5

The drivers for the approach strategy

The major natural and technical features and constraints which (along with trajectory safety) are the driving forces behind the design of the approach strategy will be discussed in this chapter. The consequences on trajectory elements and approach strategy for the various natural and technical issues will be indicated. Trajectory safety remains the overriding requirement; this always has to be kept in mind when discussing all other potential design drivers. Three examples of approach strategies with different constraints are discussed at the end of the chapter, for which, within the context of a complete approach scenario, a detailed explanation of the rationale behind the choice of trajectory elements of the different rendezvous phases is provided.

5.1 Overview of constraints on the approach strategy

The most important disturbance which has to be taken into account in the *launch strategy* is the drift of nodes due to the J_2-effect, described in section 4.2.2. Because of the difference in orbital altitude, this drift will be different for chaser and target over the duration of the approach. The difference will therefore have to be compensated for by corrective measures during launch and phasing. The *phasing strategy* is mainly driven by the difference in position between the target station and the chaser vehicle after launch and by the required arrival time at the target. The *strategy of the rendezvous and final approach phases* will be determined additionally by many other physical, technical and operational constraints that will be present as a result of:

- geometrical conditions for mating at the target station, such as the location of capture interface on the target station, the availability and location of sensor interfaces on the target station, and the attitude of the target station;

- the capabilities of the sensors for trajectory and attitude control and of other equipment aboard the chaser and target vehicles, and on ground, which are used in the rendezvous process (this includes issues such as measurement range, measurement accuracy and field of view);
- needs and capabilities for monitoring on ground and by the target crew;
- availability of crew on the target vehicle for such operations;
- rules for the approach defined by the target;
- constraints by onboard resources.

All rendezvous strategies consist of a sequence of orbital manoeuvres and trajectories, which mostly have a fixed duration, i.e. typically one or one half orbital revolution. Visual monitoring of the last few metres of approach and capture by remote operators on the target station or on ground requires, however, particular illumination and communication conditions. Unfortunately, the occurrence of such conditions is not necessarily synchronised with the sequence of orbital motions required by the approach scheme. Such monitoring conditions and constraints are as follows.

- The illumination of the capture interfaces, i.e.
  - in the case of docking, the docking port of either chaser or target, depending on the location of video equipment or viewing capability,
  - in the case of berthing, the grapple fixture for capture by a manipulator.

  Either natural illumination by the Sun or artificial illumination can be used. For Sun illumination, availability and direction of sunlight is a major issue for visual monitoring during the last part of the final approach and during the mating process (see section 5.4.1). For artificial illumination, the last few metres of approach and capture can best be performed during orbital night, in order to avoid light disturbances by the Sun.
- The visibility windows, during which ground can communicate with the chaser and target spacecraft, either directly by ground stations or via relay satellites (see section 5.4.2).
- The data rate transmission and reception capabilities of space and ground equipment, in particular when video pictures are required for monitoring (see section 9.3).
- The communications delay due to space–ground signal travel time, e.g. via relay satellites, and due to ground link delays.
- The maximum range of direct communication between chaser and target station.

- The work, rest and sleep schedule of the crew in the case of a manned target (section 5.4.3). Monitoring by crew is a 'must' for manned vehicles, but availability of crew is usually limited.

Synchronisation of the timeline will obviously become more difficult as more and more of these requirements have to be met at the same time. In particular, the combination of Sun illumination requirements during the last part of the approach and capture with the requirement for communication windows can, in some cases, lead to an extreme reduction in mission opportunities.

Another issue that plays an important role in the concept of the approach strategy is the amount of participation of the target in the final approach operations. This cooperation can range from completely uncontrolled and passive (e.g. the case of an incapacitated target vehicle); via passively cooperative (e.g. a target vehicle with fixed attitude, communication and sensor interfaces); to actively cooperative (e.g. the target performs attitude manoeuvres to facilitate docking, or performs actively the berthing operations, etc.).

If the target crew has to fulfil an active role as back-up for the automatic GNC of the chaser vehicle (see section 6.5), the trajectory has to be designed such that an immediate take-over by manual control is possible. This is generally the case only for straight line trajectories.

A further issue, which has significant repercussions on the approach strategy, concerns the above-mentioned control rules defined by the target. These include, e.g., the traffic and safety control zones around the target, as defined by the International Space Station, and the rules which have to be observed therein. This subject will be addressed in more detail in section 5.6.

The constraints due to onboard resources in terms of propellant, power and heat dissipation capabilities will be addressed shortly. Such constraints can affect both nominal and back-up strategies. The latter determine the necessary time and propellant for retreat and return, and potentially include long waiting times, with repercussions on power resources. As the nominal approach strategy has to foresee, at all points of the approach, a back-up solution, the back-up needs will in turn influence the concept for the nominal approach strategy.

5.2 Launch and phasing constraints

5.2.1 The drift of nodes

With otherwise identical orbit parameters, the rate of drift of nodes depends on the altitude of the orbit (see Eq. (4.8)). The difference in the drift of nodes between chaser and target during phasing can be several tenths of a degree per day. In the example in section 4.2.2, it was assumed that the target is in a 400 km altitude circular orbit and that the chaser is in a 350/200 km elliptical orbit, leading to a difference of 0.337 deg/day for a 52 deg inclination orbit plane and of 0.482 deg/day for a 28.5 deg inclination orbit plane.

As a result, when launched into the same orbital plane as the target, the chaser would end up, when arriving at rendezvous distance, in a plane with the same inclination but a different RAAN angle. For this reason it will have to be injected into a 'virtual target plane' with a different RAAN, such that the difference will have disappeared when the chaser arrives at the target, due to different drift of nodes along the way. This 'virtual target plane', into which the chaser has to be launched, can only be estimated, as it obviously depends on the further evolution of the actual trajectories until mating. Final adjustment of the RAAN angle has to be performed during phasing and must be concluded, except for minor calibrations, prior to the start of the close range rendezvous operations.

5.2.2 Adjustment of arrival time

Because of the problem of synchronisation of the approach timeline with the occurrence of suitable illumination conditions and communication windows, the phasing duration of the chaser after launch will not only be determined by the phase angle to the target. Since (a) launch has to take place at a certain point in time to meet the orbital plane conditions, and (b) final approach and mating have to take place at another independent fixed point in time to meet illumination and communication conditions, some flexibility in time will be needed on the way between those two points. The approach trajectories between launch and mating thus have to be designed such that the time conditions can be met in addition to compensation of the differential drift of nodes addressed above. For this purpose, time-flexible trajectory elements (see section 5.4.4) will be needed to ensure compliance with both requirements. For phasing this could be a drift for different durations at two fixed altitudes, as shown in figure 5.22.

For a given orbit plane and target position, the required conditions of launch window, Sun illumination and communication windows at mating will not be available at all times of the year. Once the point in time where all synchronisation requirements for the final approach and mating can be met has been determined, planning for the operations of all entities involved must commence. These are the control centres of the chaser and target, the target crew, and the communication infrastructure, such as relay satellites and ground links (see chapter 9). In particular, use of this infrastructure is costly, and its availability must precisely be planned for this time. After a missed final approach opportunity, the following one may be available only after some considerable waiting time, due to the necessary re-acquisition operations and the duration of all necessary preparations by the parties involved.

5.3 Geometrical and equipment constraints

5.3.1 Location and direction of target capture interfaces

The strategy for the last part of the final approach depends to a large extent on the location of the docking axis or the berthing box. This will determine whether a +V-bar, a −V-bar or an R-bar approach has to be performed, which in turn will determine the strategy of the previous phase. Depending on whether or not a potentially longer waiting time is needed, because of operational or other reasons (cf. section 9.1), a hold point on V-bar may be required. For an R-bar approach this will require first a V-bar acquisition and, after the hold point, a fly-around manoeuvre to acquire R-bar. If it can be foreseen that no such hold will be required, an R-bar approach can, of course, be initiated directly, e.g. from a lower (drift) orbit, without transferring the chaser to the target orbit first. However, as long as it is uncertain whether the possibility of a waiting time has to be envisaged, the application of an approach strategy with a hold point will be advisable.

Repercussions of port locations on the approach strategy

In cases where the target is a large orbital assembly consisting of several modules, final translation may not take place on V-bar or R-bar, but on lines parallel to it, according to the location of the docking port or the berthing box in relation to the CoM of the station. For docking this situation is shown schematically in figure 5.1, and for berthing it is shown in figure. 5.4.

Whereas the approach on a line parallel to R-bar (line through the CoM of the target vehicle) has no further consequences concerning the orbital dynamics, for an approach parallel to V-bar, orbit dynamics have to be taken into account (Eqs. (3.25), (3.44) and (3.26)). For example, the loss of control during a hold point on a docking axis above V-bar ($-\Delta z$) would result in a looping motion in the −V-bar direction and on an approach axis below V-bar ($+\Delta z$) in a looping motion in the +V-bar direction (see figure 3.11). For the approach in a closed loop controlled motion, the result would just be somewhat higher or lower propellant consumption, depending on the direction of approach, i.e. whether to a +V-bar or −V-bar port or a berthing box.

The distance in the z-direction of the approach axis from the actual target orbit line is very important for trajectory safety. In the case of loss of control, the resulting drift trajectory will become more or less safe, which means it will move either away from or toward the target. Direction and velocity of the relative motion will depend on side and direction of the approach, on the approach velocity and on the distance from V-bar. This influence, e.g., of the z-distance is shown in the two examples of figure 5.2 for the loss of control during a straight line approach. The chosen z-distances are 10 m above or below V-bar, and the assumed approach velocity is +0.1 m/s in the V-bar direction. Thrust is inhibited at a distance of $x = 60$ m from the CoM of the target. The trajectory starting below V-bar (positive R-bar side) is less safe, as it moves closer to the target.

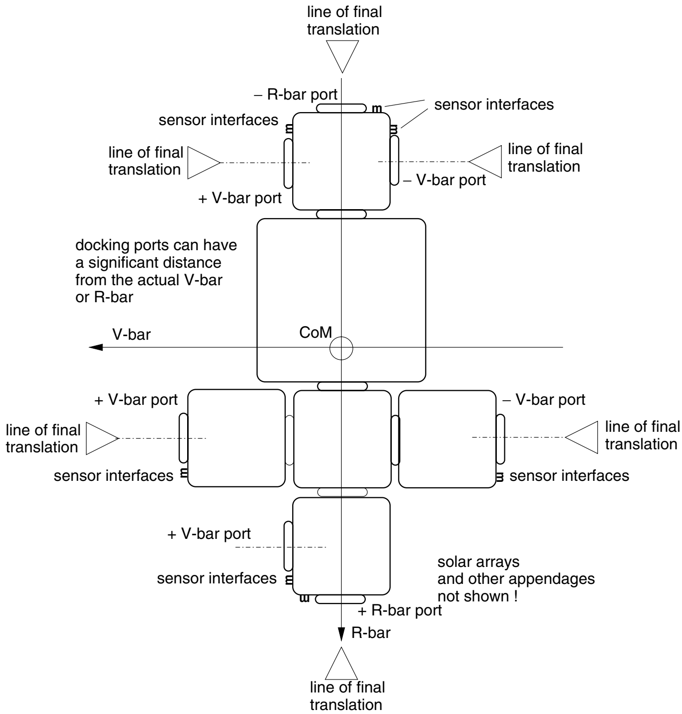

Figure 5.1. Location of docking ports w.r.t. the local orbital frame.

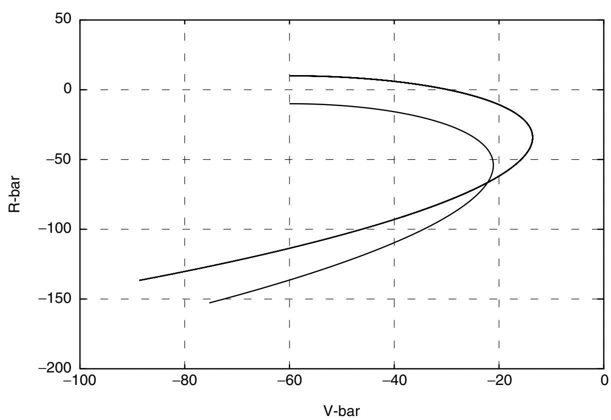

Figure 5.2. Thrust inhibit on 0.1 m/s V-bar approach at $z = \pm 10$ m.

Particularly sensitive in the cases of thrust inhibit or loss of control are hold points at a z-distance from V-bar. In these examples, shown in figure 5.3, it is assumed that the hold point is on the approach line to a docking port or berthing box at a z-distance of 10 m above or below V-bar and at a distance of $x = \pm 20$ m from the CoM of the target. For an approach on the $-$V-bar side, the critical condition concerning trajectory safety is a docking port below V-bar, i.e. with an approach trajectory at a $+\ z$-distance from V-bar. For an approach on the $+$V-bar side, a docking port above V-bar, i.e. with an approach trajectory at a $-$z-distance, will be more critical.

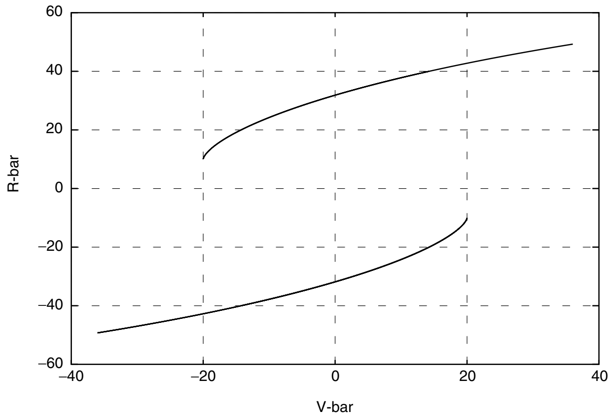

Figure 5.3. Example. Loss of control on hold points at z $= \pm 10$ m.

Any direct approach to a port located at $\pm$H-bar is inherently unsafe. In the cases of thrust inhibit or loss of control of the spacecraft due to other failures, the vehicle will continue to move toward the station, with the highest velocity achieved when crossing V-bar (see Eqs. (3.27)). For the connection to H-bar ports, berthing is preferred, whether by transfer directly from a berthing box or after initial docking from V-bar or R-bar docking ports.

Approach to a berthing box

The direction of approach to a berthing box is dictated neither by the location nor the direction of the berthing port axis. Depending on the reach and articulation capabilities of the manipulator arm used, convenient berthing box locations can be selected, from where the captured vehicle can be transferred to the structural interfaces of the berthing port, i.e. the berthing mechanism. The approach direction to a berthing box is then driven by the geometric shape of the vehicles, by the location and reach capability of the manipulator, by the location and nominal attitude of the corresponding capture interfaces (grapple fixture) and by trajectory safety considerations. The location of the berthing boxes, their relation to manipulator and berthing port positions on the target station,

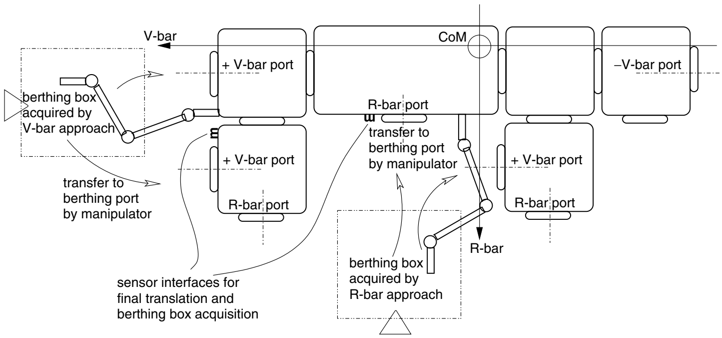

Figure 5.4. Location of berthing ports and berthing boxes w.r.t. the local orbital frame.

and their access by V-bar or R-bar approaches, are shown schematically in figure 5.4. In this example, the manipulator is located on the target, which is usually the case for large space stations (it can also be located, however, on the chaser, as in the Space Shuttle). In addition to geometric conditions and manipulator reach, availability and location of relevant rendezvous sensor interfaces will be major drivers for the location of, and approach direction to, the berthing box.

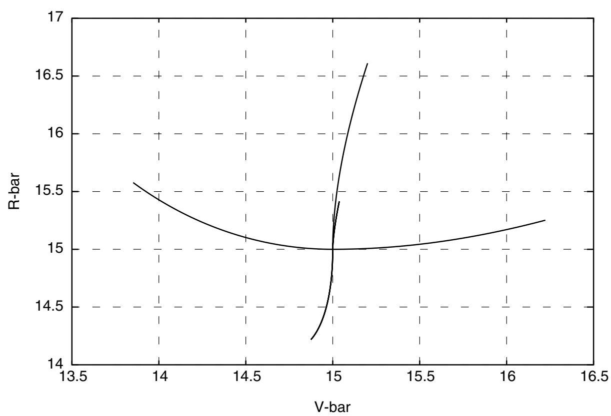

Figure 5.5. Example. Drift trajectories in berthing box, (v_x, $v_z = \pm 0.01$ m/s).

In the berthing box, control of the chaser will have to be switched off prior to capture to avoid competing control actions between chaser GNC and manipulator. In cases where the berthing box is located above or below V-bar, orbit dynamics will move the vehicle according to Eqs. (3.26) after control inhibition. For this reason, there will be only limited time for the manipulator to grapple the vehicle's capture interfaces. In the example shown in figure 5.5 the centre of the berthing box has been assumed to be 15 m below V-bar and the trajectory evolution has been calculated for 2 minutes. For the nominal case, the natural motion during this time would be approximately 0.4 m in the R-bar direction. However, there will always be residual motion after shut down of the control system, which in addition will determine the trajectory evolution. In the example, residual velocities of 0.01 m/s in the $\pm x$- and $\pm z$-directions have been assumed. As a result, drifts of more than 1.5 m occur. The example indicates three important rules for design and operation:

(1) the berthing box should be as close as possible to V-bar,
(2) the residual velocities after control inhibition shall be as low as possible,
(3) the manipulator should be able to grapple the vehicle within a very short time.

