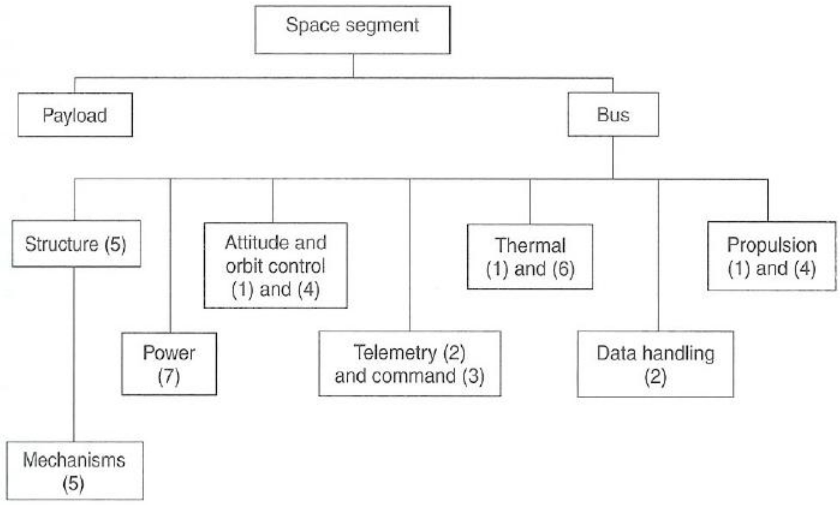
SSD

1 Spacecraft System Design

Mission concept:

- subject (what for)
- orbit and constellation
- payload, bus
- launch element
- ground element
- mission operations
- command, communication, control



2 Space Dynamics/Kepler Orbits

Typical coordinate systems:

- spacecraft-fixed
  - Mittelpunkt des Satelliten = Ursprung
  - nadir = z-Achse, nominale Geschwindigkeit = x-Achse
  - gut, um Position und Orientierung der Satelliteninstrumente festzustellen
- earth-fixed
  - Mittelpunkt der Erde = Ursprung
  - durch greenwich meridian = x-Achse
  - Geolocation, Satellitenbewegung
- roll, pitch and yaw-coordinates
- celestial coordinates
  - Mittelpunkt der Erde = Ursprung
  - Richtung Frühlingspunkt = x-Achse
  - Orbitanalyse, Astronomie

Keplergesetze

1. der Orbit eines jeden Planeten ist eine Ellipse, wobei die Sonne in einem der Fixpunkte liegt
2. die Verbindungslinie zwischen Sonne und Planet überstreicht in gleichen Zeiten gleiche Flächen
3. die Quadrate der Umlaufzeiten sind proportional zu den Kuben der großen Halbachsen

Ellipsendinge

- a ... große Halbachse
- ε, e ... Exzentrizität, "Abplattung" der Ellipse (ε=0: Kreis, ε=1: Parabel, $0 < \varepsilon < 1$: Ellipse)

Begriffe:

- Periapsis: Punkt der Ellipse, der am nähesten an dem Zentralkörper liegt (bei Sonne: Perihel, bei Erde: Perigäum)
- Apoapsis: Punkt der Ellipse, der am weitesten entfernt vom Zentralkörper liegt (bei Sonne: Apohel, bei Erde: Apogäum)
- Distanz zu Periapsis $r_p = a(1-\varepsilon)$, Distanz zu Apoapsis $r_a = a(1+\varepsilon)$

