6

The onboard rendezvous control system

The intention of this chapter is to provide the reader with a short overview of the typical tasks, functions and system hierarchy of an automatic onboard control system for a rendezvous and docking mission. It should provide a basic understanding of the concepts used for various functions, without entering into the details of actual designs. The functions required for automatic control of the vehicle's state vector, for automatic sequencing of manoeuvres and control modes and for automatic detection of failures and initiation of recovery actions are discussed. In section 6.5, the interaction of human operators with an automatic control system and the replacement of some of its functions by human operators is briefly addressed.

6.1 Tasks and functions

During the rendezvous and docking process, the automatic onboard system has to fulfil the following tasks:

(1) preparation and execution of manoeuvres and continuous control of trajectory and attitude (guidance, navigation and control = GNC);
(2) sequencing of phases, GNC modes or manoeuvres, and scheduling of equipment for such modes (mission and vehicle management = MVM);
(3) detection and recovery from system and equipment failures and from critical state vector deviations (failure detection, isolation and recovery = FDIR);
(4) data exchange concerning the rendezvous process and the onboard control system with the ground control centre and the target space station.

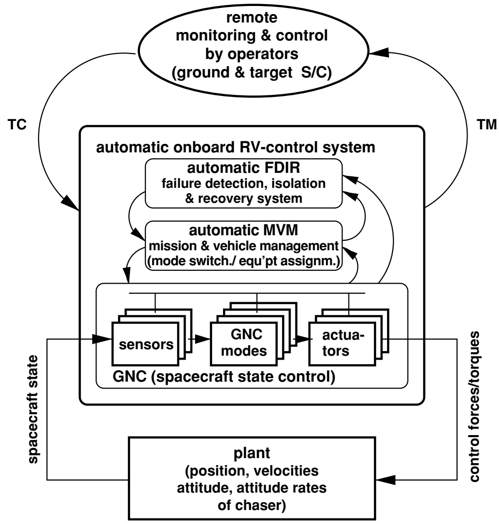

Figure 6.1. Hierarchy of control system for RVD.

There are obviously a number of other functions that the onboard system has to fulfil, such as power, thermal control and housekeeping functions. These functions, however, are not specific to rendezvous and capture, and therefore need not be addressed here. Repercussions on the trajectory strategy and GNC implementation of constraints concerning the resources for other such functions have been addressed in section 5.5.

To fulfil all the above listed tasks, the onboard control system for RVD will have to be designed according to a hierarchical structure, where the 'failure detection, isolation and recovery' (FDIR) function will have to exist at the highest level of authority. The typical hierarchy of the overall control setup for automatic rendezvous is shown in figure 6.1. This simplified figure shows only the levels of authority, not the actual functional relations within such a system. For instance, there will be the need to have failure detection functions at all levels, including on a lower level in the GNC software functions and in the sensor and actuator hardware, as will be explained in section 6.4.

As in an Earth orbit there is no need to perform the rendezvous process fully autonomously, there will be another hierarchical level above the FDIR function outside the chaser vehicle, i.e. the monitoring and control by human operators, aided by automatic tools, on ground and in the target space station. The onboard control system must be designed to allow monitoring and interaction by remote operators (see chapter 9). In the case of contingencies, operators, together with their support tools, may take over one or more of the above listed tasks. A particular case is the remote manual control of the GNC functions by human operators, which is addressed in more detail in section 6.5.

6.2 Guidance, navigation and control

The control loops for attitude and trajectory control include the sensors for position and attitude measurement, the GNC functions, which are implemented in software in the onboard computer, i.e. the navigation, guidance and control functions, and the thrusters and other actuators for attitude and position control. A block diagram of a typical control loop for one of the six degrees of freedom (DOF) is shown in figure 6.2. The disturbances acting on the spacecraft state, such as orbital disturbances and thrust errors, have been discussed already in chapter 4, whereas errors and disturbances of the sensors (block 'measurement environment and disturbances' in figure 6.2) will be covered in chapter 7.

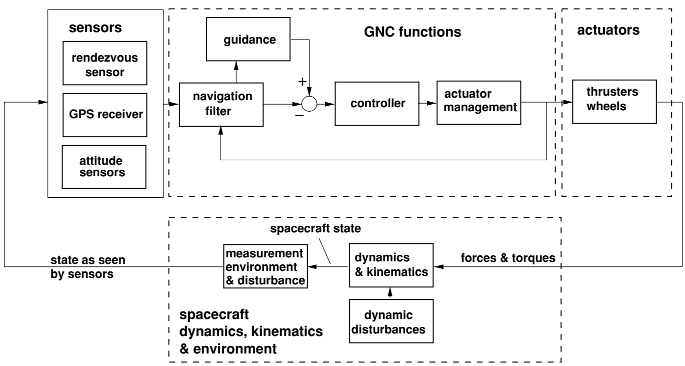

Figure 6.2. GNC functions.

Such a loop has to be implemented for each of the six degrees of freedom to be controlled, i.e. three for rotation and three for translation. During the approach, depending on the distance from the target vehicle, various translation and attitude manoeuvres have to be executed, and various types of trajectories must be controlled (see the previous chapter), for which different sensor types have to be used. This requires a reconfiguration of the control loops each time, in which algorithms and parameters of the navigation, guidance and control functions may have to be changed. The set of algorithms and parameters required for the execution of a particular manoeuvre or trajectory is termed 'GNC mode'. According to the main functions of a GNC system, the GNC modes consist of a set of different navigation, guidance and control modes. The management of these modes, i.e. the engagement of the proper algorithm and parameters, will be discussed in section 6.3.

As long as the distance between the two vehicles is large enough, each DOF rotation and translation may be controlled independently by a SISO (single-input–single-output) control system. For spacecraft with a symmetric shape, e.g. cylindrical ones, the individual body axes can be considered as decoupled. Therefore, SISO control design techniques can be applied for attitude control as well. In docking, during the last part of the approach, when the docking mechanism of the chaser vehicle has to be aligned with the docking port of the target vehicle, all motions are coupled. In this case a MIMO (multiple-input–multiple-output) control system will have advantages. MIMO control may not be necessary, however, for the approach to a berthing box, as in this case angular alignment is less critical (see section 5.3).

For each navigation mode, the navigation function consists of a Kalman filter, which processes the various information of attitude (gyros, updated by Sun, Earth and/or star sensors) and trajectory sensors (RGPS, RVS) and propagates the vehicle state in position and attitude by using the knowledge of the dynamic behaviour and information on the actual thrust commands (see section 6.2.1). The guidance function defines the set values for the nominal evolution of the spacecraft state, i.e. the references for the control of position, velocities, attitude and angular rates at each point in time (see section 6.2.2). The control function produces the force and torque commands necessary to achieve the desired corrections in attitude and trajectory and to ensure stability of the vehicle. The thruster management function transforms the torque and force commands into 'on/off' commands for the individual thrusters. This function is of particular importance for vehicles which have their thrusters located in an unbalanced arrangement w.r.t. the centre of mass. In such cases each translation force and each rotation torque has to be produced by a combination of various thrusters with burns of different duration. Control function and thruster management function are addressed in section 6.2.3.

6.2.1 The navigation filter

The task of the navigation function is to provide the controller and the guidance function with the necessary information on the present state of the vehicle. As a rule, this function is implemented as a digital filter which processes the various information inputs related to the vehicle state obtained from different sensors, from the actuators or via communication links from external sources. The purpose of such a filter is to obtain out of several inputs related to the vehicle state an estimation of the state vector with reduced noise errors. A filter, which propagates the state, will also be helpful in cases where the sensor information is only intermittently available. If there were a single sensor continuously providing all necessary information on the state vector with sufficiently low noise, the navigation function could be reduced to converting the sensor information to formats as required by the guidance and control functions.

Principle of a Kalman filter

In a navigation system that is obtaining information from different sources, a best possible estimate for the actual state vector has to be calculated from the various inputs. In systems operating in computation cycles, the output of the navigation function has, in addition, to be a propagation of this estimate as an input to the following cycle of the control output calculation. The algorithms generally used for this estimation are those relating to the Kalman filter (Kalman 1960), an optimal estimator in the 'least square' sense, which minimises the variance of the estimation error. The Kalman filter is well documented in the literature, and detailed descriptions and derivations of the filter equations can be found in many text books on digital control of dynamic processes, such as Issermann (1981) Brown & Hwang (1992) and Franklin, Powell & Workman (1998). For the description of the functional principles of a GNC system, it will not be necessary to repeat all the details relating to the Kalman filter; the intention of the following brief description is to familiarise the reader with the basic principles of the function of a Kalman filter, as far as is necessary for the understanding of the general operating principle of a navigation filter and in the possibilities of failure identification (see figure 6.18). The latter problems are discussed in more detail in section 6.4.

It shall be assumed here that the state of a discrete time invariant plant at the step '$k+1$' can be described by

$$\mathbf{x}_{k+1} = \mathbf{A}\mathbf{x}_k + \mathbf{B}\mathbf{u}_k + \mathbf{w} \tag{6.1}$$

(see also Eq. (A.16) in appendix A), and the measurements related to the state at the step 'k' can be described by

$$\mathbf{y}_k = \mathbf{G}\mathbf{x}_k + \mathbf{v} \tag{6.2}$$

where

$\mathbf{A}$ is the state transition matrix, which describes in the given dynamic process the changes of the state from step 'k' to '$k+1$';
$\mathbf{B}$ is the input matrix, which describes the relations between the inputs $\mathbf{u}$ and change of the state vector;
$\mathbf{u}$ are the external inputs to the system between steps 'k' and '$k+1$', e.g., in the case of a spacecraft, the control forces and torques;
$\mathbf{w}$ is the system noise;
$\mathbf{G}$ is the measurement model matrix (also called the output matrix), describing the theoretical relations between various measurements and the state vector;
$\mathbf{v}$ is the measurement noise.

The principle of filter operation is to provide, at each step, a prediction, i.e. a propagation in time, and a correction, i.e. an update based on measurements. The following calculations are performed. At each processing step the filter provides first a propagated value of the state vector for the new step

$$\mathbf{x}^*_{k+1} = \mathbf{A}\hat{\mathbf{x}}_k + \mathbf{B}\mathbf{u}_k \tag{6.3}$$

and a propagated (expected) value for the new measurement vector

$$\mathbf{y}^*_{k+1} = \mathbf{G}\mathbf{A}\hat{\mathbf{x}}_k \tag{6.4}$$

Both are calculated from the estimated state vector of the previous step and from the expected changes between 'k' and '$k+1$', given by the transition matrix.

The new estimate for the state vector at step '$k+1$' is calculated using the propagated value plus a correction, based on the difference between the new measurement vector $\mathbf{y}_{k+1}$ and the expected one:

$$\hat{\mathbf{x}}_{k+1} = \mathbf{x}^*_{k+1} + \mathbf{K}_{k+1}[\mathbf{y}_{k+1} - \mathbf{y}^*_{k+1}] \tag{6.5}$$

In words, we could state this as

$$\begin{matrix}\text{new}\\ \text{estimate}\end{matrix} = \begin{matrix}\text{predicted estimate,}\\ \text{based on old estimate}\end{matrix} + \begin{matrix}\text{correction}\\ \text{matrix}\end{matrix}\left(\begin{matrix}\text{new}\\ \text{measurement}\end{matrix} - \begin{matrix}\text{predicted measurement,}\\ \text{based on old estimate}\end{matrix}\right)$$

The difference between the new and the propagated observation, $\mathbf{y}_{k+1} - \mathbf{y}^*_{k+1}$, is in the literature often referred to as the 'innovation'.

We now need $\mathbf{K}$, the correction matrix (also called the 'gain matrix'), which provides a weighting factor for the contribution of the innovation information to the value of the new estimate:

$$\mathbf{K}_{k+1} = \mathbf{P}^*_{k+1}\mathbf{G}^{\mathrm{T}}[\mathbf{R} + \mathbf{G}\mathbf{P}^*_{k+1}\mathbf{G}^{\mathrm{T}}]^{-1} \tag{6.6}$$

where $\mathbf{P}^*_{k+1}$ is the estimation error covariance matrix propagated from the previous step, and $\mathbf{R}$ is the covariance matrix for the measurement noise $\mathbf{v}$.

The covariance matrix $\mathbf{P}_{k+1}$ of the state vector error for the new step '$k+1$' will be obtained using the propagated value $\mathbf{P}^*_{k+1}$ and the correction matrix K_{k+1}:

$$\mathbf{P}_{k+1} = \mathbf{P}^*_{k+1}[1 - \mathbf{K}_{k+1}\mathbf{G}] \tag{6.7}$$

The updated error covariance matrix $\mathbf{P}_{k+1}$ is not used in the gain matrix calculation of the present step, but will be used to calculate the propagated value for the subsequent step. In step '$k+1$' the propagated error covariance matrix $\mathbf{P}^*_{k+1}$ is calculated from the value obtained for the previous step 'k' as follows:

$$\mathbf{P}^*_{k+1} = \mathbf{A}\mathbf{P}_k\mathbf{A}^{\mathrm{T}} + \mathbf{Q} \tag{6.8}$$

where $\mathbf{Q}$ is a covariance matrix for the system noise $\mathbf{w}$, for which white noise is assumed. The values for $\mathbf{x}_{k+1}$ and $\mathbf{P}_{k+1}$ calculated from Eqs. (6.5) and (6.7) will become the $\mathbf{x}_k$ and $\mathbf{P}_k$ in Eqs. (6.3) and (6.8) of the calculations in the subsequent step.

