DESCRIPTION OF THE RENDEZVOUS EXPERIMENT DESIGNED FOR 2007 MARS PREMIER MISSION

M.Delpech[(1)], J-B.Dubois[(1)], J.E.Riedel [(2)], J.R.Guinn[(2)]
[1] Centre National d'Etudes Spatiales, France
[2] Jet Propulsion Laboratory, USA

Introduction

The Mars Premier mission that was to be flown in 2007 by CNES in cooperation with NASA/JPL included a rendezvous experiment to be performed in Mars orbit to validate key technologies applicable to a Sample Return mission. The experiment goal was to demonstrate the capability to detect and track a sample canister at long range (3000 km) and validate terminal rendezvous strategies including a "capture". Initially, it was envisioned to tackle the tremendous challenge of detecting, rendezvousing and capturing a small object in Mars Orbit using powerful sensors such as a Radio Direction Finder and a Lidar. Interestingly, it was later considered possible to address the challenge by reducing the sensor suite to a pair of cameras: a Narrow Angle Camera (NAC) for short to long range measurements and a Wide Angle Camera (WAC) used at very short range. For the technology experiment, a representative Target was carried by the Orbiter and deployed at the beginning of the mission. Each rendezvous phase would then be repeated several times over a six months period.

The paper focuses first on the experiment accommodation from the mission point of view and in particular the share of spacecraft resources between conflicting objectives. The next section covers the different areas of spacecraft design that are concerned by the implementation of the rendezvous experiment, namely the propulsion architecture, the AOCS (maneuver execution, attitude guidance) and rendezvous payload accommodation. It presents then the original operational concept relying on a strong implication by the ground segment that allows to minimize the complexity of the on-board algorithms while saving enough flexibility for the autonomous phases. Finally, the last section addresses the validation work that has been performed with numeric simulations to show the feasibility and particularly the robustness of this operational concept for the critical phase of Terminal Rendezvous and Capture.

Presentation of the CNES PREMIER 2007 Mission

The CNES involvement in Mars exploration is managed in the framework of the PREMIER program (Programme de Retour d'Echantillons Martiens et Installation d'Expériences en Réseau). PREMIER comprises the implementation of a complete mission to Mars, the support to the French PIs and scientists of ESA missions to Mars (starting with MARS-EXPRESS and the support to French instruments as parts of the payloads of NASA landers and rovers.

Initially, the PREMIER mission to Mars was planned as a contribution to the first Mars Sample Return mission to be launched in 2005. It consisted of an Orbiter, which was designed to capture and bring back to Earth a soil sample launched from the Mars surface by a NASA lander. Moreover, this mission included the deployment of a network of 4 Netlander probes on the Martian surface, and the first demonstration of aerocapture that represents a promising technique to insert spacecraft into Mars orbit by using a powerful and single breaking pass through the atmosphere.

After the re-architecture of the NASA Mars Exploration Program in 2000, followed in 2001 by the cancellation of aerocapture in the PREMIER mission due to its increased cost, CNES has defined a new PREMIER mission that kept the main requirements of the previous one and included a significant NASA contribution.

The high level requirements to PREMIER for the 2007 Orbiter Mission were the following :

- Launch with Ariane 5,
- Deployment of 4 Netlanders,
- Relay the data of the Netlanders during at least one year, and provide a time correlation between them,
- Demonstrate rendezvous and capture (RSC) technology to feed forward to the MSR mission,
- Perform an orbital science mission as a complement to the Netlander surface science
- Accommodate a UHF relay capability.

The following figure (Figure 1) shows the mission timeline. The launch window with Ariane 5 would extend from Sept. 1st to Sept.21st 2007 and the cruise duration would be about 11 months. After insertion by chemical propulsion in its nominal orbit (550 km altitude, sun-synchronous polar), the Orbiter would enter a one-month check-out phase. Next, the NetLander relay function would become operational and several rendezvous validation sessions would be performed. *It is obvious that the rendezvous and sample capture demonstration would be achieved without cutting off the Netlander relay service, and moreover should allow some operational margin for Orbital science.*

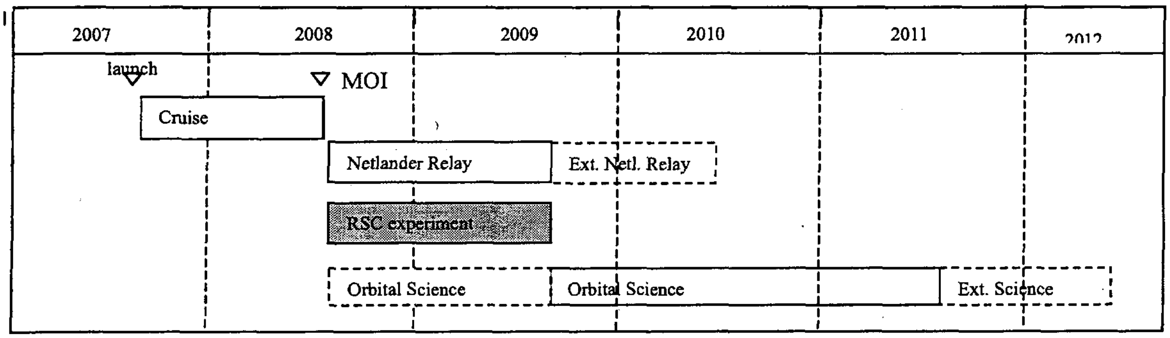

Figure 1: PREMIER mission timeline

This paper is focusing on the design of the rendezvous experiment but it is important to present the general mission context since the experiment is impacted by Netlander relay and possibly orbital science. This leads to a higher complexity from the mission design and operation point of view than what was initially planned for MSR rendezvous. In particular, the mission needs to be decomposed into sub-phases, where spacecraft resources are allocated sequentially to the different activities. The duration of these sub-phases might be comprised between a few minutes and several days.

Rendezvous Mission overview

The rendezvous experiment requires the presence of a Target in Mars Orbit that is representative of the Orbiting Sample (OS) canister to be used in a Sample Return Mission. For that purpose, a 20 cm sphere is carried by the Orbiter and released in Mars Orbit. The rendezvous experiment would comprise 5 different phases that will be repeated several times during the mission.

Search: The objective of this phase is to demonstrate the capability to detect the Target in a "MSR-like" Search. Necessary features of the Search to demonstrate include obtaining Target images at varying range and Sun phase angle and obtaining images of the Martian limb at varying Sun phase angles. The duration of a Search Phase is 4 weeks, and two may be performed contiguously, at an increasing and closing range. Search Phase operations will consume minimal spacecraft resources, allowing dominant use by other activities.

Intermediate Rendezvous:

The relative distance is decreased from 500 km to 5 km through a series of maneuvers computed on the ground and uplinked. In this phase which duration is about a week, navigation is still performed on the ground but relies on on-board picture processing. The number of observation windows (10‘ session) can be rather limited: from one to a few per day depending on the distance.

Terminal Parking: The Orbiter is maintained in a 2 km to 5 km range on a series of football orbits that guarantee a minimal risk of collision by the presence of radial and out of plane components. This phase allows to keep safely the Orbiter in the Target vicinity between repetitions of Terminal Rendezvous Phase. The Orbiter primary focus can shift to other mission activities since a quasi-permanent observation of the Target is not required. Its primary role is to prepare the departure to a Terminal Rendezvous phase and this includes occasionally the exercise of the autonomous behavior.

Terminal Rendezvous: Departing from the Terminal Parking, the Orbiter is driven autonomously to a 100m relative range. This approach that is performed by a series of transfers between progressively smaller orbits takes about 12 hours. Permanent observation of the Target is required during this phase and this forbids the use of the spacecraft for other purposes. Furthermore, during the final eclipses where the Target is the closest, some artificial illumination has to be provided to achieve the required navigation performances.

Capture: The Orbiter will not catch and secure the Target since it does not carry a capture mechanism. This phase will demonstrate anyway the ability to control the relative geometry up to the moment when the Target would enter the capture mechanism. The Orbiter is therefore driven to a distance of about 2 m with a final velocity of a few cm/s and the trajectory is followed with an accuracy consistent with a high rate of capture success. The duration of such a transfer is expected to be about 20 minutes.

