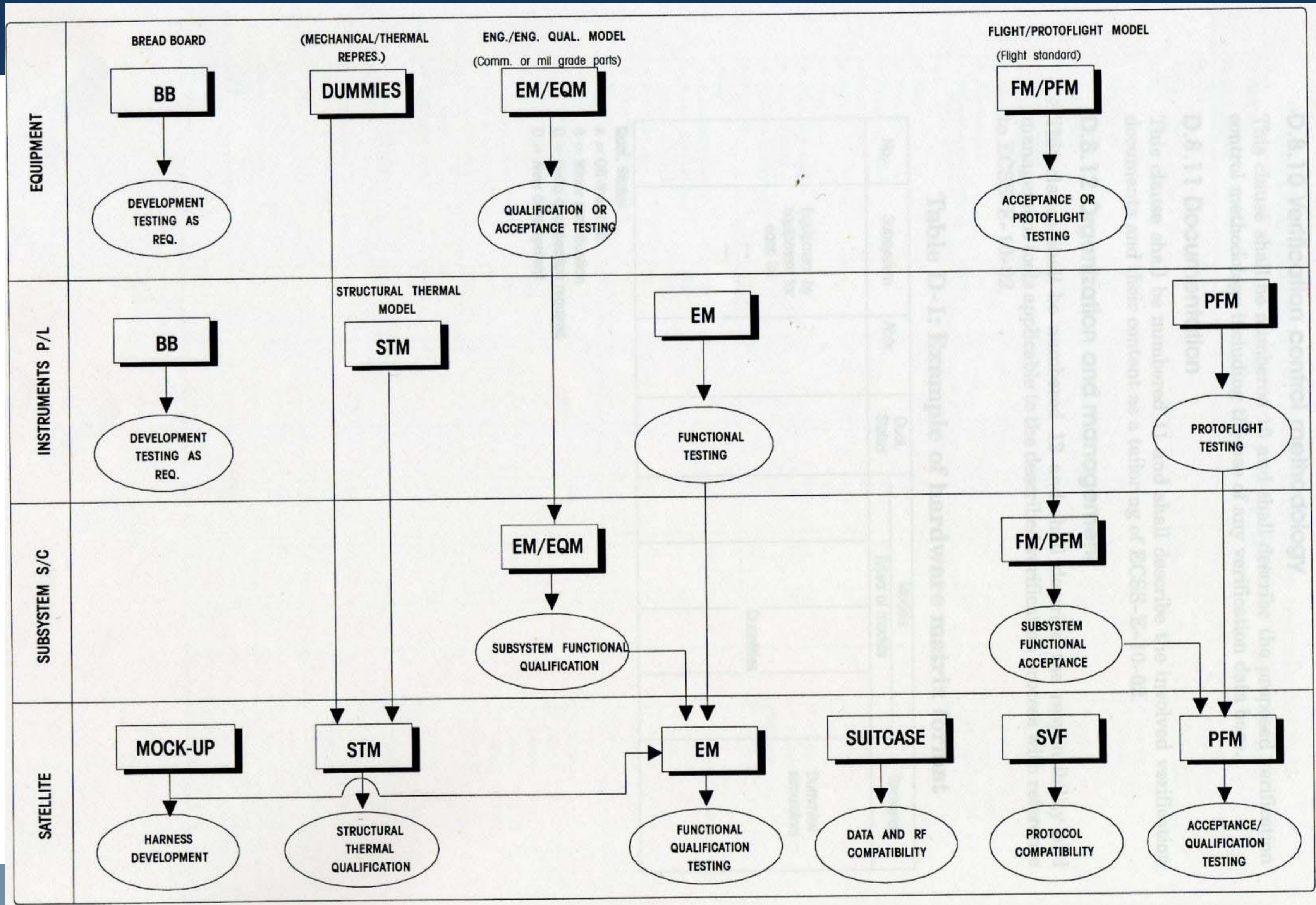
Verification is a process of confirming that a requirement or system is *compliant*
Verification answers the question:

Does the system meet its requirements?

