FAULT DETECTION AND RECONFIGURATION APPLIED TO A HELICOPTER SWASHPLATE ACTUATOR

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Abstract

The objective of this paper is to evaluate the feasibility and the performance of some fault detection and reconfiguration techniques applied to the main rotor swashplate of a helicopter. The swashplate displacements and rotations allow to change the rotor blades pitch angle and is thus responsible for controlling the magnitude and orientation of the rotor lift and thrust forces: a failure affecting the actuators moving the swashplate could have very serious consequences upon flight safety.

A simplified model of the kinematics of a swashplate, able to take into account the actuator displacements, has been developed and implemented into a numerical aero-mechanics helicopter simulation code. Then, a method based on Principal Analysis Component has been found to give very good results for the detection of actuator failures such as partial or total jamming. Finally a new method for the reconfiguration of the control law in the presence of the fault has been developed, which is based upon a LQ criterion, and has also been successfully tested in simulation.

Main notations

$\mathcal{X}^T$	=	transposee of $\mathcal{X}$
$\dot{\mathcal{X}}$	=	derivative of $\mathcal{X}$ w.r.t. time
$\lambda_i, i \in \{1;2;3\}$	=	length of i^{th} actuator
θ_p	=	longtudinal attitude of upper plate
ϕ_p	=	lateral attitude of upper plate
Λ_p	=	distance between respective centers of upper and lower plates
u, v, w	=	helicopter body translational velocities
p, q, r	=	helicopter body angular velocities
ϕ, θ	=	helicopter body bank and pitch angles
A, B	=	state and input matrices of the linear helicopter model
X, Y, U	=	state, output and input vector

1 Introduction

Fault Detection, Identification and Reconfiguration (or FDIR for short) is a major concern in aeronautics. The objective is to assess, isolate and counteract any minor or major failure which might have an incidence upon the flight performances and/or security [2]. This thematic has been widely applied for fixed-wing aircraft: due to the configuration symmetry and the redundancy of the control surfaces, a new flight equilibrium point is usually achievable in case of an actuator failure on an airplane, by using the functional ones [3, 11].

But helicopters did not receive as much attention in the literature: most applications of FDIR techniques for rotary-wing aircraft deal with small, multirotor UAVs or with tandem helicopters[8], both able to use control redundancies to overcome the loss of a control element. However, for a standard helicopter with one main rotor, the consequences of an actuator failure can be much more critical, since it is a naturally unstable aircraft and there is no redundant control surface (one path system). As a consequence, the pilot will have to cope with the stabilization of its aircraft whenever such a failure occurs, according to specific procedures described in the flight manual. Some previous works (see [5]) have addressed the question of the reconfiguration ability right from the conception of the swashplate, by designing an actuator positioning whiwh permits an optimal reconfiguration.

Generally speaking, two different objectives — often conflictive — should be addressed when considering a reconfiguration problem: the first one is naturally to find a configuration of the remaining actuators able to stabilize the system. But on the other hand, it is also necessary to take into account the existing limitations on the remaining actuators, or even on the faulty one: in a failure situation, the classical tradeoff between stabilization capacity and control authority is tightened.

The objective of this work is to assess the efficiency of a swashplate actuator fault modelization, detection, and reconfiguration scheme for a classical, 1-rotor helicopter. In the following sections, a kinematics model representing the displacements of the swashplate and taking into account the linear actuators displacements will be detailed. Then, a simulation model including this kinematics model will be used in order to evaluate the feasibility of an actuator fault detection method using PCA (Principal Component Analysis).

2 Kinematics model of the main rotor swashplate

In a conventional rotorcraft the pilot acts on the collective and cyclic sticks to change the pitch angles of the rotor blades in order to modify the magnitude and direction of the rotor thrust. These capabilities contribute to controlling the flight path of the rotorcraft, the velocities and attitudes of the vehicle are thus modified on pilot requirements. The blade pitch angles can be modified by displacing a swashplate whose rotating part is connected to the rotor blades by means of rigid rods.