6 Bahnelemente:

Lieblingsformel

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

Change of the right ascension of the ascending node

$$\Delta\Omega = -\frac{3\pi J_2 R_E^2}{a^2(1-\varepsilon^2)^2} cos(i)$$

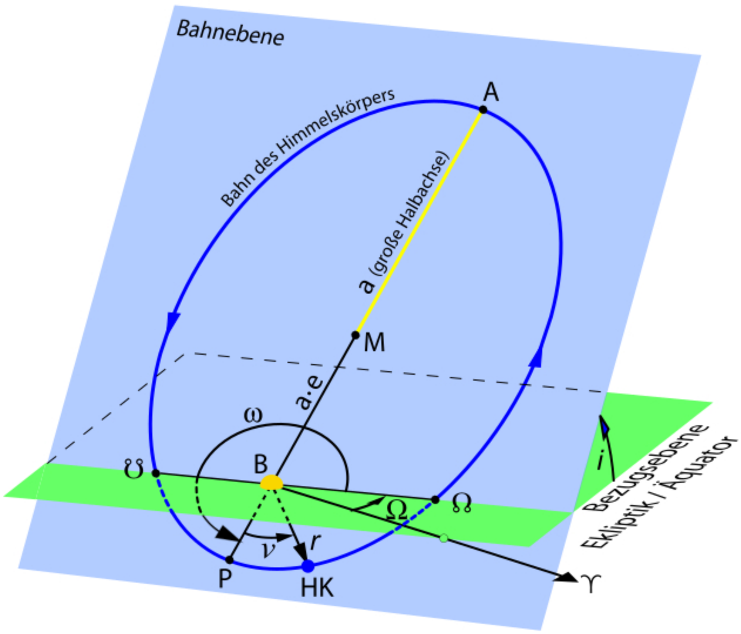
Change of the argument of perigee

$$\Delta\omega = \frac{3\pi J_2 R_E^2}{2a^2(1-\varepsilon^2)^2}(4 - 5sin^2(i))$$

Orbits

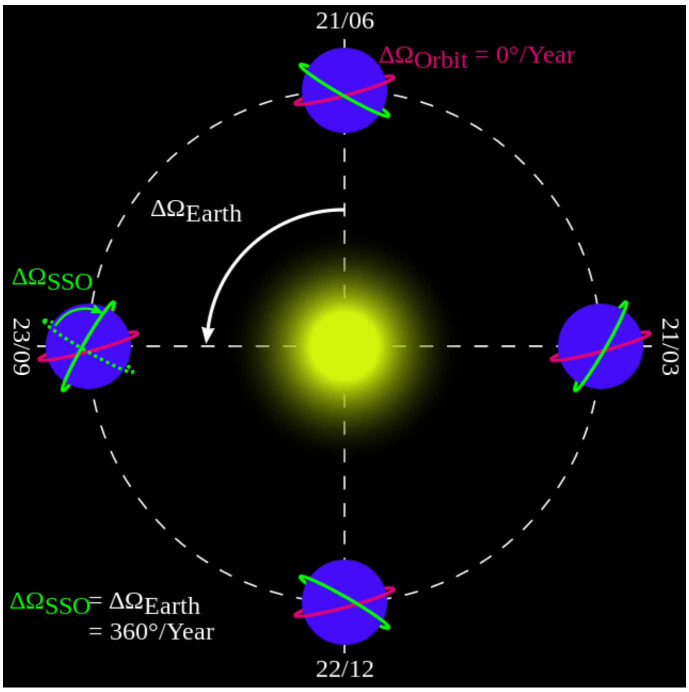
1. Highly Elliptical Orbit HEO

- große Halbachse a
- Exzentrizität ε
- inclination i
- right ascension of the ascending node Ω
- argument of perigee ω
- true anomaly ν



- hohe Exzentrizität
- große Halbachsen
- dadurch lange Kontakdauer zum Satelliten
- Werte für Perigäum: 200 bis 15.000 km
- Werte für Apogäum: 50.000 bis 140.000 km
- für Forschung (z.B. Weltraumteleskope), Telekommunikation, Militär
- Beispiel: Molniya-Orbit (feste Inklination von 63,4°, Periodendauer von einem halben Sterntag (23h56m4s))

2. Sun-Synchronous Orbit

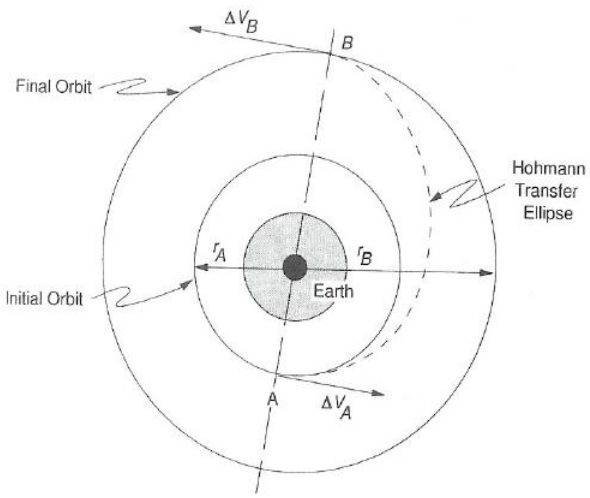


- Höhe und Inklination werden so kombiniert, dass ein Satellite aus Sicht der Sonne immer auf dem selben Orbit ist
- Höhe: 600-800 km

- Inklination: leicht retrograd ($\approx 98°$)
- Umlaufdauer: 96-100min

3. Geostationary Orbit GEO
   - kreisförmiger Orbit
   - Höhe: 35.786km
   - Umlaufdauer: 24h
   - Wettersatelliten, Kommunikationssatelliten, Fernsehsatelliten

Subsatellite Point = intersection of the line between satellite and earth center with the earth's surface
Hohmann Transfer



- Calculate a transfer between two circular orbits with radius r_A to r_B. The velocity at pericenter of the transfer ellipse:
$$v_P^2 = 2\mu\left(\frac{1}{r_A} - \frac{1}{r_A + r_B}\right) = 2\mu\frac{r_B}{r_A(r_A + r_B)}$$
- The required Δv_A to inject from the transfer orbit:
$$\Delta v_A = v_P - v_A = \sqrt{\frac{\mu}{r_A}}\left(\sqrt{\frac{2r_B}{r_A + r_B}} - 1\right)$$
- The required Δv to inject from the transfer orbit into orbit with r_B:
$$\Delta v_B = v_B - v_{\text{apo}} = \sqrt{\frac{\mu}{r_B}}\left(1 - \sqrt{\frac{2r_A}{r_A + r_B}}\right)$$
where v_B is the circular velocity at r_B.
- The Hohmann transfer is the most energy-efficient transfer between two circular orbits.
$$\Delta v_{\text{total}} = \Delta v_A + \Delta v_B = \sqrt{\mu}\left[\sqrt{\left(\frac{2}{r_A} - \frac{2}{r_A + r_B}\right)} - \sqrt{\frac{1}{r_A}} + \sqrt{\frac{2}{r_B} - \frac{2}{r_A + r_B}} - \sqrt{\frac{1}{r_B}}\right]$$

3 Mission Analyses

Earth-Synchronous Orbit

- the ground track repeats after a specific period of time
- Earth's rotation rate is the sidereal rotation period = sidereal day τ_{E}
- τ_E is varying with time $\tau_E = 86164.10555 + 0.15 \cdot C$ [s] where C is the centuries since year 2000
- as the Earth rotates eastward, the satellite is thus moving relative to the surface in westward direction by
$$\Delta\Phi_r = 2\pi \frac{T}{\tau_{\mathrm{E}}} \text{ [rad/rev]}$$
- second effect influencing the shift of the subsatellite point is the rotation of the satellite's orbit plane $\Delta\Omega$
- as $\Delta\Omega$ is positive in eastward direction, these two effects are combined to the total angular shift $\Delta\Phi$ at subsequent equator passages
$$\Delta\Phi = \Delta\Phi_r - \Delta\Omega \text{ [rad/rev]}$$
- to be Earth-Synchronous:
$$n\Delta\Phi = m \cdot 2\pi$$

Sun-Synchronous Orbit

- die Erde braucht $\tau_S = 3.155815 \cdot 10^7 s$, um einmal um die Sonne zu kreisen
- bei einem sonnensynchronen Orbit muss der Winkel zwischen Sonnenrichtung und Orbitebene konstant bleiben
- also muss sich die Ebene pro Tag um einen Winkel θ drehen
$$\theta = 2\pi \frac{\tau_{\mathrm{E}}}{\tau_{\mathrm{S}}} \text{ [rad/day]} = 2\pi \frac{\tau_{\mathrm{E}}}{\tau_{\mathrm{S}}} \frac{T}{\tau_{\mathrm{E}}} \text{ [rad/rev]}$$

