TRAJECTORY SIMULATIONS FOR THRUST-VECTORED ELECTRIC PROPULSION MISSIONS

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IEPC-03-0050

Abstract

The effects of thrust misalignment and Thrust Vector Control (TVC) have been investigated. Thrust vector misalignment about the pitch axis increases the orbit transfer time, but the main drawbacks are the increase in AOCS requirements and, if the thrust vector orientation is not corrected, the possibility of large error in final position. TVC about the yaw axis has been examined for orbit altitude and inclination change. Two guidance methods were investigated giving similar satisfactory results. The results show that a very small deflection angle and angle rates are required, leading to concerns as to whether real-world actuators and attitude sensors would be capable of the necessary precision.

1. Introduction

The ability to control the thrust vector of any spacecraft propulsion system is extremely advantageous. It can be used both to improve or optimise mission performance and also to compensate for the shift in position of the centre of mass in order to minimise attitude control requirements. In general, thrust vectoring is achieved by the use of a mechanical gimballing mechanism. This usually imposes a significant mass penalty, however. This is particularly true in the case of gimballed electric thrusters, where the mass of the pointing mechanism typically exceeds that of the thrusters.

To date, ion thruster development has concentrated on issues such as thrust level, thruster lifetime and throttleability. Now that these have reached acceptable performance levels, interest is being shown in 'second-generation' capabilities. One of the most important of these is the application of integrated thrust vectoring. Given the long burn times of electric propulsion systems, the ability to vector the thrust by only a few degrees would significantly increase the range and capability of ion thruster space mission applications.

2. Background

Thrust Vector Control (TVC) is highly beneficial, whatever the propulsion system considered. The main issues are, on the one hand, the optimisation of the propellant cost and therefore of the overall mission: on the other hand, it has to compensate for the shift in position of the centre of mass, which is similar in effects to thrust misalignment. Indeed, in the case of NSSK, this effect dominates over any other perturbations [Fearn[1]]. Thrust vectoring is usually achieved by use of mechanical gimbal systems, such as those used for the classical chemical thrusters. However, these are very expensive and represent a significant mass penalty for electric propulsion, as their mass can actually be larger than that of the ion thruster. In addition, because of the moving parts, there is a major concern about their reliability for long duration missions [Fearn[1]].

There would therefore be a major benefit in developing thrust vectoring systems dedicated to EP. Investigations of such systems have been carried out in the past forty years, including both mechanical and non-mechanical designs. However, the mission performance depends on two important things: the guidance and the controller. Extensive work has been carried out on the guidance in a point-mass analysis, but less has been conducted examining guidance and the controller together in a rigid spacecraft dynamics analysis.

Presented as Paper IEPC-03-0050 at the 28th International Electric Propulsion Conference, Toulouse, France, 17-21 March, 2003.

The spacecraft modelled in the simulations described hereafter is derived from a modified version of the ATLAS in-orbit servicer [Ellery[2]]. It is assumed to be a homogenous 1100-kg cube, including an estimated 250 kg of Xenon propellant. The external dimensions are 2 m in height (along yaw axis), 2 m in width (pitch axis) and 2.5-m long (roll axis). The two solar panels (200 kg each) are mounted on the pitch axis of the satellite. They are 12 m in length (s/c pitch) and 3-m wide with a negligible thickness. The arrays have one degree of freedom about the pitch axis, and the motion is controlled by a PID controller to follow the Sun. It is assumed that the initial orbit is a 215-km altitude circular orbit with an inclination of 5 degrees. This is very similar to a launch from the European Spaceport in Kourou, French Guyana.

A possible candidate for the main propulsion system is a T6 ion engine, whose characteristics are given by Wallace et al[3]. To briefly summarise the latter, the T6 is a UK Kaufman-type ion thruster with a SAND (Screen, Accelerator N' Decelerator) grid configuration dished inwards. The performance parameters range from a 40- to 220-mN thrust, with a corresponding estimated specific impulse of 3250 and 3500 seconds, respectively. The total input power for the T6 ranges from 1 to 5.5 kW, approximately. However, it was assumed that the thrust could be stretched to 250 mN, with the power requirement and specific impulse modified accordingly.

