LAREDO: LAunching, REndezvous and DOcking Simulation Tool

Miguel Angel Molina, Jose Prieto
GMV – c/ Isaac Newton, 11
PTM Tres Cantos
28760 Madrid
SPAIN
Tel: +34 918072100; Fax: +34 918072199
mamc@gmv.es, jjpm@gmv.es

Gonzalo Garcia, Theresa Beech
GMV Space Systems
1 Research Ct, Suite 450
Rockville, MD 20850
USA
Tel: +1 301 216 3840; Fax: +1 301 519 8001
US_marketing@gmvspacesystems.com

ABSTRACT

This paper intends to provide a description of the LAREDO tool including architecture, design choices and mathematical formulations. The goal of LAREDO, which is a six DOF simulator developed under Matlab/Simulink-based environment, is to reproduce the Rendezvous and Docking (RvD) sequences applicable to different exploration mission-scenarios and to provide detailed analysis of formation flyings (FF) satellite constellations using highly coupled GNC algorithms, which integrate attitude, navigation and maneuvers. Among those, it is possible to select: Earth, Moon, Mars, Venus, Deimos, Triton and Europa as final destination. Within the LAREDO simulator a launching optimization module is implemented to optimize the launching phase once it has lifted off from the Earth, Moon or Mars surfaces. This optimization is performed such that maximum payload is delivered into orbit. The tool simulates the rocket stages and also evaluates the launch opportunities for optimum launch. The orbit control can be performed by impulsive or continuous maneuvers both on circular and elliptical orbits. The type of maneuvers, their order and schedule is completely configurable by the users. The maneuvers are very accurate even under perturbations. In case the identified safety limits are trespassed, a collision avoidance maneuver (CAM) will be automatically activated, moving the spacecraft to a safety region. The simulator propagates considering various orbit perturbations (aerodynamic, non-sphericity, solar radiation and third body perturbations), communications (both with ground stations and between satellites), and provides full analysis of the trajectory. The GNC modules incorporate sensor models for LIDAR, camera, IMU and RF devices, and both orbit and attitude guidance and control. Furthermore, LAREDO offers powerful 3-D visualization of trajectories and attitude, as well as other important features such as consumed ΔV, ground station contacts, eclipse times, and others.

1.0 NOMENCLATURE

The following concepts and representations are used across this paper:

PAV	=	**P**lanetary **A**scent **V**ehicle. In the launch ascent context, this is the spacecraft that takes off from the planetary surface and is inserted into orbit.
CRF	=	**C**ylindrical **R**eference **F**rame. Y-axis is Perpendicular to the orbital frame, Z-axis is opposite to the Target position vector in the MEE2000 inertial frame, and the X-axis is orthogonal to the Z and Y.
Chaser	=	In the context of satellite formations, the Chaser is any of the satellites in the formation that is not the Target one.
Target	=	In the context of satellite formations, guidance and control schemes use the Target satellite motion as the reference, fixing with respect to it the nominal positions and velocities of the Chaser satellites.

Angel Molina, M.; Prieto, J.; Garcia, G.; Beech, T. (2005) LAREDO: LAunching, REndezvous and DOcking Simulation Tool. In *Integration of Space-Based Assets within Full Spectrum Operations* (pp. 11-1 – 11-16). Meeting Proceedings RTO-MP-SCI-150, Paper 11. Neuilly-sur-Seine, France: RTO. Available from: http://www.rto.nato.int/abstracts.asp.

$\left|\overline{T}\right|^{j}$, $\hat{T}^{j}$ = Thrust modulus and direction of the j[th] rocket stage

$\hat{A}^{j}$, $\overline{B}^{j}$ = Launch ascent optimization variables

$m_{payload}$ = Payload mass

$t^{j}{}_{start}$ = Ignition time of the j[th] rocket stage

t_i = Actual simulation time

$\overline{r}_0, \overline{r}_f$ = Initial and final position vectors

$\overline{r}_{0,ref}, \overline{r}_{f,ref}$ = Initial and final reference position vectors

$\overline{v}_0, \overline{v}_f$ = Initial and final velocity vectors

$\overline{v}_{0,ref}, \overline{v}_{f,ref}$ = Initial and final reference velocity vectors

t_0, t_f = Initial and final orbital maneuvers times

$\Delta\overline{v}_0$, $\Delta\overline{v}_1$ = Initial and final orbital maneuver impulses

$\overline{x}_c, \overline{x}_t$ = Chaser and Target spacecraft state vectors

ω = Orbital angular rate.

a_x, a_y, a_z = Acceleration in the x, y and z-axis.

2.0 ARCHITECTURE

The architecture of the software is represented in the Fig.1. In this figure it can be observed that the Launcher Optimization and the RvD parts run separately.

However, it has to be stressed that the RvD part can import the last state of the Launcher Optimization part to start a RvD simulation. The simulator contemplates orbital and attitude Guidance, Navigation, and Control, as well as sensors and actuators. The outputs from both the Launcher Optimization and RvD parts can be visualized in the Figures Of Merit (FOM) section.