Three different regions (volumes) can be identified for the berthing box, as defined in figure 5.6 (Lupo 1995; Bielski 1998):

- the station keeping volume (inner berthing box), which is required for the accuracy with which the chaser GNC can position the grapple interfaces w.r.t. the target coordinates and which includes the control motion;
- the capture volume, in which the capture of the vehicle has to be achieved; this adds to the station keeping volume the free drift which takes place after thrust inhibit up to the point in time when the manipulator end effector has captured the grapple interfaces;
- the total berthing box volume (outer berthing box), which adds to the capture volume the distance necessary to stop the manipulator motion.

In order to determine the allowable position of the box, the maximum outlines of the chaser vehicle (as measured from the grapple interface and including maximum possible attitude angle after capture) must be added to the outer berthing box volume, and a safety margin around the target vehicle structure must be considered. The manipulator must be able to reach the entire envelope of the outer berthing box, i.e. the capture box plus the braking envelope. For this reason, the resulting volume, taking all constraints into account, will not necessarily be of cubic form.

Since, after capture, the transfer to a berthing port does not depend on the approach direction but on the capabilities of the arm, the axis of the berthing port can have a completely different direction than the approach line. This is one of the major advantages of

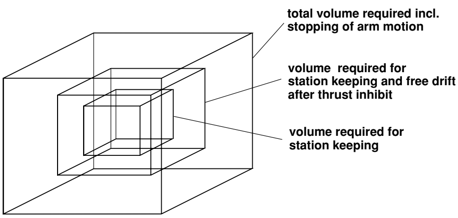

Figure 5.6. Definition of berthing boxes.

berthing. In this way connections of new elements can be made to the target station at places which are not accessible or difficult to access by direct docking. The major disadvantage of berthing is the much increased complexity of the mating process in terms of hardware, software and human operator functions involved, and of operations to be performed, with the corresponding consequences on duration and use of resources. This will also have repercussions on reliability and safety of the process, as there are now three additional active elements involved in the process, i.e. the target vehicle systems, the human operator and the manipulator arm, each adding potential failures and errors.

Approach to a docking port with an attitude angle

For a number of reasons, the target station may not be aligned with the local vertical/local horizontal (LVLH) directions. It could be pointing toward the Sun (e.g. for power reasons), be in a natural stable torque equilibrium attitude (equilibrium of major torques acting on the spacecraft, e.g. caused by gravity gradient and residual air drag effects), or pointing in a particular direction for communications reasons, etc. Although in the case of docking the chaser vehicle will eventually have to acquire the docking axis to be able to perform capture and docking, not all parts of the final approach need to be performed on the docking axis.

When the short range rendezvous sensor on the chaser acquires the sensor interfaces on the target (e.g. retro-reflectors for optical rendezvous sensors) at a distance of a few hundred metres, the parameters to be measured are the range and direction between the two vehicles. Measurement of relative attitude will typically commence (due to sensor limitations) only at a relatively short range, i.e. below 50 m. The range for which relative attitude information must be available results from the dynamics of the acquisition of the docking axis. This means that acquisition of the docking axis must take place at a distance that is large enough for the lateral trajectory oscillations to be damped out before arriving at the docking port (cf. figure 2.12).

The approach to a target with large attitude angle can be envisaged in the following way. Up to the point where relative attitude measurement is available, the attitude of the chaser will be controlled by the absolute attitude sensors, e.g. w.r.t. the LVLH frame. Assuming the use of optical sensors, the position of the chaser can be controlled such that the target reflectors will be in the centre of the field of view of the rendezvous sensor, whereby the absolute attitude of the chaser will be kept aligned to LVLH. The result will be that the approach trajectory of the chaser is a line parallel to V-bar or R-bar (depending on the direction of the docking axis), defined by the position of the sensor interfaces on the target.

This situation is shown for a V-bar approach in figure 5.7. On the right side of the figure the dispersion ellipsoid is shown, which results from the measurement accuracy of the sensor used in the previous phase. In the upper example an error radius of $r_{\max} = 10$ m is assumed; in the lower one there is an error radius of $r_{\max} = 20$ m. Further, it is assumed that the docking port is at a distance of 30 m from the CoM of the target station and that the station has an attitude angle of $+20$ deg (upper example) and -15 deg (lower example), respectively. After sensor acquisition, the GNC system will guide the chaser automatically to the approach line determined by the sensor interfaces and by its own attitude control accuracy. Note: with these measurements the GNC system of the chaser cannot determine where V-bar is (V-bar is defined in the target local orbital frame). This fact has to be taken into account for trajectory safety considerations.

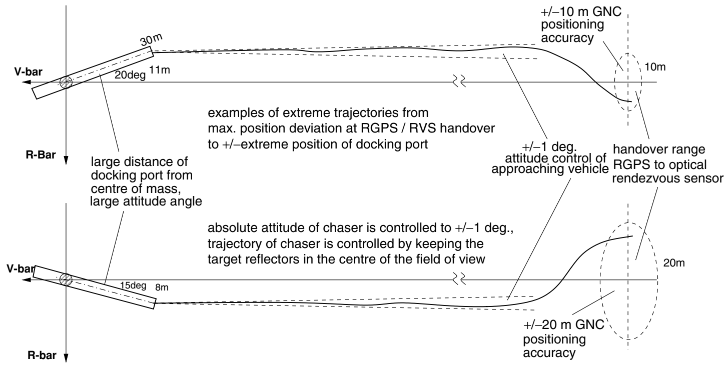

Figure 5.7. Final approach trajectory to target station with attitude angle.

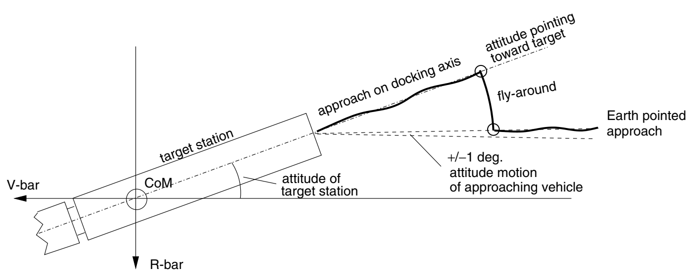

Figure 5.8. Acquisition of docking axis on target station with large attitude angle.

As the angular reception range of the docking mechanism is limited, the chaser will eventually have to acquire the instantaneous docking axis in order to perform the last part of the approach into the mechanical interfaces along this centre line. For this purpose, knowledge of the relative attitude between the two vehicles is necessary. The acquisition of the instantaneous docking axis in the case of small relative attitude deviations has been addressed already in section 2.4.2 and is illustrated in figure 2.12. For larger relative attitude angles, the visibility of the retro-reflectors may be lost, and it may no longer be possible to handle the acquisition of the docking axis by the same GNC mode as used for the approach. In this case, a fly-around manoeuvre has to be implemented. This situation is illustrated in figure 5.8. The fly-around could be implemented, e.g., as a circular fly-around manoeuvre, such as in figure 3.31, or as a two-impulse radial manoeuvre, as illustrated in figures 3.19 and 3.22. As such manoeuvres have to be performed in the very close vicinity of the target vehicle, trajectory safety will be the over-riding consideration for its implementation.

Large attitude angles of the docking port w.r.t. V-bar or R-bar direction can occur when a module carrying the port is attached to the target station under a certain angle, when the station itself flies under a certain angle, e.g. a torque equilibrium attitude (TEA), or is inertially pointing, e.g., toward the Sun for power reasons.

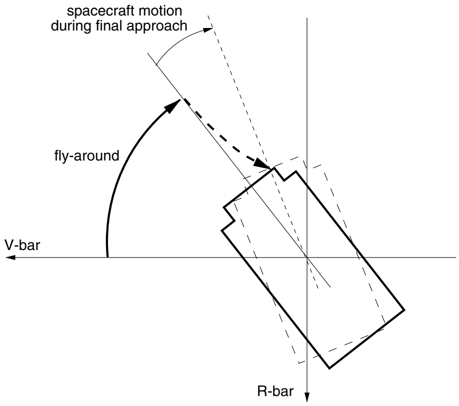

Figure 5.9. Fly-around and approach to inertially pointing target.

The case of an inertially pointing target (see figure 5.9) is very different from all the other cases, as the target is rotating, i.e. it is continuously changing its attitude w.r.t the local orbital frame $\boldsymbol{F}_{\mathbf{lo}}$. This requires the chaser, when on the docking axis, to follow the motion with the orbital rate ω of the target docking frame, which is 1.14×10^{-3} rad/s in a 400 km orbit. As a result, the chaser has to provide this rotation and the corresponding lateral velocity $r \cdot \omega$, which amounts to 0.114 m/s at $r = 100$ m and 0.0114 m/s at $r = 10$ m from the docking port. For an inertially pointing target, the acquisition of the docking axis can take place when the latter has turned, e.g., into either the $+$ or $-$V-bar direction, where the chaser is waiting. Docking axis acquisition may possibly be combined with a fly-around manoeuvre to obtain a suitable attitude of antennas w.r.t. ground or other satellites. The latter case is shown in figure 5.9. A radial impulse fly-around will have, in this context, the advantage of keeping a constant direction of chaser–target line w.r.t. the Sun. Approaches to an inertially pointing target have been used in the past in the Russian Mir Space Station scenario. However, as trajectory safety conditions are changing continuously for an approach to an inertially pointing target, a CAM would have to be calculated separately for each point of the trajectory. For this reason, V-bar and R-bar approaches are generally preferred. Approaches to an inertially pointing docking axis are nowadays mainly considered for rescue and repair missions to target spacecraft, which are in Sun-pointing mode for power reasons.

5.3.2 Range of operation of rendezvous sensors

The rendezvous approach process requires that the lateral displacements and velocities relative to the target will decrease commensurate with the decrease of distance on the approach line to the target. This results in a continuous increase of navigation accuracy requirements over the approach sequence, which cannot be fulfilled by one sensor over the entire approach range. Most sensors have a limited range of operation, which determines the extension of the particular approach phase in which it will be used. The required accuracy of the sensor measurements follows from the error ellipsoid for position and velocity, which has to be achieved in the following manoeuvre. This is explained in more detail in section 7.1.1.

The maximum operational range of a sensor is, in most cases, constrained by the power with which it can emit its signal. Exceptions are satellite navigation systems, such as GPS and GLONASS, and sensor principles, which make use of external power sources, such as Sun illumination.

- For radio-frequency based rendezvous sensors (e.g. radar) the maximum range is typically of the order of 100 km.
- For absolute GPS there is no limitation concerning the range of use in a RVD approach trajectory, as GPS satellites cover the entire surface of the Earth.
- For relative GPS, which uses the raw data of the GPS receivers on both chaser and target, the maximum range will be limited by the range of direct communication between the two vehicles.
- For optical sensors the maximum range is typically a few tens of metres to a few hundred metres, depending on the target area to be illuminated.

An overview of operational ranges and accuracies of sensors for rendezvous trajectory control is shown schematically in figure 5.10. The following remarks need to be made in support of this diagram.

The use of GPS and RGPS will be constrained at the lower end of the range of the operating range by shadowing and multi-path effects (see section 7.3.4). In particular, for large and complex structures such as a space station, with many large surfaces in various directions and even very large rotating elements, such as the solar arrays and radiators, shadowing is a significant problem.

After abandonment of the process of degradation of the GPS signal, known as 'selective availability' (SA), the accuracy of position determination by absolute GPS has improved by a factor of 10 (for further details, see section 7.3.2). The absolute GPS accuracy in figure 5.10 comes close to that of relative GPS. It has to be taken into account, however, that the calculation of the relative state of the chaser w.r.t. the target still requires measurements on the target side with the same accuracy and that the subtraction of very large numbers introduces the possibility of large errors.

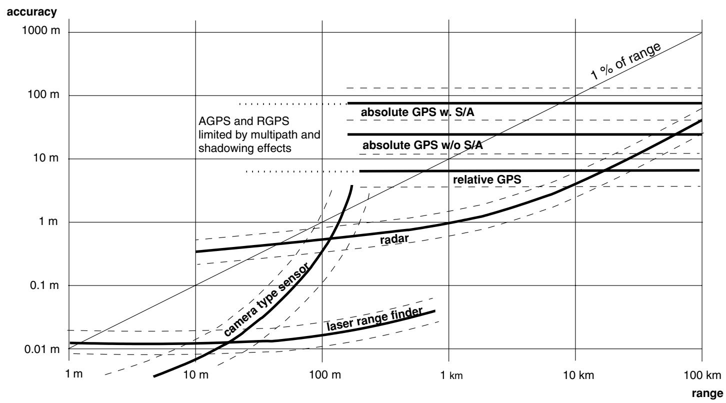

Figure 5.10. Typical operational ranges and measurement accuracies of rendezvous sensors.

Optical sensors, e.g. camera and laser range finder types (for more details see chapter 7), have an additional operational constraint for the approach trajectory design, i.e. a limited field of view (FOV). This puts limits on the lateral extension of the trajectory. An example is shown in figure 5.11: in the case of a transfer along V-bar by impulsive radial manoeuvres, the extension of the trajectory in the y-direction is one half of the advance in the x-direction, with the maximum half way. To cover such a trajectory very close to the target, an optical sensor would need to have a 60 deg FOV (30 deg half cone angle). Because of the power required to illuminate (camera sensor), or the duration required to scan such a large FOV (scanning laser), compromises have to be made, which result in relatively small FOV for most optical sensors.

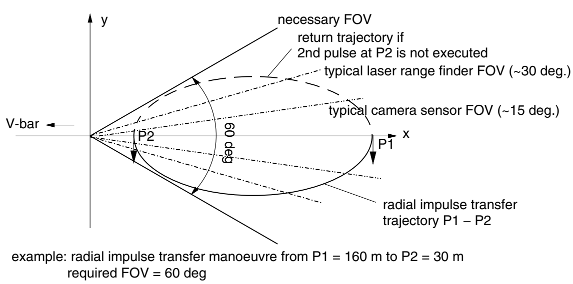

Figure 5.11. Required and available field of view (FOV) for a radial impulse transfer.

Optical sensors, with their limited FOV, are particularly suitable for the last part of the docking approach. During the final part of the approach to the docking port, the trajectory needs to be close to and the attitude aligned with the docking axis, in order to meet the reception range of the mechanical interfaces. During the last part of an approach to a berthing box, sensors with a small FOV are also quite suitable, if the chaser can point towards the sensor interface on the target. For other parts of the approach strategy, which include impulsive manoeuvres, fly-arounds and attitudes, which are not necessarily pointing towards the target, sensors with larger FOV will be required.

As a result, for the various rendezvous phases, different sensor types will have to be chosen, and, in turn, according to the operating range and performance of sensors, the various rendezvous phases will have the appropriate ranges. The following list gives the approximate ranges of approach phases and the preferred types of sensors (see section 7.1.1).

- *Final approach*. Sensor: laser range finder or camera type of optical sensor, from 100–500 m to contact.

- *Close range rendezvous*. Sensor: relative GPS, radar or other type of RF sensor, from a few thousand metres to final approach range.

- *Far range rendezvous*. Sensor: absolute GPS, radar or other type of RF sensor, from 10–100 km to the start of close range rendezvous.

- *Phasing*. Sensor: absolute GPS or ground based navigation, from launch to start of far range rendezvous.

5.4 Synchronisation monitoring needs

Most of the synchronisation problems in rendezvous operations are related to the requirement that monitoring by human operators is necessary for the last part for final approach and for capture operations. Such synchronisation problems will be most severe when the human operators are located on ground. Many problems, however, will also occur when the operator is in the target station or even in the approaching vehicle itself. Those external events which must be synchronised with the final rendezvous events are as follows.

- *Proper Sun illumination conditions* during the last part of the final approach and during docking/berthing, since, in most cases, visual monitoring will rely for power reasons on Sun illumination.

- *Availability of a communication window* for transmission of video data (data rate) to ground, either directly to a ground station or via relay satellite. Video monitoring of the last part of the approach and capture operations is required by ground operators in any case if monitoring by target crew operators is not available. Even if visual or video monitoring by operators in the target is established, additional transmission of video data to ground will be advisable for safety reasons (see section 9.1.2).

- *Availability of crew member(s)* for monitoring within the crew activity timeline (only in the case of a manned target vehicle). It is assumed that, for safety reasons, target crew must always be able to supervise the last part of the approach and mating process. In some mission scenarios, target crew may be required, in contingency situations, to take over the control of the chaser vehicle manually, in order to ensure mission success (see section 6.5.3).

5.4.1 Sun illumination

Orbital day and night occur with an orbital period depending on the actual orbital radius (in a low Earth orbit (LEO) this is every 90–95 minutes). Due to the altitude above the Earth, orbital day is generally a bit longer than half the orbital period, except when the position of the Sun is nearly orthogonal w.r.t. the orbital plane. In this case, all of the orbit would be illuminated. However, not all of the illuminated part of an orbit will be suitable for visual monitoring of rendezvous and capture operations, as shadowing may make monitoring impossible. Due to the lack of refraction from the surrounding air, the contrast between light and shadowing is much more severe in space than on ground.

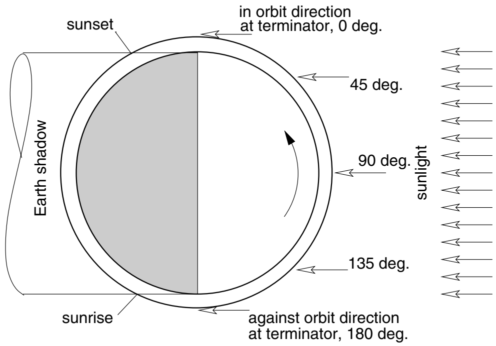

Figure 5.12. Variation of Sun elevation angle component over one orbit.

