

Space Engineering Period 1 Summary

1. Calculations

1.1 Heat Balance

To get heat balance, the total energy that the satellite or space ship absorbs, must be equal to the total energy that it transmits. Assuming that one side of the ship or satellite is towards the sun, and the other side, which is perfectly isolated, is towards cold space, the following equation applies:

$$\alpha AS = \epsilon Aq (T^4 - T_{sp}^4) \quad (1.1.1)$$

where α is the solar absorptance (fraction of solar radiation absorbed; metal 0.1, black paint 0.9, white paint 0.4), S is the solar flux (usually $1350 - 1400 \frac{W}{m^2}$ around the earth, and S is proportional to $\frac{1}{r^2}$, with r being the distance to the sun), A is the surface area, ϵ is the (infra-red) emissivity (fraction of black body radiation emitted; metal 0.04, black or white paint 0.9, glass 0.9), q is the Stefan-Boltzmann constant ($56.7051 * 10^{-9} \frac{W}{m^2 K^4}$), T is the satellite surface temperature and T_{sp} is the temperature of cold space.

1.2 Deriving Tsiolkowsky's formula of rocketry

Using the definition of impulse, it can be shown that:

$$M \frac{dV}{dt} = -\omega \frac{dM}{dt} \quad (1.2.1)$$

where M is the mass of the rocket, dV is the change in velocity, dt is the change in time and ω is the exhaust velocity of the fuel relative to the rocket. The equation can be rearranged as follows:

$$dV = -\omega \frac{dM}{M} \quad (1.2.2)$$

Assuming zero gravity and vacuum, we can derive Tsiolkowsky's formula of rocketry:

$$\Delta V_e = \omega \ln \frac{M_0}{M_e} \quad (1.2.3)$$

where ΔV_e is the velocity change, M_0 is the start mass of the rocket and M_e is the end mass. The usual value for ω in a launcher rocket is about $3000 \frac{m}{s}$. The usual value for $\frac{M_0}{M_e}$ is about 5-6.

If we also include the presence of a gravity field and air drag, the formula gets a bit more complicated. So to keep it a bit simple, we neglect drag, and assume just a homogeneous gravitational field. Now we get the following equation:

$$\Delta V_e = \omega \ln \frac{M_0}{M_e} - g_0 t_b \quad (1.2.4)$$

where g_0 is the gravitational constant at sea level and t_b is the burning time of the rocket.

1.3 Coasting

After the rocket engine has stopped working, the rocket is coasting. When coasting, the coasting time t_c (until the rocket has reached its highest point) is:

$$t_c = \frac{V_e}{g_0} \quad (1.3.1)$$

where V_e is the initial speed. The vertical distance h_c covered by the rocket during coast time is:

$$h_c = t_c V_e - \frac{1}{2} g_0 t_c^2 = \frac{1}{2} \frac{V_e^2}{g_0} \quad (1.3.2)$$

1.4 Aerodynamic Disturbance Forces

The aerodynamic disturbance force on a satellite in a low earth orbit, also known as the aerodynamic drag, is small, but can make a big difference after a lot of time. This drag D can be calculated as follows:

$$D = \frac{1}{2} C_D \rho V^2 A \cos i \quad (1.4.1)$$

where C_D is the drag coefficient, V is the speed (relative to air) with which the satellite is traveling, A is the surface area on one side of the satellite and i is the angle between the direction of movement, and the line perpendicular to the plain of the surface area. If you have already taken A as the frontal area of the satellite (which is usually the case), then $i = 0$, which means that $\cos i$ is simply 1.

1.5 Solar Radiation Disturbance Forces

The formula for the solar radiation disturbance force is not given in the space engineering book, and therefore needs not to be known. However, the solar radiation force is proportional to the following parameters:

- The surface area A .
- The solar flux S .
- The factor 2 - (α), where α is the solar absorbance. The two is there, because solar radiation that hits the space ship, often gets "bounced" back with equal speed in the opposite direction (remember that the momentum of the total situation stays the same, so the total impulse is 0). However, because of this, the radiation that gets absorbed, only counts once.
- $\cos i$, where i (rad/deg) is the angle between the sun vector and the surface normal. If you have taken A as the frontal area already, $\cos i$ is simply 1, just like with the aerodynamic drag formula.

2. Definitions

The following definitions need to be known.

- Space system: The whole of hard-and software, personnel, training, support, etc. necessary for conducting a space mission.
- Space flight: All activities, where a manned or unmanned vehicle is brought outside the Earth atmosphere with the purpose of performing a task in space
- Space research: Scientific research performed by means of a space vehicle (either manned or unmanned)
- Space technology: Technology required to design, develop and build space vehicles, including launchers
- Remote Sensing: The acquisition of information about an object without being in physical contact with it.
- Geostationary orbit: Circular equatorial orbit where the satellite remains always on the same Earth longitude.
- Apogee: Point in the orbit where the distance to the Earth is largest.
- Perigee: Point in the orbit where the distance to the Earth is smallest.
- Inclination: Angle of the orbit plane with the Earth equator
- Sun-synchronous orbit: Circular "polar" orbit, where the angle between the orbit plane and the sun vector is constant. Satellite passes at each location on the Earth at the same local time. A satellite in such an orbit that it sees the full celestial sphere in half a year.
- Geo-stationary Transfer Orbit (GTO): Orbit with a near-Earth perigee (e.g. 200 km altitude) and an apogee on geo-stationary altitude. Normal procedure to get a geo-stationary satellite in its orbit
- Apogee boost (for a geo-stationary satellite): Thrust manoeuvre required in GTO apogee to raise the perigee to geo-stationary altitude.
- Libration point: A point in space, where the attractive force of two celestial bodies is in equilibrium. A satellite in that point has a fixed position relative to these two bodies.
- Elevation: The angle of a satellite above the horizon. Also the angle of a launcher trajectory relative to the horizon.
- Azimuth: The angle of a launcher trajectory with the direction of the North Pole.
- Coverage: The part of the celestial body, that is visible from or can be communicated to by the payload of a satellite.
- Access: The part of the orbit a satellite is "visible" from an Earth located point.
- Launcher (rocket) stage: The assembly of structure, propellants, tanks, engines and other equipment that (in principle) can function as one single stage launcher. E.g. for a three-stage launcher: first stage, second stage, third stage etc. in the order as they are activated during the launch trajectory.
- Sub-rocket: The assembly of one or more launcher stages with a payload, as it occurs during the launch of a satellite. E.g. for a three-stage launcher:
 - First sub-rocket = first stage + second stage + third stage + payload
 - Second sub-rocket = second stage + third stage + payload
 - Third sub-rocket = third stage + payload
- Booster: A high-thrust rocket engine with associated propellant and structure ignited on the ground and assisting the main (first stage) launcher engines in the first flight phase.
- Mass ratio R: The ratio of start mass and empty mass of a sub-rocket or launcher stage. A practically achievable value of the mass ratio of a sub-rocket is 6 maximum. The value of R, together with the chosen propellant combination, determines the maximum achievable velocity increment of a sub-rocket.
- Structural efficiency s: The ratio of structural mass and propellant mass of a launcher stage or sub-rocket. A practically achievable value of s is 0.10.