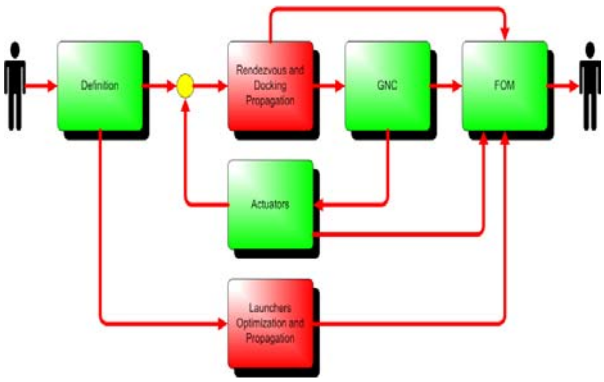

Figure 1 General Simulator Architecture.

3.0 LAUNCHER OPTIMIZATION

LAREDO tool is able to simulate the ascent trajectory of a vehicle from the surfaces of the Earth, the Moon and Mars. The software is able not only to simulate the dynamics of the rocket, but also to optimize the trajectory to reach the desired final orbit with the required relative argument of latitude. The trajectory propagation considers the atmospheric and the non-sphericity perturbations, as well as the rocket stages and the fuel mass losses.

The rocket has two configurable stages, where the structural masses, propellant masses, and other parameters can be defined. A coast arc can be inserted between the two stages, where the duration is an optimization variable.

3.1 Rocket dynamics modeling

One of the key parameters when modeling a rocket ascent trajectory is the rocket thrust vector. The thrust vector is defined by a modulus and a direction, given by Eq.(1)[1] and Eq.(2) respectively:

$$\left|\overline{T}\right|^{j} = \frac{dm^{j}}{dt} \cdot g_0 \cdot I_{sp}{}^{j} - P_a \cdot A_t{}^{j} \tag{1}$$

where $\frac{dm^{j}}{dt}$ is the jth stage flow rate, g_0 is the sea level gravity, $I_{sp}{}^{j}$ is the specific impulse in vacuum of the jth stage, $A_t{}^{j}$ is the effective exit area of the j^{th} stage nozzle and P_a is the atmospheric pressure.

$$\hat{T}^{j} = \frac{\hat{A}^{j} + \overline{B}^{j}\left(t_i - t^{j}{}_{start}\right)}{\left|\hat{A}^{j} + \overline{B}^{j}\left(t_i - t^{j}{}_{start}\right)\right|} \tag{2}$$

Eq.(2) which expresses the thrust direction, is also known as the " bi-linear tangent law[2] ". Both $\hat{A}^{j}$ and $\vec{B}^{j}$ vectors are optimized when the second stage of the rocket is burning. During the first stage, the rocket direction is considered to remain constant and thus $\overline{B}^{j} = 0$.

The mass of the burning stage is continuously reduced. The loss of mass will be determined by the rocket flow rate by the following expression:

$$m_{ti}{}^{j} = m_{ti-1}{}^{j} - \frac{dm^{j}}{dt} \tag{3}$$

where $\frac{dm^{j}}{dt}$ is the j^{th} stage flow rate, $m_{ti-1}{}^{j}$ is the j^{th} stage mass at the previous simulation step time t_{i-1} and $m_{ti}{}^{j}$ is the j^{th} stage mass at the time t_i. Note that $\frac{dm^{j}}{dt}$ will be zero if the stage is not burning.

If "*n*" is considered as the total number of stages, the mass of the rocket is defined as:

$$m_T = \sum_{j=1}^{n} m_{ti}{}^{j} + m_{payload} \tag{4}$$

The heat transfer rate is one of the main factors to be considered when launching a vehicle. If the heat transfer exceeds certain maximum allowable value, the rocket and/or its cargo and crew can suffer severe damage or even disintegrate. The heat transfer equation is given by the following equation (Sutton & Graves correlation[3]):

$$\dot{q} = C_1 \times \left(\frac{\rho_\infty}{R_n} \right)^{C_2} \times \left(\frac{U_\infty}{1000} \right)^{C_3} \tag{5}$$

where ρ_∞ is the atmospheric density, U_∞ is the velocity in the corotating reference frame (reference frame fix to the planet), R_n is the rocket nose radius and C_1, C_2 and C_3 are coefficients. For the case of Mars, the C_1, C_2 and C_3 coefficients are 18.9, 0.5 and 3 respectively.

3.2 Optimization

The goal of most rockets is to achieve the required orbit with the maximum injected payload mass. The software tries therefore, to minimize the following cost function:

$$J = -m_{payload} \tag{6}$$

The trajectory of the rocket is propagated using a full Cowell propagator with atmospheric and non-sphericity perturbations acting upon the rocket, and also taking into account the different rocket phases.

The cost function expressed in Eq.(6) is minimized while satisfying a few equality and inequality constraints which ensure that the trajectory is feasible.

- Equality constraints: The equality constraints are defined by the desired final orbital parameters. This way, the orbital parameters propagated for the launch ascent with the Cowell full- propagator must be equal to the final desired orbit parameters.
- Inequality constraints: Three inequality constraints are imposed. The first one is to make sure that the height achieved when the launch optimization is performed is greater than the planet radius. The second one is to ensure that the heat rate will never exceed a maximum value. Finally, the third one is to ensure that the PAV spacecraft is placed with respect to the other satellite within a certain relative argument of latitude limits.