In order to take into account the effect of a swashplate actuator fault, it is necessary to rely on a valid model of this mechanism. When using aeromechanical simulation codes, it is often more practical to limit the computations to the aerodynamical and structural behavior of the blades, and their influence on the global dynamics of the helicopter: the input variables are the blades pitch angles (collective, longitudinal and lateral cyclic), and the pilot inputs as well as the swashplate displacements are not considered. On the other hand, some existing simulation tools offer the ability to precisely represent the swashplate kinematics [9], but this requires to provide many geometrical data inputs such as initial lengths, cam dimensions, *etc*., which are rarely available when dealing with a generic architecture.

This is why a simple — but representative enough — kinematics model has been derived, in order to simulate realistic failures affecting the actuators displacements. The swashplate mechanism aims at controlling the pitch angles of the rotor blades through the elevation and angular position of the rotating upper plate. This upper plate is mechanically linked through rods to each blade (at trailing edge in most cases), therefore the vertical position of the upper plate defines the collective pitch angle (constant angle for every blade azimuthal position), and the longitudinal or lateral plate inclination controls the cyclic pitch angle (1-per-rev variable angle). This 3-d.o.f. motion of the upper plate is generally obtained through the extension of 3 vertical actuators. Thus, knowing the position of every actuator λ_i allows to determine the values for the collective and cyclic blade pitch angles, which in turn affect directly the aerodynamic forces upon the helicopter main rotor.

A schematic view of the swashplate positioning with respect to the rotor axis system is shown in figure 1. Here two different positions of the swashplate are represented. The parameters λ_1, λ_2, λ_3 are the extensions of the three actuators which displace the swashplate around the rotor axis z_0 from its initial position in the reference plane. The required position of the swashplate is

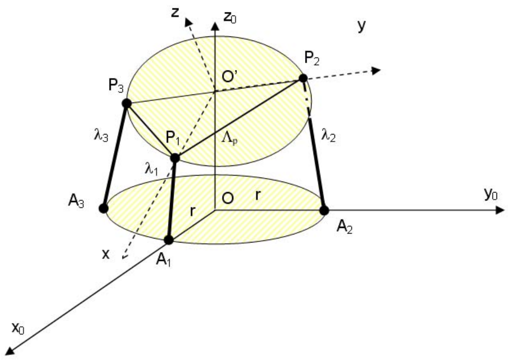

Fig. 1 Geometric configuration of a swashplate and actuators mechanism

achieved in driving three actuators (hydraulic or electric actuators) which are linked to the fixed part of the swashplate by means of swivels. Each swivel A_i displaces to P_i and the plate centre O displaces to O'. The displacement OO' of the swashplate centre is called Λ_p. The actuators allow the swashplate displacement with respect to z_0, according to the three following degrees of freedom: the position of the swashplate centre along the z_0 axis, and two angles defining the orientation of the plate with respect to the reference plane (x_0, y_0).

These actuators control:

- the collective pitch angle of the blades that is dependent on the position Λ_p of the swashplate centre along the rotor axis;
- the longitudinal and lateral cyclic pitch angles which are functions of θ_p and ϕ_p, the two Euler angles defining the orientation of the plate with respect to the (x_0, y_0) plane that is perpendicular to the rotor axis.

There are therefore three degrees of freedom to position the swashplate with respect to the rotor (Λ_p, θ_p and ϕ_p) and three actuators to do it : λ_1, λ_2, and λ_3.

From the data of the mechanical system linking the swashplate to the blades (length and position of the rigid rods, definition of their mechanical links) it is then possible to calculate the values of the pitch blade angles according to their azimuth if the position and orientation of the swashplate are known. Consequently the magnitude and direction of the rotor thrust can then be calculated via a mathematical model of the rotor aerodynamics.

A mechanical default of one of these actuators will limit the courses of the swashplate displacement, and will thus modify the capabilities for controlling the rotor blade pitch angles. In the particular case of one actuator jamming only two actuators are available for controlling the swashplate therefore one degree of freedom is definitely lost for displacing it.

In flight, such a failure is particularly severe and reduce drastically the control capabilities of the rotorcraft. It requires to quickly diagnose and identify the actuator jamming, and then to reconfigure the control laws initially designed on the base of a nominal model of the rotorcraft behavior — in order to make it possible to pursue the flight while minimizing the deterioration of the rotorcraft maneuverability and handling qualities.