Earth- and Sun-Synchronous Orbit

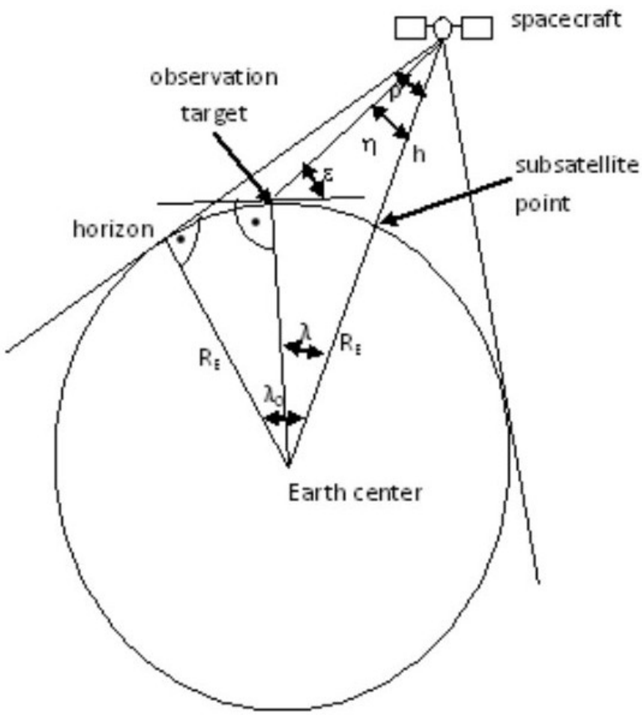
- $$\Delta\Omega = \theta \Rightarrow T\left(\frac{1}{\tau_{\mathrm{E}}} - \frac{1}{\tau_{\mathrm{S}}}\right) = \frac{m}{n}$$
- angular shift between two subsequent orbits
$$\Delta\Phi = \Delta\Phi_r - \Delta\Omega = 2\pi T\left(\frac{1}{\tau_{\mathrm{E}}} - \frac{1}{\tau_{\mathrm{S}}}\right) \text{ [rad/rev]}$$
worst case between subsequent orbits $\Delta\Phi \cdot R_E$.

Eclipse periods angle between Earth-Sun vector and normal vector to orbit plane: $\sin\beta = \vec{s} \cdot \vec{n}$
Earth central angular radius at entry into eclipse: $\beta^* = \sin^{-1}\left(\frac{R_E}{h+R_E}\right)$
Angular arc of orbit in shadow: $2\cos^{-1}\left(\frac{\cos\beta^*}{\cos\beta}\right)$ **Ground Contact and Coverage Analyses** altitude h, visible horizon characterized by angles ρ and λ_0: $\rho + \lambda_0 = 90°$

$$\begin{aligned} R_E &= (R_E + h)\cos\lambda_0 \\ &= (R_E + h)\sin\rho \end{aligned}$$

observe Λ_t, Θ_t (long,lat) from known orbit position of satellite, characterized by subsatellite point Λ_s, Θ_s.
characteristic paramters:

- nadir angle η
- earth central angle λ
- spacecraft elevation angle ε



calculate nadir angle η:

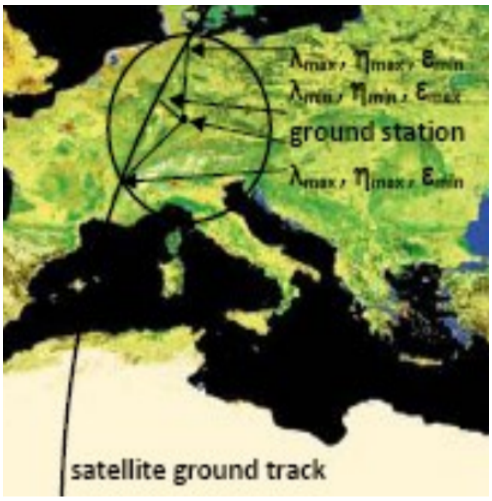
$$\tan\eta = \frac{\frac{R_E}{R_E+h}\sin\lambda}{1-\frac{R_E}{R_E+h}\cos\lambda}$$

$$\lambda + \eta + \varepsilon = 90°$$

$\lambda_{\max}$: maximum earth central angle $\Rightarrow$ swath width $2\lambda_{\max}$ perpendicular to groundtrack on surface.
Time in view T_{view} for circular orbit with period T:

$$T_{\text{view}} = \frac{T}{180°}\cos^{-1}\left(\frac{\cos\lambda_{\max}}{\cos\lambda}\right)$$

ground station contact periods



- $\sin\eta_{\max} = \cos\varepsilon_{\min}\frac{R_E}{R_E+h}$
- $\lambda_{\max} = 90° - \varepsilon_{\min} - \eta_{\max}$
- max range satellite↔groundstation: $D_{\max} = R_E\frac{\sin\lambda_{\max}}{\sin\eta_{\max}}$
- total time in view: $T_{\text{view}} = \frac{T}{180°}\cos^{-1}\left(\frac{\cos\lambda_{\max}}{\cos\lambda_{\min}}\right)$
- contact only possible, if station-orbit angle < central angle of contact cone

4 Distributed Satellite Systems

- **constellation**: similar trajectories without relative position control.
- **formation**: closed-loop onboard control for topology in the group.
- **swarm**: similar vehicles cooperating without fixed positions, selfdetermined.
- **cluster**: heterogenous system of vehicles for joint objective.

requirements on distributed satellite systems: coordination of

- orbits at different altitudes
- optimal control strategies for position/attitude of components
- activities for heterogenous sensors
- information flow and storage

Walker Delta Pattern Constellation

$$i : t/p/f$$

- i: inclination
- t: total $\sharp$ satellites
- p: $\sharp$ equally shaped orbit planes
- f: relative phase difference between satellites in adjacent planes

Example: Galileo is $56^\circ : 27/3/1$ with circular orbits ($h = 23222km$), nine satellites always in view, one spare satellite in each plane. **earth surface converage** $s = \frac{t}{p}$ number of satellites equally spaced in plane with angular distance $\Delta v = \frac{360^\circ}{s}$. There are two cases:

- $\Delta v < 2 \cdot \lambda_{\text{max}} \Rightarrow$ area of continuous coverage exists ("street of coverage")
- $\Delta v > 2 \cdot \lambda_{\text{max}} \Rightarrow$ no street of coverage

Street-width: $\cos \lambda_{\text{street}} = \frac{\cos \lambda_{\text{max}}}{\cos \frac{\Delta v}{2}}$

formation flying arcitectures and dynamics

- **virtual structure**: treated as single structure
- **behavioral strategies**: distributed control approach, following nature.
- **leader-follower**: divided into leaders and followers. followers track designated leaders with prescribed offset. absolute/relative control architecture.

communication in low-earth orbit distributed satellite systems

- comm and tele-operation infrastructure is key element for distributed systems
- transfer position and observation data for formation flying
- amount of data increases with swarm size
- analyse pre-processing procedures, intersatellite links and ground station links

conclusion on distributed satellite systems

- research field due to paradigm shift from one large spacecraft to several smaller crafts
- higher fault tolerance and robustness
- swarms are scaleable
- gun launches into orbit
- comination of big and small spacecrafts
- swarms for survailance and earth observation
- LEO $\rightarrow$ high spatial resolution
- higher temporal resolution is provided by constellations with several satellites in the same orbit

5 Mechanics

mechanical system engineering

- **mechanical specs**: requirements on satellite, components and equipment
- **verification plan**: "how to prove satellite complies specs?" test, simulations, similarity
- **test plan**: test flow, model philosophy (QM/FM↔PFM)
- **design loads**: simplified load cases for components & equipment

requirements on satellite structures

- external shape
- resonance frequency
- interfaces
- margin of safety
- mass, center of gravity
- thermo-elastic distortion
- environment (vacuum, debris, etc.)

random vibration loads

- white noise: range $20 - 2000Hz$, max levels at $80 - 300Hz$
- mainly acoustic excitation under fairing
- depends on location, orientation, mass
- equivalent design loads: 3 times root mean square (3 sigma value)

shock events

- launcher-induced: stage separation, fairing
- S/C release: clampband, discrete pyro devices
- appendage release: protechnic/deployment shock

structural engineering – fundamentals

- hooke's law: $\sigma = E \cdot \varepsilon$
- strain def: $\varepsilon = \frac{\Delta L}{L}$
- normal stress in rod: $\sigma = \frac{F}{A}$
- bending stress in beam: $\sigma = \frac{M}{W}$
- thermo-elastic strain: $\varepsilon = CTE \cdot \Delta T$

Margin of safety:

$$MOS = \left(\frac{S_a}{S_e \cdot FOS} - 1 \right) \cdot 100 \overset{!}{\geq} 0,0 \quad [\%]$$

- S_a allowable stress
- S_e applied stress
- FOS Factor of safety

safety factors:

- material safety factors on yield
- modelling safety factors covering analysis uncertainties
- specific factors, e.g. for bonded connections

material selection driven by

- ratio stiffness/mass
- functional aspects
- thermoelastic behavior
- sources
- ratio strength/mass
- compability with environment
- manufacturing complexity
- cost

test facilities

- electro-dynamic shaker testing orthogonal axes
- tasks: system identification – model correlation – model adaption

notching – reduction of dynamic loads – prevents exeeding design limit loads, preventing satellite structure damage. primary notching – whole satellite $\leftrightarrow$ secondary notching – satellite subsystems

tests

- acoustic noise tests
- separation tests

typical loads:

static	$< 10g$
quasi-static	up to $100g$
sine	up to $100g$
random	sometimes $> 100g$ (3 Sigma)
acoustic noise	$100dB$
shock	$2000g$

critical requirements

- zero grav
- launch loads
- extreme temperatues
- vacuum – lubrication is critical
- no maintainance

6 Thermal Engineering

general task

- engineering: define requirements and design
- analysis: establish thermal mathematical model (TMM), perform distribution calculations
- test: plan, perform, evaluate realistic tests

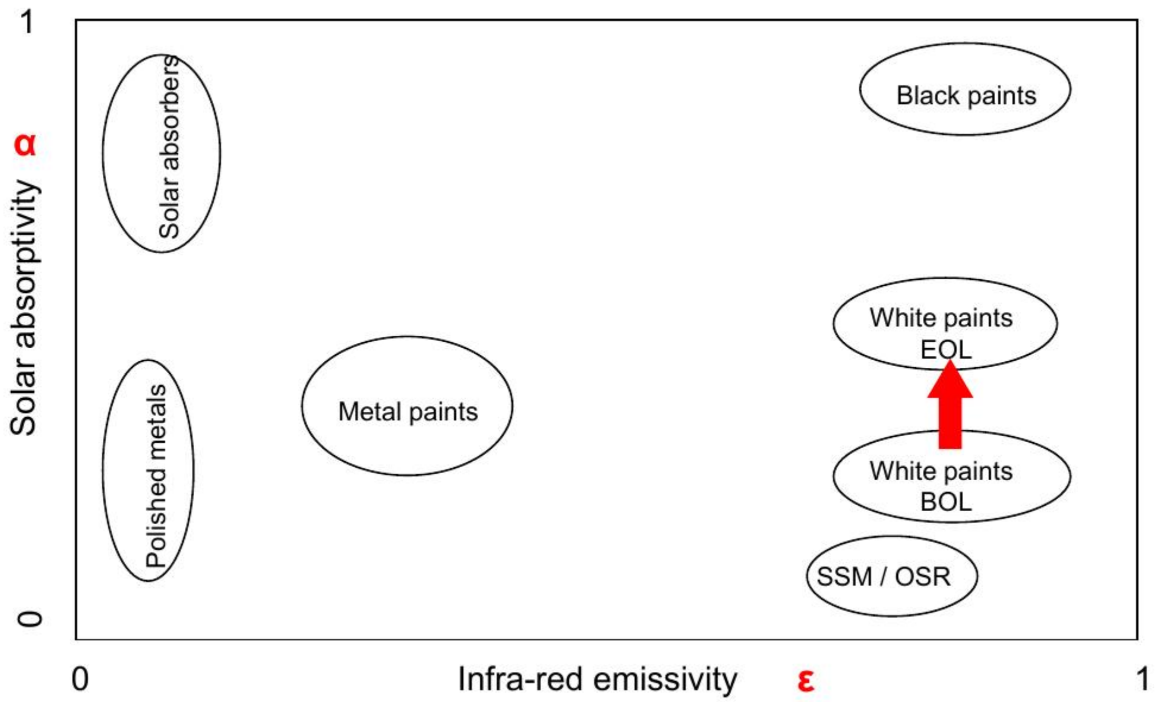
satellite thermal control – requirements

