10

Verification and validation

The final chapter of this book is dedicated to one of the most difficult issues in the development of an automated rendezvous system, i.e. the problem of gaining sufficient confidence prior to flight that the system will perform in orbit as required by the mission objectives and as intended in the design. Since orbit dynamics and the condition of 'zero-g' cannot be reproduced on ground, the function and performance of many features cannot physically be tested prior to flight. This is a general problem for all space missions. In addition to physical testing, the process of gaining this confidence will, therefore, have to include mathematical modelling of the orbital effects and spacecraft systems, and the evaluation of the behaviour of the spacecraft in the orbital environment by analysis and simulations will be based on those mathematical models. The entire process of obtaining confidence by physical testing, analysis and simulation is referred to as *verification* and *validation*, where these terms are used in the sense that:

- *verification* is the proof that
  - an item, function or process performs according to the specification, under which it has been developed;
- *validation* is the proof that
  - an item, function or process will behave as expected under real world conditions, or
  - the description by mathematical modelling represents, to a sufficient level of accuracy, the behaviour which an item, function or process would have under real world conditions.

In contrast to other space missions, in the final stages of a rendezvous and docking mission, flight operations have to be performed between two spacecraft in close proximity, and eventually physical contact has to be achieved. As these operations are safety-critical, it is particularly important to minimise prior to flight any risk concerning these operations. This requires in the development phase a particularly rigorous verification of all functions, processes and interfaces involved in the proximity operations and a proper validation of all mathematical models and tools involved in the verification process.

10.1 Limitations of verification and validation

The highest possible confidence in the proper functioning of a function or item in the real mission will be obtained when it is subjected, during testing prior to flight, to the same environment and conditions as it will experience during the mission. Worst-case conditions of this environment need to be taken into account. Wherever possible, such functions and items should therefore be tested in a realistic physical environment, including sufficient margins to cover such worst-case conditions. For functions and performances which cannot be verified by direct physical tests on ground, two other ways of verification are available in principle: mathematical modelling and simulation or testing in orbit under the conditions of the real flight.

Verification by testing in orbit is very limited, however, not only for reasons of launch cost and opportunity. Full verification by testing in orbit is even more limited because generally the ‘real world’ conditions of a mission cannot be reproduced unless the complete mission is practically duplicated. Because of the cost involved, testing or demonstration in orbit is, in most cases, a matter of flight opportunity, where the test conditions have to be taken as available. In the best case, testing in orbit can be performed under similar conditions which must be proven to be sufficiently representative of the real mission. This general problem of in-orbit verification and validation will be addressed in more detail in section 10.7.

For the majority of all features in rendezvous and docking systems and operations which include orbit dynamics, contact dynamics and ‘zero-g’ effects, verification has to rely on tools and facilities containing mathematical modelling. For this reason, detailed mathematical models have to be established of the spacecraft, its dynamics and kinematics, of the actuators, of the sensors, of the capture equipment, of the onboard data management system and of the communications links and equipment. This modelling must include all effects that the orbital environment has on these features. To make them suitable for use in verification tools, these mathematical models need to be validated w.r.t. the according properties and effects of the real world, which are set by the spacecraft design and by the orbital environment. Development of a rendezvous and docking system therefore always includes both

- the development and verification of the onboard and remote control systems and of their constituents, and
- the development and validation of the verification tools and facilities.

The validation of these verification tools is, to a large extent, the validation of the mathematical models of the features and effects, as explained above. The goal of the validation will be to provide evidence that the representation of the reality by the model is correct and sufficiently complete for the purpose of verification. The last requirement will be the most difficult one to fulfil, since it requires complete knowledge of the 'real world' with all its facets, knowledge that even with long experience will never be 100% complete.

Notwithstanding the above postulation of rigorous verification and validation of all functions, processes and interfaces involved in safety-critical operations, it has to be kept in mind that verification and validation are, in reality, tasks which by nature cannot result in absolute certainty, since they will always be limited in their extent. Furthermore, even if it were technically feasible to test the system, item or operation involving all environmental conditions relating to the mission, it would for reasons of time and cost be impossible to test all the potential variations and combinations of parameters and all possible contingency situations. The method generally applied is to test maximum and minimum values along with a certain representative number of combinations of such values. This is, however, not the same as testing the entire field of possible variations and combinations, as it leaves the possibility of unidentified harmful combinations. An additional problem is that tests can be performed only in anticipation of known effects, leaving the possibility of undetected side-effects.

The goal for all verification and validation efforts can, for the reasons given above, never be the achievement of absolute proof, but rather the acquisition of the highest possible level of confidence that an item, system or operation will perform as required in the real mission under the real conditions. In other words, even after the most rigorous verification and validation process, uncertainty and risk will always remain.

10.2 RVD verification/validation during development

In order to arrive at an acceptable level of confidence that the final product will fulfil its tasks, a number of questions must be answered positively during its development. The basic questions to be answered concerning the issues of verification/validation in any development project are as follows.

(1) Which functions/features are involved in the process; which of them have to be considered as particularly critical for the proper fulfilment of the task in question; and for which of them is experience gained from former developments and applications available?

(2) What risks have to be considered for these features; i.e. what can go wrong during development and, eventually, the operational phase?

(3) How can these risks be reduced; i.e. what needs to be done to obtain evidence for proper function and performance of these features?

These basic questions lead immediately to the next level of questions concerning the distribution of verification/validation efforts over the development life-cycle of a project (see figure 10.1):

- What features and issues have to be verified/validated and when; i.e. what risk needs to be considered for which feature at which point in the development life-cycle?
- How can such features/issues be verified/validated; i.e. which methods or means of verification/validation will have to be applied at each point in the development life-cycle?
- How much effort needs to be invested for each verification/validation task; i.e. in how much depth do features have to be verified/validated to achieve sufficient confidence at each point of the development life-cycle? Considering the fact that proof of proper functioning according to a specification or according to real world conditions is an asymptotic process, which will never reach 100%, a conscious decision has to be made regarding how much effort needs to be invested and how much residual risk can be tolerated.

The sum of the answers to all these questions will eventually lead to the definition of an overall verification/validation approach for the development project in question.

10.2.1 Features particular to rendezvous and docking

As this book is exclusively concerned with rendezvous and docking/berthing, only those features which are particular to the mission task of rendezvous and docking, and which are not used in other types of mission, shall be treated here. For RVD/B, proper function and performance of the following features must be verified:

- the algorithms of onboard systems controlling the rendezvous, i.e. the GNC, MVM and FDIR algorithms;
- the control software in which the algorithms are implemented;
- the sensors required for the rendezvous trajectory and attitude control;
- the reaction control system for full six DOF motion capability;
- the compound of algorithms, software, data management system, sensors and reaction control system forming together the onboard RV-control system;
- the capture dynamics of the docking or berthing system and the physical connection;
- the remote interaction functions (ground, space station crew) with the automatic onboard system.

There are of course many other spacecraft hardware and software items which are also involved in the rendezvous system, e.g. normal attitude sensors, such as gyros, Sun and Earth sensors, the data management and communication subsystems, thrusters and valves, etc. These items, although essential elements in any rendezvous control system, will not be discussed here, as they are not rendezvous specific. Generally, the methods of their verification are well-known from other space or even ground applications. Also, other verification issues, e.g. those w.r.t. physical space environment, such as launch loads, thermal vacuum, radiation, electrical and electro-magnetic environment or the verification w.r.t. manufacture requirements of the spacecraft, or end-to-end verification of a functional chains, etc., are not considered here, as they are covered by the normal verification tasks within any spacecraft project.

An exception to this is the reaction control system. Thruster management function and thrusters are, of course, used in practically all other spacecraft. However, in RVD/B, thrusters are used in a different mode with a much higher number of duty cycles, and the thruster management function has to select actuators for six DOF control, which is different from that of spacecraft with attitude control only. Also, trajectory errors caused by thrust imperfections and the consequences of thruster failures concerning trajectory safety, are issues specific to a rendezvous approach. For this reason the reaction control system is counted here as a rendezvous specific function.

In conclusion, three distinct lines of verification have to be followed which are, to a certain extent, independent of each other:

- the functions in charge of trajectory implementation;
- the functions responsible for the physical mating process;
- the functions and operations involved in the supervisory control of the automatic onboard system.

The first line of verification concerns the GNC and MVM functions, including algorithms, sensors and reaction control functions. The second line includes the contact and capture dynamics until insertion into the structural latching interfaces. The verification of the structural latching itself will not be discussed here, as it is in no way different from similar functions in ground applications. The third line includes the operations by the ground controllers in the chaser and target control centres, the target station crew and the functions of the dedicated support equipment for these operators. Functions for communication, such as packeting, encryption, transmitters, receivers, ground links, etc., will not be addressed, since they are not specific to RVD/B.

10.2.2 Verification stages in the development life-cycle

Verification and validation are not constrained to a particular phase at the end of a project (e.g., the *qualification phase*, during which it will be proven that everything is functioning and performing according to the requirements under all conditions of the real mission). On the contrary, verification and validation tasks start at the very beginning of a project and continue during each of the project phases. The methods of verification and validation in each phase have to be chosen such that confidence is achieved in those particular aspects which are at stake at the particular stage of development in question. In the development life-cycle of a space project the following major questions have to be answered at the various development stages.

- In the mission definition phase:
  - are mission concepts and requirements realistic and feasible?
  - do the requirements and specifications represent the real mission needs?
- In the design phase:
  - will a design be feasible which fulfils the specifications?
  - will the design be able to realise the mission concept and provide the required performance under real world conditions?
- In the development phase (concluding with qualification):
  - does the actual design function and perform according to the specification?
  - will the design implementation in hardware and software fulfil the function and performance requirements for the mission under real world conditions?
- In the flight item manufacturing phase:
  - do the flight items 'as built' in all aspects, i.e. physical, function and performance, fully correspond to the ones which passed through the qualification phase?
  - are all subsystems and items properly integrated?

It is the primary objective of the verification/validation process to ensure that the above goals of each particular phase are fulfilled. In the first instance, the verification task is to ensure that the specifications are followed. However, verification goals must of course always be related to the proper functioning and performance in the 'real world', i.e. under the conditions of the real mission. The second question at each step of the development life-cycle must therefore be

- Are there any effects in the real world, that would potentially cause a risk in the operational phase, which are not known and not sufficiently covered by the specification and verification process?

It is obvious that the detail and aspects of the 'real world' to be considered in the validation process will depend on the project phase in question. Unfortunately, the 'real world' will never fully be replicated before the mission is flown. As a result of analyses, tests and (possibly) orbital experiments, knowledge of the ‘real world’ will steadily be increased during the development stages. However, even at the time of qualification of the system, a residual uncertainty will remain. The aim must be, rather than 100% proof, that, at the end of the development life-cycle, i.e. at completion of qualification, the risks for the operational phase will be reduced to an acceptable level.

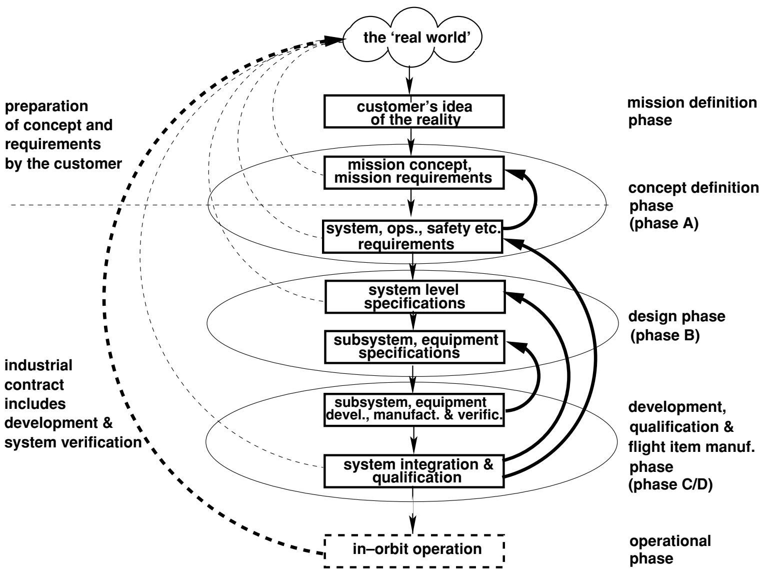

Figure 10.1. Verification and validation in the development life-cycle of a space project.

In figure 10.1, the arrows with solid lines on the right hand side represent the verification tasks, and the dotted lines on the left hand side represent the questions concerning validation. According to the above definition, verification tasks can begin only once the first products have been established according to a set of written requirements or specifications. The first task is usually the verification of the system specification w.r.t. the overall mission requirements.

A major validation task already exists, however, right at the beginning of a project. The customer, usually a national or international space agency, will have an idea of the mission to be performed. This includes also the first idea of the ‘real world’ which will exist during execution of the mission. Starting from this idea, the customer must establish a mission concept and define the mission requirements. These concepts and requirements need to be validated against the ‘real world’, which at that stage is known only to a limited extent. The investigation and description of the ‘real world’ in which the spacecraft and its subsystems and items will have to operate during the mission, is, next to the development of the spacecraft systems, subsystems and items, the other major development task during the entire development life-cycle of the project.

After the first definition of mission and system requirements, all subsequent lower level specifications and implementations have to be verified against the next higher level specifications and requirements. However, as all these sets of requirements and specifications refer to the ‘real world’ describing the future mission, there will also at each step be the need to validate the detailed description of the ‘real world’ required for the verification task of that particular step. The verification/validation effort culminates in the development phase, where it should be proven that a system or item will achieve its specified performance under all conditions of the real mission. The sum of all these activities is referred to as *qualification*. However, keeping in mind the limitations of all verification and validation efforts, it will eventually be only in the operational phase of the real mission that the spacecraft system will meet the ‘real world’ proper. Only then can final proof be achieved that the system indeed functions and performs properly, that all requirements were indeed defined correctly, comprehensively and in sufficient detail, and that the tools for verification have indeed been correctly validated to represent the ‘real world’ to the necessary extent.

10.3 Verification methods and tools

The objective of this section is to provide an understanding of the issues to be verified, the methods and types of tools employed, and the depth of modelling needed for verification in each of the various development phases. Verification concerning the effects of orbital environment, such as thermal vacuum, radiation and launch loads, will be addressed only if that environment is expected to cause a change in the parameters involved in the dynamic processes of rendezvous, contact and capture. As already stated above, the verification w.r.t. resistance to launch loads will therefore not be discussed here.

This section will also briefly address the possible and necessary depth of the validation of mathematical models or physical stimuli of test items used in the simulations of a particular development phase. Possible methods and constraints of validation for the various types of models will be treated in more detail in section 10.5.

As the intention is to show in principle the methods used for the verification and validation tasks at each development stage, the description can be limited in the case of the RV-control system to verification and validation of the guidance, navigation and control (GNC) functions, and in the case of the mating system to the verification and validation of the front end functions, i.e. the ones involved in contact dynamics and capture. In the case of a docking mechanism, the front end functions include the elements for reception, the attenuation devices and the capture latches.

For the RV-control system it will be indicated where MVM and FDIR functions will be integrated with the GNC functions. The verification of these functions will not, however, be discussed in any detail. Their final verification is, in any case, included in the overall verification of the rendezvous control software in the development phase. The description of contact dynamics and capture verification will be restricted to docking.

For berthing, the verification of capture of the interfaces of the other vehicle by the end-effector of a manipulator and the transfer to and insertion into the attachment interfaces, all of which are controlled by a human operator, are outside the scope of automated rendezvous and docking and will, therefore, not be discussed in any detail.