In order to be able to diagnose and deal with such a failure, it is first necessary to establish the mathematical model determining the values of the position Λ_p and attitudes θ_p and ϕ_p of the plate for any given values of λ_1, λ_2, λ_3, the magnitudes of the actuators extensions.

The determination of the equations of λ_1, λ_2 and λ_3 as function of Λ_p, θ_p and ϕ_p is established by expressing the distances λ_i between the swivel points A_i, which are fixed on the rotorcraft structure, to their corresponding points P_i situated on the swashplate. Expressing these distances conduct to the following system of three nonlinear equations:

$$\begin{aligned}
\lambda_1^2 &= \Lambda_p^2 - 2\Lambda_p r \sin\theta_p - 2r^2(\cos\theta_p - 1)\\
\lambda_2^2 &= \Lambda_p^2 + 2\Lambda_p r \sin\phi_p \cos\theta_p + 2r^2(1 - \cos\phi_p)\\
\lambda_3^2 &= \Lambda_p^2 - 2\Lambda_p r \sin\phi_p \cos\theta_p + 2r^2(1 - \cos\phi_p)
\end{aligned} \tag{1}$$

where r is the distance of the swivel points to the swashplate centre.

By setting $\tan\alpha = r/\Lambda_p$, the unknown parameter θ_p can be expressed as a unique function of Λ_p:

$$\theta_p = \arcsin\left[\frac{\cos\alpha}{\Lambda_p}\left(r + \frac{\Lambda_p^2 - \lambda_1^2}{2r}\right)\right] - \alpha = h(\Lambda_p) \quad (2)$$

then ϕ_p can also be expressed as a unique function of Λ_p:

$$\phi_p = \arcsin\left(\frac{\lambda_1^2 - \lambda_2^2}{4\Lambda_p r \cos\left(h(\Lambda_p)\right)}\right) = g(\Lambda_p) \quad (3)$$

Consequently a third equation in Λ_p can be extracted from the initial system (1) of three non-linear equations in λ_i, which can be formulated as follows:

$$\Lambda_p^2 + 2r^2 - \frac{\lambda_2^2 + \lambda_3^2}{2} - 2r^2\cos\left(g(\Lambda_p)\right) = 0 = f(\Lambda_p) \quad (4)$$

It can be noted that solving the initial equations system (1) is now equivalent to calculate the solutions of $f(\Lambda_p) = 0$. Initializing the value of Λ_p the system can be solved in Λ_p, θ_p and ϕ_p using an iterative method of Newton which is expressed in the classical form $\Lambda_{p_n} = \Lambda_{p_{n-1}} - \frac{f(\Lambda_{p_{n-1}})}{f'(\Lambda_{p_{n-1}})}$ where the derivative $f'(\Lambda_p) = \frac{df}{d\Lambda_p}$ can also be calculated as a unique function of Λ_p.

3 Actuator fault detection

Using the previous kinematics model coupled with a realistic helicopter aero-mechanics simulation code, it is possible to artificially introduce any default relative to the movement of the swashplate actuators. In particular, complete or partial jam can be simulated, as well as a loss of efficiency. For a real helicopter or UAV, such faulty situations could be due for example to purely mechanical defects, or to an erroneous behavior of the avionics calculators.

Generally speaking, two kinds of techniques might be employed to detect a fault occurrence in a monitored system. On one hand, some methods are based upon an explicit knowledge of both the system and the failure, and require a model for each. In this case, a model of the nominal system can be run alongside the execution of the real one, and a fault is detected if there is a discrepancy between the model outputs and the same variables measured for the real system. For example, some previous works [10] used a set of Kalman filters in order to estimate the occurrence of a fault on a sensor: for each sensor, a dual set of observers is constructed using measurements given by all sensors but the considered one. In case of a failure affecting sensor j, the only set which would be insensitive to it would be precisely the j^{th} one.

On the other hand, another set of methods do not rely on any model of the nominal system nor the failure, and are rather based on adaptive or stochastic techniques, like the one we will present thereafter.

