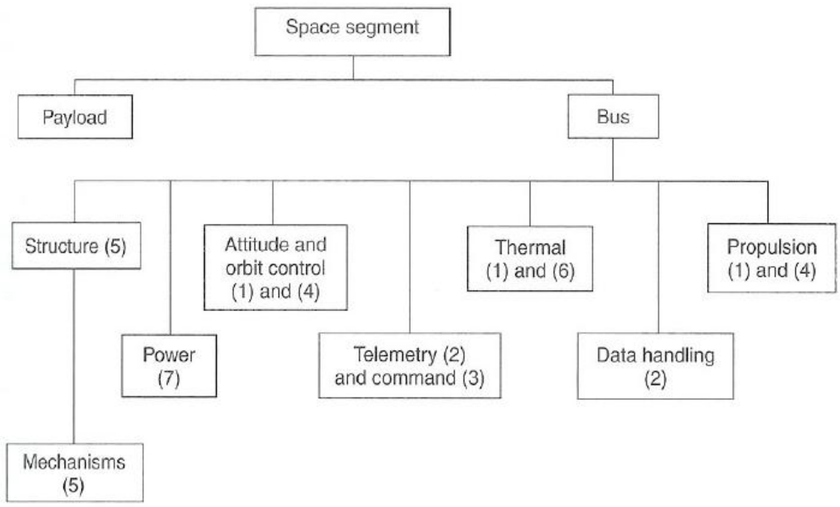
SSD

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

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

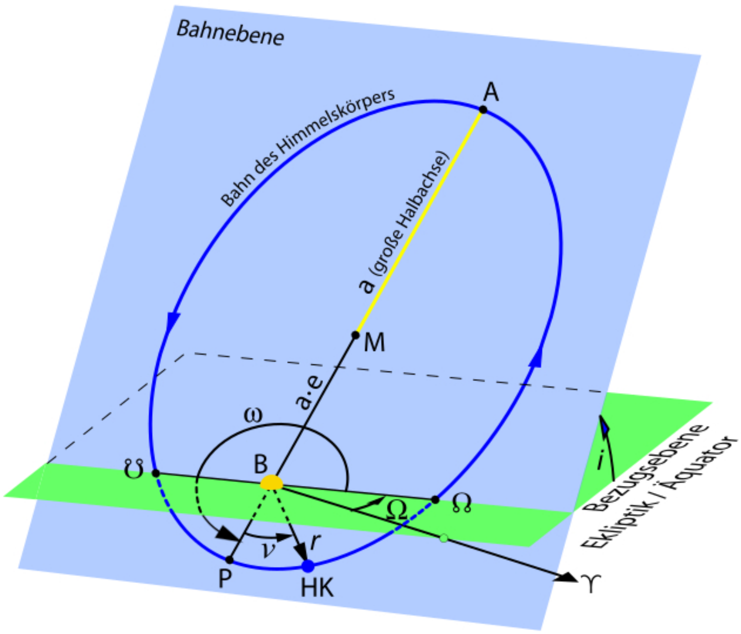
Change of the argument of perigee

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

Orbits

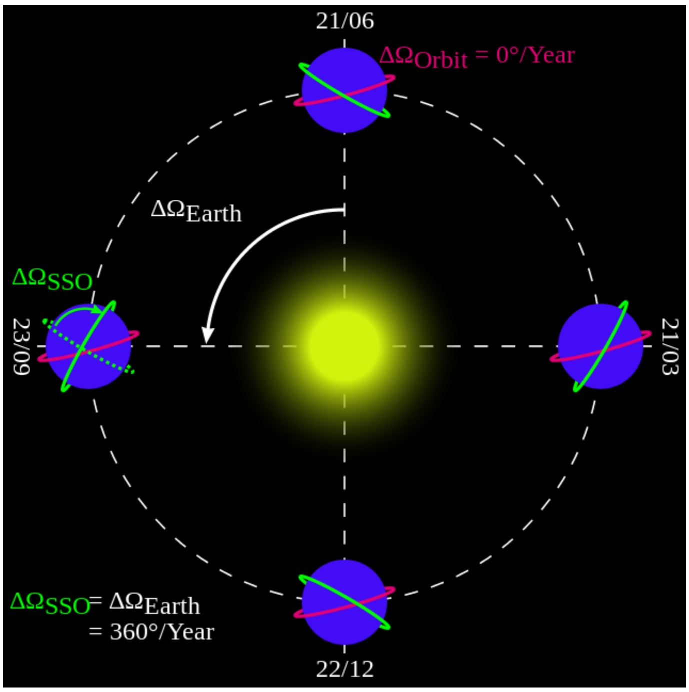
1. Highly Elliptical Orbit HEO

- 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

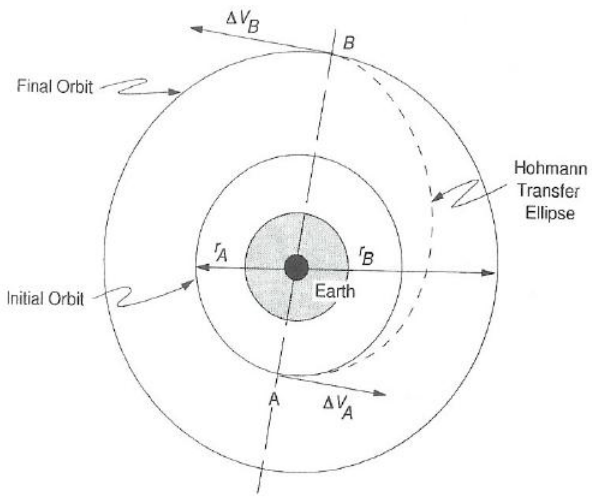


- Höhe und Inklination werden so kombiniert, dass ein Satellite aus Sicht der Sonne immer auf dem selben Orbit ist
- Höhe: 600-800 km

- 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

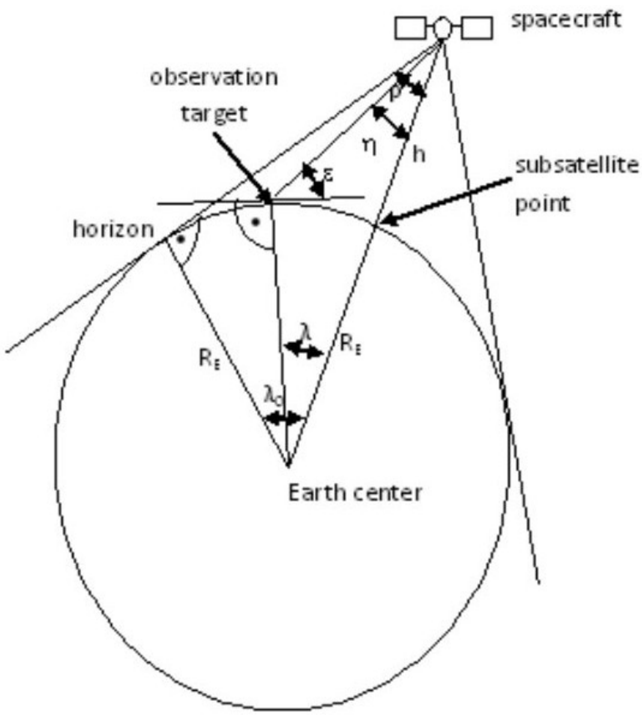
- $$\Delta\Omega = \theta \Rightarrow T\left(\frac{1}{\tau_{\mathrm{E}}} - \frac{1}{\tau_{\mathrm{S}}}\right) = \frac{m}{n}$$
- angular shift between two subsequent orbits