The principle of operation of a Kalman filter is shown in figure 6.3, which indicates schematically the flow of data and the major calculations to be performed during one cycle. The interruption of the data flow return lines from the state estimation and error covariance matrix outputs to the propagation block indicate that these outputs will be used in the subsequent cycle. The block labelled 'preparation of measurement vector' indicates that the individual measurements obtained from various types of sensors may first have to be brought into a form suitable for further use in the subsequent calculations.

The Kalman solution has some practical implementation drawbacks. It is valid, in theory, only for cases where prediction and measurement errors are equal to Gaussian random processes with zero mean. The effects of errors with a different distribution law

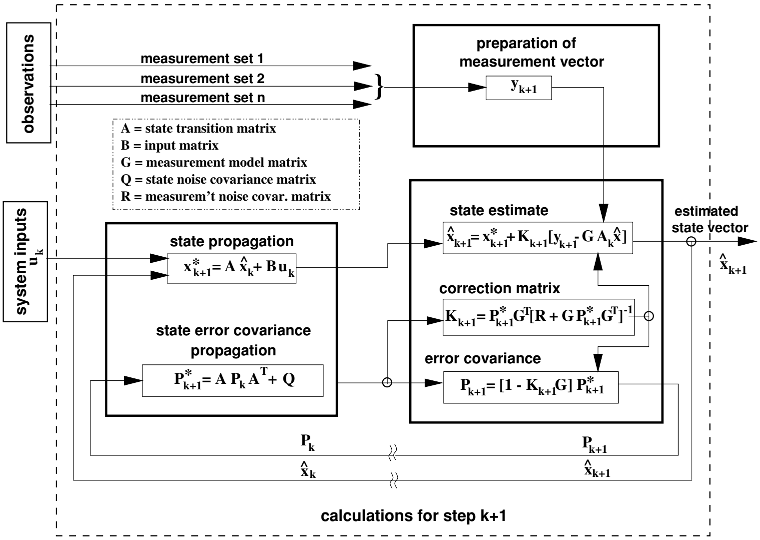

Figure 6.3. Block diagram of a Kalman filter.

(e.g. gravity errors in the prediction or measurement biases in the innovation), will have to be approximated by 'equivalent' zero-mean Gaussian random processes.

The Kalman gain computation, Eq. (6.6), and the associated covariance matrix propagation, Eqs. (6.3) and (6.8), can be extremely expensive in terms of computer load if the dimension of the state vector is not small. This can be problematic for the limited capabilities of the computer hardware available for operation in space at the time of writing, but may be less of a problem in the future, when more powerful space-qualified computers become available. The following solutions can be used to circumvent these limitations:

- the coefficients of the matrix $\mathbf{K}$ can be determined in simple SISO cases by pole placement, selecting a limited set of acceptable 'physical' coefficients, such as bandwidth and damping ratio, on the combined propagation, update equations;
- the Kalman equation can be solved to determine asymptotic gains, for k going to $+\infty$.

Performance evaluation analyses then permit convergence on an adequate set of gains. These are efficient methods for stationary systems, i.e. systems where measurements and transition matrices do not face significant variation w.r.t. time. One or the other of the above methods can be used on a sequential set of working set-points, in order to determine pre-programmed gains, which will be used in sequence as the mission advances. This feature is called 'gain scheduling'. An example is the use of a set of successive gain matrices for the rendezvous sensor based navigation estimation, in cases where the sensor accuracy improves commensurate with decreasing distance between sensor and target.

Rotational and translational motion components in the state vector

During the rendezvous phases, after relative navigation has started, the state vector $\mathbf{x}$ to be estimated consists of the instantaneous position $\mathbf{p}$ and the velocity vector $\mathbf{v}$, measured in the local orbital frame $\boldsymbol{F}_{\mathbf{lo}}$ of the target, and of the attitude angles $\boldsymbol{\alpha}$ and angular rates $\boldsymbol{\omega} = \dot{\boldsymbol{\alpha}}$, measured between the spacecraft attitude frame $\boldsymbol{F}_{\mathbf{a}}$ (body frame) and the reference frame (i.e. the local orbital frame $\boldsymbol{F}_{\mathbf{lo}}$ of the chaser), measured in the $\boldsymbol{F}_{\mathbf{a}}$ frame:[1]

$$\mathbf{x} = [\mathbf{p}, \mathbf{v}, \boldsymbol{\alpha}, \dot{\boldsymbol{\alpha}}]^{\mathrm{T}} \tag{6.9}$$

$$\mathbf{p} = \begin{bmatrix} x \\ y \\ x \end{bmatrix} \qquad \mathbf{v} = \begin{bmatrix} \dot{x} \\ \dot{y} \\ \dot{z} \end{bmatrix} \qquad \boldsymbol{\alpha} = \begin{bmatrix} \alpha_x \\ \alpha_y \\ \alpha_z \end{bmatrix} \qquad \boldsymbol{\omega} = \begin{bmatrix} \omega_x \\ \omega_y \\ \omega_z \end{bmatrix} \tag{6.10}$$

In docking, during the last part of the approach the navigation must be related to the docking frame of the chaser, which requires the availability of information on the relative attitude between the vehicles as an additional part of the state vector. The relative attitude vector $\boldsymbol{\delta} = \boldsymbol{\alpha}_{\text{chaser}} - \boldsymbol{\alpha}_{\text{target}}$ is measured in the $\boldsymbol{F}_{\mathbf{a}}$ of the chaser:

$$\boldsymbol{\delta} = \begin{bmatrix} \delta_x \\ \delta_y \\ \delta_z \end{bmatrix} \tag{6.11}$$

In the relative navigation phases the state transition matrix for the translational motion components $\mathbf{A}_{\text{trans}}$ and the input matrix $\mathbf{B}_{\text{trans}}$ in Eqs. (6.3), (6.4) and (6.8) can be derived from the Clohessy–Wiltshire equations (3.22). For the rotational motion components, the state transition matrix $\mathbf{A}_{\text{rot}}$ can be derived from the angular momentum law. A derivation of the equations of motion is provided in appendix A.

Navigation filter issues relevant to rendezvous missions

After the initiation of the filter, time is needed to reach a steady state in the feedback system of predictions and corrections. Depending on the dynamics of the process, on the noise and on the tuning of the filter parameters, this may take, in the case of navigation filters for spacecraft, up to a few minutes. Filter convergence time has to be taken into account in the planning of manoeuvre schedules, since, when a new sensor is introduced, the full navigation performance required for calculation of the manoeuvre boost will be available only after convergence of the navigation filter.

Filter convergence requires that the error covariance matrix $\mathbf{P}_{\boldsymbol{k}}$ becomes stable, i.e. that the corrections to the state vector reach a consistent level. The filter can diverge, in which case the difference between the estimated state vector and the measurements increase with each cycle, when either:

[1]The angular velocity vector $\boldsymbol{\omega}$ should not be confused with the orbital rate ω_0.

- the measurement data are inconsistent, the measurement model matrix $\mathbf{G}$ and/or the input matrix $\mathbf{B}$ do not represent the relations to the real world correctly, or
- the state noise in relation to measurement noise has not been modelled correctly.

Whether the input matrix and the measurement model matrix are representative of the situation can be verified by analysis and test prior to the mission; inconsistencies in the measurement data, however, can be caused both by sensor failures and by disturbances in the measurement environment, which may be more difficult to determine beforehand. Also, extreme changes in the measurement noise may affect the balance between the covariance matrices for state noise and measurement noise.

Depending on the type of state vector information required for the different GNC modes in the various rendezvous approach phases, and on the type of sensors or other sources of information to be used, different navigation filters will have to be designed.

- A filter for absolute attitude estimation providing the attitude information for the other filters and for use as navigation filter in contingency situations where position information is no longer available.
- A filter for absolute position estimation using, e.g., absolute GPS information, required during phasing, in contingency phases and to provide observation input for the relative navigation filter.
- Filters for relative navigation between chaser and target for the rendezvous phases proper. Due to the different measurement principles, dynamics, noise and error characteristics, different filter designs will be needed per type of sensor used, e.g. for
  - relative GPS,
  - radio-frequency sensors, such as radar,
  - scanning laser range finders,
  - camera sensors.
- Different filter designs may be needed also in the case of the same type of sensor, where a different type of information is used, during the various approach/departure phases. This is the case, e.g., in the last part of the approach, when the rendezvous sensor also has to provide relative attitude.

The primary sensors providing the input for absolute attitude $\boldsymbol{\alpha}$ and angular rates $\dot{\boldsymbol{\alpha}}$ will be the gyroscopes. However, as gyro output drifts over time, updates will be necessary in regular intervals using measurements of extero-receptive sensors such as Earth, Sun or star sensors. Intervals of updating and the switching to the updating mode will have to be controlled by the mission management function (see section 6.3). Relative attitude information, required for the last part of the approach to docking, will usually be provided by an optical rendezvous sensor. The various types of rendezvous sensors, which provide measurements necessary for trajectory control in the different rendezvous phases (i.e. measurements of x-, y-, z-position, or of range r and direction angles ψ, θ, and of the relative attitude vector $\boldsymbol{\delta}$ for the last part of the final approach) will be discussed in more detail in chapter 7 (see figures 7.28, 7.29 and 7.31).

As navigation filters for different inputs and outputs can be active in parallel, the inputs to the measurement vector block are obtained either directly from trajectory sensors and attitude sensors or from another navigation filter, as described above. Information on the thrust commands, as produced by the thruster management function, will be used as the input for the state vector propagation. As the thrust command information is related to a geometric frame of the spacecraft ($\boldsymbol{F}_{\mathbf{ge}}$, see section 3.1.5), information on the absolute attitude must also be provided to the propagation block, in order to be able to propagate the translation motion in the local orbital frame $\boldsymbol{F}_{\mathbf{lo}}$.

6.2.2 The guidance function

The task of the guidance function is to provide at each point the set values for the state vector in time, which will then be compared with the estimated actual values, provided by the navigation function, enabling the control function to prepare the control commands. Depending on the manoeuvres and trajectories to be implemented, the guidance function has to:

- pre-calculate boost manoeuvres in terms of execution time and duration;
- generate position and velocity profiles, $\mathbf{p}(t)$ and $\mathbf{v}(t)$, in all axes for closed loop controlled trajectories and hold points;
- generate attitude profiles $\boldsymbol{\alpha}(t)$, e.g. for spacecraft pointing towards Earth, the Sun or a target vehicle, and angular rate profiles $\dot{\boldsymbol{\alpha}}(t)$ for closed loop controlled slew manoeuvres (large attitude angle rotation);
- propagate the instantaneous position of the centre of mass in the vehicle body frame according to the propellant consumption during the mission.

The last of the above tasks could have been grouped under a different function. As it is, however, neither a typical navigation task nor a typical control task, it has been listed here under the guidance tasks. The propagation of the instantaneous position of the spacecraft's CoM w.r.t. a body reference frame ($\boldsymbol{F}_{\mathbf{ge}}$) is a continuous update process throughout the mission, which does not depend on the type of manoeuvre executed or the state to be achieved, but rather on the amount of propellant consumed and on the location of the tank it is drawn from. The result of this update is used in the navigation function (e.g. in the input matrix of the navigation filter), in the control function (e.g. for the thruster management function) and in the guidance function (e.g. for the coordinate transformation from the sensor or docking frame to the nominal attitude frame).

The first three tasks are the classic guidance tasks, defining the state to be achieved over time. According to the strategy chosen for rendezvous and departure (see section 5.7), some of the following modes may have to be implemented. Typical nominal approach and departure modes are

- two-boost tangential and radial transfers,
- position keeping on V-bar and R-bar,
- fly-arounds to an R-bar approach line or to arbitrary x-, y-, z- positions,

- straight line approaches on V-bar and R-bar,
- straight line approach to a docking axis pointing in arbitrary directions,
- free drift (no position control) with and without attitude control,
- slew to an arbitrary attitude.

Typical contingency modes are

- braking and hold on V-bar,
- braking and straight line retreat on docking axis (V-bar and R-bar),
- two-boost radial transfer to, and acquisition of, a hold point,
- two-boost tangential transfer to, and acquisition of, a hold point,
- slew manoeuvre to Sun-pointing attitude.

The guidance laws for translation motion will be derived from the trajectory equations in chapter 3 for the various impulsive manoeuvres and continuous thrust trajectories. Disturbances, in particular differential drag between chaser and target, may have to be taken into account in the calculation of the nominal trajectory evolution. For open loop manoeuvres this will be a necessity in any case; for closed loop controlled trajectories, drag forces and other disturbances will continuously be counteracted by the control forces, as far as they have not been taken into account in the calculation of the nominal trajectory. For rotational motion, the guidance laws define the profile over time for the attitude angles relative to Earth, the Sun or the target vehicle, and for the angular rates in the case of slew manoeuvres.

Whereas the guidance laws can be validated prior to the mission and stored in the onboard data management system, the guidance parameters will in many cases have to be calculated from the actual state and the state to be achieved immediately prior to a manoeuvre. This is particularly so for the ΔVs to be applied and the time when they have to be executed. For instance two-boost transfers may be implemented as:

- open loop manoeuvres, in which case the guidance function has to calculate the start time and the duration of each boost;
- manoeuvres with mid-course corrections (figure 6.4), in which case the guidance function has to calculate, in addition, for pre-determined points in time, the correction boosts, derived from the nominal trajectory development vs. the actual one, as determined by the navigation function;
- closed loop controlled trajectories, in which case the guidance function provides the control function with the actual nominal value for the state vector at each process cycle between the boosts (see figures 4.8 and 4.9 in section 4.4.1).