As an example, the elevation component of the Sun angle shall be considered for the easy case of an orbit with low lateral Sun angle, i.e. with the Sun being more or less in the orbital plane. This case is illustrated in figure 5.12. At orbital dawn the Sun shines in a direction opposite to the orbital velocity vector. At orbital sunset it shines in the same direction as the orbital velocity vector. At orbital noon sunlight and the orbital velocity vector are orthogonal. Assuming LVLH attitude of chaser and target, depending on the direction of the final approach, i.e. whether in +V-bar, −V-bar, the Sun will, at sunrise, illuminate fully the capture interface plane of the target station, and at sunset that of the chaser vehicle, or vice versa. For an R-bar approach, the chaser interfaces would be fully illuminated at orbital noon. If the docking port in question is fully illuminated, i.e. the Sun direction is orthogonal to the interface plane, it will be difficult to distinguish any feature on the illuminated surface. For the human eye, or a video camera, looking along the approach line, the surfaces will, in this case, all be equally illuminated and there will not be much shadowing. Also, when at close distances, one vehicle will cast its shadow on the other. On the other hand, when the Sun is orthogonal to the approach axis, structural features with extensions along the approach line will cast long shadows, resulting in little or no illumination of surfaces in the capture interface plane. This is an equally unsuitable situation for visual monitoring. Best illumination conditions for monitoring are obtained at intermediate angles, where the interface plane for capture is illuminated, shadows of structural features are not too long and shadow casting from one vehicle on the other is limited to a minimum.

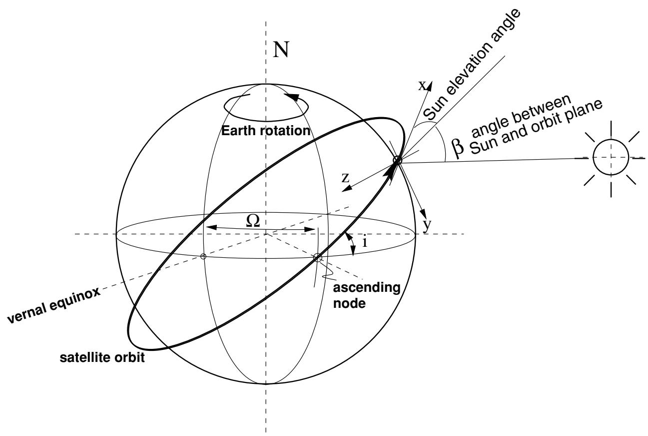

Figure 5.13. Sun angles seen from spacecraft.

In the general case, the orbit plane is not aligned with the ecliptic plane and the Sun direction will also have a lateral component β, as illustrated in figure 5.13. The maximum and minimum values that the β-angle can assume are given by the orbit inclination i plus the angle of the ecliptic w.r.t. the equator, $\varepsilon = 23.5$ deg., as shown in figure 5.14. Depending on the annual season and on the location Ω of the ascending node, the relevant component of the ecliptic angle ε has to be added to or subtracted from the orbit inclination angle i. Due to the drift of the nodes (see section 4.2.2), which can amount to several degrees per day, the β-angle can assume over time any value between the two extremes.

The direction of the Sun in the spacecraft local orbital frame F_{lo} (LVLH frame) can be calculated using the following coordinate transformations:

$$S_{\mathrm{lo}} = A_{\mathrm{lo/b}} \cdot A_{\mathrm{b/so}} \cdot A_{\mathrm{so/op}} \cdot A_{\mathrm{op/an}} \cdot A_{\mathrm{an/eq}} \cdot A_{\mathrm{eq/ec}} \cdot S_{\mathrm{ec}} \tag{5.1}$$

The position of the Sun w.r.t. the orbital plane frame is the first part of these

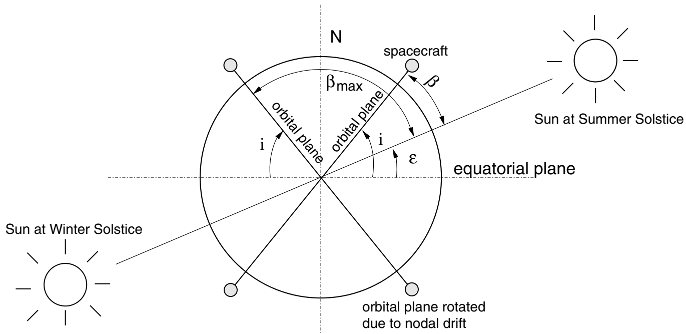

Figure 5.14. β-angle variation due to season and drift of nodes.

transformations:

$$S_{\mathrm{op}} = A_{\mathrm{op/an}} \cdot A_{\mathrm{an/eq}} \cdot A_{\mathrm{eq/ec}} \cdot S_{ec} \tag{5.2}$$

The β-angle is the arcsin of the y-component of the Sun unit vector S_{lo} in the spacecraft local orbital frame, which is the same as the $-z$-component of the Sun unit vector S_{op} in the orbital plane frame:

S_{ec} : unit vector of Sun in the ecliptic frame;

S_{op} : unit vector of Sun in the orbital plane frame;

S_{lo} : unit vector of Sun in the spacecraft local orbital frame (LVLH frame).

The transformation matrices are (see also section 3.1):

$A_{\mathrm{eq/ec}}$: from the ecliptic frame to the Earth-centred equatorial frame, rotation about x_{ec} by $-\varepsilon$;

$A_{\mathrm{an/eq}}$: from the Earth-centred equatorial frame to the ascending node frame, rotation about z_{eq} by the angle Ω (RAAN);

$A_{\mathrm{op/an}}$: from the ascending node frame to the orbital plane frame, rotation about x_{an} by the inclination i;

$A_{\mathrm{so/op}}$: from the orbital plane frame to the spacecraft orbital frame, rotation about z_{op} by the orbital phase angle ωt;

$A_{\mathrm{B/so}}$: from the spacecraft orbital frame to an auxiliary (B) frame, rotation about z_{so} by 90 deg;

$A_{\mathrm{lo/B}}$: from the auxiliary (B) frame to the spacecraft local orbital (LVLH) frame, rotation about x_{B} by -90 deg.

The relationships between the various frames used are shown in figure 5.15.

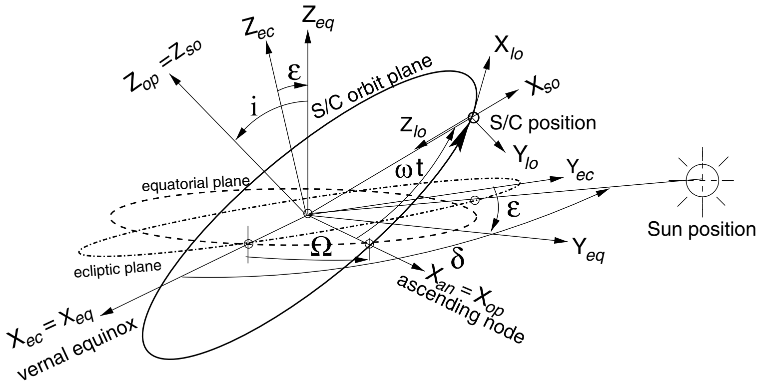

Figure 5.15. Relationships between Sun, Earth and orbit frames.

The unit vector of the Sun in the ecliptic frame is

$$S_{\mathrm{ec}} = \begin{bmatrix} \cos\,\delta \\ \sin\,\delta \\ 0 \end{bmatrix} \tag{5.3}$$

where δ is the day angle of the Sun measured from the vernal equinox.

The transformation matrices for the rotations to the various intermediate frames as defined above are

$$A_{\mathrm{eq/ec}} = A^{\mathrm{T}}_{\mathrm{ec/eq}} = \begin{bmatrix} 1 & 0 & 0 \\ 0 & \cos\,\varepsilon & -\sin\,\varepsilon \\ 0 & \sin\,\varepsilon & \cos\,\varepsilon \end{bmatrix} \tag{5.4}$$

$$A_{\mathrm{an/eq}} = \begin{bmatrix} \cos\,\Omega & \sin\,\Omega & 0 \\ -\sin\,\Omega & \cos\,\Omega & 0 \\ 0 & 0 & 1 \end{bmatrix} \tag{5.5}$$

$$A_{\mathrm{op/an}} = \begin{bmatrix} 1 & 0 & 0 \\ 0 & \cos\,i & \sin\,i \\ 0 & -\sin\,i & \cos\,i \end{bmatrix} \tag{5.6}$$

$$A_{\mathrm{so/op}} = \begin{bmatrix} \cos\,\omega t & \sin\,\omega t & 0 \\ -\sin\,\omega t & \cos\,\omega t & 0 \\ 0 & 0 & 1 \end{bmatrix} \tag{5.7}$$

$$A_{\mathrm{B/so}} = \begin{bmatrix} 0 & 1 & 0 \\ -1 & 0 & 0 \\ 0 & 0 & 1 \end{bmatrix} \tag{5.8}$$

$$A_{\mathrm{lo/B}} = \begin{bmatrix} 1 & 0 & 0 \\ 0 & 0 & -1 \\ 0 & 1 & 0 \end{bmatrix} \tag{5.9}$$

The calculation of the position of the Sun in the spacecraft local orbital frame for a given set of numerical values, for δ, ε, Ω, i and $\omega\, t$ with these transformation matrices, is easy and straightforward. The multiplication of all matrices in a general form leads, however, to very large expressions for the matrix elements, which are too complex to be used for a general assessment of the illumination situation under various conditions. It is much easier to produce the solutions in numerical or graphical form with the help of computer programs for matrix calculation.

The β-angle is variable and depends on the orbit inclination i, on the RAAN angle $\omega = f(t)$ and on the day angle δ, as shown in figures 5.13, 5.14 and 5.15. An example for the evolution of the β-angle over one year is shown in figure 5.16 for a 400 km altitude orbit with an inclination of 52 deg. It is assumed that the start time is the vernal equinox and that the ascending node of the orbit at start time is at $\Omega = 0$. The drift of nodes for this orbit is -4.989 deg/day (see section 4.2.2).

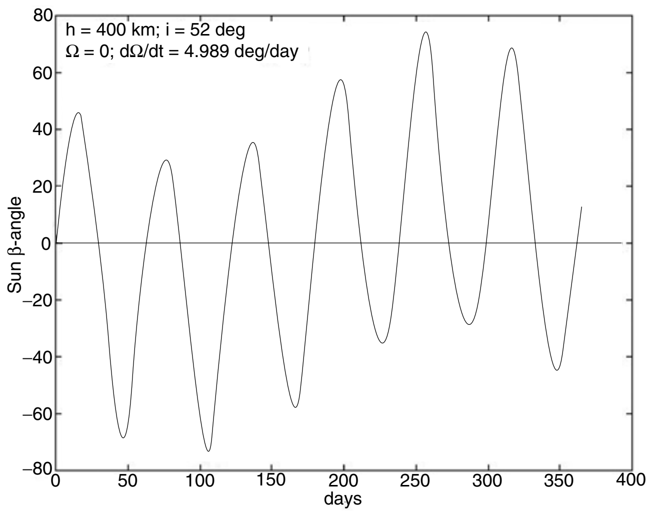

Figure 5.16. Evolution of the β-angle over time, $h = 400$ km, $i = 52$ deg.

For an approach to the target from the $-$V-bar side, at orbital sunrise the Sun will be in front of the chaser spacecraft, i.e. in the direction of the target, and will illuminate the chaser docking port side. At orbital sunset it will be the opposite way, i.e. the Sun will be behind the chaser and will illuminate the docking port side of the target (see figure 5.12). The best illumination conditions of the target docking port side will occur at intermediate angles between orbital noon and sunset, i.e. in a region around three-quarters through the orbital day corresponding to an in-plane component of the Sun angle of 135 deg. Depending on the geometric configuration of chaser and target, suitable illumination conditions may exist in a region of a maximum of $\pm$ 20–30 deg around this Sun angle point, i.e. between about 105 deg and 165 deg, corresponding to a maximum duration of a quarter of an hour. This is the case when the β-angle is zero. The useful time will be reduced with increasing β-angle. Eventually, when the β-angle becomes 90 deg, there will be no suitable illumination conditions over the entire orbital day, as the Sun direction is always orthogonal to the docking axis. To achieve suitable illumination conditions, the β-angle must differ from 90 deg by at least 20 deg.

If at a $-$V-bar approach, for monitoring reasons, a maximum of the final approach is to be performed under Sun illumination conditions, it will be necessary to plan capture towards the end of the orbital day, i.e. when the target is illuminated. As a consequence, for approaches on the $-$V-bar side, the visual target pattern is best mounted on the target side and the video camera on the chaser side. The video signal needs then to be transmitted to the target crew and to ground to enable monitoring.

For an approach to the target from the $+$V-bar side, the illumination conditions occur in the opposite order: at orbital sunrise the Sun will be behind the chaser, illuminating the target, and at orbital sunset behind the target, illuminating the chaser. In order to achieve a maximum of the final approach under Sun illumination, it is in this case better to have the visual target pattern on the chaser and the video camera on the target. For the same reason, direct visual monitoring by a target crew member looking out of a window will experience better conditions when the approach comes from the $+$V-bar side. For this approach direction, the same time constraints will result as for the approach from the $-$V-bar side: i.e. for a zero β-angle a maximum duration of about 40 minutes prior to docking under Sun illumination, out of which the last 15 minutes will provide suitable illumination conditions for the visual target pattern.

For an R-bar approach the illumination conditions are somewhat different. Sunrise will occur on the side of the chaser in the V-bar direction, with the Sun moving from the $+$V-bar- to the $-$V-bar side in the hemisphere above the chaser. Since the Sun is always in the upper hemisphere, there will be no illumination of the target docking port during a $+$R-bar approach. As a consequence, for approaches from the $+$R-bar side, the visual target pattern is best mounted on the chaser and the video camera on the target. For the opposite arrangement of camera and target pattern, an approach from the $-$R-bar side (approach direction toward Earth) will be more suitable (see figure B.8). For a β-angle near zero, i.e. the Sun in the orbital plane, optimal illumination conditions of the chaser docking port will occur in two regions around 45 deg and 135 deg of the elevation (in-plane) Sun angle. The range of suitable illumination conditions around these points will be of the order of $\pm$20–30 deg. If β is between 20 and 70 deg, favourable illumination conditions would also exist at orbital noon, whereas suitable duration would be shortened at the beginning and end. If the β-angle becomes larger than 70 deg, no suitable illumination conditions will be available during the entire orbit.

The illumination conditions for $\pm$V-bar and $+$R-bar approaches are shown schematically in figure 5.17, which indicates the local orbital frame of the chaser, the relative trajectory of the Sun, position and local horizon of the chaser (the orbital altitude is not taken into account) and the relative position of the target for the various approach directions. The approach direction of the chaser is indicated by arrows, and the target docking port for the different approach directions has to be imagined at the point of these arrows.

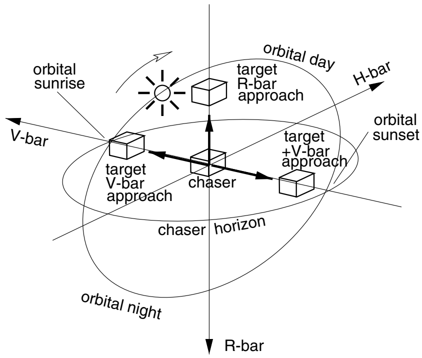

Figure 5.17. Illumination conditions during ±V-bar and R-bar approaches.

Taking all requirements together, it has to be taken into account that, for a given set of orbit parameters of the target in conjunction with the particular set of communication opportunities (ground stations, relay satellites) available for the rendezvous mission in question, proper illumination and communication conditions (see section 5.4.2) for monitoring of the last part of the approach and capture will not be achievable for all days of the year. This will be one of the most important criteria for planning the launch date of the chaser.

5.4.2 Communication windows

Communication from the chaser and target spacecraft to ground can be performed either directly between the antennas of spacecraft and ground or via a relay satellite in geostationary orbit. In the case of direct communication with a ground station, the duration of contact is limited by the altitude of the orbit, by the radiation/reception cone angle of the ground antenna (see figure 5.18) and by the part of the cone which will actually be crossed by the orbit. The resulting time for possible radio contact is called the communication window with the ground station. The useful cone of ground station antennas is

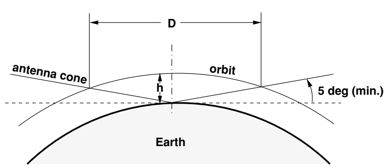

Figure 5.18. Visibility of ground stations, geometrical relation.

constrained by obstacles on ground and by reflections of ground atmosphere and other disturbances, occurring at very low elevation angles. The useful range starts at elevation angles of 5–7 deg, i.e. the maximum half cone angle of the antenna's radiation/reception cone will be less than 85 deg. In a low Earth orbit, under the best conditions, i.e. when flying over the centre of the cone, the maximum communications duration would be for a 400 km orbital altitude of the order of 10 min and for a 300 km orbital altitude about 7.5 min. When the ground track of the orbit crosses the cone at a lateral distance to the centre, the communication window will be accordingly smaller. Also, because of obstacles and disturbances the shape of the communication range may not always be circular.

Owing to the rotation of the Earth, after one orbital revolution the ground track of the next orbit will have moved by approximately $\Delta\lambda$ degrees westward, which is for a 400 km LEO orbit approximately 23 deg:

$$\Delta\lambda = \frac{-T \cdot 360\,\mathrm{deg}}{24\,\mathrm{h}} \tag{5.10}$$

where $\Delta\lambda$ is the change of longitude and T is the orbital period (see figure 5.19).

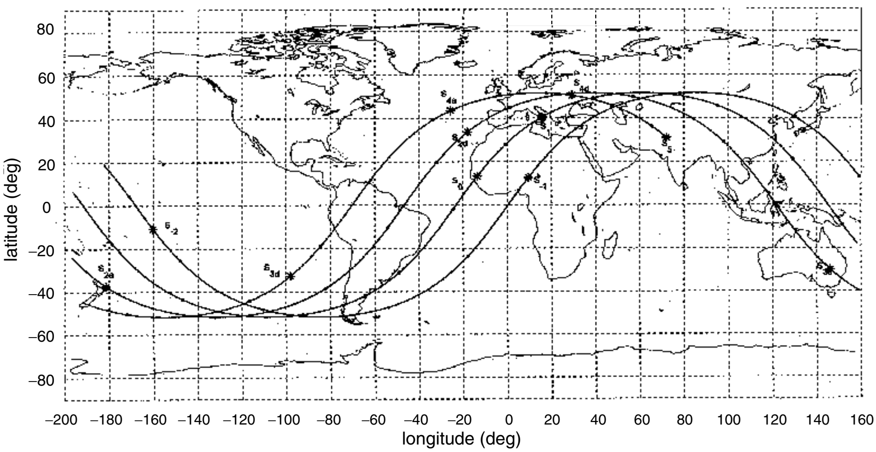

Figure 5.19. Ground track of a 400 km altitude, 51 deg inclination orbit. Dots represent 10 minute intervals.