10.3.1 Mission definition and feasibility phase

In the first phase, the mission concept and mission requirements have to be validated, which means that the feasibility of trajectories, attitudes, thruster configuration and thrust level etc. have to be confirmed by analysis or simulations. These early simulations do not need, however, to include the modelling of a closed loop for the GNC functions, or the algorithms for mode sequencing, equipment engagement (MVM) and FDIR functions, or other detailed features of the onboard system. Analysis of the trajectories, of the ΔVs to be applied and of the duration of manoeuvres will yield the basic data required to assess the required thrust level and propellant budget, the type of sensors required, the feasibility of capture, etc.

Verification/validation of trajectory control issues

Verification objectives

- Feasibility of trajectory and attitude strategy and overall ΔV requirement for the trajectory sequence.
- Feasibility of thruster configuration, thrust level, and preliminary propellant budget.
- Identification of required navigation and control performance in the various stages of approach and preliminary selection of sensor types.

The particular requirements for the individual features of the rendezvous system (subsystems, equipment, functions, operations) are derived from the initial mission and system design analyses. Verification of these individual requirements is the first important part of the overall verification task.

Verification tools The most important tool required in this phase is a non-real time trajectory simulation based on the Clohessy–Wiltshire equations. Thrust level requirements and propellant budget or trajectory and attitude control are derived from the ΔV results by applying empirical factors. Navigation performance is estimated from available sensor data and from past experience. Concerning disturbances, the trajectory simulation needs to include only the modelling of differential drag, which is essential for the analysis of manoeuvre performance and long term safety of trajectories. This requires a first estimate of the geometry of chaser and target vehicles and of the atmospheric density in the rendezvous orbit. Other orbital disturbances are of lower importance and need not to be considered at this stage.

Verification/validation of contact dynamics and capture

Verification objectives

- Feasibility of capture with the velocities, lateral and angular misalignments at contact, resulting from the expected GNC performance of the chaser and attitude dynamics of the target.
- Vice versa, the identification of requirements for contact conditions to be fulfilled by the GNC system of the chaser.

Verification tools At this early stage, verification will consist of an assessment of the feasibility of capture by a comparison between the GNC performance expected at contact and the capture capabilities of existing mating systems. In the case of berthing, this can be an assessment of whether or not the GNC system will be able to fulfil the conditions of the berthing box.

If a docking mechanism design is already available, it may be of interest at this stage to check the probability of capture by means of a non-real time contact dynamics simulation, which would include a simplified modelling of the contact surfaces and capture mechanism. If a new docking mechanism has to be developed, there will not yet be a basis for any kind of simulation at this stage. Feasibility can in this case only be assessed by comparison with known parameters resulting from the successful use of existing designs.

10.3.2 Design phase

Verification/validation of trajectory control issues

The first important task in the design phase is the development of the guidance, navigation and control algorithms. They will initially be verified by running in closed loop with an environment simulation which is developed together with the GNC algorithms. As long as no detailed models are available, spacecraft and orbital environments will initially be modelled in this simulation with consideration only to their dominant effects. For example, the spacecraft will be modelled as rigid bodies; the sensors will be modelled according to their basic function, plus some bias and noise; and only drag or differential drag according to the spacecraft cross sections will be taken into account when considering orbital disturbances. The expected/required GNC performance obtained from such early simulations will be the driving factor for the specification of the trajectory sensors and front end functions of the mating system, i.e. for reception range and capture. The MVM and FDIR algorithms are, to a large degree, independent of the GNC algorithms and will have to be developed in parallel. They can be tested by applying the appropriate stimuli to trigger the state changes. At a later stage they will have to be merged with the GNC algorithms to form the complete onboard rendezvous control software.

In the final stage of the design phase the RV-control software will eventually have to be verified in a closed loop 'all software' simulation of all environment functions and effects. This must then include, in addition to the RV-control software, detailed models of the spacecraft body, including the flexibility, rotating appendages, fuel sloshing, etc., of the onboard data management and communication architecture, of all equipment and of orbital disturbances. The RV-control software consists of the GNC, MVM and FDIR algorithms. As no hardware is involved yet, such a simulation setup can still run in non-real time, i.e. faster than the real processes, to reduce simulation and evaluation time. Running the RV-control software in a simulation environment with such detailed and complete models will provide sufficient confidence that the rendezvous control software will work properly and provide the required performance in the real mission environment.

Verification objectives

- Feasibility of the design concepts for GNC, MVM and FDIR.
- Achievability of performance requirements for GNC.
- Feasibility of design implementation with the envisaged sensor and data management hardware.
- Correctness of environment modelling (validation task).

Verification tools In the first step, separate tools will be required for GNC, MVM and FDIR algorithm development. For instance, GNC algorithms may initially be developed in non-real time development environments for control systems, such as Matlab or MatrixX, and tested in closed loop non-real time 'all software' simulations in that development environment, with basic modelling of spacecraft dynamics in orbit and including orbital disturbances. At this stage of development, the modelling of spacecraft design, subsystems and equipment may still be of limited detail. Spacecraft and equipment developments are performed in parallel, and their design is usually not yet finalised in this early phase. Under these conditions, vehicle dynamics and disturbances will have to be modelled using the available preliminary spacecraft design, and sensors, actuators, etc. will have to be modelled according to their expected GNC behaviour, rather than to their detailed design. For instance, as long as the detailed design is unavailable, sensors will usually be modelled as a basic function, plus bias, plus noise. Modelling of the measurement environment will be simplified. For GPS, e.g., only the position and visibility of GPS satellites w.r.t. the antennas on the chaser and target vehicles may be simulated, and for optical sensors the disturbance effects of a measurement environment (e.g. specular reflections) can be neglected. A block diagram of an initial GNC simulation is shown in figure 10.2.

The preliminary models of the early design phase will have to be exchanged in later stages for more refined ones. For instance, the bias and noise type sensor models, which

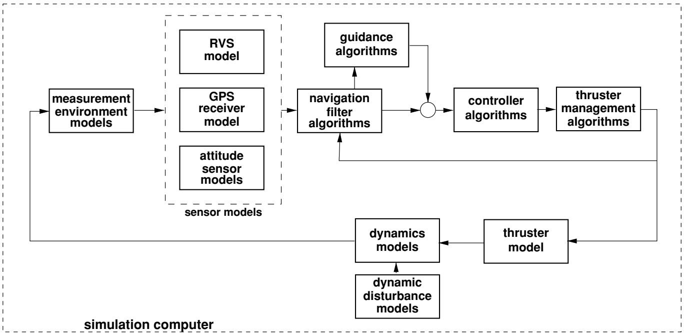

Figure 10.2. Closed loop GNC simulation on one platform.

try to approximate the behaviour of the sensor, will have to be replaced by more detailed models representing the actual design of the sensor in terms of field of view, maximum range, resolution, bandwidth, etc., and a more refined model will also have to include its operating modes. The measurement environment models will have to include the specific details of the sensor environment in question. This will include, in GPS and RGPS, e.g., the modelling of GPS satellite configurations and of multi-path and shadowing effects. In addition to run-time simulations, other tools, such as error co-variance analysis, will be used to assess the expected margins of GNC performance.

All analysis and verification tools used in the design phase will typically run in non-real time, as there is no hardware test item requiring a real time environment. Validation of the models of spacecraft and equipment will, at this stage, be limited to a comparison with the evolving design of these items. Validation of models for dynamic disturbances in the orbit and measurement environments will in most cases be limited to comparison with data known from previous missions.

Verification/validation of contact dynamics and capture

Verification of the contact and capture process will obviously depend on the type of mating used in the mission in question. For berthing this will require a simulation of a manipulator arm, with a human operator in the loop. As the manipulator and the human operator are usually located on the target station, verification of capture is the responsibility of the target project, and the chaser project will have only to provide evidence that its vehicle will be delivered into a berthing box as defined in figure 5.6, which is a GNC task.

If a docking mechanism has to be newly developed, the initial verification task is to prove that the design parameters for the reception, attenuation and capture functions have been chosen correctly. If an existing docking mechanism design has to be used, the verification objective will be to show that capture will be possible with the expected GNC performance. Otherwise, it must be determined how the GNC performance must be improved, or what parameters of the docking mechanism need to be changed, to ensure capture.

Verification objectives

- Forces and torques acting between chaser and target vehicles at contact must be below acceptable limits.
- The choice of shock attenuator parameters in the design must be correct concerning capture capability and force limits.
- Capture must be achieved with the given GNC performance of the chaser and attitude dynamics of the target.

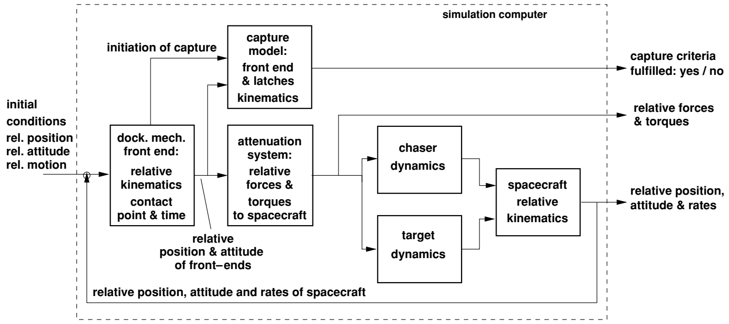

Figure 10.3. Docking system capture simulation.

Verification tools To analyse the probability of capture, a tool needs to be available which includes modelling of the following functions: the kinematics at contact of the docking front end interfaces on chaser and target; the forces and torques applied to both spacecraft via the attenuation system; the relative motion of the spacecraft as a result of these forces and torques; and the operation of the capture latches. A simplified block diagram of such a simulation is shown in figure 10.3. The first task of this simulation is the identification of the contact point and the direction of the contact forces. This requires modelling of the reception interfaces on both sides and the determination at each point of the distances of the surfaces between one side and the other as a result of relative position and attitude of the two vehicles. Contact point, force direction and the relative position and attitude of the capture interfaces determine the deformations of the attenuation system, which in turn determine the forces and torques acting on the vehicles. The closing motion of the capture latches, which will in reality be triggered by either contact force detection or an optical sensor, can be initiated in the early stage of the simulator development by a distance criterion. By changing the design parameters of the reception structure and the attenuation system, the reception range, the time available for capture and the forces acting on the spacecraft can be varied to improve capture probability. As for GNC simulations, in the design phase there will be no need for a real time simulation, as no hardware or human operator is involved.

Validation of the geometric and dynamic models used in the simulation determining contact point, force direction and dynamic reactions of the attenuation system can, at that stage of development, only be achieved by comparison with the design of the available docking mechanism, and with the results of calculations, e.g. those concerning the forces at particular deformations of the attenuation system.

10.3.3 Development phase

Verification/validation of trajectory control items

In the development and qualification phase, the onboard control system for rendezvous has to be verified for function and performance in an environment and under conditions as close as possible to the real application. This requires that the hardware and software of the system will have to be tested in closed loop in a real time operation environment with as many as possible real items in the test chain. For the rendezvous control system this means that the RV-control software will have to run in its proper computer hardware and that eventually real sensor hardware will have to be connected to it. It is generally not possible to include the actuator hardware in closed loop performance tests, as this would require the proper orbital environment for thrusters and spacecraft dynamics to close the loop. To test the rendezvous system with sensor and controller hardware in the loop, all items involved in the dynamic process, e.g. spacecraft body and actuators, must still be represented by mathematical models.

Verification/validation objectives

- Proper function and performance of the complete rendezvous control system implemented in hardware and software.
- Function and performance of the navigation hardware and software in a realistic measurement environment.
- Proper function of the onboard system together with the remote control functions (ground, target station).

GNC verification tools During the development phase, the complete rendezvous control software (GNC, MVM and FDIR functions) will become available and its performance will have to be verified. The most important tool required for verification of the onboard rendezvous control system in the development and qualification phase will, therefore, be a real time simulation, which will be able to test the RV-control software together with the data management hardware and software, i.e. the onboard computer and data bus in the loop. The rendezvous control software will be resident in the onboard computer, and the interfaces to sensors and propulsion control electronics, or to the mathematical models of these items, will be via the onboard data bus. In this way any modification of the behaviour of the control algorithms due to their implementation in the software and due to their operation in the data management environment will be included in the test results.

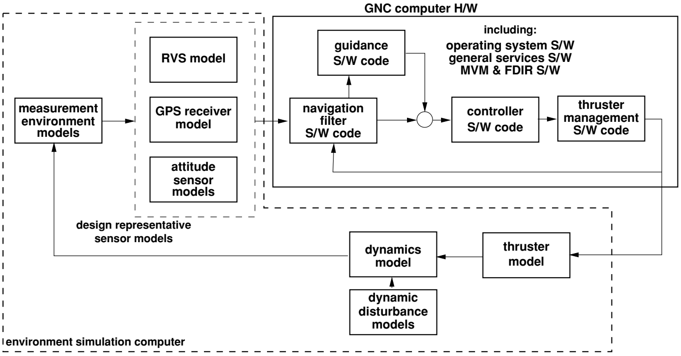

Figure 10.4. Closed loop GNC simulation with RV-control computer as test item.

The mathematical models for orbital perturbations, for the actuators and their drive electronics, for spacecraft dynamics and kinematics, and for sensors and their measurement environment, need to be available in sufficient detail and reliability to produce the same closed loop test result as in the real mission. All models need to be validated to fulfil these requirements, which means that, by test or comparison, sufficient confidence must be established that they represent the 'real world' to the extent necessary. The resulting simulation setup will be the main tool for verification of the function and performance of the rendezvous onboard control system (see section 10.4). It will be used for verification of the nominal mission and for all foreseeable non-nominal/contingency situations. A typical simulation setup for the GNC functions in the development phase is shown in figure 10.4. The MVM, FDIR and data management functions are not shown in this figure. They will, however, be included in the simulation runs and form part of the verification effort for nominal and contingency cases.

Sensor verification tools To verify the sensor hardware and software, test setups with physical stimulation of the sensors will be required. In the first instance, these will be open loop tests, where the physical stimulation is intended to provide an as realistic as possible measurement environment for the sensors in accordance with the trajectory and attitude motion of the chaser and target spacecraft and the motion of other reference points. These are, e.g., the navigation satellites for the case of GPS, RGPS or other satellite navigation systems, and, for optical sensors, the Sun at the time of the approach (figure 10.5). For attitude, Sun position and navigation satellite constellations, fixed

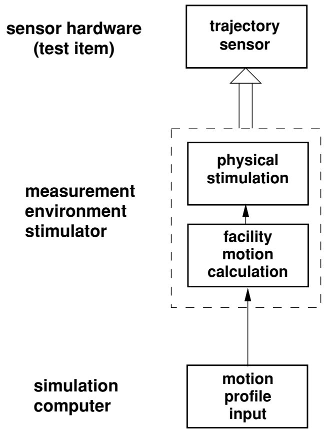

Figure 10.5. Open loop GNC stimulation of sensor hardware.

profiles can be used which have been calculated for the part of the approach to be tested. These open loop test setups with physical stimulation will be used as the primary means of validation of the mathematical models of the sensors, which will be used in the 'all software' simulations.

The non-systematic disturbance part of the measurement environment models used in the software simulations and for the stimulators can be validated only by practical experience in space, either by comparison with data from former missions or by dedicated in-orbit experiments (see section 10.7.2). Once the worst-case situations are known from experience, they can be used in the tests with physical stimulations to validate sensors and navigation function w.r.t. real world conditions.

GNC verification with sensor stimulation Eventually, stimulation facilities may also be included in the closed loop test of the GNC system (figure 10.6). These closed loop tests will be used mainly for the validation on the system level of the 'all software' simulation, in particular of the navigation function. As test preparations on such setups and test runs are complex, long and expensive, tests will have to be limited to a few particular test cases. Also, such test setups with physical sensor stimulation may not permit the reproduction of contingency situations, either because of physical limitations of the facility or because of operational safety of the test.

