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To cite this version:

Nabila Zbiri, Zohra Manseur. Control of an Aerospace Launcher. 2nd Mediterranean Conference on Intelligent Systems and Automation (CISA 2009), Mar 2009, Zarzis, Tunisia. pp.377-380, 10.1063/1.3106507 . hal-00764382

HAL Id: hal-00764382

https://hal.science/hal-00764382v1

Submitted on 5 Jul 2021

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Control of an Aerospace Launcher

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Abstract : This research is within the framework of the PERSEUS project proposed by the CNES (Centre National d'Etudes Spatiales). Its aim is to develop new concepts for the attitude control of space modules. This article presents a first study as well as the results of a robust *LQG* control system that allows stable and satisfactory performance for the attitude of a rigid launcher.

Key words: launcher, guidance, *LQG* control.

1. INTRODUCTION

Automatic guidance of airplanes, missiles and space vehicles, satellite stability and robot control constitute a privileged field of application for advanced automation methods. For future needs of the **CNES** and within the framework of the **PERSEUS** project, it will be useful to develop appropriate methodologies for piloting space vehicle launchers. It is important to use a model for a launcher whose complexity is compatible with the development of piloting rules. In order to achieve this goal, a good understanding of the system is needed including the location of its non linearities, the variation range for its in-flight parameters, and the importance of coupling between the 3 axes of motion.

The launcher is a complex, nonlinear, unstable multivariable system with parametric and dynamic uncertainties. It is also a high order system due to its flexible modes.
The main objective for the development of a control law is to confer to the system properties it does not naturally have or to reinforce those properties if they already exist.
Several control methods have been proposed in the literature. Wang and Stengel, have developed a robust nonlinear control system based on a model in which the thrust is constant and independent from the aerodynamic attitude [1]. Xu et Ioannou, have adapted their method by including the presence of uncertainties [2]. In [3], the model used is closely dependant on the launcher's geometry which makes their method difficult to generalize.
The model used in [4] was used as a reference in the work described in this article. The work presented here is about the development of a robust control system that limits the aerodynamic incidence of a rigid launcher. Indeed, a stronger incidence leads to stronger bending forces that may destabilize the launcher. However, incidence is not measurable. In this study, an *LQG* controller allows estimation of the incidence and stabilization of the launcher about a zero incidence. This article is organized in five sections as follows:
In the second section, some general concepts as well as some characteristics (piloting loop and guidance) of the launcher are presented along with its model. The third section presents the development of a *LQG* control system is to provide stability for the launcher's attitude about a zero incidence. The results of a computer simulation will be presented and discussed in the fourth section of this article. Finally, the conclusion of this work and future prospects are given in the last section.

2. MODELLING THE LAUNCHER

From a complete model, several simplified models have been developed in the literature [1-3]. Besides the launcher's dynamics which are nonlinear, other elements act on the control chain: sensors, actuators, and the computer. These elements may induce nonlinearities in the model. For example, hydraulic, electric or hybrid actuators have limited range, velocity saturation, or hysteresis effects.
However, it is not necessary to use nonlinear models in the design phase of the piloting control system. Motions of the launcher are sufficiently slow and angular variations are sufficiently small to justify the use of convenient linear approximations of the dynamics. It is therefore necessary to develop a dynamic model that combines realism and simplicity.
The launcher is considered to be a rigid structure. The problems associated with modelling the launcher's flexible modes will not be addressed. However, the effects of the flexible modes are in general considered as disturbances that can be added to the measurements in the control system [6].

It is assumed that the motions in each plane (pitch, roll and yaw) are sufficiently decoupled to be considered independently. In other words, if the motion about all three axes is to be controlled, three controllers should be implemented, one for each axis. Due to the launcher's symmetry, motions about the yaw and roll axes can be controlled using the same system. The 3-dimensional model can then be replaced by a 2-dimensional model without any loss of generality. Therefore, this study concentrates on controlling the pitch motion of the launcher only.
Figure 1 shows the motion of the launcher and the forces applied on it.