Validation is a process confirming that a set of requirements, design or system meets the *intent* of the developer or customer.
Validation answers the question:

Are the system requirements correctly defined and mean what intended?

System Engineering notes: Models and Test plan



System Engineering notes: Environmental TESTS

Test	Purpose	Equipment/Facilities Required	Process
Vibration & Shock Testing	Ensure product will survive launch Comply with launch authority's requirements Validate structural models	Vibration table and fixture enabling 3-axis testing, and/or Acoustic chamber	• Do low-level vibration survey (a.k.a. modal survey) to determine vibration modes and establish baseline • Do high-level random vibration following profile provided by launch vehicle to prescribed levels • Repeat low-level survey to look for changes • Compare results to model
Thermal & Vacuum Testing	Induce and measure outgassing to ensure compliance with mission requirements Ensure product will perform in a vacuum under extreme flight temperatures Validate thermal models	Thermal/vacuum chamber Equipment to detect outgassing (e.g. coldfinger or gas analyzer) as needed Instrumentation to measure temperatures at key points on product (e.g. batteries)	• Operate and characterize performance at room temperature and pressure • Operate in thermal and/or thermal vacuum • chamber during hot and cold-soak conditions • Oscillate between hot and cold conditions and monitor performance • Compare results to model
Electromagnetic Interference/ Compatibility (EMI/EMC)	Ensure product does not generate EM energy that may interfere with other spacecraft components or with launch vehicle or range safety signals Verify that the product is not susceptible to the range and/or launch EM environment	Radiated test: Sensitive receiver, anechoic chamber, antenna with known gain Conduction susceptibility matched "box"	Detect emitted signals, especially at the harmonics of the clock frequencies Check for normal operation while injecting signals or power losses

System Engineering notes: Environmental TESTS

Communications and Tracking Labs	Models and Simulation Labs	Thermal Chambers
Power Systems Labs	Prototype Development Shops	Vibration Labs
Propulsion Test Stands	Calibration Labs	Radiation Labs
Mechanical/Structures Labs	Biological Labs	Animal Care Labs
Instrumentation Labs	Space Materials Curation Labs	Flight Hardware Storage Areas
Human Systems Labs	Electromagnetic Effects Labs	Design Visualization
Guidance and Navigation Labs	Materials Labs	Wiring Shops
Robotics Labs	Vacuum Chambers	NDE Labs
Software Development Environment	Mission Control Center	Logistics Warehouse
Meeting Rooms	Training Facilities	Conference Facilities
Education/Outreach Centers	Server Farms	Project Documentation Centers

Test facilities

Types of Testing

There are many different types of testing that can be used in verification of an end product. These examples are provided for consideration:

- Aerodynamic
- Burn-in
- Drop
- Environmental
- High-/Low-Voltage Limits
- Leak Rates
- Nominal
- Parametric
- Pressure Limits
- Security Checks
- Thermal Limits
- Acceptance
- Characterization
- Electromagnetic Compatibility
- G-loading
- Human Factors Engineering/ Human-in-the-Loop Testing
- Lifetime/Cycling
- Off-Nominal
- Performance
- Qualification Flow
- System
- Thermal Vacuum
- Acoustic
- Component
- Electromagnetic Interference
- Go or No-Go
- Integration
- Manufacturing/Random Defects
- Operational
- Pressure Cycling
- Structural Functional
- Thermal Cycling
- Vibration

Test types

Equipment acceptance baseline

Test	Recommended sequence	Category/type of equipment											
		a	b	c	d	e	f	g	h	i	j	k	l
Physical properties	1	R	R	R	R	R	R	R	R	R	R	R	R
Functional and performance	2[1]	R	R	R[6]	R	R	R	R	R	R	-	R	R
Leak	3,5,8,11	R[3]	-	R[3]	R	R	R	O	-	-	-	-	-
Pressure	4	-	-	R[3]	R	R	R	O	-	-	-	-	-
Random vibration	6	R	R[3]	R	R	R	R	R	R	R	R	R	-
Acoustic	6	O[9]	R[4]	-	-	-	-	-	-	-	O	O	R[11]
Shock	7	O	-	-	-	-	-	-	-	O	-	-	-
Thermal vacuum[5]	9[6]	R[2]	O	R[6]	R	R	O	R	R	R	O	R	R[11]
Thermal cycling[5]	9[6]	R	O	R[6]	R	R	O	R	R	R	O	R	O
Burn-in[10]	10	R	-	-	O	-	-	O	-	-	-	-	-
Microgravity[7]	12	R	-	-	R	-	-	-	-	-	-	-	-
Audible noise[8]	13	R	R	-	R	R	-	R	-	-	R	R	-