For RGPS, the physical stimulation will have to produce the RF data input to the GPS receivers on the chaser and target sides, as they would be received by their respective antennas according to the instantaneous GPS satellite constellation and to the actual position and attitude of chaser and target vehicles. To produce the RF data input, two GPS satellite constellation simulators will be required to simulate the position of the GPS satellites as seen by each of the two vehicles. For their input they will have to obtain

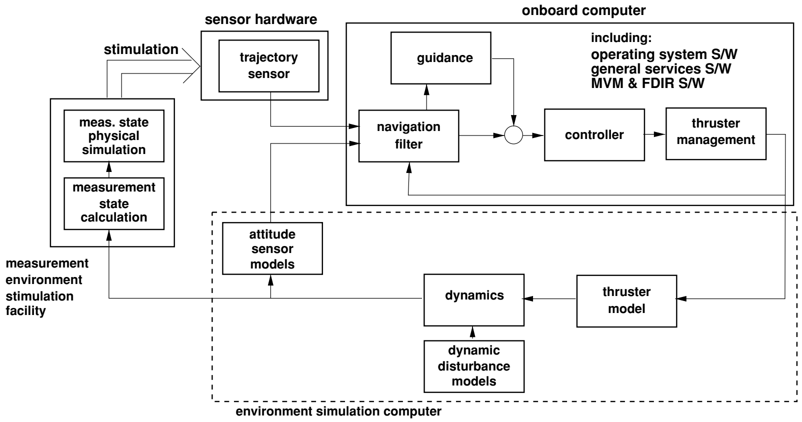

Figure 10.6. Closed loop simulation with GNC computer and navigation sensor hardware in the loop.

the actual position and attitude of chaser and target. The output of the receivers on chaser and target will be fed into the RGPS navigation filter of the GNC system, where the relative navigation data is actually produced (see figures 7.25 and 7.26). The modelling in the GPS satellite constellation simulation needs to include multi-path effects and shadowing according to the geometric conditions around the antennas on the chaser and target vehicles.

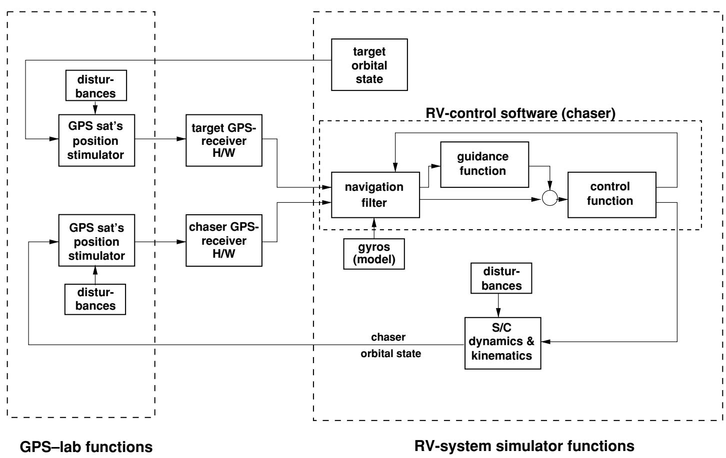

Figure 10.7. Measurement environment simulation for relative GPS navigation.

For optical rendezvous sensors, the physical stimulation needs to reproduce the conditions of the light signal to be measured by the sensor, whether emitted by the sensor illuminator and reflected by a reflector pattern on the target or emitted by target lights. This will require a motion system which changes the direction and distance of the light sources to be sensed in accordance with the relative motion between chaser and target during the trajectory part to be tested. Also, the stimulation will have to provide a realistic representation of the disturbances (measurement environment) that the sensor might be subjected to in the real mission. Such disturbances will, in particular, be specular reflections by the target surface of sunlight and light emitted by the illuminator of the sensor, and direct sunlight in the FOV of the sensor. As chaser and target will have independent attitude motions w.r.t. the Sun, the angular motion of this disturbance stimulation will need to be implemented independently of the relative motion between chaser and target, resulting in a total of at least eight DOF for the facility.

Following the above described steps of verification of the RV-control software in its data management environment and of sensor and navigation functions in their respective measurement environment, only a few tests with different sets of initial conditions and variation disturbance parameters will be possible because of time and cost reasons. In a real time simulation, the test of each part of the trajectory will take between one half and one full orbit, i.e. about 45–90 minutes. The time necessary for test preparation has to be added to this. An evaluation of the limits of the system behaviour, taking into account the full specification ranges for system and equipment and the uncertainty of knowledge of the environment, and considering all possible combinations and variations of these parameters, would require an excessive amount of time in a real time simulation setup. For the full performance verification of the system, an updated ‘all software simulation’ will be required, with which ‘Monte Carlo’ runs can be performed in faster than real time. Such simulations will be in principle similar to the one shown in figure 10.2; however they will also include the final design of the GNC algorithms, the detailed modelling of sensor and actuator functions and disturbances, and a complete representation of the MVM and FDIR functions and of the data management system.

Verification/validation of contact dynamics and capture

As for the trajectory control related systems and items, the front end functions of a docking system have to be verified under conditions as close as possible to the real application. This requires (a) the front end hardware to be tested under the same dynamic conditions as that which occur during the real docking in space, and (b) the dynamic reactions of both spacecraft to be physically available in the test setup.

Verification/validation objectives for contact dynamics and capture

- Proper reception with final kinematic and dynamic conditions of the two spacecraft, according to GNC performance at zero distance.
- Successful capture by the latches under the given dynamic and kinematic conditions at contact.

- Maximum forces and torques acting on the spacecraft during the contact and capture process.

Capture dynamics verification tools During the development of a new docking mechanism, an analysis tool (described in the previous section and indicated in figure 10.3) will be used to verify the limits of the capture capability of the design concerning approach and lateral velocities and lateral and angular misalignments. The validation of the mechanism design, and of the modelling in the simulation tool, requires a test setup in which the docking mechanism hardware will be subjected to the kinematic and dynamic conditions of contact, i.e. linear and angular velocities and misalignments. Such a test setup needs to simulate the physical motion of the front ends of the chaser and target vehicles and will have to model masses, inertias and flexibilities of the two vehicles and to calculate their kinematic reactions.

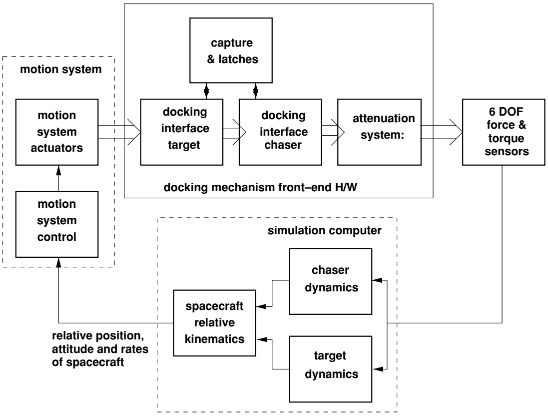

Figure 10.8. Docking system front end dynamic testing.

A block diagram of the typical functions required for such a facility is shown in figure 10.8. The six DOF physical motion will have to be provided by a motion system, which carries the front ends of the docking system on the chaser and target sides, which include the reception, capture and attenuation functions. Similarly to the physical stimulation facilities for the rendezvous sensor functions, the test of extreme or contingency situations on a contact dynamics facility will be limited because of the danger of damage to both the test item and the facility. As the time required for each test run will, however, be comparatively short, such a facility could be used also for the evaluation of optimum design parameters for capture.

The principles of verification of the other functions of a docking mechanism, i.e. retraction, structural latching and hermetic sealing, are no different from the verification of similar functions for ground applications. For this reason, there is no need to discuss them here.

Performance verification under thermal vacuum conditions

GNC functions Items which may be affected are the sensors and the actuators. Their behaviour can be tested on equipment level concerning the change in any performance parameter due to variations in environmental conditions. There will, however, be no need to test the complete system under thermal vacuum conditions.

Capture dynamic functions Considering the sensitivity of mechanisms to temperature and vacuum, all functions of the docking system need to be tested in a thermal vacuum. In particular the front end functions of attenuation and capture will have to be tested for change of performance with changing environmental conditions. For these tests, particular tools have to be developed in accordance with the actual design of those mechanisms. The performance of closed loop dynamic tests under such environmental conditions will, however, not be necessary, as the mathematical modelling can be validated at normal temperature in air, and the performance of the complete mechanism under thermal vacuum conditions can be obtained from the 'all software' capture simulation by introducing the changed parameters of the individual components, measured under those conditions. All other functions of the docking mechanism, such as retraction, structural latching and sealing must also be tested for proper functioning under thermal vacuum conditions, as postulated above. There is no need, however, to discuss such tests here in further detail.

10.3.4 Verification methods for operations and tools for remote operators

The verification of the space–ground system setup and of the requirements and concepts for support tools for remote operators will have to follow a different schedule from the verification of the automatic onboard rendezvous system, described in the previous sections. The tasks of ground operators in the chaser and target control centres and of crew members in the target station were identified and the requirements and concepts for support tools were discussed in the previous chapter. The development and verification of operational procedures governing the interactions between the players according to these tasks, and the verification of the support tools, can only start when the development of the onboard system design and its verification tools has progressed to a certain point and when real time software of both the onboard system and the environment simulation is already available. To avoid confusion with the development phases of the onboard system, the discussion and description of development and verification objectives and processes will be done here globally, without making reference to feasibility, design and development phases.

Overall verification/validation objectives for ground system tools and operations

- Proper function and performance of the support tools for ground and space operators.
- Proper interaction of chaser CC with chaser S/C during RV phases.
- Proper interaction of chaser CC with target CC during RV phases.
- Proper interaction of target crew members with the chaser vehicle in case of contingency operations.

Verification of support tool functions for ground operators and target crew

Items and issues to be verified

- Concerning the monitoring function of support tools for the ground operator and target crew member, it needs to be verified that the representation of trajectory, attitude and onboard functions status, warnings and messages is easily understandable by the human operator and will prompt him for the proper reactions.
- Concerning the immediate intervention and short term recovery functions, it needs to be verified that commands will be executed by the onboard system, that the intended trajectory/attitude changes will be achieved and that the operation of this function by the human operator will be easy and fast.
- Concerning the failure analysis function, it needs to be verified that onboard failures or contingency situations will be detected sufficiently quickly such that the human operator will be aware of them and can initiate pacifying or recovery actions in due time. Further, it will have to be verified that the identification of the location of the failure down to a certain level is possible, and that the specified tool will be able to support the operator during and following identification of the failure.
- Concerning the strategy and mission re-planning function, it needs to be verified that the tool is capable of quickly providing all data required for the implementation of a sequence of new trajectories, including the corresponding parameters of the applicable manoeuvres or GNC modes. It will further have to be verified that the results of this tool will be sufficiently accurate, such that they can safely be used for mission re-planning and trajectory implementation.

Verification methods

- *For the monitoring function.* Proper performance of the support tool can be verified by running it together with a real time simulator (representing the onboard system), which provides the relevant onboard data. The major criterion in the verification will be the subjective experience of one or more human operators concerning the ease of understanding of the displayed information and its usefulness in the operation.

- *For the immediate intervention and short term recovery functions*. As above, performance of the support tool can be verified by running it together with a real time simulator representing the onboard system, in which the onboard RV-control system accepts and executes the commands issued by the remote operator via this function.

- *For the failure analysis function*. A full verification of such a function will be difficult, as this would in principle require the availability of the complete onboard system with all hardware and software in the test setup and the capability of producing any possible failure conditions. In a setup with the real hardware and software it would be difficult, however, to reproduce all possible failures or any particular one at any particular time. The only way to test the failure analysis function would be to run it together with a simulation of the onboard system, which includes a sufficiently detailed design modelling of sensors and the data management and propulsion systems, where failure conditions can be introduced on command.

- *For the strategy and mission re-planning function*. This function will have to consist of a fast simulation with detailed GNC, propulsion and environment modelling. It can be operated and verified off-line, assuming several particular contingency situation, e.g. after a CAM, retreat, long duration waiting, etc. The verification will consist of a comparison with a simulation of the same trajectory sequence, manoeuvre and GNC mode parameters using the most accurate simulation tool available.

Verification tools

- *For the monitoring function*. This requires a setup consisting of the operator support tool running together with the real time RV-system simulator, i.e. the RV-control software in the onboard computer hardware and the real time environment simulation. In addition, a communications simulator will be needed which models effects such as the space–ground time delay, communications windows, shadowing, signal-to-noise ratio, etc.

- *For the immediate intervention and short term recovery functions*. The same setup is required, i.e. the operator support tool running together with the real time RV-system simulator.

- *For the failure analysis function*. This requires a setup consisting of the failure analysis tool together with an all-software RV-system simulator with detailed modelling of the design of all subsystems and equipment.

- *For the strategy and mission re-planning function.* A fast non-real time RV-system simulator is necessary which could be, e.g., the non-real time RV-system simulator used for Monte Carlo simulations.

Verification of interactions between control centres and vehicles

Items and issues to be verified

- The operating procedures for nominal and contingency situations, and, together with these procedures, the interaction of chaser CC with chaser vehicle during RV phases.
- The operating procedures for interactions between chaser CC and target CC in the last phases of the rendezvous mission, e.g. hand-over of mission authority, decision making for approach continuation at hold points, contingency handling, etc.
- The operating procedures for interactions of the target CC with the chaser vehicle in the case of contingencies in the last part of the final approach.
- The operating procedures for the interaction of the target crew member with the chaser vehicle in the case of contingency operations.

Verification methods The real time simulation output of the state vector, of the navigation data and of the subsystem and equipment status of the chaser vehicle, must be available for that part of the mission which is to be monitored at one or both control centres and for the last part of the approach in the monitoring equipment for the operator in the space station.

Delays, noise and other constraints or disturbances of the communication flow between space and ground must be simulated. If the control centres for the chaser and target are at different locations, the real communication lines (or technically equivalent ones) should be used in the final verification test setup for communication between ground centres.

Verification tools The basic simulation setup will consist of the real time RV-system simulator, i.e. the RV-control software in the onboard computer hardware and the real time environment simulator, plus the communication simulator, which represents the constraints and disturbances of the space–ground communication links.

If the chaser and target control centres are at different locations, the final verification tests can make use of 'distributed interactive simulation' techniques (Miro *et al.* 1998; Vankov & Arguello 1998; Arguello & Miro 2000), for which a detailed simulation of each spacecraft will run in a different place, i.e. the chaser vehicle dynamic simulation

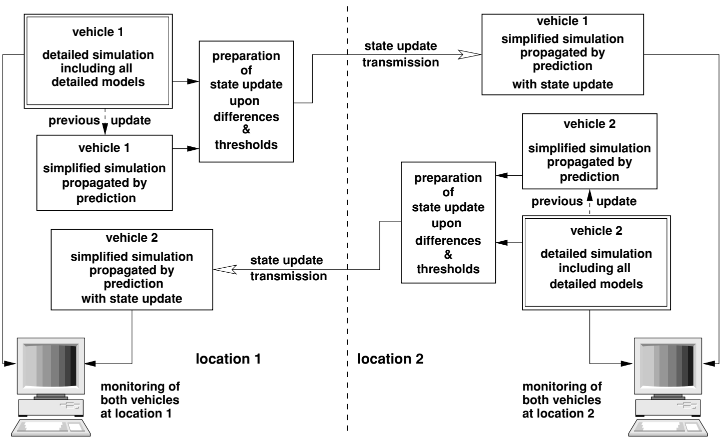

Figure 10.9. Principle of distributed interactive simulation.

in the chaser CC and the target vehicle dynamic simulation in the target CC, and a simplified dynamic simulation of each vehicle in the opposite CC. Each simplified dynamic simulation will be updated continuously via the ground communication links by state vector information from the detailed one. Communication delays between the CCs will be compensated for by a lead/lag algorithm. The principle of a distributed interactive simulation is shown in figure 10.9. The advantage of such an arrangement would be that each operator could remain in his normal environment, that the actual constraints and disturbances of the ground links will be included in the test, and that the space segment simulation of each side can be synchronised with the other to form an overall space–ground system simulation.