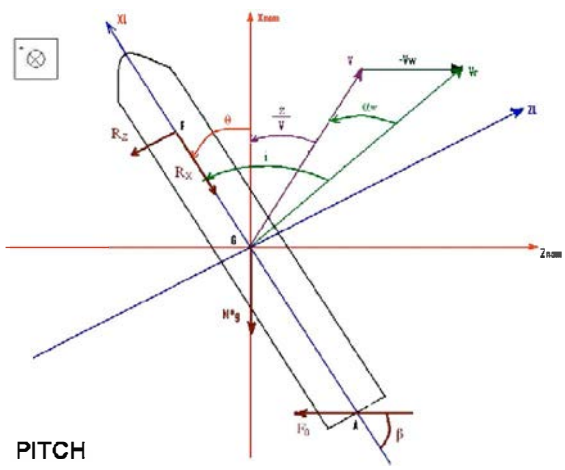

Fig. 1. Forces exerted on the launcher in the pitch plane.

Notations:

V : absolute velocity
V_r : relative velocity
V_W : wind velocity
$\dot{z}$: drift velocity along the pitch axis
i : incidence of the vehicle
β : deflection of the tailpipe
θ : pitch angle ('attitude)
α_w : angle between V and V_r
C_X : aerodynamic drag coefficient
C_Z : aerodynamic lift coefficient
R_X : drag force
R_Z : lift force
F_0 : engine thrust
M : instantaneous mass of the launcher
P_d : dynamic pressure
S : reference surface of the launcher
L_F : arm of the aerodynamic control lever
L_T : arm control lever of the propulsion force

In order to obtain a useful working model for piloting the launcher, the following assumptions must be made concerning its dynamics, as well as the control parameters:

- The motion along the trajectory is assumed to be straight and linear.
- Motions about the center of mass are small.
- The flexible modes are considered as disturbance inputs in the control system.
- The rocking effect of the propellant fuel is neglected.

The dynamic equations of the launcher allow us to determine a state space model that leads to the development of an appropriate and more robust control system for the launcher. The following inputs and outputs will be considered:
Inputs:
- deflection reference angle β
- wind disturbance (the wind velocity is V_W)

Outputs:
- bearing angle θ as measured by the inertial sensor
- attitude angle variation velocity $\dot{\theta}$ as measured by the gyroscope.

With the definitions,

$$u = \begin{pmatrix} \beta \\ \dot{V}_W \end{pmatrix} \text{ and } x = \begin{pmatrix} x_1 \\ x_2 \\ x_3 \end{pmatrix} = \begin{pmatrix} \theta \\ \dot{\theta} \\ i \end{pmatrix},$$

the system state representation is given by:

$$\begin{cases} \dot{x}_1 = \dot{\theta} \\ \dot{x}_2 = A_6 i + K_1 \beta \\ \dot{x}_3 = \dot{\theta} - \dfrac{a_2}{V_r}\theta - \dfrac{a_1}{V_r} i - \dfrac{F_0}{M * V_r}\beta - \dfrac{\dot{V}_W}{V_r} \end{cases} \quad (1)$$

where:

$$A_6 = \frac{P_d S C_Z L_F}{I_{yy}},\ K_1 = \frac{F_o L_T}{I_{yy}},\ a_1 = \frac{P_d S C_Z}{M},\ a_2 = \frac{F_o - R_X}{M} \quad (2)$$

Or, in matrix form, as:

$$\dot{x} = \begin{bmatrix} \dot{\theta} \\ \ddot{\theta} \\ \dot{i} \end{bmatrix} = \begin{bmatrix} 0 & 1 & 0 \\ 0 & 0 & A_6 \\ -\dfrac{a_2}{V_r} & 1 & -\dfrac{a_1}{V_r} \end{bmatrix} X + \begin{bmatrix} 0 & 0 \\ K_1 & 0 \\ -\dfrac{F_o}{M * V_r} & -\dfrac{1}{V_r} \end{bmatrix} \begin{bmatrix} \beta \\ \dot{V}_W \end{bmatrix} \quad (3)$$

The launcher is a non stationary system. Its physical characteristics (position, mass, inertia) and its aerodynamic parameters vary according to its flight. Typical curves describing the variation of the coefficients A_6, K_1 a_1 and a_2 are shown in figure 2 [4].
These coefficient curves are those of the Ariane 4 rocket but they have the same shapes and the same orders of magnitude as those obtained from the Ariane 5 rocket.