Categories

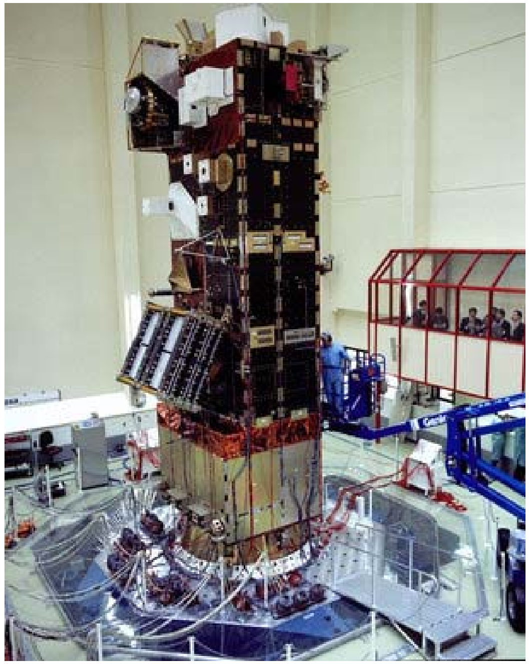
a = Electronic or electrical equipment
b = Antennas
c = Batteries
d = Valves
e = Fluid or propulsion equipment
f = Pressure vessels
g = Thrusters
h = Thermal equipment
i = Optical equipment
j = Mechanical equipment
k = Mechanical moving assemblies
l = Solar arrays