4.0 ORBITAL MANOEUVRES FOR RVD

The use of more than one spacecraft to perform a mission makes the RvD maneuvers more and more important. These maneuvers are usually performed in circular orbits, however, there is rising interest in RvD in elliptical orbits. LAREDO is able to provide a very accurate analysis in both circular and elliptical orbits, estimating the ΔV, the fuel consumption and other important features.

RvD is a very risky part of the mission as any failure can cause the satellites damage. The tool simulates a Collision Avoidance Maneuvers that is activated if the satellites trespass certain safety limits. These limits are configurable, and of course dependent on the relative satellite distances. When the satellites are far away from each other the safety zone is assumed to be a sphere, whereas if they are close the safety zone is modeled as a cone (approaching corridor). If the satellite performs some of the programmed maneuvers crossing any of these boundaries, the required orbital and attitude maneuvers are automatically activated.

4.1 Corrective maneuvers for circular orbits

The most important problem in RvD comes out from the high maneuvering precision required. This severe requirement during the RvD implies reducing as much as possible the perturbations effects acting upon the spacecraft. In order to counteract these perturbing effects, the spacecraft will perform corrective maneuvers during the whole trajectory. An increase in the number of corrective maneuvers generally implies a reduction in the final trajectory error, but with a penalty on the consumed fuel (see Fig. 2).

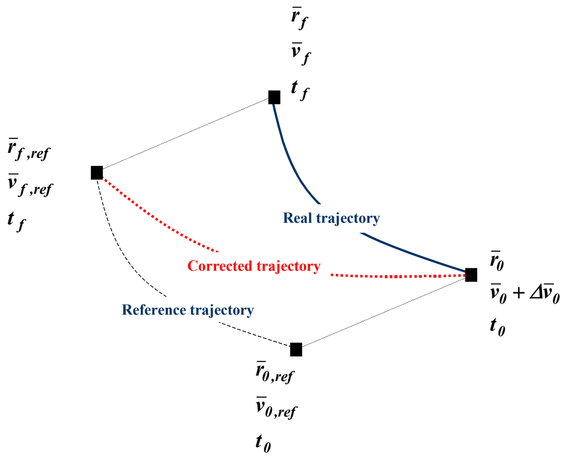

Figure 2. Corrective maneuvers.

The method implemented to compute the corrective maneuvers uses the transition matrix Φ. Mathematically the variation of the satellite state can be expressed as:

$$\Phi(t_f, t_0) = \begin{bmatrix} \Phi_{xx}(t_f, t_0) & \Phi_{xv}(t_f, t_0) \\ \Phi_{vx}(t_f, t_0) & \Phi_{vv}(t_f, t_0) \end{bmatrix} \tag{7}$$

The final position of the controlled satellite using the transition matrix can be formulated as:

$$\bar{r}_f = \bar{r}_{f,ref} + \delta\bar{r}_f = \bar{r}_{f,ref} + \Phi_{xx}(t_f, t_0)(\bar{r}_0 - \bar{r}_{0,ref}) + \Phi_{xv}(t_f, t_0)(\bar{v}_0 + \Delta\bar{v}_0 - \bar{v}_{0,ref}) \tag{8}$$

Imposing the constraint that $\bar{r}_f = \bar{r}_{f,ref}$, the initial the $\Delta\bar{v}_0$ can be obtained as:

$$\Delta\bar{v}_0 = -\Phi_{xv}^{-1}\Phi_{xx}(\bar{r}_0 - \bar{r}_{0,ref}) - \bar{v}_0 + \bar{v}_{0,ref} \tag{9}$$

In a similar way, the velocity can be expressed as:

$$\bar{v}_f = \bar{v}_{f,ref} + \Phi_{vx}(t_f, t_0)(\bar{r}_0 - \bar{r}_{0,ref}) + \Phi_{vv}(t_f, t_0)(\bar{v}_0 + \Delta\bar{v}_0 - \bar{v}_{0,ref}) \tag{10}$$

With the constrain that $\bar{v}_f + \Delta\bar{v}_1 = \bar{v}_{f,ref}$, the final impulse $\Delta\bar{v}_1$ is obtained by the following equation:

$$\Delta\bar{v}_1 = -\Phi_{vx}(\bar{r}_0 - \bar{r}_{0,ref}) - \Phi_{vv}(\bar{v}_0 + \Delta\bar{v}_0 - \bar{v}_{0,ref}) \tag{11}$$

4.2 Circular maneuvers guidance and control

So far, all of the RvD maneuvers have been performed in circular or near circular orbits due to the fuel consumption and maneuver simplicity. The formulation of these maneuvers is generally derived from the Hill-Equations, which are the simplified version of the Clohessy-Wiltshire equation for small eccentricities and relative distances, as shown in Eq. (12).

$$
\begin{aligned}
\ddot{x} &= 3\omega^2 x + 2\omega\,\dot{y} + a_x \\
\ddot{y} &= -2\omega\dot{x} + a_y \\
\ddot{z} &= -\omega^2 z + a_z
\end{aligned}
\tag{12}
$$

In case of circular orbits, the LAREDO tool orbital maneuvers are all based on the Clohessy-Wiltshire equations[4], where the set of maneuvers implemented are:

- Impulsive hohmann transfer: The aim of this maneuver is to modify the relative height of the two satellites using two impulses in the direction of the Target velocity. The maneuver duration is equal to half an orbital period (see Fig. 3).

