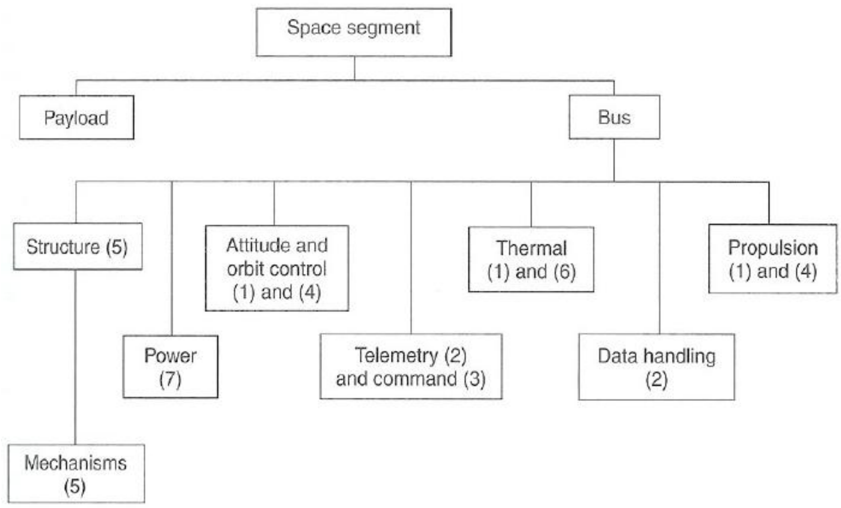
SSD

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

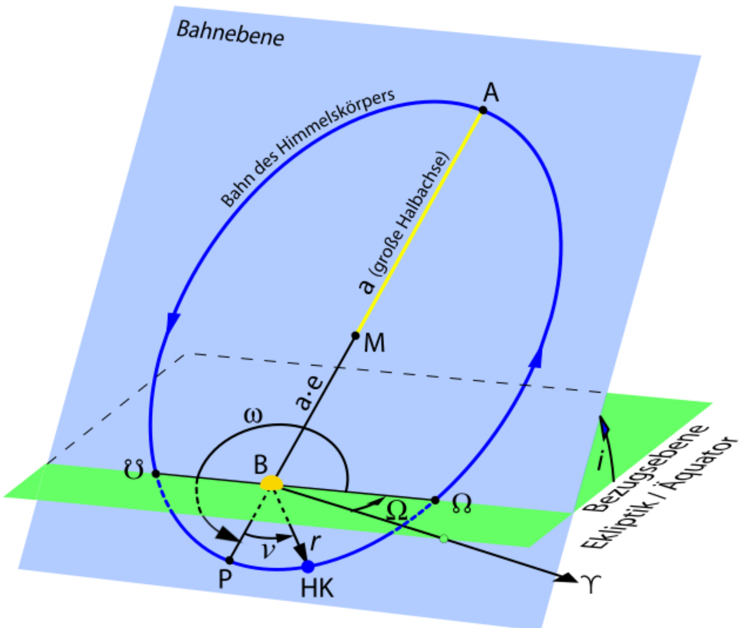
- a ... große Halbachse
- ε, e ... Exzentrizität, "Abplattung" der Ellipse (ε=0: Kreis, ε=1: Parabel, $0 < \varepsilon < 1$: Ellipse)

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:

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



Lieblingsformel

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

Change of the right ascension of the ascending node

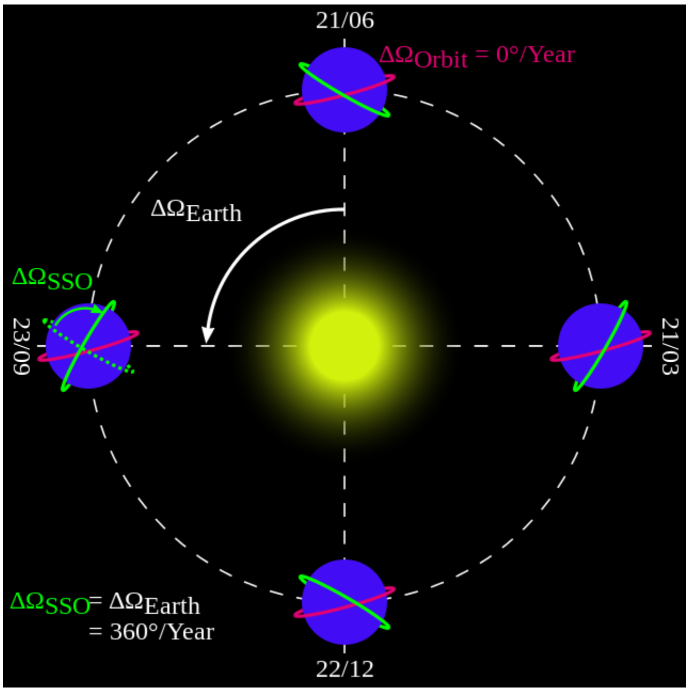
$$\Delta\Omega = -\frac{3\pi J_2 R_E^2}{a^2(1-\varepsilon^2)^2} cos(i)$$

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
    - 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