As usual, one of the first tasks is to define a set of orientation axes. Apart from the Earth-Centred Inertial axes, there are two main frames: the local horizontal, local vertical axes, noted $(X,Y,Z)_H$, and the Body frame $(X,Y,Z)_B$.

The H frame is defined as $X)_H$ being parallel to the local horizon and, considering that the orbit at every instant is almost perfectly circular, parallel to the velocity vector, and in the same direction as the latter. $Z)_H$ is pointing to nadir and $Y)_H$ completes the triad.

With neither an angle of attack nor a sideslip angle, the Body frame is identical to the H frame. With the spacecraft assumed to be cubical, $X)_B$ goes though the front panel, with the engine mounted on the rear panel $(-X)_B)$, $Z)_B$ passes through the bottom panel. Finally, $Y)_B$ goes through the side panels, completing the triad. The solar arrays, once deployed, can rotate about $Y)_B$.

The effects of thrust misalignment and vectoring on mission performance are investigated individually about the pitch and yaw axes.

3. Pitch Axis Thrust Misalignment

A change in the thrust angle about the pitch axis would create a force component in the orbital plane, tangential to the velocity vector and aligned with the position vector. This force would have an influence on the dimension of the semi-major axis of the orbit.

It was shown by Kluever[4] that the optimal in-plane steering angle α_T that maximises the rate a' (the rate of change of the semi-major axis) is found by setting the partial derivative $da'/d\alpha_T$ equal to zero:

$$\frac{\partial a'}{\partial \alpha_T} = -\frac{2a^2 v}{\mu} a_T \sin\alpha_T = 0 \qquad (1)$$

This differentiation is derived from the original equation for the rate of change of the semi-major axis:

$$a' = \frac{2a^2 v}{\mu} a_T \cos\alpha_T \qquad (2)$$

where μ is the gravitational constant, v is the velocity magnitude, and a_T is the thrust acceleration magnitude (thrust/mass).

It can be seen from equation (1) that the extremal steering law for maximum a' is for $\alpha_T = 0$. In other words, the thrust vector must be aligned with the velocity vector.

If there is no gimbal system to provide the adjustment required to the in-plane steering angle, then the spacecraft AOCS system must ensure that the thrust vector is correctly orientated by rotating the whole satellite about its pitch axis $Y)_B$. In the simulations, it was assumed that the thrust misalignment was constant.

4. Yaw Axis Thrust Misalignment and Control

If the thrust vector is steered about the yaw axis, it would create an out-of-plane force. This force could change the orbit inclination, providing that its orientation is changed when crossing the ascending and descending nodes, in order to keep this force orientated towards the equatorial plane. Let us assume a spacecraft is on an orbit with an initial inclination between 0 and 90 degrees, and it is required to bring the spacecraft in an equatorial orbit. Between the ascending node and the descending node (0<u<180 deg, u being the argument of latitude), this tangential force should be in the positive $Y)_B$, while it should be changed to the opposite direction when crossing the descending node and travelling towards the ascending node (180<u<360 deg).

When modifying the orbit of a spacecraft, it can therefore be interesting to control the change in semi-major axis and inclination together. This would however require a guidance law to achieve both goals at the same time, in order to optimise the mission by saving transfer time and propellant.

Two guidance methods exist and were compared in a LEO-GEO transfer. However, both are designed to work out the thrust steering angle in a point mass analysis, whereas we are considering the dynamics of a rigid spacecraft, therefore the guidance methods will now give us the sideslip angle.

The first method uses a cosine guidance law, very similar to the constant guidance one. The latter would set a finite yaw angle, with its sign being changed when crossing the ascending and descending nodes. This presents two main drawbacks, however. First, the corresponding guidance would be a near-square signal, making it hard for a controller to follow. The second, and probably most important in term of efficiency, is that this tangential force would be wasted at an angle of 90 degrees from the nodes anyway, where di/dt = 0 [Kluever & O'Shaughnessy[5]]. Therefore, it is better to have a wave signal dependent on the argument of latitude. The cosine guidance law is therefore defined as:

$$\beta_T = \beta_{T_{\max}} \cos u \qquad (3)$$

where β_{Tmax} is the maximum, optimised deflection angle to achieve a geostationary orbit. Then, it is modified to give more change in inclination at higher altitudes, where such a manoeuvre is more efficient. This is done by setting a linear relationship between T and the spacecraft attitude. The corresponding β_{Tmax} was determined by iteration in a point mass analysis as 7.2765 degrees. The corresponding guidance profile is shown in Figure 1.

