A STUDY ABOUT DYNAMIC VEHICLE REENTRY INTO EARTH ATMOSPHERE

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Abstract

Reentry missions request solution of critical points like: initial values of orbital injection, reentry window, coordinates of landing area, power consumption for reentry injection and others. This work introduces a numerical study for mission of scientific vehicle reentry into Earth atmosphere (ballistic trajectory). The procedure analyzes numerically the vehicle trajectory taking into account those critical points, from orbital injection time up to opening time of parachute, providing the best (in some sense) reentry window using iterative methods. The dynamic models are fully referenced to Inertial system (not usual) applicable to orbit and reentry propagation, including disturbing forces due to 6^{th} zonal and tesseral harmonics of gravitational field, aerodynamic forces and trust force[1]. The vehicle is supposed to have capability of stable aerodynamic flight (longitudinal axis aligned with velocity relative to atmosphere). The simulation of mean trajectory (ballistic type), has presented pointing errors over defined landing point lower than 10 km without trajectory correction maneuvers.

***Key words**:* Dynamic Model, Iterative Methods, Landing Point Error.

Introduction

The motivation for this study came from the AMSR project (Mission Analysis for Recoverable Satellites) at INPE. The question was: How to handle the reentry problems under point of view of orbital mechanics using rectangular coordinates and Keplerian Elements?

Usually, studies about reentry trajectory into Earth atmosphere use classical coordinate system $(\mathbf{r},\boldsymbol{\phi},\boldsymbol{\delta},\mathbf{V},\boldsymbol{\gamma},\boldsymbol{\psi})$[2] to analyze the reentry dynamics. A reentry trajectory is a descend trajectory performed by a vehicle between 95 and 15 km of altitude. This descend trajectory may be commanded or naturally.

This study analyses reentry problem since orbital phase up-to opening parachute time using the same differential equations relative to rectangular coordinates of Inertial System.

Usually, the trajectory is subjected to constraints: orbital elements of injection, landing area coordinates and size, thermal constraints, reentry conditions (flight path angle and velocity), atmospheric model, aerodynamic characteristics of the vehicle, and propulsive system capability. The trajectory to be performed must carrier the vehicle as near as possible to defined landing area.

When a landing area is requested, it is usual to query: a) What is the best (in some sense) reentry's flight path angle b) How long the propulsive sub-system must to keep activated? c) When it starts or stops? d) Once the processes are iterative: what is the criterion to select or stop the simulation process without to break the constraints?

To answer this questions was studied a criterion using the classical coordinate named flight path angle. After another criterion based on altitude of instantaneous perigee.

Flight Path Angle Criterion

Many approaches can be followed[3,4] by using classic variables. The flight path angle is related mutually with range, maximal temperature, propulsive power and consumption. The expression to calculate flight path angle, γ, by using rectangular coordinates of Inertial system is:

$$\gamma = \sin^{-1}(\hat{R}.\hat{V}_E) \qquad (1)$$

where:

$\hat{R}$ is the position versor, and

$\hat{V}_E$ is velocity versor of the vehicle relative to Earth.

Once the vehicle is in orbital level, to know γ at $\mathbf{H_r}$ =95 km (Fig. 1) is necessary to propagate trajectory from time T_0 to T_k with propulsive sub-system activated. For each step above, a second propagation must be done from T_k up to altitude $\mathbf{H_r}$ with trust force disabled. γ_k is computed by Equation 1 and compared with refereed value. This provides an intense iterative process. This criterion will be named FPA (Flight Path Angle) and is clarified by Fig. 1.

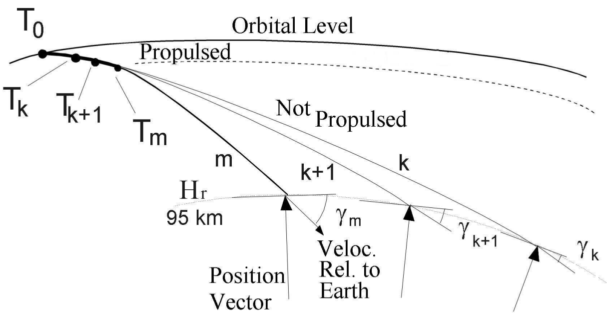

Figure 1 - FPA Criterion

Perigee Criterion

How to avoid this dual iteration of FPA criterion? What measurement is available at T_k under point of view of orbital Mechanics? Vinh[5] uses the perigee to get the entry corridor condition. The second iteration is avoided by checking the altitude of instantaneous perigee at T_k . If it is null or negative (Fig. 2) then the vehicle's trajectory will intercept the Earth's surface. These conditions are enough for a direct entry and, of course, it is a flag to stop the propulsive phase. The landing area shall be between vehicle position where the propulsion sub-system start and the position where the trajectory crosses the Earth surface.