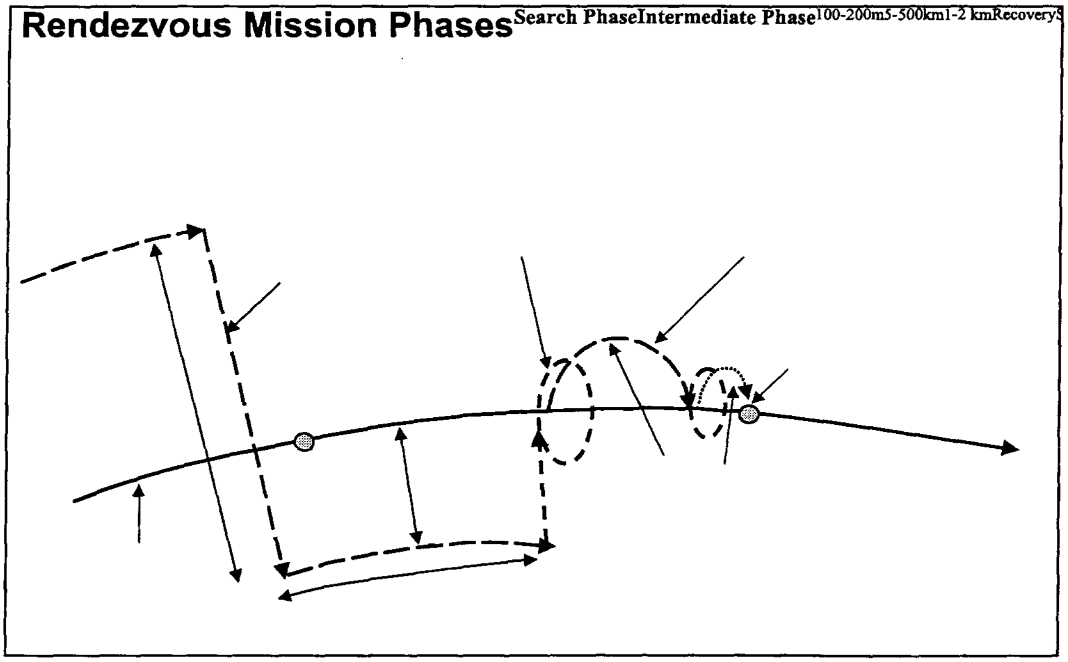

Figure 2 : Rendezvous phases

The rendezvous experiment would include also a specific phase (Target Release) that does not satisfy any demonstration objective but is critical for the mission success and safety. This phase may extend to 48-hours in order to satisfy two essential conditions : 1) achieve a safe range (e.g. 5-10km) between the Orbiter and the Target and 2) obtain sufficient optical data to allow accurate relative orbit determination by the ground. This implies dedicated spacecraft resources for 24 hours, followed by the opportunity for some shared use in the next 24 hours.

Overall mission constraints

- Orbit selection: In the first part of the mission, during which the rendezvous demonstration is to be achieved, mission design gives a high priority to Netlander relay. Therefore, the nominal orbit during this one-year period is selected to optimize the use of NetLander ressources and particularly power. Since communication is desirably performed when the Sun is closer to Netlander zenith, the selected orbit must have a mean local solar time around 12:00. However, a sufficient Sun rejection angle must be guaranteed for the rendezvous cameras and this leads finally to shift this mean ocal solar time to 11:00.
- Netlander relay: A relay session consists in communicating with one of the NetLander stations, using a UHF radio link. The data collected is first stored on-board the Orbiter before being transmitted to Earth once a day. The mission must guarantee that two relay sessions are allocated per sol to each Netlander station (one nominal and one back-up).
- Pointing requirements: The scientific payload fields of view are oriented towards the Nadir side of the Orbiter. The Orbiter attitude has to be defined in each mission phase to satisfy pointing requirements coming from both the instruments and the High Gain Antenna communicating to the Earth. Solar power output optimization is then achieved through the best orientation of the solar array.

The resulting attitude requirements for the Orbiter are the following :

Mission need	Attitude requirement	Compatibility with other mission or Support needs
Netlander relay	Nadir pointing	Communication to Earth, Orbital Science
Orbital Science	Nadir pointing	Communication to Earth, Netlander relay
Intermediate rendezvous and Search	Cameras Fov close to velocity vector	Communication to Earth,
Terminal parking, rendezvous & capture	Cameras Fov close to velocity vector	Communication to Earth

Table 1

During the mission phases where conflicts appear between NetLander relay and Rendezvous activities, the resources are shared over a single orbit in the following way (Figure 3):
- Netlander relay sessions take place over the quarter of orbit located around the sub-solar point;
- Rendezvous activities such as observations and maneuvers have to be performed before and after NetLander relay i.e. around the polar regions.
Transition phases have to be accounted for at the end of each activity to adjust the spacecraft pointing through an attitude slew maneuver.

The orbit geometry of the orbit and the various attitude requirements lead to the following observations :
- the Netlander relay sessions, as they nominally require optimal power for the stations, must take place on the quarter of orbit situated around the sub-solar point.
- Assuming a co-elliptical orbit of the target, the orbit parts favorable for Intermediate Rendezvous and Search sub-phases would be the ones over the North or South pole, where the observability of the target from cameras is the best.
- The transfer from Intermediate Rendezvous (co-elliptical orbit) to Terminal Parking orbit would optimally be achieved around the Ascending Node.

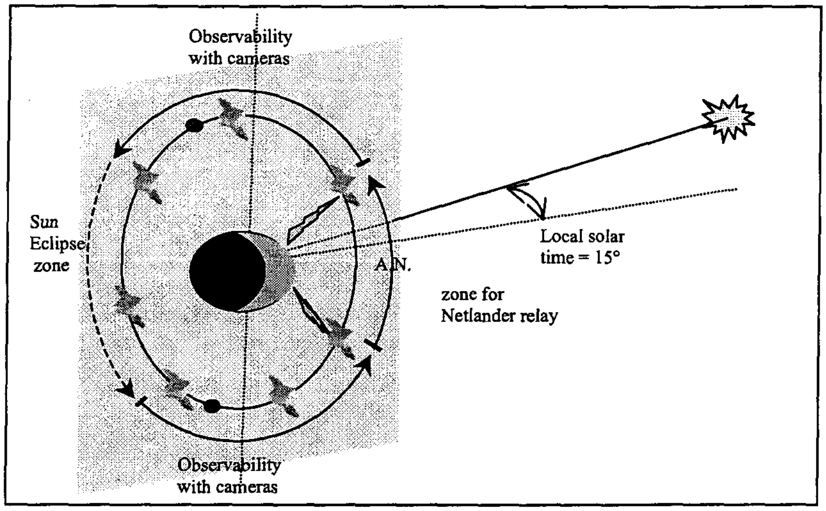

Figure 3

Impact of Rendezvous on the Orbiter design

Propulsion system
Besides the rendezvous phase, the main requirement is the capability to execute in a robust way large translational maneuvers while minimizing gravity losses (Mars insertion, orbital transfers). These maneuvers and the smaller ones used for cruise trajectory correction or Netlanders release can all be performed along a unique direction in the Orbiter frame.
However, when the rendezvous experiment is being considered, new requirements come forward:
- provide 6 degrees of freedom with the smallest possible coupling between axes,
- provide a minimum impulse along each translation axis in the mm/s range to offer the fine tuning needed during the final approach and capture.

The propulsion system addressing these requirements includes a fixed 450 N main engine and a set of 10 N thrusters providing 6 degrees of freedom (some of these thrusters participate to the maneuver direction control during main engine operation). The thruster configuration (Figure 3) is composed of 4 pods located in the same plane carrying each 4 thrusters. However, due to pluming constraints, the thrusting efficiency is not identical along all axes.