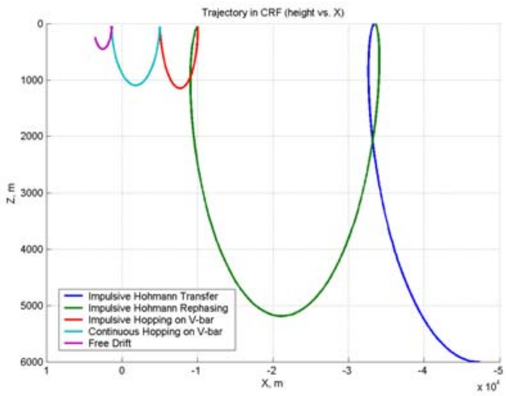

Figure 3. Impulse Hohmann transfer, Impulse Hohmann Rephasing, Impulse Hopping on V-bar and Continuous Hopping on V-bar examples.

- Impulsive hohmann rephasing: This maneuver modifies the relative along track distance of the two satellites using two impulses in the direction of the Target velocity. The difference with the Impulsive Hopping on V-bar is that the maneuver duration is a multiple of the orbital period (see Fig. 3).
- Impulsive hopping on V-bar: Maneuver to modify the relative along track distance of the two satellites using two impulses in the direction of the Target velocity. The maneuver duration is equal to half-orbital period (see Fig.3).
- Continuous hopping on V-bar: The aim of this maneuver is to modify the relative along track shift between the two satellites using a constant acceleration in the direction normal to the Target velocity. The maneuver duration is equal to an integer multiple of the orbital period (see Fig. 3).
- Continuous tangential transfer: The aim of this maneuver is to modify the relative height of the two satellites using a constant acceleration in the direction of the Target velocity. The maneuver duration is equal to an integer multiple of the orbital period (see Fig. 4).

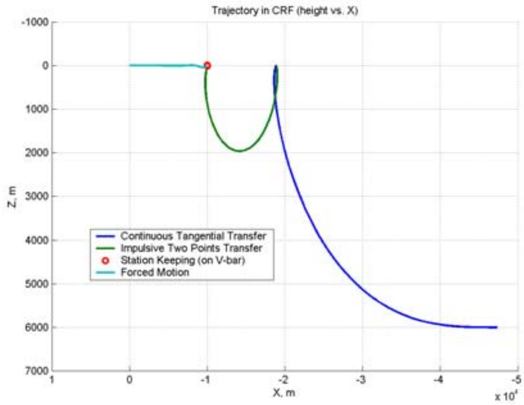

Figure 4. Continuous Tangential Transfer, Impulsive Two Points Transfer, Station Keeping and Forced Motion examples.

- Impulsive two point transfer: The aim of this maneuver is to take the Chaser satellite from an initial point to a generic final point, with generic final velocity and with generic maneuver duration (see Fig. *4*).
- Station keeping. The aim of this maneuver is to keep the relative position between the Target and the Chaser as close as possible to a static reference position in CRF, using a LQ algorithm. The user specifies the static reference position and the duration of this maneuver (see Fig. 4).
- Forced motion. The aim of this maneuver is to keep the relative position between the Target and the Chaser as close as possible to a dynamic reference position in CRF, using a LQ algorithm. This dynamic reference position moves at a constant velocity in CRF from the current position of the Chaser to a user defined final position. The duration of this maneuver, which defines the constant velocity of the dynamic reference position, is also specified by the user. The trajectory of the reference position is a straight line in CRF (see Fig. 4).

4.3 Elliptical maneuvers guidance and control

The Clohessy-Wiltshire equations described in the above section cannot be applied when the orbits have a significant eccentricity. For these orbits a different mathematical approach, based on a transition matrix, is applied.

The formulation of the elliptical maneuvers provided below is described in terms of time, but the conversion to distance is straightforward. The elliptical maneuvers will place the Chaser spacecraft in the same orbit as the Target one but with a certain time delay, t_{delay}, with respect to it. This means that if the maneuver ends at a time t_n the following relation must be satisfied:

$$\delta\bar{x}_f = \bar{x}_t(t_n) - \bar{x}_t(t_n - t_{delay}) = \bar{x}_c(t_n) - \bar{x}_t(t_n) \qquad (13)$$