- Swing by: A maneuver of a spacecraft, which uses the gravitational forces of a planet to obtain a higher speed (without hitting that planet).
- Coasting: The rocket continues its vertical flight after burn-out: it's coasting.
- Space market: The set of all actual and potential buyers of a product or service which depend or relate to having a spacecraft in orbit (Based on: Kotler et al. 1996).
- Space industry: The sellers of the above products/services.
- Service: An activity, benefit or satisfaction that is offered for sale.
- Market segment: Part of the overall market encompassing users that have more or less identical needs.

3. Spacecraft mission parts

3.1 Mission segments

A mission may be broken down in several segments. (This is the convention used in the book "Space Mission Analysis & Design" - by Larson.) Each segment has its own function, as can be seen in table 3.1.

| Mission segment | Function | Definition |
|-----------------------------|--|--|
| Subject | Observe "it"; transmit "it" | The thing that interacts with or is sensed by the payload |
| Launch segment | Provide transportation from ground to location in space | The launch facility, launch vehicle and any upper stage to place the spacecraft in orbit, as well as interfaces, payload fairing and associated ground equipment |
| Orbit & constellation | Provide access and coverage to/from object to ground | The spacecraft's trajectory or path |
| Space segment; payload | Transfer data on subject to transmittable information | The hard- and software that sense or interact with the subject |
| Space segment; bus | Support the payload with services and resources | The hard- and software that provides services and resources to the payload |
| Communications architecture | Provide telemetry, command and other datacommunication between the space segment and all users on the ground | The arrangement of components which satisfies the mission's communication, command and control (C3) requirements |
| Ground segment | Provide on-ground monitoring, command and communication services to the mission | Fixed and mobile ground stations around the globe connected by various data links |
| Mission operations | Provide the human staffing of ground and space segment | The people occupying the ground and space segment |

Table 3.1: Mission Segments

3.2 Breaking down a space craft

A spacecraft contains both a payload as a bus. The latter one contains a lot of subsystems, each with its own functions, as can be seen in table 3.2.

| Subsystem | Principal functions | Other names |
|---|---|--|
| Propulsion | Adjust orbit and attitude, dump angular momentum | Reaction Control System (RCS) |
| Guidance, Navigation & Control (GNC) | Determine and control orbit | Orbit Control System (OCS) |
| Attitude Determination & Control (ADCS) | Determine and control attitude, point and manoeuvre the spacecraft, manage angular momentum | Attitude Determination and Control System (ADCS) or Control System |
| Communications | Communicate with ground, support spacecraft tracking | Tracking, Telemetry & Command (TT&C) |
| Command & Data Handling (C&DH) | Process commands, perform data processing/formatting, provide computing power | Spacecraft Computer System or Spacecraft Processor |
| Thermal Control | Control equipment temperature | Environmental Control System |
| Power | Generate and distribute power | Electric Power Subsystem (EPS) |
| Structures & Mechanisms | Provide structural integrity, provide motion possibility to structural parts | Structure Subsystem |

Table 3.2: Breaking down a spacecraft

4. Spacecraft Market

4.1 Four main areas of space applications

- Telecommunications - Sending messages by electronic means
- Remote sensing - Information gathering from a distance by electronic means
- Navigation - Knowledge of location
- Science and technology validation - "For the benefit of knowing"

4.2 Market review

Questions to be answered when conducting a Market Review:

- Which market I am aiming for and what is its size?
- Which product I am going to offer?
 - Competitive edge
 - Price
 - Cost
 - Quality/characteristics
- Which market share can I obtain?
- What is the competition and how do I compare with it?

4.3 Market segments

Market segments are part of the overall market encompassing users that have more or less identical needs. Market segments are separated by certain segmentation principles. Segmentation principles are for example:

- Purpose (what the product/service is for)
- Geographic (world, region)
- Government or commercial
- Level of education

Two market segments are distinguished based on the type of customer:

- Commercial markets: Mission serves a paying customer base. = i highest performance per dollar.
- Institutional or governmental market: Missions are not viable commercially, require oversight or bring exceptionally high risk. Typical such missions today include mostly science and technology validation missions, launcher development and, of course, defence. = j performance driven.

4.4 Another way of segmenting the market

- Infrastructure: Including the manufacture, test, delivery and launch of satellites, other spacecraft, and related hardware.
- Telecommunications: Including transmission of international telephony services, interconnection with national telephone networks, and distribution of video signals for cable and television programmers.
- Emerging Applications: Like remote sensing, geographical information systems (GIS) or global positioning services (GPS).
- Support Services: Like publishing, business consulting, financial, legal and space insurance.

5. Spacecraft development

5.1 Steps in space system development

- Concept exploration: Results in a broad definition of the space mission and the various elements that work in unison to realize the mission. Consists of:
 - Needs analysis
 - Concept development
- Detailed design and development: Results in a detailed definition of the system components and, in larger programs, development of test hardware and/or software. Consists of:
 - Demonstration and validation
 - Engineering development
- Production and deployment: The construction of the ground and flight hardware and launch of the (constellation of) satellite(s).
- Operations and support: The day-to-day operations of the space system, its maintenance and support.
- Decommissioning: Finally its de-orbit or recovery at the end of the mission life.

5.2 What are important parameters for spacecraft?

No matter which mission, the following performance parameters are always important for a spacecraft:

- Orbit: Determines the quality and kind of observations and/or communications achievable.
- Mass: Determines the launcher and on-board propellant needed.
- Size: Determines the launcher needed.
- Electrical power: Determines the size of primary and secondary power sources (generally solar array and batteries).
- Mission duration in relation to reliability: Determines the amount of spare resources needed (propellant, over-sizing for degradation due to radiation and ageing, redundancy).
- Attitude and orbit accuracy: Determines the quality of the scientific data, the quality of communications and the safety of the spacecraft.
- On-board computer power and data storage: Determines autonomy and amount of data to be down linked.
- Up-and down-link data rate: Determines the size of antennae, receiver and transmitter power (both on board and on ground).
- Specific parameters, that are a function of the payload: For example: cooling, electrical properties of the spacecraft, shading from unwanted radiation, ground contact constraints, etc.

5.3 Why a space platform?

- Remote sensing of space (planets, stars, comets, environment, etc.) or space science: Comparable to Earth-based space observation, except that the sight is not obscured by the Earth's atmosphere and viewing distances can be reduced substantially.
- Remote sensing of the Earth or Earth Observation (EO): Comparable to observation by aircraft, balloons, etc, except that space provides global and synoptic detailed information.
- In-situ observations: Cannot be done elsewhere.

6. Spacecraft control

- Solar radiation: Can be used as propelling force.
- Aerodynamics: Can be used to capture a satellite around a planet with an atmosphere. Is used to dispose of satellites in a low (Earth) orbit. Shall be compensated to keep a satellite in its low orbit.
- Gravity: Is used to perform efficient (interplanetary) flight (celestial body swing-by); in case of Cassini, this saves 68 tons of propellants. Shall be compensated to keep satellites in their intended orbit (geostationary satellites). Is used to achieve specific orbits (sun-synchronous orbits, orbits in libration points, etc.).