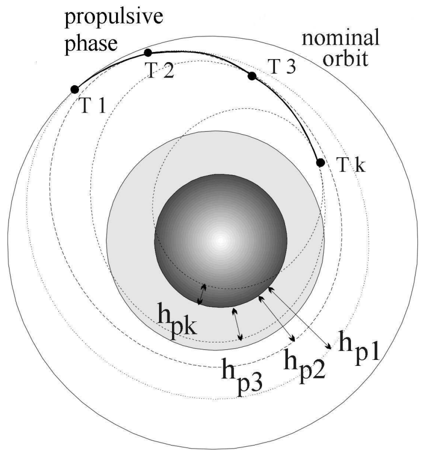

Figure 2 - Perigee Criterion

The altitude of perigee is computed converting the state vector to keplerian elements, redefining the mean anomaly to zero, and converting the redefined keplerian elements to state vector. (See Escobal, 1966 for details)

Reentry Window

Next step is to identify where (or when) to start the orbital maneuver to inject the vehicle into reentry trajectory and it reaches the landing area. This is the reentry window. Dynamically, the vehicle trajectory is subjected to environmental forces due to atmosphere and gravitational field, both changing with altitude, latitude, and longitude. The Earth is rotating, and all orbital elements and their rates are changing. By using a non-linear model is not more possible to apply analytical formulation and, again, it is necessary an iterative process. So, a new question coming up: What iterate to get the reentry window?

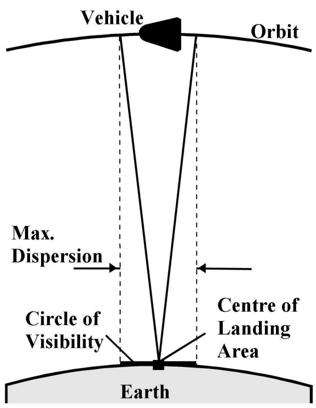

Figure 3 - Projection of Visibility Cone.

If landing area is a constraint then the orbit must be selected conveniently. By assuming no lateral maneuvers during reentry, what is happen if it starts the reentry by propulsive system when the vehicle is in orbit and over the landing area? So, it is assumed that the vehicle will cross over landing area some times at final days of the mission. The reentry window may exist some time before this passage (Fig. 3).

Let $\vec{R}_{pp}$ the orbital position vector of vehicle as near as possible from the position vector of the center of landing area. By commanding the reentry at $\vec{R}_{pp}$ and applying any criterion explained before, the vehicle will reentry and touches down at some place ahead ($\vec{R}_a$). Let β the angle between them. There will be an orbital position vector $\vec{R}_k$, before $\vec{R}_{pp}$, whose angle between them is near of β.

The iterative process for reentry window

i) By commanding reentry from $\vec{R}_k$ the vehicle will land on a new position $\vec{R}_{a+1}$. ii) β_1 is available and it is the angle between $\vec{R}_k$ and $\vec{R}_{k+1}$, that can be obtained from orbit data file and the looping is repeated. The convergence of β_i occurs at (maximum) 5th iteration. The iteration provides the time to start propulsive system and the reentry is completed (see Fig. 3)

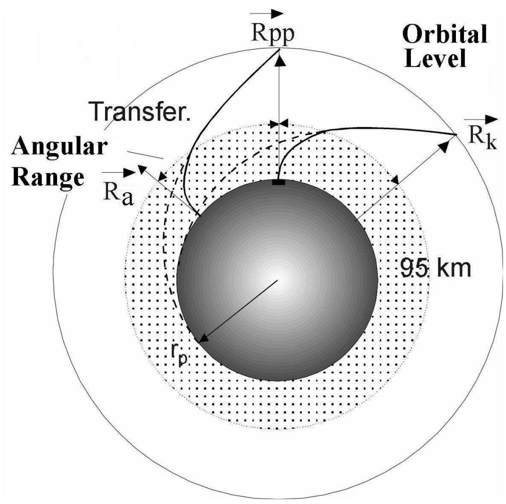

Figure 3 - Reentry Window.