3.1 Presentation of the method

The method explained here for fault detection is adapted from Hagenblad *et al.* [6, 4], and is based on a Principal Component Analysis (PCA) performed upon an estimation of the covariance matrix of the input-output transfer of the helicopter. This covariance matrix is evaluated during a nominal flight (*i.e.* without any default). Then a Principal Component Analysis (PCA) based upon a Singular Value Decomposition (SVD) of the covariance matrix allows to break down the states of the system into the most representative ones and the rest, called residuals. As these residual components are less characteristic of the nominal (default-free) input-output behavior of the system, they might also be more sensitive to events affecting the integrity of this transfer, such as a failure for example. The main advantage of this method is that it does not rely on any descriptive model of the helicopter, and could also be applied to a wide range of systems.

Let us consider that the input and output measurements are available as sampled data with time step Δt. In this section, the shorthand notation $Z(i) \triangleq Z(i\Delta t)$, $i \in \mathbb{N}$ will be used. First, the inputs $U \in \mathbb{R}^{n_u}$ and the measured outputs $Y \in \mathbb{R}^{n_y}$ of the system at a given discrete time sample i are put together into a single vector $Z \in \mathbb{R}^{n_u+n_y}$:

$$Z(i) = \begin{pmatrix} Y(i) \\ U(i) \end{pmatrix} \tag{5}$$

The objective is then to derive an implicit model characterizing the input-output nominal transfer. One of the simplest ways to do so is to calculate the covariance matrix (or at least an estimation) of Z over a given time frame, and then to use the PCA on this covariance matrix to separate the representative (explanatory) dimensions from the least important (residual) ones.

In this case, a time window of length ℓ (in number of samples) is considered:

$$t \in [i_0\Delta t; (i_0+\ell)\Delta t[\tag{6}$$

This window can be chosen to be fixed, *i.e.* i_0 is constant, or sliding with time. Naturally, if the window is fixed, its location and length should be chosen so as to contain a sample as much representative as possible of the nominal behavior of the system. On the other hand, a sliding window will be more responsive to the evolutions of the system I/O dynamics, but will necessitate more online computation for a real-time detection: a classical compromise between rapidity and performance will be carefully sorted out.

From the values of Z within the considered time frame, an estimation of the covariance matrix is constructed:

$$\hat{R} = \frac{1}{\ell} \sum_{i=i_0}^{i_0+\ell} (Z(i) - \bar{Z})\,(Z(i) - \bar{Z})^T \tag{7}$$

where $\bar{Z}$ is the averaged value of Z over the considered window:

$$\bar{Z} = \frac{1}{\ell+1} \sum_{i=i_0}^{i_0+\ell} Z(i) \tag{8}$$

In order to find the characteristics subspaces of $\hat{R}$, a singular value decomposition is performed on the covariance matrix estimation:

$$\hat{R} = WDW^T \tag{9}$$

The singular values contained in D can be sorted and separated into two sets S and T of respective dimensions $n_y + n_u - n_r$ and n_r, where T would only contain the smallest values:

$$D = \begin{bmatrix} S & 0 \\ 0 & T \end{bmatrix} \tag{10}$$

The same decomposition is performed upon the transformation matrix, which is split into two sub-matrices: $W = [W_S\ W_T]^T$. The residual vector ρ is then given by expressing the complete input-output vector Z into the residual space:

$$\rho = {W_T}^T Z \tag{11}$$

It is important to notice that the choice of n_r, *i.e.* the dimension of the residual space, with respect to the initial space dimension, may have consequences on the method efficiency. A value too high can introduce some of the legitimate dynamics of the system into the residuals, and hence might let appear some false positive detections. On the opposite, a residual space too small may not be able to represent all the considered failures.

3.2 Simulation results of actuator fault detection

This method was applied numerically for the detection of a swashplate actuator fault occurring on a 11-ton class helicopter. A simulation model of the helicopter has been completed with the kinematics model developed in section 2, and different actuator jamming have been simulated during an ordinary longitudinal flight. The results are shown in figure 2, this case corresponding to a complete locking of actuator λ_0 occurring at time $t_f = 5\ s$.