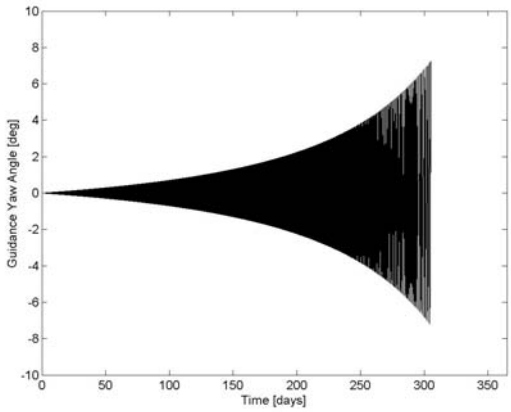


Figure 1. Guidance profile of the Constant/Cosine guidance method

The other guidance method is the so-called Inclination Change Efficiency (ICE), developed by Yoon & Tuckness[6]. Not only does it achieve the target altitude and inclination simultaneously, but it is also designed to use more thrust for altitude change where the efficiency for inclination is low and to use more thrust for inclination change where the efficiency for inclination is high. The ICE guidance is derived from the second derivative of potential function:

$$ICE(r) \equiv \frac{U''(r) - U''(r_0)}{U''(r_f) - U''(r_0)} = \frac{r_f^3}{r_f^3 - r_0^3}\left(1 - \frac{r_0^3}{r^3}\right) \tag{4}$$

where

$$U(r) = -\frac{\mu}{r} \tag{5}$$

This ICE function is then used to calculate cn and ct, the normal and tangential components of the force orientation, respectively.

$$cn(r,u) \equiv Gain \cdot ICE(r) \cdot ICE(u) \tag{6.a}$$

$$ct(r,u) = \sqrt{1 - cn^2(r,u)} \tag{6.b}$$

where

$$ICE(u) \equiv |\cos u| \tag{7}$$

The value of Gain is determined to achieve the targeted altitude and inclination simultaneously, using an iterative process, as for the previous method, as suggested by Yoon and Tuckness[6]. It was found once again in a point-mass analysis to be 0.0703, corresponding to a maximum attitude angle of about 4 degrees.

Finally, the out-of-plane steering angle is found with:

$$\beta = \tan^{-1}\left(\frac{cn}{ct}\right) \tag{8}$$

The steering angle history corresponding to this guidance method is shown in figure 2.

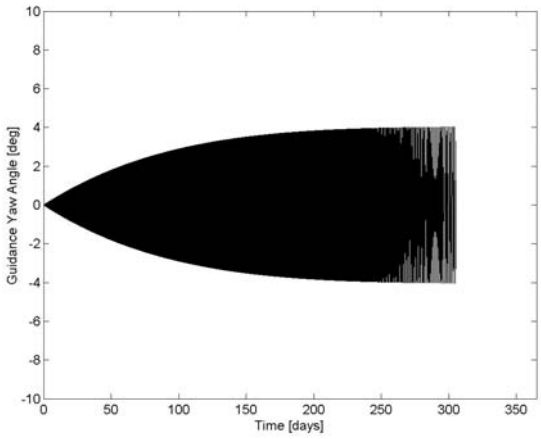


Figure 2. Guidance history for the ICE guidance method

4.1 Control Loop

In both yaw guidance methods, the control loop does not consider the guidance as the thrust steering angle but as the optimal spacecraft attitude in order to align $X)_B$ with $X)_H$. If this is achieved at any moment, then the steering angle will be set to zero, and the thrust vector will be in turn aligned with the velocity vector to satisfy the condition defined by equation (1). The difference between the guidance and the spacecraft attitude is used by the PID controller to calculate the actual out-of-plane thrust deflection angle. This has the advantage of keeping the AOCS system completely out of this process. Because the guidance profile is different in amplitude and frequency for each guidance method, the P, I and D gains are different for each guidance method.