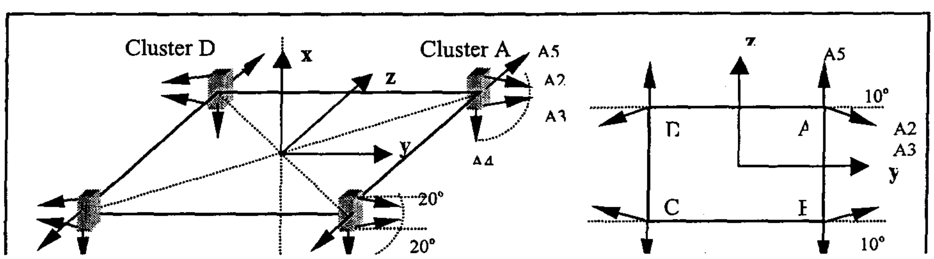

Figure 4: Nominal Thruster configuration (redundant thrusters not represented)

As for attitude control, it is achieved with 4 x 15 Nms reaction wheels mounted in a tetrahedral configuration. Except periodic wheel off-loadings, the spacecraft is guaranteed to be quiet which is required to achieve good navigation performances. This choice that satisfies rendezvous is however mainly driven by fuel usage considerations.

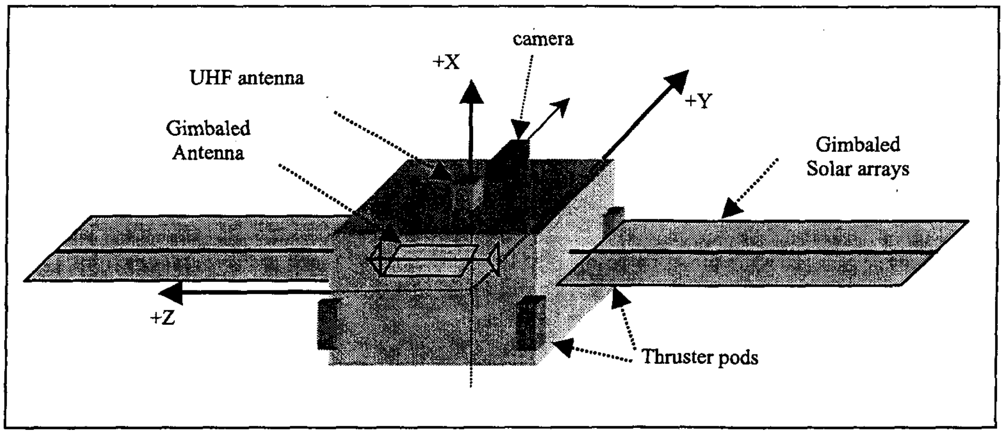

Figure 5: Orbiter configuration

Another requirement driven by the rendezvous experiment applies on maneuver execution control and knowledge since trajectory dispersions and navigation uncertainties need to be minimized. However, for Mars Premier Orbiter this requirement is also imposed by the NetLander ejection phase where very accurate insertion velocities must be achieved. These common needs lead then to the introduction of a high precision inertial measurement unit.

Maneuver execution

Because of the rendezvous experiment, the nominal mode to execute translation maneuvers becomes insufficient and an extra mode has to be implemented.

- ***X mode*** is the nominal one used to achieve maneuvers during most of the mission except terminal rendezvous. The translation maneuver is always performed along the spacecraft positive X-axis that corresponds to the direction of the main thruster. Closed loop control based on gyros and accelerometers measurements steers the thrust direction in order to achieve the desired maneuver direction and magnitude. This mode that offers the best efficiency and execution accuracy is preferably used for the maneuvers greater than 1 m/s. Prior to any translation maneuver, a spacecraft attitude change is therefore required to align the X-axis along the desired thrust direction. This constraint is acceptable even for intermediate rendezvous since the interval between maneuvers is at least in the hour range.

- ***Vector mode*** is to be used during terminal rendezvous to satisfy the mobility and controllability requirements. Here, the translation maneuver can be performed along any desired direction in the spacecraft reference frame by selecting the right subset of 10 N thrusters and adjusting the pulse duration. This capability allows to perform the requested maneuver right away without slewing the spacecraft which is required from both observability and controllability viewpoints. In this mode, closed loop control is provided only for the maneuver magnitude since residual errors in direction can be compensated by subsequent maneuvers.

In both modes, closed loop control is available if the maneuver magnitude is exceeding 10 cm/s. For smaller magnitudes, control is achieved in open loop using a thrust model depending on the thruster and the pulse duration. For both modes, the magnitude error model is given in Table 2 (3 σ values)

Δv range (m/s)	X-mode $\delta(\Delta v)$ in m/s	Vector mode $\delta(\Delta v)$ in m/s
. < 0.1	$10^{-3}+5.10^{-2}.\Delta v$	$10^{-3}+5.10^{-2}.\Delta v$
0.1 < . < 1.0	$6\ 10^{-3}$	$6.10^{-3}+9.10^{-3}.\Delta v$
> 1.0	$6.10^{-3}.\ \Delta v$	$1.5.10^{-2}.\Delta v$

Table 2

Antenna design
The Orbiter has to communicate with Earth using a High Gain Antenna that requires a fine pointing (+-0.5°). For mission activities such as Netlander relay or orbital science, simultaneous Earth communication is not a strong requirement, therefore it could be envisioned to communicate with the spacecraft inertially pointed. This becomes however forbidden during rendezvous operations since "quasi-permanent" Earth communication has to be maintained while the spacecraft is tracking the target. One option is to mount the antenna on a 2-axis gimbal to be independant from the spacecraft attitude changes. However the complexity of the mecanism increases and the limits of the motion range bring other problems. The preferred option consists in providing the antenna with a single axis gimbal while the second degree of mobility is obtained by rotating the spacecraft along some orthogonal axis. This approach is already selected to maintain the gimbaled solar arrays optimally pointed during Netlander relay activities. For commonality, the antenna gimbal is therefore made parallel to the solar arrays axis (i.e. spacecraft Z-axis). This choice drives further the accommodation of the rendezvous cameras since the target tracking axis has to be parallel to the XY plane.

Cameras Accommodation & Interface
- The Rendezvous package provided by JPL is composed of two cameras (NAC and WAC) and the Target with its Latch and Release mechanism. In order to simplify mission design and to optimize the share of Orbiter resources, it is desirable to mount the cameras on the Orbiter to allow Target observation immediately before and after without being obliged to perform a large slew. The solution adopted is to accommodate the NAC on the X+ axis face of the spacecraft with its boresight parallel to the Y axis (Figure 5). During Netlander relay activities, the spacecraft is Nadir pointed and the solar panels axes are oriented orthogonal to the Sun. The Sun direction being only 15° off the orbital plane in the worst case, this configuration guarantees a small angle (< 15°) between the NAC boresight and the Target that enables a short reacquisition time.since the Target sits close to the velocity vector in most phases of the mission.
- Cameras are interfaced via a serial bus (LVDS) to the spacecraft avionics that is in charge of controlling data transfer and image processing. Since a complete NAC image represents 2 Mbytes of data and takes up to 15s for the transfer, the load on a single computer was considered too important. Another computer has therefore been added to perform the specific tasks of acquiring and processing pictures.

Attitude guidance
Attitude guidance is particularly driven by the orbital phase of the mission where several guidance modes have to be implemented. Most of the time, two or more pointing constraints need to be satisfied simultaneously and this freezes completely the spacecraft attitude. Maintaining an optimal power and performing Earth communication represent two basic requirements that cannot be achieved altogether if another activity need to be realized at the same time (Netlander relay, translation maneuver, ..). Priorities have therefore to be allocated between power (Z axis orthogonal to Sun) and Earth communication (Z-axis orthogonal to Earth). The different guidance modes to be developed are summarized as follows:

- ***Netlander relay***: X-axis pointed to Mars with simultaneous power or communication constraint
- ***Translation maneuver***: X-axis pointed in any direction with simultaneous power or communication constraint
- ***Attitude slew***: X-axis following a specific profile with simultaneous power or communication constraint
- ***Rendezvous***: Y-axis pointed to the Target with simultaneous power or communication constraint.