The relative position of the Chaser with respect to the Target at the beginning of the maneuver can be expressed in the following way:

$$\delta\bar{x}_0 = \bar{x}_c(t_0) - \bar{x}_t(t_0) \qquad (14)$$

The Chaser position can be formulated as a linearization with respect to the Target spacecraft as follows:

$$\delta\bar{x}(t_n) = \Phi(t_n, t_0)\delta\bar{x}_0 + \begin{bmatrix} \Phi_{xv,0} & \Phi_{xv,1} & \cdots & \Phi_{xv,n-1} \\ \Phi_{vv,0} & \Phi_{vv,1} & \cdots & \Phi_{vv,n-1} \end{bmatrix} \begin{bmatrix} \Delta\bar{v}_0 \\ \Delta\bar{v}_1 \\ \cdots \\ \Delta\bar{v}_{n-1} \end{bmatrix} \tag{15}$$

Please notice that $\Delta\bar{v}_n$ is omitted since the position of the satellite at the end of the set of maneuvers is not modified. Using the Lagrange operators $\bar{\lambda}$, the function to be minimized to perform the maneuvers in an optimal way is:

$$f = Trace(\Delta\bar{v}^T \Delta\bar{v}) + Trace(\bar{\lambda}^T \left(\delta\bar{x}_f - \Phi(t_n, t_0)\delta\bar{x}_0 - \begin{bmatrix} \Phi_{xv,0} & \Phi_{xv,1} & \cdots & \Phi_{xv,n-1} \\ \Phi_{vv,0} & \Phi_{vv,1} & \cdots & \Phi_{vv,n-1} \end{bmatrix} \Delta\bar{v} \right)) \tag{16}$$

Minimizing the last equation with respect to $\Delta\bar{v}$ and $\bar{\lambda}$:

$$\frac{\partial f}{\partial(\Delta\bar{v})} = 0 = 2\Delta\bar{v} - \begin{bmatrix} \Phi_{xv,0} & \Phi_{xv,1} & \cdots & \Phi_{xv,n-1} \\ \Phi_{vv,0} & \Phi_{vv,1} & \cdots & \Phi_{vv,n-1} \end{bmatrix}^T \bar{\lambda} \tag{17}$$

$$\frac{\partial f}{\partial\bar{\lambda}} = 0 = \delta\bar{x}_f - \Phi(t_n, t_0)\delta\bar{x}_0 - \begin{bmatrix} \Phi_{xv,0} & \Phi_{xv,1} & \cdots & \Phi_{xv,n-1} \\ \Phi_{vv,0} & \Phi_{vv,1} & \cdots & \Phi_{vv,n-1} \end{bmatrix} \Delta\bar{v} \tag{18}$$

Putting these last equations in a matrix form, and solving to obtain the multiple "*n*" $\Delta\bar{v}$ necessary to achieve the required position at time t_n, the following equation arises:

$$\begin{bmatrix} \Delta\bar{v} \\ \bar{\lambda} \end{bmatrix} = \begin{bmatrix} 2I_{3x3} & 0_{3x3} & \cdots & 0_{3x3} & -\Phi_{xv,0} & -\Phi_{vv,0} \\ 0_{3x3} & 2I_{3x3} & \cdots & 0_{3x3} & -\Phi_{xv,1} & -\Phi_{vv,1} \\ \cdots & \cdots & \cdots & \cdots & \cdots & \cdots \\ 0_{3x3} & 0_{3x3} & \cdots & 2I_{3x3} & -\Phi_{xv,n-1} & -\Phi_{vv,n-1} \\ \Phi_{xv,0} & \Phi_{xv,1} & \cdots & \Phi_{xv,n-1} & 0_{3x3} & 0_{3x3} \\ \Phi_{vv,0} & \Phi_{vv,1} & \cdots & \Phi_{vv,n-1} & 0_{3x3} & 0_{3x3} \end{bmatrix}^{-1} \begin{bmatrix} 0_{3nx1} \\ \delta\bar{x}_f - \Phi(t_n, t_0)\delta\bar{x}_0 \end{bmatrix} \tag{19}$$

4.4 Orbital navigation

The orbital navigation is carried out using measurements from ground station, but also with the modeled satellite sensors, accelerometers, IMU, RF, camera, and lidar.

The navigation solution is obtained using a Kalman filter formulation.

5.0 ATTITUDE MANOEUVRES

The Attitude plays a very important role when the final part of the RvD, the docking, is carried out. The attitude needs to be accurate enough to allow the satellites to dock. The spacecraft must be able to navigate either when only inertial sensors (e.g. Star tracker) are available or when relative sensors are available. The guidance in these two cases is completely different.

5.1 Attitude maneuver guidance

The satellite implements a few sets of guidance profiles to perform the required inertial and relative maneuvers. The type of guidance will depend upon the available sensors and upon the satellite relative distances. Within LAREDO, the following guidance profiles are considered:

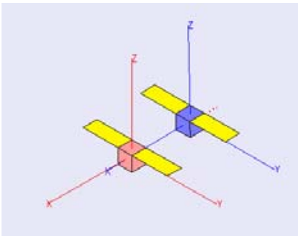

Figure 5. Fixed Relative.

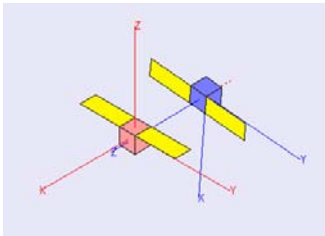

Figure 6. Target Pointing.