6.1 Disturbance torques

- External disturbance torques
 - Aerodynamic (Can be used for passive stabilisation)
 - Solar radiation (Can be used for passive stabilisation)
 - Magnetic field (Can be used for active and passive stabilisation)
 - Gravity gradient (Can be used for passive stabilisation)
- Internal disturbance torques
 - Thrust misalignment (generally: actuator misalignment)
 - On-board mechanisms

6.2 Means to control the attitude of a spacecraft

- Passive
 - Gravity gradient, magnetic, aerodynamic (rarely), solar radiation
 - Gyroscopic (spinning spacecraft)
- Active
 - Thrusters
 - * Cost propellants, which are depleted
 - * Can remove disturbance torque
 - Reaction or momentum wheels
 - * Cost electrical energy
 - * Cannot remove disturbance torque, can only store it
 - Magnetic coils or torque rods
 - * Cost electrical energy
 - * Work only in a relatively large magnetic field
 - * Can remove disturbance torque

6.3 Disturbance torque and spacecraft sizing

- The maximum disturbance torque sizes the maximum actuator (control) torque. But spacecraft manoeuvring may require more torque.
- The time integrated cyclic disturbance torque sizes the angular momentum storage capacity required. A reaction or momentum wheel can store momentum, a thruster system cannot (it only eliminates angular momentum).
- The time integrated non-cyclic disturbance torque sizes the angular momentum dump capacity required. To dump angular momentum you need an actuator than can generate a torque through interaction with the environment or can generate a torque by mass expulsion.

Space Engineering Period 2 Summary

1. Two-Dimensional Orbital Mechanics

1.1 Kepler's Laws

Kepler came up with three laws (which are in *italic*). These laws were equivalent with three rules which we now often use. **Kepler's Laws** are:

1. *Planets move in ellipses with the Sun at one focus.* (Planets/satellites/comets all move along conical sections.)
2. *Planets sweep out equal areas in equal times.* (Law of conservation of angular momentum.)
3. *The square of the orbital period is proportional to the cube of the length of the (semi-)major axis of the ellipse.* ($T^2 \sim a^3$)

1.2 Conservation of Angular Momentum (Second Law)

Define the **gravitational parameter** $\mu = MG$, where M is the mass of the central body (assumed to be much greater than the masses of the objects orbiting it) and G is a universal constant. The gravitational acceleration now is:

$$\ddot{\mathbf{r}} = -\frac{\mu}{r^2} \frac{\mathbf{r}}{r} \quad (1.2.1)$$

From this can be derived that **angular momentum** is conserved (thus there is **conservation of angular momentum**). In formula:

$$\mathbf{r} \times \dot{\mathbf{r}} = \mathbf{r} \times \mathbf{V} = \text{constant} = \mathbf{H} \quad (1.2.2)$$

This implies that the motion is in one plane, and $H = r^2 \dot{\phi} = \text{constant}$.

1.3 Conservation of Energy

Define the **energy** E as:

$$E = \frac{V^2}{2} - \frac{\mu}{r} \quad (1.3.1)$$

It can be shown that the energy stays constant. Thus there is **conservation of energy**.

1.4 Conical Sections (First Law)

It can also be shown that $\ddot{r} - r\dot{\phi}^2 = -\frac{\mu}{r^2}$. Combining this with conservation of angular momentum ($r^2\dot{\phi} = H$), and solving the differential equation, gives the **equation for conical sections**:

$$r = \frac{p}{1 + e \cos \theta} \quad (1.4.1)$$

where r is the distance from the origin, θ is the **true anomaly** (the angle with the **pericenter**, being the point of the orbit closest to the origin), p is the **semi-latus rectum** ($p = H^2/\mu$) and e is the **eccentricity**. The eccentricity determines the shape:

- $e = 0 \Rightarrow$ The orbit has the shape of a circle.

- $0 < e < 1 \Rightarrow$ The orbit has the shape of an ellipse.
- $e = 1 \Rightarrow$ The orbit has the shape of a parabola.
- $e > 1 \Rightarrow$ The orbit has the shape of an hyperbola.

1.5 Useful Equations for Elliptical Orbits

Most orbits are **elliptical orbits**. Suppose a is the **semi-major axis** (half of the longest diagonal) and b is the **semi-minor axis** (half of the shortest diagonal) of the ellipse. Then the following equation holds:

$$p = a(1 - e^2) \quad \Rightarrow \quad r = \frac{a(1 - e^2)}{1 + e \cos \theta} \quad (1.5.1)$$

Now if r_p is the minimum distance from the origin (**pericenter**) and r_a is the maximum distance from the origin (**apocenter**), then also:

$$r_p = a(1 - e) \quad r_a = a(1 + e) \quad (1.5.2)$$

$$a = \frac{r_a + r_p}{2} \quad e = \frac{r_a - r_p}{r_a + r_p} \quad (1.5.3)$$

1.6 Velocities

From conservation of angular momentum, conservation of energy, and elliptical properties, it follows that the velocities in the pericenter and apocenter can be calculated with:

$$V_p^2 = \frac{\mu}{a} \left(\frac{1+e}{1-e} \right) \quad V_a^2 = \frac{\mu}{a} \left(\frac{1-e}{1+e} \right) \quad (1.6.1)$$

With some help of these equations, the **energy equation for elliptical orbits** can be derived:

$$V^2 = \mu \left(\frac{2}{r} - \frac{1}{a} \right) \quad (1.6.2)$$

(Note that inserting $r_a = a(1+e)$ and $r_p = a(1-e)$ into equation 1.6.2 transforms it back to the equations in 1.6.1.) For a circle it is clear that $r = a$. So inserting this in equation 1.6.2 gives us the **local circular velocity**:

$$V_c = \sqrt{\frac{\mu}{r}} \quad (1.6.3)$$

1.7 Orbital Period (Third Law)

From Kepler's second law, and by using the equation $H = \sqrt{\mu p}$, the **orbital period** T can be derived, which indicates Kepler's third law was correct:

$$T = 2\pi \sqrt{\frac{a^3}{\mu}} \quad (1.7.1)$$

1.8 The Eccentric Anomaly

The **eccentric anomaly** E is an alternative for the true anomaly θ and often shows up when travel times in elliptical orbits need to be calculated. First of all, it can be derived that:

$$r = a(1 - e \cos E) \quad (1.8.1)$$

From this the relationship between θ and E can be derived:

$$\tan \frac{\theta}{2} = \sqrt{\frac{1+e}{1-e}} \tan \frac{E}{2} \quad (1.8.2)$$

where $\theta/2$ and $E/2$ have to be in the same quadrant. After some derivation, **Kepler's Equation** can be found:

$$E - e \sin E = \sqrt{\frac{\mu}{a^3}}(t - t_p) = n(t - t_p) = M \quad (1.8.3)$$

where t_p is the time of the last passage of the pericentre, n is the mean angular velocity ($n = \sqrt{\mu/a^3}$), M is the mean anomaly, E is the eccentric anomaly and θ is the true anomaly. Using the past equations, one can use the following steps to find the time needed, given a certain position: $\theta \rightarrow E \rightarrow M \rightarrow t - t_p$. The other way around is a bit more difficult. Given a certain time span, one can find the change in mean anomaly M . From this value M , one can approximate E recursively, using $E_{n+1} = M + e \sin E_n$ (where $E_0 = 0 \Leftrightarrow E_1 = M$).