The first version of the controller had free range in angular rate. However, following the first set of results for the ICE guidance, the latter was also tested with a maximum angular rate of ± 1 degree per second.

5. Results

5.1 Pitch Axis Thrust Misalignment

It is assumed that the change in attitude is performed before the ion engine is ignited. Therefore, the satellite is already in position to optimise the thrust. As could be expected, the Time of Flight (ToF) and the propellant consumption remains identical for any deflection angle, at 305.1 days and 191.927 kg, respectively. The reason is that, in every case, the thrust is aligned with the velocity and therefore all the thrust is used to increase the semi-major axis of the orbit.

The only difference is the torque induced by the fact that, although it is parallel to the velocity vector, the thrust is not aligned with the Centre of Mass (CoM). Obviously, the greater the deflection angle, the greater the magnitude of the torque, as shown in figure 3.

The induced torque is of the magnitude of 10^{-2} N.m, which is in the same order of magnitude as the sum of the other disturbing torques (gravity gradient, solar radiation, aerodynamic forces and magnetic field), especially in LEO. This would at least double the total disturbing torques for a misalignment angle greater than 2 degrees, and increase to more than 500 % in a worst-case scenario (10-degree misalignment). Based on the latter figure, this would more than double the AOCS design characteristics (such as the position gain). It should be noted however that maintaining the spacecraft in this position makes it more demanding for the AOCS as it also increases the disturbing torque due to gravity gradient. It is therefore worth looking at the mission performance in the case where the AOCS keeps the Body frame aligned with the H frame.

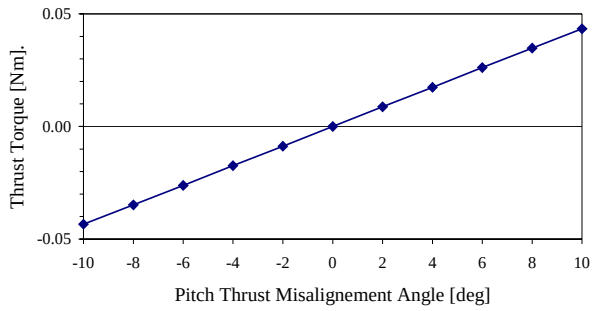


Figure 3. Thrust torque for a range of thrust misalignment angle in pitch

In the second case, the AOCS does not align the thrust, but still has to compensate for the disturbing torque. The Time of Flight (ToF), propellant mass and final orbit eccentricity are shown in figure 4 for a range of thrust misalignment angle. In the worst-case scenario, an extra 4.5 days can be added to the ToF of 305 days for the optimal case, which represents an increase of only 1.5%. This ratio is the same for the mass of propellant consumed, due to the linear relationship between the two parameters for an unthrottled engine. The increase in propellant mass used for the mission is less than 3kg. The final eccentricity can vary quite randomly from one misalignment angle to another. But in any case, this proves not to be a major issue, owing to the fact that the misalignment is kept constant and the eccentricity remains very small. The major

problem lies in fact in the delivered position (in terms of argument of latitude) of the spacecraft, this is especially true for a geostationary satellite or for a rendez-vous mission. Assuming that the desired position is obtained for zero deflection angle, the error in position due to thrust misalignment is clearly shown by Figure 4d. If the spacecraft is allowed to stop before the targeted altitude or, on the contrary, to overshoot it in order to meet the desired position, a random pattern profile for the altitude error is obtained with a pick error of 200 km.