10.3.5 Flight item manufacture phase

Following production of the flight items, the risk introduced by manufacturing errors has to be eliminated. It has to be verified that the flight items ‘as built’ are in all aspects ‘as qualified’, i.e. that the qualification in terms of physical properties, functioning and performance performed on the final development items are fully applicable to the flight items.

To make this verification, not all of the tests performed for qualification need to be repeated. The verification of conformance of an item ‘as built’ with the design ‘as qualified’ usually will require less effort, as confidence in proper manufacture can be established by a few physical tests and simulation runs. This will have to include the verification of functioning and performance under nominal (and possibly a few contingency) conditions, the verification of all interfaces and the verification of physical identity with the qualification items. It is important that flight item verification is performed up to the highest level, i.e. system integration and integration into the spacecraft, in order to verify proper integration and connection of all interfaces.

GNC verification tools

The rendezvous control software must, in addition to being subjected to comprehensive testing on software level, be acceptance tested in the real time RV-simulation setup with the data management hardware and software in the loop. This will be the easiest way of ensuring that the flight item hardware and software will work together without problems. The acceptance tests can be limited to the nominal mission sequence and a few contingency cases, in order to establish that there is no deviation in behaviour between qualification and flight software.

Except for the testing of the RV-control software in the data management hardware environment, there will be, in the flight item manufacture phase, no need for a GNC performance test with hardware in the loop. Hardware items will be tested individually in their own acceptance test programme using their own equipment level test tools and facilities. GNC level tests with sensor hardware in realistic measurement environments will not be necessary for acceptance, as this behaviour can be considered to be design dependent, rather than manufacture dependent. An end-to-end test with all hardware and software in a chain will have to be performed on a spacecraft level, using electric stimuli for the sensors and other hardware functions. The objective of such tests will be the verification of proper functioning, rather than performance, of the individual items and of the complete chain.

A similar final electrical test of the GNC system will have to be performed prior to flight, which will form part of the checkout of all functions of the spacecraft. As the spacecraft is completely assembled, this test can be, however, less comprehensive than the previous one.

Capture dynamics verification tools

The acceptance testing of the front end of the docking mechanism flight item will consist of both the individual tests of the attenuation and capture mechanisms under laboratory and thermal vacuum conditions and the test of the complete system under laboratory conditions on a six DOF motion facility. The verification tools will be the same ones as used in the development phase.

In contrast to the real time testing of the RV-control software, which will, as explained above, always take considerable time and will, therefore, be limited to a few test cases, the test of a single case of contact and capture will not take more than a few minutes.

For this reason, it will be worthwhile testing the flight item of the docking system within the full range of dynamic conditions specified for the GNC performance at contact.

10.4 Modelling of spacecraft items and orbital environment

In a test performed to evaluate the behaviour of an item or function under 'real world' conditions, every other item or feature involved in the system or process of the 'real world' must either be present in reality or be represented by an equivalent mathematical model. In a verification test setup, we call the function or item to be tested the *test item* and all other functions and features representing the 'real world' the *environment*.[1] For instance, when testing a sensor in its system environment, i.e. with the GNC system in closed loop, the sensor is the test item and the GNC algorithms and data management system belong to the environment. On the contrary, when testing the GNC algorithms in their system environment, i.e. with the sensor hardware in a closed loop, the algorithms are the test item and the sensors belong to the environment. Mathematical modelling is always related to items or features of the *environment*.

As explained in the previous section, modelling will start in the early stages of a project with simple behaviour models. For spacecraft equipment, such as sensors and actuators, the initial models will include the basic output function plus some errors or disturbances, modelled as bias and noise. For the measurement environment, this will be the basic effects; e.g., for GPS the basic effects are the positions of the individual navigation satellites. Orbital disturbances, such as drag or differential drag, will initially be modelled as constant forces. In the very early trajectory feasibility analysis, i.e. in the beginning of the mission definition and feasibility phase, ideal sensors and actuators with no errors are assumed, and ΔVs are applied as an impulse, rather than a force with a certain duration (see chapter 3). During the course of development, these models will have to be improved and refined, such that they will eventually represent realistically the spacecraft items, measurement environment and orbital disturbances required for the qualification of system and items for flight.

This section will identify the type and the contents of the models needed to satisfy the verification requirements for the final stage of the development process. Model requirements for the RV-control system and for the capture system of the docking mechanism will be addressed. Since these models depend on the actual design of spacecraft and equipment, it would not make much sense, however, to discuss their design details here. In many cases, the basic functions of the models have been addressed in the previous chapters of this book, which is indicated in the following by reference to the appropriate sections, figures or equations.

[1]Note, the term 'orbital environment' includes only that part of the simulation or test environment that relates to the effects in orbit experienced by the spacecraft.

10.4.1 Modelling of environment simulation for RV-control system test

In the example shown in figure 10.10, it is assumed that the test item is the complete RV-control system, i.e. the GNC, MVM and FDIR algorithms processed in the onboard computer. This is the type of test shown in figure 10.4. Figure 10.10 shows the various models of the environment simulation required for a closed loop rendezvous simulation with the described test item. It also shows the input and output data interfaces between the environment simulation and the test item. The inputs to the environment simulation are the GNC commands to the propulsion system for the execution of the actuation forces and torques and the information on orbit parameters and spacecraft attitude for the determination of the orbital disturbances. The outputs of the environment simulation to the RV-control system are the raw data of the GPS receivers of chaser and target (see figure 7.25), the absolute attitude of the chaser and (in the last part of the approach) range, line-of-sight and relative attitude angles provided by the optical rendezvous sensor.

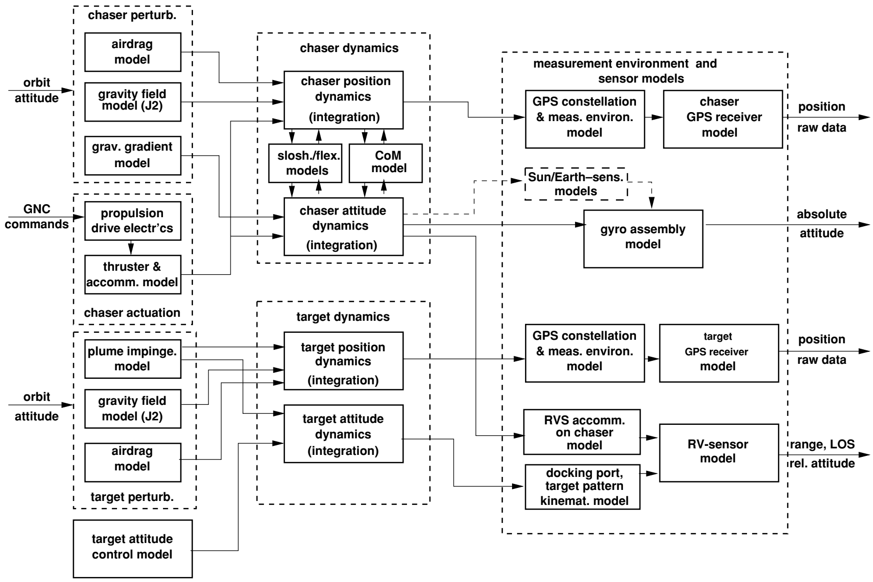

Figure 10.10. Simulation models for spacecraft items, dynamics, kinematics and orbital environment.

The models used in the environment simulation can be broken down into the following major groups:

- chaser actuation models,
- chaser perturbation models,

- chaser dynamics models,
- target perturbation models,
- target dynamics models,
- sensor and measurement environment models.

The major inputs/outputs to/from each model are also indicated in figure 10.10. However, in order not to comlicate the diagram, not all connections are shown (e.g. the inputs to the perturbation models, taken from the position and attitude dynamics output of chaser and target are not shown). In addition to the above model groups, the figure shows a 'target attitude control model', which models, in a simplified way, the control torque inputs to the target attitude dynamics. Where a more detailed description of the target GNC is not available, or such a detailed modelling is not necessary for the purpose of performance testing of the chaser GNC, the attitude motion of the target can be modelled by inputting a fixed command profile, e.g. a saw-tooth profile, into the target attitude dynamics model.

The major features, which have to be represented in these models are listed in the following.

Chaser actuation models

- *Propulsion drive electronics model:*
  - the time delay between the actuation of thruster valves and commands from the thruster management function, which is part of the GNC algorithms;
  - a representation of 'built-in test equipment' producing data on failed thrusters as input to MVM and FDIR;
  - data on redundant string main valve status as input information to MVM and FDIR.
- *Thruster accommodation model:*
  - location and angles of thruster accommodation on spacecraft;
  - errors of location and angles of thruster accommodation (see section 4.3.2);
  - thrust direction and magnitude errors due to plume impingement of spacecraft structure (see section 4.2.4) – this has to be modelled individually for those thrusters affected, depending on accommodation location and angle and on the design of the spacecraft structure.
- *Thruster equipment:*
  - minimum impulse bit (MIB), thrust magnitude and thrust pulse characteristics as a function of 'thruster-on time' (see figures 6.14 and 6.15);

- thrust direction and magnitude uncertainties (see section 4.3.2);
- 'thruster open' and 'thruster closed' failure modes.

Chaser perturbation models

- *Absolute and differential air drag (see section 4.2.1):*
  - evolution of air density along the orbit due to the temperature difference between the illuminated side and the shadow side of the orbit;
  - change of the cross section of the chaser along orbit due to the solar arrays rotating with the Sun direction;
  - determination of the absolute drag force acting on the chaser, calculated from the evolution along the orbit of the chaser cross section and the air density;
  - determination of torques due to drag. These occur when the centre of pressure and the centre of mass do not coincide in the cross section of the vehicle normal to the velocity vector. The CoM position will be obtained from the mass, inertia and CoM model below;
  - determination of the differential drag by subtraction of the absolute drag values of chaser and target;
  - correction of the differential drag value at close distance by determination of the areas on the chaser which are shadowed by the target structure.
- *Gravity field, J_2-effect (see section 4.2.2):* forces, due to the deviation of the gravity field of the Earth from an ideal sphere, which change the orbit of the vehicle (see Eqs. (4.8) and (4.9)).
- *Gravity gradient:* torques acting on the spacecraft body due to the gradient in the gravity field of the Earth in the radius direction (this attitude disturbance has not been discussed further in this book, since the effect is well analysed and described in all textbooks on attitude control).
- *Thruster plume interaction (see section* 4.2.4)*:*
  - forces and torques due to the impact on the surfaces of the chaser vehicle of the thrust plumes emitted by the target vehicle;
  - kinematic model of location of thruster on target vehicle and of relative position and attitude between the two vehicles.

Note: the modelling of the target thruster plume interaction may not be necessary if target thrusters are inhibited during docking.

Chaser dynamics models

- *Chaser position dynamics:*
  - numerical integration of the Hill Eqs. (3.21) (note: the Clohessy–Wiltshire linearised solutions Eqs. (3.22) would be not accurate enough). Parameters are the vehicle properties, i.e. the mass and the CoM position of the spacecraft body, determined by the 'mass, inertia and CoM' model described below. Input values are the forces acting on the spacecraft, i.e. the thrust forces, the air drag forces and the J_2-effect;
  - internal features, modifying the solid body dynamic behaviour in terms of resonant frequencies, bandwidth, damping behaviour, etc., are flexible appendages, e.g. solar arrays and antennas, and the sloshing of liquid propellant in the tanks (see below);
  - correction of thrust force vectors by results from CoM position model. These will be necessary because there is in the general case not a single tank located in the geometric centre (see below).
- *Chaser attitude dynamics:*
  - numerical integration of Eq. (A.83), for which the input values are the inertia tensor of the vehicle determined by the mass, inertia and CoM model described below and the torque inputs, i.e. the control torques by the thrusters and the disturbance torques by drag and gravity gradient effects;
  - as for the position dynamics, internal features, modifying the solid body dynamic behaviour in terms of resonant frequencies, bandwidth, damping behaviour, etc., are flexible appendages, e.g. solar arrays and antennas, and the sloshing of liquid propellant in the tanks (see below);
  - correction of thrust force vectors by results from CoM position model, as described for the position dynamics model.
- *Flexible appendages:* part of the dynamics model block, modelling the dynamic interaction between items flexibly attached to the main body of the spacecraft and the rigid body dynamics. These are items such as solar arrays or antennas, which are usually modelled as separate masses attached to the main body by spring-damper links.
- *Fuel sloshing:* this model is part of the dynamics model block, modelling the motion of the propellant liquid in the tanks as a result of linear and angular accelerations. Several models are available, the most simple one being a pendulum model. The pendulum model, in which a single mass representing the propellant mass is hinged with three DOF motion capability at a point representing the centre of the tank, is typically used in simulations of the concept feasibility stage. In the final design stage more detailed models have to be used, taking into consideration the

shape and type of tank (e.g. surface tension tank, diaphragm or bladder tank) and the motion of liquid in those tanks under zero-g conditions.

- *Mass, inertia and CoM position:*
  - the changes of mass, inertia and CoM of the spacecraft during the mission, due to the consumption of propellant. The input to the model is the actual propellant consumption, determined from the thrust commands and from the scheme governing the depletion of tanks;
  - the inertia changes during one orbital revolution due to the rotation of the solar arrays.

Target attitude control modelling

- *Simplified modelling*. As already noted above, the modelling of the target can be kept relatively simple, as long as only the orbital motion of its CoM is of interest for the relative position determination of the chaser, and as long as only its typical attitude motion needs to be simulated to obtain realistic results for the docking port motion as inputs for the optical rendezvous sensor in the last part of the approach. If there is no need to represent the target attitude control by a closed control loop, attitude disturbances resulting from gravity gradient and air drag effects do not need to be modelled for the target. The target attitude evolution over time can then be modelled by a bias attitude, representing the torque equivalent attitude, where all external torques are in balance, and by a simplified attitude command model, representing, e.g., the torque commands in a limit cycle type of attitude control.
- *Closed loop modelling*. If a closed loop attitude control system is to be modelled for the target, disturbances and dynamic modelling will be the same as for the chaser. As actuators, the target may have, in addition to thrusters, reaction wheels or control-moment gyros. In the following, the simple type of modelling is assumed.

Target perturbation models

- *Absolute air drag model:*
  - evolution of air density along the orbit due to the temperature difference between the illuminated side and the shadow side of the orbit (input from chaser modelling);
  - change of the cross section of the target along orbit due to the solar arrays rotating with Sun direction;
  - determination of the absolute drag of the target, calculated from the evolution along the orbit of the target cross section and the air density.

- *Gravity field, J_2-effect (see section 4.2.2):* the forces due to the deviation of the gravity field of the Earth from an ideal sphere, which changes the orbit of the vehicle (see Eqs. (4.8) and (4.9)) (as in the chaser model, the numeric values will be different from the chaser values only as long as the target is at a different altitude from the chaser).

- *Thruster plume interaction model (see section 4.2.4):*

  – forces and torques due to the impact on the target surfaces of thrust plumes emitted by the chaser;

  – kinematic model of location of thruster on chaser vehicle and of relative position and attitude between the two vehicles.