For exact calculations, the orbital motion of the Earth around the Sun and the drift of the nodes $\dot{\Omega}$ (see section 5.2.1) will have to be taken into account, resulting in $\Delta\lambda = -T(15.041 + \dot{\Omega})$. Measured from the centre of the Earth, the half cone angle of a ground station visibility zone will be no more than 20 deg. Therefore, a communication window with a ground station, contacted in the previous orbit, may in the following one be reduced or non-existent.

In planning recovery operations after contingencies, it is desirable to achieve the same conditions as for the nominal approach. Neglecting the drift of nodes, a ground station passes the orbital plane after each 12 and 24 h, where after 12 h the direction of the orbital velocity vector will be opposite. However, due to the fact that orbital period (e.g.

92.3 min for a 400 km orbit and 90 min for a 300 km orbit) is not synchronised with the rotation of the Earth, the vehicle will not necessarily be directly over that ground station after 12 or 24 h. As a result, the nearest pass over the same ground station may happen somewhat before or after this ground station passes the orbital plane, resulting in a different contact duration.

It is obvious from the above considerations that a single ground station will not be sufficient in order to monitor the close range rendezvous phases. To achieve communication windows of 20 min or more, which would be desirable for monitoring the last part of the final approach including capture, a large number of adjacent ground stations would have to be established. This number would have to be even larger should a second approach and docking attempt be possible in the event of contingencies. Such an arrangement of adjacent ground stations has been established by the former Soviet Union for the rendezvous and docking of the space station programmes Salyut and Mir, and this is still operated by Russia. This unique system comprises seven ground stations, covering the major part of the Eurasian continent (see figure 5.20).

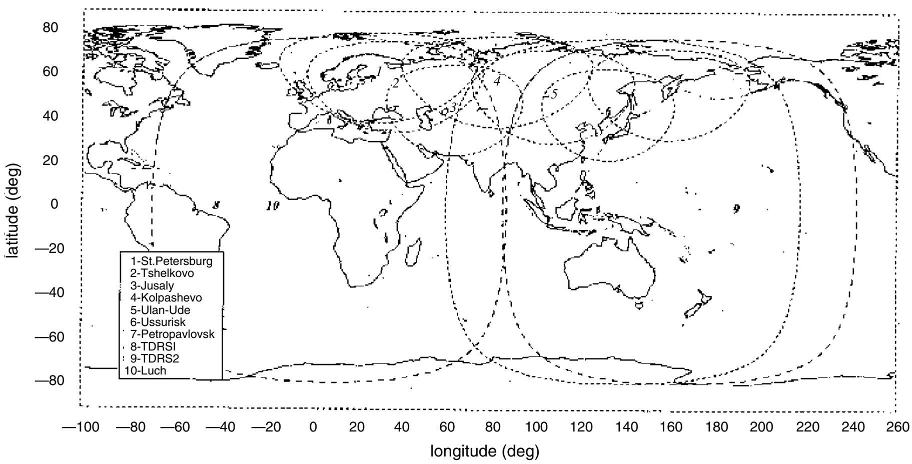

Figure 5.20. Coverage of relay satellites and Russian ground stations for RVD.

Longer communication windows can be achieved via relay satellites in geostationary orbit, which provide approximately half an orbit coverage each. The dashed lines in figure 5.20 show the boundaries of communication of the American TDRS1 and TDRS2, and of the Russian Luch relay satellites. In contrast to the ground stations, where communication is possible inside the closed dashed and dotted line, for relay satellites communication is possible outside the closed lines.

Because of the location of the presently available data relay satellites over the Earth, there will be an area where no communication is possible. In addition, the visibility constraints due to the antenna accommodation on the chaser vehicle must be taken into account. For the data exchange with a satellite in GEO, the antenna of the satellite

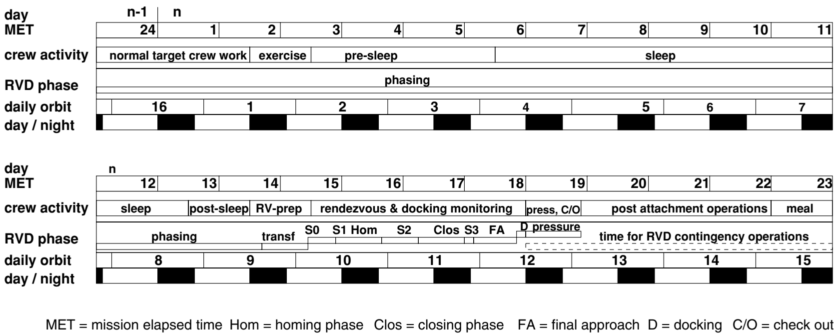

Figure 5.21. Typical 24 h crew activity timeline with RVD operations.

in LEO needs continuously to change its direction in order to point toward the GEO satellite. This requires an articulated antenna mount with two degrees of freedom. The available FOV will be limited by the articulation range of the antenna mount and by the geometry of the satellite. The available communication window via relay satellites will be reduced by these constraints.

The most important problem concerning transmission of monitoring data is, as we have seen, to arrange the approach timeline in such a way that the chaser enters the communication window when the last part of the final approach and capture commences. An additional constraint is that monitoring by video pictures requires a very large data transmission rate (high bandwidth), which may not necessarily be available on the frequencies and channels generally assigned to the mission (see section 9.3). Because of the relatively high cost, high data rate channels will usually be booked for the envisaged monitoring period only.

5.4.3 Crew activities

A large part of the available crew time will be taken up with the supervision of the approach and the preparation of docking/berthing and post-contact activities. As the total number of crew at the target station will be limited, it cannot generally be assumed that shift work around the clock is a realistic option. The time needed for approach/contact monitoring and mating operations, including the time for potential contingency handling, must therefore fit into the work, rest and sleeping schedule of the target crew. A maximum of 10 hours can be assumed for one working session, into which all nominal and contingency operations must fit, i.e. rendezvous and docking/berthing operations up to utility connection, plus potential contingency operations until acquisition of a hold point. Such a hold point must be at a sufficient distance from the target, to be at least one night safe, until crew can again take action.

The target crew will have to monitor all rendezvous phases, which include a potential collision danger. These are all phases, starting from the transfer to the target orbit and including the possibility of reaching the target within very few orbital revolutions. In the examples of figures 5.25 and 5.28, the manoeuvres, beginning with the transfer to the target orbit, take a duration of typically three half orbits. Including hold points, the total time of the approach which has to be monitored by the crew may take in the order of 2.5–3 hours. Adding the preparation time for the rendezvous related activities aboard the target, e.g. for check-out activities and for the time needed for post contact activities, e.g. pressurisation, verification of connections, preparation for hatch opening, the nominal duration of crew involvement in the rendezvous and docking process will be 4–6 hours. Because of the additional tasks for the manipulator operations, the total nominal duration of crew involvement in a berthing mission will be longer. The planning must include sufficient margins for contingency operations.

Although there will be a certain flexibility in the work/rest schedule by perhaps an hour forward or backward, larger shifts to provide synchronisation with the approach timeline and illumination conditions will have to be planned well in advance. Recovery operations after a contingency or a CAM and retreat to a far hold point can be performed at the earliest opportunity during the following working day.

5.4.4 Time-flexible elements in phasing and approach

Synchronisation of the rendezvous timeline with external constraints is, as shown above, a major issue in the design of the approach strategy at mission planning. During the mission, corrective synchronisation starts right after launch, and continues up to the final approach, to compensate for launch margins and manoeuvre tolerances. The trajectory elements and operational methods used to achieve synchronisation have been termed ‘time-flexible elements’. They can be grouped into drifts at different orbital heights and hold points. To a limited extent, the velocity profile of the straight line final approach can be modified to fine tune the arrival time.

Drift at different orbital heights

Three drift cases can be distinguished:

- drift on elliptic orbits – the advance depends on the size of the major axis;
- drift on circular orbits – the advance depends on the orbit radius, i.e. on the orbital height above ground;
- drift on two fixed major axes or orbital heights – variation of time of transfer.

The last method, which allows for adjustments during the mission, is shown in figure 5.22. Since phasing lasts for a number of orbital revolutions, the basic idea is to prepare at all times for two standard orbits of different sizes, either of different radii, in the case of circular orbits, or of different major axes in the case of elliptical orbits. According to synchronisation needs, the total duration of phasing can be decreased by leaving the chaser for a longer time on the lower orbit and it can be increased by transferring the chaser earlier to the higher orbit. The standard orbits can be used for all missions of a certain type of vehicle, so such a strategy would be well suited for an automatic system.

Because of the relatively long time available during phasing and the complexity of the rendezvous phases after approach initiation (‘S1’ in figures 5.25 and 5.28) in terms of manoeuvres and of parties involved, it is usual to perform the synchronisation tasks, as far as possible, during phasing. The chaser arrives then at the approach initiation point at the proper time, leading to the proper illumination and communication conditions at capture, when all subsequent manoeuvre elements are executed nominally. In this case, hold points will be kept as short as possible, and they will be used for functional check-out and last fine tuning of arrival time only.

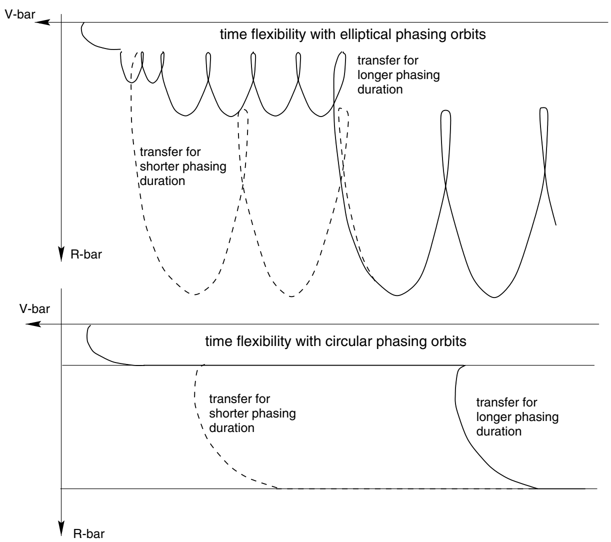

Figure 5.22. Time flexibility during phasing.

Hold points

Three trajectory elements with nominally zero average motion after one orbit exist:

- passive hold points only on V-bar,
- active hold points,
- safety ellipse.

Because of the increasing forces to be applied (see Eqs. (3.55) and (3.58)) with increasing distance from V-bar, this type of trajectory element is feasible only when the vehicle is on V-bar, or at least very near to it. Hold points on V-bar are in principle a very effective time-flexible element, as (theoretically) for their maintenance no forces are required. However, exact positioning on V-bar will not be possible, considering the measurement and manoeuvre errors discussed in section 4.3. In reality, on passive hold points with no position control, the chaser will always drift either toward or away from the target. Drift direction and velocity will be as given in Eq. (3.24), according to its actual z-position below or above V-bar. A further cause of drift will be the effect of differential drag, discussed in section 4.2. Both sources can lead, even after very few orbits, to a significant change of position along V-bar. Because of this drift sensitivity, passive hold points can be used only for relatively short duration and at sufficiently long distances from the target.

The drift can of course be avoided by actively controlling the hold point position; this requires proper sensors and more propellant. The duration of a closed loop controlled hold point is constrained only by the consumption of propellant. However, in the case of a failure which results in a loss of control, it still must be ensured that no drift toward the target which may lead to collision will occur. A possible solution to the drift problem is the location of the hold point at a sufficiently large z-distance for an approach from the $-$V-bar side above the target orbit and for an approach from the $+$V-bar side below it. This z-distance must be chosen such that, for all conditions of differential drag and navigation errors, it is ensured that the chaser will never move toward the target. For a closed loop controlled hold point this will increase the propellant consumption significantly. In the open loop case, one could no longer call it a 'hold point'; a 'slow retro-drift' would be more correct.

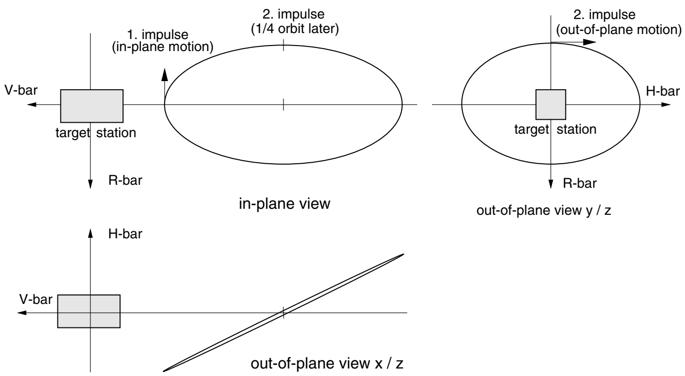

Figure 5.23. Safety ellipse.

Safety ellipse

The safety ellipse shown in figure 5.23 can be applied in cases where active hold point control or the above described 'slow retro-drift' is not possible (for whatever reason) and where a hold point type of time-flexible element is required for operational reasons. It consists of a combined in-plane and out-of-plane elliptic motion, which ensures that even in case of a drift toward the target, there will never be a collision, as the chaser will move around it on a distance given by the in-plane and out-of-plane excursions.

A safety ellipse can be implemented by applying a radial impulse (± R-bar direction) at the starting position on V-bar and an out-of-plane impulse (± H-bar direction) a quarter of an orbit later. It has to be pointed out also that the 'safety ellipse' will be only 'short term safe', as over a larger number of orbits the uncertainty of the differential drag has to be taken into account.

Velocity of straight line approach

The last chance of adjusting the arrival time is theoretically the variation of the approach velocity on the closed loop controlled straight line approach to a docking port or a berthing box. In practice this option will not be used, as the final approach will start only when it is proven that all conditions, including synchronisation, are nominal. Also, for reasons of approach safety, i.e. the assessment of the 'safe state' of the incoming vehicle by human operators in the target and on ground, it is preferred that the final part of the approach is performed with a standard velocity profile.

5.5 Onboard resources and operational reserves

Limitations concerning the availability of onboard resources can have significant repercussions on the approach concept. The most important resources in this respect are:

- electrical power,
- heat rejection capacity,
- propellant.

Limitations in available electrical power can be a reason to change the vehicle's attitude at regular intervals during the approach, e.g. from LVLH to Sun-pointing attitude, in order to obtain maximum output from the solar generators. This may lead to a strategy in which a Sun-pointing attitude will have to be assumed after any impulsive manoeuvre boosts and in the case of hold points. Hold points may be required to have a longer duration, or a larger number of hold points may be included in the approach to re-charge batteries than in cases without power constraints.

Limitations in the heat rejection capabilities may also result in a particular strategy, changing the attitude of a vehicle during approach whenever possible to a direction in which the heat radiating surfaces point toward deep space. Such requirements may have repercussions on the possibility of closed loop control of trajectories or hold points, as trajectory control sensors may not be available when the chaser is, e.g., in Sun-pointing attitude.

Propellant limitations may lead to the avoidance particularly of ΔV consuming trajectory types, such as straight line forced motion trajectories. Also, closed loop control of a trajectory after a boost, as described in section 4.4.1, may have to be excluded because of the significant additional propellant cost.

Mission planning for any rendezvous mission has to take into account the resources necessary for, e.g., a potential waiting time of a specified number of hours prior to start of the final rendezvous, or for the capability to repeat the approach a specified number of times. Such requirements are intended to cover the possibility of failures and delays caused by any party involved, i.e. chaser, target or all other elements of the space and ground segment. Considerations on how to minimise resources for recovery already have an influence on the design of the nominal approach strategy. For example, during close range rendezvous there will be a requirement to be able to retreat in the case of problems to a hold point from any point of the approach trajectory. This may be to the last hold point for those problems that are expected to be solved within same working session, or to a more distant hold point, where a longer period for resolution of the problems is expected. In order to minimise the cost for acquiring such hold points, the trajectory and manoeuvre of the nominal approach must be designed such that a retreat to the previous hold point is possible without excessive boost manoeuvres and within reasonable time. For example, such a requirement can lead, amongst other reasons, to the choice of a two-pulse *radial* boost transfer (see figures 3.20 and 3.21) for a certain part of the nominal approach strategy, as this type of trajectory will return to its starting point when the second boost is not executed. This means that whenever a retreat to the previous hold point is ordered, it can be achieved without extra cost within a time of between a half and one orbital revolution.

5.6 Approach rules defined by the target

Whereas during launch and phasing the approaching vehicle does not have many operational interfaces with the target vehicle and will be controlled by its control centre just as any single spacecraft, from some point during the far range rendezvous onwards, operations of chaser and target vehicles have to be planned and supervised in a coordinated way. As there will be different control teams responsible for the control of chaser and target vehicle, and as these teams may even belong to different organisations and space powers, a hierarchy of control authority and procedures of joint operation have to be established for the rendezvous operations in the proximity of the target (see chapter 9). In cases where one of the two vehicles, chaser or target, is manned and the other one is unmanned, the highest authority will have to be with the manned vehicle. This means that for an automated rendezvous of an unmanned chaser vehicle, the manned target space station, or its control centre on ground, will at a certain distance take over mission control authority for the rest of the rendezvous mission (for a definition of the various types of control authority see section 9.1.2).

In all cases a hand-over of control authority can best be achieved at defined points in the approach strategy, where the actual state of the vehicle w.r.t. the nominal one can be verified, or where there is subsequently sufficient time and opportunity for such verification. Hand-over of control authority does not necessarily include the hand-over of the actual command authority to control the vehicle, but concerns rather the highest level decision making concerning trajectories and operations to be performed (see sections 9.1.1 and 9.1.2). Because of the need for defined interface and verification conditions at hand-over, this may have, amongst other conditions and rules, repercussions on the choice of the type of trajectory element during which hand-over has to take place.