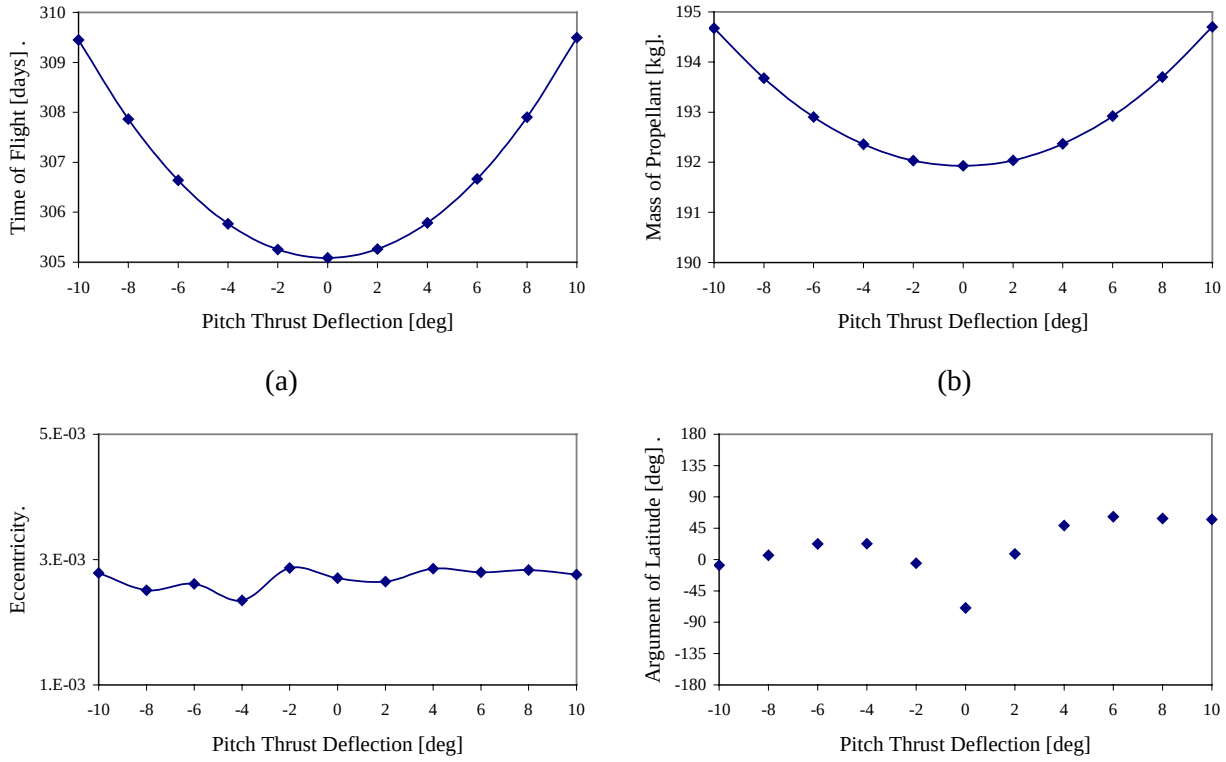


Figure 4. Mission performances for thrust misalignment in pitch axis

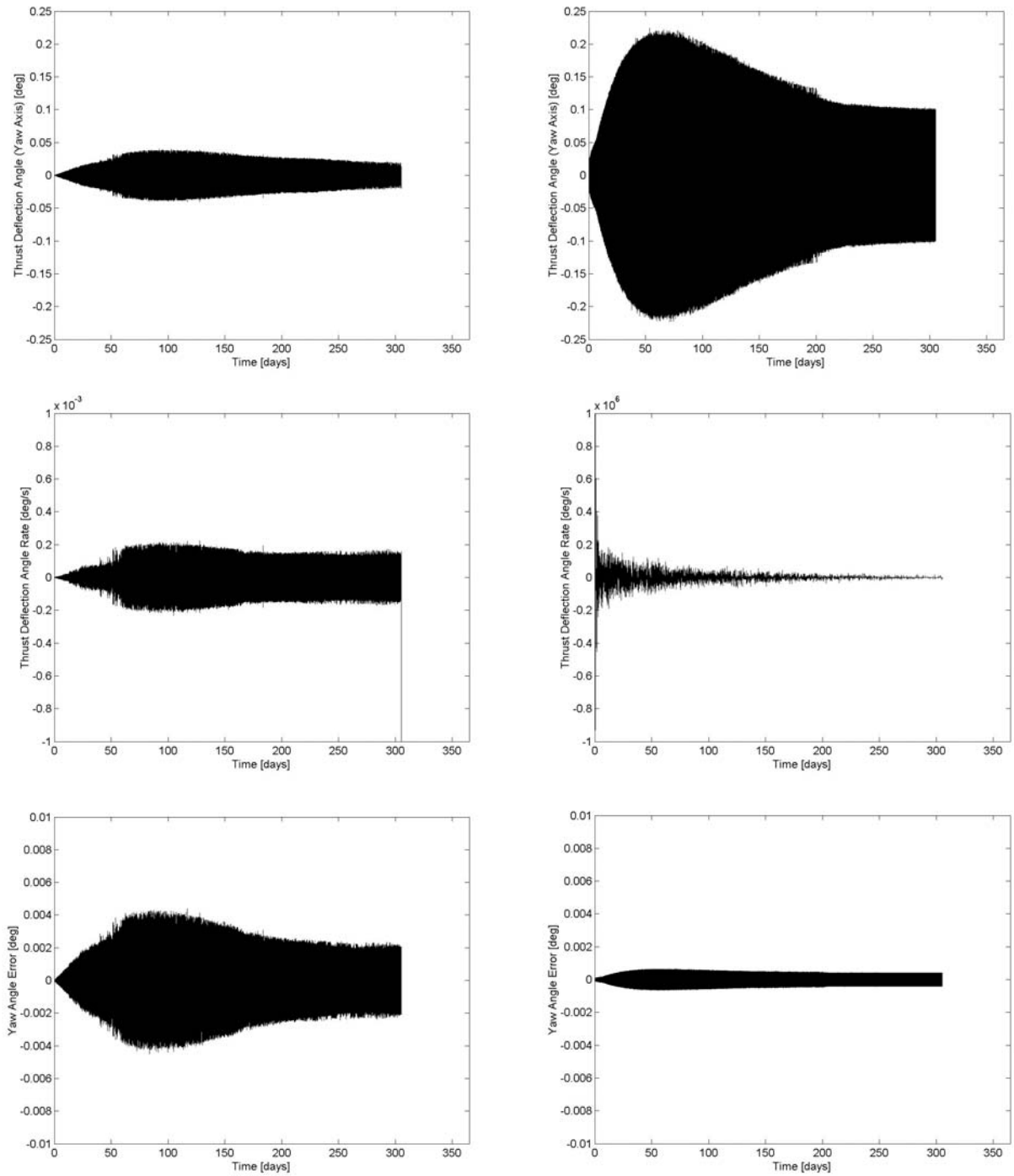
It is therefore better to align the thrust with the velocity vector. If a gimbal system is provided, not only can it help nulling the effect of thrust misalignment but it can also offload the AOCS actuators during the orbit transfer. The effect due to the subsequent tangential force should not be as dramatic as described in the second case as it would only be occasional and not constant. However a control loop taking into account the spacecraft position is highly desirable in any case.

	Cosine	ICE	ICE rate-limited
Altitude [km]	35786.0	35786.0	35786.0
Inclination [deg]	-2.999E-03	-3.509E-03	4.157E-03
Time of Flight [days]	305.263	305.322	305.322
Propellant Mass [kg]	192.040	192.077	192.077
Maximum Thrust Deflection Angle [deg]	0.0401	0.223	0.2267
Maximum Thrust Deflection Angle Rate [deg.s^{-1}]	2.31E-04	1.00E+06	1

Table 1. Comparison of mission performances and actuator requirements for different guidance methods for TVC in yaw

5.2 Yaw Axis Thrust Vector Control

Both guidance methods give similar mission performances, as shown in Table 1 and Figure 5. The targeted altitude and inclination are matched to within a negligible fraction, and the ToF and propellant mass are quasi-identical. The differences are mainly due to the fact that the amplifier gain as well as the controller gains are tuned for each method. If this were not done and a single controller were used for both cases, one would work while the other would fail completely.