1.9 Parabolic and Hyperbolic Orbits

Parabolic orbits are the orbits at which $V \rightarrow 0$ as $r \rightarrow \infty$. Using this data and the energy equation, one finds the **local escape velocity**:

$$V_o = \sqrt{\frac{2\mu}{r}} = \sqrt{2}V_c \quad (1.9.1)$$

where V_c still is the local circular velocity. While objects in a parabolic orbit lose all their speed as $r \rightarrow \infty$, objects in **hyperbolic orbits** still have a velocity when $r \rightarrow \infty$. Using the energy equation once more, one finds for the actual local satellite velocity:

$$V^2 = V_o^2 + V_\infty^2 \quad (1.9.2)$$

where V_∞ is the velocity the satellite would have as $r \rightarrow \infty$.

2. Three-Dimensional Orbits

2.1 Orbit Descriptions

The projection of the satellite on the Earth's surface is called the **sub-satellite point**. The trace of the sub-satellite point is called the **ground track**. The ground track is a way to visualize the shape of the orbit. But to describe an orbit, we want to know the satellite's position and velocity at a given point, which can not be derived from a ground track.

2.2 Cartesian Elements

There are multiple ways to fully describe an orbit. One such description (which is not often used) is in **Cartesian Elements**: $X, Y, Z, \dot{X}, \dot{Y}, \dot{Z}$. This Cartesian coordinate system has as origin the center of the earth. The X -axis points in the direction of the **vernal equinox**, which is the position of the sun as it cross the equator around March 21st. The Z -axis points to the North Pole and the Y -axis is perpendicular to the past two (its direction can be found with the right-hand-rule). These 6 orbital elements fully determine the orbit, orbital plane and satellite position. Note that X, Y and Z describe the position, while \dot{X} and \dot{Y} and \dot{Z} describe the velocity.

2.3 Keplerian Elements

Another description is in **Keplerian Elements** (see figure 2.1): $a, e, i, \Omega, \omega, \theta$. a is the **semi-major axis** of the ellipse the satellite is traveling on. e is the **eccentricity** of the ellipse. i is the **inclination** of the orbital plane, being the angle between the orbital plane and the equatorial plane. Ω is the **right ascension of the ascending node**, being the angle between the X -axis (pointing to the vernal equinox) and the ascending node. (The **ascending node** is the point at which the sub-satellite point of the satellite crosses the equator in a northward direction.) ω is the **argument of perigee**, being the angle between the perigee and the ascending node. And finally, θ is the true anomaly, which was discussed in the last chapter. Note here that i and Ω describe the orbital plane, a and e describe the shape of the ellipse, ω describes the orientation of the ellipse and θ describes the position of the satellite on the ellipse.

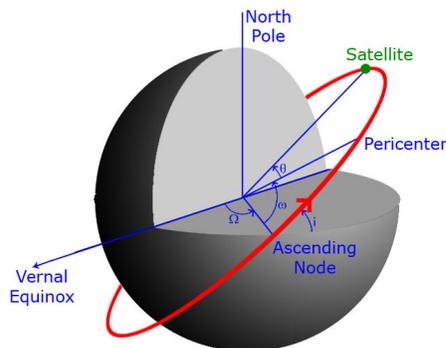


Figure 2.1: Indication of the Keplerian Elements.

2.4 Orbit Perturbations

Orbits are in reality not exact ellipses in a 2D plane. The real (3D) orbits are influenced by a lot aspects. For example, irregularities on earth (non-spherical shape, non-homogeneous division of mass), gravity from moon/sun, atmospheric drag, solar radiation pressure, etcetera.

However, these **orbit perturbations** do not have to be bad, as they can be used. Due to the non-spherical shape of the earth, there is extra mass around the equator, effecting the satellite orbit. This causes **precession** (rotation of the orbital plane). If the angle of inclination is $\phi = 97^\circ$, the precession is about 1° per day, which is sufficient to keep the sun always on the same side of the satellite. More about this will be discussed in chapter 5.10.

3. Rocket Motion

3.1 Tsoilkowski's Formula

Suppose we have a rocket with mass M flying at velocity V in some direction, expelling fuel with mass dM with a velocity w (with respect to the rocket) in the opposite direction. The change in momentum now is:

$$dI = I_{final} - I_{start} = ((M + dM)(V + dV) - (V - w)dM) - (M V) = M dV + w dM \quad (3.1.1)$$

where (in the case of rockets, where fuel is exhausted, and thus the mass decreases) dM will be negative. If we assume no external forces are present (thus $dI = 0$) and if we assume w is constant, then integrating the last equation with respect to time results in **Tsoilkowski's formula**:

$$\Delta V = w \ln \frac{M_0}{M_e} = I_{spec} g_0 \ln \Lambda \quad (3.1.2)$$

where M_0 is the **initial mass** of the rocket, M_e is the **final mass**, $\Lambda = M_0/M_e$ is the **mass ratio**, g_0 is the gravitational acceleration at ground level and I_{spec} is the **specific impulse** (per definition equal to $I_{spec} = w/g_0 = T/mg$).

3.2 Thrust and Burn Time

The thrust T of a rocket can be calculated as follows:

$$T = m \frac{dV}{dt} = -w \frac{dM}{dt} = w m \quad (3.2.1)$$

where $m = -dM/dt$ is the mass flow. Assuming the mass flow is constant, the **burn time** t_b can be calculated as follows:

$$t_b = \frac{M_0 - M_e}{m} = \frac{w}{T}(M_0 - M_e) = \frac{w}{g_0} \frac{M_0 g_0}{T} \left(1 - \frac{M_e}{M_0}\right) = \frac{I_{spec}}{\Psi_0} \left(1 - \frac{1}{\Lambda}\right) \quad (3.2.2)$$

where Ψ_0 is the **thrust-to-weight-ratio** (defined as the thrust per unit of initial weight $\Psi_0 = \frac{T}{M_0 g_0}$).

3.3 Traveled Distance

To find the distance that a rocket has traveled when the rocket has burned out, we once more use $m = -dM/dt$ and find the following:

$$s_e = \int_0^{t_b} V dt = \int_0^{t_b} w \ln \frac{M_0}{M} dt = - \int_0^{t_b} w (\ln M - \ln M_0) dt = \int_{M_0}^{M_e} \frac{w}{m} (\ln M - \ln M_0) dM \quad (3.3.1)$$

Let's also assume that the mass flow is constant. Integrating and working out the result gives the following equation:

$$s_e = w t_b \left(1 - \frac{\ln \Lambda}{\Lambda - 1}\right) + V_0 t_b \quad (3.3.2)$$

3.4 Gravity and Drag

When gravity and drag are included in the calculations, things get more difficult. Using $T = m w = -w \frac{dM}{dt}$, it follows that:

$$M \frac{dV}{dt} = T - Mg - D \quad \Rightarrow \quad \Delta V = w \ln \Lambda - \int_0^{t_b} g dt - \int_0^{t_b} \frac{D}{M} dt \quad (3.4.1)$$

This is a complicated integral. When it is used, it is often simplified in several ways. First of all, when a rocket is launched from earth, the burn times are often short enough to assume the gravity stays constant (thus $g = g_0$). We also still assume that w is constant. And finally drag is often neglected. This gives the following two equations:

$$\Delta V = w \ln \Lambda - g t_b \quad (3.4.2)$$

$$s_e = w t_b \left(1 - \frac{\ln \Lambda}{\Lambda - 1} \right) - \frac{1}{2} g t_b^2 + V_0 t_b \quad (3.4.3)$$

Note that usually $V_0 = 0$ and thus $\Delta V = V_e$.