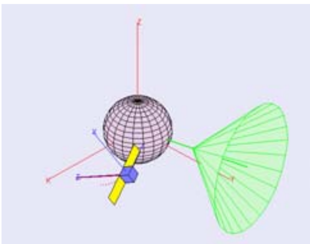

Figure 7. Sun Pointing.

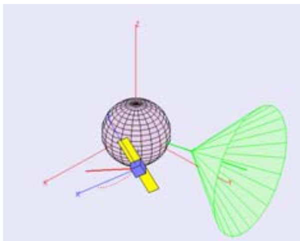

Figure 8. Nadir Pointing.

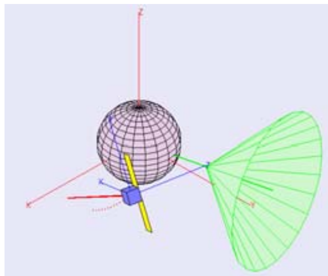

Figure 9. GS pointing.

- Fixed relative attitude. Guidance to maintain the Chaser in a constant orientation with respect to the Target (see Fig. 5).
- Target pointing (see Fig. 6). Guidance to point towards the Target satellite. Very useful if the Chaser satellite is desired to point always towards a sensor placed on the Target spacecraft (e.g. lasers or cameras).
- Sun pointing. Guidance to point towards the Sun. Useful mode if batteries are desired to be charged or if Collision avoidance maneuver is activated (see Fig. 7).
- Nadir pointing. Guidance to point towards the Earth Center (see Fig 8).
- GS pointing. Guidance to point towards the Ground Station (GS). This type of guidance is required if the antenna is not directional when uploading or downloading information from GS (see Fig. 9).

5.2 Control

The control used in attitude is a Proportional Derivative Controller (PD) [5]. This type of attitude control is simple, but its performance is very good. The equation of the implemented controller is as follows:

$$
\begin{aligned}
T_{Cx} &= 2K_x\, q_{1E} q_{4E} + K_{xd}\, w_{E_x} \\
T_{Cy} &= 2K_y\, q_{2E} q_{4E} + K_{yd}\, w_{E_y} \\
T_{Cz} &= 2K_z\, q_{3E} q_{4E} + K_{zd}\, w_{E_z}
\end{aligned} \tag{20}
$$

Where K_x, K_y, K_z are the proportional gains of the x, y, z axis; K_{xd}, K_{yd}, K_{zd} are the derivative gains of the x, y, z axis; q_{1E}, q_{2E} and q_{3E} are the vectorial part of the quaternion error (error between the desired and actual attitude); q_{4E} is the scalar part of the quaternion error; w_{E_x}, w_{E_y} and w_{E_z} are the x, y and z components of the angular velocity error.

5.3 Attitude navigation

Attitude navigation is carried out autonomously since no measurements from the ground are required. The sensors modeled are Star trackers, RF system, Camera system, Gyroscopes and Sun sensors. The sensor noises are filtered using a Kalman Filter[6] to obtain a clear signal.

6.0 TEST CASES

This paper reports the simulation results obtained for a Mars Sample Return mission scenario[7]. The simulation of the this mission scenario includes:

- Launch ascent trajectory optimization of the PAV to insert it in a circular orbit.
- A change of plane maneuver to place both satellites in the same orbital plane.
- RvD phase of the satellites in circular orbits.

Further more, the paper also shows the results of a RvD in elliptical orbits to illustrate the tool's performances. The attitude guidance and control is already presented in the Attitude Guidance Maneuvers section (Fig. 5, Fig. 6, Fig. 7, Fig 8 and Fig. 9). These figures are obtained directly from the LAREDO, where all the sensors, and actuators were activated.

6.1 Mars sample return launch ascent results

The launch is performed from 45 degrees latitude and 0 degrees longitude on the Mars surface. In this simulation, the atmospheric and non-sphericity perturbations are taken into account. The launch ascent vehicle characteristics and the desired final orbit is provided in the tables shown bellow (Table 1 and Table 2). Note that the Target spacecraft is also simulated, and its initial argument of latitude is 78.70 deg.

Table 1. Launch ascent Final orbit

Semi-major axis (km)	3897.514
Eccentricity	0
Inclination (deg.)	45
Right ascention node (deg)	265.811
Argument of perigee (deg)	253.629
Max. relative Argument of latitude (deg)	4
Min. relative Argument of latitude (deg)	2

Table 2 Launch ascent Rocket characteristics

	First Stage	Coast Arc	Third Stage
Reference diameter (m)	0.563	0.563	0.563
Nominal burn time (s)	197.566	-	672.197
Propellant mass (kg)	360	-	116
Gross mass (kg)	456.6	-	213.8
Specific Impulse (s)	306	-	325

Fig. 10 and Fig. 11 show that the circular orbit of 500 km height is achieved. The atmospheric pressure losses can be clearly observed in the thrust profile of Fig. 12, where the thrust increases when higher altitudes are reached. The rocket loss of mass is represented in the Fig. 13. This figure clearly shows that the mass is reduced when the first stage of the rocket is deployed. With the optimized trajectory, the rocket can deliver **10.32 kg** of payload mass, deploying it at a relative argument of latitude with respect to the Target of 2 degrees.