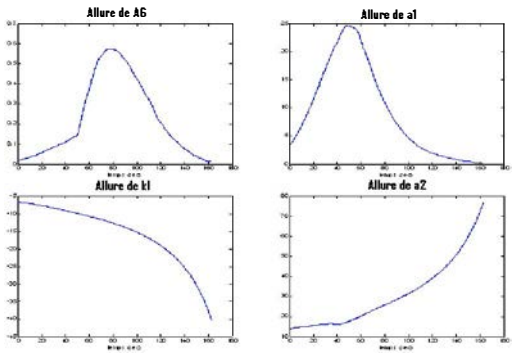

Fig. 2. Variation curves for A_6, K_1 a_1 and a_2.

3. DESIGN OF THE *LQG* CONTROL

The state representation (3) can take the form:

$$\begin{cases} \dot{x} = Ax + B_1\beta + v \\ y = Cx + \omega \end{cases} \quad (4)$$

where $v = B_2\dot{V}_W$ and ω is the noise for the sensor measurements. They will be considered as white noise, centered with variance $v > 0$ and $\omega > 0$.

The measurable variables are the pitch angle θ, and the pitch velocity $\dot{\theta}$ obtained with the help of an inertial sensor and a gyro. The output matrix can then be written as:

$$C = \begin{bmatrix} 1 & 0 & 0 \\ 0 & 1 & 0 \end{bmatrix}$$

One difficulty encountered in the control of the launcher is that the incidence, a variable to be controlled, is not measurable. The solution is to improve the state model using a state estimator. The use of the *LQG* control method is thus necessary for this application.

It can be verified that:

$$rank\begin{bmatrix} B_1 & AB_1 & A^2B_1 \end{bmatrix} = rank\begin{bmatrix} C & CA & CA^2 \end{bmatrix} = 3$$

Therefore, the system is controllable and observable.

Reconstruction of the state vector

The estimate $\hat{x}$ is determined by a Kalman filter optimized by minimizing the quadratic norm of the estimation error. Its state equation is given by:

$$\dot{\hat{x}} = A\hat{x} + B_1\beta + K(y - C\hat{x}) \quad (5)$$

where the filter gain is determined by the following expression:

$$K = PC^TW^{-1} \quad (6)$$

where the matrix P is obtained by solving the Ricatti equation :

$$AP + PA^T + V - PC^TW^{-1}CP = 0 \quad (7)$$

By using the estimated state vector $\hat{x}$, it is possible to find the optimal controller that minimizes the quadratic criterion:

$$J = \int_0^\infty (x^TQx + u^TRu)dt \quad (8)$$

where Q and R are diagonal weight matrices. The control input is therefore given by: $u = -L\hat{x}$

The gain matrix L is computed by using the expression:

$$L = R^{-1}B^T\tilde{P} \quad (9)$$

where $\tilde{P}$ is a symmetric definite positive matrix solution of the Ricatti equation :

$$A\tilde{P} + \tilde{P}A^T + Q - \tilde{P}BR^{-1}B^T\tilde{P} = 0 \quad (10)$$

4. SIMULATION RESULTS

Computer simulations are conducted by using the values $A_{6\min}$ and $A_{6\max}$ retrieved from the curves of figure 2.

At first, the wind disturbance is not considered. The results for the simulations are given by the curves in Figure 3 for $A_{6\min}$ and the curves of Figure 4 for $A_{6\max}$.