The guidance laws have to take into account that the propulsion system can realise neither perfect impulses nor continuous accelerations. Thruster operation will always consist of a series of boosts of limited time. For two-pulse transfers the thrust duration t_2 of each pulse τ and the start time of the second pulse have to be calculated, as indicated for two examples in section 3.3.3. For constant thrust manoeuvres, such as straight line V-bar and R-bar approaches, the required thrust level has to be realised by pulse width modulation. Accordingly, the guidance laws have to take into account that there will be

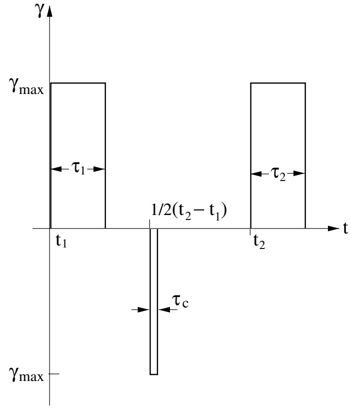

Figure 6.4. Two-boost transfer manoeuvres with mid-course correction.

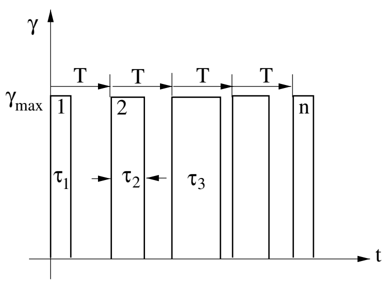

Figure 6.5. Pulse width modulation.

a minimum or threshold acceleration level, since the minimum pulse duration for τ in figures 6.4 and 6.5 will be determined by the characteristics of the thruster (minimum impulse bit, MIB) and by thrust efficiency considerations (see also figure 6.14). The minimum granularity will thus be given by the thrust duration $\tau_{\min}$ and the duration T of the computation cycle, i.e. the minimum average thrust level is

$$\gamma_{\min} = \frac{\tau_{\min}}{T}\gamma_{\max}$$

The maximum average thrust level is obviously achieved when the 'on' time τ is equal to the cycle duration T. In this case, the thruster will not be switched off at the end of the cycle. When large accelerations are required, this condition can last over many cycles. For modes where such large accelerations are not required, the condition of continuous 'thruster on' may be used as a failure criterion (see section 6.4). In any case, the guidance laws must be designed such that the acceleration request does not exceed the maximum possible thrust level.

For forced motion straight line approaches and similar trajectory elements, an acceleration profile has to be implemented at the start of the motion to achieve a desired approach velocity. Correspondingly, at the end a deceleration profile has to be implemented to arrive at the desired position with the desired velocity. At the start of motion a constant acceleration will bring the vehicle in the shortest possible time to the desired approach velocity, and thrust in this direction can be stopped when the velocity is achieved. For the deceleration phase constant thrust is less desirable, as not only a final velocity but also a final position has to be achieved, and thrust level errors would translate into position errors with the square of the time. For this reason, other guidance laws have been developed, such as exponential deceleration.

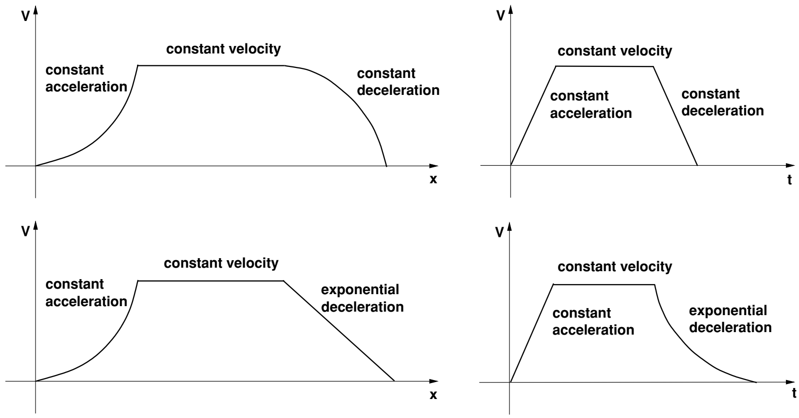

Figure 6.6. Examples of velocity profiles.

The exponential braking law, which has been formulated to fulfil passive safety criteria in case of loss of thrust (see figure 4.14), is characterised by an exponential change of position and velocity with time of the type

$$x(t) = X_0 \cdot \mathrm{e}^{-\frac{t-t_0}{\lambda}}$$
$$\dot{x}(t) = \dot{X}_0 \cdot \mathrm{e}^{-\frac{t-t_0}{\lambda}}$$

which results in a linear profile in the approach phase plane of

$$\dot{x}(t) = \dot{X}_0 - \lambda(x(t) - X_0)$$

with λ being a proportionality factor, tuned to the safety area required by the target station.

Other schemes, such as proportional braking, are conceivable, which reduce the applied deceleration proportional to the elapsed time in order to allow a smooth acquisition

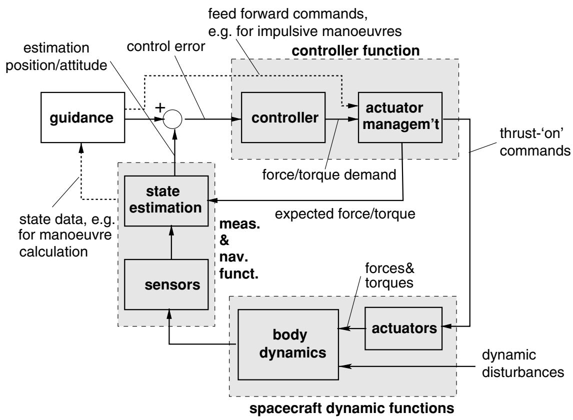

Figure 6.7. Functional block diagram of a closed loop controlled spacecraft.

of a pre-determined velocity at a particular position. Such a scheme would, however, result in even longer braking times.

Finally, another potential task of the guidance function should be mentioned, which has already been addressed in section 4.4.1, i.e., for closed loop controlled trajectories, the calculation of safety boundaries around each component of the state vector for each point in time, as discussed in section 4.4.1. This information will have to be provided to the FDI function (see section 6.4, figure 6.20) in case such state vector 'corridor' checks are included in the system design. The state vector safety boundaries have to be calculated together with the nominal trajectory.

6.2.3 The control function

The task of the control function is to provide the force and torque commands which will be executed by the reaction control system of the spacecraft to correct the deviations of the actual state vector from the nominal one. While the guidance function provides the nominal or reference state (previous section), and the navigation function estimates the actual state (addressed in section 6.2.1), from the differences of the two states the control function produces actuation commands to compensate for the effects of disturbances and errors. The performance of such a control loop is determined by the dynamic behaviour and errors of its elements and by the disturbances acting on them. The various disturbances and their points of interaction are shown in figure 6.8.

For the purpose of discussion of the control function, the functional blocks of figure 6.2 have been re-arranged into the reference functional block diagram, figure 6.7. Groups of functions which will be treated later in this section by a single mathematical model (transfer function) are marked by shaded boxes.

The *plant* in figure 6.7 is the block representing the six degree of freedom motion dynamics of the spacecraft, i.e. the dynamics of translational motion (position dynamics) and rotational motion (attitude dynamics). The coupling between translational and rotational motion is due to the orbital rotation (small) and the translational components $r \cdot \sin\alpha$ that result from a rotation angle α at a distance r from the centre of rotation. The latter effect plays a role when the vehicle is controlled w.r.t. a frame which does not originate in its CoM. This is the case when both lateral and rotational alignment with the target docking axis has to be achieved prior to contact. In all other cases, coupling of translational and rotational motions is rather small, and can usually be neglected. The position dynamics of motions of the CoM in orbit can be described, as already mentioned, by the Hill equations, Eq. (3.21), and the attitude dynamics by Euler's moment equations (see appendix A).

The *measurement and navigation function* includes the sensors and the state estimation, i.e. the navigation filter discussed in section 6.2.1. As already stated above, for measurement of the rotational motion the sensors used are inertial sensors, i.e. any type of gyroscopes. The measurement principles of sensors used for translational motion or position measurement are treated in chapter 7. Since all sensors are imperfect (bias, noise, disturbance by measurement environment, bandwidth), measurement performance enters the loop and impacts on the overall system behaviour.

The *controller function* includes, for the purpose of control loop analysis, the controller proper and the actuator management function, which translates the force/torque commands into 'on/off' commands for the individual thrusters. As the coupling between rotational and translational motion is rather small, for most of the approach, except for the last metres prior to docking, where the position and attitude loops can be separated. For the actuator management function this is true only provided the propulsion system allows independent control of position and attitude.

The *actuation system* is composed of thrusters, which in a spacecraft generally may be supported by rotating actuators (wheels) and magneto-torquers to reduce fuel consumption. In the case of a chaser vehicle for rendezvous, usually no rotating actuators are used, because (a) for such a short mission the added mass and complexity due to the additional hardware would outweigh the propellant mass saving by wheels, and (b) because the presence of an angular momentum would complicate the spacecraft dynamics in the last phase prior to contact. Concerning the forces and torques produced, the actuation system belongs in the control loop analysis to the block 'spacecraft dynamics'.

During the final phase of the rendezvous, the six degrees of freedom of motion of the vehicle must be controlled simultaneously. This could in principle be done by selecting a set of thrusters with components in three directions, and generating for each operating cycle the needed force/torque vector. Due to operational constraints of the thrusters, this may lead, however, to imperfect force and torque realisation, which creates input disturbances to both the position and attitude dynamics (see Eq. (6.30)).

Analytical relations in a control loop

For convenience of analysis, the closed loop system is usually described by a linear approximation about the operating point of the real function, which may be non-linear in its full range. This permits modelling of the system by a set of first order differential equations and the application of Laplace transforms, which transform the differential equations into easier to solve algebraic ones. From the controller design and analysis point of view, the block diagram, figure 6.7, may be redrawn into figure 6.8, which will be used as the reference diagram for the short explanation of control analysis in this section. The control loop is characterised by the transfer functions of its elements:

$K(s)$ = transfer function of the controller

$G(s)$ = transfer function of the plant (spacecraft dynamics)

$M(s)$ = transfer function of the sensing function (measurement and navigation function in figure 6.7)

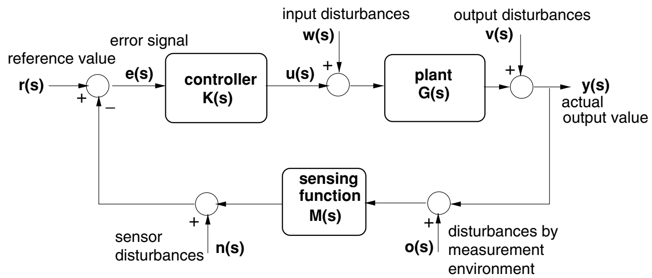

Figure 6.8. Reference control loop.

In our reference control loop diagram the guidance function becomes simply the reference input $r(s)$, the actuation system is merged with the plant in the transfer function $G(s)$, the navigation function and sensor dynamics are merged in the transfer function $M(s)$. In these functions s is the new variable in the Laplace domain of a function $f(t)$ in the time domain:

$$\mathcal{L}[f(t)] \equiv F(s)$$

which has the convenient property

$$\mathcal{L}[\dot{f}(t)] = sF(s) - f(0)$$

Without the disturbances, the ratio of reference and output values is the transfer function of the closed loop system:

$$\frac{y(s)}{r(s)} = T(s) = \frac{K(s)G(s)}{1 + K(s)G(s)M(s)} \tag{6.12}$$

The inputs, outputs and disturbances in figure 6.8 are as follows:

$r(s)$: reference signal (guidance value),

$e(s)$: error signal (difference between reference and measurement),

$u(s)$: control output,

$w(s)$: input disturbances (external disturbances on plant),

$v(s)$: output disturbances (internal disturbances of plant),

$o(s)$: disturbances from the measurement environment,

$n(s)$: sensor disturbances (bias, noise, etc.),

$y(s)$: plant output.

Depending on the characteristics of the errors and disturbances, some of them may be combined for convenience of analysis, e.g. the disturbance from the measurement environment $o(s)$ and from the sensor itself $n(s)$ are usually combined in $n(s)$.

The quality of the control action is determined by the *steady state error*, i.e. the difference between the reference and the output state achieved after settlement of transients, by the *transient response* characteristics and by the *noise rejection capability*. The transient response describes the time required to achieve steady state conditions and the output behaviour during this time. The latter includes the natural frequency of the system and the damping behaviour. These characteristics entirely depend on the transfer functions of the elements shown in figure 6.8.

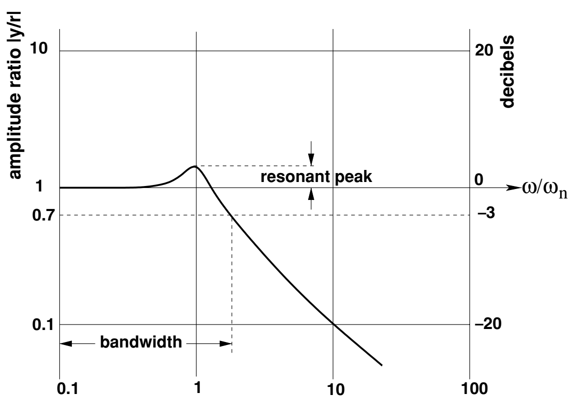

Figure 6.9. Bandwidth of a control loop.

The *transient response* will depend on the frequencies contained in the input signal. The response will be different if the input frequency is below or above the natural frequency of the loop. Far below the natural frequency, the relation of response to input amplitude will be equal to unity (or the value given by the loop gain); far above the natural frequency, it will approach zero. There is thus a limit in frequency up to which the output of a system will track a sinusoidal input. By convention, the frequency at which the output amplitude has dropped by 3 dB, i.e. the output is 0.707 of the input magnitude, is called the *bandwidth*, ω_b.