All guidance modes can be implemented through the use of a single and generic approach. It is based on the definition of two varying direction vectors using ephemeredes ($\vec{v}_P, \vec{v}_{\perp Z}$): $\vec{v}_P$ represents the inertial direction to point the spacecraft X-axis or Y-axis (center of Mars, velocity axis, target direction, delta-v maneuver axis), $\vec{v}_{\perp Z}$ represents the direction to which the spacecraft Z-axis needs to be maintained orthogonal (Sun, Earth). This definition allows to compute the components of the spacecraft reference axes: if X-axis is directly defined by $\vec{v}_P$, Z-axis is obtained by the formula: $\vec{v}_Z = \vec{v}_X \times \vec{v}_{\perp Z}$ and Y-axis is derived as orthogonal to the previous ones. Afterwards, from the knowledge of this triedrum, the on board computer determines the quaternion representing the desired attitude.

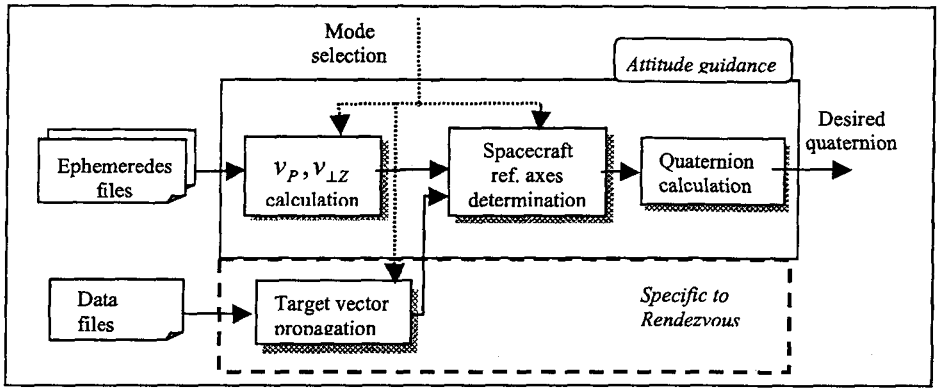

Figure 6: Attitude Guidance

The mode selection consists in creating a mapping between the spacecraft reference axes (X-axis, Z-axis) and two ephemeredes files defining the evolution of the $\vec{v}_P, \vec{v}_{\perp Z}$ vectors in the inertial frame.

The ephemeredes files contain polynomials coefficients with very different update rates since the evolution of the direction vectors varies from the 10^{-4} mrad range when specifying Sun and Earth directions to a few milliradians when defining slew maneuvers.

The principle is to elaborate these ephemeredes files on the ground and upload them some time before they are needed (usually at least one day in advance). This operational approach is applicable for most of the mission since the directions are known in advance.

Impact from RdV: The terminal rendezvous phase represents here an exception since the Target direction has to be updated on board using localization data from camera pictures. Attitude guidance can still rely on ephemeredes files for the definition of $\vec{v}_{\perp Z}$ but it includes now another functionality that is continuously computing the relative target direction. This function is implemented as a simple propagator that extrapolates to the current time the updated relative vector produced by orbit determination. For that purpose, the propagator relies on transition matrices generated on the ground and available in data files. Further explanations are given in the Target tracking paragraph of the Rendezvous system description.

Specific requirements coming from the search phase

In the MSR reference mission, the Target is not completely lost in space at the time of search. The initial uncertainties in the Target orbital parameters propagate as follows: the mean anomaly is unknown and the ascending node is fixed within +/- 1 deg.

The ability to detect the OS implies a scanning of this annulus by the Long Range Camera that will acquire a mosaic of images. On Mars Premier Orbiter, this camera was initially equipped with a 2D gimbal, but was later descoped and the camera ended up body mounted: the scanning is therefore to be entirely performed by the spacecraft. This generates stringent requirements both on orbiter pointing stability and attitude controllability that are summarized herebelow.

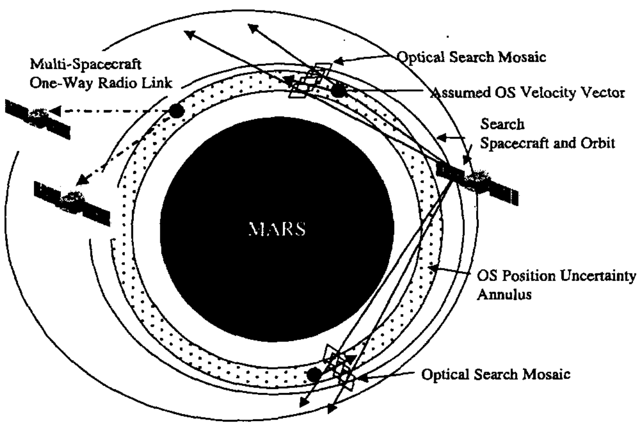

Figure 7

Attitude guidance objective: The Target must be searched over a zone of the sky that is larger than the NAC field of view. Therefore, the NAC boresight has to be displaced so that its field of view will sequentially cover for image acquisition purpose the entire zone. This area can be described in a moving frame by a geometric pattern called "mosaic". Each element of the "mosaic" is referred as a "tile".

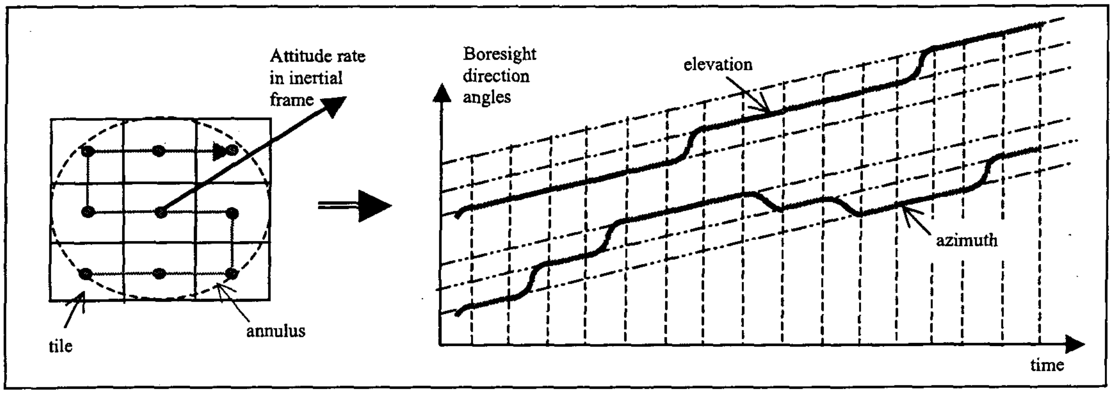

Figure 8: Example of attitude profile for a 3x3 mosaic

The guidance system has to elaborate the attitude profile that satisfies the image acquisition constraint at each tile and that steers the NAC boresight from tile to tile over the entire mosaic. An example of attitude profile is illustrated on Figure X.

The Target detection principle is the following: during image acquisition, the camera boresight is moved to cancel the a priori relative angular motion of the OS; if the OS is within the camera FOV, the light being reflected produce a bright spot on the camera detector whereas stars generate a series of distinguishable rays. The objective to maintain the OS within +/-1 pixel during the image exposure would imply that the attitude rate of the spacecraft matches the OS motion with a +/-24 μm accuracy. The spacecraft is actually required to achieve a pointing stability that is expressed as follows: "the attitude displacement due to the rate error shall be less than 50 microrads (3-sigma) during any period of 1 second".

Control objective: The attitude profile to be tracked represents the a-priori motion of the Target relative to the Orbiter and the control objective is twofold when tracking this profile:

- the profile rate has to be achieved with a high accuracy (highest priority),
- the attitude accuracy must guarantee that the Target remains always in the NAC field of view.

Therefore, this is achieved by implementing two different control modes illustrated on Figure 8:
- nominal attitude control is active when moving from one tile to the next,
- rate control is used during the period of image acquisition (all mechanisms such as solar arrays and antenna gimbals are switched off to reduce perturbations).

After switching into rate control, a several seconds stabilization period allows the rate to converge to the desired value. Image acquisition is then triggered 2-3 times for a maximum period of 10s at each occurrence. Over the rate control period, the cumulative attitude error due to rate bias and jitter must remain within bounds that correspond to the tile overlap amplitude.