As it can be seen in fig. 2, the evolution of the output variables (the vertical speed w and body pitch angle θ are shown here) after the fault occurrence does not follow a recognizable pattern. However, the residual components (bottom plots) are much more clearly correlated with the fault occurrence: a strong deviation from the respective average values of residual vector components r_1 and r_2 appears from $t > t_f$, and by setting a user-defined threshold (as shown as a black dot-

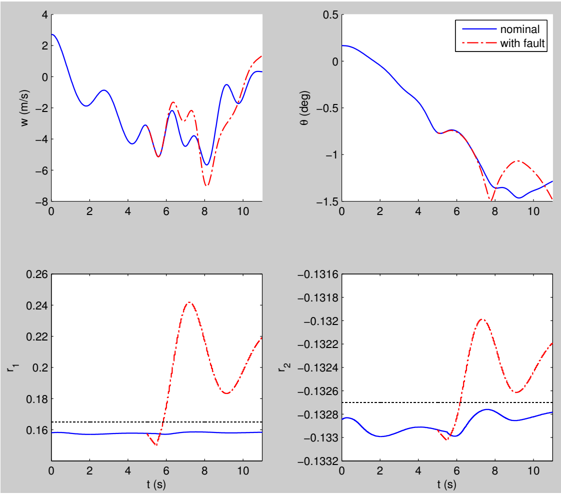

Fig. 2 Evolution of helicopter dynamics and residual components in presence of actuator failure

ted line on fig. 2) it is possible to detect precisely the occurrence of an actuator failure.

4 Control law reconfiguration after failure

After having developed models and tools for the simulation and detection of an actuator fault occurring in the swashplate, the next step would be to ensure that the detected fault will have only limited consequences upon the degradation of the helicopter dynamics and the safety of the flight.

In complex systems, or wherever safety is a critical issue, the loss of some functions must at any cost be avoided. As a consequence the main components, being sensors, calculators or actuators, are physically redunded in sufficient number, so that the failure probability of all redundant elements simultaneously would be small enough (usually 10^{-9} fault per flight hour). However, such redundancies are also tightly associated with extra weight and maintenance. With the development of fly-by-wire technologies — bringing more onboard computational capacities — and the emergence of an increasing number of unmanned rotary-wing air vehicles configurations, for which safety of the crew is no longer a concern, it is now more foreseeable to substitute material redundancies with real-time (or pre-planned) adaptations of the stabilisation and control law of the helicopter.

In this section, an overview of two existing reconfiguration methods which have been applied for helicopters will be presented. Then, an innovative technique, based upon a reconfiguration-specific LQ criterion, will be detailed, accompanied with simulation results.

Moreover, we will assume that the open-loop dynamics of the fault-free helicopter can be written under a classical linear state-space form:

$$\dot{X} = AX + BU \tag{12}$$

where as usual $X \in \mathbb{R}^{n_x}$ is the state vector, $U \in \mathbb{R}^{n_u}$ the input vector, A and B are respectively the state and control matrices. We will also suppose that the considered fault affects linearly the inputs of the system:

$$\tilde{U} = PU \tag{13}$$

where P is the fault matrix of dimension $n_u \times n_u$, characteristic of a given occurring default, and $\tilde{U}$ is the fault-affected input. For example, a 60% loss of efficiency in the i^{th} actuator would be represented by $P = \text{diag}(1, \ldots, \delta_i, \ldots, 1)$, where $\delta_i = 0.4$ [12].

Finally, the presence of a state feedback control law is also taken into account:

$$U = V - KX \tag{14}$$

where K is the control matrix gain and V the objective (reference) value (without loss of generality V might be considered equal to zero in most calculations).

4.1 Overview of existing techniques

4.1.1 Pseudo-inverse method (or control remixing)

This method is presented in [7], and explained more in detail in [8]. It applies when the open-loop helicopter is stabilized through a state feedback. The actuator efficiency loss is modelized through a real matrix P, such as introduced in (13). The state-space control matrix $\tilde{B}$ of the