Space stations, with their large and complex structures, can allow approach and departure of visiting vehicles only within certain spherical sectors, originating for docking at the docking ports or, in the case of berthing, at the nominal locations of grapple and release by the manipulator. In both cases, at implementation the origin and centre line of such cones will be determined by sensor and target pattern accommodation on chaser and target. Furthermore, safety of the station requires monitoring of the approaching or departing vehicles by the crew of the station and/or by the ground control centre of the station, in order to ensure that the trajectory and the other state vector parameters do not deviate from the planned ones. To establish references for control, volumes around the station can be defined in which visiting vehicles are subjected to certain rules (NASA 1994). Within such control zones, the hierarchy of control authority of the parties involved may be defined, maximum ΔVs allowed, operational procedures determined, approach and departure corridors defined, etc. Drivers and constraints for the definition of control zones can, for example, be

- the range where direct communication between the station and the visiting vehicle is available;
- the range from which the station could be reached within one orbital revolution, with a ΔV exceeding some defined small value;
- the range within which the visiting vehicle must fly inside approach and departure corridors in order to protect the structure of the station sufficiently against collision in the case of failures of the visiting vehicle;
- the range within which monitoring by video camera is possible, etc.

There are of course no exact physical laws to establish the size of such control zones, and their definition will, therefore, always be somewhat arbitrary. The purpose will rather be the definition of easily understandable and easily controllable boundaries at which the trajectory and state of the visiting vehicle can be checked against fixed criteria. The order of magnitude of potential effects will roughly determine the range of such boundaries. For instance, the boundary of the zone in which the visiting vehicle must move inside a defined corridor will be related to the size of the target vehicle. To be comfortable, the length of the corridor should be roughly one order of magnitude larger than the geometric extensions of the target station in that direction. If, e.g., the extension of the station in the x-direction is of the order of 20 m from the CoM, the length of the corridor can be envisaged to be 200 m. The definition of a control zone, in which execution of manoeuvres of a visiting vehicle will have to be agreed by the target, will depend on the typical size of such manoeuvres. If the ΔVs used in the rendezvous phase are, e.g., up to 0.1 m/s, a related control zone must be larger than 1800 m in $\pm$ V-bar direction (see Eqs. (3.28)). Drivers and constraints for the definition of the diameter of the approach and departure corridors include the following.

- *Observability.* In addition to the onboard data transmitted by the visiting vehicle, ground controllers and crew must be able to judge quickly upon the safety of the trajectory by means of direct vision or video. The nominal trajectory must be within a defined region of the field of view.
- *Thermal loads and contamination* by the thruster plumes of the approaching or departing vehicle on the surfaces of the target station must be limited.
- *Safety margins w.r.t. collision* around the geometry of the target station are necessary to prevent immediate impact in the case of failure of the visiting vehicle functions. This is a protection on the target side, which is independent of the failure tolerance requirements to be fulfilled on the side of the visiting vehicle (see section 4.1.1).

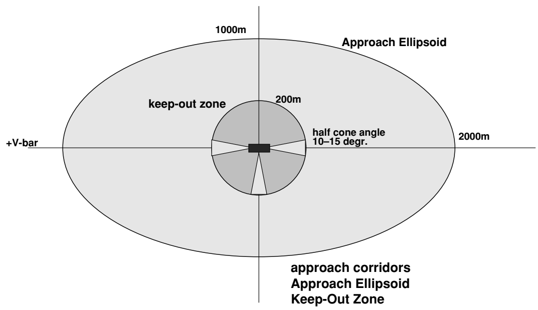

Figure 5.24. Control zones of the ISS.

Approach and departure corridors, however, must not be confused with the trajectory safety boundaries discussed in section tolerance requirements w.r.t. the target station, whereas the approach and departure corridors permit the target station to ensure observability and to limit thermal loads and contamination for the nominal approach and departure cases. Nominal approach and departure trajectories, including the safety boundaries, must be inside these corridors. Also, the corridors will also have to include some margin for the initiation of contingency operations by the visiting vehicle in the case of failure. In addition, margins must be taken into account for the attitude motion of the station itself. Depending on the range and the uncertainty of the attitude motion of the target, half cone angles of approach corridors (defined for the CoM of the approaching vehicle) may be between $\pm$ 5 deg and $\pm$ 15 deg.

For trajectories without thruster firings, e.g. after inhibition of thrust, after the nominal departure boost, after a boost for a contingency departure, or after a CAM, wider corridors may be defined. For this type of corridors, only protection against collision is the issue, rather than observability by cameras and protection against thruster plumes. It has to be noted that these wider approach and departure corridors will have to be defined not for the motion of the CoM but for that of the envelope of the vehicle, i.e. no part of the geometry of the visiting vehicle must exceed the corridor.

As an example, the control zones and approach/departure corridors of the ISS are shown in figure 5.24. The outer zone, called the 'Approach Ellipsoid' (AE), has an extension of $\pm$ 2000 m in the x-direction and $\pm$ 1000 m in the other directions. Prior to entering the AE, overall control authority will be taken over by the ISS Control Centre. The inner control zone, the so-called 'Keep-Out Zone', is a sphere of 200 m radius, which can be entered only through one of the approach and departure corridors, which are available for $\pm$ V-bar and $+$R-bar approaches. If they are defined w.r.t. the docking port axes or other body features of the station, the precise position and direction of these corridors will depend not only on the exact location of the docking ports and berthing boxes, but also on the main attitude of the station at the time of approach or departure.

5.7 Examples of approach strategies

In order to show the repercussions of various constraints on the choice of the trajectory elements used in the approach sequence, three examples of approach strategies will be discussed in this section. It is not the intention to analyse existing approach strategies of American and Russian vehicles, since these strategies have evolved over a very long time; some of the considerations leading to their design may have been driven by historical situations and are generally not known to the author. A brief description of the approach schemes of the US Space Shuttle and of the Russian vehicles Soyuz and Progress can be found in appendix B.

The three examples chosen are characterised by different directions of the final approach axis, by different sensor sets and by different mating methods, i.e. docking or berthing. These are as follows.

(1) Approach to a docking port on $-$V-bar; space station scenario with control zone setup; RGPS and optical laser scanner as rendezvous sensors.
(2) Approach to a berthing box on R-bar; space station scenario with control zone setup; RGPS and optical laser scanner as rendezvous sensors.
(3) Approach to a docking port on the $+$V-bar side with the docking axis skewed under an angle β w.r.t. the orbital velocity vector; radar and optical camera sensor as rendezvous sensors; no control zone constraints.

For all three examples a 400 km altitude quasi-circular orbit has been assumed. As launch and phasing will not be affected by the constraints for the relative navigation phases of a rendezvous mission, the discussion will start at the transition from phasing to the far range rendezvous.

5.7.1 Approach strategy, example 1

Strategy overview

This example of an approach to a $-$V-bar docking port in the ISS scenario is illustrated in figure 5.25. It is similar to the approach strategy of the European ATV (Cornier *et al.*

1999; Fabrega, Frezet & Gonnaud 1996) to the ISS. It is not the intention, however, to discuss here the actual approach strategy of the ATV project. The purpose of this example is rather to address the typical considerations for the design of a −V-bar approach concept to a docking port.

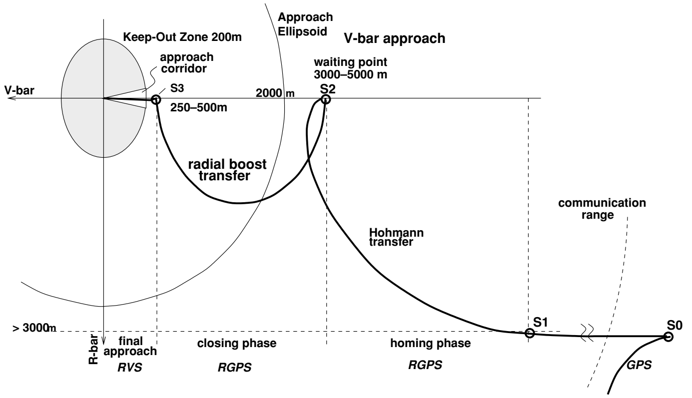

Figure 5.25. Approach strategy to −V-bar docking port (example 1).

The trajectory strategy consists of the following:

(1) *A free drift.* After the last phasing manoeuvre the chaser moves on a quasi-circular, 3000–5000 m lower orbit, parallel to V-bar. During the drift on this orbit, acquisition of the communication link to the target takes place and the navigation filter for RGPS converges.

(2) *A Hohmann transfer to the target orbit under RGPS navigation.* The transfer trajectory with all possible dispersions must not enter the approach ellipsoid of the ISS.

(3) *A hold point on the target orbit outside the Approach Ellipsoid.* At this point, last check-outs of the chaser system and last synchronisation corrections with external events, such as lighting conditions, crew schedule, etc., can take place. Further approach into the AE can commence only after permission from the ISS.

(4) *A radial boost transfer manoeuvre into the AE under RGPS navigation.* This transfer leads to a point where acquisition of the target reflectors by the optical rendezvous sensor can take place.

(5) *A forced motion straight line approach on, or parallel to, V-bar under optical RVS navigation.* This trajectory leads to the docking port. There may be optional stops on the straight line approach, e.g. to acquire a new navigation mode with relative attitude measurements.

The nominal attitude during the entire rendezvous approach is assumed for this example to be LVLH, i.e. aligned with the local orbital frame $\boldsymbol{F}_{\mathrm{lo}}$ of the chaser.

Acquisition of the target orbit

Location of the final aim point for phasing The rendezvous scenario starts at a point, identified in the figures as S0, which is the final point of the last phasing manoeuvre (initial aim point) and which is located below and behind the target. The location of this point will be defined by a number of desirable conditions on one side and constraints on the other side. The desirable conditions for initiation of far range rendezvous at S0 are as follows.

- Relative navigation should start at a distance behind the target, which is large enough to have sufficient time and range available for the manoeuvres necessary to reduce step-by-step position errors and velocities to those required for docking.
- The chaser orbit after the final phasing manoeuvre should be as close as possible to the altitude of the target orbit, to reduce the difference in orbital rate between the two vehicles and to create sufficient time for the above-mentioned manoeuvres.

The navigation sensor prior to S0 is, in this approach strategy example, assumed to be absolute GPS. RGPS will be acquired only during the drift between S0 and S1, after the chaser has entered the communication range with the target. The communication range is assumed to be 30 km for the purpose of this example. Navigation accuracy for absolute GPS is assumed to be of the order of 100 m and 0.1 m/s (i.e. the accuracy before abandonment of 'selective availability' for GPS; see section 7.3.2). Navigation errors can be cumulative, i.e. the existing position error at the start of the manoeuvre and the additional error after the second boost have to be taken into account. The initial velocity measurement error translates after the manoeuvre into a position error. Thrust errors of the last phasing manoeuvre depend on the type of thruster used and on the size of boost applied. The errors are assumed to be of the order of 0.2 m/s. The knowledge of the target position is assumed to be within 100 m and the evolution of the position error over one orbit is known to be not more than 50 m. The extension in the z-direction of the Approach Ellipsoid of the ISS is 1000 m (see figure 5.24). Summing up, the contributions to be taken into account for the definition of the altitude of S0 are:

- the required altitude difference due to rules for approach safety, set by the target, e.g. the Approach Ellipsoid in the case of the ISS (Δz_{AE} = 1000 m);
- the altitude uncertainties $\Delta z_{\epsilon\mathrm{targ}}$ due to the knowledge of the instantaneous position and due to the forecast accuracy for the evolution of the orbit of the target in the time between the manoeuvre boosts ($\Delta z_{\epsilon\mathrm{targ}} = 150$–200 m);
- the altitude errors $\Delta z_{\epsilon\mathrm{nav}}$ due to the navigation sensor accuracy, which is available to acquire S0 (due to position measurement error, $\Delta z_{\epsilon\mathrm{nav}} = 100$–200 m; due to velocity measurement error, $\Delta z_{\epsilon\mathrm{nav}} = 350$–450 m);
- the altitude errors $\Delta z_{\epsilon\mathrm{thr}}$ due to the thrust errors of the manoeuvres leading to S0 ($\Delta z_{\epsilon\mathrm{thr}} = 700$–800 m).

Additionally, a margin $\Delta z_{\mathrm{margin}}$ of 250–500 m has to be added for the definition of the nominal altitude of S0. This margin will have to cover such effects as the change of relative orbital height between the vehicles due to differential drag during the drift phase after S0. The nominal relative altitude of the aim point w.r.t. the target orbit can then be calculated to be

$$\Delta z_{\mathrm{S0}} = \Delta z_{\mathrm{AE}} + \Delta z_{\epsilon\mathrm{targ}} + \Delta z_{\epsilon\mathrm{nav}} + \Delta z_{\epsilon\mathrm{thr}} + \Delta z_{\mathrm{margin}}$$

Adding up all errors and margins, the aim point for the final phasing manoeuvre will have to be of the order of 3000–5000 m below the nominal target orbit.

The contributions which have to be taken into account for definition of the location of S0 in orbit direction are as follows.

- The required distance Δx_{AE} due to the rules for approach safety set by the target, e.g. the Approach Ellipsoid in the case of the ISS ($\Delta x_{\mathrm{AE}} = 2000$ m).
- The position uncertainties $\Delta x_{\epsilon\mathrm{targ}}$ due to the knowledge of the instantaneous position and the forecast accuracy for the evolution of the orbit of the target between the manoeuvre boosts ($\Delta x_{\epsilon\mathrm{targ}} = 150$–$200$ m).
- Position errors $\Delta x_{\epsilon\mathrm{thr}}$ due to the thrust errors of the manoeuvres leading to S0 ($\Delta x_{\epsilon\mathrm{thr}} = 3400$–$3600$ m).
- Position errors $\Delta x_{\epsilon\mathrm{nav}}$ due to the navigation sensor accuracy available to acquire S0 (position measurement error, $\Delta x_{\epsilon\mathrm{nav}} = 100$–$200$ m, velocity measurement error, $\Delta x_{\epsilon\mathrm{nav}} \approx 3500$ m).
- The distance in orbit direction Δx_{Hoh} required for the Hohmann transfer from the S0 altitude to the target orbit, which is $\Delta x_{\mathrm{Hoh}} = \frac{3\pi}{4}\Delta z$ (see Eqs. (3.31)). With a Δz at S0 of 3000–5000 m, the x-distance for the Hohmann transfer becomes $\Delta x_{\mathrm{Hoh}} = 7000$–$12000$ m.
- A free drift distance $\Delta x_{\mathrm{drift}}$ between S0 and the starting point of the transfer manoeuvre, S1, equivalent to the time required for the preparation of the transfer manoeuvre. This may require manoeuvre confirmation by ground. Since the advance on a lower orbit per orbital revolution according to Eqs. (3.25) is $\Delta x = 3\pi\Delta z$, a total time for preparation of the manoeuvre and validation by ground of, e.g., 3 minutes, would result in an advance of the order of $\Delta x_{\mathrm{prep}} = 900$ m for a Δz of 3000 m and of $\Delta x_{\mathrm{prep}} = 1500$ m for a Δz of 5000 m.

  As the range of the communication link between chaser and target is assumed not large enough for RGPS navigation to be acquired prior to arrival at S0, a drift distance equivalent to the time required for convergence of the RGPS navigation filter (see chapters 6 and 7) needs to be added. With a time for filter convergence of 10 minutes, the advance in orbit direction would approach $\Delta x_{\mathrm{converg}} = 3000$ m for a Δz of 3000 m and of $\Delta x_{\mathrm{converg}} = 5000$ m for a Δz of 5000 m. The minimum required drift range is then $\Delta x_{\mathrm{drift}} = \Delta x_{\mathrm{prep}} + \Delta x_{\mathrm{converg}}$, which would amount to 3900 m for a 3000 m altitude difference and to 6500 m for a 5000 m altitude difference.

A margin must be added to these contributions, which takes into account further uncertainties and ensures that the hold point S2 on the target orbit will be outside the

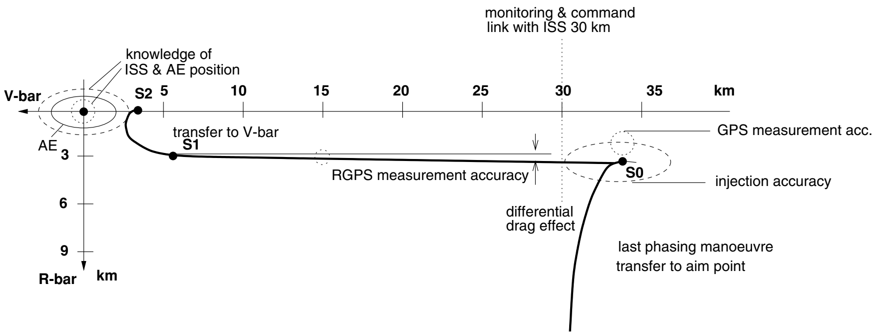

Figure 5.26. Acquisition of the target orbit (example 1).

Approach Ellipsoid. This margin should be at least of the order of 500–1000 m. The minimum relative distance in the orbit direction of the aim point w.r.t. the target position can then be calculated to be

$$\Delta x_{\mathrm{S0}} = \Delta x_{\mathrm{AE}} + \Delta x_{\epsilon\mathrm{targ}} + \Delta x_{\epsilon\mathrm{nav}} + \Delta x_{\epsilon\mathrm{thr}} + \Delta x_{\mathrm{Hoh}} + \Delta x_{\mathrm{drift}} + \Delta x_{\mathrm{margin}}$$

Adding up all worst case errors and margins, the minimum x-distance of the aim point for the final phasing manoeuvre would have to be of the order of 28 000 m behind the nominal target position. However, as S0 cannot be placed sufficiently inside the communication range of 30 000 m to prepare the second transfer manoeuvre under RGPS navigation, it will be better to place S0 such that its nominal position plus all possible dispersions are outside the communication range. This will make the approach more robust, as it ensures that sufficient time for filter convergence, manoeuvre preparation S1–S2, and validation by ground is available, and that short time communication interruptions can be covered.

Transfer to the target orbit From the above considerations it can be deduced that the hold point S2, i.e. the position after the Hohmann transfer to the target orbit, should, under no circumstances, be closer than 500 m to the boundary of the Approach Ellipsoid. For the definition of the nominal position and the dispersions of S2, the location of the starting point of the Hohmann transfer, S1, needs to be identified first. Three different strategies for the location of S1 can be considered.