In the following, a few basic properties and characteristics of a control loop will be briefly explained in the example of a simple second order system (mass/inertia-control stiffness-control damping), to provide a reference for the discussion of the performance of the control function in the rendezvous process. It goes without saying that in real applications systems may be more complex. The relations of resonant frequency, damping and steady state error can be explained in the example of a unity feedback system (figure 6.10), in which the full output is fed back. In most cases a given control loop can be reduced by manipulation of the transfer functions in a block diagram to a unity feedback system, so that the results of this discussion will be widely applicable.

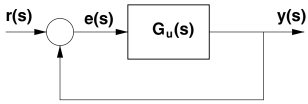

Figure 6.10. Unity feedback system.

Combining the transfer functions of controller and plant into one transfer function, i.e. the open loop transfer function $G_\mathrm{u}(s)$, the closed loop transfer function, Eq. (6.12), of the unity feedback system becomes

$$T(s) = \frac{G_\mathrm{u}(s)}{1 + G_\mathrm{u}(s)} \tag{6.13}$$

The equation of motion of a second order system can generally be written in the form $A\ddot{x} + B\dot{x} + C = 0$, where x is the motion variable for translation or rotation, A is, for translational motion, the mass m and, for rotational motion, the inertia I, B is the factor of the rate depending term, and C is a constant gain representing the 'spring stiffness' of the system. The closed loop transfer function in a second order system can be expressed accordingly as

$$T(s) = \frac{C}{As^2 + Bs + C} \tag{6.14}$$

The *natural frequency* of the loop is given by

$$\omega_\mathrm{n} = \sqrt{\frac{C}{A}} \tag{6.15}$$

In our reference case of a simple second order feedback system, the value of ω_b is very close to the *natural frequency*, ω_n, of the system (see figure 6.9), so that in calculations the bandwidth is often taken for the resonant frequency.

The *damping ratio* is given by

$$d = \frac{B}{2\sqrt{CA}} \tag{6.16}$$

The amplitude ratio (magnitude) of the *resonance peak* in figure 6.9 is determined by the damping ratio d.

With Eqs. (6.15) and (6.16) the closed loop transfer function Eq. (6.14) can be written in a form which reveals the dynamic behaviour of the system,

$$T(s) = \frac{\omega_{\mathrm{n}}^2}{s^2 + 2d\omega_{\mathrm{n}}s + \omega_{\mathrm{n}}^2} \tag{6.17}$$

The transient response can be obtained from the roots of the denominator of Eq. (6.17) (characteristic equation). For $d = 1$ the system is critically damped; for $d = 0$ the system is undamped.

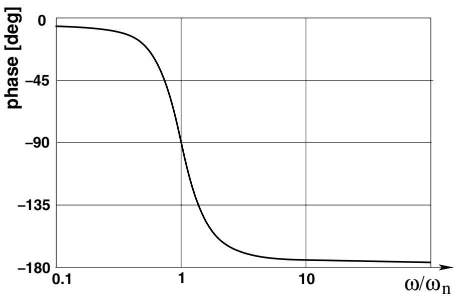

Figure 6.11. Phase shift over frequency.

As not only the amplitude response but also the phase response changes with the signal frequency, a feedback system can become unstable. The phase shift of the output signal due to the dynamic behaviour of the control loop is shown for a second order system in figure 6.11. Due to the fact that at frequency ratios $\gg 1$ the phase shift approaches -180 deg, an oscillation could in fact be amplified instead of damped if the amplitude of the feedback signal remained large enough. The assurance of stability is one of the major objectives of control analysis, and the according methods are well described in the literature, e.g. Anand (1984), Franklin *et al.* (1994) and D'Azzo (1995). For the purpose of this section, which will explain the effects and constraints of control loop features on the performance of trajectory and attitude control, these methods do not need to be discussed in detail. We can assume here that, for a control system implemented in a spacecraft, stability has been assured in the design and development process. With regard to the performance of a control loop, what has to be kept in mind is that stability requirements may put constraints on the achievable values for bandwidth, steady state error, etc.

The *steady state error* is defined as the difference between the reference signal and the output after settlement of transients:

$$e_{\mathrm{ss}}(t) = r(t) - y(t) \quad \text{for} \quad t \to \infty \quad \text{i.e.} \quad s \to 0$$

$$e_{ss}(t) = \lim_{s\to 0} s \left[\frac{r(s)}{1 + G_u(s)} \right] \tag{6.18}$$

For a *position step* change at $t = 0$, e.g., the reference signal for all $t > 0$ is $r(t) = A$. The Laplace transform of the position step input is $r(s) = A/s$, with which the position steady state error of our unity feedback system becomes

$$e_{ss}(t) = \lim_{s\to 0} \left[\frac{A}{1 + G_u(s)} \right] \tag{6.19}$$

$\lim_{s\to 0} G_u(s)$ is the zero-frequency gain of the open loop transfer function.

A velocity step is a ramp input $r(t) = At$, which has the Laplace transform $r(s) = A/s^2$, and an acceleration step is a parabolic input $r(t) = At^2$, which has the Laplace transform $r(s) = A/s^3$, for which the steady state error can be derived accordingly.

The steady state error is thus proportional to a fraction with the zero-frequency open loop gain in the denominator, i.e. the larger the open loop gain, the smaller the steady state error e_{ss}. The steady state error can also be decreased by including an integrating behaviour in the feedback loop (integral feedback), at the expense, however, of increasing the transient response errors.

In the discussion of the control loop behaviour it is convenient to introduce an additional function, the so-called *sensitivity function* $S(s)$, which is the quotient of closed loop and open loop transfer function. For the unity feedback system of figure 6.10 this is

$$S(s) = \frac{1}{1 + G_u(s)} \tag{6.20}$$

The sum of the closed loop transfer function and the sensitivity function functions fulfils the relation

$$T(s) + S(s) = 1 \tag{6.21}$$

The sensitivity function indicates the advantage of close loop control: the error of a controlled parameter due to variations in the open loop gain $K(s)G(s)$ (e.g. by an input disturbance) is, in the case of closed loop control, by a factor of the 'sensitivity function' lower than in the case of open loop control.

General objectives of controller design

To simplify the discussion of the behaviour of our reference control loop, the reference block diagram figure 6.8 will now be transformed into a unity feedback system. This may be done as long as the bandwidth of the measurement system $M(s)$ is much higher than the closed loop control bandwidth, which is a desirable condition for the control loop design.[2] If this requirement is fulfilled, the transfer function $M(s)$ may be modelled

[2]This requires that the sensor has a sufficiently high bandwidth, which may not always be the case.

by a simple gain, which can be regarded to be resident within the controller $K(s)$ or within the plant $G(s)$. The closed loop transfer function Eq. (6.12) then becomes

$$T(s) = \frac{K(s)G(s)}{1 + K(s)G(s)} \tag{6.22}$$

and the sensitivity function becomes

$$S(s) = \frac{1}{1 + K(s)G(s)} \tag{6.23}$$

Making these assumptions, we can now assess the influence of the various inputs or disturbances shown in figure 6.8. The system output is driven by four input signals: the reference signal r, the input disturbances w, the output disturbances v and the sensor disturbances n (measurement environment $o(s)$ is included in sensor errors $n(s)$). It follows directly from figure 6.8 that the *system output* is

$$y(s) = T(s)r(s) + G(s)\frac{1}{1 + K(s)G(s)}w(s) + \frac{1}{1 + K(s)G(s)}v(s) - T(s)n(s)$$

which can be rewritten using the definition of $S(s)$, Eq. (6.23), as

$$y(s) = T(s)[r(s) - n(s)] + S(s)[G(s)w(s) + v(s)] \tag{6.24}$$

The *controller output* is

$$u(s) = \frac{K(s) \cdot [r(s) - n(s) - v(s) - G(s)w(s)]}{1 + K(s)G(s)}$$

which can be reduced to

$$u(s) = \frac{T(s)}{G(s)}[r(s) - n(s) - v(s) - G(s)w(s)] \tag{6.25}$$

From Eqs. (6.24) and (6.25), the following conclusions may be drawn for the control loop:

- Sensor disturbances are amplified by the closed loop gain $T(s)$ in the same way as the reference signal, i.e. there is no reduction of sensor errors/disturbances due to the feedback.
- Input and output disturbances are reduced due to the sensitivity function $S(s)$. As a result, tracking and disturbance rejection requires that the sensitivity function becomes small in the frequency domain of the disturbances, i.e. in the low and medium frequency range.
- To keep propellant consumption low, the energy of the control output $u(s)$ needs to be small, which requires the closed loop transfer function $T(s)$ to be small, at least where the transfer function of the plant, $G(s)$, is small.

- If the sensitivity function becomes small, $S(s) \ll 1$, the plant disturbances $w(s)$ and $v(s)$ are highly damped, i.e. the system has a large disturbance rejection capability. In this case, however, $T(s)$, according to Eq. (6.21), becomes very large and the control loop thus becomes very sensitive to the measurement noise $n(s)$ (cf. section 6.2.1, where one of the major objectives of the navigation filter was to reduce sensor noise).
- It should be noted that the system reference inputs, as well as the system disturbances, are generally in the low frequency range, whereas measurement noise lies in the high frequency range. The overall design guideline is, therefore, to achieve a small $T(s)$ for a high frequency range and a small $S(s)$ for the low frequency range. Consequently, the open loop transfer function $K(s)G(s)$ should be large for low frequencies and small for high frequencies.

As an '*a priori*' definition (setting) of the closed loop transfer function $T(s)$ may result in a non-realisable controller $K(s)$, the controller is therefore often designed via the open loop transfer function $K(s)G(s)$, for which specific characteristics may be required.

- The steady state error of, e.g., position, velocities and attitude angles may be required to not exceed certain limits. As a consequence, a requirement for a low steady state error for these parameters will result in a requirement for a high gain at low frequencies.
- Concerning the response time, the bandwidth must be selected such that the required frequency content of the guidance profiles and of the sensing function output can be tracked and that the disturbances are rejected with an acceptable damping ratio.
- Sufficient gain and phase margin is required for stability of the loop.

In conclusion, the controller will be designed such that it achieves sufficient performance in tracking the reference signal $r(s)$, without being too sensitive to measurement noise $n(s)$ and plant disturbances $v(s)$ and $w(s)$. Furthermore, the controller needs to be sufficiently robust w.r.t. plant parameter uncertainties, such as mass and inertia, and the control signal $u(s)$ should be kept small to minimise the propellant consumption.

Modelling of plant and disturbances

The description of the plant in the control loop design needs to include dynamic and kinematic models (see figure 10.10); the dynamic models are derived from the equations of motion and the kinematic models are derived from the spacecraft design. The modelling of input disturbances $w(s)$, such as air drag, gravity gradient, J_2 and sloshing effects, which are directly related to the dynamic and kinematic characteristics of the spacecraft, will need the spacecraft dynamic and kinematic models as input. These models will be needed in the design analysis of the control loop but are not necessarily part of the controller. To the extent that these disturbances are predictable, they can be compensated for in the guidance function. The modelling of dynamics and disturbance is addressed again in section 10.4.1 concerning the verification of the onboard system.

The major sources of disturbance for rendezvous trajectories have been addressed in section 4.2.

As already stated, the relative translational motion dynamics of one space vehicle w.r.t. another can be described by the Hill equations (3.21). For orbital motion, it was shown in section 3.3 that in-plane motion (x,z) and out-of-plane motion (y) are decoupled. The latter is an undamped oscillation with the orbital frequency. The in-plane coupling terms correspond to Coriolis and gravity forces in the Clohessy–Wiltshire solution, Eqs. (3.22) (see appendix A). For a closed loop controller design, these terms may be considered as low frequency disturbances and, for this reason, no longer need to appear in the design model. Consequently, a double integrator model, independent for each axis, can be applied for the position controller design. For the rotational motion of a three axis stabilised, Earth pointing satellite, Euler's moment equations can be linearised (Kaplan 1976) (see appendix A), so that for rotational motion basically a double integrator dynamic model can also be used for the controller design.

With this result, once damping d and closed loop bandwidth ω_{b} have been selected, a simple PD controller can directly be defined by

$$K(s) = K_{\mathrm{P}} + K_{\mathrm{D}} \cdot s$$

where for translational motion (position control) $K_{\mathrm{P}} = m \cdot \omega_{\mathrm{b}}^2$; for rotational motion (attitude control) $K_{\mathrm{P}} = I \cdot \omega_{\mathrm{b}}^2$; and the damping term $K_{\mathrm{D}} = 2d\sqrt{K_{\mathrm{P}}}$ (see also Eqs. (6.14)–(6.17)). For more detailed analysis, e.g. for stability analysis, the neglected coupling terms have to be taken into account.

In the phases prior to the final approach (see figure 2.1) position and attitude control is related to the CoM of the vehicle. This requires proper kinematic modelling of, e.g., the location w.r.t. the position of the CoM of the vehicle. The calculation of the actual position of the CoM has been assumed in section 6.2.2 to be the task of the guidance function. In the last part of an approach to docking, position and attitude of the chaser's docking port have to be aligned with that of the target (see figure 6.12). As a result, the vehicle has to be controlled w.r.t. its docking frame, which requires modelling of the geometric relations between the docking frame and the spacecraft attitude frame, which has its origin in the (moving) CoM. The corresponding kinematic model describing the motion coupling, resulting from location and direction of view of the sensors, is part of the navigation filter (see box labelled 'preparation of measurement vector' in figure 6.3).

Generally, the use of proper mathematical modelling of the disturbances and the system to be controlled will improve the performance of the control function. It is clear, however, that the real world cannot be described fully deterministically. The choice of the model fidelity therefore determines the quality of results, not only in the guidance but also in the control algorithms. This choice, i.e. whether the model is more or less accurate, most often also determines the complexity of the model and the computer effort for its processing. For this reason, there may be a limit in the degree of accuracy in the modelling of certain functions and effects.