3.5 Rocket Restrictions

When a rocket is lifting off from earth, there are several restrictions to its parameters. First of all, there are restrictions on the thrust-to-weight-ratio and to the burn time, if the rocket ever wants to come loose from the ground:

$$T > M_0 g_0 \quad \Rightarrow \quad \Psi_0 > 1 \quad \Rightarrow \quad t_b < I_{spec} \left(1 - \frac{1}{\Lambda} \right) \quad (3.5.1)$$

There is also a restriction for the maximum thrust-to-weight-ratio and the maximum burn time. It can be shown that the acceleration at burn-out (at which it is at a maximum, for constant thrust, since the mass is at a minimum) is $a_e = g_0(\Psi_0 \Lambda - 1)$. Thus the following restrictions are in order:

$$\Psi_0 < \frac{1}{\Lambda} \left(\frac{(a_e)_{max}}{g_0} + 1 \right) \quad \Rightarrow \quad t_b > \frac{I_{spec}(\Lambda - 1)}{\frac{(a_e)_{max}}{g_0} + 1} \quad (3.5.2)$$

where the second part was derived using equation 3.2.2.

3.6 Coasting

After burn-out, the rocket pursues its vertical flight. During this flight, its velocity is $V = V_e - gt$. This vertical flight continues for a time of $t_c = V_e/g_0$ (where g is approximated as being constant) until it has reached its highest point. The distance traveled after burn-out of the rocket, can be calculated as follows:

$$s_c = \int_0^{t_c} V dt = V_e t_c - \frac{1}{2} g_0 t_c^2 = \frac{1}{2} \frac{V_e^2}{g_0} \quad (3.6.1)$$

Now the time taken to reach the highest point (the **culmination point**) and the maximum height reached can be calculated, simply by adding things up:

$$t_{tot} = t_b + t_c \quad h_{tot} = h_b + h_c \quad (3.6.2)$$

3.7 Multi-Stage Rockets

Normal rockets usually do not reach the high velocity necessary for low earth orbit. That's why multi-stage rockets are used. As the name indicates, a multi-stage rocket consists of multiple stages, each with its own mass. Usually a division is made as follows:

$$M_0 = M_p + M_e = M_p + M_c + M_u \quad (3.7.1)$$

where M_0 is the total mass, M_p is the mass of the fuel of the current stage, M_e is the empty rocket, M_c is the structural weight of that stage and M_u is the payload of that stage. So the rocket burns up its fuel with mass M_p to give the rocket with mass M_e a certain velocity, which can be calculated using Tsoilkowski's equation. Then the structure with mass M_c is ejected, and the new stage, consisting of M_u is activated. This stage can now be seen as a new rocket, so if we call $M_{0_2} = M_{u_1}$, we can do all the calculations again for the new stage and calculate the velocity of the rocket at the end of the second stage. The primary advantage lies in the fact that structural weight is dumped as soon as it's unnecessary, and thus does not need to accelerate further.

4. Satellite Tracking Systems

4.1 Measuring Ways

There are multiple tracking systems, and they sometimes measure different aspects of the satellite trajectory. Some tracking systems measure the **range**, which is the distance between the ground station and the satellite. Others measure the **range-rate**, which is the time derivative of the distance between the ground station and the satellite.

4.2 Visibility

When a satellite is passing over a ground station, it can be seen. But when it's on the other side of the earth it can not. When an observer can see a satellite (indicated by the visibility circle radius s), depends on the satellite altitude h and the cut-off elevation E . Figure 4.1 shows more details. From this picture can be derived that:

$$s = R\alpha = \frac{h}{\tan(\alpha + E)} \quad (4.2.1)$$

The cut-off elevation depends on buildings and trees around the observatory, but it can minimally be $E = 0^\circ$. Then the line PS in figure 4.1 will be tangent to the earth's surface.

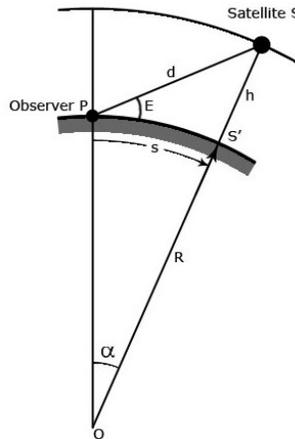


Figure 4.1: Definition of the variables concerning visibility circles.

4.3 Contact Time

If there's a ground station observing a satellite in a circular orbit, how long would it be able to see it if the satellite passes directly over it? This is an important aspect, since it's an indication of how long data can be transmitted/recieved. This time depends on the height h , the orbit time T and the cut-off angle E . For simplicity we will assume that the ground station's view doesn't get hindered, so $E = 0$. The angle α can now be calculated using:

$$\cos \alpha = \frac{R}{R + h} \Rightarrow \alpha = \arccos \frac{R}{R + h} \quad (4.3.1)$$

The part of the orbit which the observer can see spans an angle of 2α . So the satellite is visible for a part $(2\alpha)/(2\pi) = \alpha/\pi$. The total contact time now is:

$$T_{contact} = \frac{T}{\pi} \arccos\left(\frac{R}{R+h}\right) \quad (4.3.2)$$

4.4 Number of Ground Stations

To calculate how many ground stations are necessary to continuously keep track of a satellite, use is made of a simplification. Simply divide the area of the earth by the area of one visibility circle:

$$\text{Number of stations} = \frac{4\pi R^2}{\pi s^2} = \left(\frac{2R}{s}\right)^2 \quad (4.4.1)$$

5. Traveling to Planets

5.1 Minimum Energy Travel

There are many ways to travel to other planets. However, the method that uses minimum energy is the so-called **Hohmann transfer orbit**. This orbit is half of an ellipse, with one planet at its apocenter and the other at its pericenter. Various aspects of this Hohmann orbit can be calculated, but they are based on a few assumptions. It is assumed that the orbit trajectories of planets are circles, and it is also assumed that the orbital planes of all planets are in the **ecliptic** (the orbital plane of the earth). When calculating these aspects, use is made of the **patched conic approach**. The entire orbit is split up into pieces, which all have the shape of a conic section, and those pieces are then 'patched' together.

5.2 Example Travel to Mars

Suppose we want to travel to Mars. We start in a parking orbit around earth (at height $h_e = 300 \text{ km}$, and want to wind up in a parking orbit around Mars (at height $h_m = 200 \text{ km}$). What would our velocity be at the important parts of our journey? We know the trajectory for the satellite in the sun-centered part already, so it's easiest to start calculating there.