(1) Start the Hohmann transfer at a fixed time after acquisition of the communication link between chaser and target, taking into account the time required for RGPS filter convergence plus a margin. This strategy results in a wide range of S2 locations, leading to a large variation of the ΔV required for the subsequent manoeuvre. The rationale for this strategy would be to obtain a more or less fixed duration of the drift phase between S0 and S1.

(2) Calculate the starting point S1 of the Hohmann transfer according to the altitude difference between drift orbit and target orbit such that the final point S2 will

always have the same nominal distance from the target. The rationale for this strategy would be to enable a standard manoeuvre strategy after arrival at S2.

(3) At a fixed time after acquisition of the communication link between chaser and target, start a two-pulse transfer with tangential and radial thrust components (Lambert targeting), aiming at a nominal position of S2. The rationale for this strategy would be to combine the rationales of (1) and (2).

Because of the necessary effort of verification and validation of manoeuvres and trajectories in a proximity operations scenario, and in particular within a multiple control authority environment, such as the ISS scenario, the second strategy is considered preferable, as for the transfer between S1 and S2 the verification and validation of tangential boost manoeuvres will be easier and more credible. Also, monitoring by operators on ground and in the target station will be easier with a nominal trajectory, which ends at a fixed nominal position and which does not cross V-bar. The variation in duration of the drift between S0 and S1 in the second strategy can easily be compensated for by a variable stay time at the hold point S2. For the approach after S2, in both cases (2) and (3) only one single manoeuvre plan has to be verified, which can be done prior to launch. The validation of the actual manoeuvres prior to execution will then be much easier, as it is then only necessary to check that the dispersions are within the tolerable margins. It must be kept in mind, however, that even with a standard manoeuvre plan the individual manoeuvres still have to be calculated and will vary due to the dispersions of the previous manoeuvre. These dispersions need to be corrected in the following manoeuvre, to remain as close as possible to the nominal trajectory plan.

Taking into account all the above considerations, the nominal position of the hold point S2 on the target orbit has been chosen, for the purpose of this example, to be 3000 m behind the target position. The start of the Hohmann transfer, i.e. the manoeuvre point S1, would then be for a 3000 m altitude difference of the drift orbit at $x_{\mathrm{S1}} = -10\,070$ m and for a 5000 m altitude difference at $x_{\mathrm{S1}} = -14\,780$ m.

Correction of out-of-plane errors Residual out-of-plane errors will be corrected successively during the last phasing manoeuvre and during the Hohmann transfer between S1 and S2. Such corrections can be executed either at a node crossing, or by manoeuvres separated by one quarter of an orbit, or by continuous corrections. The latter possibility would typically be used for closed loop controlled trajectory transfers.

Trajectory safety S0–S2 The final aim point for phasing, S0, has been placed, as discussed above, sufficiently below the approach ellipsoid such that the subsequent drift trajectory will never enter the AE, if no further manoeuvre is executed.

The passive trajectory safety features of a Hohmann transfer have been discussed already in section 4.4.2. If the second manoeuvre at S2 cannot be executed, the trajectory would loop below the AE and would, according to Eq. (3.29), return to the target orbit altitude in front of the target. This will be at a distance of about 4070 m, not taking into account differential drag, for a transfer which had started from 3000 m below and about 8800 m behind the target. This distance of about 4000 m in front of the target for the return of the trajectory leaves sufficient margin for dispersions and drag effects. The figures indicate also that the nominal position of S2 should not be further than 5000 m behind the target position for this altitude difference. For partially executed manoeuvres, the possibility of collision cannot be excluded, and therefore a CAM must be available, the magnitude of which at least cancels the nominal boosts at S1 or S2. As a result, the ΔV of a CAM during a Hohmann transfer from, e.g., a 5000 m lower drift orbit must be at least 1.5 m/s.

Further, it has been assumed for this strategy example that the trajectory of the Hohmann transfer is closed loop controlled, as described in section 4.4.1, which adds active trajectory protection. The additional propellant expenditure for closed loop control is quite substantial when compared with the theoretical expenditure for a pure impulsive manoeuvre. As all ΔV cost for the rendezvous phase is, however, small compared with the ΔV required for phasing, the extra costs for closed loop control are considered worth the gain in trajectory safety and accuracy.

Approach recovery For all mission interruptions between the manoeuvre points S0 and S2, whether resulting from a cease of propulsion at S1 or S2, or from a CAM, the chaser will end up on a drift trajectory and will be a couple of kilometres in front of the target after one orbital revolution. As the communication link with the target may no longer exist, the first part of the recovery manoeuvres may have to be performed under absolute GPS navigation accuracy (for the purpose of this example, in the order of 100 m, the accuracy with SA). The recovery strategy would then be as follows:

(1) Acquire V-bar to reduce the relative motion between chaser and target as much as possible. Because of the assumed relatively low navigation accuracy of absolute GPS, V-bar acquisition will have to take place at a sufficiently large distance in front of the target ($\geq$10 km).

(2) Tangential boost transfer to a position at the $-$V-bar side at large distance ($>$50 km). To save propellant, this can be done in several loops, provided that the trajectory never enters the AE.

(3) From the position on $-$V-bar, further transfer to a 3000–5000 m lower drift orbit, such that on entering the communication range (30 km) the automatic approach can be re-acquired.

Acquisition of the final approach corridor (V-bar approach)

This phase is also called ‘closing’. Its task is to transfer the chaser vehicle from the hold point S2, outside the AE, to a point S3, which is inside the AE but outside the ‘Keep-Out Zone’ (KOZ). S3 is the point from which the final approach to docking can commence. The two major issues which determine the strategy for this phase are:

- the best location of the target point for closing, S3,
- the type of trajectory to be used for the transfer from S2 to S3.

The location of S3 will be determined essentially by four features:

(1) the radius of the KOZ;
(2) the maximum operational range of the rendezvous sensor, used for the final approach;
(3) the worst case position dispersion at S3 in case of loss of control after S2;
(4) the minimum useful range of RGPS, which is the trajectory sensor used during closing.

As already indicated in section 5.3.2, the lower boundary of the useful range of GPS and RGPS is given by multi-path and shadowing effects. The geometric extensions of the ISS are of the order of 100 m and include very large rotating solar arrays. Shadowing and multi-path effects on the GPS antenna of the chaser will become more pronounced when the distance is close to a few times the extension of the target. Since the visibility of GPS satellites by the ISS GPS antenna will also be constrained by the various structural elements of the station (see figures C.1 and C.2 for ISS configuration), the GPS receiver on the ISS will not be able to track all GPS satellites on the hemisphere. As a result, the number of common GPS satellites which can be tracked by both chaser and target receivers will become more and more reduced the closer the chaser gets. There will be a danger that the RGPS measurement function will be interrupted for longer than tolerable periods. In the ISS scenario it has, therefore, been concluded that GPS cannot safely be used below a range of 300 m. The maximum range of a laser range-finder, the sensor for the final approach, is a function of the emitting power of the laser beam (see section 7.4.1). Because of the limited power available on a spacecraft, sensors are designed to cover, along with the necessary margin, just the operational range needed. Laser range-finder sensors on the market or under development have, therefore, a maximum range of typically 500–1000 m. From these considerations, it can be concluded that, for the conditions of this strategy example, manoeuvre point S3 should be located between 300 m and 500 m from the target.

When selecting of the type of trajectory to be used for closing, the following features need to be considered:

- passive trajectory safety (no collision with ISS in the cases of missed or partial boosts);
- open loop dispersion of S3 (no penetration of KOZ in the case of loss of control after the first boost);
- ease of approach recovery (in the case of missed or partial boost or loss of control, trajectory should stay on −V-bar side);
- propellant expenditure (for comparison between trajectory types);
- transfer duration (short transfer times are preferable, because of the overall operational constraint addressed in section 5.4.3).

Concerning the type of trajectory, three options for closing are considered here:

(1) tangential boost transfer along V-bar,
(2) radial boost transfer along V-bar,
(3) straight line forced motion on V-bar.

These three types of trajectories are shown as solid lines in figure 5.27. Their follow-on drift trajectories after cease of thrust, indicating passive trajectory features, are shown as dotted lines.

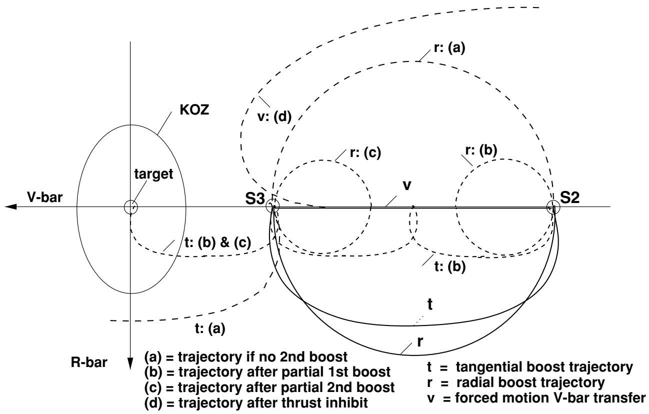

Figure 5.27. Passive safety of closing trajectories (example 1).

Tangential boost transfer The tangential boost transfer has the lowest propellant consumption of all trajectory types: the reference value for a theoretical impulsive transfer from 3000 m to 300 m is 0.33 m/s for the tangential transfer, whereas it is 4.7 times ($\frac{3\pi}{2}$) higher for the radial transfer. The transfer duration is one revolution, which may still be acceptable, but is twice as long as for the radial boost transfer. In the case of a missed burn in S3, the trajectory would loop forward under the target and reach the target orbit again at $\approx$2700 m in front of the target. However, passive trajectory safety requirements are not fulfilled in the case of partial burns, where collision with the target cannot be excluded (see section 4.4.2 and figure 5.27).

Sensitivity to thrust errors in terms of absolute values is higher than with the radial transfer, e.g. a thrust error of 0.01 m/s would result, for tangential transfer, in an x-position error of about 170 m, whereas for radial transfer there is an error of about 36 m. Approach recovery would have to start from the +V-bar side, in the same way as for the previous phase, whereas for radial transfer approach recovery it can commence in practically all cases (except for thruster open failures) from S2, i.e. from the conditions given in the mission interruption case (see below).

Straight line forced motion transfer The major disadvantage of the straight line forced motion approach for closing is the high propellant consumption. For transfer from 3000 m to 300 m the reference value for a theoretical transfer as defined in figure

3.24 would be, for a transfer duration of one orbital revolution, a factor of nearly 22 higher, and for one half revolution, 44 times higher than that for the tangential transfer. The straight line forced motion is passively safe. The type of trajectory, resulting after thrust inhibit, is shown in figures 3.13, 4.14 and 5.27. As the trajectory moves backward and returns to the target orbital altitude on the −V-bar side, approach recovery is easier than for tangential transfer, which, in the case of thrust inhibit, continues to move forward, always ending up at the +V-bar side. Trajectory recovery would not be as straightforward, however, as for radial transfer.

Choice of trajectory type Considering all criteria, conditions and features assumed for this strategy example, the radial boost transfer comes out as the best choice for the closing phase.

Trajectory safety of chosen strategy S2–S3 The basic features of passive trajectory safety of the radial boost transfer are discussed in section 4.4.2 and illustrated in figure 4.13. For completeness of the picture, these features are repeated here.

- If control ceases during the arc S2–S3, or second boost cannot be executed, return after one orbit to the starting point S2.
- If the first boost can be executed only partially, a loop of smaller size through S2 will commence.
- If the second boost can be executed only partially, a loop of smaller size through S3 will commence.

To compensate for the effects of differential drag in an open loop trajectory, a small tangential component could be added (its magnitude depending on the ratio of the ballistic coefficients), which ensures that the centre of the trajectory loop moves slowly away from the target. Since there will always be significant uncertainties concerning the actual value of differential drag, worst case assumptions have to be applied to achieve trajectory safety. As for the Hohmann transfer it has been assumed for the purpose of this strategy example that the trajectory will be closed loop controlled w.r.t. a nominal trajectory. This nominal trajectory will be calculated before the manoeuvre, taking the above-mentioned compensatory tangential component into account. If closed loop control ceases, the chaser remains on a short term safe trajectory.

Approach recovery Except for the case of a thruster-open failure and a subsequent CAM, approach recovery is, for this part of the approach, straightforward, as the free drift trajectories after thrust inhibit or partial boosts remain within the loop (plus some dispersions) described by the nominal S2–S3 arc and its corresponding return trajectory. Approach recovery can, therefore, be achieved in these cases by return to S2, and by application under ground control of a stop pulse to achieve a hold point. From S2 the automatic approach can be resumed. In the case of a CAM, the approach recovery procedure is as described for the Hohmann transfer.

Acquisition of the docking axis

The over-riding requirement for the last approach phase is that the trajectory, including all dispersions, has to be inside the approach corridor defined by the target station (discussed in section 5.6). The trajectory type of the last rendezvous phase is a straight line forced motion V-bar approach of the type described in section 3.3.2, which has been chosen for the following reasons.

- The setup of a narrow approach corridor of $\pm$ 10 deg (see figure 5.24) excludes any other trajectory choice for the last 200 m, if the approach time shall not become excessively long. Radial and tangential boost transfers would require a number of smaller loops of one or one half orbital revolution each to stay inside the corridor.
- The engagement of the docking interfaces of the two vehicles requires for the last few metres a straight approach line anyway. As the docking port may be located at a z-distance from V-bar (see figure 5.1), the approach line will not necessarily be exactly on the target orbit (V-bar) but at some distance from it (see figure 5.7).
- Monitoring of the final approach trajectory is much easier when this trajectory is a straight line.

The velocity profile along the approach line will consist of an acceleration phase at S3, a constant velocity phase and a deceleration phase when approaching the docking port. Velocity profiles are addressed in section 6.2.2. The deceleration phase ends when the final docking velocity is reached. It is followed by a constant velocity phase for the last few metres until contact with the target interfaces. The problem of acquisition and control of the instantaneous docking axis has been addressed already in section 2.4.2 and will be addressed again in section 6.2.3 (concerning the controller requirements) and in section 8.3.6 (concerning the required reception range of the capture mechanism).

If docking is to be performed under Sun illumination, the starting time w.r.t. orbital day and night at S3 and the approach velocity have to be chosen such that for the last few tens of metres optimal illumination conditions for optical monitoring (discussed in section 5.4.1) will be achieved. Another option would be to perform the final approach during orbital night and to illuminate the relevant docking and monitoring interfaces artificially. This would require, however, sufficient power resources and illumination equipment (floodlight, stroboscopic lamps, etc.) on at least one of the spacecraft.

Trajectory safety S3–contact Due to the trajectory rules of the Keep-Out Zone, which exclude any trajectory outside the approach corridor, in this phase passively safe trajectories as illustrated in figure 4.13 cannot be considered as a practical solution for approach safety. As a result of the geometric extension of the target in the z-direction, such safe trajectories would exist, in any case, only for large distances of 200 m or more.

The straight line approach on, or near, V-bar has, however, one very advantageous feature w.r.t. approach safety, i.e. that it can be stopped and held at any point, and that a hold is not very expensive in terms of propellant consumption. This can be seen as an additional safety feature for those types of failures which are not related to the control and propulsion systems of the chaser, but would not allow continuation of the approach. If either a serious malfunction of a GNC function or of the propulsion system occurs during this phase, a CAM is again the sole safety means available.

Approach recovery The strategy for approach recovery after a CAM is identical to that of the previous phases. Approach recovery after a hold on V-bar is trivial, as only an acceleration to the nominal approach velocity at the particular distance is needed, from where switch-over to the nominal automatic approach is possible.

5.7.2 Approach strategy, example 2

Strategy overview

The second example is an R-bar approach to a berthing box in the ISS scenario. The example is, to a certain extent, similar to the approach strategy of the Japanese HTV (Kawasaki *et al*. 2000). Again, it is not intended to discuss here the approach strategy of an actual space project, but rather to address the typical constraints and considerations for the choice of trajectories. The trajectory sequence is shown in figure 5.28.The strategy for the acquisition of the target orbit is, in principle, the same as for example 1. It has been assumed, however, that in this case the last phasing manoeuvre to acquire manoeuvre point S0 can be performed inside the communication range with the target station. As a result, this last phasing manoeuvre can be performed with RGPS navigation accuracy, and the aim point S0 can be located more accurately at an altitude range of 2500–3000 m below the target orbit. The trajectory elements of the approach strategy after S0 in this second example consist of:

(1) a free drift from S0 until the first Hohmann boost at S1;
(2) a Hohmann transfer from S1 to a position S2 on the target orbit outside the AE;
(3) a hold point at S2 for approach synchronisation and check-out;
(4) a Hohmann transfer S2–S3 to a drift orbit 500 m below the target orbit;
(5) a drift trajectory toward the target at an altitude of 500 m below V-bar;
(6) a stop boost at S4 to cancel the drift velocity; S4 is located at, or close to, the x-position of the berthing box;
(7) a straight line closed loop controlled R-bar approach to the berthing box, which is assumed to be 15 m below V-bar;
(8) a closed loop controlled position keeping in the berthing box until the manipulator is ready for grappling.