Target dynamics models

- *Target position dynamics*. Numerical integration of the Hill equations Eqs. (3.21) with the absolute drag as the input. In the simplified modelling of the target motion it can be assumed that no position control of the target during the rendezvous approach of the chaser takes place. If the target performs position control during the approach of the chaser or other control forces are applied, an open loop control force profile can be used as the input to the position dynamics model in the same way as for the attitude control torques. In addition to the position control, residual forces may result from thruster operations during de-saturation of reaction wheels or control moment gyros.

- *Target attitude dynamics*. If the target attitude is not just defined by a simple kinematic model, numerical integration of the Eq. (A.83), with fixed inertia characteristics and control torque inputs from the target attitude control model.

Sensor and measurement environment models

Satellite navigation models *(see section 7.3)*

- *Navigation satellite position constellation for chaser:* determination of the navigation satellite positions w.r.t. the instantaneous position of the chaser.

- *Satellite navigation disturbance models (shadowing, multi-path) for chaser:*

  – fixed visibility constraints of chaser antennas for satellite navigation due to antenna characteristics, chaser structure and chaser attitude;

  – shadowing of navigation satellites by moving structural items on chaser, e.g. solar arrays. This model provides additional temporary masking features to the previous one;

- shadowing of navigation satellites by target structure;
- multi-path effects.

- *Satellite navigation receiver model for chaser*. Model of chaser satellite navigation receiver representing the basic functions (see section 7.3.2) and the particular features of the individual design. This model needs to include all the features that have any kind of interaction with the GNC, MVM and FDIR functions of the RV-control software. As a minimum the following values must be available from the model:

  - raw measurements with accuracy representative for receiver design,
  - output rate of data, accuracy of time measurement representative for receiver design,
  - time required for filter convergence,
  - criteria for switching between antennas,
  - criteria and functions for redundancy switching to redundant channels,
  - built-in test functions.

- The satellite navigation receiver model must represent the internal functions of the receiver equipment in such detail that the listed features and performances result from the modelling.

- *Navigation satellite position constellation for target:* determination of the navigation satellite positions w.r.t. the instantaneous position of the target.

- *Satellite navigation disturbance models (shadowing, multi-path) for target:*

  - fixed visibility constraints of chaser antennas for satellite navigation due to antenna characteristics, target structure and target attitude;
  - shadowing of navigation satellites by moving structural items on target, e.g. solar arrays. This model provides additional temporary masking features to the above one.

- *Satellite navigation receiver model for target:* same modelling as for chaser.

Optical rendezvous sensor models (*see section 7.4*)

- *Optical rendezvous sensor accommodation and kinematics model*:

  - relationship between the linear and angular motion at the position of the sensor and the angular motion of the CoM of the chaser;

   - relative position and relative attitude between sensor location and optical axis on chaser, and reflector pattern on target from relative position and attitude at spacecraft CoMs, target docking port kinematics model and chaser sensor location model.

- *Target docking port and reflector pattern kinematics model:*
   - relationship between the angular motion of the CoM of the target and the linear and angular motion at the docking port;
   - position of the sensor reflector pattern in the docking port plane.

- *Measurement environment model for rendezvous sensor:*
   - received light power as a function of range (Eq. (7.45) for laser range finder and Eq. (7.54) for camera type of sensor), signal-to-noise ratio criteria;
   - Sun in FOV of sensor determination;
   - specular reflection representation (if needed – sensor may not be sensitive).

- *Optical rendezvous sensor equipment model.* Rendezvous sensor model, representing the basic measurement function (see sections 7.4.1 and 7.4.2) and the particular features of the actual design of the equipment. The model must include all features and parameters that have any interaction with the GNC, MVM and FDIR functions of the RV-control software. As a minimum the following values must be available from the model:
   - sensor FOV,
   - performances,
   - operational limits for measured parameters,
   - bandwidth, output rate, delay of information,
   - built-in test functions.

  As for the satellite navigation receiver model, the rendezvous sensor model has to represent the internal functions of the equipment in such detail that the above features and performances are a product of the modelling.

Other sensors

- *Gyro assembly model.* Due to the host of applications on ground and in space, detailed models of gyro assemblies exist which can be adapted to the particular mission application. The model will have to include particular redundancy features, such as skewed gyros as backup for more than one measurement axis, with the corresponding reduction in accuracy. Also, as for rendezvous sensors and GPS receivers, all features and parameters that have any interaction with the GNC, MVM and FDIR functions of the RV-control software have to be modelled.

- *Sun and Earth sensors*. These models are well established, since they are used in the attitude control systems of a very large number of satellites. Modelling of the basic measurement function plus noise and bias will be sufficient in most cases. Existing models can easily be adapted to the particular items used in the mission.

10.4.2 Modelling for contact dynamics simulation

The basic groups of models used in a contact and capture simulation are shown in figure 10.3. The spacecraft features that play a role in the modelling of the dynamics of contact and capture are shown in figure 10.11. The detailed modelling will very much depend on the type of mechanism design (see section 8.2.5).

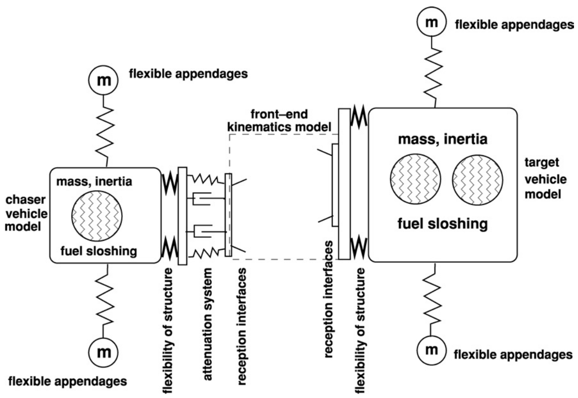

Figure 10.11. Principle of spacecraft modelling for contact dynamics analysis.

The following features have to be included in the modelling:

- *Relative kinematics between chaser and target:* determination of the relative position and relative attitude angles between the spacecraft at their CoMs. The initial conditions, which will be given as the input to the simulation, are the approach velocity and GNC performance values at the test start distance. In the subsequent computation steps, relative position and attitude are determined from the difference of the position and attitude output provided by the dynamics models of chaser and target.

- *Front-end kinematics model:*
  - geometric representation of reception configuration, i.e. rod–cone or contact rings with petals;
  - determination of relative position and angles of front end geometric features on chaser and target from the relative kinematics of the spacecraft and from the distance of these features from the spacecraft CoM;

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  - determination of shortest distance between front end geometric features of chaser and target, and determination of contact point when distance becomes zero;
  - determination of contact force direction from the geometric features of chaser and target front ends at the contact point.

- *Capture latch kinematic model.* The latch model needs to include, as a minimum:

  - position of capture latches as a function of time after initiation;
  - comparison of latch positions with the position of their interfaces on the target side;
  - determination of latched condition.

  This model is closely related to the attenuation dynamics and front end kinematics models. Its design will depend on whether the capture latch is connected to the attenuation system (as, e.g., in the central docking system, see figures 8.8 and 8.25), to the contact ring (as in the peripheral docking system with passive capture latches, shown in figure 8.26), or to the docking ring (as in the peripheral docking system with active capture latches, shown in figure 8.27).

  In the case of a passive, spring-loaded capture latch, the model will have to determine whether the position and angles for engagement of the latches are fulfilled. It will also have to model the spring and friction forces acting between the spacecraft due to the capture latches.

  For active latches, the model will have to include the criteria for initiation of the latch operation, the kinematics of the latch motion, the determination of latch contact with the target interfaces and the forces and their direction applied by the latches to chaser and target.

- *Attenuation dynamics model.* Determination of the evolution over time of the deformation of the shock attenuation devices and of the forces acting between the spacecraft after contact, as described in section 8.3. Inputs are the contact point and force direction, determined by the front-end kinematics model and the instantaneous velocity vector.

- *Chaser body dynamics model.* Determination of the linear and angular accelerations of the chaser spacecraft body as a result of the forces transmitted by the attenuation system. For large spacecraft, the flexibility of the substructure of the docking port, flexible appendages and fuel sloshing may have to be taken into account for the calculation of the dynamic reactions, as shown in figure 10.11.

- *Target body dynamics model.* This model is, in structure and content, similar to the chaser body dynamics model, representing the structural design of the target, relevant to forces applied at the location of the docking port.

10.5 Validation of models, tools and facilities

All the models discussed above, and eventually also the tools and facilities in which they are used, need to be validated prior to use in verification tools for the onboard system. According to the definitions given at the beginning of this chapter, this means that sufficient confidence must be obtained that the models (individually for a particular effect) and the tools/facilities (globally for the complete environment of the test item) represent reality to the extent necessary for the verification to be performed. A validation can be achieved in principle by:

(1) comparison of model/simulation output with data obtained from physical testing of the item or the effect in question under the same conditions;

(2) comparison of model/simulation output with data derived from real space missions, where model or simulation parameters are tuned to that particular mission – if results conform to the conditions of an existing mission, confidence in the applicability of the model for other mission conditions will be increased;

(3) comparison of a mathematical model with a model of the same kind which has been obtained from other sources and which has been already validated;

(4) comparison of results of the complete simulation with results of other simulations of the same kind which have been obtained from other sources and are already validated.

10.5.1 Validation of GNC environment simulation models

Orbital perturbation models

For orbital features which are independent of the design of the spacecraft, its subsystems or equipment, validated models are, as a rule, already available from the flight experience and development work of former missions. This is the case for the orbital environment, e.g. for models of the residual atmosphere and for the anomaly (J_2-effect) and the gradient of the Earth gravity field. Although such environment models may be well established and validated, there may still be a residual uncertainty concerning the actual values to be expected when the mission is performed. This is true in particular for the residual air density, which varies with solar flux (see section 4.2.1).

In cases where the perturbation model depends on the actual design of the spacecraft in addition to the orbital environment, such as in the case of absolute and differential air drag forces and gravity gradient torques, the relevant geometric models of the spacecraft which are representative for the particular case must first be established and validated:

- For air drag, the model must represent all surfaces of the vehicle which are perpendicular to the direction of flight. To determine the disturbance torques due to drag, the centres of pressure of these surfaces and their distance from the CoM of the vehicle must be calculated.

- For the gravity gradient effect, a model has to be established which represents the distribution of masses on the spacecraft.

These models will have to be refined commensurate with the design evolution of the spacecraft, starting from rough preliminary values in the concept definition and feasibility phase up to the final versions, where the detailed design of the spacecraft, its moving parts, such as solar arrays and articulated antennas, the spacecraft attitude and the change of mass, CoM and inertias during flight have to be taken into account. Although the measured data for the mass and geometry of parts, and of the complete vehicle, will eventually become available once the spacecraft has been manufactured, validation of these geometric models can be obtained only by analysis and comparison of the model with the hardware design of the vehicle.

Since this type of disturbance model is obtained by a combination of orbital environment models and spacecraft geometric models, the combined model cannot easily be validated experimentally. The method of modelling can be validated, however, by applying it retroactively to spacecraft designs which have already been flown and for which flight data are available. For instance, the drag forces can be calculated from the decay of a spacecraft orbit. As the cross section of the vehicle is known, the drag coefficient C_D (see Eq. (4.1)) can be calculated if the residual air density is known; or, vice versa, if the drag coefficient is known with sufficient confidence, the density can be determined.

Plume interaction models

As in the case of drag, this is a combined model, depending on the plume properties of a thruster, on the geometric accommodation of the thruster, on the properties of the surface geometry of the opposite spacecraft and on the relative position and attitude between the two vehicles. Concerning the thruster plume properties, the model for the pressure field of the plume (see Eq. (4.11)) has to be validated. This has been done in many cases by measuring in a vacuum chamber the pressure distribution over the cross section and/or the forces acting on a plate at various distances from the thruster. Plume forces of a thruster exist and have measurable effects also at distances larger than those available in a vacuum chamber. As the plume expansion in a vacuum is well known, the pressure magnitude and distribution as a function of range can be calculated from the measured pressure field at shorter distances.

Spacecraft dynamic models

The equations of motion, i.e. Eqs. (3.21) for translational motion and Eq. (A.83) for rotational motion, are the basic laws of mechanics, which can be assumed to be proven. The accuracy of the numerical integration method used can be proven by mathematical means. The remaining models for the spacecraft dynamics which depend on the spacecraft design and which, for this reason, need to be validated are:

- the evolution during flight of spacecraft mass, CoM position and inertia,

- dynamic interaction caused by flexible appendages,
- dynamic interaction caused by fuel sloshing.

For the final spacecraft design, none of these models can be validated directly by experiment, as this would require the mass changes during flight and the dynamic reactions under zero-g conditions to be observed on the real spacecraft. In these cases, the method of validation will be to use a proven method of modelling and cross-checking by independent analysis to verify that the model provides correct, or at least credible, results for particular sets of parameters.

Evolution of mass, CoM position and inertia The evolution of mass, CoM position and inertia, which is a function of the propellant consumption and tank usage, can be validated only by analysis, e.g. by calculating precisely, for particular points in the mission time-line, the propellant consumption and the change of propellant mass in the tanks. Considering the uncertainties in propellant consumption and the liquid motion in the tanks, the values for mass, CoM position and inertia will also have a margin of uncertainty which cannot be improved by better modelling.

Flexible appendages Dynamic interaction of flexible appendages affects the dynamic response of a body to input forces and torques and has repercussions on the control performance, most significantly on attitude control. The methods used to model flexible appendages and the design of controllers to overcome the effects are well understood and have been experimentally verified and validated for particular configurations. Validation of the model for the actual spacecraft design in question can be performed with a sufficiently high level of confidence by analysis.

Fuel sloshing Models for the motion of liquids in a vessel under zero-g conditions are difficult to be validated experimentally on ground.

Fuel sloshing provides dynamic uncertainties in the attitude control, a phenomenon which is present in practically every satellite. For this reason, a lot of theoretical and experimental work has already been carried out. Theoretical work includes finite element fluid dynamics analysis. Experimental work has been done, using, e.g., drop towers and parabolic flights, which provide a short-time zero-g environment. A number of attempts have been made with orbital experiments, the most recent and most comprehensive one being the Sloshsat 'Flevo' project, was planned, at the time of writing, to be launched by the US Space Shuttle in 2003 (Vreeburg 1999 a,b).

Validation of a sloshing model for a particular spacecraft design can be performed only by analysis. The confidence level in such a model, depending on its level of detail, will be moderate. Comparing the magnitude of effects with other disturbances, however, simulation errors will not be very high for the purpose of a rendezvous mission.

Sensors

For satellite navigation receivers, optical sensors, gyro packages, etc., the models will be derived from the actual design and will be validated by comparison with the results of equipment tests on stimulation facilities as indicated in section 10.3.3, figure 10.5 and as described in section 10.6 below. Such tests need to include the evaluation of the types of values and parameters which have been listed for satellite navigation receivers and optical rendezvous sensors in section 10.4. The objective of orbital demonstrations of the sensor equipment will rather be to increase the confidence in proper performance of the equipment than to validate the model.

Measurement environment

Correct representation of sensor performance in a simulation depends both on the modelling of the sensor functions and on the modelling of the measurement environment.

The systematic part of the measurement environment, which is part of the sensing process, consists of defined geometric and time relations. For satellite navigation receivers, this is the position constellation of the navigation satellites w.r.t. the receiver position and satellite and receiver clock times. For optical sensors, this is the target reflector position, which is obtained from the accommodation and kinematics models for sensor and reflector hardware. This part of the model can be validated at high confidence level by analysis.