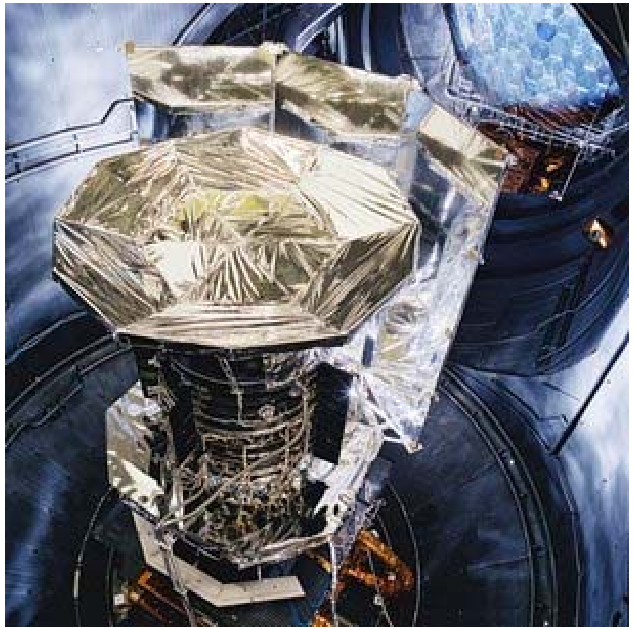
Legend

R = Required   O = Optional   - = Not required

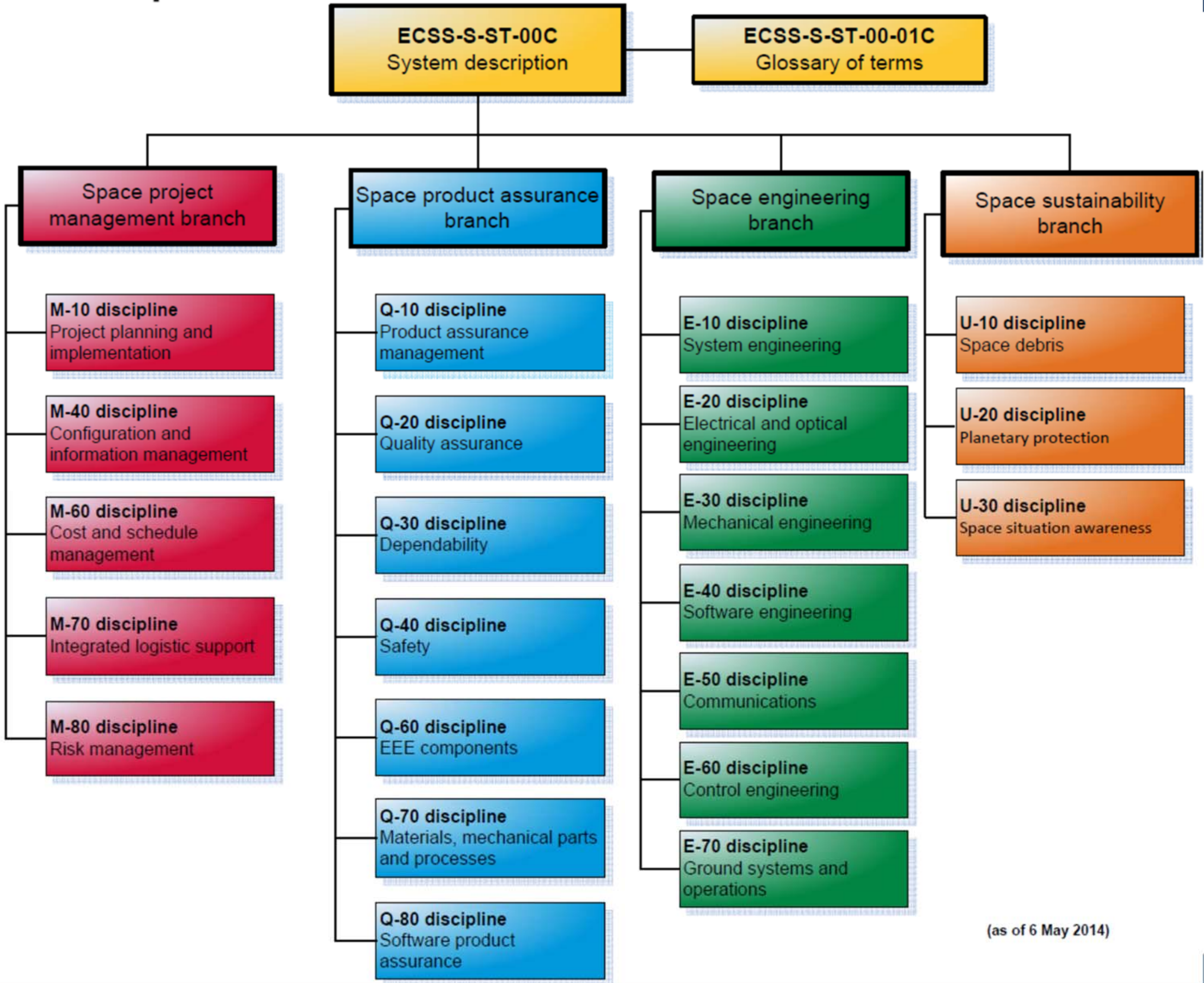
ECSS-E-10-03A

ESA tests facilities



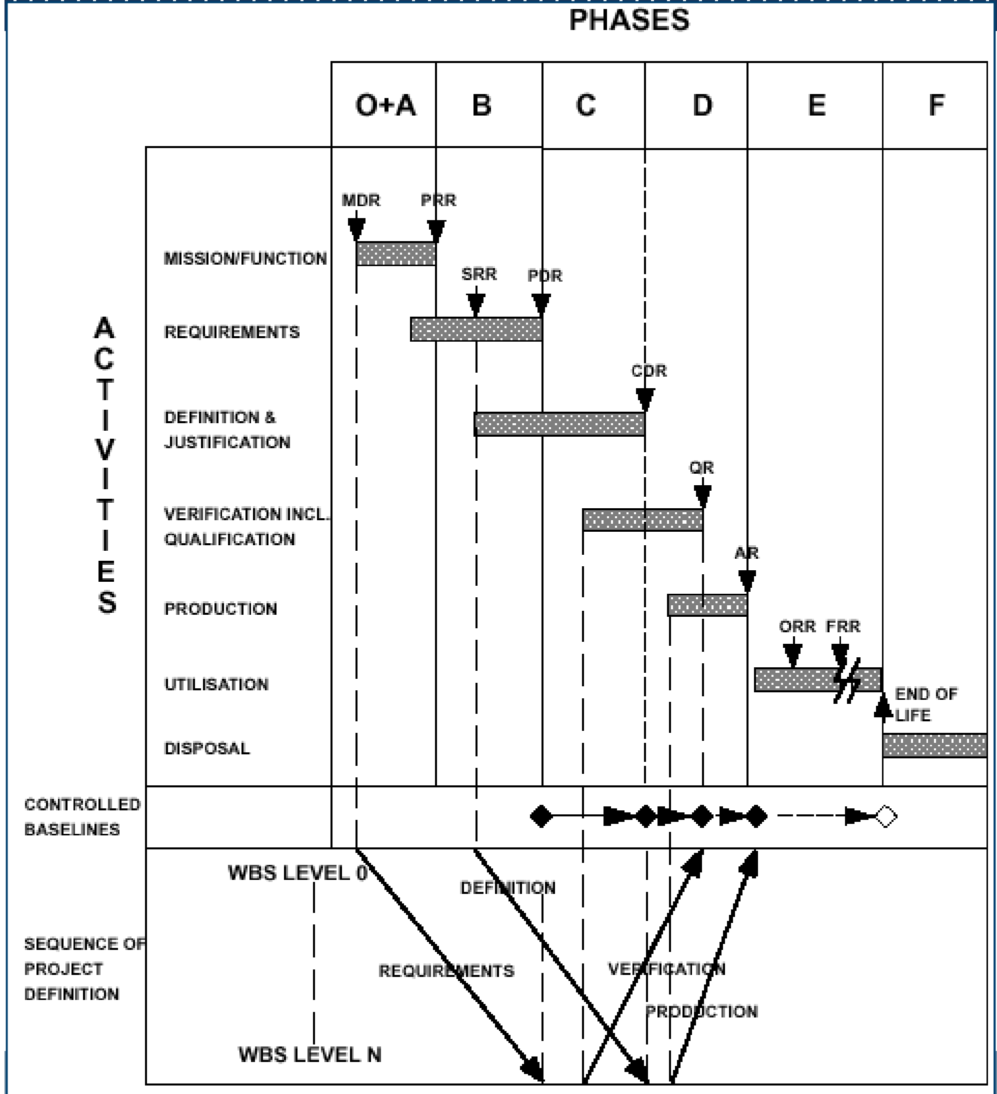


System Engineering notes:
European Cooperation for Space Standardisation (ECSS)



http://www.ecss.nl/

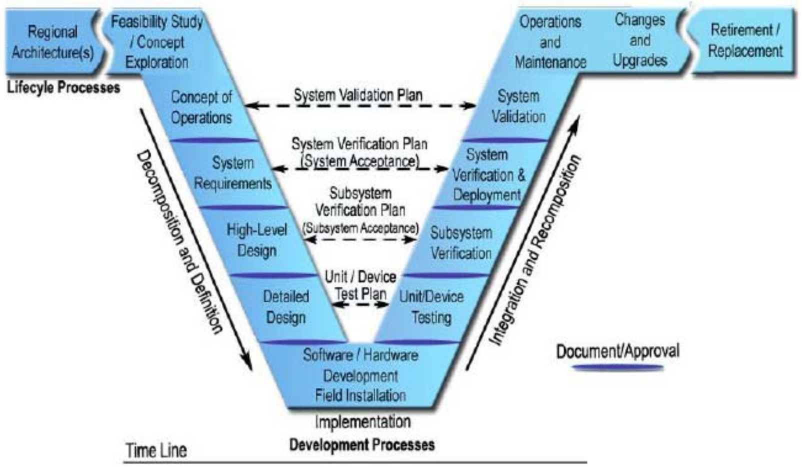
System Engineering notes



ECSS 30A ESA

NOTE AR = Acceptance Review
CDR = Critical Design Review
FRR = Flight Readiness Review
MDR = Mission Definition Review
ORR = Operational Readiness Review
PDR = Preliminary Design Review
PRR = Preliminary Requirements Review
QR = Qualification Review
SRR = System Requirements Review
WBS = Work Breakdown Structure