5.3 Example Travel - Sun-Centered Part

The circular velocity of the earth with respect to the sun is:

$$V_{c_e} = \sqrt{\frac{\mu_s}{r_e}} = \sqrt{\frac{132.7 \cdot 10^9}{149.6 \cdot 10^6}} = 29.784 \text{ km/s} \quad (5.3.1)$$

where r_e is the distance between Earth and the sun. The circular velocity of Mars around the sun is now easily calculated:

$$V_{c_m} = V_{c_e} \sqrt{\frac{r_e}{r_m}} = 29.784 \sqrt{\frac{1 \text{ AU}}{1.524 \text{ AU}}} = 24.126 \quad (5.3.2)$$

where r_m is the distance between Mars and the sun. Also, since the satellite will be traveling in an ellipse from Earth to Mars, we can calculate the eccentricity of the ellipse:

$$e = \frac{r_m - r_e}{r_m + r_e} = \frac{1.524 \text{ AU} - 1 \text{ AU}}{1.524 \text{ AU} + 1 \text{ AU}} = 0.2076 \quad (5.3.3)$$

Since we'll travel to Mars (which is further away from the sun than the earth), the earth is the perihelion (pericenter, but then for the sun) of the elliptical orbit and Mars is the apohelion (note that this is opposite if we traveled to Venus and Mercury!). Using these three numbers, the velocities at perihelion and the apohelion (so the velocities at earth and at Mars) can be found:

$$V_p = V_{c_e} \sqrt{1 + e} = 29.784 \sqrt{1 + 0.2076} = 32.729 \text{ km/s} \quad (5.3.4)$$

$$V_a = V_{c_m} \sqrt{1 - e} = 24.126 \sqrt{1 - 0.2076} = 21.477 \text{ km/s} \quad (5.3.5)$$

These velocities come in very handy during the upcoming calculations. So now we will lift-off at earth, and see which velocities we will need.

5.4 Example Travel - Earth-Centered Part

We're in a parking orbit around earth at height $h_e = 300 \text{ km}$. So our velocity is the circular velocity around earth at $r = R_e + h_e$ (where $R_e = 6378 \text{ km}$ is the radius of the earth), which is:

$$V_{e_c} = \sqrt{\frac{\mu_e}{r}} = \sqrt{\frac{\mu_e}{R_e + h_e}} = \sqrt{\frac{398.6 \cdot 10^3}{6378 + 300}} = 7.726 \text{ km/s} \quad (5.4.1)$$

But we want to get away from the earth, so we need to escape. For this, the escape velocity of $\sqrt{2}V_{e_c}$ isn't sufficient. We want to escape, but also want to have sufficient velocity to travel to Mars. We just found out that we needed a velocity of $V_p = 32.729 \text{ km/s}$ to travel to Mars. But this velocity is with respect to the sun. We want to know the velocity with respect to the earth. So we simply subtract the velocity of the earth with respect to the sun. So the final velocity V_{n_e} (after escape), with respect to the earth, would be:

$$V_{\infty_e} = V_p - V_{e_c} = 32.729 - 29.784 = 2.945 \text{ km/s} \quad (5.4.2)$$

(Note that if we wanted to travel to Venus, the velocity at the earth would be the velocity at the apohelion, so we would have to use $V_{\infty_e} = V_{c_e} - V_a$.) To reach this velocity, we get a velocity increase. Suppose we have a velocity of V_{n_e} after the increase. This velocity can be calculated (using the energy equation):

$$\frac{V_{n_e}^2}{2} - \frac{\mu_e}{R_e + h_e} = \frac{V_{\infty_e}^2}{2} - \frac{\mu_e}{\infty} = \frac{V_{\infty_e}^2}{2} \quad (5.4.3)$$

$$V_{n_e} = \sqrt{V_{\infty_e}^2 + 2\frac{\mu_e}{R_e + h_e}} = \sqrt{V_{\infty_e}^2 + 2V_{e_c}^2} = \sqrt{2.945^2 + 2 \cdot 7.726^2} = 11.316 \text{ km/s} \quad (5.4.4)$$

So the necessary velocity increase at earth is:

$$\Delta V_e = V_{n_e} - V_{e_c} = 11.316 - 7.726 = 3.590 \text{ km/s} \quad (5.4.5)$$

5.5 Example Travel - Mars-Centered Part

And finally we arrive at Mars with a velocity $V_a = 21.477 \text{ km/s}$ with respect to the sun. This velocity with respect to Mars is:

$$V_{\infty_m} = V_{c_m} - V_a = 24.126 - 21.477 = 2.649 \text{ km/s} \quad (5.5.1)$$

We want to get in a parking orbit at $h_m = 200 \text{ km}$. So first we need to descend, but while doing that, our velocity increases. And if we do not do anything, then we will just leave Mars again. So we need a velocity decrease. To be able to stay in orbit at $h_m = 200 \text{ km}$, we need the following circular velocity:

$$V_{m_c} = \sqrt{\frac{\mu_m}{r}} = \sqrt{\frac{\mu_m}{R_m + h_m}} = \sqrt{\frac{43.01 \cdot 10^3}{3397 + 200}} = 3.458 \text{ km/s} \quad (5.5.2)$$

Suppose we have a velocity of V_{n_m} when we are $h_m = 200 \text{ km}$, before the velocity decrease. Then V_{n_m} can be calculated:

$$\frac{V_{n_m}^2}{2} - \frac{\mu_m}{R_m + h_m} = \frac{V_{\infty_m}^2}{2} - \frac{\mu_m}{\infty} = \frac{V_{\infty_m}^2}{2} \quad (5.5.3)$$

$$V_{n_m} = \sqrt{V_{\infty_m}^2 + 2\frac{\mu_m}{R_m + h_m}} = \sqrt{V_{\infty_m}^2 + 2V_{m_c}^2} = \sqrt{2.649^2 + 2 \cdot 3.458^2} = 5.562 \text{ km/s} \quad (5.5.4)$$

So the necessary velocity decrease will be:

$$\Delta V_m = V_{n_m} - V_{m_c} = 5.562 - 3.458 = 2.104 \text{ km/s} \quad (5.5.5)$$

5.6 Example Travel - Summary

We were in orbit around the earth at $h_e = 300 \text{ km}$ with a velocity of $V_{e_e} = 7.726 \text{ km/s}$. Then we got a velocity increase of $\Delta V_e = 3.590 \text{ km/s}$, and thus got a velocity of $V_{n_e} = 11.316 \text{ km/s}$. We ascended, and when we finally left earth, we only had a velocity left of $V_{\infty_e} = 2.945 \text{ km/s}$ with respect to the earth, which was a velocity of $V_p = 32.729 \text{ km/s}$ with respect to the sun. We traveled all the way to Mars, and when we finally got there, our remaining velocity was $V_a = 21.477 \text{ km/s}$ with respect to the sun, which was $V_{\infty_m} = 2.649 \text{ km/s}$ with respect to Mars. We descended, and picked up velocity, until at $h_m = 200 \text{ km}$ we had a velocity of $V_{n_m} = 5.562 \text{ km/s}$. Then we got a velocity decrease of $\Delta V_m = 2.104 \text{ km/s}$, such that we remained in orbit around Mars with a velocity of $V_{m_c} = 3.458 \text{ km/s}$.