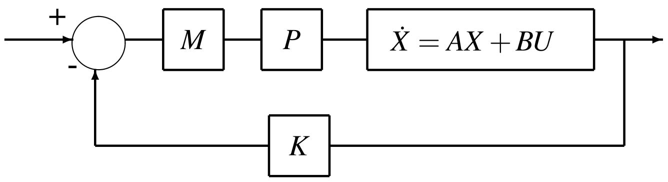

Fig. 3 Closed-loop reconfigured with pseudo-inverse method

faulty system is then given by post-multiplying the nominal (fault-free) control matrix B by P — which is equivalent to say that P pre-multiplies the control U. The objective of the reconfiguration procedure is to find a matrix M such as:

$$\tilde{B}M = B \tag{15}$$

This system is *a priori* overdetermined and could be resolved only in an approximate way. The classical Moore-Penrose pseudo-inverse is used to give the optimal (in terms of least-square distance) solution:

$$\begin{aligned} M &= \tilde{B}^{+}B \qquad (16)\\ \tilde{B}^{+} &\triangleq (\tilde{B}^T\tilde{B})^{-1}\tilde{B}^T \qquad (17)\end{aligned}$$

The closed-loop after reconfiguration can be represented as in figure 3: it appears that the introduction of the reconfiguration matrix M can also be seen as a modification of the control gain to $K_R \triangleq MK$, so that the closed loop matrix after reconfiguration becomes:

$$\tilde{A} = A - \tilde{B}MK = A - \tilde{B}K_R \tag{18}$$

Whenever a complete jamming of an actuator (equivalent to a 100% loss of efficiency) is not involved, the P matrix remains invertible. The solution to the reconfiguration problem is trivially given by $M = P^{-1}$, and the system (15) is solved exactly. As a consequence, a 50% loss on a given actuator needs for compensation an input twice as much as in the nominal case, obtained through matrix M. It is important to notice that this reconfiguration technique has an effect upon the system dynamics (via the equivalent closed-loop matrix $A - \tilde{B}K_R$), but also upon the controls. The control after reconfiguration U_R is indeed given by:

$$U_R = M(V - KX) = V_R - K_R X \tag{19}$$

with $V_R = MV$ is the new reference value. As a consequence, the pilot or the external control loop does not have to bother giving a greater input in order to compensate the failure, since the reference value V, albeit affected by the fault $(A - \tilde{B}MK)$, is also reconfigured by matrix M and becomes V_R. On the other hand, the helicopter handling qualities might be altered. In the case of a full actuator locking, represented as a null diagonal term in fault matrix P, the solution is given through a least-square, pseudo-inverse approach, as mentioned earlier, and the actual needs on the faulty control are distributed along the healthy inputs, hence the "control remixing" nomenclature.

This method presents however some drawbacks. First, and as it was previously mentioned, the control after reconfiguration might involve an input requirement such as actuator elongation, pressure, or voltage (depending on the technology), which may be physically unattainable given the mechanical constraints inherent to the system. In order to circumvent this issue, previous research [8] proposed to use a weighted least-square method, by introducing some constraints upon the coefficients in M. The objective is to find a tradeoff between the verification of equality (15) and the magnitude of the post-reconfiguration inputs. But the efficiency of the method lies upon a considerate choice of weighting parameters, which are depending *a priori* on the actual fault type. Another drawback is that this method does not take into account the severity of the occurring fault: it is clear that the reconfigured controls will generate a higher solicitation on a presumably already damaged actuator, which may worsen the situation and lead to a complete breakup in the control channel. An improvement over this shortcoming will be proposed in section 4.1.3.

4.1.2 Reference model

This technique is also presented in detail in [8], and assumes that the helicopter is controlled in closed-loop with a reference (objective) model. A LQ-criterion, associated with the difference between the closed-loop behavior and the reference dynamics, is minimized, thus ensuring that the system acts nominally as close as possible to the reference model. The proposed reconfiguration method is based on a simple inversion of fault matrix P, in order to derive a new control gain $K_R = P^{-1}K$. The closed-loop stability is preserved, but the reference input V is not taken into account in the reconfiguration process. Moreover, this method remains unapplicable in case of a full efficiency loss, since P would not be invertible.