- temperature limits!
- temperature gradients, -stability, -uniformity
- heatflux, -storage
- power and mass-allocation

heat mechanisms:

- radiation: transfer via electromagnetic waves
- conduction: transfer via fluids and solids in absense of fluid motion
- convection: transfer in a flowing fluid and between fluid and wall (mostly irrelevant in satellites)

thermo optical properties of materials



global energy balance

- Solar intensity $\dot{q}_S = 1316 \ldots 1428 \frac{W}{m^2}$
- Earth IR Radiation $\dot{q}_E = 189 \ldots 261 \frac{W}{m^2}$
- Emissivity Coefficient $\varepsilon = 0 \ldots 1$
- Earth albedo $\dot{q}_A = 0.2 \ldots 0.4 \dot{q}_S$
- Absorbtivity Coefficient $\alpha = 0 \ldots 1$
- View Factor $\phi = 0 \ldots 1$

orbital environment and load cases:

- external loads (sun, earth, moon...)
- LEO, GEO, lagrange, eclipse?
- orientation (earth, sun, deepspace?)
- operating modes

- mission scenario

typical design load cases

	cold case	hot case
environment	cold external, BOL	hot external, EOL
solar intensity	min ($\sim 1320 \frac{W}{m^2}$)	max ($\sim 1420 \frac{W}{m^2}$)
earth albedo	min (~ 0.2)	max (~ 0.4)
earth IR radiation	min ($\sim 200 \frac{W}{m^2}$)	max ($\sim 260 \frac{W}{m^2}$)

thermal design – approach

- insulate against environment
- minimize absorbed heat
- balance internal heat
- define radiator-areas and distribute heat
- install thermal control hardware
- analyze and verify TCS (thermal control system)

critical components

- batteries
- detectors and sensors (instrument & star)
- optical equipment
- mechanisms, tubes, propulsion systems

thermal control hardware

- insulation, surface coating
- thermal interfillers
- thermal doublers, heat straps, heatpipes
- electrical heaters, temperature sensors

7 Rocket Propulsion

Propulsion systems: *(experimental)*

- chemical: solid, liquid, hybrid, *gelled*
- electrical: *electo-thermal, electrostatic*, electromagnetic
- photonic: *photon, solar sails*
- nuclear: *solid core, gas core, nuclear electric*
- cold gas thrusters

principals

- ejection of mass, provided by onboad means
- conservation of momentum, no momentum transfer to external medium
- continuous acceleration

staged vehicles

- tandem staged
- parallel staged

liquid Propulsion systems

- pressure feed system: high-pressure gas supply, pressure regulation, most simple and reliable
- turbopump feed system: propellant pressurized by pump, driven by turbine, high thrust and long duration

selection criteria

- performance: specific impulse, energy release per propellant mass, combustion, ignition, coolant performance
- economic: availability, cost, logistics
- handle: condition at ambient, non-toxic, non-corrosive, hazards

mono-propellants

- energy release by decomposition, stable under controlled environment
- ignition: thermally, catalytic
- advantages: simple tankage, feeding, flow, injection
- e.G.: hydrogen peroxide (H_2O_2), hydrazine (N_2H_4)

bipropellants

- chemical reaction of two propellants (O_2, H_2 or O_2, kerosene)
- separate storage, mixing
- high performance, safe operation
- **hypergolic propellants:** toxic, trained personel required, pollution risk at launch failure
- **cryogenic propellants:** gaseous at ambient, need thermal insulation, high power

combustion

- before chem. reaction, fuel has to atomize/evaporate
- mixing of propellants
- timescale: chem $\ll$ atomization, evaporation, mixing

- temperatur increase → gas volume increase → velocity increase
- chamber cooling: cooling fluids (fuel), film injection, thermal emission

ignition

- pyro: solid propellant, electrilly ignited
- spark plug: sparks ignite in combustion chamber
- spark torch: seperate igniter combustion chamber
- laser: beam focused in combustion chamber

solid propellants

- long time storage
- range of thrust levels: $2N \ldots 10MN$
- no moving parts, no service
- no shutoff, toxic
- applications: boosters, upper stage engines, tactical missiles, gas generation

electric propulsion

- electrothermal: heating of propellant by contact with hot metal
- electrostatic: acceleration of charged particels
- electromagnetic: acceleration of highly ionized plasma

launchers

	first launch	space ports	LEO	GTO
HII (Japan)	1994	Tanegashima	19 T	4-8 T
Soyuz (Russland)	1957	Baikonur/Plesetsk	6 T	1.3 T
Ariane 5 (Europa)	1996	Kourou	–	9.6 T