Acquisition of the target orbit

As mentioned above, the scenario until S2 is very similar to the first example, except for the fact that the altitude difference between S0 and the target orbit is reduced due to the

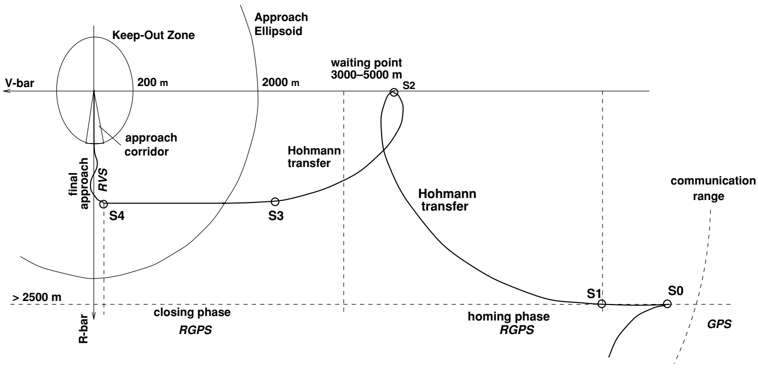

Figure 5.28. Approach strategy to berthing box on R-bar.

better navigation performance. The immediate consequence of this assumption is that the x-distance of S0 can be significantly reduced, because (a) there is no time needed for the RGPS filter convergence (this is assumed to have taken place prior to S0) and (b) the Δx required for the Hohmann transfer is smaller for the reduced altitude difference. The contributions which have to be taken into account for the definition of the altitude of S0 are, for this type of strategy:

- the extension in the x-direction of the AE (2000 m);
- the position errors due to the thrust errors of the manoeuvre leading to S0, when only the second boost can be performed under RGPS navigation (estimated 1500–3000 m);
- the distance on the drift orbit, due to the time required for the preparation of the manoeuvre S1–S2 (3 min drift at 3000 m altitude difference ≈900 m is assumed);
- the Δx required for the Hohmann transfer to S2 (7000 m for an altitude difference of 3000 m);
- the distance required between S2 and the AE boundary (this distance will depend on the safety features of the manoeuvres to be performed for the strategy of the subsequent approach phase – see below – and will be between 500 and 3000 m);
- a margin (500–1000 m) to cover boost errors, disturbances, etc.

For the initial aim point S0 of the last phasing manoeuvre, this adds up to a total Δx of 15 000–17 000 m behind the target position. This value also includes navigation errors, which are, however, comparatively small when using RGPS. Trajectory safety and approach recovery considerations for this approach phase from S0 until S2 are the same as in the first example.

Alternative strategies The question is, of course, why a transfer to the target orbit and a hold point on the target orbit at S2 would be necessary at all, since the approach to the berthing box along a line parallel to R-bar needs to start for a lower orbit altitude anyway. Alternative strategies would be:

- a straight line forced motion R-bar approach from a 3000 m lower position to the berthing box;
- a tangential transfer from the 3000 m lower drift orbit to a point S2 on an orbit 300–500 m below the berthing box. This is followed by a drift to point S4 (in figure 5.28) from where the straight line R-bar approach to the berthing box can commence.

The first alternative has to be excluded because of excessive propellant cost. The second one would, on the contrary, have a lower propellant consumption and approach duration than the one proposed in the example and would also provide good trajectory safety properties. It would not provide, however, much time flexibility to achieve final approach conditions and to cover holds for operational reasons. If for any reason a stop of the approach for more than a few minutes became necessary, a transfer to a hold point on the target orbit would become indispensable. For this reason a hold point S2 on V-bar prior to the manoeuvres for acquisition of the final approach corridor has been considered advantageous and included in the chosen strategy. If, in a real mission, operational experience showed that so much time flexibility were not really required and the hold point on V-bar not necessary, the second alternative strategy would be the natural choice.

Acquisition of the final approach corridor (R-bar approach)

The retro-reflectors for final approach sensors are assumed to be mounted on the Earth-facing side of one of the modules of the station (see figure 5.30). Therefore, the approach taken to acquire the berthing box needs to start from a position S4 directly below these target reflectors. The location of manoeuvre point S4 has to be 300–500 m below the location of the target station to comply with both safety considerations (KOZ) and operational range of the RV-sensor. The selected strategy is

- a Hohmann transfer S2–S3 to a 500 m lower drift orbit and a subsequent drift to S4.

According to Eqs. (3.31), the x-distance for a transfer to a different altitude of $\Delta z = 500\,\mathrm{m}$ will be $\Delta x = \frac{3\pi}{4}\Delta z = 1178\,\mathrm{m}$ (without drag). If the second boost of the transfer must be applied for safety reasons outside the AE, the distance of S2 must be, including some margin, at least 1500 m from the AE or 3500 m from the target CoM. Actually, for the selected strategy of this example, a distance of 5000 m from the target CoM has been chosen. The rationale is that, if the second transfer boost cannot be executed, it will be better to have the subsequent apogee (trajectory returns to the target orbit) outside the AE.

Alternative strategies There are many other strategies with which it is possible to acquire an R-bar approach axis. A direct transfer from S2 to the final approach corridor by tangential or radial impulsive manoeuvres raises too many safety questions, as the trajectory may penetrate the KOZ if it cannot be stopped when arriving on the approach axis. In order to show the considerations for strategy selection, therefore, the following three trajectory examples (see figure 5.29) will also be discussed, for comparison with the selected one:

(1) straight line forced motion transfer from S2 to a point S3f on V-bar, plus forced motion circular fly-around S3f–S4, as shown in figure 3.31;
(2) tangential two-boost transfer from S2 to a point S3t on V-bar, plus a tangential two-boost fly-around S3t–S4;
(3) radial two-boost transfer from S2 to a point S3r on V-bar, plus a radial two-boost fly-around S3r–S4.

For these alternative three cases, the distance of the hold point S2 from the AE can be much smaller, i.e. not more than 500–1000 m, as it has to include only a safety margin which is needed to cover the dispersions of the acquisition of S2. As it is necessary to reduce the relative x-velocity w.r.t. the target to zero, to start the R-bar approach, a stop pulse has to be applied at S4 in all four cases.

Both the tangential (trajectory type 2) and the radial (trajectory type 3) fly-around trajectories have a fixed ratio of $\Delta x : \Delta z$, which requires a start from the location of a particular point S3 on V-bar. For the radial boost fly-around with a Δz of 500 m, this is a distance of 1000 m, and for the tangential one it is about 1180 m. For the forced motion circular fly-around, the location of S3 would be, of course, 500 m from the target CoM.

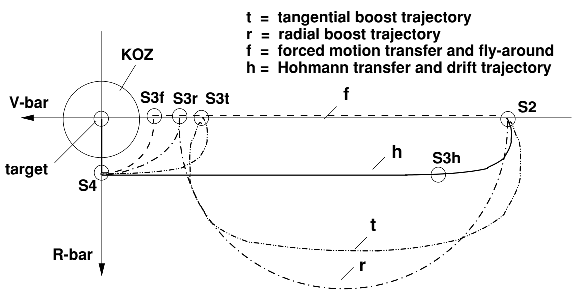

Figure 5.29. Alternative strategies for closing and fly-around (example 2).

Forced motion strategy The straight line transfer S2–S3f of alternative strategy (1) is extremely costly in terms of ΔV, as we have seen already in the first approach strategy example (section 5.7.1). This disadvantage outweighs by far the good monitoring and trajectory safety properties. The forced motion circular fly-around (strategy 1) is also much more propellant consuming than the other trajectory types, and does not provide any particular advantage over the other strategies in terms of trajectory safety or approach recovery. For this reason trajectory type (1) has to be discarded.

Radial boost strategy The advantage of the easy approach recovery with the radial transfer, which contributed to its selection in the case of the V-bar approach to docking (approach strategy example 1), would be of course the same for the transfer from S2 to the starting point S3r of the fly-around. For the fly-around itself, this advantage is theoretically also present. However, the short distance to the KOZ and the lack of monitoring capabilities on the upper ($-z$) side of the ISS would make a 360 deg fly-around unsafe. Also, the safety measure of a small additional ΔV in the tangential direction, as in the case of the closing transfer of strategy example (1), to ensure that the centre of the elliptic trajectory moves slowly away from the target, does not work for a fly-around.

The total duration of the closing and fly-around transfers with radial boosts would take three quarters of an orbital revolution, not including a potential hold after the closing transfer. Comparing the ΔV values for the theoretical impulsive transfers, the radial boost strategy (3) costs 4.2 times as much as the chosen strategy.

Tangential boost strategy The tangential transfer from S2 to the start of the fly-around is unsafe when considering the trajectory evolution after missed or partial thrusts. The same considerations as discussed in section 5.7.1 apply here (see figure 5.27). The tangential boost fly-around, which consists of the same size manoeuvre boosts as those of the last Hohmann transfer of the chosen strategy, is relatively safe. The next apogee would come back to V-bar in front of the target at a distance of about 1100 m, and the trajectory would loop out further in the forward direction. The total duration of the closing and fly-around transfers with tangential boosts would take one and a half orbital revolutions, not including a hold after the closing transfer. Comparing the ΔV values for the theoretical impulsive transfers, tangential boost strategy (2) costs 1.4 times as much as the chosen strategy.

Chosen strategy The easiest and least propellant consuming transfer method is, therefore, the Hohmann transfer to a 500 m lower orbit with subsequent drift (chosen strategy). The duration of the transfer S2–S4 is about 1.3 times an orbital revolution (T), i.e. $0.5T$ for the Hohmann transfer and $0.8T$ for the drift trajectory. Compared with the chosen strategy, the overall trajectory safety and approach recovery features of the radial boost strategy (2) would be no better, but those of tangential boost strategy (3) would be even worse.

Trajectory safety of the chosen strategy S2–S4 The sole critical part concerning trajectory safety is the Hohmann transfer to the 500 m lower drift orbit. Starting on the target orbit at S2, at $x = -5000$ m, the second manoeuvre boost has to be applied at S3h, at $x = 3822$ m. If the second boost cannot be applied, the undisturbed trajectory would return back to V-bar at an x-distance from the target of -2644 m (outside the AE) and, after a second revolution, at a distance of 288 m (outside the KOZ). The next return to V-bar would be at a safe distance in front of the target. However, considering differential drag and thrust errors, the free drift trajectory, following the missed second burn, could become safety-critical (depending on the ratio of the ballistic coefficients) at the second revolution. A shift in position of S2 further outwards would not change the situation, as, in the case of a partial first burn, the target could be hit after a few revolutions anyway. It is, therefore, essential for this trajectory strategy to have a CAM available for the Hohmann transfer from S2 to S3h. The drift after S3h is safe: if the stop pulse in S4 cannot be applied, the chaser vehicle moves forward and eventually leaves the AE. In the case of a partial stop burn, the resulting velocity is slower than that of the corresponding circular orbit anyway, which results in a looping trajectory, starting in a downward ($+z$-direction) motion. The apogee of this trajectory is at the altitude of the drift orbit.

Approach recovery Approach recovery from positions on the drift orbit part of the trajectory S3–S4 would start with a continuation of the drift until the chaser vehicle is outside the KOZ or even outside the AE. This could become necessary, e.g., if the station denies further approach because of operational problems. As the relative drift velocity on a 500 m lower orbit is, at 0.85 m/s (about 4700 m per orbital revolution), still relatively slow, there will be sufficient time to plan and execute recovery manoeuvres. Whether the first recovery manoeuvre can still be executed inside the AE will depend on the ISS control authorities. The aim of the recovery strategy, which, due to the vicinity to the station, can probably be performed under RGPS navigation, will be to return to the hold point S2. From there, automatic approach can be re-initiated. In the case of a CAM, e.g. after failure during the S2–S3 transfer, approach recovery is identical to that from a CAM in the previous example.

Acquisition of the berthing box

The trajectory type for the approach to the berthing box is a straight line forced motion R-bar approach in the type described in section 3.3.3. As for the final approach in the previous example, the definition by the ISS of a narrow approach corridor of ± 10 deg excludes any other trajectory choice. Similar to the V-bar final approach, the velocity profile will contain an acceleration phase, a constant velocity phase and a deceleration phase, as illustrated in figure 6.6. The motion will be stopped at the nominal capture point, followed by a closed loop controlled position keeping mode. At that point, the manipulator end-effector will be moved into the close vicinity of the grapple fixture mounted on the target vehicle, and once it is ready for grappling the active control of the target will be switched off. The manipulator has then a limited time available to perform the grappling (section 5.3.1). It is important also that, during this time, the target station does not perform any manoeuvres which would make the grappling task more difficult. Station attitude motions should be as low as possible.

Trajectory safety S4–berthing box Passive trajectory safety properties, as illustrated in figure 4.14 for the forced motion V-bar approach, are in principle similarly available on a forced motion R-bar approach. Approaching on R-bar with velocity v_z, the trajectory which results after a thrust inhibit at a point z_0 is equivalent to a trajectory starting

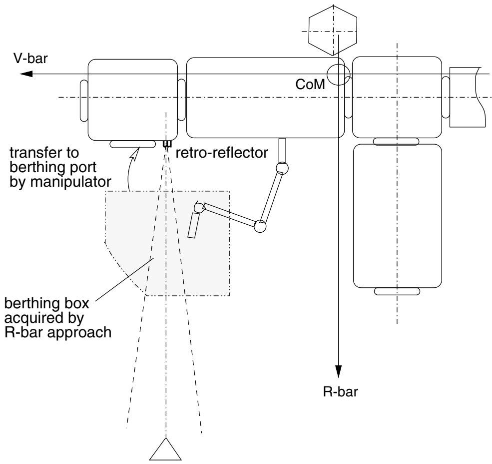

Figure 5.30. Approach to berthing box (ISS scenario).

at the same point z_0 with a radial ΔV_z of the same size. The trajectory evolution of this case can be obtained by addition of Eqs. (3.26) and (3.34). As the orbit dynamic forces, which move the trajectory away from the approach line, are, for a straight line R-bar approach, even stronger than in the V-bar case, it would be possible to design an approach velocity profile such that the trajectory remains passively safe until arrival in the berthing box. This requires, however, that the approach velocities are rather low (see

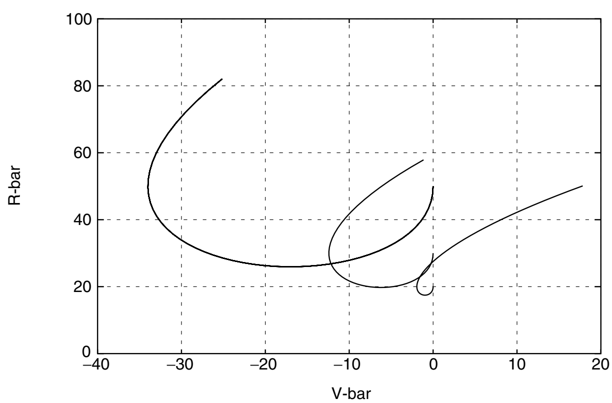

Figure 5.31. Example: trajectories after thrust inhibit during R-bar approach.

figure 5.31). Assuming an extension of the adjacent structure of the target vehicle in the z-direction, as shown in figure 5.30, of not more than 15 m, the approach velocity at

50 m should not exceed 0.1 m/s, at 30 m 0.05 m/s and at 20 m 0.02 m/s. This would result in a very long approach duration for the last few metres. Also, as in example 1, an approach/departure corridor is defined for free drift motions (no thruster activities). Taking into consideration the safety margins around the geometry of the ISS and the geometric extension of the chaser vehicle, the boundaries of this corridor would be exceeded by the free drift trajectories for practically all cases of position and velocity at thrust inhibit. For these reasons also in an R-bar scenario a CAM will be the sole safety measure for the final approach. A suitable CAM would, in contrast to the previous phases and to strategy example 1, consist of a ΔV in the $+z$-direction plus a ΔV in the $-x$-direction, as shown in figures 4.19 and 4.20.

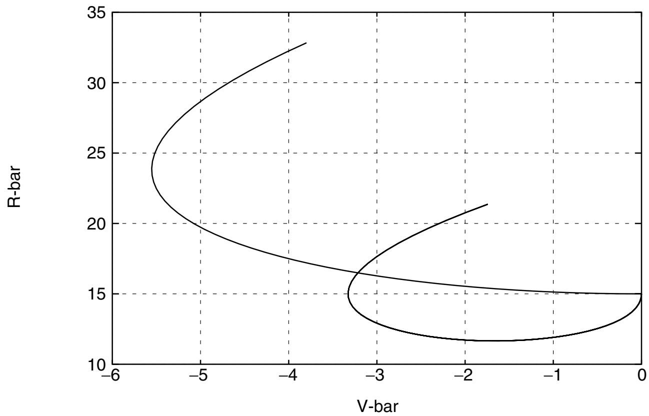

Figure 5.32. Example: trajectory safety at position keeping in berthing box.

The final trajectory element in this example of an R-bar approach to a berthing box is the active position keeping at a z-distance of 15 m from the target orbit. The safety of this part depends on the residual velocities at the time of loss of control.

Figure 5.32 shows the drift trajectories after inhibit of control for two cases: (a) a residual velocity of 0.02 m/s in the $-x$-direction and (b) a residual velocity of 0.02 m/s in the $-z$-direction. After sufficient time (shown are 10 min. for case (a) and 15 min. for case (b)), the free drift trajectories have turned away from the target. The two examples show, however, that within that time, with excursions of >3 m in the $-z$-direction and >5 m in the $-x$-direction, the trajectories would at least penetrate the safety margin around the station structure, if not lead to collision. It is obvious from this example that the residual velocities at thrust inhibit or loss of control in the berthing box would have to be much lower than 0.02 m/s, if one wanted to rely on passive trajectory safety. Otherwise, there would be the danger that the structure of the chaser, which may have geometrical extensions of a few metres, collides with that of the target within the first 5–10 min. As such a requirement would be unrealistic, for any malfunction of control and actuation in the berthing box a CAM will have to be applied.

The CAM for the berthing box will again be a combination of a ΔV in the $+z$- and in $-x$-directions, as in the final approach phase. As one will try for practical reasons to have only one fixed CAM manoeuvre stored per rendezvous phase, this CAM has to be defined according to the part of the trajectory with the maximum needs. For the R-bar trajectory, those maximum needs occur at the end of the approach, i.e. when arriving in the berthing box. In order to ensure that the trajectory after one orbital revolution will be, with sufficient margin, outside the AE, the CAM chosen for the final approach of this example is a boost of 0.5 m/s in the z-direction and 0.15 m/s in the $-x$-direction. The resulting trajectory is shown in figure 5.33. Starting at $z = 15$ m, the trajectory has moved at $z = 30$ m about 4 m into the $-x$-direction, reaching its maximum excursion in the $-x$-direction of less than 10 m at about $z = 80$ m. It can, therefore, be considered safe in the close vicinity of the berthing box. The trajectory does not return along the x-direction after one or more orbits closer than 2500 m to the target. The same CAM applied at the beginning of the trajectory, i.e. at $z = -500$ m, would be after one orbit at an x-distance of more than 21 km in front of the target, with an apogee at the same z-distance as the one at CAM initiation.