The total velocity change we have had during our trip was $\Delta V_t = \Delta V_e + \Delta V_m = 3.590 + 2.104 = 5.694 \text{ km/s}$, which is about the minimum possible for a travel from Earth to Mars.

5.7 Travel Time

When traveling from one planet to another, a certain amount of time is taken. This time depends on the orbit type. But for a Hohmann transfer orbit, the travel time can be rather easily calculated. If we travel from planet 1 to planet 2, then suppose those two planets have a distance r_1 and r_2 , respectively, to the star they orbit. The ellipse with its pericenter at one of the planets and its apocenter at the other has a semi-major axis of:

$$a = \frac{r_1 + r_2}{2} \quad (5.7.1)$$

We only travel half an ellipse, so the travel time is only half of the orbit time, which is:

$$T_{travel} = \pi \sqrt{\frac{a^3}{\mu}} = \pi \sqrt{\frac{(r_1 + r_2)^3}{8\mu}} \quad (5.7.2)$$

And since the orbit times of the two planets, T_1 and T_2 respectively (which are for the whole orbits), can also be calculated using $T = 2\pi\sqrt{r^3/\mu}$, it can be determined that:

$$T_{travel} = \frac{1}{2} \sqrt{\frac{a^3}{r_1^3}} T_1 = \frac{1}{2} \sqrt{\frac{a^3}{r_2^3}} T_2 = \frac{\sqrt{2}}{8} \sqrt{(T_1^{2/3} + T_2^{2/3})^3} \quad (5.7.3)$$

5.8 Synodic Period

Interplanetary missions can not be started at any time. It takes a time until the planet configuration is right. If you miss an opportunity to launch, you have to wait until the opportunity occurs again. This time between two consecutive opportunities (thus the time after which the planets have the same configuration again) is called the **synodic period**. With the same configuration is meant that the angle between the two planets and the sun is equal.

Let's define an arbitrary axis somewhere, and let's suppose we have two planets, planet 1 and planet 2, just like in figure 5.1. The two planets have angles ϕ_1 and ϕ_2 , respectively, with the axis. Their relative angle is defined as $\phi_{rel} = \phi_2 - \phi_1$. Suppose the relative angle ϕ_{nes} is necessary to start a mission. Also suppose the last time the relative angle was ϕ_{nes} was on $t = t_0$. We have to wait until ϕ_{rel} once more becomes ϕ_{nes} .

To calculate the time taken, we first express ϕ_1 and ϕ_2 as a function of the time t :

$$\phi_1(t) = \phi_{1_0} + \omega_1(t - t_0) = \phi_{1_0} + \frac{2\pi}{T_1}(t - t_0) \quad \text{and} \quad \phi_2(t) = \phi_{2_0} + \omega_2(t - t_0) = \phi_{2_0} + \frac{2\pi}{T_2}(t - t_0) \quad (5.8.1)$$

And now we can express ϕ_{rel} as a function of t :

$$\phi_{rel}(t) = \phi_2(t) - \phi_1(t) = (\phi_{2_0} - \phi_{1_0}) + \left(\frac{2\pi}{T_2} - \frac{2\pi}{T_1} \right) (t - t_0) = \phi_{rel_0} + 2\pi \left(\frac{T_1 - T_2}{T_1 T_2} \right) (t - t_0) \quad (5.8.2)$$

We assumed that at $t = t_0$ also $\phi_{rel} = \phi_{nes}$ so $\phi_{rel_0} = \phi_{nes}$. To be able to launch again, we know that $\phi_{rel}(t)$ should be ϕ_{nes} plus or minus $k \cdot 2\pi$ (since if one planet has turned exactly k more rounds than the other, the configuration would be the same again, but ϕ_{rel} would have increased by $k \cdot 2\pi$). So we need to solve the following equation:

$$\phi_{rel}(t) = \phi_{nes} + 2\pi \left(\frac{T_1 - T_2}{T_1 T_2} \right) (t - t_0) = \phi_{nes} + k \cdot 2\pi \quad \Rightarrow \quad (t - t_0) = k \frac{T_1 T_2}{T_1 - T_2} \quad (5.8.3)$$

This equation gives, for different k , the occurrences of all launch possibilities (and for $k = 0$ the launch possibility we just missed). However, we only want to know when the next launch could take place. This is for $k = \pm 1$ (where the sign depends on whether $T_1 - T_2 > 0$). So the synodic period can be calculated using:

$$T_{syn} = (t - t_0) = \left| \frac{T_1 T_2}{T_1 - T_2} \right| \quad (5.8.4)$$

It is interesting to note that if $T_1 = T_2$, the synodic period is infinite.

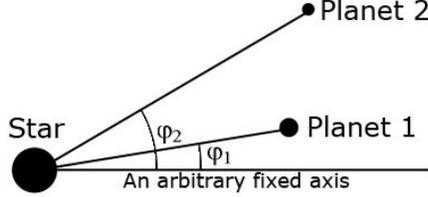


Figure 5.1: Clarification of the synodic period.

5.9 Lay-Over Time

Suppose we have traveled all the way from planet 1 to planet 2, and are in orbit there, but want to get back. How much time must there minimally be between arrival and departure? Or in other words, what is the **lay-over time**? For that, we once more need to look at angles between the two planets and some arbitrary axis. Let's call the lay-over time T_w . The angle which the satellite has at the start and the end of the mission are:

$$\phi_{sat_{begin}} = \phi_0 \quad \text{and} \quad \phi_{sat_{end}} = \phi_0 + \pi + \pi + T_w \omega_2 \quad (5.9.1)$$

where the first π is for the road to planet 2 and the second π is for the road back. Also note that $\omega_2 = \frac{2\pi}{T_2}$. In the meanwhile, the angle which the satellite has at the start and the end of the mission are:

$$\phi_{earth_{begin}} = \phi_0 \quad \text{and} \quad \phi_{earth_{end}} = \phi_0 + T_h \omega_1 + T_h \omega_1 + T_w \omega_1 \quad (5.9.2)$$

where T_h is the time needed for the trip. Also note that $\omega_1 = \frac{2\pi}{T_1}$. At the end of the mission, the condition $\phi_{earth} - \phi_{sat} = k \cdot 2\pi$ must be true, and so:

$$\phi_0 + 2T_h \frac{2\pi}{T_1} + T_w \frac{2\pi}{T_2} - \phi_0 - 2\pi - T_w \frac{2\pi}{T_2} = k \cdot 2\pi \quad (5.9.3)$$

Working this out gives:

$$k + 1 = 2 \frac{T_h}{T_1} + \frac{T_w}{T_1} - \frac{T_w}{T_2} = 2 \frac{T_h}{T_1} \pm \frac{T_w}{T_{syn}} \quad \Rightarrow \quad T_w = \pm T_{syn} \left(k + 1 - 2 \frac{T_h}{T_1} \right) \quad (5.9.4)$$

The \pm is present, because the equation for the synodic period was in absolute value. If $T_2 > T_1$ there should be a plus, and otherwise there should be a minus. To get the minimum lay-over time, we have to find the k for which T_w is minimal, but not smaller than 0. This is usually a matter of trial and error, based on the numbers given.