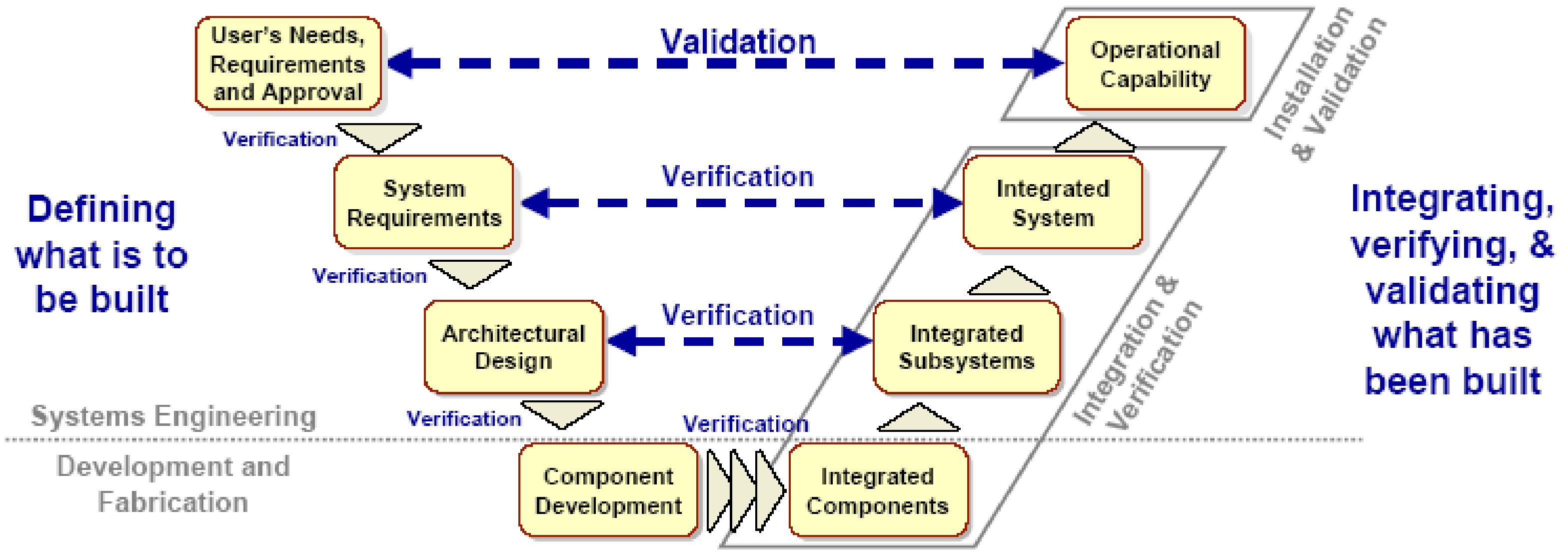
Figure 1: Typical project life cycle



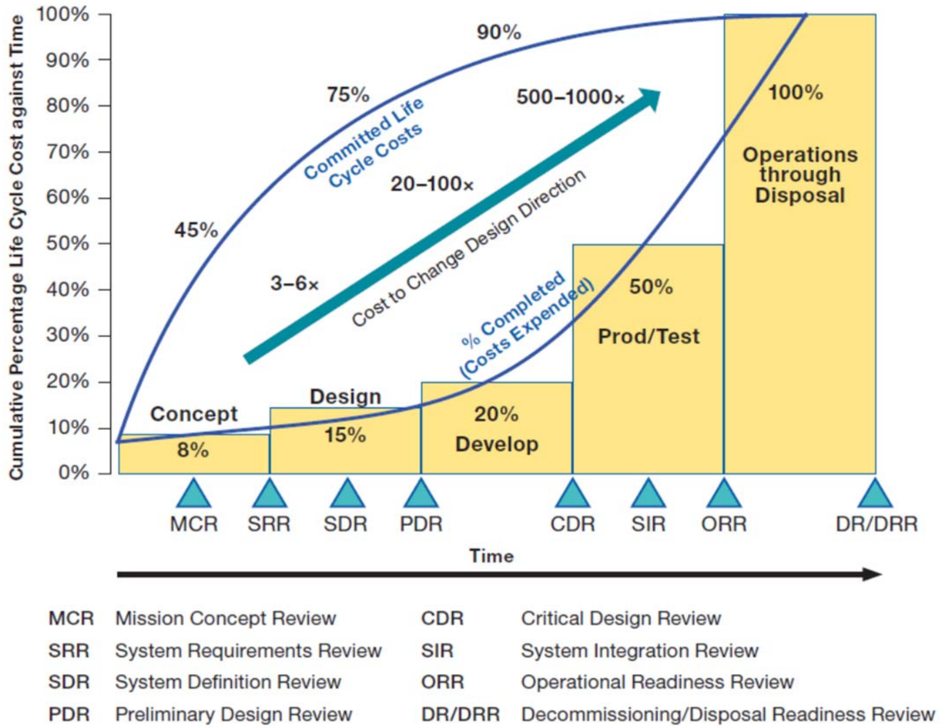
System Engineering notes: V-diagram

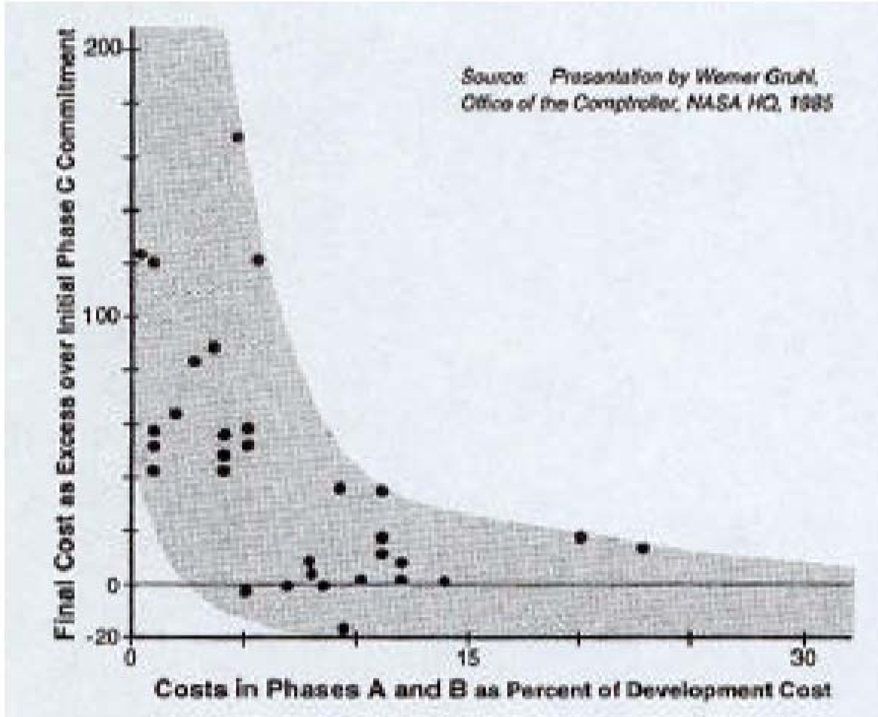
The simple life cycle can be re-organized as a V-diagram to emphasize:

- Verification between phases, checking what has been built against its reqs
- Validation as end-to-end verification ensuring that the complete system meets the user needs
- Decomposition and definition of what is to be built
- Integrating and verifying what has been built



System Engineering notes





Overruns are very likely if phases A and B are underfunded

Alpbach Summer School July 17-26,2018

POLITECNICO
MILANO 1863

System Engineering and Technology

Michèle Lavagna

michelle.lavagna@polimi.it