8 TT&C

terminology

- radiocommunication service RR20 – involving the transmission/reception of radio waves
- frequency allocation RR17 – entry in the table of frequencies of given band for radiocommunication services
- frequency assignment RR18 – authorisation to use radio frequency under conditions

communication delay

Location	Light-Time to get There
Across the Room	0.02 micro-seconds
Moon	a bit over 1 second
Sun	8 minutes
Edge of the Solar System	5 ½ hours
Nearest Star (Alpha Cent.)	4.3 years
Center of the Galaxy	~25 to 28 thousand years
Across Milky Way Galaxy	100,000 years
Nearest Galaxy (Andromeda)	1 million years

bands – frequencies & wavelength

Frequency band	Wavelength (cm)	Frequency (GHz)
Ka	0.8-1.1	40 - 26.5
K	1.1-1.7	26.5 - 18
Ku	1.7-2.4	18 - 12.5
X	2.4-3.8	12.5 -8
C	3.8-7.5	8 - 4
S	7.5-15	4 - 2
L	15 -30	2 - 1
P	30 -100	1 - 0.3

band usage

- S: SOHO, XMM-Newton, Cluster, Integral
- X: Mars Express, Rosetta, Venus Express, Herschel, Plank
- K_a: LISA Pathfinder, Gaia, James Webb Space Telescope, BepiColumbo

transponder operations

- Uplink Carrier, Carrier + Telecommand, Carrier + TC + Ranging
- Downlink Carrier, Carrier + Telemetry, Carrier + TM + Ranging
- auto-switch into coherent mode

reasons for modulation

- to separate signals
- to select correct frequency
- easier transmission

typical modulation types

- uplink: $2\frac{kbits}{s}$ bitstream is phase-modulated onto 16kHz carrier
- downlink: bi-phase stream directly phase-modulated onto carrier, residual carrier recovered at ground-station before demodulation

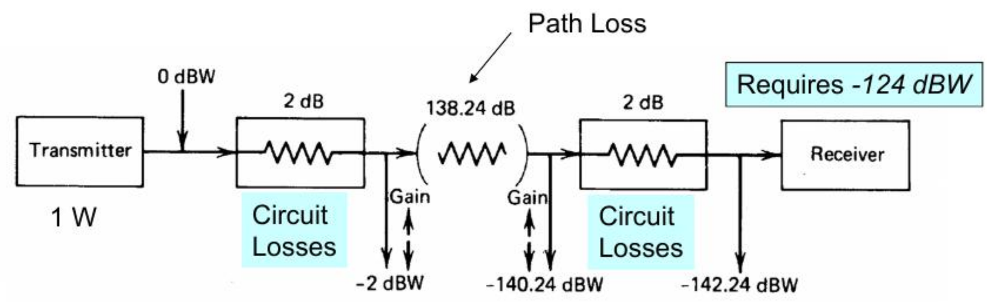
Link-Design Key Parameters

- Antenna Directivity and Gain
- Antenna Effective Area
- Dish Antenna Gain
- Free-Space Path Loss
- Effective Isotropic Radiated Power (EIRP) – product of transmit power and transmit antenna gain
- Thermal Noise – voltage fluctuations by moving charge carriers in conducting medium
- Figure-of-Merit (G/T) – capability to recieve signal

Directivity and gain

- all Antennas are stronger in one direction
- inverse square law of electromagnetic radiation $\frac{1}{r^2}$
- effective area of antenna is proportional to gain
- gain-beamwidth tradeoff: narrower beam↔more gain, less coverage → more stringent positioning of SC
- beamwidth inversly proportional to antenna size
- free-space path loss: doubling frequency implies 6 dB increase in path loss

basic link design



synchronisation

- signal influenced by: frequency offset, phase offset, hardware delays
- recievers try to:
  - detach information from carrier (frequency)
  - estimate and remove offsets
- demodulation and estimation
- phase lock loop techniques
- Typical Bandwidth: 800 Hz (near-Earth), or 20 Hz (Deep-Space)

system- & error budgets

- predicting and managing variability
- propagete errors through a system
- link aspects of design and environment to capabilities and tolerances

- determine and track critical parameters

channel encoding

- encoder takes k incoming bits, maps them to n outgoing bits ($n > k$)
- $n - k$ bits for error detection
- decoder does reverse process

9 Power Generation

- individual atoms have discrete energy levels
- probability of occupation at energy E

$$f(E) = \frac{1}{1 + e^{\frac{E-E_F}{kT}}}$$

- current is conducted via electrons in the conduction band (electrons)
- since there are vacant positions in the valence band, the electrons there can contribute to the current as well (holes)
- electrons and holes are treated as quasi free particles
- doping of semiconductors: replace a group of atoms by a group of atoms with lower or higher ordinal number (acceptors/donors)
- n-doped: higher ordinal number, p-doped: lower ordinal number
- so far: semiconductor in equilibrium, now: under illumination
- energy of light is added to the electron's energy which can lift the electron from the valence band to the conduction band
- recombination:
  - light creates electron hole pairs
  - if light is switched off: recombination
  - but also present: radiative recombination (cannot be prevented)
- pn junction: bring p- and n- doped semiconductor in contact
- upon forming the junction, there is a large concentration gradient and an associated diffusion current from holes leaving the p region and e- leaving the n region
- at the same time, a space charge is created by the ionized dopant atoms, in the resulting electrical field an opposing drift current develops
- in equilibrium, both currents are equal (for e- as well as holes) and no net current flows
- pn-junction under illumination = solar cell
- Summary solar cell principles
  - a semiconductor has a gap in the energy band diagram
  - at T¿0 free charge carriers (electrons, holes) exist
  - by doping, one type is increased dramatically which leads to the distinction majority/minority carrier
  - under illumination, minority carriers are created
  - due to the fact that there is a gap in the allowed energy levels, they don't relax immediately but have a finite lifetime τ
  - if they can be extracted before they recombine, they provide an external current $\Rightarrow$ solar cell
  - a pn junction does exactly that. The built-in field creates an asymmetry in the band structure. Majority carrier cannot cross it, but minority carriers can.
  - if a charge carrier crosses the pn junction, it is transformed from a minority carrier (e.g. e- in the p doped material into a majority carrier in the n doped side) with essentially infinite lifetime

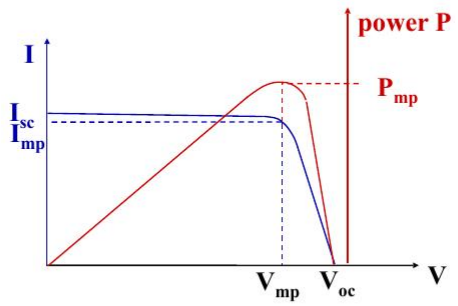
9.1 solar cell basics

For real solar cells, the following idealizations are not valid any more:

- infinite cell dimensions: real solar cells have surfaces, which are ideal recombination sites
- homogeneous carrier generation G: the absorption is wavelength and depth dependent (longer wavelength has larger penetration depth)
- recombination in the depleted region cannot always be neglected

Typically, the I-V curves are plotted in the first quadrant. Key parameters describing the IV curve:

- ppen circuit voltage V_{oc}
- short circuit current I_{sc}
- maximum power (P_{max} – V_{mp}/I_{Mp})



9.2 solar cells for space

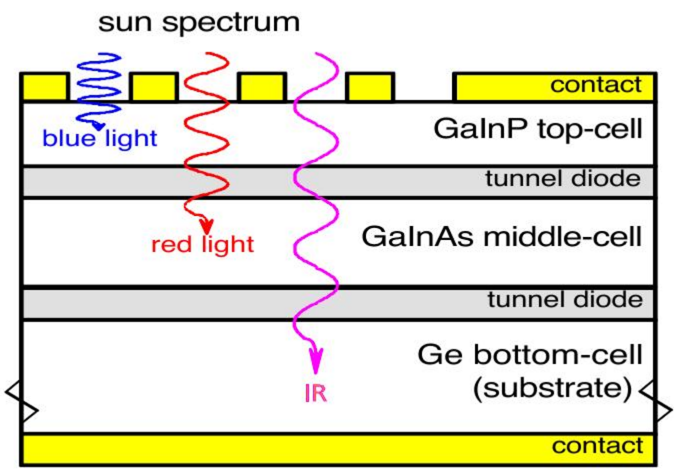
Main requirements:

- high efficiency
- low mass
- radiation resistant

Evolution: Si cells ⇒ cells based on direct semiconductors ⇒ multijunction cells.

multijunction cells

- efficiency of Si cells: 18%
- additional junction reduces thermalization losses and increases efficiency ⇒ multijunction cells



9.3 solar array technology

radiation environment in space:

- protons and electrons trapped in the earth magnetic field
- solar protons

damage caused by particle radiation:

- ionization damage
- displacement damage

matching of solar cells:

$$\text{current} I(S) = \sum I_\nu(U)$$

$$\text{voltage} V(S) = \sum V_\mu(I)$$

$$I(S) = n \cdot I_{cell}$$

$$V(S) = m \cdot V_{cell}$$

power prediction

1. mission profile
   - launch date
   - launcher
   - transfer orbits
   - final orbit
   - lifetime
   - power requirements and power profile
   - solar array orientation
2. satellite configuration
   - power control and power conditioning (fixed voltage or maximum power tracking)
   - solar generator type (body mounted, deployable fixed or sun oriented)
3. main parameters derived from orbit
   - intensity and incidence angle of sun insolation over mission time
   - effective Earth/planet radiation and albedo
   - type, spectrum and intensity of charged particle irradiation
   - loss factors BOM and EOM
   - optimum solar cell and coverglass type

power limiting factors
basic and design related:

- temperature
- calibration inaccuracy
- mismatch
- coverglass gain/loss
- cable losses
- random failures

mission related:

- sun intensity

- irradiation angle
- charged particles
- micrometeorites/debris

mechanical solar array design

so far: situation in orbit, now: mechanical criteria during satellite launch

- it has to be folded to the satellite sidewall in order to fit inside the launch vehicle
- it has to supply power during transfer orbit ($\rightarrow$ partial deployment)
- it has to be fully deployed once in geosynchronous orbit
- the mechanical design has to survive the acoustic loads (created by the main engines of the launch vehicle, reflected from the launch pad) and vibrational loads

10 Power System

- Power System Design
- Energy: Source and Generation
- Energy storage (Battery)
- Power Conditioning (PCU)
- Power Distribution (PDU)
- Thermal Control
- Reliability Aspects

10.1 Power System Design

- supply electrical power to spacecraft loads
- control and distribute electrical power
- meet average and peak electrical loads
- provide power conditioning and conversion
- provide command and telemetry capability
- protect spacecraft against EPS failure
- suppress transient bus voltage spikes
- provide energy storage for eclipse and peak demands
- provide specialized power for specific functions such as firing ordinance for mechanism deployment

10.2 Power System Functions

- power source (e.g. solar array, radio-isotope thermoelectric generator, nuclear reactor, primary batteries, ...)
- source control (regulators)
- power management and distribution
- power processors (dc/dc- or dc/ac-converters, regulators)
- energy storage control (charger, regulator)
- energy storage (batteries, flywheels)

10.3 Power Sources

- photovoltaic: conversion of solar radiation (light) to electrical current; solar generator equipped with Silicon (Si) or Gallium-Arsenid (GaAs) cells
- radio-isotope thermoelectric generators
  - deep space missions, military missions in low earth orbit
  - advantages: continuous power supply, no external supply needed, high reliability, small volume, low mass, high lifetime
  - disadvantages: safety measures needed during launch and launch preparation, shielding necessary to protect spacecraft, regulation in case of launch failure