Particular control issues in rendezvous missions

The performance requirements for the reduction of trajectory errors, i.e. the steady state errors for position and approach velocity, increase with decreasing range to the target. They will be particularly stringent in the last part of the approach, when entering the docking reception range. A short overview of the typical performances required is given in section 7.1, where sensor requirements are discussed.

For the coarse approach up to several hundred metres toward the target, orbital arc transfers are used (see sections 3.3.2, 3.3.3 and 5.7). If an impulsive transfer is used, boost and free flight phases have to be distinguished. For the rendezvous operations proper, which start in the vicinity of the target (typically 30 to 50 km), the necessary velocity increments ΔV per boost are rather small compared with the orbit transfer boosts needed during phasing to arrive at the target orbit. The small ΔV requirements either lead to large accelerations and short burn durations, when using large thrusters (e.g. the ones used during phasing), or to small accelerations and correspondingly long burn times, when using small thrusters (e.g. the ones for attitude control). In the first case, the ΔV errors may become larger and control performance may be lower; in the second case the burn time will, for the main boosts, cover a larger portion of the orbital arc. These considerations will be some of the driving factors in the selection of thrusters for a rendezvous spacecraft to be developed.

Open loop boost manoeuvres At larger distances from the target, trajectory manoeuvres can be performed with sufficient accuracy in open loop, i.e. the thrusters are fired for a duration calculated from the expected acceleration level and the required velocity increment. This will particularly be the case in the far range rendezvous phases, where accuracy requirements are still moderate. A major error contributor for the open loop trajectory implementation is the propulsion system itself (see also section 4.3.2).

In the free flight phase between the boosts, the errors made in the implementation of the first boost and the perturbations, which are not covered by the guidance function, can be observed and corrected open loop by mid-course manoeuvres.

Closed loop control of long boost manoeuvres Where longer boost duration is needed in the case of small thrusters, performance improvement can be achieved by closed loop control, using a high bandwidth velocity controller. For such a scheme, the navigation function (sensors and navigation filter) must be able to provide the state information with sufficiently high bandwidth.

It should be noted here that specific attention must be paid to the attitude control during the boost. Small centre of mass uncertainties and differences in the individual thrust levels can create significant disturbance torques, which have to be properly counteracted to maintain the correct orientation of the velocity increment (see also the ‘Thrust direction errors’ subsection in section 4.3.2). This requires a relatively high bandwidth controller, which may challenge the assumption of a rigid body dynamics for cases where solar arrays or other flexible appendages are attached to the spacecraft.

Closed loop control between boost manoeuvres This case has been discussed in view of trajectory safety in section 4.4.1. During the free flight between manoeuvres, perturbations are caused by the air drag and other disturbances. As far as they can be predicted, they should be taken care of by the guidance function. The residual disturbances of the control loop are the unknown and neglected terms in the applied guidance model. Both are small and of low frequency, so that a low bandwidth controller will be sufficient, if closed loop control is applied at all during free flight phases. In fact, in this case the use of a controller is related to the correction of trajectory errors, caused by the preceding boost (navigation errors and thrust errors), or to the implementation of an active trajectory safety scheme, rather than to the compensation of the disturbance effects. Consequently the often applied alternative to closed loop control is the introduction of open loop correction boosts (mid-course manoeuvres), described above.

An alternative orbital arc transfer is based on low but continuous acceleration during the transfer (see section 3.3.3). Although this generally leads to a doubling of the transfer duration, it is well suited to a low bandwidth closed loop control concept, since the disturbances are small and constant.

Closed loop controlled straight line trajectories While approaching the target, the accuracy requirements in position and velocity control increase. Open loop techniques are no longer usable below a certain range. Constraints due to the angular operating range of optical sensors (visibility of target reflector pattern, see section 5.3.2) and due to approach corridors, defined by the target station (see section 5.6), require the spacecraft to stay on pre-defined trajectories.

For closed loop controlled trajectories, the required controller bandwidth will depend on the reference trajectory and velocities according to the guidance profile, on the performance requirements (response time, accuracy, consumption) and on the disturbances. The bandwidth of the necessary tracking system must be, on one hand, high enough to reach the required accuracy and, on the other hand, small enough not to waste fuel. The latter property can be improved when an appropriate dead-band is introduced in the closed loop.

Last metres approach to docking In the last part of the approach, an important driver for the controller design is the requirement that the chaser must be able to follow the motion of the docking port of the target. In the case of target thruster firings and of motions due to structural flexibility of the target during the final approach of the chaser, the frequency content of these motions may lead to higher bandwidth requirements than in the other approach phases. Although steady state errors of position and angular alignment may fit into the reception range of the docking mechanism, in case of (e.g.) rapid target motions, transient errors may be larger. The transient response of the controller must be such that, for the expected frequency content of such motions, the sum of the instantaneous values of lateral position and angular misalignments between the docking interfaces of chaser and target will be smaller than the reception range (see figure 8.30).

As the motion between the coordinate frames of the docking interfaces of chaser and target has to be controlled in this phase, coupling of rotational and translational motion has to be considered, as indicated above. Any attitude motion of chaser or target will result both in a change of relative attitude and relative lateral positions of the docking frames of chaser and target.

At contact, the control system of the chaser must have achieved lateral position and angular alignment with the target docking mechanism, such that the residual errors fit into the reception range of the docking interfaces. To explain the control problem of the last phase prior to contact, three concepts of approach control shall be considered, where the chaser GNC system tracks different parameters of the target state:

(a) the position of the target CoM,
(b) the position of the target docking port,
(c) the position and the relative attitude of the target docking port.

Three different cases are shown in figure 6.12, in which the bold arrows represent the vehicle attitude frames, and the length of the x-component indicates the distance of the docking port from the CoM. The attitude control margins are shown on the target side, the position control margins on the chaser side. The maximum possible lateral and angular misalignments are indicated for each case above the figure.

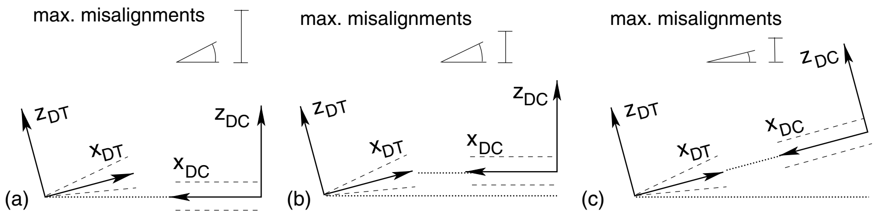

Figure 6.12. Control concepts for docking. (a) Tracking of target CoM, chaser attitude nominal. (b) Tracking of target docking port, chaser attitude nominal. (c) Tracking of target docking port and of chaser to target relative attitude.

Figure 6.12(a) represents the most simple control scheme, i.e. instead of controlling relative position and orientation of the docking ports, only the relative motion of the centres of mass and the absolute attitude of the vehicles are controlled, as in the previous approach phases. The aim point is the nominal, not the actual, location of the centre of the target docking port. The systematic alignment error at the docking interfaces is the sum of the angular error due to the target attitude deviation and the lateral error due to the distance of the docking port from the target CoM.

In figure 6.12(b), the lateral position of the chaser docking port is controlled w.r.t. the target one. A line of sight measurement to the target docking port is necessary. The attitude of each vehicle is controlled independently w.r.t. a nominal reference frame (e.g.

LVLH). The systematic alignment error at the docking interfaces is just the angular error due to attitude deviations between chaser and target.

In figure 6.12(c), both the relative lateral position and the relative attitude have to be controlled simultaneously to achieve full translational and rotational alignment of the docking interfaces. This needs, in addition to the lateral position, an onboard estimation of the relative orientation of the docking axes of chaser and target. A consequence of this scheme is that rotational and translational motions are now coupled, which has important repercussions on the controller, as it requires MIMO design.

The third case is the only one without systematic alignment errors. Alignment errors due to residual attitude and position control errors will be the smallest of the three cases. In RVD missions only cases shown in figure 6.12 (b) and (c) are actually used. In the following 'case (a)' will refer to figure 6.12(a), etc.

- *Case (a)* has never been applied, as generally it is not only too inaccurate as compared with realistic reception ranges of docking mechanism, but also it is not practical, as there is no sufficiently accurate sensing capability available to measure the relative position of the CoMs of chaser and target.
- *Case (b)* has been applied in some docking missions, whereby the position of the chaser has been controlled manually and the attitude of each vehicle automatically by its respective attitude control system. This option still requires a relatively large angular reception range of the docking interfaces, due to the steady state and transient errors of the individual attitude control systems (bang-bang control in most cases).
- *Case (c)* is applied in all automatic GNC systems for rendezvous and docking. In this case relative position and attitude measurements are processed in the navigation function to obtain an estimated six degrees of freedom relative state vector, the components of which have to be controlled simultaneously and will require the application of advanced controller design techniques. Case (c) is now also the standard concept for manually controlled approaches to docking (see section 6.5.3). Lateral position and relative attitude information can be obtained by the human operator, if a video camera and a suitable target pattern (see figure 6.25) is available. The nominal attitude will then be adjusted by commanding appropriate offsets.

During most of an RVD mission, the attitude control and the position control systems can be considered as de-coupled, except for the final approach to docking (case (c)). Furthermore, accuracies are, in these phases, less stringent than at close range. It shall nevertheless not be forgotten that for the position there are also the low frequency couplings for the in-plane motion of the Clohessy–Wiltshire equations, mentioned already under the 'Modelling of plant and of disturbances' section above (see also Eq. (3.22), for the 2ω terms).

These cross couplings can be taken into account during either the control design or as feed forward terms from the guidance part of the system in order to reduce transient errors, increase precision and reduce fuel consumption. For this part of the GNC design, classical PID type of designs will often suffice, but when low damped flexible modes are present in the system, more advanced designs are needed. This can be a classical control design combined with separately tuned notch filters, but a more elegant and efficient design is to apply H_∞ design, which is well suited for such problems.

During the last metres of the final approach, the position and attitude angles and the translational velocities and angular rates are coupled, as a result of the control w.r.t. the target docking port (case (c)), and need to be controlled simultaneously. Such a control problem typically calls for a multi-variable design method. Owing to the nature of the system, the flexible modes to be controlled and the way in which the disturbances are described, a multi-variable robust design method is recommended for the synthesis, as well as for the analysis, part of the design.

As already discussed above, the actuators for this type of vehicle are usually thrusters, and their non-linear characteristics (see figure 6.15) require special attention during the stability analysis. For the part where SISO designs are applicable, the use of the negative inverse describing function is recommended, and for the MIMO methods an appropriate frequency domain weighting function should be chosen.

Discrete time control

Since onboard control systems are nowadays always implemented in software on digital computers, we are in fact dealing with discrete computer controlled systems rather than with analogue ones, assumed in Eqs. (6.12)–(6.25). Proper use of the discrete domain theories has, therefore, to be made in the entire design process. The type of systems we are dealing with here are sampled data systems with a fixed sampling time T for the feedback loop, in contrast to some interrupt driven free running discrete systems.

A sampled data system is described by difference equations, and a signal is described by a number sequence. These number sequences are obtained by sampling a continuous or analogue signal. An ‘analogue to digital converter’ (ADC) converts the analogue sensor signal to a digital signal, and a ‘digital to analogue converter’ (DAC) converts the digital controller output, which is fed to the actuators in the system. One can either view the real world as analogue, and the discrete the system as a special case, or, more conveniently, one can view the world as discrete and the continuous part as the special system; the latter is recommended. An analogue and an equivalent discrete control loop are illustrated in figure 6.13.

The sampling of the continuous signal must be performed at a sufficiently high frequency such that not too much phase margin is lost. The Nyquist frequency, which is twice the highest frequency in the signal, is the theoretical minimum. For practical purposes this is too slow, and one needs to sample at a frequency which is seven to ten times the fastest mode in the closed loop system. In this context, one also has to pay attention to aliasing of high frequency noise from sensors, which could show up as warped low frequency discrete signals. To avoid this, analogue anti-aliasing filters might need to be applied.

The output from the discrete system to the continuous one is typically through the DAC as well as a zero order hold (ZOH) network, which keeps the output signal constant over one sample.

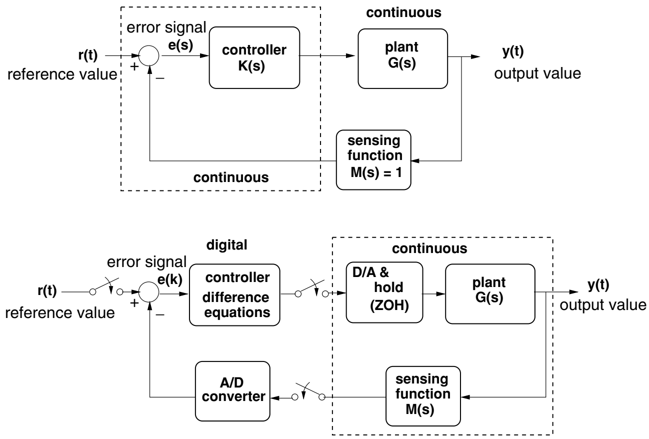

Figure 6.13. Continuous and digital control loop.

As a tool for the analysis of discrete systems (equivalent to the Laplace transform for the continuous domain), the $\mathcal{Z}$-transformation is available; it is defined as follows:

$$y(z) \triangleq \mathcal{Z}[y(k)] = \sum_{k=0}^{\infty} y(k) z^{-k} \tag{6.26}$$

The variable z can be viewed as a shift operator or as a complex variable in the z-plane, just as we view s as a differential operator or as a complex variable in the Laplace plane.