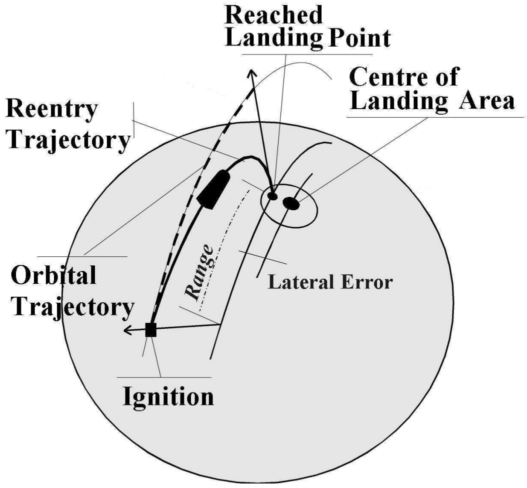

Figure 3 - Illustration of lateral error, reentry trajectory and landing area.

Dynamic Equations:

Differential Equations:

$$\dot{\vec{R}}_I = \vec{V}_I \ , \ \dot{\vec{V}}_I = \vec{A}_G + \frac{\vec{F}_D}{m} + \vec{A}_R \qquad (2)$$

where:

$$\vec{V}_r = \vec{V}_I - \vec{\Theta} \times \vec{R} \, , \hat{V}_r = \frac{\vec{V}_r}{|\vec{V}_r|} \, , \ \vec{\Theta} = \dot{\theta}\hat{Z} \qquad (3)$$

and

$\dot{\theta} =$ 7.292115854682 x 10^{-5} rad/s

It is assumed that atmosphere is locked over Earth surface. So it is valid to assume that velocity of vehicle relative to Earth is equal to that velocity of vehicle relative to Atmosphere. The atmosphere model follows US Standard Atmosphere 1976 pattern, performing the necessary interpolation.

Drag force:

$$\vec{F}_D = -\frac{1}{2} C_D \rho A V_r^2 \hat{V}_r \ , \quad \hat{H} = \frac{\vec{H}}{|\vec{H}|} \, , \qquad (4)$$

$$\vec{H} = \vec{R} \times \vec{V}_r$$

Trust force:

$$\vec{A}_R = \frac{T}{m}\hat{T}\ ,\ \hat{T} \Leftarrow \frac{(\vec{R}_I \times \vec{V}_I)\times \vec{R}_I}{\left|\vec{R}_I\right|^2 \left|\vec{V}_I\right|} \qquad (5)$$

Gravitational Field

The vector $\vec{A}_G$ is provided by GEM-10 model, obtained from the field model:

$$U = \frac{\mu}{r}\left\{1 + \sum_{n=2}^{k}\left[\left(\frac{R_e}{r}\right)^n C_n P_n(sen\delta) + \right.\right.$$

$$\left.\left. + \sum_{m=1}^{n}\left(\frac{R_e}{r}\right)^n P_{nm}(sen\delta)\left(C_{nm}\cos(m\lambda) + S_{nm}sen(m\lambda)\right)\right]\right\} \qquad (6)$$

Harmonics Order/Degree of Gravitational Field

Although the drag is the main disturbing force at reentry phase, the gravitational field has significant influence over precision of reached landing point. To evaluate this influence, under point of view of harmonic order, it was done a set of simulation. Using an available model GEM-10, the results are condensed into two tables. Table 1 shows the coordinates of vehicle at 95 km propagated from the same initial condition. Note that, by taking spherical Earth (harmonics 0/0) the final coordinates are different, of 4 degrees in longitude and 0.3 degrees of latitude at 95 km from those obtained by using non-spherical models. This difference represents a error of 440 km. Of course, this is a position error not yet in reentry phase, but it compromises the mission.

Table 1 - Coordinates of vehicle at 95 km.

HARMONIC DEG./ORDER		COORDINATES OF VEHICLE AT 95KM OF ALTITUDE		
Zonal	Tesseral	TIME (s)	LONG. (DEG)	LAT. (DEG)
0	0	1593,3	60,57220°	-3,796299°
2	0	1531,1	56,54872°	-4,019510°
3	0	1529,8	56,54646°	-4,019905°
4	0	1529,7	56,53690°	-4,020375°
5	0	1529,7	56,53742°	-4,020310°
6	0	1529,6	56,53335°	-4,020493°
6	6	1529,9	56,55200°	-4,019212°
30	30	1530,0	56,55685°	-4,018638°