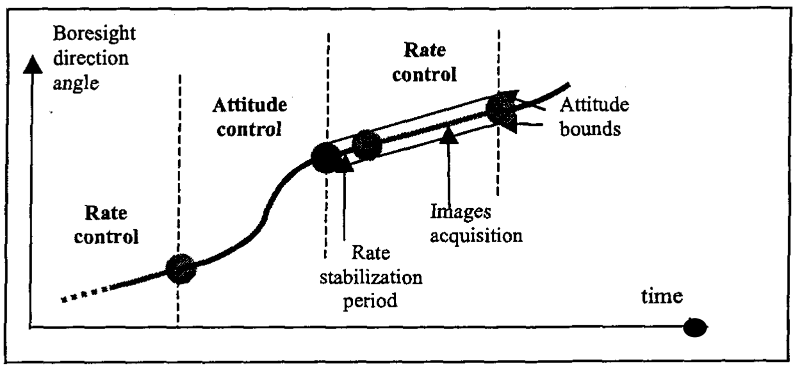

Figure 9: Attitude control modes during Search

Beyond control, another area potentially impacted is the programming of ephemeredes. We have seen in the previous chapter that any attitude profile could be generated on the ground and uploaded as a set of 2 ephemeredes files defining the vectors $\vec{v}_P$ and $\vec{v}_{\perp Z}$. Generalizing this approach to the attitude profile segments where rate control is applied can be envisioned but it requires the derivation of these vectors and this generates some extra noise. Adding specific ephemeredes defining the rate evolution of the $\vec{v}_P$ and $\vec{v}_{\perp Z}$ vectors for the rate control periods was therefore under analysis.

For any given mosaic, the durations of the rate and attitude control periods are respectively 70 s and 50 s. The first figure includes the 10 s stabilization period, 3 picture exposure (10 s each) and 2 picture data transfer (15 s each). As for the second figure, it is driven by the torque authority of the reaction wheels. A 3x3 mosaic lasts therefore 17 minutes where as a 4x4 mosaic requires 31 minutes to completed.

Presentation of the Rendezvous system

To reduce the implementation cost on the flight segment, the rendezvous system has been designed to rely for most phases on ground control and on a severely constrained autonomy during the critical periods like final approach and capture.
First, this autonomous behavior is made compatible with the execution of only deterministic time-tagged functions that are programmed beforehand in sequences by ground operators. Second, the implementation of navigation and guidance functionalities is distributed between the flight and ground segments. Most of the computations that correspond to updating quantities like transfer and gain matrices are performed on the ground. To execute algorithms, the flight segment combines later these quantities uploaded in data files with on board measurements using generic mathematical operators. [1]. To make this implementation viable, the strategy is to constrain the Orbiter on a predefined trajectory and timeline so that the Orbiter absolute and relative states remain during the autonomous phase in the close vicinity of some a priori values at any given time. As long as this condition is satisfied, quantities computed on the ground stay valid for a later use by the on board algorithms.

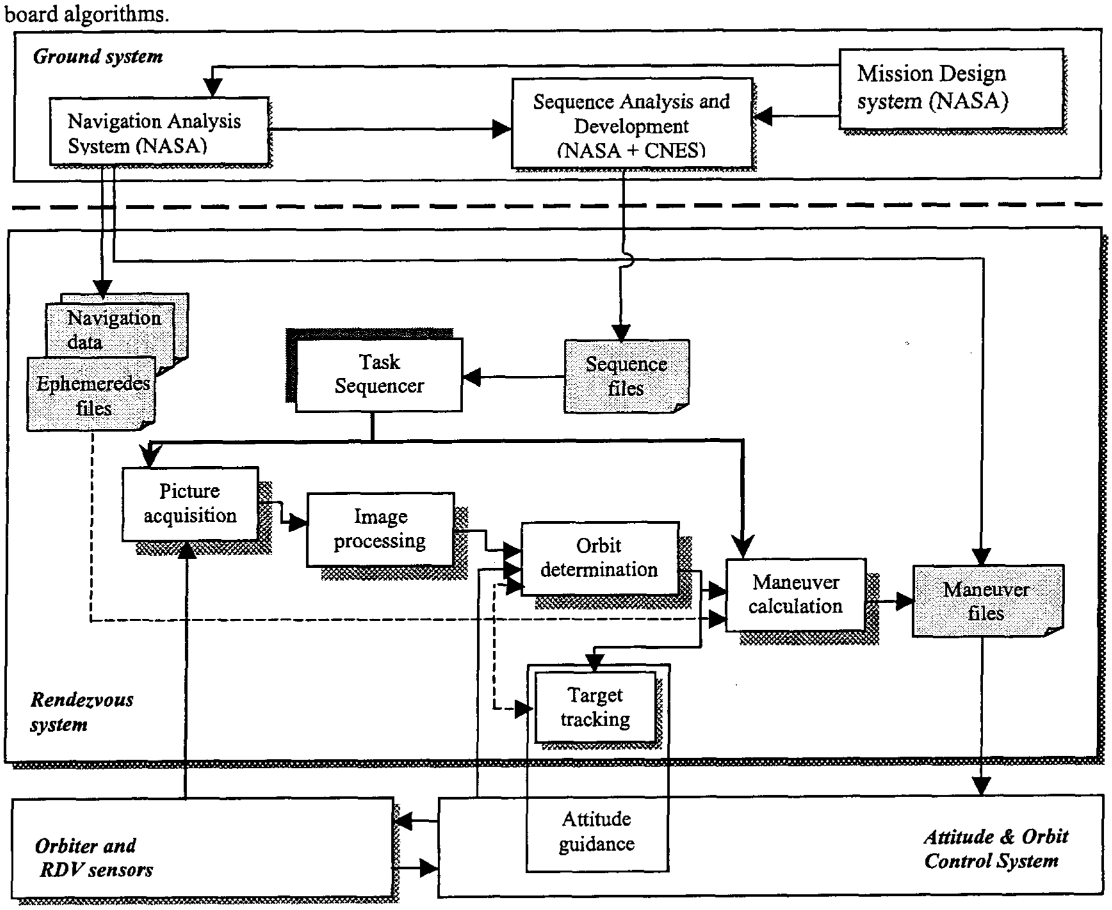

Figure 10: Rendezvous System Architecture

The different data files needed to perform the rendezvous experiment are the following:
- Ephemeredes file (defining the Orbiter and Target nominal states),
- Navigation & guidance file (containing the gain and transfer matrices),
- Maneuver file (defining the dates of all maneuvers and the value of deterministic maneuvers)
- Sequence file (scheduling the activities to be performed).

These data are elaborated in the JPL/NASA Rendezvous control center and transmitted to the CNES mission center where they are cross-checked before being uploaded.
The volume of data to be uplinked increases when the Orbiter is approaching the Target since navigation and guidance imply a higher frequency of observations and maneuvers. The Terminal Rendezvous Phase represents therefore a sizing case. Considering such a phase with a conservative 24 hours duration, the overall volume gets into the 75 kiloBytes range. Fortunately, the Orbiter antenna allows a 2 kbps telecommand rate and the uplink of the whole scenario takes about 5 minutes. The burden on the communication channel is then regarded as perfectly acceptable.

The functionalities to be developed in the Orbiter Flight System for the specific needs of rendezvous are limited to the following ones: 1) Target tracking, 2) Picture acquisition and processing, 3) Orbit determination, 4) Maneuver Calculation and some details of their implementation are given in the next paragraphs.

Image processing

- During the simulation of the Target search, Target identification is performed on the ground and on board processing is limited to some basic image compression algorithm allowing to reduce drastically the telemetry volume. Image compression consists in extracting the bright objects obtained by grouping all contiguous pixels whose luminance exceeds a given threshold. Data stored and sent later to the ground corresponds to the geometric characteristics of each visual object.

- During terminal rendezvous that requires closed loop control, image processing is still based on the extraction of regions which luminance exceeds a given threshold but it includes now the functions of target identification plus the determination of its diameter and center coordinates. The capability of the simple threshold technique to yield the right target contour is guaranteed for two reasons: (1) Mars never appears in the target background thanks to the selected approach scenario, (2), image is acquired with the optimal time of exposure that is computed using the target reflective characteristics and the estimated distance.