4.1.3 Reconfiguration LQ criterion

A new method is proposed here, intended to overcome the issues of the pseudo-inverse technique. The objective is mainly to formulate a quadratic criterion whose minimization will reduce the difference between the reconfigured control matrix and the nominal one, while alleviating the solicitations on the faulty input. The following formulation satisfies these goals:

$$\begin{aligned} J \;=\; & \int_0^\infty U^T(\tilde{B}M-B)^T Q(\tilde{B}M-B)U \\ & +U^T M^T RMU\,dt \end{aligned} \tag{20}$$

where Q and R are symmetric positive-definite weighing matrices. The coefficients in R are intended to alleviate the effect of the reconfiguration matrix M on the faulty input, in order to avoid a potentially increased stress applied to a weakened actuator, as we stated before. Hence, R matrix should be symmetric, but not necessarily diagonal.

In order to calculate the reconfiguration matrix minimizing criterion (20), let us assume that the nominal system is stabilized with a state feedback $U = -KX$. By substituting into the previous expression, we get:

$$J = \int_0^\infty X^T K^T SKX\,dt \tag{21}$$

with $S = (\tilde{B}M-B)^T Q(\tilde{B}M-B)+M^T RM$.

Furthermore, since $A_K \triangleq A-BK$ is a stabilizing state feedback, previous expression is also equal to:

$$J = {X_0}^T P X_0 \tag{22}$$

where

$$P = \int_0^\infty e^{{A_K}^T t} K^T SK e^{A_K t}\,dt \tag{23}$$

and $X_0 = X(t=0)$. It is easy to verify [1] that P is also solution of the Lyapunov equation:

$${A_K}^T P + PA_K + K^T SK = 0 \tag{24}$$

Let us denote with M^* the optimal value of M minimizing criterion (21), and with J^* and P^* the respective corresponding values for J and P. By considering small variations ΔM around M^* and the resulting small variations ΔP around P^*, we have the following equations:

$$J = {X_0}^T(P^*+\Delta P)X_0 = J^* + {X_0}^T \Delta P X_0 \tag{25}$$

$$\begin{aligned} & (A-B(M^*+\Delta M)K)^T(P^*+\Delta P) \\ & +(P^*+\Delta P)(A-B(M^*+\Delta M)K) \\ & +K^T\Big[(\tilde{B}(M^*+\Delta M)-B)^T Q(\tilde{B}(M^*+\Delta M)-B) \\ & +\,(M^*+\Delta M)^T R(M^*+\Delta M)\Big]K = 0 \end{aligned} \tag{26}$$

and by subtracting equation (24) to the previous one, we get:

$$\begin{aligned} & (A-\tilde{B}(M^*-\Delta M)K)^T \Delta P \\ + \quad & \Delta P(A-\tilde{B}(M^*-\Delta M)K) \\ + \quad & (\Delta MK)^T\big[-\tilde{B}^T P + (\tilde{B}^T Q(-B)+RM \\ & +\tilde{B}^T Q\tilde{B}M)K\big] \\ + \quad & \big[-P\tilde{B}+K^T((\tilde{B}M)^T Q\tilde{B}+(-B)^T Q\tilde{B} \\ & +M^T R)\big]\Delta MK \\ + \quad & (\Delta MK)^T R\Delta MK + (\tilde{B}\Delta MK)^T Q\tilde{B}\Delta MK = 0 \end{aligned} \tag{27}$$

Since M^* minimizes J, any variation ΔM will result in an augmentation for J, and when considering (25), then necessarily $\Delta P > 0$. The system in closed-loop is stable, consequently in application of Lyapunov theorem ΔP is positive if and only if:

$$\begin{aligned} & (\Delta MK)^T \left[-\tilde{B}^T P + (\tilde{B}^T Q(-B) + RM \right. \\ & \qquad \left. + \tilde{B}^T Q\tilde{B}M)K\right] \\ + & \left[-P\tilde{B} + K^T((\tilde{B}M)^T Q\tilde{B} + (-B)^T Q\tilde{B} \right. \\ & \qquad \left. + M^T R)\right] \Delta MK \\ + & (\Delta MK)^T R\Delta MK + (\tilde{B}\Delta MK)^T Q\tilde{B}\Delta MK > 0 \end{aligned} \tag{28}$$

And since Q and R are positive-definite, the above equation holds for any ΔM if necessary:

$$-\tilde{B}^T P + (\tilde{B}^T Q(-B) + RM + \tilde{B}^T Q\tilde{B}M)K = 0 \tag{29}$$

which leads to:

$$K_R = MK = (R + \tilde{B}^T Q\tilde{B})^{-1} \times (\tilde{B}^T P + \tilde{B}^T QBK) \tag{30}$$

If the nominal, fault-free control gain K can be directly modified when reconfiguration is needed, then the new value K_R is given directly by the equation (30) above. Otherwise, if the control law cannot be altered, then the pre-multiplying reconfiguration matrix M can be obtained through a pseudo-inversion of the previous expression.