Table 2 shows the trajectory propagation from positions of table 1 up to 4 km of altitude. Table 2 shows the influence of gravitational field over the reentry phase. Since Table 1 presents a difference of 4 degree between (0/0) and others harmonics in longitude at 95 km, how to explain the difference of 5 degrees at 4 km between (0,0) and others harmonics from Table 2? Continuing the simulation of harmonic 0/0, but changing the harmonic order/degree to 6/6 at 95 km and propagating up to 4 km, the simulated results have no more presented this discrepancy. Since 1 degree means distance of 110 km (approximately), it can not be neglected. So, non-spherical gravitational field is recommend unless the landing area is big enough to compensate this error. At this work it is assumed a gravitational field model with harmonic order/degree (6/6) since this order/degree is also recommended for propagation in orbital level.

Table 2 - Coordinates of vehicle at 4 km.

DEG./ORDER HARMONIC		COORDINATES OF VEHICLE AT 4 KM DE ALTITUDE AFTER REENTRY		
Zonal	Tesseral	TIME (s)	LONG. (DEG)	LAT. (DEG)
0	0	2038,4	78,55180°	-2,501944°
2	0	1958,5	73,54476°	-2,880521°
3	0	1958,4	73,54192°	-2,881075°
4	0	1958,4	73,53018°	-2,881888°
5	0	1958,4	73,53084°	-2,881793°
6	0	1958,4	73,55054°	-2,882121°
6	6	1958,9	73,55054°	-2,880005°
30	30	1958,9	73,55744°	-2,879184°

Simulations

By testing all procedures in this work, it was simulated a reentry mission, ballistic type. The orbital elements and terrestrial coordinates of the center of landing area are presented in Table 3. The mission must finish after 7 days and before 10th day. The landing area is a circle with radius of 20 km. The vehicle has a cone shape, rounded nose, with C_d=0.1362, mass=150 kg (including propellant).

For simulations, criteria FPA (Flight Path Angle) and *Perigee* were applied. For FPA, the propulsive sub-system keeps on until it gets γ=-4° at 95 km of altitude (means a height of perigee $\approx$ -870km). For *Perigee* criterion, the propulsive sub-system keeps on until it gets an altitude of instantaneous orbital perigee equal zero, or $\mathbf{r_p = R_e}$, where $\mathbf{R_e =}$ Equatorial radius of Earth (means a $\gamma \approx$ -1.4°).

Table 3 - Initial Condition for orbit and coordinates of center of landing area.

ORBITAL ELEMENTS	
Date:	30/07/1990 12:00:00
a: (semi-major axis)	6678139 (m)
e (excentricity)	0.001
i (inclination)	5°
Ω (Right Asc of A.Node)	300°
ω:(perigee)	20°
M (Mean Anomaly):	30°
COORDINATES OF CENTER OF LANDING AREA	
Longitude	320°
Latitude	-2°
Altitude	0 (m)

Visibility Problem

The orbit of the mission was simulated for 10 days. The visibility was analyzed and filtered. Table 4 shows the passages with lateral error less than 20 km (radius of landing area). The reentry must be commanded at 9th day because all others are out of mission life-time

The signals (+/-) of *lateral error* means that vehicle is distant to/from center of landing area.

Table 4 - Crossing time and lateral error between the vehicle position and the center of landing area.

Time from launching (mission day)	(sec)	Lateral Error (km)
5	435042	+18,26
5	435044	+7,08
5	435046	-12,27
6	550614	+13,27
6	550616	-5,02
6	550618	-16,13
9	856846	+9,09
9	**856848**	**-6,49**
9	856850	-19,64

Propellant Consumption

It is assumed a propulsive system with a trust force, $\mathbf{T} = 500$ N, using a propellant with specific impulse, $\mathbf{I}_{sp}$ = 280s. The propellant load, Δm, is:

$$\Delta m = \frac{T\,\Delta t}{g\,I_{sp}} \tag{7}$$

where: $\mathbf{T}$ is trust force; Δt is period that the propulsive system is kept on; $\mathbf{g}$ is the local gravity.

Table 4 - Estimated Propellant Load, Time to Start Orbital Maneuver to reentry.