Target identification is then easily achieved using criteria of size, luminance and possibly shape that suffice to eliminate the groups of pixels corresponding to stars.

Next, the determination of the observables that are the target diameter and the center coordinates can be performed using different techniques. A technique that is regarded as powerful from the computation and accuracy viewpoints consists in using the luminance weight (sum of luminances) and the barycenter of the target visible area.

The target aspect in the image depends on the Orbiter-Target-Sun angle (or "lighting" angle) and it varies over a complete orbit from a full moon to a narrow crescent (see Figure 11). Assuming a diffuse and isotropic reflection, the luminance weight S_l is analytically related to the diameter and the lighting angle θ by the formula (1):

$$S_l = \frac{1}{6}.k_r.D^2_{pixels}.(\sin^3\theta + \cos\theta.(\pi - \theta - \sin(2\theta)/2)) \qquad (1)$$

where k_r is a coefficient depending on the target distance and the reflection ratio.

The "lighting" angle is easily derived on board from the spacecraft attitude and the Sun direction that is uploaded through slowly varying ephemeredes. The kr coefficient must be calibrated on the ground and recalibrated on board. Then, from pixels representing the visible target area, the diameter is therefore computed by applying formula (2).

$$D = (6.S_l / k_r(\sin^3\theta + \cos\theta.(\pi - \theta - \sin(2\theta)/2)))^{1/2} \qquad (2)$$

The coordinates of the Target center are computed in a similar way by extracting the barycenter of the visible area and by applying formulas taking into account the phase angle and the Sun direction in the picture frame.

These observables are then differenced from the estimated quantities (x, y, range) obtained by mapping the relative position vector into the camera frame. The residuals are finally provided to the Orbit Determination process.

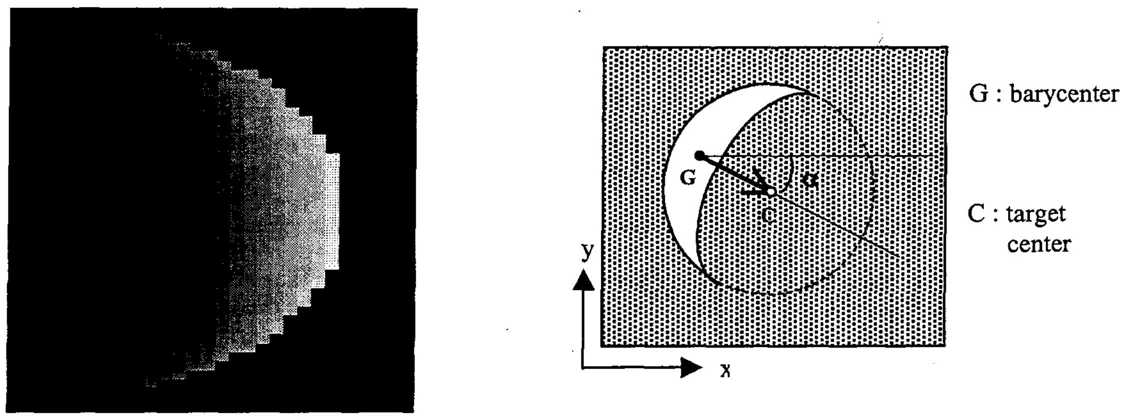

Figure 11: Target aspect (phase angle = 120°)

Model of accuracy
A preliminary evaluation of this technique has been performed in simulation. The target is modeled as a perfect sphere with isotropic diffuse reflective properties and the simulated noise includes the spatial digitization in pixels and the luminance noise per pixel. For any distance and lighting angle considered, the diameter of the Target is extracted 500 times for different target locations obtained from a uniform random distribution over a pixel.
The following table shows the accuracy obtained for different distances in the range [20 m – 5000 m] and the worst phase angle (150°). The diameter is characterized by two quantities: bias and standard deviation that are expressed in pixels

Distance (m)	Diameter (pixels)	**d** bias	**d** std. dev.
5000	1.66	0.142	0.313
2500	3.33	0.094	0.127
1000	8.33	0.018	0.045
500	16.66	0.007	0.017
100	83.33	0.002	0.003
20	416.66	0.001	0.001

Table 3

The simulation results can be synthesized with the following error model considered independent from the lighting angle and the relative distance.
Center coordinates: bias 0.15 pixel (1 σ) + noise 0.4 pixel (1 σ)
Diameter: bias 0.25 pixel/diameter (1 σ) + noise 0.5 pixel / diameter (1 σ)

The noise on the center coordinates is not a concern since it is masked by the attitude knowledge noise that is larger by one order of magnitude.

Target tracking
During the various rendezvous phases of the mission, the spacecraft attitude must be steered to maintain the Target within the camera FOV while satisfying the solar arrays and high gain antenna pointing constraints. This target tracking functionality is actually integrated in the common attitude guidance module which principle has been described hereabove.
The addition consists in a simple propagator that enables to update continuously the Orbiter and Target estimated states that are used afterwards to compute the current relative geometry. This propagator gets initialized with the states provided by the ephemeredes. The propagated estimated states are then periodically adjusted by adding the state corrections ΔX produced by Orbit Determination.
These computations are not triggered each time via sequence commands. They are performed instead in a synchronous way at the attitude guidance rate (1-2 Hz). The target tracking functionality is only activated at the beginning of a rendezvous activity session.

Orbit determination

Orbit determination (OD) starts automatically when the Image Processing has completed its task. The OD function consists in estimating the Target state relative to the spacecraft through the use of a Kalman filter technique. The output of the Picture acquisition – Image processing process is the difference expressed in the camera frame between the target measured position and the estimated position. This latter is obtained by propagating the estimated state computed at the previous cycle (prediction). The state update equation can then be expressed as the product of these residuals R_i by the Kalman gain M_1 provided in the data files:

$$R_i = \Delta(x, y, range)$$

$$\Delta\overline{X}_i = \cdot M1_{i(6x3)} \cdot R_i'$$

where $\mathbf{\Delta X_i}$ represents the state corrections already mapped to a given time.

The state corrections $\mathbf{\Delta X_i}$ are then provided to the Target tracking functionality that updates the current Target propagated states.

As part of the OD process, the Orbiter states should be corrected as well. However, according to this simplified design, the update is only based on the information produced by the AOCS of accumulated translational velocity due to all thruster firings. This process not relying on observations is therefore performed within the Target tracking function.

Maneuver calculation

The vector representing the Target state relative to the Spacecraft ***SC2OS*** is continuously computed by the Target tracking functionality. Maneuver calculation uses this input to produce the difference vector ***ΔSC2OS*** between the nominal and the estimated relative states. A pair of maneuvers (Δv_1, Δv_2) can then be determined to drive the estimated relative state back to the nominal over a given time horizon T. By providing the transfer matrix M2(T) that relates the difference of the relative states at date t and date t+T, the maneuvers are computed as follows:

$$M2_{6x6}(T) = \begin{bmatrix} M2_{11} & M2_{12} \\ M2_{21} & M2_{22} \end{bmatrix} \quad where \quad \Delta SC2OS(t+T) = M2(T).\Delta SC2OS(t)$$

$$\Delta\overline{v}_1 = [\Delta SC2OS_{1:3} \cdot M2_{11} + \Delta SC2OS_{4:6} \cdot M2_{21}] \cdot M2_{21}^{-1}$$

$$\Delta\overline{v}_2 = [\Delta SC2OS_{1:3} \cdot M2_{12} + (\Delta SC2OS_{4:6} - \Delta\overline{v}_1) \cdot M2_{22}]$$

The values are added to the current values on the Maneuver File, and that file is then updated with the sum:

Rendezvous executive with command sequencing

The operation of the Rendezvous Control System is purely deterministic by design since all sensing, navigation and guidance activities necessary to drive the orbiter back on the nominal trajectory are performed at specific dates that are predefined on the ground and stored in sequences uploaded beforehand. A deterministic behavior does not imply however the inability to react to some anomalies when the behavior steps out of the nominal without putting the mission in immediate danger. Here we want to discriminate these anomalies from the real contingency situations where the execution of abort maneuvers is required. An example of this type of anomaly is the failure to detect the target in the camera FOV that may occur if the navigation error gets too large. This situation can be expected once in a while after a long period of non observation like an eclipse and it is important to deal with it without triggering a systematic abort.