It should be noted that $y(z)$ often can be obtained from a known Laplace transform description, without going via the impulse response function $y(k)$ and the summation as given in the definition equation (6.26). This is typically performed by various approximations of the complex variable s for Euler forward, backward or bilinear transformations. These are approximations to the $\mathcal{Z}$-transformation, and will lead to a need for higher sampling frequencies followed by bigger computer load. A better method is to use the *pole-zero mapping*, where

$$z = \mathrm{e}^{sT} \tag{6.27}$$

and s is the Laplace variable and T is the sampling time. This leads to the smallest sampling frequencies and it is also mathematically exact. This means that at the sampling times the discrete signal is exactly equal to the continuous one and corresponds to the $\mathcal{Z}$-transformation.

The stability for discrete systems can be evaluated with respect to the unit circle, as we know from the conformal mapping that the left half-plane of the Laplace domain maps onto the inner part of a unit circle in the z-plane. Therefore systems having closed loop poles inside the circle are stable.

The best performance, lowest sampling rates and better stability margins will be achieved when designing discrete GNC systems directly in the discrete domain, i.e. transforming the plant into that domain. This will give better results than implementing first a continuous design and thereafter designing the controllers with some approximate transformations.

As a detailed discussion of digital control theory would be outside the objectives and scope of this chapter, the interested reader is referred to the literature, e.g. Franklin & Emami-Naeini (1990).

The thruster selection function

The task of the thruster management function is to translate the force and torque commands generated by the control function into 'on/off' commands for the individual thrusters according to their direction and to their location w.r.t. the momentary centre of mass of the vehicle. Depending upon the control error, a request for a force of varying level to be applied along the individual axes is made. The position controller generates force commands, and the attitude controller generates torque commands, of varying amplitude along and around the body axes. With the thruster hardware available for the realisation of such requests, two problems exist:

(1) The thruster either provides the nominal force or nothing, i.e. it is not possible to change the force level from zero to a maximum value.
(2) Due to thruster accommodation and redundancy constraints, dedicated thrusters for the individual controller outputs are most often not available, i.e. generally there is coupling between forces and torques, which has to be accounted for properly.

One solution to the first type of problem is the application of non-linear controllers with switching elements. For thruster control, pulse width pulse frequency (PWPF) modulation is usually applied (Noges & Frank 1975) (see figure 6.5).

During rendezvous, the sample interval for the discrete controller (typically 1 s) is large compared with the minimum pulse length of a thruster. Consequently, within the control cycle, the pulse length can be varied from zero to the control sample interval. The realised impulse is approximately a linear function of the pulse length. Therefore, as a first order approximation, the effect corresponds to an amplitude modulation of the force within the control cycle, where the average force is (see figure 6.14)

$$F_{\text{average}} = F_{\text{nom}} \frac{t_{\text{pulse}}}{t_{k+1} - t_k} \tag{6.28}$$

The output of the controller corresponds to F_{average}, and the command to the thruster control electronics is the pulse length t_{pulse}.

The pulses cannot of course be made arbitrarily small. The lower limit is the 'minimum impulse bit'. For a bi-liquid thruster of 200 N, for example, the MIB is in the order of 50 ms. Moreover, for small impulses, the steady state thrust level is not yet reached. Consequently, small force (or torque) requests are suppressed by a dead zone at the output of the controller.

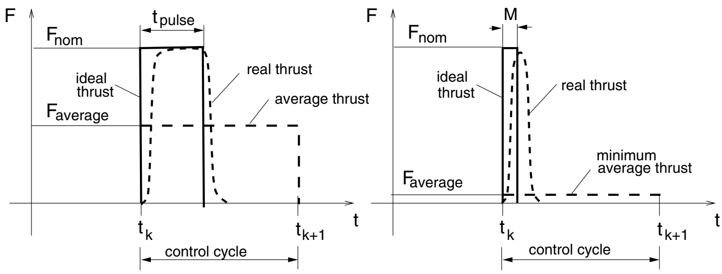

Figure 6.14. Thrust pulse characteristics.

When the thruster is operated close to the MIB, the simple equation (6.28) above may be modified by a calibration factor C_F, providing an appropriate correction as a function of the force request. We then have

$$t_\mathrm{cmd} = C_\mathrm{F} \frac{F_\mathrm{ctrl}}{F_\mathrm{nom}} (t_{k+1} - t_k) \tag{6.29}$$

Including the dead zone, the shape of the correction factor is shown in figure 6.15. The curve above the dashed level 1 may be approximated by an exponential function.

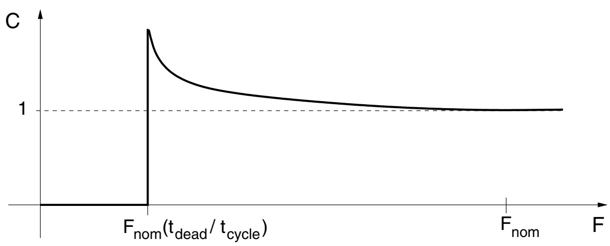

Figure 6.15. Control force correction factor as a function of nominal thrust.

The MIB of a thruster is generally a function of its maximum thrust. This may create a problem when the same set of thrusters has to be used both for larger thrusts, e.g. for transfer manoeuvres, and for fine control, e.g. for the last metres to docking or, e.g., for minimisation of the residual velocities in a berthing box. The minimum average force which can be applied during a control cycle will then be a constraint on the achievable GNC performance.

The coupling problem addressed under point (2) above may be treated by two different principles, depending on the complexity of coupling and on the envisaged thruster redundancy concept. If the accommodation can be kept close to the ideal configuration, where dedicated thrusters are used in the individual axes for force and torque requests, then so-called 'look-up tables' may be used. These tables allocate the thrusters to be applied for specific force/torque requests. The actual commands are then composed of a scaled combination of the reference points.

The tables are only valid, however, for a specific thruster accommodation with a fixed CoM, and provide only sub-optimal solutions w.r.t. both fuel consumption and accuracy. However, this approach is rather fast and invokes only small CPU load. Since spacecraft systems have to be failure tolerant, a dedicated table is needed for any possible propulsion configuration, i.e. for any thruster (or thruster cluster) failure. Consequently, a large number of tables needs to be defined. If, furthermore, the CoM is subject to large variations, as is the case for a space station servicing vehicle during the various approach phases, even more tables are needed.

This problem can be reduced, when the optimal set of thrusters and the appropriate thruster-open duration are determined onboard. For this, the actual CoM position must be known precisely, which remains one of the major sources of errors. This task corresponds to a linear optimisation problem, which can be solved by a standard mathematical procedure such as the *Simplex* algorithm (see Press *et al.* (1992)). This algorithm works on a tableau representing the constraints on the optimisation variables, defined by linear equalities and the objective function. The latter contains the contribution of each free variable to the optimisation result.

The following problem has to be solved for the selection of thrusters:

$$\begin{bmatrix} f_{1x} & f_{2x} & \cdots & f_{nx} \\ f_{1y} & f_{2y} & \cdots & f_{ny} \\ f_{1z} & f_{2z} & \cdots & f_{nz} \\ t_{1x} & t_{2x} & \cdots & t_{nx} \\ t_{1y} & t_{2y} & \cdots & t_{ny} \\ t_{1z} & t_{2z} & \cdots & t_{nz} \end{bmatrix} \cdot \begin{bmatrix} u_1 \\ u_2 \\ \vdots \\ u_n \end{bmatrix} = \alpha \begin{bmatrix} F_x \\ F_y \\ F_z \\ T_x \\ T_y \\ T_z \end{bmatrix} \tag{6.30}$$

or

$$\mathbf{A} \cdot \mathbf{B} = \alpha \cdot \mathbf{C}$$

where

f is the x,y,z-force component of thruster 'n',
t is the x,y,z-torque component of thruster 'n',
n is the number of the available or usable thruster,
u is the normalized thruster open duration, $|u_i| \leq 1$.

The maximum value of u is limited to 1, i.e. the thruster is opened for the complete control cycle. A zero value for u_i means that the thruster is not used at all. The factor α allows us to account for thruster saturation. When the commanded force/torque control vector $\mathbf{C}$ cannot be realised with the available thruster, then at least the commanded direction will be kept, i.e. the $\mathbf{C}$-vector is appropriately scaled.

The optimisation criteria is the minimisation of the fuel consumption, while simultaneously maximising the scaling factor α. The cost function then can be given as

$$Z = -(c_1 u_1 + c_2 u_2 + \cdots + c_n u_n) + c_\alpha \alpha \implies \max. \tag{6.31}$$

where c_i is a coefficient for propellant consumption, and c_α is a weighting factor for the realisation.

Nominally, the simplex algorithm always finds a solution. Theoretically, however, the algorithm may enter the so-called ‘circling’ régime, i.e. the algorithm doesn’t converge because a constantly repeated exchange series within the simplex tableau occurs. This rarely happens, but if it does it can be avoided, e.g. by enforcing the iteration loop to stop after a pre-defined number of iterations, or, in other words, accepting a potentially non-optimal solution.

Due to the optimisation scheme, the CPU load is significant, in particular when numerous thrusters ($>$20) are to be treated simultaneously. It provides, however, very high inherent flexibility with respect to thruster failures and CoM locations. The only things to be changed are the CoM location and a set of flags indicating whether or not a particular thruster can be used.

Since the processing power required for spacecraft computers is increasing over time (though much slower than in commercial products on Earth), the CPU load may become of lesser importance in the future. The inherent flexibility of this algorithm will then make its application interesting for a larger number of spacecraft, and it may be implemented in future, e.g. as a general service for thruster control.

6.3 Mode sequencing and equipment engagement

Because of the many different manoeuvres, trajectories and attitudes and the different sensors used in the various steps of the rendezvous approach, there will be, for each approach step, a different set of software functions (algorithm, parameter) and hardware functions (sensor), as explained above. For this reason, there must be a management function which activates for each new step the proper software modes for guidance, navigation and control (phase/mode management) and the sensor configuration (vehicle management) at the proper point in time, when a new manoeuvre or trajectory element has to commence. In addition, there will be the need for redundancy management of hardware functions, such as processors, data buses, sensors and reaction control equipment for contingency cases. All these functions can be grouped together under the term ‘mission and vehicle management’.

In single satellite missions, only the redundancy management of critical hardware needs to be implemented as an onboard function; all other management functions can be performed by ground operators. In the case of a rendezvous mission of an unmanned vehicle many of these management functions are better performed on board. This is because of the possibility of interrupted communication with ground and the sensitivity of the trajectories to time delays of the manoeuvres. In particular it is the case for the rendezvous phases proper, i.e. when the chaser is close enough to the target to perform relative navigation. The automatic mission and vehicle management functions are, therefore, key elements in the capability to perform automated rendezvous and capture.

The major task for the phase/mode management function is to determine the proper point in time when a new GNC mode has to be initiated. For this purpose both time and spacecraft state (position and attitude) criteria have to be checked to ensure that the intended state prior to initiation of the next trajectory element has been achieved. Except for the last phase of the approach, all trajectory elements last for a certain part of an orbital revolution (typically of the order of half or one orbital revolution), which can be pre-calculated according to the type of manoeuvre and the necessary thrust duration of the boosts to be applied. Considering the time as the only criterion will be, however, not sufficient, as e.g. thrust errors or failures may result in a position that is unsuitable to start the next approach step. In the same way, position or other spacecraft state parameters alone will not be sufficient, as e.g. the vehicle may (because of thrust errors) never reach the position that the criteria is requesting. Only if both time and the state parameter are within a certain margin can the next step can be initiated. Otherwise a contingency strategy has to be applied.

Figure 6.16 shows a flow diagram of nominal and contingency modes for the approach strategy given in the first example of section 5.7. From each nominal mode there are three possible transitions: (a) next mode, (b) mission interrupt, (c) mission abort. As explained in chapter 4, the mission abort (CAM) can be implemented as a simple retrograde boost, which may be, however, of different sizes in the different approach phases. The mission interrupt will consist of different guidance modes according to the approach phase. In the chosen example (see figure 5.25), mission interrupt modes in the rendezvous phases would be as follows.

- *Drift phase from S0–S1*. Continuation of drift after passing S1, if first pulse of homing manoeuvre cannot be executed because of onboard problems, or if it is not executed for reasons not related to the rendezvous onboard control system.
- *Homing S1–S2*. Either transition to next mode 'hold at S2', if mission has to be interrupted for reasons not related to the rendezvous onboard control system, or continuation of drift after passing S2, if second pulse of homing manoeuvre cannot be executed.
- *Hold point S2*. Tangential pulse to initiate a retrograde motion, looping away from the target. This pulse must be small enough to avoid the first part of the resulting trajectory entering the approach ellipsoid. By a further boost the motion could be stopped to acquire a new hold point.
- *Closing S2–S3*. Transition to next mode 'hold at S3' if mission has to be interrupted for reasons not related to the rendezvous onboard control system or, if the second pulse of the closing manoeuvre cannot be executed, continuation of drift after passing S3. In this case a second manoeuvre would have to be performed after half an orbit in order to stop the vehicle at S2, preventing it from moving again toward the target.