For the disturbance part of the measurement environment, i.e. shadowing and multipath effects in the case of satellite navigation receivers and, for optical sensors, spurious reflections of sensor illuminator or sunlight or direct sunlight in the FOV, validation of the models will be a very difficult task. Many experimental investigations, using measurement data from test setups and from real space missions, will be necessary to achieve sufficient confidence in such models. The requirement of validation of these disturbance effects in scope, magnitude and frequency of occurrence is one of the driving forces for building stimulation facilities (see section 10.6) and for performing experiments and demonstrations in orbit (see section 10.7).

Thrusters

Validation can be performed to a large extent by analysis. Due to the long experience with thrusters of all sizes and of many different design types, detailed modelling is available, and good confidence exists in modelling of thruster behaviour and performance. Nevertheless, for each individual design, the thrust level, minimum impulse bit, on/off profile and similar parameters should preferably be validated experimentally at least once for each design. Such experimental validation can be performed by comparison of the model output with results of thruster testing in a vacuum chamber.

10.5.2 Validation of contact dynamics simulation models

Chaser dynamic model

The model can be based on the one established for the GNC simulation, but needs to be modified to include the stiffness of the chaser body at the docking port. Validation by analysis will provide a sufficiently high level of confidence.

Target dynamic model

As for the chaser, the model established for GNC simulation can be used as a basis for and modified to include the stiffness of the target body at the docking port. Validation by analysis will provide a sufficiently high level of confidence.

Spacecraft relative kinematic model

This is a purely geometric model of relative position of the CoMs and relative attitude of vehicles. The current values in the simulation will be obtained by propagation of the initial conditions by integration of the dynamics output. Validation can easily be performed by analysis.

Front end kinematics and contact detection model

Validation is required of the determination of the points of shortest distances at each point in time and the determination of the force direction at contact. Validation will be performed in the development life-cycle, first by analysis for a particular set of parameters comprising initial position and velocities and relative attitude and angular rates. Final validation will be achieved by comparison with results from tests of real mechanisms on a docking dynamics test facility.

Attenuation dynamic model

For the validation of the spring and damper models, the elements can be physically tested and the results compared with the model output. For complex spring-damper arrangements further validation will be performed by analysis.

The model of the complete attenuation system could be validated by physical testing of the integrated attenuation system hardware, i.e. by measuring forces and displacements following well-defined impacts. Final validation can eventually be achieved when the complete docking mechanism is tested on a docking dynamics test facility.

Capture latch kinematic model

The modelling of the relative motion of all capture latches w.r.t. their counterparts (catches, interface ring) has to be validated. This includes the planar motion of a single capture latch and the six DOF relative motion of the structures that they are mounted on. Capture success depends on both the kinematic condition of the latch and the front ends on both sides. The model is purely kinematic and can easily be validated by analysis. Again, final validation will be achieved by comparison of model and test results, once the complete docking mechanism is tested on a docking dynamics test facility.

10.5.3 Validation of simulator programs and stimulation facilities

Even when all models in a simulation program are properly validated, residual risks remain: the various parts of the program may not interact correctly, i.e. they may interfere with each other because of dynamic incompatibilities; there may be undetected hardware and/or software problems. Validation of proper functioning and performance can be achieved:

- by comparison with results from other simulations, on running the same test case on both simulations,
- by comparison of test results with flight data from a previous mission using the same parameters for initial conditions, disturbances, etc. in the simulation.

The validation of stimulation facilities depends not only on the type of sensor, but, as for the measurement environment models, also on the objective of the stimulation to be produced, i.e. whether the objective is:

- to test if correct measurement data are produced from the input data obtained by the sensor from the measurement environment,
- to test, if the sensor output is sensitive to disturbances by the measurement environment.

In the first case, the validation of the facility essentially consists of providing evidence that the geometric constellation, the velocities and time values produced by the facility are correct. This is the case, e.g., of a stimulator for a satellite navigation receiver, where the accuracy of the input signals for the test item, i.e. the receiver, have to be validated w.r.t. the real navigation satellite constellation. It is also the case for a stimulation facility for optical sensors, where it has to be proven that the geometric positions, angles and rates of the sensor and target pattern indicated by the facility represent, with the accuracy necessary for the verification test, the values produced by the facility.

In the second case, the validation consists of proof that the modelled disturbances are realistic. To verify the sensors it is important that measurement data are not affected under worst case disturbance conditions. Validation is, therefore, the proof that the facility provides such worst case conditions. For example, this is the case of a stimulator of light disturbance of optical sensors. It must be proven that the illumination source is equivalent to sunlight, both for the illumination of the target and for the simulation of the Sun in the FOV of the sensor. Worst case situations of reflection disturbances will be specular reflections of sunlight and of the sensor's own illuminator. It must be proven that the stimulation facility creates such conditions. In the case of a stimulator for satellite navigation, it must be proven that disturbances due to multi-path effects and shadowing are worst case but realistic. The problem of validation has been addressed already above at the discussion of the measurement environment model validation.

To validate the physical motion stimulation output of docking facilities, in the first instance simple contact geometries (e.g. ball against plate) and simple models for the spacecraft bodies (e.g. spheres and cubes) can be used for which the resulting motion after contact can easily be verified by analysis. Such initial validation exercises will verify the proper functioning of the complete setup from force vector reconstitution and transformations via the spacecraft dynamics into actuator motion. The validation of the models has been addressed already in the previous section. The validation of the complete facility, including all models, can best be performed by comparison with results of an already validated simulation. The compatibility of results of two independently developed simulations, e.g. in this case the test result of the docking hardware on the facility and the results of a simulation based entirely on mathematical modelling, will also increase the confidence in both tools, even if neither of them has been validated before.

10.6 Major simulators and facilities for RVD

10.6.1 Verification facilities based on mathematical modelling

The term *verification facility* is used here for a simulator which provides the environment for verification of hardware and software items in closed loop with a simulated environment. In section 10.3.3 we have seen that such facilities are required for the verification of the RV-control software resident in the onboard computer (figure 10.4) and for the verification of trajectory sensor equipment together with the navigation function of the GNC system (figures 10.6 and 10.7). In these figures, only a high level representation of the GNC functions, sensors and models of the other features that play a role in the performance of the test item under 'real world' conditions are shown. In addition, a verification facility will have to provide a number of functions for the interface with the test item, for running the simulation environment and integration of the dynamics equations, for proper engagement of models, for data inputs and outputs, for pre-, post-processing and storage of data, as required for the tests. The major functions of such a test facility are summarised in figure 10.12.

In order to be able to communicate with the test item, the facility will have to provide all those data interfaces which the test item has in its nominal environment, e.g. data bus and hardwired signal line interfaces. In addition it may have to establish for the test item particular test data interfaces, e.g. if there is a need for measurement of values which are not included in the data stream through the data bus and direct lines, or for input of particular data, e.g. for the creation of a failure condition in a particular function of the test item.

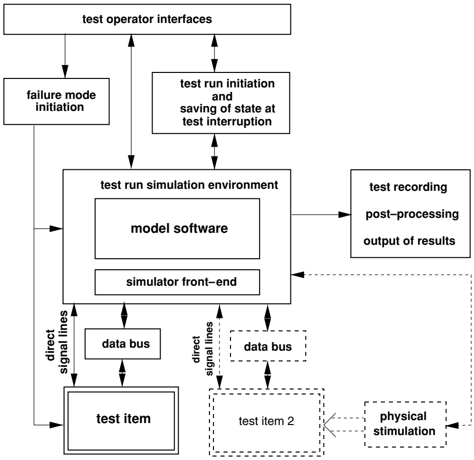

Figure 10.12. Functions of a simulation facility.

The primary function of the facility is to provide the simulation run environment, i.e. the real time execution of the integration of the differential equations describing the dynamic processes to be simulated, the engagement and calculation of the models and the transmission of the input and output data to and from the test item. Input and output data flow must be synchronised with the actual processing sequence and rate of the test item. Other features required for a verification facility are the introduction of failure conditions in the test run and the capability to stop and re-start the simulation run. Finally, a test facility must be capable of recording, post-processing and providing output of the test history and test results.

The introduction of failure conditions in the test item or in one of the functions modelled in the environment simulation can either be at pre-planned points during the simulation run or at the discretion of the test operator. In testing the FDIR functions, such failure simulation may concern both the consequences of failure conditions in the test item, and of equipment modelled in the environment simulation.

At any time during the simulation run, the test operator must be able to stop the simulation, e.g. for off-line analysis, and to resume it at the same state, i.e. with the same conditions and values of the models. The facility must provide all the necessary functions for the initiation of the test at a certain point of the mission, i.e. with the state of the models of the environment simulation according to that mission point. At interruption of the test, it must be possible to save this state and use the data as the initial values for the subsequent test run.

A second or third test item may be connected to the facility, e.g. a sensor or the GPS receivers of chaser and target, which may be physically stimulated by external stimulators, as already indicated in figures 10.4 and 10.7. In that case, the stimulator will be synchronised with the simulation environment and will be driven concerning the motion of chaser and target by models contained in the model software. The physical stimulation can be in the form of electric signals, e.g. the antenna signal fed to a satellite navigation receiver, or in the form of light and motion, as for optical sensors. For the latter case, an example of a stimulator is described in more detail in the following section.

10.6.2 Example of a stimulation facility for optical sensors

A stimulation facility for optical rendezvous sensors must provide variable relative positions, relative attitudes, relative velocities and angular rates in all directions between the sensor head and the target pattern. It must further provide an emulation of the Sun in the FOV of the sensor and of sunlight illuminating reflective surfaces around the target pattern. The first features are required to test sensor performance within the limits of their operating ranges, the latter ones are required to test their sensitivity against disturbances of the measurement environment.

Although it would be desirable to have a test setup which covers the complete operational range of an optical sensor (typically a few 100 m in the LOS direction), the combination of all test functions with such an extreme range would be extremely difficult to implement and not worth the effort. Also, with increasing distance between the sensor head and the reflector pattern, slight differences of density due to temperature differences cause the air to move and will introduce increasing disturbances. The most critical range for optical rendezvous sensors is that of the last few tens of metres prior to docking. In this range, not only the highest performance requirements exist for the range and LOS angle measurements, but also the three relative attitude angles have to be measured. A stimulation facility for optical rendezvous sensors should, therefore, cover a significant part of this range. If the effects of spurious specular reflections of sensor illuminator or sunlight are to be tested, the target reflector pattern needs to have the capability of three DOF angular motion independently of the sensor head.

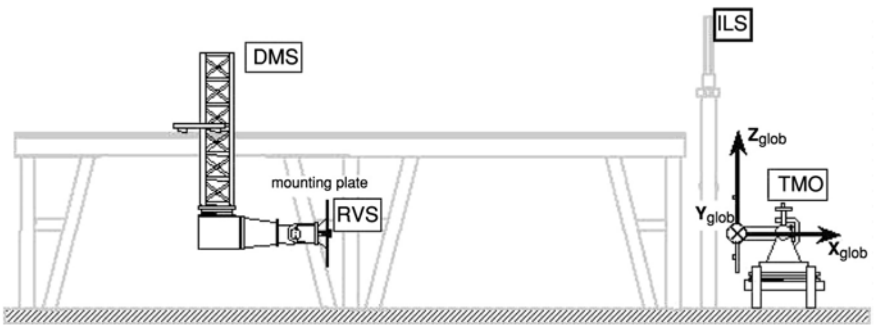

Figure 10.13. Measurement environment facility for optical sensors, EPOS (courtesy DLR).

For the operation as a GNC test facility in a closed loop with the rendezvous control software, as shown in figure 10.6, the motion system needs to be able to follow smoothly, in a slave mode, the state vector values produced by integration of the spacecraft dynamics equations in the environment simulation facility (see also figure 10.12). This requires that the bandwidth of the facility is at least as large as that of the closed loop system of the GNC and the spacecraft, and that the motion system of the facility does not produce additional oscillations to those contained in the spacecraft motions. The bandwidth requirement is another constraint for the size of the facility: the larger its dimensions, the lower the eigenfrequency.

An example of such a stimulation facility is the European Proximity Operations Simulator (EPOS) (Heimbold, Prins & Fehse 1987; Heimbold & Zunker 1996), which has been jointly developed by the European Space Agency and by DLR, the German Aerospace Centre. The facility is located at the DLR Flight Operations Centre in Oberpfaffenhofen, Germany. EPOS (figures 10.13 and 10.14) consists of a motion system (indicated as DMS) providing six DOF of motion capability, a target mount (indicated as TMO) with additional three DOF of angular motion and an illumination system (indicated as ILS) having a two DOF capability in translation and two DOF in rotation. The test item (indicated as RVS) is mounted on the front end of the DMS.

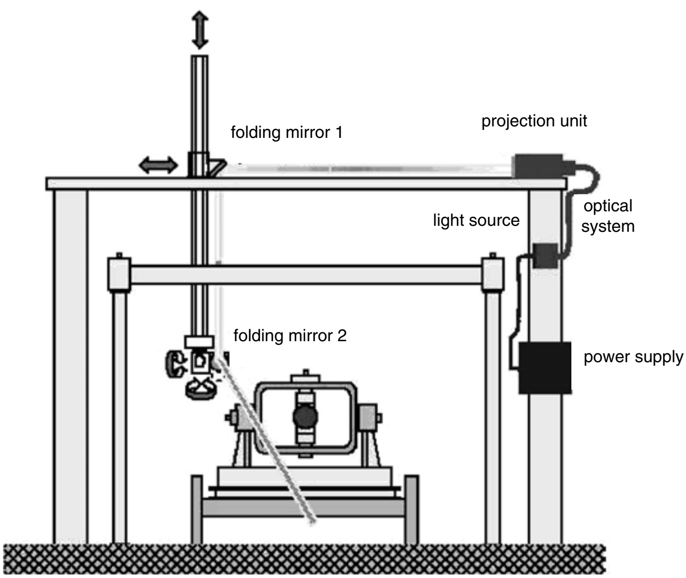

Figure 10.14. Illumination system of EPOS (courtesy DLR).

The *motion system* is a gantry robot, which has a working space of 12 m $\times$ 3 m $\times$ 2 m and provides maximum translation velocities of 0.5 m/s in all directions. The first carriage (gantry) moves in the $\pm x$-direction on two rails mounted at a distance of >3 m on a base structure firmly connected to the foundation. The second carriage moves on the gantry in the $\pm y$-direction, and the third carriage, a vertical beam structure connected to the second one, moves in the $\pm z$-direction. Connected to the lower end of the vertically moving structure is the front end, which carries the test item. The front end has rotational freedom (without test item) of $360 \times 180 \times 360$ deg (yaw, pitch, roll) and provides maximum rotation rates of 6 deg/s. The bandwidth of the motion system is of the order of 2 Hz.

The *target mount* provides an operating range of $360 \times 180 \times 360$ deg (yaw, pitch, roll), which will be reduced, however, by the accommodation of the test item, i.e. the target pattern and possibly a model of the surrounding surface of the target spacecraft. The target mount can be re-located to increase the distance between the reflector pattern and the sensor head up to 25 m.

The *illumination system* (figure 10.14) is on another gantry type robot, which provides lateral translation capability in the y- and z-directions, and a pitch and yaw rotation capability by its gimballed front end. The Sun simulator consists of a projector which produces parallel light with the intensity of 1 solar constant and two folding mirrors, the first mounted on the moving part of the gantry and the second on the outer (pitch) gimbal of the front end. The illumination system produces a light beam of 12 cm diameter, which can point either to the sensor optics or to the target mount.

To test the sensors, the three systems can be operated by pre-defined profiles for position, angles, velocities and angular rates. All three robot systems can be steered in position and direction by the output of a real time GNC simulation facility connected to the EPOS facility, where the measurements of the sensor mounted on the DMS provide the input to the simulation.