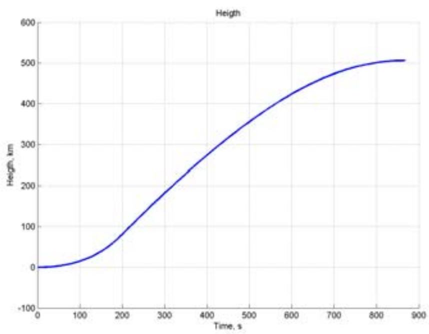

Figure 10. Launch ascent height.

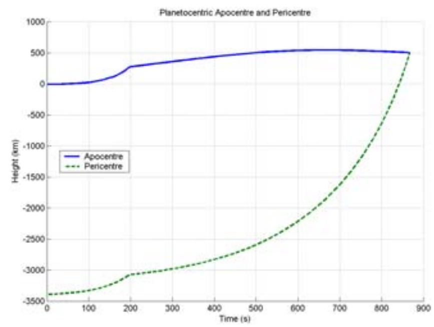

Figure 11. Launch ascent apocentre and pericentre.

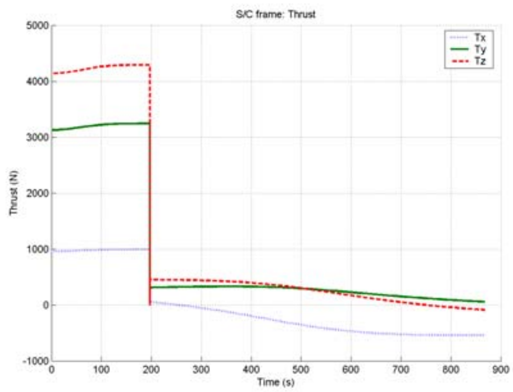

Figure 12. Launch ascent Thrust.

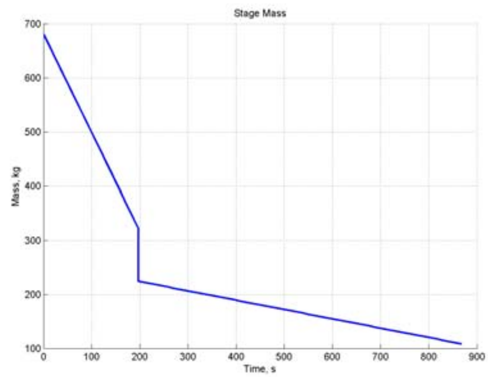

Figure 13. Launch ascent Mass.

6.2 Mars sample return change of plane

Generally, the launcher does not deploy the PAV exactly in the desired orbit. The PAV is delivered into a circular orbit that is two degrees different in inclination and Right ascension with respect the Target Orbit, and 20 km below (see Table 3). Therefore, a change of plane maneuver is performed.

Table 3. Change of Plane orbital elements

	Initial orbit before change of plane	Final orbit after change of plane
Semi-major axis (km)	3877.515	3897.514
Eccentricity	0	0
Inclination (deg)	45	47
Right ascension node (deg)	265.811	268.811
Argument of perigee (deg)	253.629	253.629
True anomaly (deg)	121.035	123.035

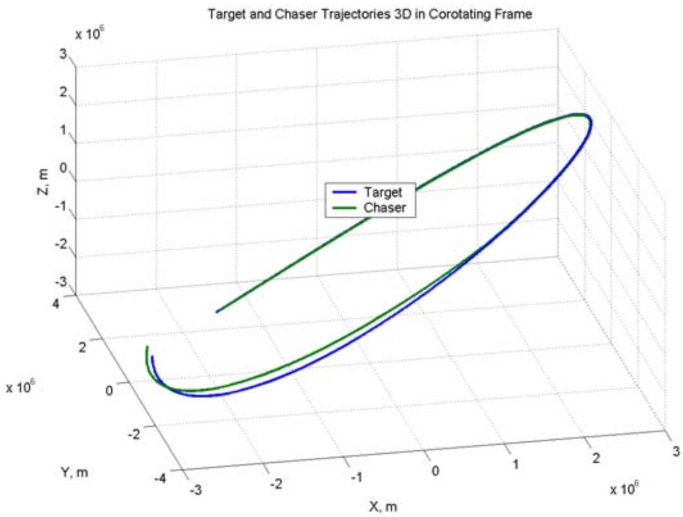

Figure 14. Change of Plane Maneuver.

6.3 Mars sample return RvD in circular orbit

Once the change of plane is performed, the RvD can be done nicely in the coplanar circular orbits. The satellites starts from the change of plane maneuver at a relative height of 20 km and –60 km of along track, performing the required maneuvers (also applying corrective maneuvers) until the docking is achieved (see Fig. 15 and Fig. 16). The maneuver Timeline is as follows:

- Impulsive Hohmann Transfer to place the satellites at the same height
- Impulsive two point transfer to place the Chaser –2.7 km in along track
- Station keeping at –2.7 km along track for 1000 seconds
- Impulsive Hopping on V-bar to place the Chaser –300 km in along track
- Station Keeping at –300 km along track for 1000 seconds
- Forced Motion for 2000 seconds to place the Chaser –100 km
- Forced Motion for 2000 seconds until final Docking.