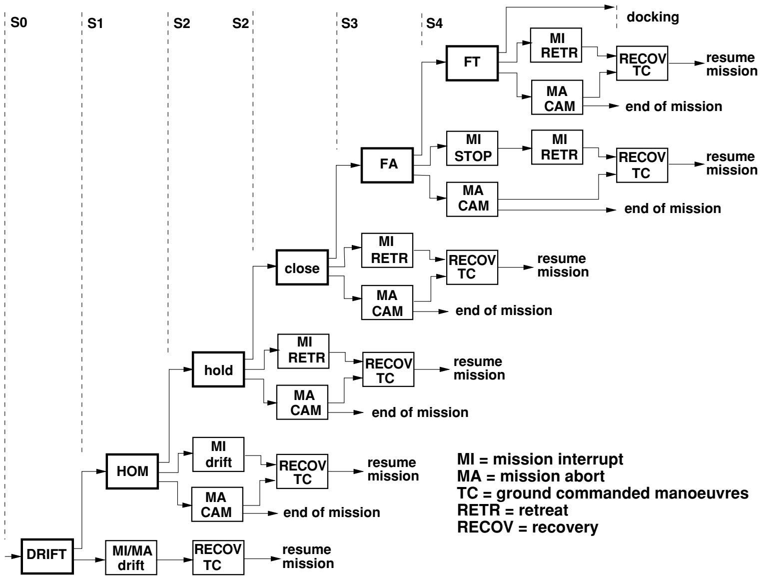

Figure 6.16. Sequencing of nominal and contingency modes.

- *Hold point S3*. Radial pulse manoeuvre to return to the previous hold point S2. Again, in this case a second manoeuvre would have to be performed after half an orbit in order to prevent the vehicle from moving again toward the target.
- *Final approach S3–S4*. Stop and hold on V-bar. In contrast to the previous modes, during the straight line approach the vehicle can be stopped at each point of the trajectory at the same cost. This braking manoeuvre may be followed by a retreat to the previous hold point S3.
- *Final translation to contact*. Retreat to S4. In this case stop and hold in very close vicinity of the target would be considered too hazardous.

Except for cases when the mission is stopped at a hold point or on V-bar, recovery from mission interrupts or mission aborts requires the calculation of a series of different recovery manoeuvres. A re-planning of the mission may have to be performed for all interrupts of more than a few minutes, as synchronisation with illumination conditions and communication windows may be lost. All mission re-planning will have to be done on ground, as it will be impossible to pre-plan all potential recovery manoeuvres. The recovery manoeuvres will have to be established based on the actual situation, and then would have to be transmitted to the vehicle by telecommand.

There are of course many ways in which such a phase/mode management function can be implemented. In order to provide an idea of how such a function would operate, a simplified concept is shown as an example in figure 6.17. This concept would work as follows.

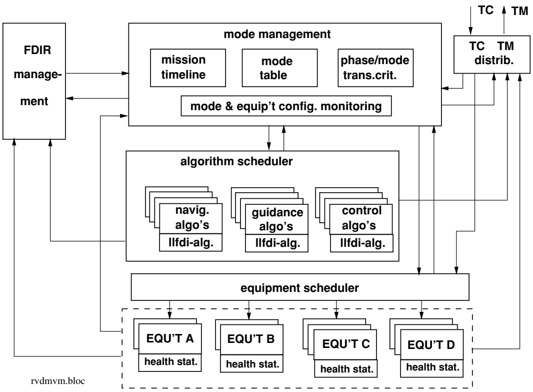

Figure 6.17. Mission and vehicle management functions.

(1) A ‘mission timeline’ function is implemented as a look-up table, which contains a schedule of the planned manoeuvres and trajectories.

(2) At a point in time when a pre-planned change of trajectory or attitude is due, the ‘mission timeline’ function requests a change of control mode (new manoeuvre, trajectory or attitude mode).

(3) Upon this request, the ‘mode management’ function fetches from the ‘mode transition criteria’ table the applicable criteria and checks whether the GNC data (navigation and guidance output) fulfil the criteria for a mode change.

(4) When the criteria are fulfilled, the new GNC mode will be loaded from the mode table and the corresponding commands will be transmitted to the ‘algorithm scheduler’ and the ‘equipment scheduler’.

(5) If the switching criteria are not fulfilled, a contingency mode, pre-determined for each phase, will be loaded and the corresponding commands will be issued to the ‘algorithm scheduler’ and the ‘equipment scheduler’.

(6) According to the commands from the ‘mode management’ function, the ‘algorithm scheduler’ fetches from the algorithm tables the different N, G and C algorithms belonging to the new mode, and the ‘equipment scheduler’ switches the relevant equipment combination.

The ‘mode management’ and ‘equipment scheduler’ functions in this concept would have to register the instantaneous configuration of software modes and of equipment, including their redundancy status, and would provide this information to the failure detection and isolation and recovery (FDIR) function. If the FDIR function detects a failure in equipment or the faulty behaviour of the complete string, it would request the ‘equipment scheduler’ function to execute the redundancy switching of sensor and actuator equipment. The mission timeline in the example concept would be a table containing the latest planning of the time sequence of all mission events. It would be loaded prior to launch and would include the latest mission updates, based on the planned launch time and the evolution of the target orbit. For smaller changes of the actual timeline, this table and related parameters, e.g. time criteria for mode switching, could be adapted automatically by the onboard system. For larger deviations, the timeline table could be modified by telecommand from ground.

In any kind of design, the 'mission and vehicle management' functions will have to interact closely with the FDIR function, as the examples of mode switching failure and equipment redundancy management already indicate. Possible methods for failure detection and recovery will be discussed in more detail in the following section.

6.4 Fault identification and recovery concepts

The intention of this section is to describe some basic principles concerning how fault tolerance and recovery concepts can be implemented in the space and ground segments of rendezvous systems in general and in the design of onboard control systems in particular. As the first necessary step toward recovery from a failure is its detection and identification, the first question to be answered must be, therefore, 'What are the observables, and what are the criteria by which failures can be identified?'. On the highest level it may be possible to detect the deviation of the actual state of the vehicle from the planned one, e.g. in terms of position, velocity or attitude. On a subsystem level it may be possible to detect the deviation of a number of parameters and conditions from the nominal ones. On the lowest level it may be possible to detect the proper functioning of equipment by checking output data characteristics, power consumption and other physical parameters. Accordingly, the FDIR system of a chaser spacecraft will have to provide failure detection capabilities at three different levels.

- At the lowest level, all equipment will have to be checked either by internal or external agents (built-in test functions, criteria for voltage, temperature, pressure, etc.).
- At the GNC subsystem level, a number of criteria have to be checked, which will be related to the GNC modes engaged and to the manoeuvres and trajectories to be executed.
- At the highest level, concerning the spacecraft state, the transgression of safety margins around the nominal values of position, velocity, attitude and angular rates have to be checked.

The general concept for onboard FDIR implementation will always be to start the recovery at the lowest possible (equipment) level. If failure detection and recovery is not possible at equipment level, the onboard control system will have to try to recover the situation at the next highest level, i.e. the GNC subsystem level by, e.g., switching to a redundant string of the GNC subsystem. If recovery actions on the GNC subsystem level still do not bring the vehicle back into the allowable margins of state, actions at

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mission level have to be taken. The options for the onboard FDIR at this level are, depending on the approach phase, e.g. a hold on V-bar, a free drift with contingency attitude mode and a mission abort (CAM).

In figures 6.1 and 6.17, the FDIR system has been presented as a single functional block. In a real design, the failure detection and identification (FDI) function will have to be, following the above three levels of failure detection, very closely related to the equipment (sensors, reaction control system, computers) and to the GNC functions, and will not form a separate subsystem. As we have seen for recovery, probably the same functions which are needed already for the management of the nominal mission will be used. In a real design implementation, the FDIR functions cannot, therefore, be separated from the MVM functions.

The following can be used for fault detection on the onboard GNC equipment.

- Information on equipment power status (i.e. switched-on or -off) or power consumption.
- Equipment can provide information on its health status (self-test data).
- Where equipment has different modes of operation, information on the mode status can be provided.
- For certain sensors, where the measurement principle includes different types of data, consistency checks of measurement data may be performed. This may be possible, e.g., for GPS and for a laser scanner type of rendezvous sensor, but not for sensor types where only one parameter is measured, as in the case of single gyros, Earth and Sun sensors, e.g. For single-parameter sensors, even where one hot redundant equipment is available, a failure condition cannot be allocated to one of the redundant sensors on the basis of the output signal alone.
- Failure of thrusters can, in principle, be detected on the equipment level by measurement of the pressure in the combustion chamber and of the equipment temperature. The first possibility is not always available with the hardware on the market, and the drawback of the second possibility is the duration after failure before the condition is observable. Hard failures of thrusters, i.e. stuck open and stuck closed, can probably be detected more quickly by observing the output of the control function (see below). The clear identification of a thruster failure will be, however, quite difficult, with complex thruster configurations, which are not fully symmetrical w.r.t. the CoM of the vehicle.

In accordance with the general description of a Kalman filter, figure 6.3, the navigation filter of an onboard system for automatic rendezvous can be represented by a general block diagram of the form shown in figure 6.18, in which the functional blocks correspond to the update and propagation blocks which process Eqs. (6.5)–(6.8). Comparison of the results produced by the measurement, propagation and update blocks can help to identify malfunctions in the navigation filter.

For the detection of failure conditions on the GNC subsystem level, including fault conditions on the sensor and thruster level and of the GNC functions proper, parameters and conditions such as the following can be checked.

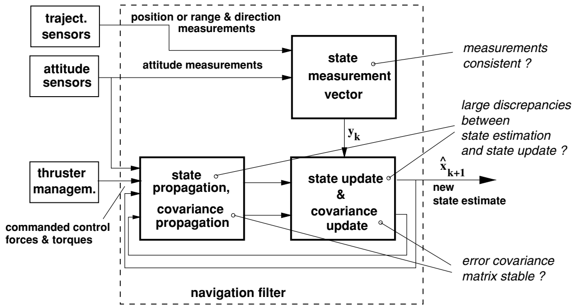

Figure 6.18. Navigation filter simplified: error detection possibilities.

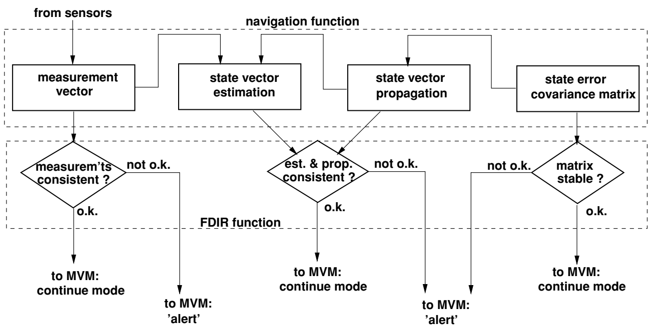

Figure 6.19. Failure detection related to the navigation function.

- Discrepancies between measured and propagated states (see figure 6.19). This condition points to a major problem in the navigation of the vehicle, which can be caused either by a faulty operation of one of the sensors or of the software of the navigation filter.

- Convergence of the navigation filter (see figure 6.19). Delay of convergence or even divergence of the filter can be caused by any of the data inputs, i.e. sensor data or thrust command feedback, or by errors in the filter software. Again, with this check only a failure condition can be identified without pointing to the failure source.

- Duration of manoeuvres (see figure 6.20) and check that the desired state has been achieved (this has been addressed in the previous section). The result of such checks relates to the achievement of a planned dynamic condition. The detection of a deviation will, however, not immediately point to the failure source.

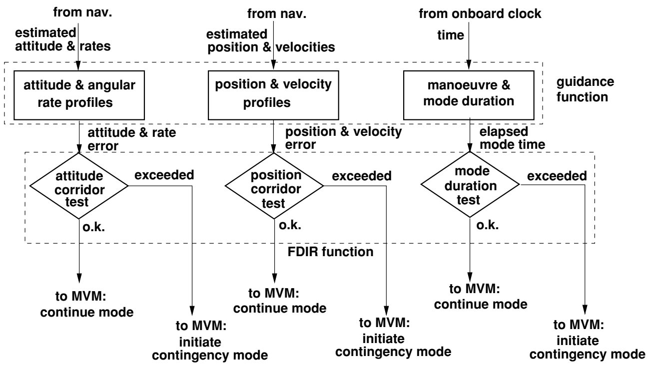

Figure 6.20. Failure detection related to the guidance function.

- Limits of forces and torques requested by the controller (see figure 6.21), e.g. saturation of command output to thrusters. Except in the case of boost manoeuvres, the continuous command of thrust in one direction can be caused either by a large deviation of the nominal state from the set value commanded by the guidance function or by a thruster-open failure.

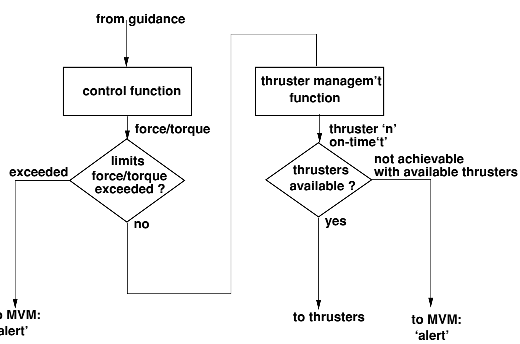

Figure 6.21. Failure detection related to the control function.

To detect a failure condition at the highest level, i.e. the transgression of safety margins around the nominal values of position, velocity, attitude and angular rates, in the first instance the nominal sensors and GNC functions can be used (see figure 6.20). This will cover failures of thrusters and 'hard' failures of sensors and of the GNC software. 'Soft' failures of sensors and GNC software can, however, not be detected in this way. The term 'soft failures' is used here as a general term for faulty outputs of the sensor and GNC functions, which are not immediately identifiable, but will cause over time critical deviations in the actual from the nominal spacecraft state. Serious malfunctions, such as cease of operation, completely inconsistent output data or high noise, are what is meant by 'hard failures'.