5.10 Swing-By's

Swing-by's are very handy for long trips, as they give a (free) velocity increase. This velocity increase seems to be inconsistent with conservation of energy, but this is not the case. The satellite approaches the target planet with a certain velocity (with respect to the planet), and it leaves the planet with the same velocity (still with respect to the planet). With respect to the sun, the velocity vector of the satellite has changed direction, and thus, seen from the sun, the velocity of the satellite has increased.

The change in angle of the velocity vector can be calculated, based on the (hyperbolic) orbit around the planet. The deflection angle δ can be calculated using the simple relation $\delta = 2\theta - 180^\circ$, where θ is the angle such that $r \rightarrow \infty$. Thus $\cos\theta = -1/e$ and this make δ :

$$\delta = 2 \arccos\left(-\frac{1}{e}\right) - 180^\circ = 2 \arcsin\left(\frac{1}{e}\right) \quad (5.10.1)$$

The change of velocity of the satellite now depends on the eccentricity, the velocity of the planet and the way of approaching the planet.

6. Special Earth Orbits

6.1 Orbital Properties

The orbital plane (determined by \mathbf{r} and $\dot{\mathbf{r}}$), with the angular momentum vector \mathbf{H} perpendicular to it, always runs through the center of gravity of the earth. The point where the satellite passes the equator in northward direction is called the **ascending node** (AN), while the southward crossing is the term descending node. The line between the nodes, called the **line of nodes**, also goes through the earth's center of gravity.

The **inclination**, as was already discussed, is the angle between the orbital plane and the equatorial planes, but it's also the angle between the angular momentum \mathbf{H} and the earth's rotation vector. If $0^\circ \leq i < 90^\circ$, then the rotation is **prograde**, meaning in the same direction as the rotation of the earth. If $90^\circ < i \leq 180^\circ$, the rotation is **retrograde** - against the rotation of the earth. The inclination for **geostationary orbits** is always $i = 0^\circ$ and for **polar orbits** it always is $i = 90^\circ$.

6.2 Orientation Changes

If a satellite, going in some direction, wants to change its direction by an angle of β , while not changing the magnitude of the velocity, the necessary velocity change is:

$$\Delta V = 2V \sin \frac{\beta}{2} \tag{6.2.1}$$

This is called an **out-of-plane manoeuvre**, and changes the orientation of the plane. For out-of-plane manoeuvres executed in the equatorial plane, only the inclination changes. For out-of-plane manoeuvres in any other point, both the inclination as the line of nodes changes.

When launching from a point on earth with latitude ϕ , the inclination of the orbit is limited to the range $|\phi| \leq i \leq 180^\circ - |\phi|$. So if a launch site not on the equator wants to put a satellite in a geostationary orbit, an additional velocity change is needed when the satellite crosses the equatorial plane. At the end of its transfer orbit from **Low Earth Orbit** (LEO) to **Geostationary Orbit** (GEO), normally a velocity change of $\Delta V = V_{geo} - V_a$ is needed, where V_a is the velocity in the apogee of the transfer orbit and V_{geo} is the velocity for a geostationary orbit. This velocity changes sets the satellite in the right orbit. However, when also a change in angle, due to the inclination of the orbit, is necessary, the velocity change, due to the cosine rule, is the following:

$$\Delta V = \sqrt{V_{geo}^2 + V_a^2 - 2V_{geo}V_a \cos i} \tag{6.2.2}$$

6.3 Orbit Perturbations

As was already discussed earlier, real orbits aren't perfect **unperturbed (Kepler) orbits**, but are influenced by **perturbing** forces. These forces don't have to be bad, since they can be used. The largest perturbation is the so-called **J_2 -effect**, caused by the flattening of the earth. The **geometric flattening** is $f = (R_e - R_p)/R_e = 1/298.257$, where R_e is the equatorial radius and R_p the polar radius.

The geometric flattening causes the earth's gravitational field to be flattened to. This is expressed by the parameter $J_2 = 1.082627 \cdot 10^{-3}$. J_2 causes periodic orbit perturbations which cancel out over an entire orbit. However, secular effects cause a rotation of the orbital plane around the polar axis, and a slow precessing of the orientation of the ellipse in the orbital plane. The secular effects thus influence ω and Ω (remember the Keplerian elements), but do not influence i , a and e .

6.4 Nodal Regression

The extra mass of the equator causes a torque which changes the angular momentum vector. This causes the line of nodes to rotate against the rotation of the satellite in its orbit. This is called **Nodal Regression**. After one revolution, the line of nodes has rotated by:

$$\Delta\Omega = -3\pi J_2 \left(\frac{R_e}{p}\right)^2 \cos i \quad (6.4.1)$$

where p is still the Semi-latus Rectum of the orbit (for circles it is just the radius r). Note that this is maximum when $i = 0^\circ$ and minimum when $i = 90^\circ$. A special application of nodal regression is the so-called sun-synchronous orbit. Now the orbit is chosen such that $\dot{\Omega} = 360^\circ$ per year. The sun always makes the same angle with the orbit. This is used because the satellite can continuously see the sun, the satellite overflies a certain latitude always at the same local solar time (illumination conditions are the same) and instruments looking outward can cover the entire celestial sphere in half a year, without disruption by the sun.

When calculating the inclination of a circular, sun-synchronous orbit, you first have to calculate $\dot{\Omega} = \frac{\Delta\Omega}{\Delta T}$. We know $\Delta\Omega$ after one revolution. The change in time after one revolution is $\Delta T = 2\pi\sqrt{r^3/\mu}$. Now $\dot{\Omega}$ is:

$$\dot{\Omega} = \frac{\Delta\Omega}{\Delta T} = \left(-3\pi J_2 \left(\frac{R_e}{p}\right)^2 \cos i\right) \left(\frac{1}{2\pi} \sqrt{\frac{\mu}{r^3}}\right) \quad (6.4.2)$$

For a sun-synchronous orbit, $\dot{\Omega} = 360^\circ/\text{year} = (2\pi)/(365.25 \cdot 24 \cdot 60 \cdot 60) = 1.99102 \cdot 10^{-7}$ rad/s. From this condition the necessary inclination i can be found.

6.5 Precession of Perigee

Also the perigee 'suffers' from precession. The **precession of the perigee** per orbital revolution is expressed by:

$$\Delta\omega = \frac{3}{2}\pi J_2 \left(\frac{R_e}{p}\right)^2 (5 \cos^2 i - 1) \quad (6.5.1)$$

When $\cos i = 1/\sqrt{5}$, the orientation of the ellipse in the orbital plane is fixed. This is called the **critical inclination** and is for earth: $i = 63.43^\circ$ and $i = (180 - 63.43) = 116.57^\circ$. This principle is used by the Russian Molnya satellite orbits. To have long contact time, the Molnya satellite orbits are eccentric orbits with its apogee (where the velocity is low) above Russia. To prevent the position of the apogee (and thus also the position of the perigee) from changing, these orbits have the critical inclination.