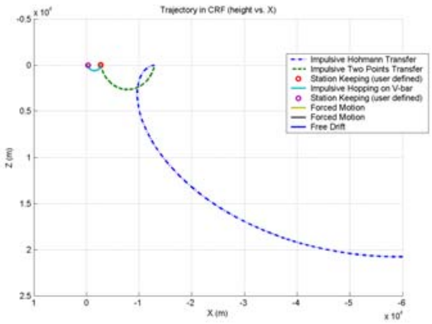

Figure 15. Rendezvous and Docking trajectory.

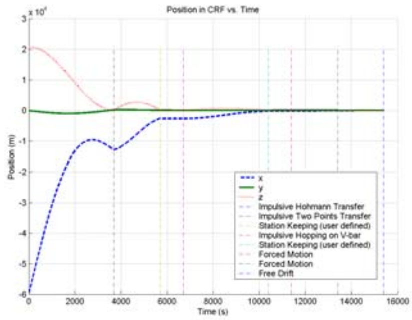

Figure 16. Rendezvous and Docking position.

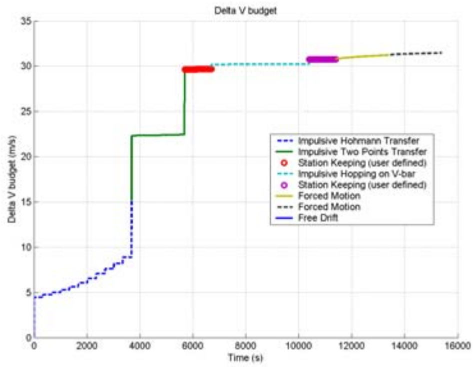

Figure 17. Rendezvous ΔV.

6.4 RvD in elliptical orbit

This test case is presented to show the results obtained using the RvD algorithms implemented for elliptical orbits. The initial satellite states used for this simulation are provided in the table shown below (see Table 4):

Table 4. Elliptical RvD orbital elements

	Chaser Orbital Elements	Target Orbital Elements
Semi-major axis (km)	9257.165	9306.165
Eccentricity	0.607	0.607
Inclination (deg)	45	45
Right ascension node (deg)	232.934	232.934
Argument of perigee (deg)	345.076	345.076
True anomaly (deg)	-0.233	-0.033

The initial relative position of the Chaser with respect to the Target spacecraft is a difference of 20 km in height and nearly 13 km in along track. It can be observed from the table above (Table 4) that both the Chaser and Target spacecrafts are in high elliptical orbits. The Elliptical RvD maneuvers Timeline is as follow (see Fig. 18):

- Elliptical Transfer to place the satellites at the same height

- Elliptical Re-phasing in Time. Chaser is placed at 1-second delay with respect to the Target.
- Elliptical Hopping on V-bar to place the Chaser – 5km in along track.
- Elliptical Station Keeping at –5km along track.
- Elliptical two points transfer to place the Chaser -1 km in along track.
- Elliptical Forced Motion until Docking is performed.

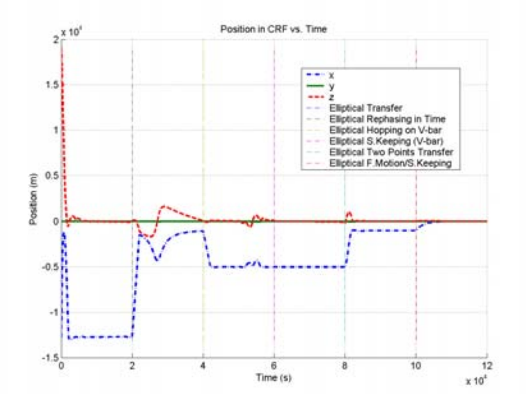

Figure 18. Elliptical Rendezvous.

7.0 CONCLUSION

LAREDO is a very sophisticated tool to perform Launch ascent trajectory optimizations and RvD for elliptical and circular orbits. The launch ascent optimizer provides a very accurate way to obtain an optimized trajectory when launching from the Moon, Mars and the Earth. For the RvD phase, Venus, Deimos, Europa and Triton can also be used as central bodies in addition to the Earth, the Moon and Mars. The RvD in circular orbits uses the Clohessy-Wiltshire equations with the aid of corrective maneuvers. The RvD in elliptical orbits is based on the transition matrix to cope with the non-linearity of the problem.

The results presented in this document show the good performances of the launch ascent trajectory for a two-stage rocket. The simulations presented for RvD are performed with all perturbations acting upon the satellite and using sensors and actuators. Both the Orbital and Attitude algorithms implemented in LAREDO are extensively proved and documented.

Future work will be done to prove the orbital and attitude algorithms in a test bench to see their performances in a more realistic way.

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BIOGRAPHY

Miguel Angel Molina is a Senior Aeronautical Engineer currently responsible of the Flight Engineering Business Unit at GMV S.A. This Business Unit is mostly dedicated to Flight Dynamics and Mission Analysis activities.