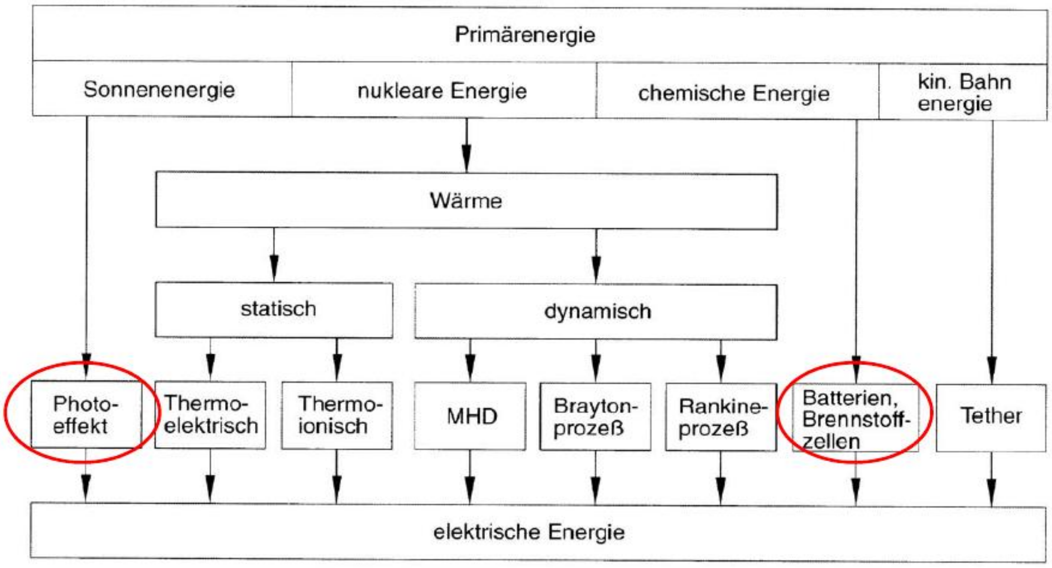


Abbildung 1: Energy Sources

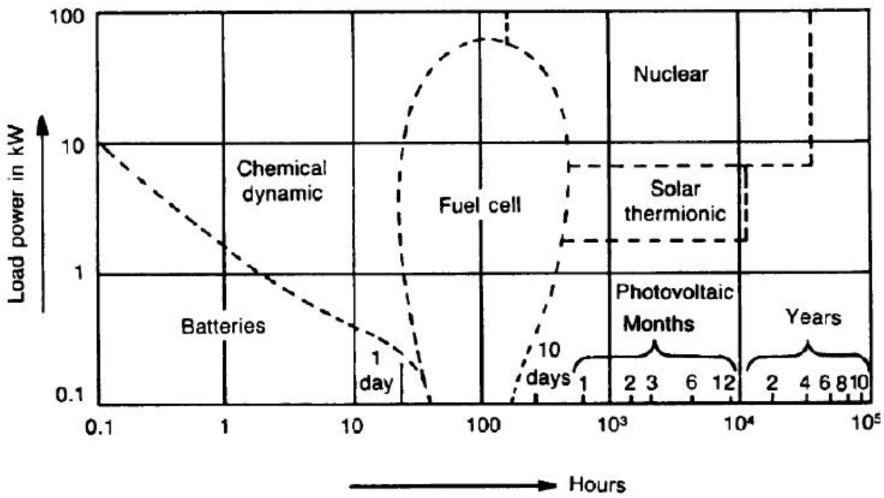


Abbildung 2: Comparison of Energy Sources

10.4 Energy Storage

- battery types: NiCD (no more used), NiH2 (mainly in telecommuniction satellites), Li-Ion (state of art), Li-Polymer (not yet used in space)
- fuel cells (ISS)

10.5 Source Control

- shunt regulator
- series regulator
- linear regulator
- peak power point tracker

10.6 Main Requirements and Design Parameters

- average and peak power: determines the size of the solar array
- peak power: determines size of battery capacity

- battery charging: determines the solar array
- maximum discharge DoD: $< 40\%$ for Li-Ion
- mission lifetime: determines the degradation und subsequently sizing of battery and solar array
- orbit geometry: determines the available solar energy, radiation environment and eclipse durations
- solar constant: $1358\frac{W}{m^2}$ mean above atmosphere. Seasonal variation: $1310 - 1400\frac{W}{m^2}$
- satellite must be able to recover from total power loss without assistance from ground

10.7 Radiation Effects

The following impacts on operation of electronic systems are associated with the charged particle environment:

- TID (total ionization dose): degradation of electronics which result from proton and electron degradation in semiconductor devices
- SEL (single event latch-up):
- SEB (single event burnout):
- SEU (single event upset):

$$\frac{\text{EOL}}{\text{BOL}} = \text{degradation}$$

10.8 Efficiency and Degradation Consideration

- production efficiency η of solar cells (14-22%)
- path efficiency from solar array through batteries to loads: $X_e = 0.65, X_d = 0.85$ (direct energy transfer), $X_e = 0.60, X_d = 0.80$ (peak power tracking)
- inherent degradation: $I_d \approx 0.77$, ranges from $0.49 - 0.88$
- cosine loss, angle Θ between array normal and sun vector; typically use worst-case sun-angle
- life degradation: micrometeorites, radiation, etc. (2-4% per year)

$$L_d = (1 - \text{degradation per year})^{\text{satellite life}}$$

10.9 From Begin of Life to End of Life

$$P_o = \eta \cdot 1358\frac{W}{m^2} \qquad \text{output power}$$
$$P_{BOL} = P_o \cdot I_d \cdot cos(\Theta)$$
$$P_{EOL} = L_d \cdot P_{BOL}$$

solar array size to meet power requirement:

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

mass of solar array ranges from 14 to $47\frac{W}{kg}$:

$$M_{sa} = 0.04 \cdot P_{sa} \qquad (\text{for } 25\frac{W}{kg})$$

10.10 Maximum Power Point Tracker (MPPT)

- continuously measures the power from the solar array and determines the maximum power point
- adjusts the solar array interface voltage such that the actual power demand of the spacecraft can be delivered
- maximum efficiency ($> 99\%$) can be achieved when the solar array interface voltage is close to the battery voltage

10.11 Pros and Cons of Power Regulation

- power damper: excessive power will be absorbed in high power resistors, switch control system for resistors ⇒ impact on thermal control system, as significant power dissipation occurs
- linear regulator: excessive power will be absorbed in high power transistor ⇒ impact on thermal control system, as significant power dissipation occurs
- shunt regulator: simple, robust, failsafe ⇒ requires high number of cells per string to ensure battery charging and minimum power
- MPPT: highest efficiency and minimum solar array size, complex in redundancy concept, each wing requires a dedicated MPPT

10.12 Battery Sizing

Power need:

$$P_{avg} = V_{bus} \cdot I$$

$$Ah_{avg} = \frac{T_e}{1h} \cdot I$$

$$Ah_{total} = \frac{Ah_{avg}}{DoD}$$

Capacity:

$$C_r = \frac{P_{avg} \cdot T_e}{DoD \cdot N_{bat} \cdot \eta}$$

10.13 Power Distribution

- protection by fuse
- electronic protections
- limit current in failure case
- isolate failed components from bus

11 Thermal Testing

12 Spacecraft Operations