In this situation, the strategy is to start a target search involving a series of image acquisitions to scan the region around the last estimated target direction. If the target is detected, it is possible to return to the nominal sequence of operations, otherwise the abort is triggered and the mission is interrupted.

In order to limit the number of combinations and the amount of tests to perform on the ground, it is envisioned to introduce only a few branching points (one each time the target gets out of eclipse to allow a reacquisition, one at the end of each reacquisition phase).

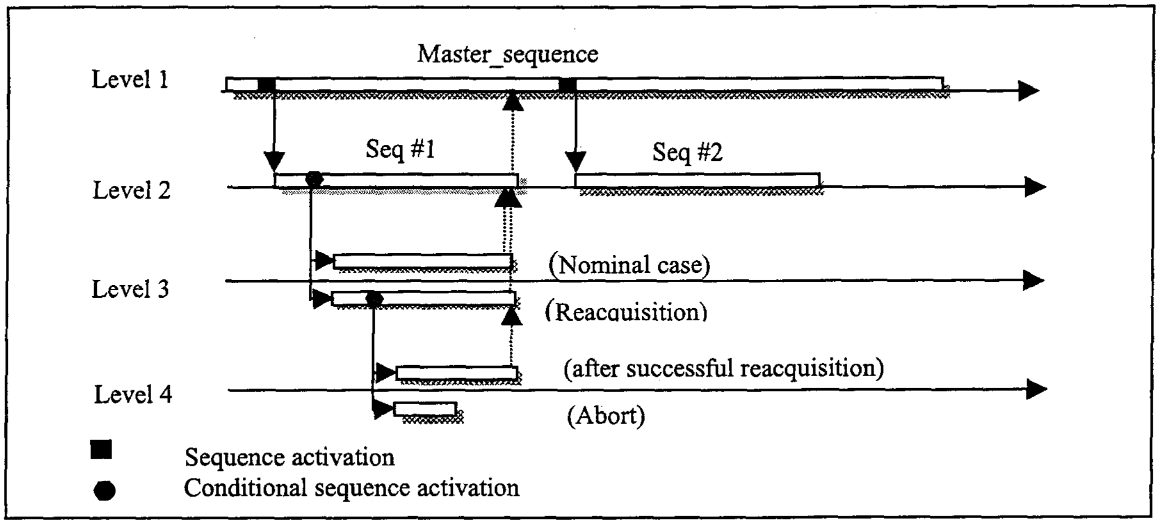

Figure 10:

To further simplify the executive, the branching mechanism is implemented through a conditional execution of sequences. It is programmed using the following instruction:

- COND_SEQ condition_name, val_min, val_max,
  f ilename1, filename2

If the specified condition has a value comprised between ***val_min*** and ***val_max***, the sequence 'filename1' is activated, if not the system activates sequence 'filename2'.

This programming flexibility implies: 1) the decomposition of the mission into smaller sequences, 2) the capability by the sequence manager to deal with up to 4 levels of sequences. However, numerous sequences will be identical since the same type of activity is most of the time repeated every orbit. They will be made reusable by specifying the date of each command in a relative sense.

The programming of the rendezvous experiment can be performed using the list of commands listed in Table 4.

Command name	Description	Frequency of use
CAM_POINT	Request the AOCST to point camera at target, specifying particular camera.	One request at beginning of tracking session.
PICT_ACQ	Take a picture and deliver to the RSCS, the command will trigger OD calculations	One command per picture
PLAN_MAV	Plan a maneuver based on current OD, and store result on the Maneuver File.	One command per maneuver opportunity.
SET_DFILE	Assignment of data file to the RSCS, specify one of several files.	Before the start of autonomous session
COND_SEQ	Conditional sequence activation	Up to 2 requests per orbit
RESET_DATA	Null currently estimated state biases.	At start of rendezvous

Table 4: Sequence commands

It was initially proposed to generalize this operational concept to the whole mission by upgrading the executive with the new sequence management functionalities. However, to reduce the on-board software validation cost, it has been decided to separate the rendezvous experiment from the rest of the mission. Rendezvous is then performed using a single mode of operation that covers the different phases of the experiment from release to "capture".

Concept validation by computer simulation

The computer simulation is built according to the operational approach described hereabove and therefore allows to exercise the share of responsibilities that has been defined between CNES and JPL. The part developed by CNES includes the dynamic behavior of both Orbiter and Target in the Mars environment, the actuators and sensors models, the Orbiter AOCS and the rendezvous module. It is implemented in Matlab/Simulink and each block is developed as a S-function coded in C language.

The rendezvous scenario is defined by JPL and is provided to CNES as a set of variables stored in Matlab files. They are further converted into several ASCII files (Ephemeredes file, Maneuver file, Navigation & propagation data file). For the purpose of validation, we are considering only the nominal scenario and sequencing is therefore trivial since it is directly driven by picture acquisition and maneuver calculations. Therefore no sequence file needs to be provided.

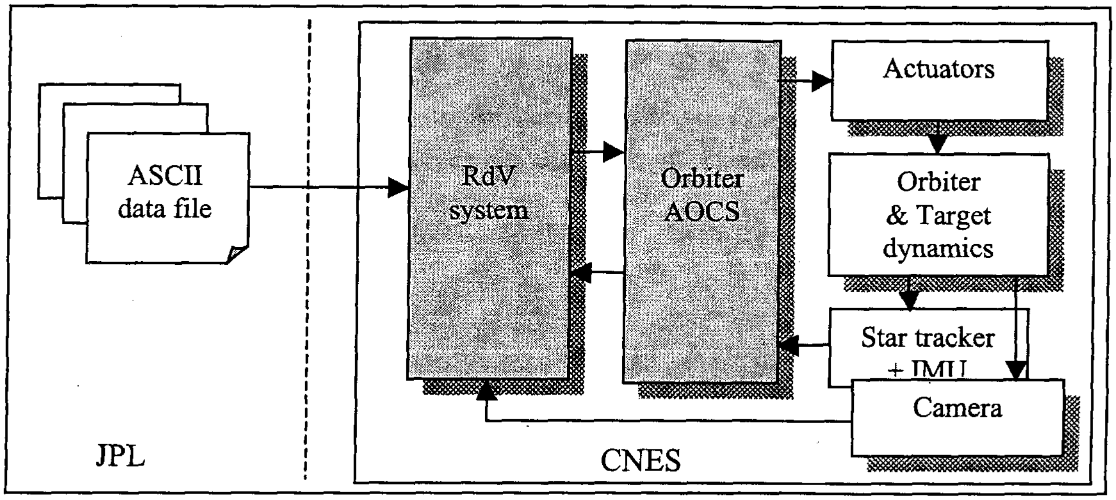

Figure 11: Bloc diagram of the CNES-JPL joint simulation

The scenario consists in driving the spacecraft from a distance of several kms down to an hypothetical capture point located at 2 m from the spacecraft center of mass. The approach that spans 12 orbits (24 hrs) is performed through a series of football orbits and their dimensions get smaller when the Orbiter approaches the Target .

The scenario is illustrated on Figures X.Y that display the Orbital position in a moving frame attached to the Target. The dotted and solid line represent respectively the desired and achieved trajectories. Deterministic and statistical maneuvers are indicated by diamonds and stars.