10.6.3 Dynamic stimulation facilities for docking

Many attempts have been made to build a test facility for docking dynamics entirely by mechanical means. Each of the following requirements alone is already difficult to implement:

(1) six DOF motion capability;

(2) two spacecraft models with the correct mass, inertia and CoM position;

(3) compensation for the effects of gravity;

(4) the correct contact velocities;

(5) realistic translational and rotational misalignments;

The combination of all these requirements is practically impossible to achieve without reduction of the degrees of freedom and without arriving at extremely constraining compromises concerning the choice of test conditions. These constraints will apply to the range of contact conditions, spacecraft masses and inertias which can be tested and to the freedom of motion of the emulated spacecraft bodies motion after contact. The design of a fully mechanical facility with five DOF has been described in Syromiatnikov (1990). This test setup has been used for the verification of the Apollo–Soyuz docking mechanism and for other Russian (Soviet) docking mechanisms. In figure 10.15 it can be seen how the mass and inertia of the two spacecraft are emulated (item 2 for target, item 4 for chaser), how the gravity effect is compensated for by hanging the two masses on cables, suspended in their CoM, how the pendulum effects are compensated for by spring compensators (items 1 and 6), how the chaser body is replaced w.r.t. the target body (item 7), etc.

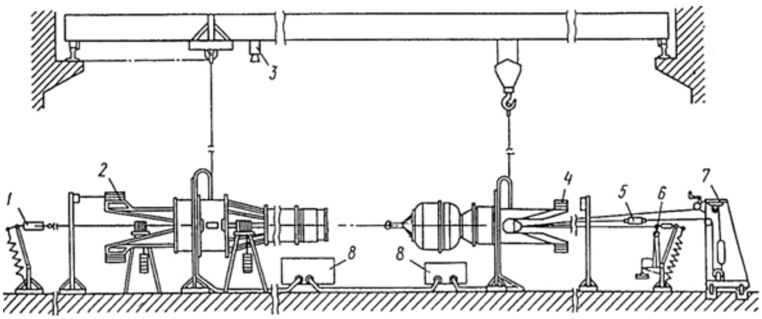

Figure 10.15. Mechanical docking test facility (Syromiatnikov 1990).

It is obvious that with increasing mass and inertia of the vehicles involved in the docking process, such an entirely mechanical facility will be even more difficult to realise. The main disadvantage of such a facility is that, due to the reduction of degrees of freedom, the actual dynamic reactions of the two model bodies will not properly represent the real dynamic reactions of the chaser and target spacecraft. Such setups may provide some indication of the contact point and magnitude of forces in the attenuator system. They may also be useful for testing the proper functioning of the attenuation system or of the latches, but they cannot prove that capture will be successful.

In order to be able to test docking systems in a full six DOF environment, computer controlled electro-mechanical facilities have been developed according to the concepts described in section 10.3.3 and shown in figure 10.8. In practically all the developments of such facilities performed so far, the principle of a Stewart platform has been used. (A Stewart platform is the configuration which provides a six DOF motion capability at maximum stiffness and a minimum number of actuators and moving parts.) The basic design principle of a dynamic docking test facility is shown in figure 10.16. Such facilities have been built for the American (Tobbe & Naumann 1992) and Russian (Syromiatnikov 1990) space programmes, and later also in Europe (Brondino *et al.* 1990) and Japan (Inoue 1991).

From the front end kinematics model, the required position of the platform will be calculated and transformed into the necessary extension of the linear actuators. The requirements for the mathematical models which have to be engaged for the computation

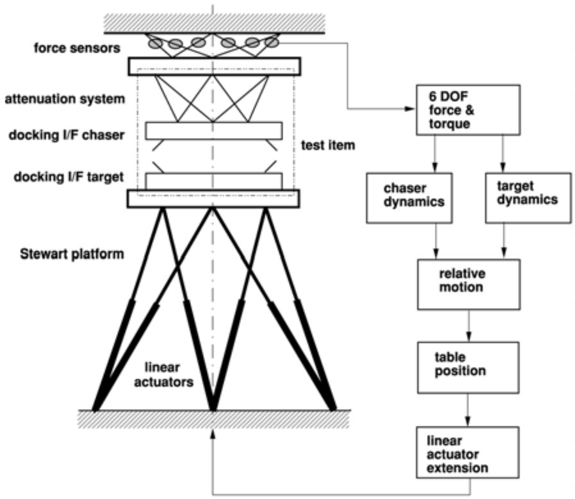

Figure 10.16. Principle of a docking dynamics test facility.

of the motion of the platform have been discussed already in section 10.4.2. Because of the large mass of manned docking mechanisms and the high loads in the case of impact docking, in most cases hydraulic actuators are used. Linear actuators driven by electric motors have been used in smaller test facilities originally designed for unpressurised docking test facilities (Brondino *et al*. 1990; Inoue 1991). Attempts to verify contact dynamics of docking mechanisms for manned scenarios on such small facilities have led to the development of scaled models and to the derivation of scaling laws (MATRA 1993). It has to be kept in mind, however, that although such tests on scaled down hardware is a useful approach for the investigation and verification of the capabilities of the design, and also for validation of models, they will not be sufficient for the final qualification of the flight item.

For better load carrying capability and compensation of the undesirable 1 g loads, the main motion axis (approach axis) of the facility should be vertical. In some implementations (Grimbert & Marshal, 1987; Brondino *et al*. 1990; Inoue 1991; RDOTS 1997) this has been compromised in favour of an additional capability for testing rendezvous sensors. Such a combination of objectives is, however, not advisable:

- A docking test facility requires a high stiffness to achieve a high bandwidth. This is necessary to be able to perform also the motions resulting from structural oscillations of the docking mechanism and the spacecraft substructure at impact. The range required in the approach direction is not much more than what is necessary to accelerate the platform plus the test item to the contact velocity.

- A sensor test facility must have relative large ranges in all directions, but only moderate stiffness requirements (see the previous section).

Large range and high stiffness are incompatible technical requirements. Whereas with very small and lightweight docking mechanisms, such as the ones for unpressurised mating, a combination of the types of facilities might just be possible, for large manned docking systems, such a combination of facility objectives would not lead to useful results for either type of test.

10.7 Demonstration of RVD/B technology in orbit

10.7.1 Purpose and limitations of in-orbit demonstrations

The term *demonstration* is used here for the operation of an item in front of witnesses, with the aim of providing evidence of proper function and performance. A demonstration is neither verification, as there will be no proof that a test item fulfils *all* specifications, nor is it a complete validation, as there will be no proof that a test item functions and performs as required under *all* real world conditions. Generally, a demonstration can show in the best case that for the single set of demonstration conditions the requirements are fulfilled. A successful demonstration in orbit, however, can significantly add to the level of confidence in the proper performance of an item.

Demonstration in orbit will provide in the first instance experience concerning the behaviour of an item under orbit dynamic and zero-g conditions. Further, depending on the demonstration item and the objectives, it may also provide experience concerning other environmental conditions, e.g. survival under launch load conditions, performance under real measurement environment conditions, etc. However, if the demonstration cannot be performed exactly under the same conditions as the real mission, its 'real world conditions' may be similar, but not equal, which reduces the value as a means of verification.

A particular problem of using the results of a demonstration in orbit with two spacecraft for verification/validation purposes, is the fact that this would require an independent measurement capability which is better than or at least equal to the performance of the item or process that is to be verified. For instance, the performance of a GNC system is mainly determined by the accuracy of its sensors. The GNC sensors have been selected, however, because they are the best available for this purpose. For this reason, it will generally not be possible to verify the performance of a GNC system and of its sensors in orbit by comparing it with independent measurements of higher accuracy. At best, a sensor of different or independent design with equal accuracy can be used.

The objective of a demonstration in orbit will, therefore, in most cases not be a verification or validation by direct measurement. The success of a demonstration will usually have to be judged either by the final state achieved in an operation, e.g. the end point of a manoeuvre, the capture of the docking interfaces, etc., or by indirect criteria which can more easily be performed either in orbit or from the ground. Orbital parameters and a position in an orbit at a particular time can, e.g., be reconstituted with relatively good accuracy after flight.

Flight demonstrations are extremely costly when this requires a dedicated mission including spacecraft, launch and mission operations. For this reason, in most cases, flight opportunities will be sought where the flight experiment is an add-on to an existing mission. Such opportunities are rare, however, in particular if a demonstration opportunity in the same type of orbit as the target mission is sought. For this reason, considering both the limited flight opportunities and the general limitations of achieving a demonstration in orbit, in most cases any flight opportunity will be accepted for a demonstration provided the orbit conditions are roughly similar (e.g. LEO for a LEO target mission).

10.7.2 Demonstration of critical features and equipment

In-orbit testing will be sought in particular for items or features where doubts exist that all potential effects or disturbances could be identified and covered to a sufficient extent by analysis, simulation or physical testing on ground. Such doubts typically exist for

- new technologies, for which no space experience yet exists;
- complex measurement environments, where disturbances may be caused by the complex structure of one or both spacecraft, by the atmosphere or by interference from ground, which cannot easily be modelled;
- complex dynamic disturbances and interactions.

A typical case in which it would be desirable to test a new technology in orbit is the demonstration of a rendezvous sensor. Since the performance of the sensors is crucial for the success of a rendezvous mission, it must be ensured that there will be, once in orbit, no effects which may compromise function and performance of the sensor during the mission. The objectives of a flight demonstration will be, therefore, to

- uncover potential side effects, due to orbital conditions, which may have slipped through analysis and testing;
- achieve a better understanding of the operational environment in which the equipment will have to function;
- obtain better information on the disturbance part of the measurement environment.

The possibile validation by an in-orbit demonstration of particular features has already been addressed, e.g. for drag and propellant sloshing, in section 10.5.1. The objective will be, in this case, confirmation or improvement of an existing model. The concept for the demonstration/test setup could be, e.g.,

- the measurement of dynamic reactions of the host spacecraft itself (for example, excitation of slosh by applying linear and angular accelerations, measurement of excitation and of reactions by accelerometers and by gyros (Vreeburg 1999), or

- the measurement of differences in trajectory and attitude between the host spacecraft and a sub-satellite (for example, measurement by optical rendezvous sensors or relative GPS of the trajectory of a sub-satellite with different ballistic coefficients to determine differential drag).

Relative GPS (see section 7.3.3) is an example of a complex sensor function which involves a measurement environment formed by a set of navigation satellites, sensor equipment, i.e. satellite navigation receivers on chaser and target spacecraft, a communication link between the vehicles and a navigation filter, receiving inputs from other functions on the chaser vehicle. Because of the complexity of this measurement principle and its dependency on many conditions and features, which are present only under the condition of two vehicles being in relative close vicinity in orbit, a demonstration prior to operational use is extremely desirable. Unfortunately, a flight opportunity providing all of the features of

- two spacecraft in close vicinity,
- GPS receivers on both chaser and target spacecraft,
- communication between spacecraft,
- an RGPS navigation filter with the real time inputs of actuation commands and attitude measurements of one of the vehicles,

will be rare, unless a dedicated demonstration mission can be implemented (see the ETS-VII demonstration mission in section 10.7.3). Obviously, any rendezvous mission to a space station and any deployment and recovery of spacecraft by the US Space Shuttle will provide some of these features. However, since in the Mir and ISS scenarios no spacecraft has used, up to the time of writing, RGPS for rendezvous and docking navigation, and since most deployment and retrieval missions did not include GPS receivers, all demonstrations of RGPS have so far required a particular experiment setup. A number of orbital experiments have already been performed at the time of writing, and the according experiment plans and results have been reported in, e.g., Hinkel, Park & Fehse (1995), Park *et al*. (1996), Ortega *et al.* (1998), Cislaghi *et al*. (1999) and Mokuno, Kawano & Kasai (1999). However, to achieve sufficient confidence for operational use, additional experience with RGPS will have to be gained in the proper environment of the envisaged spacecraft systems and mission operations.

RGPS flight demonstration example

As an example of the complexity of measurements and data reconstitution, an in-orbit demonstration of RGPS will be described which was performed by the European Space Agency in 1997 on the STS-84 and STS-86 rendezvous missions of the US Space Shuttle to the Russian Mir Space Station (Ortega *et al.* 1998; Cislaghi *et al.* 1999). The demonstration campaign included also the demonstration of the optical rendezvous sensor RVS. However, for the purpose of this example, the description will concentrate on the RGPS part of the demonstration only. The objectives of the demonstration were threefold:

(1) a validation of the receiver equipment under actual space environment conditions;

(2) a validation of the RGPS navigation filter with real flight data inputs;

(3) a validation of the mathematical model of the GPS receiver.

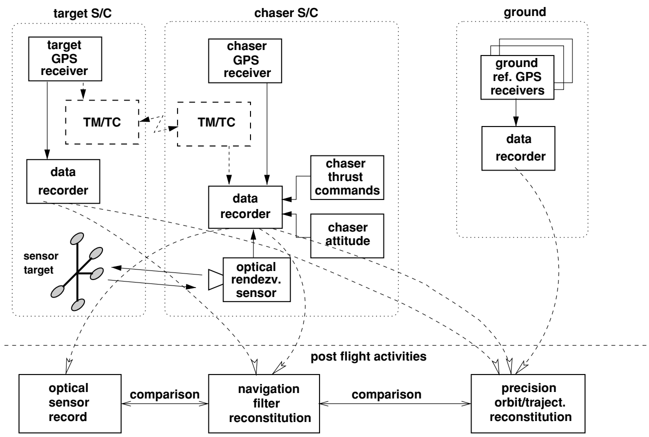

Figure 10.17. Experimental setup for flight demonstration of RGPS.

The RGPS demonstration setup consisted of the following elements:

- GPS receivers on both chaser and target vehicles. These receivers were of different design:
  - the target had a Motorola Viceroy receiver, which was part of the German navigation package MOMSNAV, mounted on the Priroda Module of the Mir Space Station;
  - the chaser had a Laben GPS receiver (based on the Loral TENSOR design), which was mounted as an ESA experiment near to the docking port on the orbiter. The antennas of both vehicles were nominally zenith pointing during the last part of the R-bar final approach of the Orbiter.
- An optical rendezvous sensor on the chaser to provide reference data for the short range. There were two sensors involved, the ESA rendezvous sensor (RVS) and the NASA trajectory control sensor (TCS). Both sensors were of the laser range finder type and mounted near to the docking port of the Orbiter. Corresponding target

reflectors were mounted on the docking module for the Orbiter, which was attached to the Kristal Module of the Mir station.

- A data recorder on the chaser vehicle (Orbiter):
  - to record the raw data of the chaser GPS receiver;
  - to record the accelerations and attitude values of the chaser;
  - to record the optical rendezvous sensor output.
- A data recording device in a laptop computer on the target vehicle (Mir), to record the raw data of the target GPS receiver. Other data pertaining to Mir could not be recorded. The actual time of the attitude changes of the station had to be obtained from the flight plan post-flight.
- A radio link for the transmission of the GPS raw data from the target (Mir) to the chaser (Orbiter) was originally planned, but could not be implemented.

To obtain the maximum opportunities for measurements, the GPS experiments were planned to take place during both approach to and departure from Mir by the Orbiter. The first part of the departure trajectory was similar to the approach trajectory (shown in figure 10.18), only in the opposite direction, with Mir in a LVLH attitude. The second part of the departure trajectory was a fly-around, while the Mir Station was in a Sun-pointing inertial attitude, with the Orbiter remaining opposite to the docking module on Mir at a constant distance.