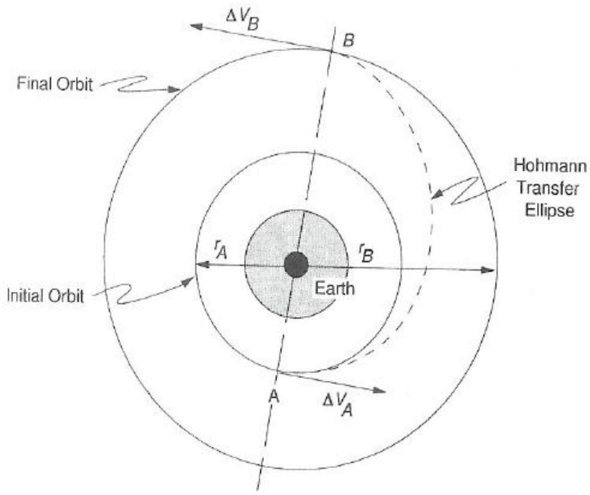
Subsatellite Point = intersection of the line between satellite and earth center with the earth's surface
Hohmann Transfer

- 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

$$\Delta\Phi_r = 2\pi\frac{T}{\tau_{\text{E}}} \text{ [rad/rev]}$$

- 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

$$\theta = 2\pi\frac{\tau_{\text{E}}}{\tau_{\text{S}}} \text{ [rad/day]} = 2\pi\frac{\tau_{\text{E}}}{\tau_{\text{S}}}\frac{T}{\tau_{\text{E}}} \text{ [rad/rev]}$$

Earth- and Sun-Synchronous Orbit

- $$\Delta\Omega = \theta \Rightarrow T\left(\frac{1}{\tau_{\text{E}}} - \frac{1}{\tau_{\text{S}}}\right) = \frac{m}{n}$$

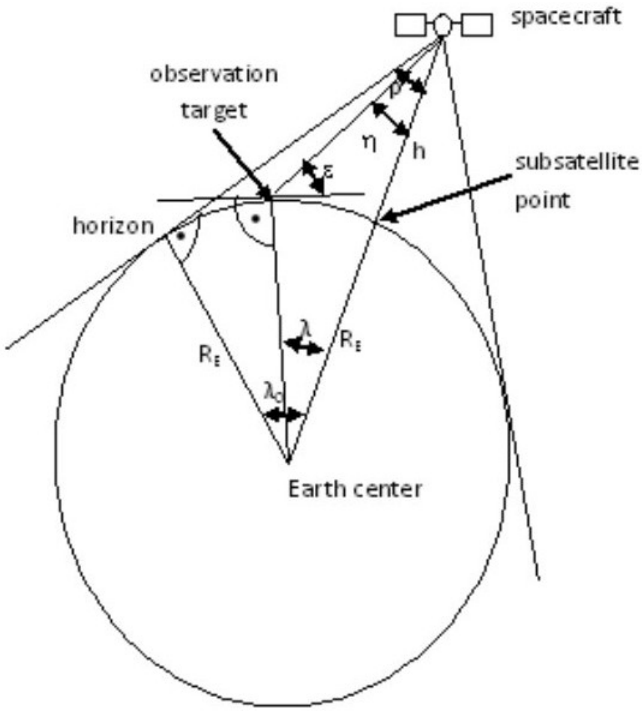
- angular shift between two subsequent orbits
$$\Delta\Phi = \Delta\Phi_r - \Delta\Omega = 2\pi T\left(\frac{1}{\tau_{\text{E}}} - \frac{1}{\tau_{\text{S}}}\right) \text{ [rad/rev]}$$
worst case between subsequent orbits $\Delta\Phi \cdot R_E$.

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: $\rho + \lambda_0 = 90°$

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

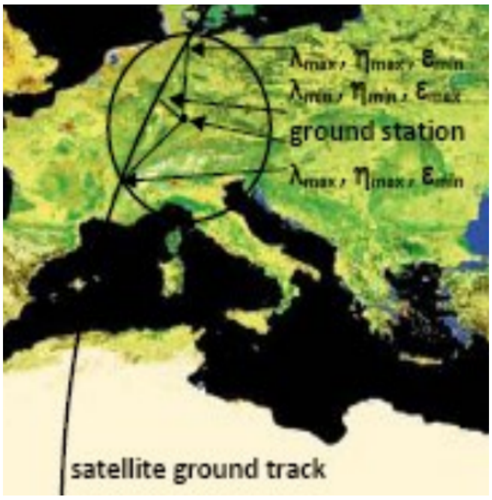
observe Λ_t, Θ_t (long,lat) from known orbit position of satellite, characterized by subsatellite point Λ_s, Θ_s.
characteristic paramters:

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



calculate nadir angle η:
$$\tan\eta = \frac{\frac{R_E}{R_E+h}\sin\lambda}{1 - \frac{R_E}{R_E+h}\cos\lambda}$$
$$\lambda + \eta + \varepsilon = 90°$$
λ_{max}: maximum earth central angle $\Rightarrow$ swath width $2\lambda_{\text{max}}$ perpendicular to groundtrack on surface.
Time in view T_{view} for circular orbit with period T:
$$T_{\text{view}} = \frac{T}{180°}\cos^{-1}\left(\frac{\cos\lambda_{\text{max}}}{\cos\lambda}\right)$$

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

4

5 Mechanics

6 Thermal Engineering

7 Rocket Propulsion

8 TT&C

9 Power Generation

10 Power System

11 Thermal Testing

12 Spacecraft Operations