Criteria	FPA	*Perigee*
Burst start time	856086,4 s	854998,4 s
(Δt)	72 s	25,5 s
(Δm)	13,1 kg	4,6 kg

Since the *Perigee* criterion is equivalent to a lower flight path angle (absolute value), it explains the different results between the criteria. A small flight path angle (absolute values) requests a small mass of propellant and small period of time, but the start-time of maneuver must be done in advance. The simulation takes into account that propulsive sub-system reduces the transversal component of velocity in plane RV (position & velocity)

Deceleration

The deceleration curves for trajectories are showed by Fig. 4 level for *perigee* criterion is lower than FPA criterion because the flight path angle at 95 km of first criterion ($\gamma = -1.4^\circ$) is lower than the second one ($\gamma = -4^\circ$).

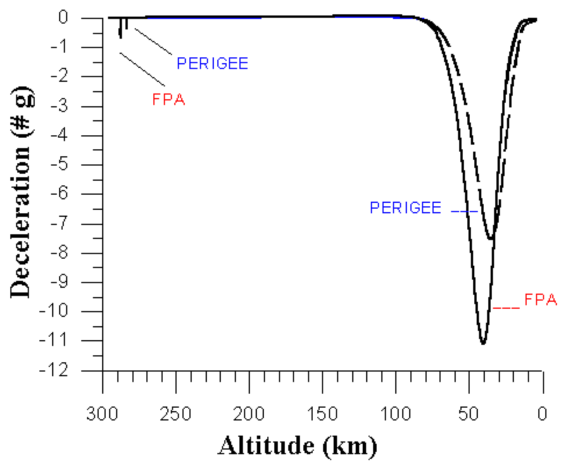

Fig. 4 - Deceleration versus Altitude

Touching down

Figure 5 shows the terrestrial angular coordinates of burst time and of landing point. It is import knows the coordinates of burst time to locate critical ground stations that will support the mission. The vehicle reaches the landing coordinates by using both criteria.

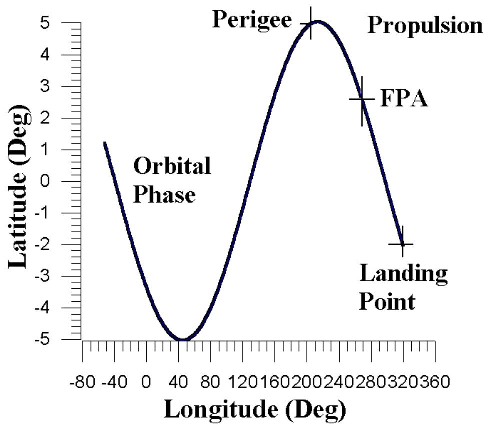

Figure 5 - Orbital phase, start time of reentry injection and Landing Point.

Fig. 6 shows details of landing trajectories, at altitude below 25 km. The solid lines represent the trajectory using FPA criterion and the dashed ones represent that one using the *Perigee* criterion. The straight lines are in plane of coordinates. The curved lines are in plane of altitudes & longitude of vehicle. At middle top, the mark is the center of landing area.

Below 10 km of altitude the trajectory is completely vertical. Note that for a lower nominal orbit this vertical decay can occur at higher altitude.

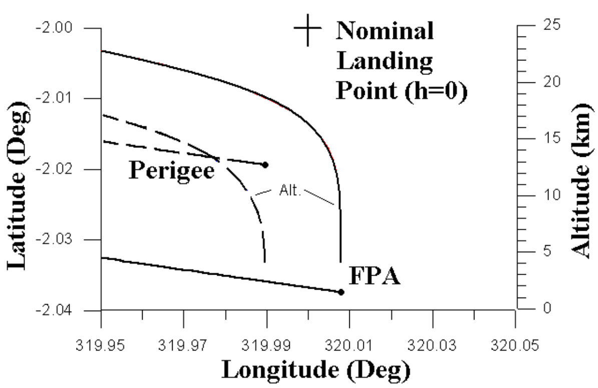

Figure 6 - Details of the trajectories of vehicle below 25 km for criteria FPA and *Perigee*.

The distances between touching down point and the nominal landing point (center of the landing area over the plane Latitude versus Longitude), are approximately 3.3km and 4.4km for criteria Perigee and FPA, respectively.

Conclusions

a) Degree/order of harmonic used in the gravitational field has significant influence to reaches the landing point with precision in reentry problem.
b) It is necessary to take into account the landing area constraints on the initial condition of orbital launching otherwise no reentry window can exist during the mission life-time and may be necessary lateral maneuvers.
c) The numeric method to identify the reentry window shows to be efficient to provide a lateral error lower than precision less then 4.4 km for a ballistic reentry from an orbital level of 300km.

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