During the mission no navigation filter processing was to be performed. Instead, the experiment plan was to reconstitute the data processing and output of the navigation filter from the recorded GPS raw data of the chaser and target and from the recorded attitude and thrust acceleration data of the inertial measurement unit of the Orbiter. GPS time was to be used for synchronisation of data. The absolute trajectories of chaser and target were planned to be reconstituted by differential GPS techniques (DGPS, see section 7.3.3) using the IGS (International Geo-dynamics GPS Service) network of GPS receivers at various locations on the Earth as references. These absolute trajectories were intended to be used for the validation of the RGPS performance and for the validation of the mathematical modelling of the GPS receiver.

According to this concept, the post-flight data processing included the following steps:

- Reconstitution of the navigation filter data flow from the recorded GPS raw data of chaser and target and chaser attitude and thrust data.
- Reconstitution of 'best estimated' *absolute* trajectories of chaser and target by DGPS processing of the outputs of the chaser and target GPS receivers with the records of the reference GPS receivers on ground.

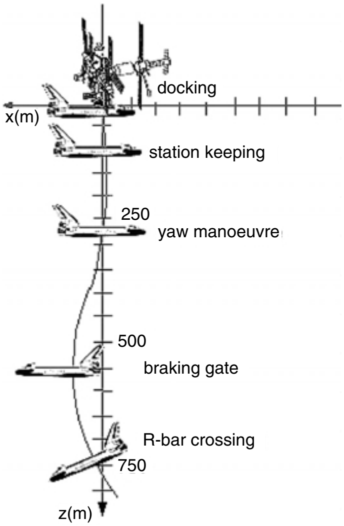

Figure 10.18. Final approach trajectory of Orbiter to Mir (courtesy ESA).

- Comparison of the relative trajectory from the output of the navigation filter with the 'best estimated' *relative* trajectory, obtained from the differences between the absolute trajectories of chaser and target obtained from the DGPS results using the IGS receivers.
- Comparison of the relative trajectory from the output of the reconstituted navigation filter with relative trajectory data from the optical sensor measurements. Optical sensor data were, however, not available for all ranges and all trajectory parts.
- Comparison of the actual absolute trajectory output of the test GPS receiver with the output of the mathematical model of the receiver, fed by the 'best estimated' absolute trajectory obtained from the DGPS results using the IGS receivers and by the GPS satellite constellation at the time of demonstration.

The results of the RGPS experiments during the two flights have been analysed in MATRA-MARCONI (1998 a,b). During only a few short parts of the approach and departure trajectories, all measurements required for the evaluation were simultaneously available. For example, during the entire first flight, no optical sensor measurements and useful GPS measurements of both receivers could be obtained at the same time. During the second flight a complete set of data could be recorded only during the departure phase. The reasons were: a partial non-availability of one receiver and a less suitable attitude or shadowing by one or both vehicles, such that the minimum of four commonly tracked GPS satellites was not achieved; or that the target reflectors were outside the FOV of the optical sensor.

The accuracy of the best estimated relative trajectories obtained from the DGPS processing with the IGS receivers was about 15–25 m in position and about 3–5 cm/s in velocity, which is about twice the required performance of RGPS ($<$10 m). Although this was not sufficient for a validation of RGPS performance, the DGPS data helped in the general assessment of results and in the detection and interpretation of disturbances. The accuracy of the best estimated absolute trajectory was, however, sufficient for the validation of the GPS receiver mathematical model. The accuracy of the relative trajectory obtained from the optical sensor measurements (TCS) of $<$1 m eventually provided the proper reference for RGPS performance validation.

The experience with this flight experiment revealed:

- the difficulties in implementing a suitable RGPS experiment setup on a mission which was not planned for this purpose;
- the problems of obtaining simultaneously suitable measurements from all necessary contributors to the demonstration experiment, during flight operations which were not designed for such a demonstration;
- the complexity of the evaluation process from indirect data concerning the performance of a sensor system as complex as relative GPS.

Shadowing and multi-path disturbances encountered during the flight experiment were found to be difficult to re-establish by simulation. This was mainly due to the high geometric complexity of the Mir Station and the Orbiter, which were difficult to represent in the multi-path model. Multi-path effects became observable at ranges below 200 m. The errors were generally below 10 m; however, short peak values of $>$100 m have also been encountered. The RGPS navigation filter was found not to be very sensitive to temporary multi-path effects. In conclusion, the flight demonstrations were successful and the results were very encouraging. They confirmed also, however, that more flight experience in an operational environment as close as possible to that of the target mission would be desirable to reduce risks of disturbances by shadowing and multi-path effects.

10.7.3 Demonstration of RV-system and operations in orbit

The objectives of an in-orbit demonstration on a systems and operations level can be:

(1) to gain general experience about a new orbital technique;

(2) to prove the readiness of systems and operations design prior to an operational rendezvous mission.

In the first case the objectives of the demonstration will be to demonstrate the general ability of performing in-orbit rendezvous and mating of two spacecraft, to build confidence in the design concept for an automated onboard rendezvous control system, and to gain experience about all issues of planning and execution of orbital operations for such missions, including the communication and command capabilities with/from ground. One of the major interests will be to identify side effects, omissions (‘not thought ofs’) and any design weaknesses of any part of the space and ground systems which may have slipped through the verification exercise on ground.

In the second case, the demonstration is a final proof of operational readiness under ‘real world’ conditions, i.e. the final step in the validation of a complete rendezvous system, consisting of the space and ground segment functions and operations, including all infrastructure and auxiliary functions. Whereas in the first case the emphasis will be on the technical success of a new concept, in the second case the safety of the target crew (in the case of a manned mission, otherwise it will be the security of investment) during all rendezvous and capture operations is the most important feature to be demonstrated.

Example of a rendezvous and docking technology demonstration on system level

The Japanese Engineering Test Satellite ETS-VII (Kawano *et al*. 1998; Mokuno *et al.* 1999; Tsukui *et al.* 1999) is the best recent example of an RVD system technology demonstration in-orbit. The satellite system, launched in 1997, consisted of the main spacecraft, which acted during the rendezvous demonstration as the chaser, and a sub-satellite, which had only attitude control capability and acted as the target (see figure 10.19). With ETS-VII, two technologies and techniques were demonstrated: rendezvous and docking and space robotics. For the purpose of this book only the rendezvous and docking part is of interest. The three major objectives of the rendezvous demonstration were:

(1) Validation of the RVD specific equipment technology, i.e. demonstration of proper functioning and performance of RGPS, of the optical rendezvous sensors and of the docking mechanism.

(2) Validation of rendezvous control technology, i.e. demonstration of proper functioning and performance of the GNC and flight management functions (the latter contains the functions called MVM and FDIR in this book).

(3) Validation of the RVD operations techniques, i.e. demonstration of supervisory control techniques of monitoring and high level controlling chaser and target during automatic rendezvous operations, demonstration of proper functioning of telecommunication techniques via relay satellite (TDRS), and demonstration of teleoperations techniques, performing trajectory control from ground.

The chaser vehicle, i.e. the main spacecraft of 2500 kg mass, had the following RVD specific equipment:

- a double-redundant onboard control computer with voting function;

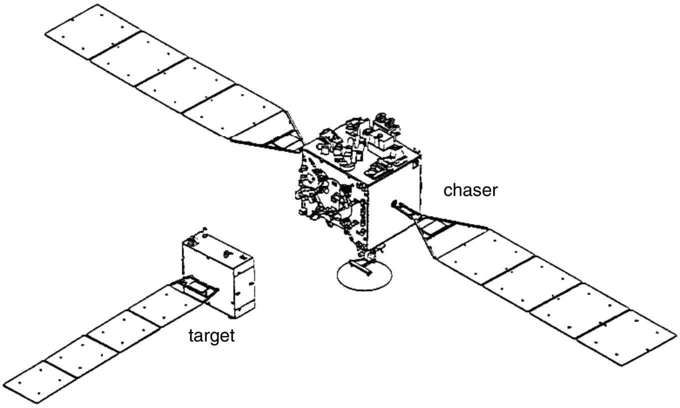

Figure 10.19. ETS-VII spacecraft for rendezvous demonstration (Kawano *et al*. 1998).

- the rendezvous control software resident in the control computer, which includes the algorithms for the GNC modes and for the flight management functions (mode sequencing, FDIR, approach abort and CAM implementation);
- a GPS receiver for absolute position measurement in all ranges and for relative position measurement between 9 km and 500 m;
- a laser radar type rendezvous sensor for relative measurement between 500 m and 2 m;
- an optical proximity sensor for relative position and relative attitude measurement between 2 m and contact;
- an unpressurised docking mechanism consisting of three latches of the type shown in figure 8.6.

Along with the other equipment necessary for the operation of a satellite, such as the reaction control system, the communication system for data exchange with the target satellite and ground (via TDRS) and attitude control sensors, such as gyros and Earth sensors, video cameras and a floodlight installation were available to observe and record the approach and docking operations.

Because of its more passive function in the rendezvous demonstration, the target vehicle had a smaller size and a mass of only 400 kg. Its GNC functions were reduced mainly to attitude control. It carried the following RVD specific equipment:

- a GPS receiver,
- the target reflectors for the laser radar rendezvous sensor,
- the target pattern for the proximity sensor,

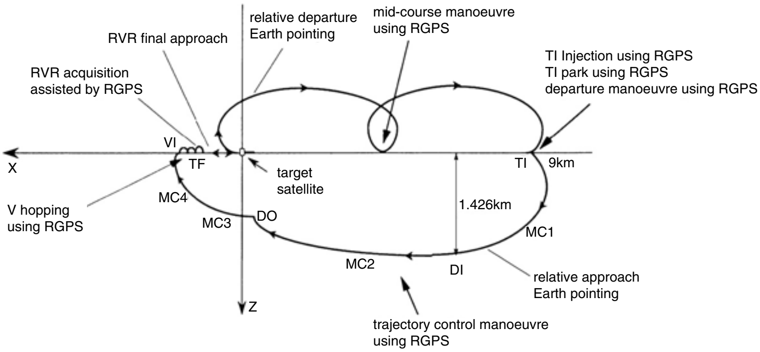

Figure 10.20. Planned trajectories of third ETS-VII RVD flight campaign (Kawano 1997).

- the handlebars for the docking latches,
- an optical target for the video camera.

Attitude control electronics, gyros and Earth sensors, a reaction control system and equipment for an inter-satellite link with the chaser vehicle enabled this sub-satellite to function as an independent spacecraft.

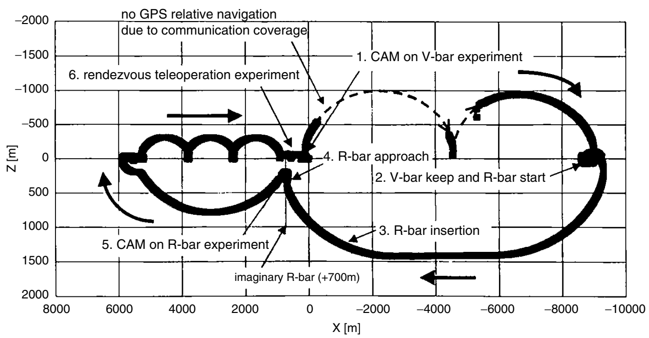

Figure 10.21. Actual trajectories of third ETS-VII RVD flight campaign (Yamanaka 2000).

The RVD demonstration was planned to be performed in several flight campaigns with trajectory strategies designed to demonstrate particular features, such as V-bar separation and docking, V-bar final approach, complete V-bar approach sequence including all ranges of relative navigation and docking, contingency operations, R-bar approach, and remotely manually controlled approach. In spite of thruster failures, all demonstration objectives could be achieved, i.e. all equipment and system functions addressed above have been demonstrated successfully. As an example, the planned trajectories for the third flight campaign are shown in figure 10.20 (Kawano 1997). The actual strategy of the third campaign had been modified, however, to include features which were originally planned for an additional flight campaign. The flight trajectories recorded from the RGPS navigation data are shown in figure 10.21 (Yamanaka 2000).

In-flight demonstration of system readiness prior to operational use

As indicated above, the objective of such a demonstration would be to test all the systems and operations in their proper context prior to a real mission, i.e. prior to the first mating with the target spacecraft. The problem of validation of a rendezvous system is similar to the problem of validation of a launcher: the necessary confidence in proper functioning and performance can be obtained only by demonstration of systems and operations with the real vehicle. Demonstrations with other vehicles will always lead to different body and contact dynamics, which will require an adaptation of GNC and docking systems to those dynamics. Also, the infrastructure of the reaction control system, the data management system, the communications system and other subsystems will be different on another vehicles, so that, in the end, such a demonstration would be no more than the technology type of demonstration described above. The dilemma is that, when using different vehicles, the objective of proving flight readiness of the operational system cannot be achieved; on the other hand, with an unproven system, proximity operations and mating with the target form a risk, in particular if the target vehicle is manned.

Let us consider in more detail what the problems and limitations of validity are if the same or different vehicles are used:

- The target can be the real one down to a safe distance, whereas at close proximity and contact the target for the demonstration should be different for safety reasons. However, if the target is different:
  - the target attitude motion will be different, which is a drawback for the verification of the RV-control system of the chaser for the last part of final approach;
  - target mass and inertia will be different, which is a drawback for contact/capture dynamics verification.

  The result would be, in both cases, that, even if the demonstration were successful, there would be no guarantee for success of close proximity operations and docking with the real target.

- If the chaser vehicle is different, practically the entire onboard GNC system has to be adapted to the vehicle properties of the demo vehicle – in particular the navigation filter and the control function. The demonstration would then be in essence a demonstration of the approach strategy and operations only. Except in very particular cases, where such a mission, including launch, could be implemented very cheaply and where some particular systems or operational features are to be proven, a demonstration with a different chaser vehicle for the proof of flight readiness of the complete RVD system will probably not be worth the investment.

- If the chaser vehicle is of the same design as that for the operational mission, all GNC system and operations demo-objectives can be realised, except for the last part of final approach and contact/capture dynamics verification. The last task would, in this case, require safety-critical operation of a yet undemonstrated system with the manned space station as the target.

It appears then that the dilemma that

(a) system functions and operations for the most critical part of final approach and capture cannot satisfactorily be proven in a demonstration with a different target of opportunity, and

(b) a first time proximity operation and contact with the real target vehicle should be avoided, since it is safety-critical,

cannot be solved without taking the risks of endangering the operational target either in the demonstration mission or in the first operational mission. For this reason, the following solution has been proposed, which tries to reduce the risk to be taken to the maximum possible extent:

(1) Perform the demonstration of the RVD system and the operational approach strategy with the *real chaser* vehicle to the *real target* vehicle up to a point where the safety risk is acceptable. For vehicles visiting the International Space Station, this could be, e.g., for the approach strategies described in section 5.7, the point S3 in example 1 (figure 5.27) or the point S4 in example 2 (figure 5.28).

(2) Perform the rest of the approach in steps with additional stop points, where system and trajectory verifications can be performed and where the vehicle could be commanded to back off, if necessary.

(3) Demonstrate prior to the start of the final approach a CAM, and, during the final approach, a back-off manoeuvre, to prove commandability and proper functioning of such manoeuvres and to prove capability of ground to implement recovery operations and new flight plans.

(4) Perform the last metres of approach from a hold point, at which availability and functioning of all systems and equipment required for the docking operations have been checked.

In the previous demonstration steps, proper functioning of the onboard system and of all onboard and ground operations for the contingency case have been demonstrated.

At the last hold point, availability of sensor functions and GNC modes for the last few metres have been checked. If, at that point, all systems are functioning and no major failures have occurred before, which may have reduced redundancies, the residual risk for the last part of approach and capture should be considered bearable.