(a) (b)

Figure 5. History of Thrust Deflection Angle, Deflection Angle Rate and Attitude Error for (a) the Cosine Guidance and (b) the ICE Guidance

Although the maximum thrust deflection angle is very small in both cases, its rate becomes unmanageable for the controller with the original ICE guidance, where a peak value of 10^6 deg.s^{-1} can be observed. The ICE

guidance was again tested but the control signal was limited to a rate of change of ± 1 deg.s^{-1}, and its results are shown in the right hand side column of Table 1. It can be noted that the mission performances remain almost the same, although the amplifier gain had to be increased to be able to follow the guidance. The thrust deflection angle and the spacecraft attitude error are hardly altered (Figure 6).

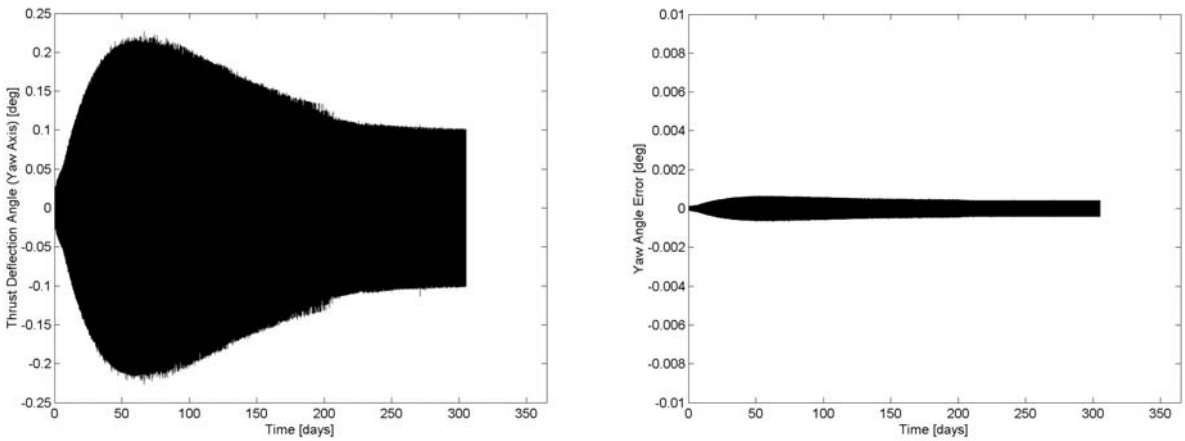


Figure 6. ICE Guidance Method with Thrust Deflection Angle Rate Limitation

Although the results are very satisfactory, it should be kept in mind that, even if the TVC half-cone is very small and the angle rate is acceptable, the precision required is such that, on the one hand, it would require a very fine determination of the spacecraft attitude and, on the other hand, no actuator could possibly meet it. Indeed, a preliminary analysis has shown that attitude error precision greater than 10^{-5} deg would not allow the controller to follow the guidance. Furthermore, the actuator step-size was not taken into account. This would not be a major problem since the actuator step-size could be larger than the maximum required TVC angle, but the controller would obviously need to be redesigned. However, the smaller the step-size, the better the precision, and the smaller the required deflection angular rate. The use of integrated thrust vectoring could have a major impact, as performances are expected to be much better than mechanical systems, not only in precision but also in response time. As the results above showed, this would have a major impact on the controller and the mission performances, as the thrust deviation could be kept as small as possible with fast response. However, thrust vector stability for the whole mission duration is compulsory.

6. Conclusions

The effects of thrust misalignment and the benefits from TVC have been shown. Thrust misalignment about the pitch axis can multiply the disturbance torque by a factor of 5, which would in turn more than double the requirements on the AOCS. If the AOCS does not orientate the spacecraft to align the thrust vector with the velocity vector, minor increase in ToF is observed but a major error in the final position of the spacecraft can occur.

Thrust Vector Control about the Yaw axis can be used to modify the inclination of the orbit. However, a guidance law is required to optimise the orbit transfer. Two guidance methods were tested with both the cosine and the ICE methods giving similar, satisfying results. The requirements on the controller are very small, however parameters such as the actuator step size and the attitude determination would require further work to be done. It is believed that integrated thrust vectoring would be highly beneficial because of a better deflection precision and faster response time. However, these advantages depend strongly on the capability of maintaining a stable thrust vector deviation.

7. References

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