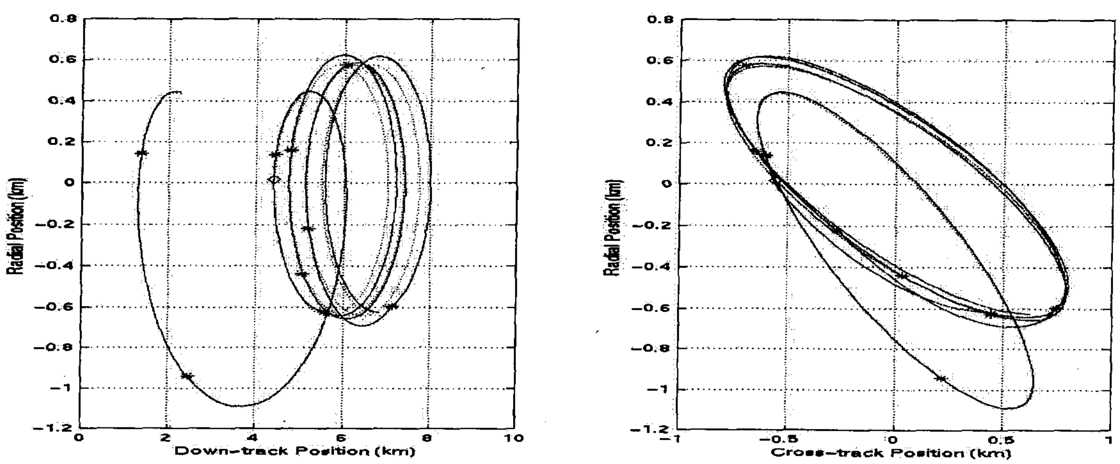

Figure 12.1: Relative trajectory during the first 8 hours period

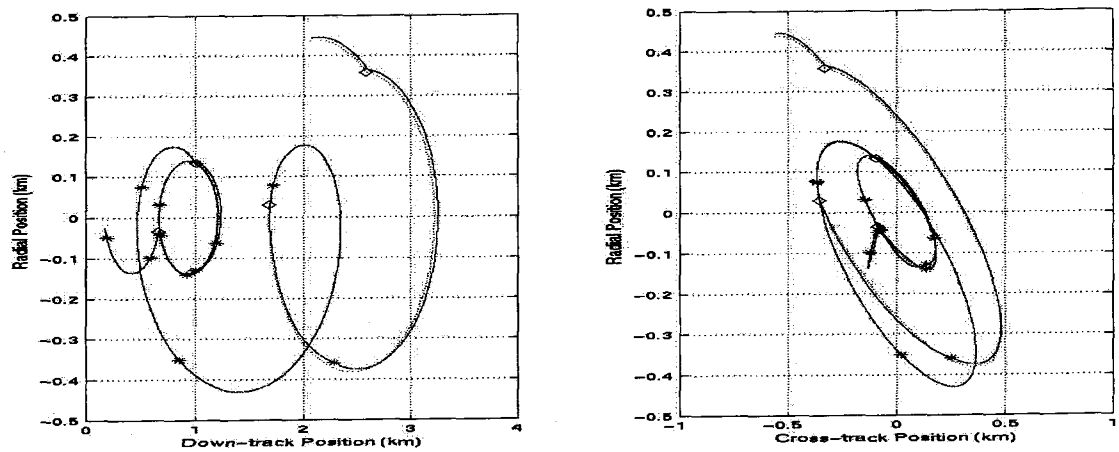

Figure 12.2: Relative trajectory during the intermediate 8 hours period

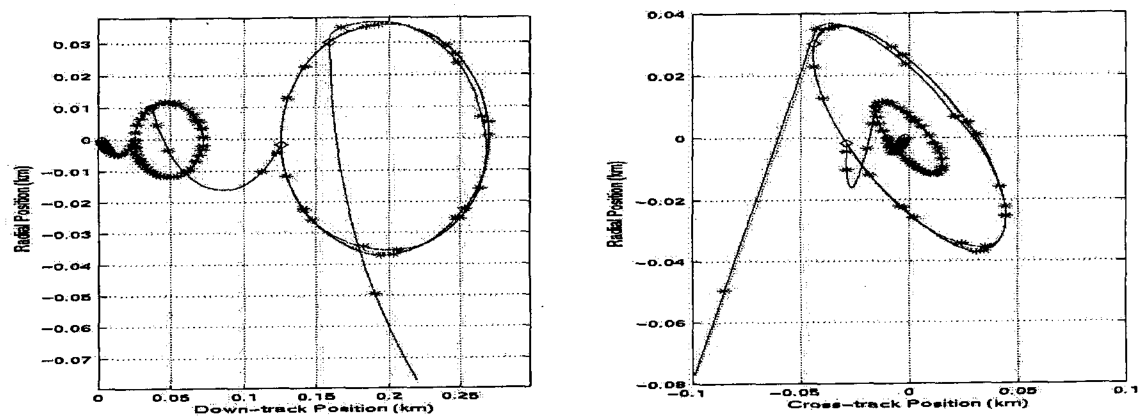

Figure 12.3: Relative trajectory during the last 8 hours period

The nominal scenario includes 600 picture acquisitions for navigation, 12 deterministic maneuvers and 68 pairs of statistical maneuvers. During most of the scenario, picture acquisition is only performed outside eclipse periods when the phase angle is above 30°.

The nominal parameter configuration is the following:

- Initial navigation error: 10 m (1 σ):, 1 cm/s on each axis (1 σ) for both Orbiter and Target,
- Observation error: _ pixel for the Target location (1 σ), _ pixel / nominal diameter (1 σ)
- Maneuver execution accuracy: $\delta\Delta V = 5.10^{-2}\,\Delta V$ (3 σ) and 2 m/s minimum impulse
- Pointing knowledge error is 0.25 mrad (1 σ) on each axis.

Series of Monte Carlo simulations (500 runs) have been performed to test the system robustness to the variation of different parameters. We are particularly interested by the following: initial navigation error, camera measurement noise, maneuver execution errors, statistical maneuver frequency and picture acquisition frequency.

Performance is measured according to the following criteria:

- number of failures,
- total delta-v consumption including deterministic and statistical maneuvers (mean and variance),
- number of target losses,
- position accuracy at "capture".

	Number of failures	Target losses mean/ variance	Target centering along X,Y axis (pixels)	"Capture" accuracy (1σ)	Total delta-v mean (m/s) / variance (m/s)
Nominal case	0	0.03 / 0.17	19.1 - 28.6	1.03 cm	2.32 / 0.17
Larger maneuver execution error	0	1.51 / 1.25	49.3 – 74.3	1.42 cm	2.71 / 0.42
Diameter measurement noise x 5	0	0.23 / 0.51	24.2 / 33.2	1.17 cm	2.45 / 0.28
Maneuver frequency/ 2	0	0.44 / 1.46	35.2 – 42.7	1.28 cm	2.31 / 0.26
Picture acq. rate / 2	0	1.03 / 0.97	40.3 – 71.5	1.16 cm	2.66 / 0.37

Table 5: Monte Carlo results

In the nominal case, the rendezvous system shows an excellent robustness with no case of failure and only one observation missed every 30 runs of the Monte Carlo simulation. Performances are not noticeably affected when the maneuver execution error is increased up to 10% of the magnitude and the minimum delta-v becomes 4 mm/s.
The system still shows no noticeable sensitivity to some increase of the noise on the Target diameter (noise x 5) as long as it is inversely proportional to the diameter and the error does not include any significant bias term.
As far as the scenario is concerned, the number of picture acquisitions can definitely be divided by 2 (evenly over the 12 orbits) with a slight degradation of the performances: the target gets out of the camera FOV only once per run. However, this represents a limit since a further reduction brings a significant number of failures (division by 3). The failures occur generally within the first orbits when the frequency of picture acquisitions is the lowest (period = 240 s x 3 = 720 s).

A similar conclusion applies to the reduction of the number of statistical maneuvers. An even reduction by a factor 2 remains acceptable but failures start to occur when the reduction factor gets larger (3). In those cases, the guidance strategy does not allow to maintain the Orbiter close enough from the desired trajectory. Navigation and guidance data become more and more inaccurate and stability is lost.

Anyway, these results do not express the system absolute performances since they depend on the tuning of the navigation filter. Another tuning with different noise parameters would possibly provide better performances. The purpose is to illustrate the achievable robustness in presence of realistic perturbations.

References:
(1) A combined Open-Loop and Autonomous Search and Rendezvous Navigation System for the CNES/NASA Mars Premier Orbiter Mission. J.E.Riedel and al.