To cover 'soft failures' (e.g. a slow build-up of bias on a sensor) by the onboard system, independent sensors and an independent guidance and navigation function would have to be installed on board, which in fact would have to be more reliable than the nominal GNC system. It is obvious that any design will eventually approach some technical and economical limits. This is one of the reasons that even a well-designed onboard FDIR system can assure protection against failures to a very large extent, but never 100% of the time. The availability of external agents, having independent means for remote monitoring, for assessment of contingency situations and for interaction with the automatic onboard system, will, therefore, always be highly desirable and will be necessary for safety reasons for the final part of the approach and capture. Such agents will be ground operators and the crew in the target station with their support tools, and the independent means of assessment may be direct vision, video or other measurement devices for range and direction (see chapter 9). Although fully automatic external supervision systems are conceivable, in the vicinity of a manned target station, monitoring by human operators will be indispensable.

Next in importance to failures in the onboard system is the interruption of the communication links; this can endanger the rendezvous approach, as these links may be essential both for the remote monitoring and control and for navigation, as in case of RGPS (see section 7.3.3). The following concepts concerning the fault detection of communication links can be envisaged:

- The link from the target station to the chaser vehicle can be checked by the chaser onboard system, e.g. reception of GPS data, where RGPS is used as rendezvous sensor. Otherwise, e.g., a continuous pulse or tone signal emitted by the target station could be monitored.
- In the close range rendezvous phases, the link from the chaser to the target station can continuously be checked by the target receivers, as the chaser will have to broadcast continuously its GNC and housekeeping data. For RGPS this will include implicitly also a check of the link from the target station to the chaser vehicle, as the data broadcast by the chaser will include the navigation data. Otherwise, if deemed necessary, a continuous signal transmitted by the target can be re-transmitted by the chaser, allowing (on the target side) monitoring of both directions of the link.
- The links between chaser and ground, whether via DRS or via ISS/TDRSS, including all ground links, can be checked by ground receiver equipment.

As we have seen, various criteria are available to enable the identification of a faulty situation, covering the large majority of potential failures. These criteria do not lead, however, in most cases to an immediate and unequivocal identification of the failure source. Only in the case of equipment failures, e.g., where health status information is available, will immediate decisions for redundancy switching on this level be possible. In most other cases, the sole recovery option will be to switch over to a redundant string on at least subsystem (e.g. GNC) level. To ensure the availability of the redundant chain, the redundancy status of all necessary functions has to be monitored, and cross straps have to be managed, to engage equipment which has already lost its redundancy, as the choice of recovery operations or eventual safety measures (e.g. CAM) will depend on whether redundancy is available or the function has been lost.

It can be concluded from the above discussion that the recovery from the various types of contingencies will consist mainly of three types of actions:

(1) Switch-over to redundant single equipment, if faulty equipment can be identified.
(2) Switch-over to redundant string, if failure could not be isolated. This includes a switch to a redundant processor with identical RVC software.
(3) Interruption or abortion of mission, i.e.
   - interruption of mission (stop on V-bar, retreat to hold point),
   - execution of a CAM, where danger of collision exists,
   - inhibit of trajectory control actuation to leave the vehicle on a safe drift orbit (if available),

   in all cases where problems cannot be solved by redundancy switching.

In order to fulfil the failure tolerance requirements (described in section 4.1.1) for essential equipment such as data management equipment, reaction control system hardware, gyros, etc., double redundancy must be available. For other equipment, which is not essential for safety operations (CAM, survival mode), such as the rendezvous sensor, single redundancy will be sufficient. Functional redundancy will have to be included in the design, to the largest extent possible, in order to reduce the complexity of the system.

Detection of and recovery from contingencies due to violation of corridors or due to loss of functions can either be handled by the onboard system or by the target crew, or by ground operators assisted by support tools (see chapter 9). As mentioned above, re-planning and re-synchronisation of the mission after a contingency resulting in a change of the timeline can be achieved only by the ground system. Fast re-planning will require a faster than real time simulation capability, as the recovery strategy will have to be verified prior to up-linking to the chaser vehicle (see section 9.2.2).

6.5 Remote interaction with the automatic system

Although the onboard system of automated unmanned vehicles must be sufficiently failure tolerant to fulfil safety and mission success requirements (see section 4.1.1), it is obvious this can never cover all potential failure cases. Contingencies can be caused by external events (e.g. by problems in the target station), by design mistakes left undiscovered during the verification/validation process, or by manufacturing faults left undiscovered during the acceptance procedure. Remote control techniques could help to resolve those contingency situations during flight which are caused by such problems. This requires interaction with the various functions of the automatic onboard control system depending on the contingency situation and type of failure. The following four levels of intervention into the automatic onboard system can be considered:

- High level intervention in the mode management concerning the mission sequence and in the equipment scheduler concerning the vehicle configuration. This type of intervention will be discussed in more detail in chapter 9.
- Intervention in the navigation, guidance or control functions while the automatic GNC system is still operating. In this case certain functions are modified or partially taken over by a remote operator.
- Manual navigation, guidance and control, i.e. the N, G and C functions of the automatic system are fully taken over by a remote operator. There is still a closed loop control, as the human operator continuously corrects the actual state w.r.t. the nominal one. Also, the actuator management function of the onboard system is still required.
- Open loop intervention, i.e. command by remote operators of discrete thrust manoeuvres to change trajectories or attitudes.

(The term ‘remote operator’ is applied here to human operators, computerised operator functions, or a combination of both, which are outside the chaser spacecraft.)

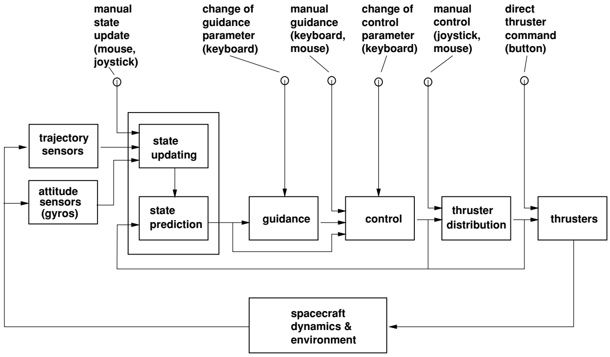

Figure 6.22. Potential points of intervention in the GNC loop.

6.5.1 Interaction with the GNC functions

Concerning the GNC functions, the possibilities available, in principle, of remote interaction with an automatic system are shown schematically in figure 6.22 and listed below:

- input of measurement data to the navigation function (manual state update);
- inputs to the guidance function:
  - fixed set values,
  - new guidance laws,

  - new guidance parameters;
- inputs to the control function:
  - new control laws,
  - new control parameters;
- input of force and torque commands to the actuator management function;
- input of valve open/close commands to individual thrusters.

Where guidance and control laws and parameters are concerned, interaction is by way of a software update, where the new software will be verified and validated on ground and up-linked to the onboard system off-line, e.g. during drift phases or hold points, when these guidance and control laws and parameters are not in use by the onboard system.

The direct command to thrusters from outside requires, in principle, a remote thruster management function, as the realisation of pure forces and torques will require generally the combined action of several thrusters. In the case of a CAM, however, where a fixed force in one body direction of the spacecraft has to be produced, a fixed set of thrusters with a fixed 'on'-time for each can be commanded.

Three of the points shown in figure 6.22 are of particular interest concerning a more continuous interaction by human operators in an automatic GNC system:

- manual state update, i.e. manual input of correction data to the propagated state, as produced by the navigation filter, to the actual state, as observed by the human operator;
- manual guidance, e.g. input of set points for velocities or angular rates by the human operator;
- manual control, i.e. input of force or torque commands into the thruster management function by the human operator.

6.5.2 Manual state update for the automatic GNC system

The manual state update can be useful as a backup mode, when the rendezvous sensor has failed but the automatic GNC system is still available and the human operator has continuous information from a video camera. If, e.g., an artificial contour of the target vehicle is produced and superimposed on the video picture, so simulating the same field of view as the video camera and driving the direction and size of the artificial view by the state update output of the navigation filter, the human operator can match the two images by varying the artificial view (i.e. by increasing/decreasing size, by changing position, and by rotation). This concept is illustrated by figure 6.23. Because the computer generated image has all the necessary data, such as range, line-of-sight angles etc., the modified values (after matching the images) can be entered into the update function of the navigation filter (MATRA 1993; Vankov *et al.* 1996).

In order to explain the principle, the real shape of the target spacecraft has, in figure 6.23, been replaced by that of a pyramid. The complete geometry of the target is, in principle, not needed for the artificial image. A minimum number of four points, of which

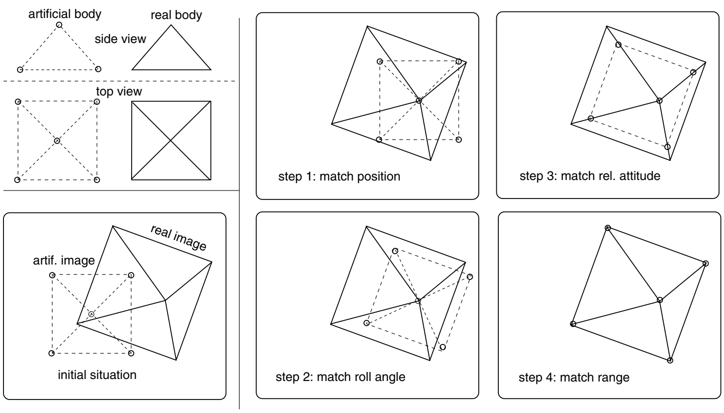

Figure 6.23. Manual state update by superposition of real and artificial images.

one must be outside the plane described by the other three, are required to define and manipulate a body. An artificial image more closely describing the spacecraft geometry will, however, be easier to match with the real target image because of the redundancy of information.

In the initial situation, the operator sees on the screen the unmatched video image of the target and the artificial image; this represents the state of the vehicle as propagated by the navigation filter. In the first step the operator will match the position (or azimuth/elevation angles, respectively) of the images by moving the artificial image, using, e.g., a computer mouse or a joystick. In the second and third steps the operator will rotate the artificial body about three axes to adjust the relative attitude, and in the final step he or she will match the size to adjust the range to the target. These steps may be performed iteratively. When the match is satisfactory, the result, which will be re-transformed by the computer into position and angles, will be supplied as a new measurement into the navigation filter. It has been found during simulations that an untrained operator can perform this task in real time with sufficient accuracy (Vankov *et al*. 1996).

6.5.3 Automatic GNC system with man-in-the-loop

Whereas in the rendezvous approach of the US Space Shuttle, the last part of the approach up to contact will always be controlled manually, the Russian vehicles, i.e. the manned Soyuz and the unmanned Progress, have manual control as a backup mode, which would be used only in case of failure of the automatic system. It should be noted that for any automatic vehicle such a manual backup control mode is a means by which to increase mission success rather than to increase safety. By itself, a manual control will be less safe, as there will be no automatic check of safety boundaries and the human controller will be the highest level of authority.

In the manual operating mode, as used, e.g., for the Russian Progress transporter, the unmanned vehicle will be manually remotely controlled by a crew member in the target station, or by a ground operator, using video images. Video pictures are sent by the approaching vehicle to the target station or ground. From the video pictures the remote human operator extracts the navigation information, and inputs commands to the system in the same way as if he were the pilot in the vehicle.

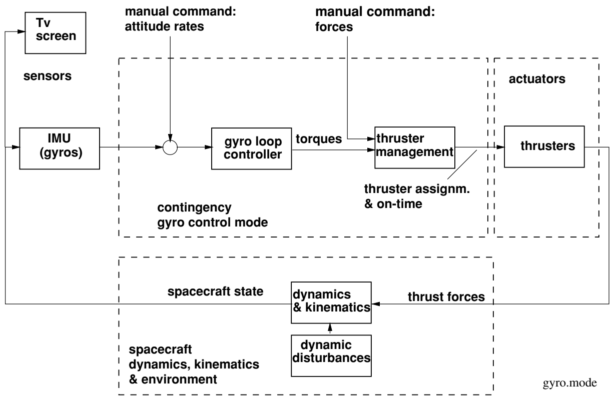

Figure 6.24. Manual operating mode with manual guidance and manual control inputs.

In the manual operating mode the human operator fulfils all GNC tasks except for stabilisation of attitude, which is performed by an automatic control loop based on gyro measurements (see figure 6.24). Changes of attitude will be entered by joystick command as change of attitude rate (guidance set value) into the gyro stabilisation loop. This is actually a replacement of the guidance function by the human operator, as he introduces the set values for the angular rates. Changes of translation velocities will be entered by joystick command directly into the thruster management function, which selects the thrusters and the length of thrust. In this case the human operator replaces the control function, as he directly commands the forces and torques to be executed.

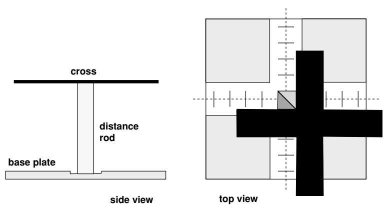

Figure 6.25. Visual target pattern for short range manual approach.

Navigation is performed by the human operator by assessing on the screen (with the help of reticles and fixed marks)

- the distance to the target (range) from the apparent size of the target image,
- the position or azimuth and elevation angles from the position of the target image relative to the centre of the screen,
- the relative attitude angles from the position of principal points of the target geometry w.r.t. the screen axes.

As long as the distance to the target is large enough that the full image of the target is on the screen, the operator will use the complete image for assessment. At shorter distances, only a part of the spacecraft will be on the screen. The operator will then use the largest available features for navigation assessment. Eventually, only a small section near to the docking port will be visible on the screen. It will then be necessary to have a particular visual target pattern, which must be sized such that its image does not exceed the screen up to contact. For monitoring the visual target pattern at medium and short range, cameras with two sets of lenses can be used. The concept of such a target pattern, as used in Russian and American space programmes, is shown in figure 6.25. The pattern consists of a base plate and a cross at a distance from the ground plane. Range, position and relative attitude can be assessed in the same way as described above from the relative position of the pattern w.r.t. the screen and from the size and position of the cross w.r.t. the base plate.