$$\Delta\Phi = \Delta\Phi_r - \Delta\Omega = 2\pi T\left(\frac{1}{\tau_{\mathrm{E}}} - \frac{1}{\tau_{\mathrm{S}}}\right) \text{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

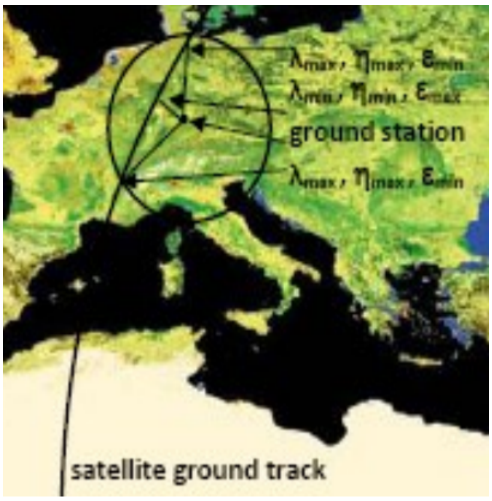
$$\tan \eta = \frac{\frac{R_E}{R_E+h} \sin \lambda}{1 - \frac{R_E}{R_E+h} \cos \lambda}$$

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

$\lambda_{\max}$: 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 $56° : 27/3/1$ 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 $\Delta v = \frac{360°}{s}$. There are two cases:

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

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

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

6 Thermal Engineering

7 Rocket Propulsion

8 TT&C

9 Power Generation

10 Power System

11 Thermal Testing

12 Spacecraft Operations