4.2 Simulation results

As previously for the fault detection, a certain number of numerical simulations were performed in order to test the efficiency of the proposed reconfiguration method. The same helicopter model was used, and this time was linearized around a forward cruise equilibrium point. Then starting from this point, a wind gust has been simulated on several axes simultaneously at time $t = 1\ s$, in order to apply an external perturbation to the system. Then, a 90% loss of efficiency is introduced on the different actuators λ_i, $i \in \{1;2;3\}$ separately at $t = 2\ s$. Finally, in order to allow a small time amount for the fault detection processing, the reconfiguration process starts one second later, at $t = 3\ s$. The simulation results are presented in figs. 4 and 5. Fig. 4 shows the evolution of the helicopter motion through the body velocities and attitudes: the comparison between both simulated cases — respectively without failure and with failure and reconfiguration — shows that the reconfiguration method is able to stabilize the system despite the failure, while ensuring that the dynamics of the reconfigured system remains close enough to the nominal case.

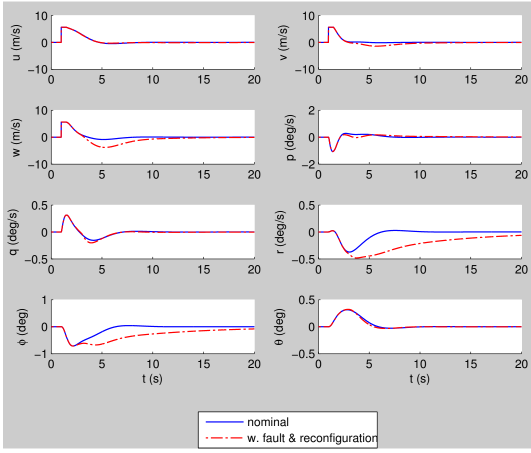

Fig. 4 Evolution of helicopter states: nominal and after fault & reconfiguration

When looking at the evolution of the lengths of the actuators (which are in this case the controlled inputs of the system) in fig. 5, it is also noteworthy to observe that a faulty actuator would not receive a higher solicitation than in the fault-free case, which is consistent with the command weighing introduced in the reconfiguration criterion (20).

5 Conclusion and further work

In this article, some techniques for the detection of a given fault on a helicopter swashplate, as well as for the reconfiguration of the flight control laws after the fault occurrence, have been presented and successfully tested. The next steps will consist in evaluating the robustness of the reconfiguration method, as well as the performances of the fault detection and reconfigura-

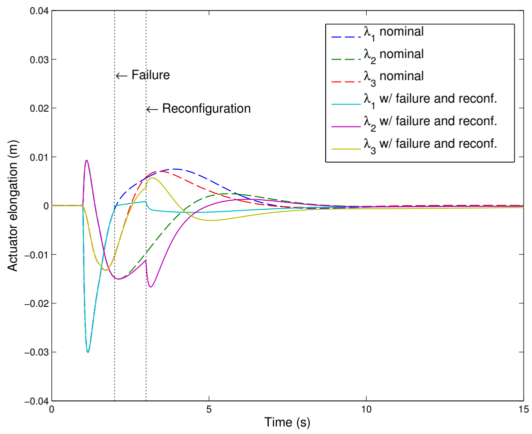

Fig. 5 Evolution of actuators lengths: nominal and after fault & reconfiguration

tion algorithms during a piloted flight. To do so, FDIR simulations will be run on the prototyping and simulation environment LABSIM of ONERA Center of Salon de Provence, which includes a real-time complete helicopter simulation code, control inputs, visual displays and a completely modular and programmable architecture.

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5.1 Copyright Statement