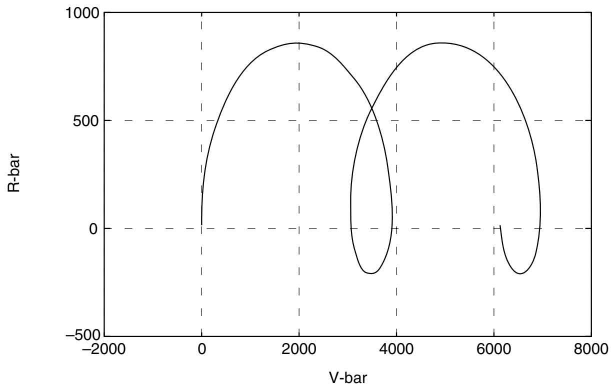

Figure 5.33. CAM trajectory $\Delta V_z = 0.5\,\mathrm{m/s}$, $\Delta V_x = -0.15$ m/s for R-bar approach.

Approach recovery The drift trajectory for the straight line R-bar approach following a CAM is, as we have seen, much more variable than for a straight line V-bar approach. The time and ΔV required for recovery will also depend much more on the conditions at CAM initiation, as for a V-bar CAM. The recovery strategy will therefore have to take time and propellant saving measures into account.

If the CAM had been initiated at the beginning of the straight line approach, i.e. at several hundred metres from the target orbit, the strategy will be to acquire V-bar as soon as possible, e.g. after the first revolution, to save time and propellant for recovery. Because of the large distance, navigation accuracy will be that of absolute GPS. The recovery strategy to return to S2 may include, e.g., a Hohmann transfer to a higher orbit, after which the vehicle is drifting at a safe $-z$-distance above the target, and another one to acquire the manoeuvre point S2 on V-bar, from where the automatic approach can be initiated again.

If the CAM had been initiated at the end of the trajectory, i.e. near or in the berthing box, the x-distance after one orbit is still not very large and the apogee will even be above V-bar. The consequences are as follows: (a) there will be in this case more time available to implement corrective actions; (b) since the apogee of the trajectory is anyway above V-bar (see figure 5.33), it might be convenient to start a two-pulse or three-pulse transfer from there directly to S2.

5.7.3 Approach strategy, example 3

Strategy overview

The third example is a fictitious rescue mission to an incapacitated target vehicle, the docking port of which has a large attitude angle w.r.t. to the LVLH frame. It is assumed that the target has lost power and control, but that it had been designed to function as a target in a rendezvous mission. Because of the loss of control, the spacecraft has assumed a natural torque equilibrium attitude of the order of -30 deg w.r.t. V-bar, and because of the loss of power no communication of the target vehicle with the chaser or ground is possible. As a consequence, GPS on the target does not function, and therefore no RGPS navigation is available. The far and medium range rendezvous sensor used by the approaching vehicle is assumed to be radar. Furthermore, it is assumed that the incapacitated target vehicle possesses as interfaces for an optical rendezvous sensor the usual retro-reflectors near to the docking port and that the rescue vehicle has a camera type of rendezvous sensor.

The target vehicle is on a quasi-circular LEO orbit, the parameters of which are measured from ground by optical telescopes or radar with an accuracy of the order of 1 km. The chaser vehicle may have a GPS receiver, which can be used for absolute navigation, but would not be of much help for relative navigation with the target during rendezvous. In addition to the radar and the optical rendezvous sensor, it is assumed that the rescue vehicle has one or more navigation/observation cameras (e.g. star tracker type) available, which can be used for measurement of the direction of the target (azimuth and elevation angles) in the chaser attitude frame and for video monitoring of the target during approach.

For the far range, the approach strategy is to bring, relatively early, the apogee of the elliptic phasing orbit of the chaser to the altitude of the target orbit. This makes it possible, even at a large distance behind the target, to measure the elevation angle of the target using the navigation camera of the chaser. Using the camera measurements, the apogee of the chaser orbit can thereafter be adjusted more precisely to the target orbit, and also its perigee can be reduced successively. The aim of this series of manoeuvres is that the trajectory passes with its last apogee behind the target through a 'gate', as described in section 2.2.7. This 'gate' is a checkpoint which ensures that, at a defined

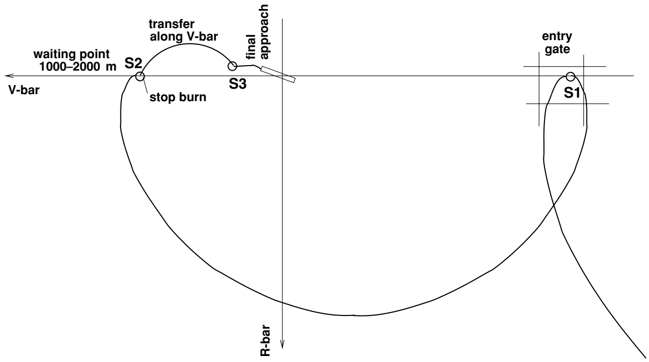

Figure 5.34. Approach strategy to docking port with large skew angle.

range, a defined set of orbit parameters w.r.t. the target is achieved; these parameters are necessary to initiate the close range rendezvous operations. The radar is assumed to have an operational range of the order of 50 km, from which point information on the range to the target is also available.

As the docking port is assumed to be on the +V-bar side, the further approach strategy from the entry gate onward is (see figure 5.34) as follows:

(1) Execution of a tangential boost manoeuvre in the +x-direction at the last apogee about 10 000 m behind the target (manoeuvre point S1), reducing the perigee such that the following apogee (manoeuvre point S2) will be at an x-position of 1000–2000 m in front of the target.

(2) Application of stop boost to acquire the closed loop controlled hold point S2 on the target orbit. The stay time in S2 is used for synchronisation with lighting conditions at docking,

(3) Radial boost transfer to a point S3 at 100–200 m from the docking port, where acquisition of the target retro-reflectors by the optical rendezvous sensor can take place.

(4) Short hold at S3 for inspection of the target docking interfaces by the navigation/observation camera prior to the final approach.

(5) Forced motion straight line approach parallel to V-bar down to a range between the sensor interfaces of 20–30 m.

(6) Circular fly-around of 30 deg to acquire the docking axis.

(7) Straight line approach up to contact on the docking axis, 30 deg canted w.r.t. V-bar.

Acquisition of the target orbit

As described above, the manoeuvres implemented to acquire the target orbit are assumed to be determined with the help of a camera. Assuming an angular resolution of the navigation camera measurements of the order of <0.05 deg, and a knowledge of the instantaneous attitude of the chaser in the LVLH frame of the same accuracy, the altitude uncertainty of the last apogee prior to S1 w.r.t. the target orbit can be reduced to a value of the order of 100 m. The above-mentioned 'entry gate' at S1 has to have an extension in the z-direction, which is equal to the navigation accuracy prior to radar acquisition, plus a margin. The extension of the 'gate' in the x-direction must be larger, according to Eq. (4.16), by a factor of at least 3π more. With the assumed operational range of the rendezvous radar of about 20–50 km, there may be, after radar acquisition, up to half an orbital revolution available to prepare the manoeuvre at S1 under radar navigation. This allows the subsequent transfer S1–S2 with a navigation accuracy of less than 10 m.

To see the target during radar navigation, the vehicle must point with the measurement axis of the radar toward the target, unless if the radar itself has articulation devices for pointing that are independent from the chaser vehicle. In the first case, because of the orbital arcs, the attitude of a vehicle pointing toward the target will change continuously, and with it the angles w.r.t. the Sun and the Earth. This fact will then have repercussions concerning (1) the resolution of ΔV commands into thrust commands for the individual thrusters, in the case of mid-course corrections, and (2) the power budget and the antenna coverage for communication with ground.

Because the mission assumed for this example 3 is not a routine mission, and that the position of S1 is not firmly fixed, the transfer S1–S2 will, in contrast to example 1, not be closed loop controlled. With mid-course corrections, the accuracy of the acquisition of S2 will be of the order of a few tens of metres. However, as the position of S2 will be closed loop controlled under radar navigation after the stop boost of the S1–S2 transfer, the dispersion of S2 can be reduced during station keeping to about 10–20 m. Since for this mission scenario no traffic control zones and rules are to be observed, as for the previous two examples, the hold point S2 could be moved closer to the target.

Trajectory safety S1–S2 For this phase, similar trajectory safety considerations exist as for the same phase of example 1. The big difference is, however, that if no, or only a partial, boost can be executed at S1 and S2, the trajectory is still safe. As the manoeuvre at S1 is a braking boost, a partial boost will lead to a larger loop than planned, putting the chaser further out in front of the target. A loss of boost or a partial boost at S2, which is, in contrast to example 1, now in front of the target, results in a forward motion, leading away from the target. Although the manoeuvres at, and the trajectory between, S1 and S2 are safe, it would still be advisable to have a CAM available to cover other malfunctions of the onboard system.

For the execution of a CAM during the phase S1–S2, the situation will be more complex if the radar antenna is not articulated and the vehicle has to point the docking axis and the axis of the radar antenna toward the target. At S1 and S2 the chaser is aligned with the LVLH frame and its docking port points toward the target. In this attitude, a CAM in the opposite direction of the docking axis, as in example 1, can be executed. Between S1 and S2 there will be a point where the docking axis is pointing in such a direction that a simple CAM in the opposite direction of the docking axis will resolve to such ΔV_x and ΔV_z components that the resulting trajectory can hit the target. An example which would lead to a near collision with the target is given in figure 5.35, which is based on the following assumptions:

- The first part of the trajectory is the nominal transfer from S1 $= -10\,000$ m to S2 $= +1000$ m.
- After 0.76 orbital revolution, a severe malfunction of the attitude control system occurs and a CAM is initiated, i.e. a single boost of 1 m/s in the negative direction of the docking axis of the spacecraft.
  - The trajectory position at this instant is $x = 690$ m, $z = 1094$ m.
  - The pointing angle is $\phi = 57.8$ deg w.r.t. V-bar.
  - With this pointing angle, a CAM of 1 m/s has the components $\Delta V_x = 0.533\text{m/s}$ and $\Delta V_z = 0.846$ m/s.

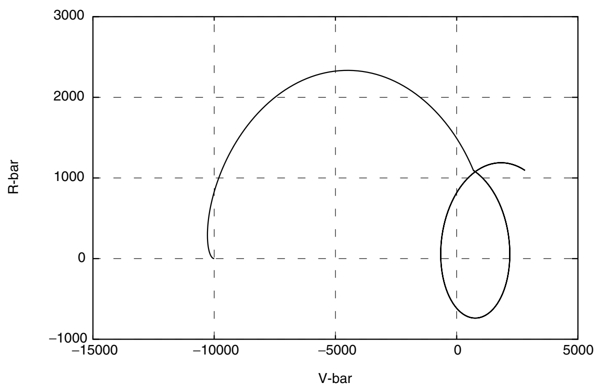

Figure 5.35. Example: CAM at transfer to S2 with target pointing chaser.

This example shows that, whenever the attitude changes significantly over the transfer trajectory, there is no immediate simple solution for a set of boosts which is valid for all points of the trajectory. In such cases, either a particular set of boosts has to be stored in the GNC system for each point, or the vehicle has first to be slewed to a LVLH attitude before the CAM is executed. Both possibilities require basic GNC functions to be available, which is not compatible with the concept of a CAM as a last resort.

Acquisition of the final approach corridor

For the transfer from S2 to S3, a radial boost manoeuvre has been chosen for the same reasons as in example 1. The transfer trajectory S2–S3 is assumed to be closed loop controlled to achieve the necessary accuracy in manoeuvre point S3. The navigation accuracy of the radar improves with the approach, as indicated in figure 5.10. It is assumed that the radar can track particular known features on the target structure, so that the accuracy of the measurement is not corrupted by reflections from different unknown surfaces of the target. The requirements for the position accuracy in S3 come from two factors:

- *Safety*: the uncertainty of the position must not be more than 10% of the range to the target structure, leading to a position accuracy requirement in approach direction of about 10 m at S3.
- *RV-sensor*: the sensor has a field of view of $\pm$ 15 deg. This leads, for sensor acquisition, including margins for attitude control, to a position accuracy requirement in the lateral directions of $<$ 20 m. If the radar measurement has a different reference point on the target than the centre of the docking port, this difference, if not known, will reduce further the permitted lateral performance value.

The camera type of rendezvous sensor is assumed to have a position measurement performance at 200 m of better than 4 m and at 100 m of better than 1 m.

After optical RV-sensor acquisition, the hold point S3 will be position controlled w.r.t. the target reflector pattern on the target. With the assumed TEA angle of 30 deg and an assumed distance of the docking port from the target CoM of 20 m, S3 has a z-position of -11.5 m above V-bar.

Trajectory safety S2–S3 Because of the z-distance of more than 11 m above V-bar, the hold point S3 is not passively safe. In the case of loss of control, the chaser would advance during one orbital period according to Eqs. (3.25) by an amount $\Delta x = 3\pi\Delta z = 108\,\mathrm{m}$ toward the target. Passive trajectory safety and approach recovery considerations for the trajectory element S2–S3 are, for the rest of approach, the same as those of the same phase in example 1, as long as the chaser has an LVLH attitude.

If the radar antenna has no articulation and the chaser is pointing towards the target, as discussed already in the previous phase, the situation with a CAM would be more complicated. Due to the resulting changes of the vehicle attitude over the trajectory S2–S3, the CAM would have thrust components in the $+z$-direction, depending on the position of initiation along the trajectory. The effect of this z-component has to be considered in the evolution of the CAM trajectory. In the assumed scenario, the chaser attitude angle can have values up to $\phi = 35.2$ deg, resulting in a reduction in the x-component of the thrust by a factor of $\cos\phi = 0.817$, which still produces a large ΔV_x-component. In addition, for this trajectory the pointing angle always produces a ΔV_z-component in the $-z$-direction, which helps the escape.

An example for a CAM trajectory initiated at the position with the largest pointing angle ϕ is given in figure 5.36:

- the first part of the trajectory (small arc) is the nominal transfer from $\mathrm{S2} = +1000\,\mathrm{m}$ to S3$= +100$ m;
- after 0.418 orbital revolution, a CAM is initiated;

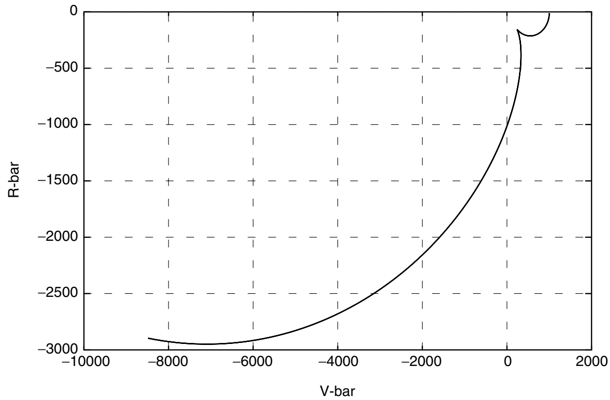

Figure 5.36. Example: CAM at transfer to S3 with target pointing chaser.

- the trajectory position is, at this point, $x = 240.9$ m and z $= -156.6$ m;
- the pointing angle at this point is $\phi = 33$ deg w.r.t V-bar;
- With a CAM of -1 m/s, the components are $\Delta V_x = 0.838$ m/s and $\Delta V_z = -0.545$ m/s.

For all other points, the pointing angle will be smaller and, because of the larger ΔV_x-component, the arc of the CAM trajectories will be larger. A single retro-boost in the opposite direction along the body axis of the chaser will, therefore, be sufficient as a CAM for this approach phase, even for a target pointing chaser.

Acquisition of the docking axis

The final approach up to S4 is in principle identical to that of example 1, described in section 5.7.1, with the exception that it will take place at a larger z-distance from V-bar. The repercussions of this fact on trajectory safety have been discussed already in section 5.3.1. There is in this third example no limitation by an approach corridor. However, the limited FOV of the camera type RV-sensor of ± 15 deg puts a comparable constraint on the trajectory design.

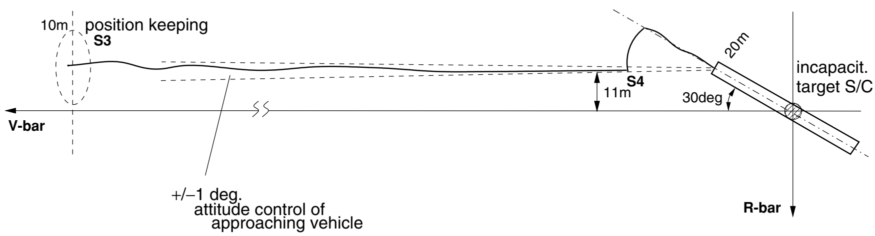

Figure 5.37. Final approach to docking port with large skew angle.

At a range of 20–30 m from the docking port, a closed loop controlled circular fly-around (see Eqs. (3.77) and (3.79)) will be performed. This can be implemented by keeping the range constant, controlling the centre of the target reflector pattern to remain in the centre of the FOV of the sensor, and commanding an attitude slew velocity, moving the chaser docking axis downward. The attitude slew, together with the control of a constant range and the target pattern in the centre, will result in an upward motion ($-z$-direction), which will be stopped when the relative attitude measured by the RVS becomes zero. From this position a straight line forced motion approach along the docking axis with a constant velocity of 0.05 m/s is performed until contact.

Trajectory safety S4–contact As in example 1, passive trajectory safety features cannot be exploited for the final approach. In this example, the passive trajectory features are even worse than in example 1 because of the z-distance of the approach line above V-bar . For this reason a CAM is the sole safety means available in all cases of loss of control. Similar to the first example, a hold and retreat on the trajectory is possible at reasonable propellant cost, as long as the GNC system is still functioning. The CAM for the final approach is again a boost in opposite approach direction. Even at an angle of 30 deg the component of the thrust in the x-direction is still 86% of the total thrust, which will result in an escape trajectory looping in a $-x$-direction. With a boost of, e.g., 0.5 m/s, this would cause the trajectory to return to the target orbit at about 7000 m behind the target.