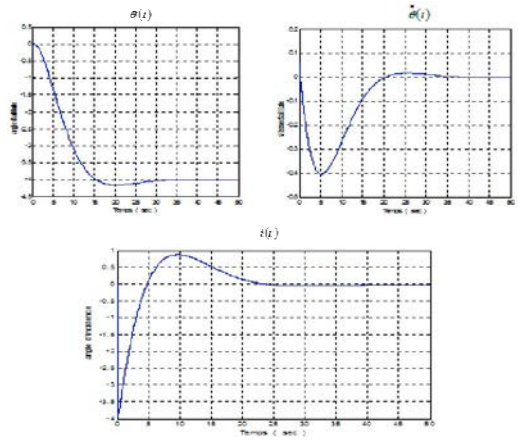

Fig. 3. State variable curves for $A_{6\min}$

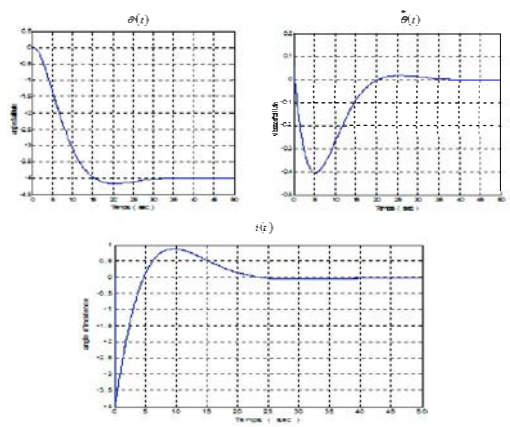

Fig. 4. State variable curves for $A_{6\max}$

It is important to note that without considering the disturbance effects, the curves for both cases are similar for all three state variables $(\theta, \dot{\theta}, i)$. It is however worth noticing

that the time performance criteria (overflow and response time) are different. It has also been observed that the system behavior remains stable for intermediate values of A_6 but this article does not discuss this observation for lack of space. What is discussed here is that for variations of the characteristic parameter A_6 from its minimal value to its maximal value, the *LQG* controller provides stability of the launcher's attitude. In a first study, this allows the development of a control system without considering the non stationarity of the launcher.

Disturbance Rejection

The system represented by equation **(2)** is influenced by the wind. This disturbance, the rate of change of the wind, can be considered to be a wind burst, as described by Figure 5.

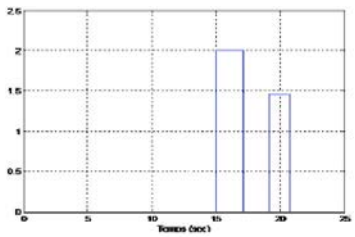

Fig. 5. Wind disturbance.

For $A_{6\max}$, the disturbance appears at t = 15 s, when steady state is reached. For $A_{6\min}$, a similar disturbance is also introduced in steady state at t = 45 s.

The simulation results obtained are shown on Figures 6 and 7. The state variables of the system are affected by the wind disturbance. However, the disturbance effects are quickly weakened in the two cases described in both figures. It is shown that the *LQG* controller allows the attitude of the launcher to be stable about a zero incidence while eliminating the effect of a wind disturbance.

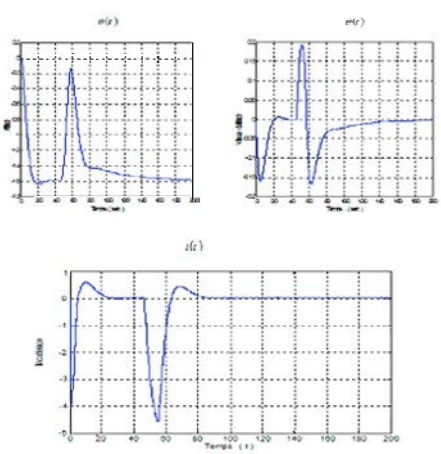

Fig. 6. State variables curves for $A_{6\min}$

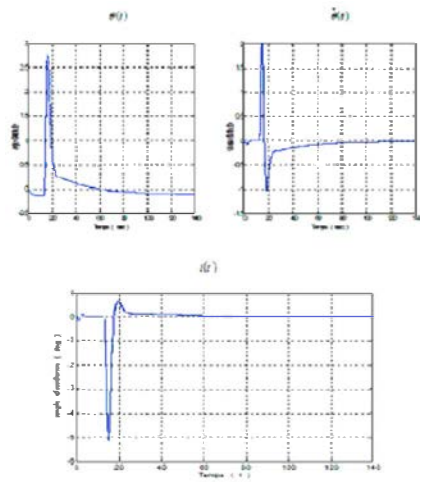

Fig. 7. State variables curves for $A_{6\max}$.

5. CONCLUSIONS

The results of the simulation that we presented show the *LQG* system ability to control the attitude of a launcher while rejecting the effect of a constant disturbance due to the wind. The nonstationarity of the system has been partly taken into account by considering values included between the maximum and minimum values of the characteristic coefficient A_6 of the launcher.

The *LQG* controller stabilise the launcher's attitude by maintaining its incidence angle close to zero. However, the model is considered linear. Further studies are pursued to develop a more robust guidance system, taking in consideration the system non linearities, especially the launcher parameters' variations.

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